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FLYING QUALITIES OF PILOTED AIRCRAFT



AMSC N/A FSC 15GP

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FOREWORD

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1. SCOPE

1.1 Purpose.

This standard contains the requirements for the handling qualities, in flight and on the ground, of US military piloted air vehicles. It is intended to assure flying qualities that provide adequate mission performance and flight safety regardless of design implementation or flight control system mechanization. The structure of the standard allows its use to guide these aspects in design tradeoffs, analyses, and tests.

1.2 Application.

The requirements of this standard, with blanks filled in, are to be applied during the design, construction, testing, and acceptance of the subject air vehicle. The requirements apply as stated to the combination of airframe and related subsystems.

2. APPLICABLE DOCUMENTS

2.1 General.

The documents listed in this section are specified in sections 3, 4, or 5 of this standard. This section does not include documents cited in other sections of this standard or recommended for additional information or as examples. While every effort has been made to ensure the completeness of this list, document users are cautioned that they must meet all specified requirements of documents cited in sections 3, 4, or 5 of this standard, whether or not they are listed.

2.2 Government documents.

2.2.1 Specifications, standards, and handbooks.

The following specifications, standards, and handbooks form a part of this document to the extent specified herein. Unless otherwise specified, the issues of these documents are those cited in the solicitation or contract.

DEPARTMENT OF DEFENSE SPECIFICATIONS

JSSG-2001	Air Vehicle
JSSG-2006	Aircraft Structures
JSSG-2008	Vehicle Control and Management System (VCMS)
JSSG-2010	Crew Systems
JSSG-2010-12	Crew Systems Deployable Aerodynamic Decelerator (DAD) Systems Handbook
MIL-D-8708	Demonstration: Aircraft Weapon Systems

MIL-F-8785	Flying Qualities of Piloted Airplanes *(Inactive for New Design;document is cited for reference.)*
MIL-A-8861	Airplane Strength and Rigidity Flight Loads
MIL-F-9490	Flight Control Systems - Design, Installation and Test of Piloted Aircraft
MIL-C-18244	Control and Stabilization Systems: Automatic, Piloted Aircraft
MIL-F-83300	Flying Qualities of Piloted V/STOL Aircraft

DEPARTMENT OF DEFENSE STANDARDS

MIL-STD-882	System Safety
MIL-STD-1472	Human Engineering

DEPARTMENT OF DEFENSE HANDBOOKS

MIL-HDBK-217	Reliability Prediction of Electronic Equipment
MIL-HDBK-244	Guide to Aircraft/Stores Compatibility
MIL-HDBK-338	Electronic Reliability Design Handbook

(Copies of these documents are available online at http://assist.daps.dla.mil/quicksearch/ or from the Standardization Document Order Desk, 700 Robbins Avenue, Building 4D, Philadelphia PA 19111-5094.)

2.2.2 Other Government documents, drawings, and publications.
The following other Government documents, drawings, and publications form a part of this document to the extent specified herein. Unless otherwise specified, the issues of these documents are those cited in the solicitation or contract.

ADS-33	Handling Qualities Requirement for Military Rotorcraft

(Application for copies should be addressed to USA AMCOM, Engineering Directorate, AMSAM-AR-E, Redstone Arsenal AL 35898.)

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Navy Rpt No. SA-14R-81	Navy Evaluation of the F/A-18A Airplane with Roll Rate Improvements Incorporated; Copeland, W.; K. Grubbs; et al.; March 1981

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ASD-TR-70-18	Development of Tracking Error Frequency Response Functions and Aircraft Ride Quality Design Criteria for Vertical and Lateral Vibration; Rustenburg, J. W.; January 1971.
ASD-TR-72-48	Criteria for Predicting Spin Susceptibility of Fighter-Type Aircraft; Weissman, R.; June 1972
ASD-TR-72-64	Ride Quality Design Criteria for Aircraft with Active Mode Control Systems; Rustenburg, J. W.; October 1972.
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WADC-TR-52-298	Artificial Stability Flight Tests of the XF-88A Airplane; Moore, N. B.; July 1954
WADC-TR-54-594	Flight Evaluations of Variable Short Period and Phugoid Characteristics in a B-26; Newell, F. D.; and G. Campbell; December 1954
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AFFDL-TR-65-15	Human Pilot Dynamics in Compensatory Systems-Theory, Models, and Experiments with Controlled Element and Forcing Function Variations; McRuer, D.; D. Graham; et al.; July 1965
AFFDL-TR-65-39	Ground Simulator Evaluations of Coupled Roll-Spiral Mode Effects on Aircraft Handling Qualities; Newell, F. D.; March 1965
AFFDL-TR-65-138	A Study of Conventional Airplane Handling Qualities Requirements. Part I: Roll Handling Qualities; Ashkenas, I. L.; November 1965
AFFDL-TR-65-190	An Analysis of Flexible Aircraft Structural Mode Control; Wykes, J.; and A. Mori; June 1966.
AFFDL-TR-65-198	A Handling Qualities Theory for Precise Flight-Path Control; Bihrle, W., Jr.; June 1966
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AFFDL-TR-66-2	Flight Evaluation of Various Phugoid and $1/T_{h_1}$ Values for the Landing Approach Task; Chalk, C. R.; February 1966
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AFFDL-TR-67-51	In-Flight Simulation and Pilot Evaluation of Selected Landing Approach Handling Qualities of a Large Logistics Transport Airplane; Rhoads, D. W.; July 1967
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AFFDL-TR-73-2	Direct Side Force Control (DSFC) for STOL Crosswind Landings; Boothe, E. M.; and H. J. Ledder; February 1973
AFFDL-TR-73-76	Recommended Revisions to Selected Portions of MIL-F-8785B(ASG) and Background Data; Ashkenas, I. L.; R. H. Hoh; and S. J. Craig; August 1973
AFFDL-TR-74-9	A Two-Phase Investigation of Longitudinal Flying Qualities for Fighters; Boothe, E. M.; R. T. N. Chen; and C. R. Chalk; April 1974
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AFFDL-TR-76-78	Direct Side Force Control Criteria for Dive Bombing. Vol I: Summary. Vol II: Analysis and Results; Brulle, R. V.; W. A. Moran; and R. G. Marsh; September 1976
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2.4 Order of precedence.

In the event of a conflict between the text of this document and the references cited herein, the text of this document takes precedence. Nothing in this document, however, supersedes applicable laws and regulations unless a specific exemption has been obtained.

3. DEFINITIONS

3.1 Acronyms and abbreviations.

A/B	Afterburner	
AC	Advisory Circular	
A/C	Aircraft	
ACAH	Attitude-Command-Attitude-Hold	
ACM	Air Combat Maneuver	
AD	Aerial Delivery Flight Phase	4.1.2
ADS	Aeronautical Design Standard	
ADS	Aircraft Development Service	
Adv	Adverse	
Aero	Aerodynamic(s)	
AF	Air Force	
AFAMRL	Air Force Aerospace Medical Research Laboratory	
AFB	Air Force Base	
AFCS	Automatic Flight Control System	
AFFDL	Air Force Flight Dynamics Laboratory	
AFFTC	Air Force Flight Test Center	
AFGS	Air Force Guide Specification	
AFTI	Advanced Fighter Technology Integration Program	
AFWAL	Air Force Wright Aeronautical Laboratories	
AGARD	Advisory Group for Aerospace Research & Development	
AGL	Above Ground Level	
AIAA	American Institute of Aeronautics and Astronautics	
AMRL	Aerospace Medical Research Laboratory	
AMST	Advanced Medium STOL Transport	
AND	Aircraft Nose Down	
ANL	Aircraft Nose Left	
ANR	Aircraft Nose Right	
ANU	Aircraft Nose Up	
AOA	Angle of attack	

APU	Auxiliary Power Unit	
AR	Advisory Report	
AR	Aerial Recovery Flight Phase	4.1.2
ARC	Aeronautical Research Council	
ARI	Aileron-Rudder Interconnect	
AS	Anti-submarine Search Flight Phase	4.1.2
ASD	Aeronautical Systems Division	
ASRM	Automatic Spin Recovery Mode	
AT	Accelerating Transition Flight Phase	4.1.2, 5.2.7, 5.2.7.1.2, 5.2.7.1.8, 5.2.7.2.2
ATLAS	Adaptable Target Lighting Array System	
Aug	Augmented	
AYC	Adverse Yaw Compensator	
B	Bolter Flight Phase	4.1.2, 5.2.6, 5.2.6.4, 5.2.6.6
BAC	British Aircraft Corporation	
BIT	Built-In Test	
BuAer	Bureau of Aeronautics	
C	Carrier-borne aircraft (suffix to Air Vehicle Class, i.e., Class IV-C)	4.1.1
C	Climb Flight Phase	4.1.2
CA	Carrier Approach Flight Phase	4.1.2, 5.2.2.1.2.3, 5.2.2.2.1, 5.2.2.2.2, 5.2.3.10, 5.2.3.10.1, 5.2.6, 5.2.6.3, 5.2.6.6, 5.2.7.1.3
CAL	Cornell Aeronautical Laboratory	
CAP	Control Anticipation Parameter	5.2.2.1.1.2
CAS	Calibrated Airspeed	
CAS	Command Augmentation System or Control Augmentation System	
CCV	Control Configured Vehicle	
CDR	Critical Design Review	
CFD	Computational Fluid Dynamics	
c.g. or CG	Aircraft center of gravity	4.1.3, 4.1.3.1, 4.1.3.2
CH or C-H	Cooper-Harper (as in Cooper-Harper pilot rating)	

CL	Carrier Landing Flight Phase	4.1.2, 5.1.1.1, 5.2.2.1.7.6, 5.2.2.2.1, 5.2.2.2.2, 5.2.3.10, 5.2.3.10.2.1, 5.2.6, 5.2.6.3, 5.2.6.6, 5.2.7, 5.2.7.1.4
C_L	Centerline	
cm	Centimeter(s)	
Co.	Company	
CO	Air-to-air Combat Flight Phase	4.1.2, 5.1.1.1, 5.2.1.6, 5.2.3.5.1, 5.2.3.12.2
Conf. or Config.	Configuration	
Corp.	Corporation	
cos	Cosine	
CP	Conference Proceedings	
cps	Cycles per second	
CR	Contractor Report	
CR	Cruise Flight Phase	4.1.2
CT	Catapult Takeoff Flight Phase	4.1.2, 5.2.2.1.7.5.1, 5.2.6, 5.2.6.2, 5.2.6.6
CTOL	Conventional Takeoff and Landing	
CV	Aircraft carrier	
D	Descent Flight Phase	4.1.2
dB	Decibel(s)	
DE	Emergency Deceleration Flight Phase	4.1.2, 5.2.2.1.8.3
deg	Degree(s)	
DFC	Direct Force Control	5.2.3.12, 5.2.3.12.1, 5.2.3.12.2, 5.2.3.12.3, 5.2.3.12.4, 5.2.3.13, 5.2.3.13.1, 5.2.3.13.2, 5.2.3.13.3, 5.2.3.13.4, 5.2.7.1.9, 5.2.7.2.1.4, 5.2.7.2.1.5, 5.2.7.2.1.6, 5.2.8.3.7
DFCS	Digital Flight Control System	
DFRC	Dryden Flight Research Center	
DFVLR	Deutsche Forschungs- und Versuchsanstalt für Luft- und Raumfahrt	
DIGITAC	YA-7D used in the Digital Flight Control Systems for Tactical Fighters Program	

DLC	Direct Lift Control	5.2.2.1.2.3, 5.2.2.1.2.4, 5.2.2.1.2.4.1, 5.2.2.1.3.3, 5.2.2.1.6, 5.2.2.2, 5.2.2.2.1, 5.2.2.2.2, 5.2.2.2.3, 5.2.7, 5.2.7.1, 5.2.7.1.1, 5.2.7.1.2, 5.2.7.1.3, 5.2.7.1.4, 5.2.7.1.5, 5.2.7.1.6, 5.2.7.1.8, 5.2.7.1.9, 5.2.7.2, 5.2.7.2.1, 5.2.7.2.1.1, 5.2.7.2.1.4, 5.2.7.2.1.7, 5.2.7.2.2, 5.2.8.3.2, 5.2.8.3.7
Doc.	Document	
DOF	Degrees of Freedom	
DOT	Department of Transportation	
DSFC	Direct Side Force Control	5.2.3.2.4, 5.2.3.12, 5.2.3.12.1, 5.2.3.12.2, 5.2.3.12.3, 5.2.3.12.4, 5.2.3.13, 5.2.3.13.1, 5.2.3.13.2, 5.2.3.13.3, 5.2.3.13.4, 5.2.8.3.2, 5.2.8.3.7
DT	Decelerating Transition Flight Phase	4.1.2, 5.2.7, 5.2.7.1.8, 5.2.7.2.2
Dyn Press	Dynamic Pressure	
EAS	Equivalent Airspeed	
ED	Emergency Descent Flight Phase	4.1.2
eff	Effective	
e.g.	Exempli gratia (for example)	
ESDU	Engineering Sciences Data Unit	
exp	Exponent	
FAA	Federal Aviation Administration	
FAR	Federal Aviation Regulations	
FCS	Flight Control System	5.2.8, 5.2.8.1, 5.2.8.2, 5.2.8.2.1, 5.2.8.3, 5.2.8.3.1, 5.2.8.3.2, 5.2.8.3.3, 5.2.8.3.4, 5.2.8.3.5, 5.2.8.3.6, 5.2.8.3.7, 5.2.8.4, 5.2.9, 5.2.9.1, 5.2.9.1.1, 5.2.9.1.2, 5.2.9.1.3, 5.2.9.1.4, 5.2.9.1.5, 5.2.9.2, 5.2.9.2.1, 5.2.9.3
FDL	Flight Dynamics Laboratory	
FF	Close Formation Flying Flight Phase	4.1.2, 5.1.1.1, 5.2.1.6
FFT	Fast Fourier Transform	
FLT	Flight	
FMECA	Failure Modes and Effects Criticality Analysis	5.1.1.3.2, 5.2.1.5, 5.2.1.5.1, 5.2.1.5.2

FMET	Failure Modes and Effects Test	5.1.1.3.2, 5.2.1.5, 5.2.1.5.1, 5.2.1.5.2
fn	Function	
fnsp	Short-period natural frequency	
FOD	Foreign Object Damage	
fpm	Feet per minute	
fps	Feet per second	
FRM	Flight Research Memorandum	
FSAA	Flight Simulator for Advanced Aircraft	
FSD	Full-Scale Development	
ft	Foot or feet	
FTC	Flight Test Center	
FTM	Flight Test Manual	
FTR	Flight Test Report	
FWD	Forward	
GA	Ground Attack Flight Phase	4.1.2, 5.1.1.1, 5.2.1.6, 5.2.3.5.1
GCA	Ground-Controlled Approach	
GCAH	Gamma- or Flightpath-Command-Attitude-Hold	
GCGH	Gamma- or Flightpath-Command-Gamma-Hold	
GDCGH	Gamma-Dot- or Flightpath-Rate-Command-Flightpath-Hold	
GH	Ground Handling	4.1.2, 5.1.1.1, 5.2.2.1.7.5, 5.2.2.1.7.6, 5.2.3.10, 5.2.3.10.2, 5.2.3.10.3, 5.2.3.11.1, 5.2.6, 5.2.6.1, 5.2.7, 5.2.7.1.7
GM	Gain Margin	5.2.2.1.3.1, 5.2.3.12.1
GRATE	Ground Attack Targeting Equipment	
GW	Aircraft Gross Weight	
H	High speed range	5.2.3.5
H	Hover Flight Phase	4.1.2, 5.2.7, 5.2.7.1.5, 5.2.7.1.9, 5.2.7.2.1, 5.2.7.2.1.1, 5.2.7.2.1.2, 5.2.7.2.1.3, 5.2.7.2.1.4, 5.2.7.2.1.5, 5.2.7.2.1.6, 5.2.7.2.1.7, 5.2.7.2.1.8
HDBK	Handbook	
HFB	Hamburger Flugzeugbau	
HGI	Hot Gas Ingestion	

HOS	Higher-Order System	5.2.2.1.1, 5.2.2.1.1.1, 5.2.2.1.1.2, 5.2.2.1.3, 5.2.2.1.3.1, 5.2.3.1, 5.2.3.1.1, 5.2.3.1.2, 5.2.3.1.3, 5.2.3.1.4, 5.2.3.1.5, 5.2.3.1.6, 5.2.3.2.7
HQDT	Handling Qualities During Tracking	5.1.1.1, 5.2.1.6
HQR	Handling Qualities Rating	
HST	Hypersonic Transport	
HT	Hovering Translation Flight Phase	4.1.2, 5.2.7, 5.2.7.1.9, 5.2.7.2.1.5, 5.2.7.2.1.6
HUD	Head-Up Display	
HWR	Hover Weight Ratio	
Hz	Hertz	
IAS	Indicated Airspeed	
IAS	Institute of the Aeronautical Sciences	
ICAS	International Council of the Aeronautical Sciences	
i.e.	Id est (that is)	
IFFC	Integrated Fire and Flight Control	
IFR	Instrument Flight Rules	
ILAF	Identical Location of Accelerometer and Force	
ILS	Instrument Landing System	
IMC	Instrument Meteorological Conditions	
in.	Inch(es)	
Inc.	Incorporated	
INTER	Intermediate	
ISO	International Organization for Standardization	
J. C.	Jesus Christ!, as in J. C. Maneuver (PIO)	
JSSG	Joint Service Specification Guide	
KCAS	Knots Calibrated Airspeed	
KEAS	Knots Equivalent Airspeed	
KIAS	Knots Indicated Airspeed	
kt or kts	Knot(s)	
L	Land-based aircraft (suffix to Air Vehicle Class, i.e., Class IV-L)	4.1.1

L	Landing Flight Phase	4.1.2, 5.1.1.1, 5.2.2.1.7.6, 5.2.2.2.1, 5.2.2.2.2, 5.2.3.10, 5.2.3.10.2, 5.2.7.1.4
L	Left	
L	Low speed range	5.2.3.5
LAHOS	Landing Approach Higher-Order System	
LAHS	Low Altitude High Speed	
LAPES	Low Altitude Parachute Extraction System	4.1.2
LATHOS	Lateral Higher-Order System	
lb	Pound(s)	
LCDP	Lateral Control Divergency Parameter	5.2.5.5, 5.2.5.5.1, 5.2.5.5.2, 5.2.5.5.3
LHA	Landing Helicopter Assault amphibious assault ship	
LHD	Landing Helicopter Dock amphibious assault ship	
ln	Natural logarithm	
LO	Loiter Flight Phase	4.1.2
LOES	Lower-Order Equivalent System	5.2.2.1.1, 5.2.2.1.1.1, 5.2.2.1.1.2, 5.2.3.1, 5.2.3.1.1, 5.2.3.1.2, 5.2.3.1.3, 5.2.3.1.4, 5.2.3.1.5, 5.2.3.1.6
LOS	Low Order System; i.e., conventional, unaugmented aircraft dynamics	
Lt	Left	
Lt.	Lieutenant	
LT	Lateral Translation	5.2.3.13, 5.2.3.13.1, 5.2.3.13.2, 5.2.3.13.3, 5.2.3.13.4, 5.2.7.2.1.6, 5.2.8.3.7
LWD	Left Wing Down	
m	Meter(s)	
M	Mach number	
M	Medium speed range	5.2.3.5
MAC	Mean Aerodynamic Chord	
MAT	Maximum Augmented Thrust, maximum thrust augmented by all means available for the Flight Phase	
max or MAX	Maximum	
MDC	McDonnell Douglas Corporation	
Memo	Memorandum	

MIL	Military	
Min or MIN	Minimum	
min	Minute(s)	
MPO	Manual Pitch Override	
MRT	Military Rated Thrust, the maximum thrust at which the engine can be operated for a specific period	
MSL	Mean Sea Level	
MVLW	Maximum Vertical Landing Weight	
NACA	National Advisory Committee for Aeronautics	
NADC	Naval Air Development Center	
NAECON	National Aerospace Electronics Conference	
NASA	National Aeronautics and Space Administration	
NATC	Naval Air Test Center	
NATO	North Atlantic Treaty Organization	
NATOPS	Naval Air Training and Operating Procedures Standardization Program (aka Navy's Attempt To Operate Planes Safely)	
NLR	Nationaal Lucht- en Ruimtevaartlaboratorium (National Aerospace Laboratory, The Netherlands)	
NORM	Normal	
NRC	National Research Council	
NRT	Normal Rated Thrust, the maximum thrust at which the engine can be operated continuously	
oct	Octave	
OPT	Optimum	
ORD	Operational Requirements Document	
OVRD	Override	
PA	Power Approach Flight Phase	4.1.2, 5.1.1.1, 5.2.1.6, 5.2.2.1.2.3, 5.2.2.2.1, 5.2.2.2.2, 5.2.3.10, 5.2.3.10.1
PH	Precision Hover Flight Phase	4.1.2, 5.2.7, 5.2.7.1.5, 5.2.7.1.9, 5.2.7.2.1, 5.2.7.2.1.1, 5.2.7.2.1.2, 5.2.7.2.1.3, 5.2.7.2.1.4, 5.2.7.2.1.5, 5.2.7.2.1.6, 5.2.7.2.1.7, 5.2.7.2.1.8

PIO	Pilot-In-The-Loop Oscillation (previously known as Pilot-Induced Oscillation)	5.2.1.6, 5.2.2.1.1.2, 5.2.2.1.2.2, 5.2.2.1.3.1, 5.2.2.1.3.2, 5.2.2.1.3.3, 5.2.2.1.4, 5.2.2.1.5, 5.2.2.1.6, 5.2.2.1.8.1, 5.2.2.1.8.2, 5.2.2.1.8.4, 5.2.3.1.1, 5.2.3.1.4, 5.2.3.1.5, 5.2.3.1.6, 5.2.3.2.1, 5.2.3.2.2, 5.2.3.2.3, 5.2.3.2.5, 5.2.3.2.6, 5.2.3.2.7, 5.2.3.3, 5.2.3.4, 5.2.3.5, 5.2.3.5.1, 5.2.3.5.2, 5.2.3.5.3, 5.2.3.8.2, 5.2.3.8.3, 5.2.8.2.1, 5.2.8.3.1, 5.2.8.3.2, 5.2.8.3.5
PIOR	PIO Rating	
PL	Precision Landing	
PLA	Power Lever Angle	
p/p	Peak to peak	
PR	Pilot Rating	
PRCA	Pitch/Roll Channel Assembly	
PRV	Proverse	
PSD	Power Spectral Density	
psf	Pounds per square foot	
PSG	Post-Stall Gyrations	
R	Right	
rad	Radian(s)	
RAE	Royal Aeronautical Establishment	
RC	Reconnaissance Flight Phase	4.1.2
RCAH	Rate-Command-Attitude-Hold	
Ref.	Reference	
rms	Root-Mean-Square	
RORH	Region of Recoverable Handling	4.1.4, 4.1.4.3, 5.1.1.3.3, 5.2.2.1.8.8
ROSH	Region of Satisfactory Handling	4.1.4, 4.1.4.1
ROTH	Region of Tolerable Handling	4.1.4, 4.1.4.2
Rpt	Report	
RR	In-flight Refueling (Receiver) Flight Phase	4.1.2, 5.1.1.1, 5.2.1.6
RSS	Relaxed Static Stability	
Rt	Right	
RT	In-flight Refueling (Tanker) Flight Phase	4.1.2
RWD	Right Wing Down	

R&M	Reports and Memoranda	
SAE	Society of Automotive Engineers	
SAS	Stability Augmentation System	
SAWE	International Society of Allied Weight Engineers	
SCAS	Stability and Control Augmentation System	
SCR	Supersonic Cruise Research	
SD	Substantiating Document	
sec	Second(s)	
SETP	Society of Experimental Test Pilots	
SFCS	Survivable Flight Control System	
Sim	Simulation	
sin	Sine	
SK	Station Keeping Flight Phase	4.1.2, 5.2.7, 5.2.7.1.5, 5.2.7.1.9, 5.2.7.2.1, 5.2.7.2.1.1, 5.2.7.2.1.2, 5.2.7.2.1.3, 5.2.7.2.1.4, 5.2.7.2.1.5, 5.2.7.2.1.6, 5.2.7.2.1.7, 5.2.7.2.1.8
SL	Sea Level	
SL	Short Landing Flight Phase	4.1.2, 5.2.7, 5.2.7.1.4
S/MTD	STOL and Maneuver Technology Demonstrator	
SPEC	Specification	
SRP	Seat Reference Point	
ss	Steady state	
SST	Supersonic Transport	
STD	Standard	
STEMS	Standard Evaluation Maneuver Set	
STI	Systems Technology Incorporated	
STO	Short Takeoff Flight Phase	4.1.2, 5.2.7, 5.2.7.1.1
STOL	Short Takeoff and Landing	
STOVL	Short Takeoff/Vertical Landing	
SUDAAR	Standord University Department of Aeronautics and Astronautics Report	
Sym	Symbol	
tan	Tangent	

TAS	True Airspeed	
TBD	To Be Determined	
TD	Technology Document	
TDR	Technical Documentary Report	
Tech	Technical	
Tech	Technology	
TED	Trailing Edge Down	
TEL	Trailing Edge Left	
TER	Trailing Edge Right	
TEU	Trailing Edge Up	
TF	Terrain Following Flight Phase	4.1.2
TIFS	Total In-Flight Simulator	
TIH	Technical Information Handbook	
TLF	Thrust for Level Flight	
TLR	Technical Letter Report	
TM	Technical Memorandum	
TN	Technical Note	
TO	Takeoff Flight Phase	4.1.2, 5.1.1.1, 5.2.2.1.7.5, 5.2.2.1.8.6, 5.2.3.10, 5.2.3.10.2, 5.2.3.10.2.1, 5.2.3.11.1, 5.2.3.11.2
TP	Technical Paper	
TP	Technical Publication	
TPR	Transient Peak Ratio	5.2.2.1.2.2
TR	Technical Report	
TSS	Transport SuperSonique	
Univ	University	
USAF	United States Air Force	
USAFTPS	United States Air Force Test Pilot School	
USMC	United States Marine Corps	
USNTPS	United States Navy Test Pilot School	
VCMS	Vehicle Control and Management System	
VFR	Visual Flight Rules	
VL	Vertical Landing Flight Phase	4.1.2, 5.2.7, 5.2.7.1.6, 5.2.7.2.1.7

VL	Very Low speed range	5.2.3.5
Vol	Volume	
V/STOL	Vertical/short takeoff and landing	
VTO	Vertical Takeoff Flight Phase	4.1.2, 5.2.7, 5.2.7.1.2
VTOL	Vertical Takeoff and Landing	
WADC	Wright Air Development Center	
WADD	Wright Air Development Division	
WD	Weapon Delivery/Launch Flight Phase	4.1.2, 5.2.1.3, 5.2.1.4
WL	Wright Laboratory	
WLT	Wings-Level Turn	5.2.3.12, 5.2.3.12.1, 5.2.3.12.2, 5.2.3.12.3, 5.2.3.12.4, 5.2.8.3.7
W.O.	Washout	
WO	Waveoff/Go-Around Flight Phase	4.1.2, 5.2.3.7, 5.2.3.11.3, 5.2.6, 5.2.6.5, 5.2.6.6, 5.2.7.1.6
WOD	Wind Over Deck	
WP	Working Paper	
WRDC	Wright Research and Development Center	

3.2 Terms.

Airspeed	Magnitude of the velocity with respect to the air mass	
Air Vehicle Class	An air vehicle falls under one of six classes defined in table I. The Class designation is used in this standard to help particularize the requirements according to broad categories of intended use.	4.1.1
Air Vehicle State	The State of the air vehicle is defined by the selected configuration together with the functional status of each of the air vehicle components or systems, throttle setting, weight, moments of inertia, center-of-gravity position, and external stores complement. The trim setting and the position of the pitch, roll, and yaw controls are not included in the definition of Air Vehicle State since they are often specified in the requirements. There are four types of Air Vehicle States: Normal States, Extreme States, Failure States, and Special Failure States.	4.1.3
Normal States	Air Vehicle Normal States are Air Vehicle States with no component or system failure.	4.1.3.5
Extreme States	Air Vehicle Extreme States are Air Vehicle Normal States with extremely heavy loads or extreme asymmetries or unusual aerodynamic characteristics for which some Level 1 requirements may not be met.	4.1.3.6
Failure States	Air Vehicle Failure States consist of Air Vehicle Normal States modified by one or more malfunctions in air vehicle components or systems.	4.1.3.7
Special Failure States	Special Failure States are Air Vehicle Failure States which have extremely remote probabilities of failure during a given flight.	4.1.3.8
Atmospheric disturbances	Atmospheric disturbances are defined in two atmospheric disturbance models for the purpose of showing compliance with this standard: a low-altitude model (below 2000 ft AGL) and a medium/high-altitude model (above 2000 ft AGL).	5.1.2
Breakout forces	Breakout forces refer to the cockpit control force required to affect movement of the surface.	5.2.2.1.8.5, 5.2.3.8.1, 5.2.4.4
Calibrated airspeed	Airspeed-indicator reading corrected for position and instrument error but not for compressibility	
Calm air	In the context of this standard, Calm air is considered to be no wind, no turbulence, no gusts, and no shears.	5.1.2
Class	See Air Vehicle Class.	

Combat ceiling	The highest altitude at which the maximum rate of climb is 500 ft/min for a given weight and engine thrust	4.1.4.1
Common atmospheric disturbances	Levels of atmospheric disturbances which may be encountered commonly in operations (i.e., probability of exceeding these levels is approximately 1 x 10^{-2})	5.1.2
Configuration	A configuration is defined by the positions and adjustments of the various selectors and controls available to the crew, except for pitch, roll, yaw, throttle, and trim controls. Examples are: the flap control setting and the yaw damper ON or OFF.	4.1.3.4
Control Anticipation Parameter	The ratio of initial pitching acceleration to steady-state normal acceleration	5.2.2.1.1.2
Control effectors	The aerodynamic surfaces, thrust vectoring, reaction controls, etc., which produce forces and moments to control the air vehicle	
Control power	Control power is the effectiveness of control surfaces in applying forces or moments to an air vehicle. For example, 50 percent of available roll control power is 50 percent of the maximum rolling moment available to the pilot with allowable roll control force.	5.1.3, 5.2.2.1.7 through 5.2.2.1.7.6, 5.2.2.2.2, 5.2.3.5 through 5.2.3.5.3, 5.2.3.6, 5.2.3.6.1, 5.2.3.9.4, 5.2.3.10.1 through 5.2.3.10.3, 5.2.3.11.1 through 5.2.3.11.5, 5.2.3.12.2, 5.2.3.13.2, 5.2.4.2, 5.2.5.4.2, 5.2.5.4.3, 5.2.7.2.1.8
Controllable	See “Qualitative Degrees of Suitability.”	
Controllers	Controllers are the cockpit controls; e.g., the stick or wheel and pedals manipulated by the pilot to produce pitching, rolling, and yawing moments.	
Conversion	Conversion is defined as the physical changes in the aircraft configuration that are required to achieve transition from powered-lift flight to fully wing-borne flight.	
Cooper-Harper rating scale	A pilot rating scale used to evaluate air vehicle flying qualities (See figure 5.)	
Cruise ceiling	The highest altitude at which the maximum rate of climb is 300 ft/min at NRT at a given weight	4.1.4.1
Dangerous	A condition in which loss of control, loss of the air vehicle, or death or injury to the crew is probable	
Direct force controllers	Direct force controllers are devices producing direct forces such as direct lift control systems and lateral translations systems.	
Discrete gust	A deterministic implementation (as opposed to a probabilistic implementation) of atmospheric disturbances	5.1.2

Dryden turbulence	The Dryden form of the spectra for random turbulence velocities	5.1.2
Equivalent airspeed	True airspeed multiplied by $\sqrt{\sigma}$ where σ is the ratio of free-stream density at the given altitude to standard sea-level air density	
Effective time delay (t_1)	A time-domain measure of phase lag, determined by plotting the pitch rate time response to a step input and drawing a line tangent to the pitch rate curve at the point of maximum slope. The effective time delay is the time at which this line intersects the time axis (q = 0).	5.2.2.1.2.2
Extraordinary atmospheric disturbances	Levels of atmospheric disturbances which may be encountered only under extraordinary circumstances in operations (i.e., probability of exceeding these levels is approximately 1×10^{-5})	5.1.2
Extreme State	See "Air Vehicle State."	
Fail-operate	The capability of the FCS for continued operation without degradation following a single failure and to fail passive in the event of a related subsequent failure	
Fail-soft	The capability of the FCS to continue with degraded operation that does not result in dangerous or unsafe flight conditions following a single failure (also known as fail-degrade)	
Failure State	See "Air Vehicle State."	
Flight control system	The flight control system includes any stability and control augmentation systems, manual and automatic control and trim functions, the pitch, roll, and yaw system controls, direct force controls, including leading-edge and trailing-edge flaps, trim selectors, and all mechanisms and devices that they operate, including the feel system.	
Flight Envelope	The flight envelope defines boundaries in terms of speed, altitude, load factor, angle-of-attack, sideslip, and any other parameters which form flight limits which encompass all regions in which operation of the air vehicle is both allowable and possible, and which the aircraft is capable of safely encountering.	
Flight path bandwidth	The frequency at which the response of flight path angle, measured at the aircraft center of gravity, lags the cockpit control input, including the feel system, by 135°	

Flight Phase	Air vehicle missions are subdivided into segments known as Flight Phases. Together, the Flight Phases constitute the entire mission, with no gap between successive Flight Phases and smooth transitions between them. For example, the Flight Phases for a ground attack mission might consist of ground operation, takeoff, climb, cruise, in-flight refueling, ground attack, descent, approach, and landing.	4.1.2
Flight Phase Categories	The similarity of tasks in many Flight Phases, plus the limited amount of evaluation data on specific Flight Phases, has led to grouping the Phases into three Categories defined in table II.	4.1.2
Flying qualities	The flying qualities of an air vehicle are defined as the stability and control characteristics that have an important bearing on the safety of flight and on the pilots' impressions of the ease of flying the air vehicle in steady flight and in maneuvers.	
Handling qualities	Those qualities or characteristics of an air vehicle that govern the ease and precision with which a pilot is able to perform the tasks required in support of the air vehicle's role	
High-gain task	A closed-loop piloting task with such demanding task performance requirements that the task requires continuous pilot attention and control corrections to accomplish the task successfully	
Hovering flight	Flight at ground speeds below 35 kts	
Intolerable	"Intolerable" is to be interpreted in the context of controllability: an annoyance, distraction, or discomfort so great as to interfere with the ability to maintain control.	
Lateral Control Divergence Parameter (LCDP)		
Lateral translation	Lateral translations occur at essentially zero bank angle and zero change in heading.	
Levels of flying qualities	The handling characteristics described in this standard are specified in terms of qualitative degrees of suitability and Levels of flying qualities. These Levels and degrees of suitability are part of the structure of this standard, and are based on the Cooper-Harper Scale. (See figure 5.)	5.1.1, 5.1.2
Loadings	The loading of an air vehicle is determined by what is in (internal loading) and attached to (external loading) the air vehicle. The loading parameters that influence flying qualities are weight, center-of-gravity position, and moments and products of inertia. In addition to these, external stores also affect aerodynamic coefficients.	4.1.3.1, 4.1.3.2

Longitudinal control force	The component of applied force exerted by the pilot on the cockpit controller in the direction to command pitch	5.2.2.1.8
Maneuver margin	The distance between the c.g. and the maneuver point	
Maneuver point	Stick-fixed maneuver point: the c.g. position at which the change in elevator angle required to accelerate the aircraft equals zero ($\lvert\delta_e\rvert/n = 0$) Stick-free maneuver point: the c.g. position at which the change in stick force required to accelerate the aircraft equals zero ($\lvert F_{es}\rvert/n = 0$)	
Neutral point	The c.g. position at which the aircraft is neutrally stable ($\lvert C_m\rvert/C_L = 0$)	
Normal State	See "Air Vehicle State."	
Objectionable	Objectionable is to be interpreted in the context of operational missions: an annoyance, distraction, or discomfort so great as to interfere with task performance.	
Phase delay (τ_{θ_p})	A frequency-domain measure of phase lag in pitch attitude response, $\tau_{\theta_p} = -(\varphi_{2\omega_{180}} + 180°)/(57.3 \cdot 2 \cdot \omega_{180})$	5.2.2.1.3.1
Pilot-In-the-loop Oscillation	An unintentional sustained or uncontrollable oscillation that results from the efforts of the pilot to control the aircraft	5.2.1.6, 5.2.2.1.5, 5.2.3.3, 5.2.3.4
PIO rating scale	A pilot rating scale used to evaluate PIO tendencies (See figure 19.)	
Pitch acceleration peak time ($t_{\dot{q}}$)	A time-domain parameter, pitch acceleration peak time is determined from the pitch acceleration time response to a step or pulse input. It is the time at which the pitch acceleration reaches its maximum value.	5.2.2.1.2.2
Pitch attitude bandwidth (ω_{BW_θ})	A frequency-domain parameter, pitch attitude bandwidth is the highest frequency at which the responses of aircraft pitch attitude to pilot control force and control deflection inputs have both 45° or more of phase margin and 6 dB or more of gain margin.	5.2.2.1.3.1
Pitch attitude dropback	A time-domain parameter, pitch attitude dropback is determined from the pitch attitude time response to a pulse input. As originally defined by Gibson (DB_θ), it is the difference between the value of pitch attitude at the time the input is removed and the final pitch attitude. As used in this standard (θ_{DB}), it is the difference between the peak pitch attitude and the steady-state pitch attitude following cessation of the pulse input.	5.2.2.1.2.2
Pitch attitude dropback ratio	A time-domain parameter, pitch attitude dropback ratio is the pitch attitude dropback divided by q_{ss}.	5.2.2.1.2.2

Pitch rate overshoot ratio	A time-domain parameter, pitch rate overshoot ratio is determined from the pitch rate time response to a step or pulse input. It is the ratio q_{max}/q_{ss}.	5.2.2.1.2.2
Pitch rate peak time (t_q)	A time-domain parameter, pitch rate peak time is determined from the pitch rate time response to a step or pulse input. It is the time at which the pitch rate reaches q_{max}.	5.2.2.1.2.2
Post-stall	The flight regime involving angles of attack greater than nominal stall angles of attack. The air vehicle characteristics in the post-stall regime may consist of three more or less distinct consecutive types of air vehicle motion following departure from controlled flight: post-stall gyration, incipient spin, and developed spin.	5.2.5 through 5.2.5.6.4
Post-stall gyration	Post-stall gyration is uncontrolled motions about one or more air vehicle axes following departure from controlled flight. While this type of air vehicle motion involves angles of attack higher than stall angle, lower angles may be encountered intermittently in the course of the motion.	5.2.5 through 5.2.5.6.4
Powered-lift	Powered-lift is the flight regime of any aircraft in which controlled, level flight is possible below the power-off stall speed and in which part or all of the lift and/or control moments are derived directly from powerplant(s).	
Qualitative degrees of suitability	The handling characteristics described in this standard are specified in terms of qualitative degrees of suitability and Levels of flying qualities. These Levels and degrees of suitability are part of the structure of this standard, and are based on the Cooper-Harper Scale. (See figure 5.) The qualitative degrees of suitability are defined as: **Satisfactory** — Flying qualities clearly adequate for the mission Flight Phase. Desired performance is achievable with no more than minimal pilot compensation. **Tolerable** — Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists. **Controllable** — Flying qualities such that the air vehicle can be controlled in the context of the mission Flight Phase, even though pilot workload is excessive or mission effectiveness is inadequate, or both. The pilot can transition from Category A Flight Phase tasks to Category B or C Flight Phases, and Category B and C Flight Phase tasks can be completed.	5.1.1, 5.1.2

	Recoverable — Flying qualities such that control can be regained following loss of control due to departure, failures, or atmospheric disturbances	
Random turbulence	A probabilistic implementation (as opposed to a deterministic implementation) of atmospheric disturbances	5.1.2
Recoverable	See "Qualitative Degrees of Suitability."	
Refusal speed	This is the maximum speed to which the aircraft can accelerate and then stop in the available runway length.	
Regions of Handling	For each Aircraft Normal State, the Flight Envelope is divided into three regions defined in terms of speed, altitude, and load factor, plus any other parameters which may form flight limits (such as sideslip), within which the flying qualities requirements apply.	4.1.4
	Regions of Satisfactory Handling (ROSH) — The boundaries of the ROSH are determined by the speed, altitude, and load factor ranges which the aircraft must be able to achieve in order to accomplish the missions of 4.1.1 with Level 1 flying qualities for normal operation.	4.1.4.1
	Regions of Tolerable Handling (ROTH) — The boundaries of the ROTH are based on speed, altitude, and load factor limits which are derived from aircraft performance margins, such as service ceiling, at or beyond the limits of the ROSH and within which the aircraft must have at least Level 2 flying qualities for normal operation.	4.1.4.2
	Regions of Recoverable Handling (RORH) — The boundaries of the RORH encompass all regions of the Flight Envelope in which operation of the aircraft is both allowable and possible, and which the aircraft is capable of safely encountering.	4.1.4.3
Ride Discomfort Index (D_i)	A measure of the impact of ride quality on the crew over time during flight in turbulence	5.2.1.8
Rise time (Δt)	Rise time is a time-domain measure of responsiveness, determined by plotting the pitch rate time response to a step input and drawing a line tangent to the pitch rate curve at the point of maximum slope. The rise time is the difference between the time at which this line intersects q=0 and the time at which this line intersects $q=q_{ss}$.	5.2.2.1.2.2
Roll control force	This is the component of applied force exerted by the pilot in the direction to command roll.	5.2.3.8
Satisfactory	See "Qualitative Degrees of Suitability."	

Service ceiling	The service ceiling is the highest altitude at which the maximum rate of climb is 100 ft/min at a given weight and engine thrust.	
Special Failure State	See "Air Vehicle State."	
Spin	Spin is that part of the post-stall air vehicle motion which is characterized by a sustained yaw rotation. The spin may be upright or inverted, flat (high angle of attack) or steep (low but still stalled angle of attack); and the rotary motions may have oscillations in pitch, roll, and yaw superimposed on them. The incipient spin is the initial, transient phase of the motion during which it is not possible to identify the spin mode, usually followed by the developed spin—the phase during which it is possible to identify the spin mode.	5.2.5 through 5.2.5.6.4
Stall speed	At 1g normal to the flight path, the highest (in equivalent airspeed) of : a. speed for steady straight flight at $C_{L_{max}}$, the first local maximum of the curve of lift coefficient ($L/\bar{q}S$) versus angle of attack which occurs as C_L is increased from zero b. speed at which uncommanded pitching, rolling, or yawing occurs (5.2.5.2) c. speed at which intolerable buffet or structural vibration is encountered	
Static margin	The static margin is the distance between the c.g. and the neutral point.	
Steady wind	The steady wind is the mean wind speed. In the absence of a wind shear, the mean wind speed and direction are constant.	5.1.2
Takeoff	Takeoff includes the ground run, rotation, and liftoff, the ensuing acceleration to V_{max}(TO), and the transient caused by assist cessation. Takeoff encompasses operation both in and out of ground effect.	
Takeoff thrust	This is the maximum thrust available for takeoff.	
Thrust	Thrust is the driving force exerted on the air vehicle by its rotating propeller or rotor blades, jet engine or engines, or other propulsive device or force.	
Tolerable	See "Qualitative Degrees of Suitability."	

Transient Peak Ratio (TPR)	Transient peak ratio is a time-domain measure of damping, determined by plotting the pitch rate time response to a step input. Δq_1 is the difference between the maximum value of pitch rate at the first peak and q_{ss}, and Δq_2 is the difference between the minimum value of pitch rate at the second peak and q_{ss}. The TPR is the ratio $\Delta q_2/\Delta q_1$.	5.2.2.1.2.2
Transition region	The transition region is defined as the range of speeds between hover and the maximum at which conversion is initiated (inbound) or complete (outbound), including all of the conversion process.	
Uncommon atmospheric disturbances	These are Levels of atmospheric disturbances which may be encountered infrequently in operations (i.e., probability of exceeding these levels is approximately 1×10^{-3}).	5.1.2
Von Karman turbulence	The von Karman form of the spectra for random turbulence velocities	5.1.2
Wind shear	The gradient of change in mean wind speed and/or direction as a function of time or position	5.1.2
Yaw-control pedal force	The difference push-force components of forces exerted by the pilot on the yaw control pedals	5.2.3.8

3.3 Symbols.

a	Break frequency	5.1.2
a	Inverse time constant of a first-order polynomial	
A	Amplitude	
a_p	Digital filter break frequency for p_g component of Dryden turbulence	5.1.2
a_q	Digital filter break frequency for q_g component of Dryden turbulence	5.1.2
a_r	Digital filter break frequency for r_g component of Dryden turbulence	5.1.2
a_u	Digital filter break frequency for longitudinal gust component of Dryden turbulence	5.1.2
a_v	Digital filter break frequency for lateral gust component of Dryden turbulence	5.1.2
a_w	Digital filter break frequency for vertical gust component of Dryden turbulence	5.1.2
a_y	Lateral acceleration	
a_{y_p}	Lateral acceleration at the pilot's station	
A_{y_p}	Lateral acceleration at the pilot's station	
a_{z_p}	Vertical acceleration at the pilot's station	
A_γ	High-frequency gain of the flight path angle response to control input	
A_θ	High-frequency gain of the pitch attitude response to longitudinal control input	
b	Wing span	
c	Control stick displacement	
C	Magnitude of a steady crosswind	5.2.3.10
$\bar{c}$	Mean aerodynamic chord	
CAP	Control Anticipation Parameter	5.2.2.1.1.2
CAP′	Control Anticipation Parameter attenuated by higher-order dynamics	5.2.2.1.1.2
CAP_e	Equivalent Control Anticipation Parameter, i.e., CAP determined from equivalent system parameters	5.2.2.1.1.2

C_d	Amplitude of the Dutch roll component of sideslip following a step roll control input
C_D	Nondimensional drag coefficient
C_{Du}	Nondimensional coefficient of drag due to longitudinal velocity
$C_{D\alpha}$	Nondimensional coefficient of drag due to angle of attack
$C_{D\delta}$	Nondimensional coefficient of drag due to control surface defleciton
$C_{hr\beta}$	Nondimensional coefficient of rudder hinge moment due to sideslip angle
$C_{hr\delta r}$	Nondimensional coefficient of rudder hinge moment due to rudder deflection
C_{ℓ}	Nondimensional rolling moment coefficient
C_L	Nondimensional lift coefficient
C_{L_1}	$C_{L_1} = W/(\bar{q}\,S)$
$C_{L_{max}}$ or $C_{L_{MAX}}$	Maximum lift coefficient
C_{ℓ_p}	Nondimensional coefficient of rolling moment due to roll rate
C'_{ℓ_p}	Nondimensional coefficient of rolling moment due to roll rate modified to account for kinematic inertial coupling
C_{ℓ_r}	Nondimensional coefficient of rolling moment due to yaw rate
C'_{ℓ_r}	Nondimensional coefficient of rolling moment due to yaw rate modified to account for kinematic inertial coupling
C_{Lstall}	Lift coefficient at stall angle of attack, α_S, defined in 5.2.5.2
C_{Lu}	Nondimensional coefficient of lift due to longitudinal velocity
$C_{L\alpha}$	Nondimensional coefficient of lift due to angle of attack
C_{ℓ_β}	Nondimensional coefficient of rolling moment due to sideslip angle
C'_{ℓ_β}	Nondimensional coefficient of rolling moment due to sideslip angle modified to account for kinematic inertial coupling
$C^*_{\ell_\beta}$	Nondimensional coefficient of rolling moment due to sideslip angle modified to account for rudder hinge moments
$C_{L\delta}$	Nondimensional coefficient of lift due to control deflection

$C_{\ell_{\delta_a}}$	Nondimensional coefficient of rolling moment due to aileron deflection
$C'_{\ell_{\delta a}}$	Nondimensional coefficient of rolling moment due to aileron deflection modified to account for kinematic inertial coupling
$C_{\ell_{\delta_{as}}}$	Nondimensional coefficient of rolling moment due to lateral stick (or wheel) deflection
$C_{\ell_{\delta r}}$	Nondimensional coefficient of rolling moment due to rudder deflection
$C'_{\ell_{\delta r}}$	Nondimensional coefficient of rolling moment due to rudder deflection modified to account for kinematic inertial coupling
C_m	Nondimensional pitching moment coefficient
C_{m_0}	Pitching moment coefficient at zero angle of attack
C_{m_q}	Nondimensional coefficient of pitching moment due to pitch rate
C_{mu}	Nondimensional coefficient of pitching moment due to longitudinal velocity
$C_{m\alpha}$	Nondimensional coefficient of pitching moment due to angle of attack
$C_{m_{\dot{\alpha}}}$	Nondimensional coefficient of pitching moment due to rate of change of angle of attack
C_{m_δ}	Nondimensional coefficient of pitching moment due to control surface deflection
C_n	Nondimensional yawing moment coefficient
C_N	Nondimensional normal force coefficient
C_{n_p}	Nondimensional coefficient of yawing moment due to roll rate
C'_{n_p}	Nondimensional coefficient of yawing moment due to roll rate modified to account for kinematic inertial coupling
C_{n_r}	Nondimensional coefficient of yawing moment due to yaw rate
C'_{n_r}	Nondimensional coefficient of yawing moment due to yaw rate modified to account for kinematic inertial coupling
$C_{N\alpha}$	Nondimensional coefficient of normal force due to angle of attack
C_{n_β}	Nondimensional coefficient of yawing moment due to sideslip angle
C'_{n_β}	Nondimensional coefficient of yawing moment due to sideslip angle modified to account for kinematic inertial coupling

$C^{*}_{n_\beta}$	Nondimensional coefficient of yawing moment due to sideslip angle modified to account for rudder hinge moments	
$C_{n_{\delta_a}}$	Nondimensional coefficient of yawing moment due to aileron deflection	
$C'_{n_{\delta_a}}$	Nondimensional coefficient of yawing moment due to aileron deflection modified to account for kinematic inertial coupling	
$C_{n_{\delta_{as}}}$	Nondimensional coefficient of yawing moment due to lateral stick (or wheel) deflection	
$C_{n_{\delta_r}}$	Nondimensional coefficient of yawing moment due to rudder deflection	
$C'_{n_{\delta_r}}$	Nondimensional coefficient of yawing moment due to rudder deflection modified to account for kinematic inertial coupling	
COST	Cost function of the match between a lower-order equivalent system and a higher-order system	5.2.2.1.1
C_y	Nondimensional lateral force coefficient	
C_{y_β}	Nondimensional coefficient of lateral force due to sideslip angle	
$C^{*}_{y_\beta}$	Nondimensional coefficient of lateral force due to sideslip angle modified to account for rudder hinge moments	
$C_{y_{\delta a}}$	Nondimensional coefficient of lateral force due to aileron deflection	
$C_{y_{\delta r}}$	Nondimensional coefficient of lateral force due to rudder deflection	
$C_{y_{\delta rp}}$	Nondimensional coefficient of lateral force due to rudder pedal deflection	
D	Drag	
DB_θ	Pitch attitude dropback as originally defined by Gibson, the difference between the value of pitch attitude at the time a pulse input is removed and the final pitch attitude.	5.2.2.1.2.2
D_i	Ride Discomfort Index	5.2.1.8
d_m	Generalized discrete gust length (always positive)	5.1.2
f	Frequency	
F	Applied force	
F_a	Force applied to the control stick by the pilot to command roll	
F_{as} or F_{AS}	Force applied to the control stick by the pilot to command roll	
$F_{as_{MAX}}$	Maximum force applied to the control stick by the pilot to command roll during a manuever	

F_{DFC}	Force applied to the direct force controller by the pilot	
F_e	Force applied to the control stick by the pilot to command pitch	
F_{es}	Force applied to the control stick by the pilot to command pitch	
fnsp	Short-period natural frequency	
F_r	Force applied to the rudder pedals by the pilot	
F_{rp}	Force applied to the rudder pedals by the pilot	
F_s	Force applied to the control stick by the pilot	
F_s/n	Gradient of steady-state pitch control force versus n at constant speed	5.2.2.1.8.1
$(F_s/n)_{max}$	Maximum pitch maneuvering force gradient	5.2.2.1.8.1
$(F_s/n)_{min}$	Minimum pitch maneuvering force gradient	5.2.2.1.8.1
$(F_s/n)_{ss}$	Steady-state pitch maneuvering force gradient	
F_s/δ or F_s/δ_s	Force/deflection gradient	
f_t	Truncation frequency beyond which aeroelastic responses are no longer significant in turbulence	5.2.1.8
F_w	Force applied to control wheel by the pilot	
F_w/n	Wheel force/response gradient	
g	Acceleration of gravity	
G_{HOS}	Gain of the higher-order system at a given frequency	5.2.2.1.1
G_{LOES}	Gain of the lower-order equivalent system at a given frequency	5.2.2.1.1
G_{HOS_q}	Gain of the pitch response of the higher-order system at a given frequency	5.2.2.1.1
G_{HOS_α}	Gain of the angle-of-attack response of the higher-order system at a given frequency	5.2.2.1.1
G_{LOES_q}	Gain of the pitch response of the lower-order equivalent system at a given frequency	5.2.2.1.1
G_{LOES_α}	Gain of the angle-of-attack response of the lower-order equivalent system at a given frequency	5.2.2.1.1
g_y	Lateral component of acceleration	
g_z	Vertical component of acceleration	
GM	Gain margin	5.2.2.1.1
h or H	Altitude	

$\dot{h}$	Rate of change of altitude (climb rate or sink rate)	
h_c	Calibrated test pressure altitude	
H_e	Engine angular momentum	
h_L	Limit altitude, the maximum attainable altitude	4.1.4.3
h_m	Maneuver margin	
h_{max}	Service ceiling, the maximum altitude boundary for the ROTH	4.1.4.2
h_n	Static margin	
$\dot{h}_0$	Sink rate at flare initiation	
$h_{o_{max}}$	Maximum altitude boundary for the ROSH	4.1.4.1
$h_{o_{min}}$	Minimum altitude boundary for the ROSH	4.1.4.1
I_e	Moment of inertia of the engine	
i_T	Thrust incidence angle	
I_{xp}	Moment of inertial of the aircraft about the principal x-axis	
I_{xx}	Moment of inertia of the aircraft about the x-axis (body axes or stability axes)	
I_{xy}	Product of inertia of the aircraft with respect to the x-y plane (body axes or stability axes)	
I_{xz}	Product of inertia of the aircraft with respect to the x-z plane (body axes or stability axes)	
I_{yy}	Moment of inertia of the aircraft about the y-axis (body axes or stability axes)	
I_{yz}	Product of inertia of the aircraft with respect to the y-z plane (body axes or stability axes)	
I_{zp}	Moment of inertia of the aircraft about the principal z-axis	
I_{zz}	Moment of inertia of the aircraft about the z-axis (body axes or stability axes)	
j	$\sqrt{-1}$	
k	Ratio of the bank angle attained to the bank angle required in the stated time of 5.2.3.5 and 5.2.3.5.1 for the Class, Flight Phase Category, and Flying Qualities Level under consideration	5.2.3.2.5
K	Gain	
K_c	Controlled element gain	

K_d	Amplitude of the Dutch roll component of roll rate response to a lateral step input (See figure 105.)	5.2.3.1.1
K_e	Gain on an error signal	
K_F	Forward-loop gain	
K_p	Gain of the pilot's forward-path transfer function	
K_{pm}	Command gain on roll rate, scheduled with dynamic pressure, in the DIGITAC flight control system (See figure 178.)	
K_{q_e}	Equivalent high-frequency gain on pitch rate response to control input	5.2.2.1.1
K_R	Amplitude of the roll mode component of roll rate response to a lateral step input (See figure 105.)	5.2.3.1.1
K_s	Sensor gain	
K_S	Amplitude of the spiral mode component of roll rate response to a lateral step input (See figure 105.)	5.2.3.1.1
k_x	Normalized radius of gyration of the aircraft with respect to the x axis	
k_y	Normalized radius of gyration of the aircraft with respect to the y axis	
k_z	Normalized radius of gyration of the aircraft with respect to the z axis	
K_{α_e}	Equivalent high-frequency gain on angle-of-attack response to longitudinal control input	5.2.2.1.1
K_{β_e}	Equivalent high-frequency gain on sideslip response to directional control input	5.2.3.1
K_θ	High-frequency gain on pitch response to control input	
K_{ϕ_e}	Equivalent high-frequency gain on bank angle response to lateral control input	5.2.3.1
$K_{\dot{\phi}_{\delta_{AS}}}$	High-frequency gain on bank angle rate response to lateral stick input	
L	Lift force, aerodynamic lift plus the thrust component normal to the flight path	
L	Rolling moment	
L	Scale length of random turbulence	5.1.2
ℓ_b	Distance between an equivalent point-mass bobweight and the c.g., positive for the point-mass ahead of the c.g.	
LCDP	Lateral Control Divergence Parameter	5.2.5.5

ℓ_c	Effective control arm of the longitudinal control surface, positive forward from the c.g. (negative for an aft tail)	5.2.2.1.6
ℓ_{CR}	Distance between the c.g. and the pitch center of rotation measured along the x-axis, positive for the center of rotation ahead of the c.g.	
$L_{F_{as}}$	Dimensional stability derivative of rolling moment with respect to lateral stick force, the high-frequency gain on roll rate response to lateral stick force	
$L'_{F_{as}}$	Dimensional stability derivative of rolling moment with respect to lateral stick force modified to account for kinematic inertial coupling	
$L_{F_{as_e}}$	Equivalent high-frequency gain on roll rate response to lateral stick force	
L_p	Dimensional stability derivative of rolling moment with respect to roll rate	
L'_p	Dimensional stability derivative of rolling moment with respect to roll rate modified to account for kinematic inertial coupling	
L_p	Dimensional stability derivative of rolling moment with respect to roll rate	
L'_r	Dimensional stability derivative of rolling moment with respect to yaw rate modified to account for kinematic inertial coupling	
L_u	Scale length for longitudinal gust, u_g	5.1.2
L_v	Dimensional stability derivative of rolling moment with respect to side velocity	
L_v	Scale length for lateral gust, v_g	5.1.2
L_w	Scale length for vertical gust, w_g	5.1.2
L_w	Dimensional stability derivative of lift with respect to vertical velocity	
ℓ_x	Distance between the c.g. and the pilot's station measured along the x-axis	
L_α	Dimensional stability derivative of lift with respect to angle of attack	
L_β	Dimensional stability derivative of rolling moment with respect to sideslip angle	
L'_β	Dimensional stability derivative of rolling moment with respect to sideslip angle modified to account for kinematic inertial coupling	

L_{δ}	Dimensional stability derivative of rolling moment with respect to control deflection	
L_{δ_a}	Dimensional stability derivative of rolling moment with respect to aileron deflection	
L'_{δ_a}	Dimensional stability derivative of rolling moment with respect to aileron deflection modified to account for kinematic inertial coupling	
$L_{\delta_{as}}$	Dimensional stability derivative of rolling moment with respect to lateral stick input	
$L'_{\delta_{as}}$ or $L'_{\delta_{AS}}$	Dimensional stability derivative of rolling moment with respect to lateral stick input modified to account for kinematic inertial coupling	
$L_{\delta_{rp}}$	Dimensional stability derivative of rolling moment with respect to rudder pedal input	
$L'_{\delta_{rp}}$	Dimensional stability derivative of rolling moment with respect to rudder pedal input modified to account for kinematic inertial coupling	
m	Aircraft mass	
M	Pitching moment	
M	Mach number	
M_{aero}	Aerodynamic pitching moment	
M_{F_s} or $M_{F_{es}}$	Initial pitch acceleration per pound of longitudinal stick force	
M_m		
M_{max}	Maximum service Mach number, the maximum Mach number boundary for the ROTH	4.1.4.2
M_{min}	Minimum service Mach number, the minimum Mach number boundary for the ROTH	4.1.4.2
M_u	Dimensional stability derivative of pitching moment with respect to forward velocity	
M_w	Dimensional stability derivative of pitching moment with respect to vertical velocity	
$M_{\dot{w}}$	Dimensional stability derivative of pitching moment with respect to vertical acceleration	
M_{α}	Dimensional stability derivative of pitching moment with respect to angle of attack	
M_{δ}	Dimensional stability derivative of pitching moment with respect to control surface deflection	

$M_{\delta e}$	Dimensional stability derivative of pitching moment with respect to elevator deflection	
n	Load factor, measured at the c.g. unless the location is otherwise specified	
N	Yawing moment	
n_c	Commanded load factor	
n_f	Load factor normal to the flight path	
n_L	Symmetrical flight limit load factor for a given Air Vehicle Normal State, based on structural considerations	
$n_{L_{max}}$	Maximum limit load factor (positive-g limit load factor), the maximum load factor boundary for the RORH for a given altitude	4.1.4.3
$n_{L_{min}}$	Minimum limit load factor (negative-g limit load factor), the minimum load factor boundary for the RORH for a given altitude	4.1.4.3
n_{max}	Maximum service load factor, the maximum load factor for the ROTH for a given altitude (See figure 4.)	4.1.4, 4.1.4.2
n_{min}	Minimum service load factor, the minimum load factor for the ROTH for a given altitude (See figure 4.)	4.1.4, 4.1.4.2
$n_{o_{max}}$	Maximum load factor for the ROSH for a given altitude (See figure 4.)	4.1.4, 4.1.4.1
$n_{o_{min}}$	Minimum load factor for the ROSH for a given altitude (see figure 4.	4.1.4, 4.1.4.1
n_{o-}	For a given altitude, the lower load factor boundary in the V-n diagrams depicting the ROSH (See figure 4.)	4.1.4, 4.1.4.1
n_{o+}	For a given altitude, the upper load factor boundary in the V-n diagrams depicting the ROSH (See figure 4.)	4.1.4, 4.1.4.1
N_m	Maneuver point	
N_p	Dimensional stability derivative of yawing moment with respect to roll rate	
N'_p	Dimensional stability derivative of yawing moment with respect to roll rate modified to account for kinematic inertial coupling	
N_r	Dimensional stability derivative of yawing moment with respect to yaw rate	
N'_r	Dimensional stability derivative of yawing moment with respect to yaw rate modified to account for kinematic inertial coupling	

n_y	Lateral load factor	
$N_{Y_{c.g.}}$	Side acceleration at the center of gravity	
$n_{y_{p0}}$	Initial lateral load factor at the pilot's station	
$n_{y\,pilot\,max}$	Maximum lateral load factor at the pilot's station	5.2.3.2.4
n_z or N_z	Normal load factor measured at the instantaneous center of rotation for pitch control inputs	
$n_{z_{peak}}$	Peak normal acceleration in the flare	5.2.2.2.2
$n_{z_{ss}}$	Steady-state normal acceleraton following a unit step stick force input	
n/α or n_z/α	Steady-state normal acceleration change per unit change in angle of attack for an incremental pitch control deflection at constant speed (airspeed and Mach number)	5.2.2.1.1.2
N_β	Dimensional stability derivative of yawing moment with respect to sideslip angle	
N'_β	Dimensional stability derivative of yawing moment with respect to sideslip angle modified to account for kinematic inertial coupling	
N_{δ_a}	Dimensional stability derivative of yawing moment with respect to aileron deflection	
$N_{\delta_{as}}$	Dimensional stability derivative of yawing moment with respect to lateral stick input	
N'_{δ_a}	Dimensional stability derivative of yawing moment with respect to aileron deflection modified to account for kinematic inertial coupling	
$N'_{\delta_{as}}$ or $N'_{\delta_{AS}}$	Dimensional stability derivative of yawing moment with respect to lateral stick input modified to account for kinematic inertial coupling	
$N_{\delta_{rp}}$	Dimensional stability derivative of yawing moment with respect to rudder pedal deflection	
$N'_{\delta_{rp}}$	Dimensional stability derivative of yawing moment with respect to rudder pedal deflection modified to account for kinematic inertial coupling	
n_-	For a given altitude, the lower load factor boundary in the V-n diagrams depicting the ROTH (See figure 4.)	4.1.4, 4.1.4.2
n_+	For a given altitude, the upper load factor boundary in the V-n diagrams depicting the ROTH (See figure 4.)	4.1.4, 4.1.4.2

$N_{\delta}^{a_z}$	Numerator of the a_z/δ transfer function	
N_{δ}^{u}	Numerator of the u/δ transfer function	
$N_{\delta_{as}}^{\beta}$	Numerator polynomial of the sideslip to lateral stick input transfer function	
$N_{\delta_{rp}}^{\beta}$	Numerator polynomial of the sideslip to rudder pedal input transfer function	
$(N_{\delta_{as}}^{\beta} / N_{\delta_{rp}}^{\beta})_{aug}$	Ratio of the numerator polynomials of sideslip to lateral stick and sideslip to rudder pedal transfer functions for the augmented aircraft	5.2.3.2.7
N_{δ}^{γ}	Numerator polynomial of the γ/δ transfer function	
N_{δ}^{θ}	Numerator polynomial of the θ/δ transfer function	
$N_{\delta_{stk}}^{\phi}$	Numerator polynomial of the bank angle to lateral stick input transfer function	
p	Perturbation roll rate about the x-axis	
P	Total roll rate (steady-state plus perturbation)	
$\dot{p}$	Roll acceleration	
$\dot{p}_0$	Initial roll acceleration	5.2.3.2.7
p_1	Roll rate at the first peak following a yaw-control-free step roll control command	5.2.3.2, 5.2.3.2.3
p_2	Roll rate at the second peak (the first minimum) following a yaw-control-free step roll control command	5.2.3.2, 5.2.3.2.3
p_3	Roll rate at the third peak following a yaw-control-free step roll control command	5.2.3.2
p_c	Commanded roll rate	
p_{cr}	Critical roll rate, the roll rate at which the aircraft becomes unstable due to inertial coupling	5.2.4.3
p_g	Rotational atmospheric disturbance velocity about the x-axis	5.1.2
p_{max}	Maximum roll rate	5.2.3.2.4
p_{osc}/p_{av}	A measure of the ratio of the oscillatory component of roll rate to the average component of roll rate in response to a yaw-control-free step roll control command	5.2.3.2, 5.2.3.2.1
p_{ss}	Steady-state roll rate	
$P_{SS\ MAX}$	Maximum steady-state roll rate	
p_{∞}	Steady-state roll rate following a step roll control input	

$(p/F_{as})_{ss}$ or $(p/F_{AS})_{ss}$	Sensitivity of steady-state roll rate to lateral stick input	
$\overline{PR}$	Average pilot rating	
q	Perturbation pitch rate	
Q	Total pitch rate	
$\bar{q}$	Dynamic pressure	
$\dot{q}$	Pitch acceleration	
$\dot{q}_0$	Instantaneous pitch acceleration in response to a step input	
q_g	Rotational atmospheric disturbance velocity about the y-axis	5.1.2
$\bar{q}_{LO}$	Dynamic pressure at lift-off speed	
q_{max}	Maximum value of pitch rate following a step or pulse input.	5.2.2.1.2.2
$\dot{q}_{max}$	Maximum pitch acceleration in response to a step input	
q_{peak}	Maximum value of pitch rate following a step or pulse input	5.2.2.1.2.2
q_{ss}	Steady-state pitch rate following a step input.	5.2.2.1.2.2
r	Perturbation yaw rate	
R	Total yaw rate	
$\dot{r}$	Yaw acceleration	
$\dot{r}_0$	Initial yaw acceleration	5.2.3.2.7
$(\dot{r}_0/\dot{p}_0)_{step}$	The ratio of initial yawing acceleration to initial rolling acceleration for a step roll control input (equivalent to $N'_{\delta_{as}}/L'_{\delta_{as}}$)	5.2.3.2.7
r_g	Rotational atmospheric disturbance velocity about the z-axis	5.1.2
R_{H_Y}	Cross track component of perceived hit point (air-to-ground gun attack)	
s	Laplace operator	
S	Aircraft wing area	
$\overline{S}$	Magnitude of the average slope of the θ/F_s transfer function in dB/oct over the interval from 1 to 6 rad/sec	5.2.2.1.5
t	Time	
T	Thrust	
T	Time iteration interval in a digital simulation	
T	Modal time constant	

t^*	Midpoint of the control input transient; used as t = 0 for time-domain requirements specifying response to a step input	
t_0	Time at which a failure occurs	5.2.1.5
$t_{0.5\Delta\gamma_{max}}$	Rise time to 50 percent of peak flightpath change in response to a step change in the designated flight path controller (See $t_{R_{\gamma T}}$)	
t_1	Time at which a pilot is alerted to the occurrence of a failure by a suitable cue	5.2.1.5
t_1	Effective time delay, the time at which a straight line tangent to the pitch rate response to a step input at its maximum slope intersects the time axis (with time measured from the instant of the input)	5.2.2.1.2.2
T_1	Time constant of the control system lead element in the Neal-Smith configurations	
$t_{1_{roll}}$	Effective roll time delay, the time at which a straight line tangent to the roll rate response to a step input at its maximum slope intersects the time axis (with time measured from the instant of the input)	
$T_{1/2}$	Time to half amplitude	
t_2	Time at which the pilot first moves the controls to commence recovery from a failure transient	5.2.1.5
t_2	Time at which a straight line tangent to the pitch rate response to a step input at its maximum slope intersects the value of the steady-state pitch rate (with time measured from the instant of the input)	5.2.2.1.2.2
T_2	Time constant of the control system lag element in the Neal-Smith configurations	
T_2	Time to double amplitude based on the value of an unstable root. $T_2 = -(\ln 2)/\zeta\omega_n$ for oscillations, $T_2 = -(\ln 2)\ T$ for first-order divergences, where T is the modal time constant.	
t_{30}	Time to roll through 30º	
t_{90}	Time to roll through 90º	
t_{360}	Time to roll through 360º	
t_{40°	Time to roll through 40º	
t_{50°	Time to roll through 50º	
t_{90°	Time to roll through 90º	

$t_{360°}$	Time to roll through 360º	
T_a	Time constant of the actuator ram	
T_{cs}	Acceleration transmissibility at the crew station	5.2.1.8
T_d	Damped period of the Dutch roll oscillation; $T_d = \frac{2\pi}{\omega_d \sqrt{1-\zeta_d^2}}$	5.2.3.2.1, 5.2.3.2.5
T_{eff}	Effective time constant of command-path plus forward-path control loop elements such as prefilters and actuators	
T_F	Filter time constant	
T_F	Flare-mode time constant	
T_{h1}	Time constant of the first-order zero in the $\dot{h}/\delta_e$ transfer function	
T_{LAG}	Time constant of a first-order lag	
T_{LEAD}	Time constant of a first-order lead	
t_{n_β}	Time for the Dutch roll oscillation in the sideslip response to reach the n^{th} local maximum for a right step or pulse roll-control command, or the n^{th} local minimum for a left command	5.2.3.2.1
T_{p_1}	Time constant of the lead term in the pilot's forward-path transfer function	5.2.2.1.4
T_{p_2}	Time constant of the lag term in the pilot's forward-path transfer function	5.2.2.1.4
TPR	Transient Peak Ratio	5.2.2.1.2.2
t_q	Pitch rate peak time, time at which pitch rate reaches its maximum value following a step or pulse input	5.2.2.1.2.2
$t_{\dot{q}}$	Pitch acceleration peak time, time at which pitch acceleration reaches its maximum value following a step or pulse input	5.2.2.1.2.2
T_R	Time constant of the first-order roll mode, positive for a stable mode	5.2.3.1.1
T_{R_e}	Equivalent roll mode time constant	5.2.3.1, 5.2.3.1.1
$t_{R\gamma T}$	Effective flight path rise time	5.2.2.2.1
T_S	Time constant of the first-order spiral mode, positive for a stable mode	5.2.3.1.3
T_{S_e}	Equivalent spiral mode time constant	5.2.3.1, 5.2.3.1.3

T_{sp_1}	Time constant of the stable airframe subsidence pole in the open-loop transfer function of the longitudinal response of an RSS aircraft	
T_{sp_2}	Time constant of the unstable airframe subsidence pole in the open-loop transfer function of the longitudinal response of an RSS aircraft	
T_{u_1}	Time constant of the first-order zero in the u/δ transfer function	
T_{u_4}	Time constant of the random ship-wake disturbance contribution to the longitudinal component of gust velocity in the carrier landing disturbance model	5.2.6
T_{β_1}	Time constant of the first first-order zero in the sideslip response to directional control input	
$T_{\beta_{1e}}$	Equivalent time constant of the first first-order zero in the sideslip response to directional control input	5.2.3.1
$T_{\beta_{2e}}$	Equivalent time constant of the second first-order zero in the sideslip response to directional control input	5.2.3.1
$T_{\beta_{3e}}$	Equivalent time constant of the third first-order zero in the sideslip response to directional control input	5.2.3.1
$T_{\beta_{as}}$	Lateral stick (or wheel) sideslip time constant	
$T_{\beta_{rp}}$	Rudder pedal sideslip time constant	
t_γ	Flight path delay	
T_{θ_1}	Long-term (or low-frequency) numerator time constant in the transfer function for pitch response to longitudinal control input	5.2.2.1.1
$T_{\theta_{1e}}$	Equivalent long-term numerator time constant of the pitch response to longitudinal control input	5.2.2.1.1
T_{θ_2}	Short-term (or high-frequency) numerator time constant in the transfer function for pitch response to longitudinal control input, also known as flight path-attitude lag	5.2.2.1.1
$T_{\theta_{2e}}$	Equivalent short-term numerator time constant of the pitch response to longitudinal control input	5.2.2.1.1
$(1/T_{\theta_2})_{eff}$	Effective $1/T_{\theta_2}$, the lowest frequency where $\Phi(\gamma/\theta) = -45^\circ$ for pitch control inputs	
$t_{\phi=30^\circ}$	Time to roll through 30º	

T_{ϕ_1}	Time constant of the non-minimum phase first-order root in the numerator of the bank angle to control deflection transfer function	
u	Perturbation longitudinal velocity	
u	Generic input	
U	Total longitudinal velocity	
U_0	Total initial velocity in the x-axis direction	
u_1	Random free-air turbulence contribution to the longitudinal component of gust velocity in the carrier landing disturbance model	5.2.6
u_2	Ship-wake turbulence contribution to the longitudinal component of gust velocity in the carrier landing disturbance model	5.2.6
u_3	Periodic ship-motion-induced turbulence contribution to the longitudinal component of gust velocity in the carrier landing disturbance model	5.2.6
u_4	Random ship-wake disturbance contribution to the longitudinal component of gust velocity in the carrier landing disturbance model	5.2.6
u_{20}	Mean wind speed at 20 ft above the ground	5.1.2
u_a	Component of airspeed in the direction of the x-axis	
u_g	Longitudinal component of gust velocity, positive forward	5.1.2
u_w	Mean wind speed	5.1.2
v	Perturbation side velocity	
V	Airspeed along the flight path	
V	Ground speed (inertial velocity)	
$\dot{v}$	Lateral translational acceleration	
V_0	Total initial velocity	
v_1	Random free-air turbulence contribution to the lateral component of gust velocity in the carrier landing disturbance model	5.2.6
v_4	Random ship-wake disturbance contribution to the lateral component of gust velocity in the carrier landing disturbance model	5.2.6
v_a	Component of airspeed in the direction of the y-axis	
V_a	Airspeed	

V_A	Maneuvering speed	
V_c	Calibrated airspeed	
V_C	Catapult end airspeed	
$V_{C_{min}}$	Minimum catapult end airspeed	5.2.2.1.7.5.1
V_{end}	Speed for maximum endurance	
v_g	Lateral component of gust velocity, positive to the pilot's right	5.1.2
V_G	Ground speed	
V_G	Gust penetration limit speed	
$v_{g_{PAST}}$	Past value of v_g in digital filter implementation of r_g component of Dryden turbulence	5.1.2
V_H	Level flight maximum speed, the maximum continuous level-flight speed in the basic configuration at the basic-flight-design gross weight commensurate with the specified operational use of the airplane	
V_i	Indicated airspeed	
V_L	Limit airspeed, the maximum attainable airspeed	
V_{LF}	Takeoff, approach, and landing limit airspeed	
V_{LO}	Lift-off speed	
v_m	Generalized discrete gust intensity, positive along the positive axes	5.1.2
V_{MAT}	Maximum level speed with maximum augmented thrust	
V_{max}	Maximum service speed, the maximum speed boundary for the ROTH for a given altitude.	4.1.4.2
V_{MC}	Airborne minimum control speed (as used by FAA)	5.2.3.11
V_{MCA}	Airborne minimum control speed	5.2.3.11
V_{MCG}	Minimum controllable speed while on the ground	5.2.3.11.1
V_{min}	Minimum service speed, the minimum speed boundary for the ROTH for a given altitude.	4.1.4.2
$V_{min_{power-off}}$	Minimum service speed with power off	
V_{MRT}	Maximum level speed with military rated thrust	
V_{NRT}	Maximum level speed with normal rated thrust	
$V_{o_{max}}$	Maximum speed boundary for the ROSH for a given altitude	4.1.4.1
$V_{o_{min}}$	Minimum speed boundary for the ROSH for a given altitude	4.1.4.1
v_p	Lateral velocity at the pilot's station	

$V_{PA\ min}$	Minimum approach speed	
V_{range}	Speed for maximum range in zero wind conditions	
$V_{R/C}$	Speed for maximum rate of climb	
V_{ref}	Reference speed	
V_S	Stall speed (equivalent airspeed), at 1g normal to the flight path.	4.1.4.3, 5.2.5.2
V_{SG}		
V_T	True airspeed	
$\dot{V}_w$	Time rate of change of the magnitude of a horizontal wind shear	
$V_{w/d}$	Magnitude of the wind over the aircraft carrier deck	5.2.6
w	Acceleration weighting function for Ride Discomfort Index	5.2.1.8
w	Vertical velocity	
W	Aircraft weight	
W	Weighting coefficient	
W	Wind speed	
w_1	Random free-air turbulence contribution to the vertical component of gust velocity in the carrier landing disturbance model	5.2.6
w_2	Ship-wake turbulence contribution to the vertical component of gust velocity in the carrier landing disturbance model	5.2.6
w_3	Periodic ship-motion-induced turbulence contribution to the vertical component of gust velocity in the carrier landing disturbance model	5.2.6
w_4	Random ship-wake disturbance contribution to the vertical component of gust velocity in the carrier landing disturbance model	5.2.6
w_g	Vertical component of gust velocity, positive down	5.1.2
$w_{g_{PAST}}$	Past value of w_g in digital filter implementation of q_g component of Dryden turbulence	5.1.2
x or X	Distance from the origin measured along the longitudinal axis, positive forward	
X	Horizontal distance between the c.g. and the main gear axle	
X	Distance from air vehicle c.g. to ship center of pitch, negative aft of ship	5.2.6
x_{cg}	Location of the center of gravity along the x-axis	

X_{MP}	Distance along the x-body axis between the c.g. and the pilot's station	
x_p	Distance along the x-axis between the c.g. and the pilot's station	
X_{PCR}	Distance along the x-body axis between the instantaneous center of pitch rotation and the pilot station	
X_u	Dimensional stability derivative of longitudinal force with respect to longitudinal velocity	
X_w	Dimensional stability derivative of longitudinal force with respect to vertical velocity	
X_α	Dimensional stability derivative of longitudinal force with respect to angle of attack	
X_{δ_e}	Dimensional stability derivative of longitudinal force with respect to elevator deflection	
y	Distance from the origin measured along the lateral axis, positive right	
Y	Lateral force	
Y	Vertical distance between the c.g. and the main gear axle	
Y_c	Transfer function of the commanded element, the element of the pilot-vehicle system that the pilot's feedback loops are wrapped around, in the case of an aircraft, the aircraft and it's flight control system	5.2.2.1.4
Y_{CF}	Transfer function of a pilot's lateral stick or wheel crossfeed to the rudder pedals to coordinate a turn	5.2.3.2.7
Y'_{CF}	Lower-order approximation of the transfer function of a pilot's lateral stick or wheel crossfeed to the rudder pedals to coordinate a turn	5.2.3.2.7
$Y_{CF}(3)$	The magnitude at t = 3 sec of the time response of Y_{CF} to a unit step input	5.2.3.2.7
Y_p	Pilot compensation, the pilot's forward-path transfer function	5.2.2.1.4
y_T	Distance between the thrust line and the cg measured along the y axis	
Y_v	Dimensional stability derivative of lateral force with respect to lateral velocity	
Y_γ	Pilot's forward-path gain on flight path error	
$Y_{\delta a}$	Dimensional stability derivative of lateral force with respect to aileron deflection	

$Y_{\delta_{as}}$	Dimensional stability derivative of lateral force with respect to lateral stick or wheel input	
$Y_{\delta_{rp}}$	Dimensional stability derivative of lateral force with respect to rudder pedal input	
$Y^{*}_{\delta_{rp}}$	Dimensional stability derivative of lateral force with respect to rudder pedal input divided by airspeed	
$Y_{\delta_{SF}}$	Dimensional stability derivative of lateral force with respect to side force control surface deflection	
Y_{θ}	Pilot's forward-path gain on pitch attitude error	
z	Distance from the origin measured along the vertical axis, positive down	
z	z-transform operator	
z_0	Reference height for wind scalar shear	5.1.2
z_p	Distance along the z-axis between the c.g. and the pilot's station	
z_{p_s}	Distance between the pilot station and the stability x-axis, negative for the pilot above the x-axis	
z_T	Distance between the thrust line and the c.g. measured along the z-axis	
Z_u	Dimensional stability derivative of vertical force with respect to longitudinal velocity	
Z_w	Dimensional stability derivative of vertical force with respect to vertical velocity	
Z_{α}	Dimensional stability derivative of vertical force with respect to angle of attack	
Z_{δ}	Dimensional stability derivative of vertical force with respect to control surface deflection	
$Z_{\delta e}$	Dimensional stability derivative of vertical force with respect to elevator deflection	
α	Angle of attack; the angle in the plane of symmetry between the fuselage reference line and the tangent to the flight path at the aircraft center of gravity	
α_0	Initial angle of attack	
$\dot{\alpha}_g$	Rate of change of angle of attack due to atmospheric turbulence, positive opposite right hand rule	5.1.2
α_{gust}	Change of angle of attack due to atmospheric disturbance	

α_S	Stall angle of attack at constant speed for the configuration, weight, center-of-gravity position and external store combination associated with a given Air Vehicle Normal State	5.2.5.2
α_{ss}	Steady-state change in angle of attack following a step input	
β	Sideslip angle at the center of gravity, the angle between the undisturbed flow and the plane of symmetry; positive, or right, sideslip corresponds to incident flow approaching from the right side of the plane of symmetry.	
$\dot{\beta}$	Rate of change of sideslip angle	
β_d	Dutch roll component of sideslip response following a step lateral control input	
γ	Flight path angle, positive for climbing flight $\gamma = \sin^{-1}$ (vertical speed/true airspeed)	
$\dot{\gamma}$	Rate of change of flight path angle	
γ_0	Initial flight path angle	
γ_{aero}	Aerodynamic flight path angle	
γ_c	Commanded flight path angle	
$\gamma_{inertial}$	Inertial flight path angle	
γ_{max} or γ_{MAX}	Maximum achievable flight path angle	
γ_{min} or γ_{MIN}	Minimum achievable flight path angle	
γ_{PA}	Normal power approach flight path angle	
γ_{ss}	Steady-state flight path angle following a step input	
$\dot{\gamma}_{ss}$	Steady-state rate of change of flight angle following a step input	
γ_{trim}	Trim flight path angle	
γ_ε	Error between commanded flight path angle and actual flight path angle	
δ	Control surface deflection	
$\dot{\delta}$	Control surface deflection rate	
δ_a	Aileron deflection	
$\delta_{a_{avg}}$	Average aileron deflection during a specified maneuver	
δ_{ac} or δ_{a_c}	Commanded aileron deflection	
$\delta_{a_{max}}$	Maximum aileron deflection	

δ_{as} or δ_{AS}	Deflection of the cockpit roll controller	
$\delta_{a_{step}}$	Step roll control input	
δ_{DFC}	Direct force cockpit control deflection	
δ_e	Elevator deflection	
δ_{es}	Longitudinal controller deflection	
δ_F	Flaperon deflection	
δ_{FL}	Left flaperon deflection	
$\dot{\delta}_{FQ}$	Control surface deflection rate needed to achieve the pitching moment specified by the CAP′ flying qualities requirement	
δ_{FR}	Right flaperon deflection	
δ_{HT}	Horizontal tail deflection	
δ_{max}	Maximum (i.e., most positive) control surface position	
δ_{MAX}	Maximum control surface position (positive or negative)	
δ_{min}	Minimum (i.e., most negative) control surface position	
$\dot{\delta}_{pr}$	Control surface deflection rate needed to counter the pitching moment due to roll-yaw coupling while rolling about the x-stability axis	
δ_r	Rudder deflection	
$\delta_{r_{max}}$	Maximum rudder deflection during a specified maneuver	
δ_{rp}	Rudder pedal deflection	
δ'_{rp}	Rudder pedal deflection produced by the pilot's lateral stick or wheel crossfeed to coordinate a turn multiplied by $N'_{\delta_{rp}} / L'_{\delta_{as}}$	5.2.3.2.7
$\delta_{rp}(0)$	Initial rudder pedal deflection required to coordinate a unit step lateral stick (or wheel) input	5.2.3.2.7
$\delta_{rp}(3)$	Rudder pedal deflection required to coordinate a unit step lateral stick (or wheel) input at t = 3 sec after the input	5.2.3.2.7
$\delta'_{rp}(3)$	Yaw-to-roll crossfeed parameter; $\delta_{rp}(3)$ multiplied by $N'_{\delta_{rp}} / L'_{\delta_{as}}$	5.2.3.2.7
δ_s	Stick deflection	
δ_S	Stabilator deflection	
δ_{SL}	Left stabilator deflection	
δ_{SR}	Right stabilator deflection	

$\dot{\delta}_{stab}$	Control surface deflection rate needed to provide stability for a RSS aircraft	
δ_T	Throttle deflection	
δ_w	Wheel deflection angle	
$\delta_{w_{eff}}$	Effective wheel angle, wheel angle for maximum rolling moment	
Δ	Prefix indicating a change in a parameter	
Δ	Denominator polynomial of a transfer function	
Δcentrifugal force	Incremental centrifugal force	
$\Delta\dot{h}$	Change in climb rate	
Δm	Incremental mass	
ΔM	Incremental pitching moment	
Δn	Change in load factor	
Δn_c	Commanded increment of normal acceleration	
Δn_z	Change in vertical acceleration	
$\Delta n_{z_{ss}}$	Steady-state change in normal acceleration following a step input	
Δq_1	Difference between maximum pitch rate and steady-state pitch rate following a step input	5.2.2.1.2.2
Δq_2	Difference between pitch rate at the first minimum and steady-state pitch rate following a step input	5.2.2.1.2.2
$\Delta q_2/\Delta q_1$	Transient Peak Ratio	5.2.2.1.2.2
Δt	Effective rise time, the difference between t_2 and t_1	5.2.2.1.2.2
ΔT	Change in thrust	
$\Delta\beta_{max}$	Maximum change in sideslip occurring within 2 sec or within one half-period of the Dutch roll mode, whichever is greater, following a step roll control command (figure 119 through figure 121d)	5.2.3.2, 5.2.3.2.5, 5.2.3.2.6
$\Delta\gamma$	Change in flight path angle	
$\Delta\gamma_{ss_{down}}$	Steady-state downward change in flight path angle following a step input of the designated flight path controller	5.2.2.2.2
$\Delta\gamma_{max}$	Peak change in flight path angle following a step input of the designated flight path controller	5.2.2.2.1
$\Delta\gamma_{min}$	Greatest downward change in flight path angle	

$\Delta\gamma_{ss}$	Steady-state change in flight path angle following a step input of the designated flight path controller	5.2.2.2.1, 5.2.2.2.2
$\Delta\gamma_{ss_{up}}$	Steady-state upward change in flight path angle following a step input of the designated flight path controller	5.2.2.2.2
$\Delta\delta_{FQ}$	Control surface deflection margin needed to achieve the pitching moment specified by the CAP′ flying qualities requirement	
$\Delta\delta_{marg}$	Control surface deflection margin needed to counter pitching moments due to stochastic disturbances such as turbulence and sensor noise	
$\Delta\delta_{pr}$	Control surface deflection margin needed to counter the pitching moment due to roll-yaw coupling while rolling about the x-stability axis.	
$\Delta\delta_{stab}$	Control surface deflection margin needed to provide stability for a RSS aircraft	
$\Delta\delta_{tran}$	Control surface deflection margin needed to provide the pitching moment required for transient agility and stability	
$\Delta\phi_{2\omega_{180}}$	Difference between phase angle at ω_{180} and phase angle at twice the ω_{180} frequency	5.2.2.1.3.1, 5.2.3.12.1
$\Delta\tau_e$	Additional time delay	
ε	Angle between the principal x axis and the flight path measured in the x-z plane	
ε	Difference between ground speed heading and airspeed heading	
ε	Inclination of the engine rotor axis to the principal x axis	
ζ	Damping ratio	
ζ_a	Damping ratio of the zeroes of the vertical acceleration to longitudinal control input transfer function	
ζ_d	Damping ratio of the second-order Dutch roll mode	5.2.3.1.2
ζ_{d_e}	Equivalent Dutch roll damping	5.2.3.1, 5.2.3.1.2, 5.2.3.2.1, 5.2.3.2.2
ζ_p	Damping ratio of the phugoid mode	5.2.2.1.1.1
ζ_{p_e}	Equivalent phugoid damping	2.2.1.1, 5.2.2.1.1.1
ζ_{RS}	Damping ratio of a second-order coupled roll-spiral mode	5.2.3.1.4
ζ_{sp}	Damping ratio of the short-period mode	5.2.2.1.1.2
$\zeta_{sp_{cl}}$	Closed-loop short-period damping	

ζ_{sp_e}	Equivalent short-period damping	5.2.2.1.1, 5.2.2.1.1.2
ζ_{α_e}	Equivalent numerator damping ratio of the angle-of-attack response to longitudinal control input	5.2.2.1.1
ζ_{ϕ}	Damping ratio of the numerator quadratic in the transfer function for bank angle response to lateral control input	
ζ_{ϕ_e}	Equivalent damping ratio of the numerator quadratic in the bank angle response to lateral control input	5.2.3.1, 5.2.3.1.2
η	Normally-distributed random number	
η_1	Normally-distributed random number input to digital filter implementation of u_g component of Dryden turbulence	5.1.2
η_2	Normally-distributed random number input to digital filter implementation of v_g component of Dryden turbulence	5.1.2
η_3	Normally-distributed random number input to digital filter implementation of w_g component of Dryden turbulence	5.1.2
η_4	Normally-distributed random number input to digital filter implementation of p_g component of Dryden turbulence	5.1.2
θ	Pitch attitude, the angle between the x axis and the horizontal plane	
$\dot{\theta}$	Rate of change of pitch attitude	
$\ddot{\theta}$	Acceleration of change of pitch attitude	
$\ddot{\theta}_0$	Instantaneous acceleration of change of pitch attitude	
$\ddot{\theta}_{max_{HOS}}$	Maximum pitch acceleration of the higher-order system following a unit step stick force input	5.2.2.1.1.2
θ_c	Commanded pitch attitude	
θ_{DB}	Pitch attitude dropback, the difference between the peak pitch attitude and the steady-state pitch attitude following cessation of a pulse input.	5.2.2.1.2.2
θ_e	Error between commanded pitch attitude and actual pitch attitude	
θ_s	Ship pitch amplitude	5.2.6
θ_ε	Error between commanded pitch attitude and actual pitch attitude	
$\lvert\theta/\theta_c\rvert_{MAX}$	Closed-loop resonance	5.2.2.1.4
λ	Root of a first-order polynomial	
λ_R	First-order roll mode pole, $\lambda_R = -1/T_R$	

λ_S	First-order spiral mode pole, $\lambda_S = -1/T_S$	
Λ	Wing sweepback angle	
μ	$\mu = (W/S)/(g\,\rho\,\bar{c})$	
μ	Coefficient of rolling friction of the main landing gear	
μ	Rudder shaping parameter, the separation between the lateral stick (or wheel) and rudder pedal sideslip zeros, normalized by the rudder pedal sideslip zero $\mu = \delta_{rp}(3) - 1$ where $\delta_{rp}(3)$ is the rudder pedal deflection that would be required for perfect coordination 3 sec after a step roll control input	5.2.3.2.7
ξ	Inclination of the thrust line to the axis of motion, $\xi = \alpha + i_T$	
π	Pi, the ratio of the circumference of a circle to its diameter	
ρ	Air density	
ρ_{50}	Standard atmospheric air density at 50,000 ft	5.1.2
σ	Difference between mean wind heading and ground speed heading	
σ	Real part of the root of a polynomial	
σ	Root-mean-square disturbance intensity, where $\sigma^2 = \int_0^\infty \Phi(\Omega)\,d\Omega = \int_0^\infty \phi(\omega)\,d\omega$	5.1.2
σ_p	Root-mean-square intensity of p_g	5.1.2
σ_{PR}	Standard deviation of a set of pilot ratings	
σ_q	Root-mean-square intensity of q_g	5.1.2
σ_r	Root-mean-square intensity of r_g	5.1.2
σ_s	Root-mean-square intensity of sensor noise	
σ_u or σ_{u_g}	Root-mean-square intensity of the longitudinal component of gust velocity	5.1.2
σ_{u_4}	Root-mean-square intensity of the random ship-wake disturbance contribution to the longitudinal component of gust velocity in the carrier landing disturbance model	5.2.6
σ_v	Root-mean-square intensity of the lateral component of gust velocity	5.1.2
σ_w	Root-mean-square intensity of the vertical component of gust velocity	5.1.2

σ_δ	Root-mean-square of control surface deflection response	
$\sigma_{\dot{\delta}}$	Root-mean-square of control surface deflection rate response	
σ_η	Variance of normally-distributed random number source	5.1.2
τ	Time delay	
τ_e	Equivalent time delay	
τ_p	Pilot's time delay	5.2.2.1.4
τ_{α_e}	Equivalent time delay of angle-of-attack response to longitudinal control input	5.2.2.1.1
τ_{β_e}	Equivalent time delay of sideslip response to directional control input	5.2.3.1, 5.2.3.1.2, 5.2.3.1.6
τ_{γ_e}	Equivalent time delay of flight path response to longitudinal control input	
τ_{θ_e}	Equivalent time delay of pitch response to longitudinal control input	5.2.2.1.1, 5.2.2.1.1.1, 5.2.2.1.1.2
τ_{θ_p}	Pitch attitude phase delay	5.2.2.1.3.1
τ_{ϕ_e}	Equivalent time delay of bank angle response to lateral control input	5.2.3.1, 5.2.3.1.2, 5.2.3.1.5
ϕ	Bank angle, measured in the y-z plane, between the y-axis and the horizontal plane	
ϕ or Φ	Phase angle	
ϕ_1	Bank angle change in 1 sec in response to an abrupt roll control command initiated from steady bank angles or from wings-level flight, with time measured from the initiation of the control force application	
ϕ_1	Bank angle at the first peak following a pedals-free impulse aileron control command	5.2.3.2.2
ϕ_2	Bank angle at the second peak following a pedals-free impulse aileron control command	5.2.3.2.2
ϕ_3	Bank angle at the third peak following a pedals-free impulse aileron control command	5.2.3.2.2
$\phi_{1_{max}}$	Maximum bank angle change in 1 sec	
$\phi_{2\omega_{180}}$	Phase angle at a frequency twice that of the ω_{180} frequency in a Bode plot of frequency response	5.2.2.1.3.1

$\phi_{a_{z_p}}(\omega_c)$	Phase angle of normal acceleration at the pilot's station evaluated at ω_c	5.2.2.1.5
ϕ_c	Commanded bank angle	
ϕ_e	Error between commanded bank angle and actual bank angle	
ϕ_{osc}/ϕ_{av}	A measure of the ratio of the oscillatory component of bank angle to the average component of bank angle following a pedals-free impulse roll control command	5.2.3.2.2
ϕ_t	Bank angle change in a given time, t, in response to an abrupt roll control command initiated from steady bank angles or from wings-level flight, with time measured from the initiation of the control force application	5.2.3.5, 5.2.3.5.1
$\phi_{t\,command}$	Bank angle actually attained in the stated time for a given step roll command with yaw control pedals employed as specified in 5.2.3.5 and 5.2.3.5.1	5.2.3.2.5
$\phi_{t\,requirement}$	Bank angle required to be attained in the stated time for a given step roll command with yaw control pedals employed as specified in 5.2.3.5 and 5.2.3.5.1	5.2.3.2.5
$\lvert\phi/\beta\rvert_d$	At any instant, the ratio of the amplitudes of the bank angle and sideslip angle envelopes of the Dutch roll mode	5.2.3.1.2, 5.2.3.2.1, 5.2.3.2.5
Φ_{HOS}	Phase angle of the higher-order system	5.2.2.1.1
Φ_{HOS_q}	Phase angle of the pitch response of the higher-order system	5.2.2.1.1
Φ_{HOS_α}	Phase angle of the angle-of-attack response of the higher-order system	5.2.2.1.1
Φ_{LOES}	Phase angle of the lower-order equivalent system	5.2.2.1.1
Φ_{LOES_q}	Phase angle of the pitch response of the lower-order equivalent system	5.2.2.1.1
Φ_{LOES_α}	Phase angle of the angle-of-attack response of the lower-order equivalent system	5.2.2.1.1
Φ_M	Phase margin	5.2.2.1.3.1, 5.2.3.12.1
Φ_{u_l}	Spectrum for the random free-air turbulence contribution to the longitudinal component of gust velocity in the carrier landing disturbance model	5.2.6
Φ_{u_g}	Spectrum for the longitudinal component of gust velocity, where $\Phi_{u_g}(\Omega) = V\phi_{u_g}(\omega)$	5.1.2

Φ_{v_1}	Spectrum for the random free-air turbulence contribution to the lateral component of gust velocity in the carrier landing disturbance model	5.2.6
Φ_{v_g}	Spectrum for the lateral component of gust velocity, where $\Phi_{v_g}(\Omega) = V\phi_{v_g}(\omega)$	5.1.2
Φ_{w_1}	Spectrum for the random free-air turbulence contribution to the vertical component of gust velocity in the carrier landing disturbance model	5.2.6
Φ_{w_g}	Spectrum for the vertical component of gust velocity, where $\Phi_{w_g}(\Omega) = V\phi_{w_g}(\omega)$	5.1.2
χ	Heading of the airspeed vector	
χ_a	Heading of the airspeed vector	
ψ	Yaw angle	
$\dot{\psi}$	Rate of change of sideslip angle	
ψ_G	Heading of the ground speed vector	
ψ_p	Phase angle of the Dutch roll oscillation in roll rate response to a lateral step input	
ψ_W	Heading of the mean wind (See figure 190.)	
ψ_β	Phase angle expressed as a lag for a cosine representation of the Dutch roll oscillation in sideslip	5.2.3.2, 5.2.3.2.1, 5.2.3.2.2, 5.2.3.2.5
ω	Temporal frequency, where $\omega = \Omega V$	
ω_3	Undamped natural frequency of the second-order control system lag in the Neal-Smith configurations	
ω_{135}	Frequency corresponding to -135º of phase in a Bode plot of frequency response	5.2.2.1.3.1
ω_{180}	Frequency corresponding to -180º of phase in a Bode plot of frequency response	5.2.2.1.3.1, 5.2.3.12.1
ω_a	Undamped natural frequency of the zeroes of the vertical acceleration to longitudinal control input transfer function	
ω_{BW}	Bandwidth frequency, with stated minimum phase and gain margins (generally 60º and 6 dB respectively).	5.2.2.1.3.1, 5.2.2.1.3.3, 5.2.3.12.1
$\omega_{BW_{gain}}$	Frequency at which amplitude is 6 dB greater than the amplitude at ω_{180} in a Bode plot of frequency response	5.2.2.1.3.1, 5.2.3.12.1

$\omega_{BW_{hT}}$	Bandwidth of climb rate response to throttle or other designated flight path controller	
$\omega_{BW_{phase}}$	ω_{135}, the frequency corresponding to -135º of phase in a Bode plot of frequency response	5.2.2.1.3.1, 5.2.3.12.1
$\omega_{BW_{\gamma}}$	Flight path bandwidth	5.2.2.1.3.3
$\omega_{BW_{\theta}}$	Pitch attitude bandwidth	5.2.2.1.3.1, 5.2.2.1.3.3
ω_c	Crossover frequency	
ω_c	Smith-Geddes criterion frequency	5.2.2.1.5, 5.2.3.3
ω_{cs}		
ω'_{cs}	Stick-free undamped natural frequency of the elevator feel system	
$\omega_{c_{\phi}}$	Bank angle crossover frequency (bank angle bandwidth)	
$\omega_{c_{\psi}}$	Heading crossover frequency (heading bandwidth)	5.2.3.12.1
ω_d	Undamped natural frequency of the second-order Dutch roll mode	
ω_{d_e}	Equivalent Dutch roll undamped natrual frequency	5.2.3.1, 5.2.3.1.2
ω_e	Rate of rotation of the engine	
ω_i	Forcing function bandwidth	
ω_n	Undamped natural frequency	
ω_p	Ship pitch frequency	5.2.6
ω_p	Undamped natural frequency of the phugoid mode	
ω_{p_e}	Equivalent phugoid natural frequency	5.2.2.1.1
ω_R	Resonant frequency, any frequency between 1 and 10 rad/sec at which lightly damped oscillations in pitch attitude can result from disturbances	
ω_{RS}	Undamped natural frequency of a second-order coupled roll-spiral mode	5.2.3.1.4
ω_s	Sensor bandwidth	
ω_{sp}	Undamped natural frequency of the short-period mode	5.2.2.1.1.2, 5.2.2.1.3.3

ω'_{sp}	Stick-free undamped natural frequency of the short-period mode	
$\omega_{sp_{cl}}$	Closed-loop short-period natural frequency	
ω_{sp_e}	Equivalent short-period frequency	5.2.2.1.1, 5.2.2.1.1.2
$\omega_{sp_{ol}}$	Open-loop short-period natural frequency	
ω_α	Undamped natural frequency of the zeroes of angle-of-attack response to longitudinal control input	
ω_{α_e}	Equivalent numerator natural frequency of the angle-of-attack response to longitudinal control input	5.2.2.1.1
ω_ϕ	Undamped natural frequency of the numerator quadratic in the transfer function for bank angle response to lateral control input	
ω_{ϕ_e}	Equivalent undamped natural frequency of the numerator quadratic in the bank angle response to lateral control input	5.2.3.1
ω_ψ	Heading bandwidth	5.2.3.12.1
Ω	Spatial (reduced) frequency	
$£^{-1}$	Inverse Laplace operator	
$\angle p/\beta$ or $\angle p/\beta\|_d$	Phase angle between roll rate and sideslip in the free Dutch roll oscillation. Angle is positive when p leads β by an angle between 0 and 180º	5.2.3.2
$\angle_{pc}$	Pilot compensation, the phase angle of the pilot's forward-path lead-lag at the minimum bandwidth frequency	5.2.2.1.4
$\angle\frac{\theta}{\delta}$	Phase angle of the frequency response of pitch attitude to control surface deflection	
∞	Infinity	

4. GENERAL REQUIREMENTS

4.1 Preliminary steps in development and application of flying qualities requirements.

REQUIREMENT RATIONALE

The following paragraphs are needed to guide the application of the rest of the requirements.

4.1.1 Determining operational missions.

The contractor shall derive the flying qualities requirements from the operational mission requirements of the air vehicle as defined by the customer/user. Operational missions include the entire spectrum of intended usage, to include aircrew upgrade and training. These missions shall include all associated Flight Phases and tasks, such as takeoff, takeoff abort, landing, and missed approach. These operational mission requirements shall also include consideration of secondary missions as well as the primary missions, and consideration of any special conditions required, such as day/night, adverse-weather operation.

REQUIREMENT RATIONALE

The very reason for procuring aircraft is to perform one or more missions. In keeping with overall guidance to relate requirements to the intended mission, the mission statement has been found to be necessary to guide selection of flight conditions and to tailor requirements.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.1 and 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 1.3, 1.3.1, and 3.1.1
MIL-STD-1797: 3.1.

The contractor needs a specific mission statement to furnish guidance for consistent selection of quantitative requirements and for interpreting qualitative requirements. Unfortunately, the word "mission" is used in several contexts not only in this standard, but throughout the documents pertinent to a new weapon system acquisition. In the broadest sense, "operational missions" applies to the classification of the aircraft as a fighter, bomber, reconnaissance, etc.; or to "accomplishing the mission" of bombing, strafing, etc. The objective of the mission statement is to define the function of the vehicle in general terms. It shall be sufficient for the contractor to refer to 3.1 of the Joint Service Specification Guide for Air Vehicles (JSSG-2001) to define the overall performance requirements, the operational requirements, and the employment and deployment requirements.

The operational missions considered shall not be based on the design mission profiles alone. However, such profiles serve as a starting point for determining variations that might normally be expected in service, encompassing ranges of useful load, flight time, combat speed and altitude, in-flight refueling, etc., to define the entire spectrum of intended operational use. "Operational missions" include training and ferry missions.

The intended use of an aircraft must be known before the required configurations, loadings, and the Regions of Satisfactory Handling (ROSH) can be defined and the design of the aircraft to meet the guidance of this standard can be undertaken. If additional missions are foreseen at the time the detailed specification is prepared, it is the responsibility of the contractor to define the operational requirements to include these missions. Examples of missions or capabilities that have been added later include in-flight refueling (tanker or receiver), aerial pick-up and delivery, low-altitude penetration and weapon delivery, and ground attack for an air-superiority fighter or vice versa.

Once the intended uses or operational missions are defined, the contractor shall conduct a Flight Phase analysis of each mission. With the Flight Phases established, the contractor can define the configurations and loading states which will exist during each Phase. After the configuration and loading states have been defined for a given Flight Phase, the contractor can determine the Regions of Tolerable and Recoverable Handling (ROTH and RORH, respectively) and more fully define the ROSHs.

The contractor shall assign the air vehicle to one of the six Air Vehicle Classes (table I). The Class designation is used in this standard to help particularize the requirements according to broad categories of intended use. The contractor shall apply the requirements appropriate to the chosen Air Vehicle Class. When no Class is specified in a requirement, the requirement is meant to apply to all Classes. When operational missions so dictate, an aircraft of one Class shall be required by the contractor to meet selected requirements ordinarily specified for aircraft of another Class. The classification scheme simplifies mission definition and aids in interpreting guidance in this standard and in selecting requirements for the Air Vehicle Guide Specification. Basically, the first four classes are related qualitatively to maximum design gross weight and symmetrical flight limit load factor at the basic flight design gross weight, as shown on figure 1. The presentation of figure 1 makes it obvious that highly maneuverable aircraft such as fighter and attack types, together with certain trainer and observation aircraft, shall be designed for high limit load factor. These vehicles tend to group in the weight range from 5,000 to 100,000 lbs. There are a few small, light-weight trainers and observation aircraft which are also designed for fairly high load factors, which could be in either Class I or Class IV. Classification of these aircraft shall be on the basis of more detailed information about the intended use; or alternatively the detail specification shall be a combination of appropriate requirements.

Figure 1 also illustrates that all other aircraft are required to be designed for a limit load factor of less than 4 g, and that current aircraft span the weight range from 1,000 to almost 1,000,000 lbs. In addition, there may be significant differences in the way each vehicle responds to atmospheric turbulence or wind shear. Another factor of possible significance is the location of the pilot in the vehicle relative to the center-of-gravity and the extremities of the vehicle. The location of the pilot in the vehicle affects its motion and ride qualities. If the effects of each of these factors on handling or flying qualities were fully understood and a sufficient data base existed, then the quantitative requirements could be stated as mathematical or empirical functions of the significant factors, and there would be no need for any classification breakdown to accommodate these effects in the specification requirements.

TABLE I. Air Vehicle Classes.

Class I	Small, light air vehicles such as light utility, primary trainer, or light observation.
Class II	Medium weight, low-to-medium maneuverability air vehicles such as heavy utility/search and rescue; light or medium transport/cargo/tanker; early warning/electronic countermeasures/airborne command, control, or communications relay; antisubmarine; assault transport; reconnaissance; tactical bomber; heavy attack; or trainer for Class II.
Class III	Large, heavy, low-to-medium maneuverability air vehicles such as heavy transport/cargo/tanker; heavy bomber; patrol/early warning/electronic countermeasures/airborne command, control, or communications relay; or trainer for Class III.
Class IV	High-maneuverability air vehicles such as fighter/interceptor; attack; tactical reconnaissance; observation; or trainer for Class IV.
Class V	Rotorcraft
Class VI	V/STOL air vehicles

If the characters "-L" follow a class designation, they identify an air vehicle as land-based; carrier-based air vehicles are similarly identified by "-C". When no such differentiation is made in a requirement, the requirement applies to both land-based and carrier-based air vehicles.

It should also be recognized that as vehicles become larger, practical design considerations may dictate compromises between the degree of maneuverability and the values of flying qualities parameters that are desired and what can be accepted, through relaxation of operational requirements or through modification of operational procedures or techniques.

How best to handle the factors discussed above is not completely clear at this time. Ideally the requirements should be expressed as mathematical functions of the significant factors. The current state of knowledge and the experimental data available do not permit this, so it is necessary to make the relatively arbitrary Class definition. Further research into possible scaling parameters, simulation study, and operational expertise is required in this area.

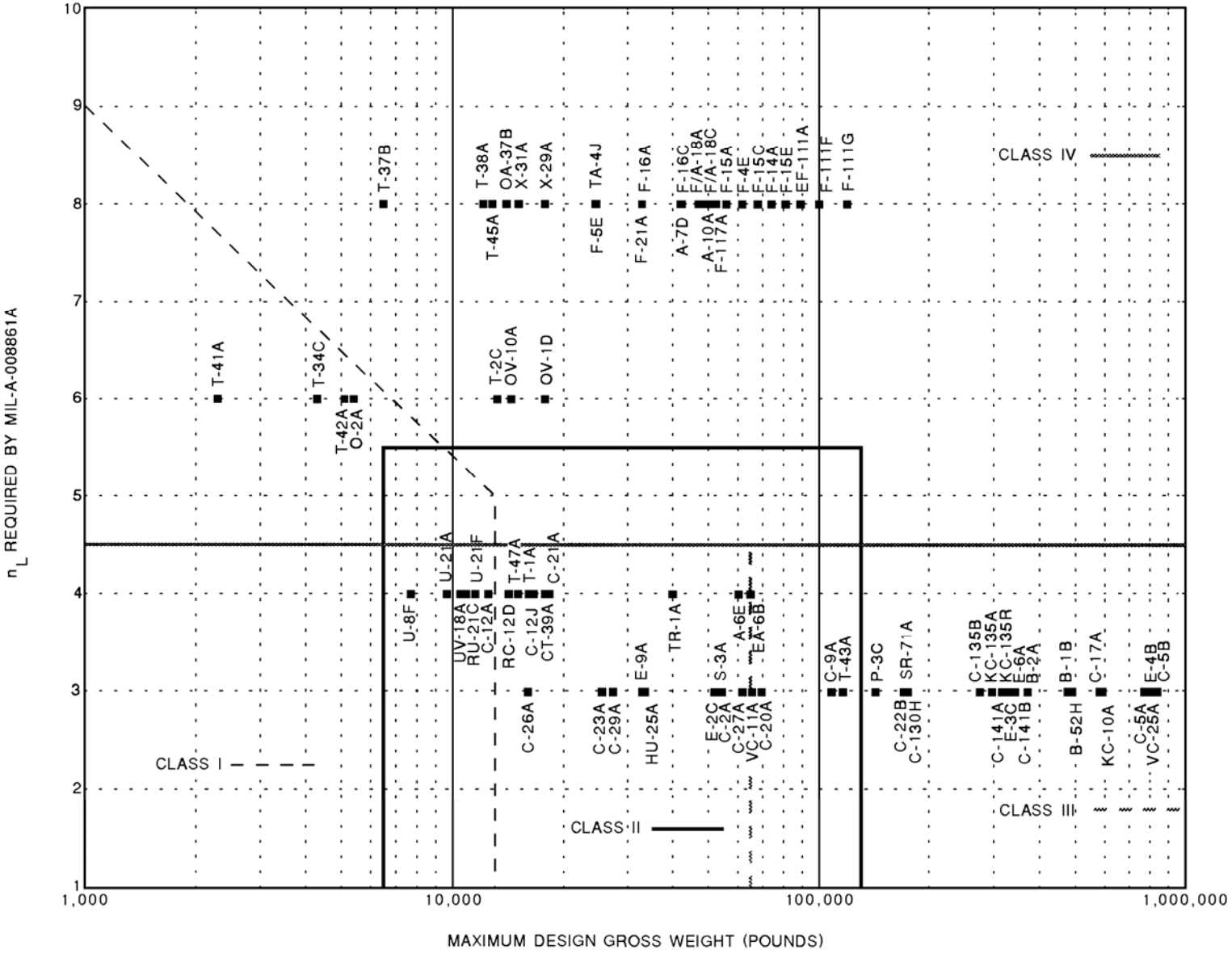


FIGURE 1. Classification of Air Vehicles.

REQUIREMENT LESSONS LEARNED

Operational missions generally depart significantly from the design mission profile, even for the same type of mission. It is important to allow enough latitude to cover likely variations. Also, an air vehicle's operational missions will likely change in both type and detail over its lifetime. There are, of course, tradeoffs with cost, weight, and the like. For a particular procurement the extent of the ROSH beyond minimum operational needs should be as large, then, as these tradeoffs will reasonably permit. While stability and control augmentation can do wonders, such factors as basic control authority and rate, aeroelasticity, and stall speed are limiting at operational extremes and difficult and costly to change after the design freeze. Skimping on the ROSHs, then, can cause difficulties.

When designing for aerial refueling, consideration must be given to interference effects on the receiver from the tanker's flow field. When the air vehicle is a tanker, the air vehicle and its tanker aerial refueling subsystem(s) shall provide an adequately stable platform for the receiver air vehicles to fly behind to permit safe contact and engagement of the tanker's aerial refueling subsystem interface with the appropriate receiver aerial refueling subsystem interface. When the air vehicle is a receiver, it shall have adequate flying qualities and sufficient power within the flow field(s) behind the targeted tanker platform(s) to permit safe contact and engagement of its aerial refueling subsystem interface with the appropriate tanker aerial refueling subsystem interface (s) on the targeted tanker(s).

4.1.2 Determining applicable flight phases.
The contractor shall determine the flight phases (i.e., aerial refueling, terrain following/terrain avoidance, landing, takeoff, etc.) necessary to accomplish the operational missions.

REQUIREMENT RATIONALE

Flying qualities requirements vary for different phases of a mission. To the extent permitted by available data, these variations should be taken into account.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 1.4
MIL-STD-1797: 3.2.

The appropriate Flight Phases to be considered shall be chosen from the Flight Phases defined in table II. Experience with aircraft operations indicates that certain Flight Phases require more stringent values of flying qualities parameters than do others (e.g., air-to-air combat requires more Dutch roll damping than does cruising flight). Also, a given mission Flight Phase will generally have an Air Vehicle Normal State associated with it (e.g., flaps and gear down for landing approach and up for cruising flight).

In flight and simulator evaluations, pilots generally rate a set of flying qualities on suitability for a given mission segment like one of the these Flight Phases. In an appropriate environment, the pilots assign an overall rating based on ability and effort required to perform certain appropriate tasks such as precision tracking of a target or a glide slope, trimming, and making heading changes at constant altitude. The similarity of tasks in many Flight Phases, plus the limited amount of evaluation data on specific Flight Phases, has led to grouping the Phases into three Categories (table II) which are referred to in the recommended values for each requirement. When necessary, the contractor may recategorize, add new Flight Phases, or delineate requirements for special situations; e.g., a zoom climb.

These Flight Phases are to be considered in the context of the total mission so that there will be no gap between successive Phases of any flight, and so that transition will be smooth. In certain cases, a requirement is directed at specific Flight Phases identified in the requirement. When no Flight Phase or Category is stated in a requirement, that requirement is meant to apply to all three Categories.

For the most part, the Flight Phase titles are descriptive enough to facilitate picking those applicable to a given design. The Close Formation Flying (FF) Flight Phase is intended to be used, if desired, where there is no other requirement for rapid maneuvering, precision tracking, or precise flight-path control in up-and-away flight. An example might be a Class I trainer for which the procuring activity desires Category A flying qualities (note the use of the T-37, T-38, etc., in non-training roles).

TABLE II. Flight Phase Categories.

Nonterminal Flight Phases:	
Category A	Those nonterminal Flight Phases that require rapid maneuvering, precision tracking, or precise flight-path control. Included in this Category are: a. Air-to-air combat (CO) b. Ground attack (GA) c. Weapons delivery/launch (WD) d. Aerial recovery (AR) e. Reconnaissance (RC) f. In-flight refueling (receiver) (RR) g. Terrain following (TF) h. Antisubmarine search (AS) i. Close formation flying (FF) j. Low-altitude parachute extraction (LAPES) k. Precision hover (PH)
Category B	Those nonterminal Flight Phases that are normally accomplished using gradual maneuvers and without precision tracking, although accurate flight-path control may be required. Included in this Category are: a. Climb (C) b. Cruise (CR) c. Loiter (LO) d. In-flight refueling (tanker) (RT) e. Descent (D) f. Emergency descent (ED) g. Emergency deceleration (DE) h. Aerial delivery (AD) i. Hover (H) j. Station keeping (SK) k. Hovering translation (HT) l. Accelerating transition (AT) m. Decelerating transition (DT)

Terminal Flight Phases:	
Category C	Terminal Flight Phases are normally accomplished using gradual maneuvers and usually require accurate flight-path control. Included in this Category are: a. Takeoff (TO) b. Catapult takeoff (CT) c. Power approach (PA) d. Waveoff/go-around (WO) e. Landing (L) f. Carrier approach (CA) g. Bolter (B) h. Ground handling (GH) i. Carrier landing (CL) j. Short takeoff (STO) k. Vertical takeoff (VTO) l. Short landing (SL) m. Vertical landing (VL)

Not all of these Flight Phases apply to a given aircraft. Those that are appropriate to design operational missions and emergencies should be chosen for each design. The list of Flight Phases in Section 3 cannot be exhaustive because new mission requirements continue to be generated, thus the contractor may delete some Phases and add others. Choosing applicable Flight Phases, as with selecting appropriate requirements, is initially the contractor's responsibility. The contractor shall make the necessary additions to assure that his list is inclusive and exhaustive for the stated primary and alternate missions. It is the Flying Qualities Working Group's responsibility either to agree with the contractor's suggestions or to recategorize the Flight Phases.

In certain cases, both flying qualities requirements and aircraft capabilities may be less than one would ordinarily expect. An example is a zoom climb—a dynamic maneuver in which qualities such as speed stability and natural frequency can not be measured in flight, and the effectiveness of aerodynamic controls is necessarily low at low dynamic pressure. Insufficient data prevents formulation of general quantitative requirements for these cases, thus the contractor must provide specific requirements as specific mission needs dictate.

For each Flight Phase or Flight Phase Category, (depending upon the data available) typical flight conditions, maneuvers, disturbances, side tasks, etc. have been assumed in setting the suggested numerical values. Accurate flight path control for landing, as an example, may well be a high-gain piloting task to which some Category A requirements apply. In tailoring the requirements for a particular procurement, any envisioned operating conditions more lax or more stringent than normal (such as aerial refueling or landing at zero fuel states in turbulence) shall be taken into account to the extent possible.

REQUIREMENT LESSONS LEARNED

As an example of the last caveat above, consider the A-10 experience documented by Brandeau in AFFDL-TR-78-171. That airplane appeared to meet MIL-F-8785B Level 1 requirements for Category A (which includes ground attack) and it was rated Level 1 during flight tests using a

straight-in approach. Its flying qualities were unsatisfactory, however, when evaluated in an operationally realistic ground attack task.

In close air support, a wide variety of attack maneuvers may be characterized by three general phases, as shown on figure 2:

Target acquisition - rapid rolling toward target while developing 4 to 5 g's; bank and g's held until rollout onto target (return to zero bank and 1 g)

Weapon delivery or tracking/firing - errors eliminated and pipper maintained on target

Break - a gross maneuver to reposition for another attack while looking after aircraft survival.

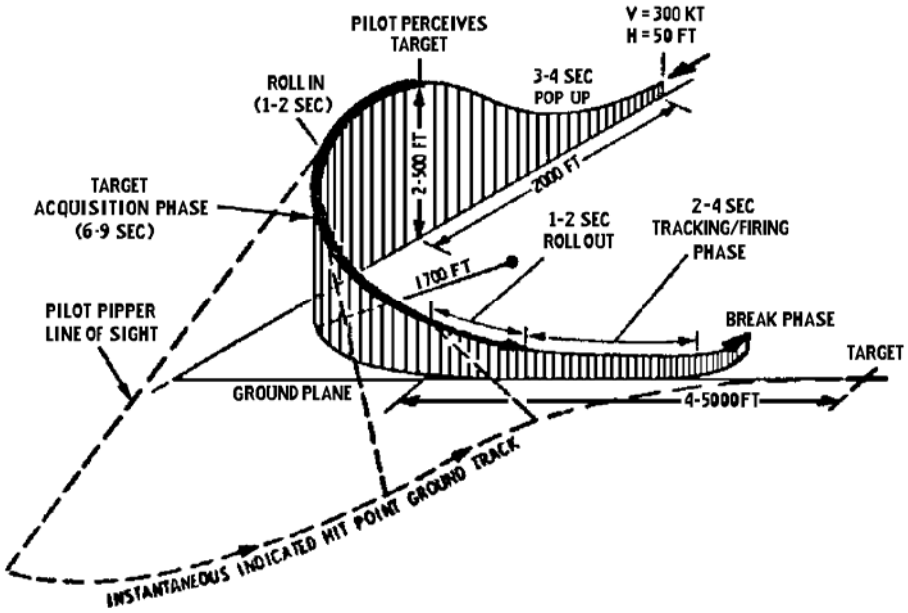


FIGURE 2. Ground attack maneuver scenario.

For gross target acquisition maneuvers, highly predictable terminal orientation of the velocity vector is vital in order to minimize the duration of the relatively vulnerable weapon delivery phase. Excellent roll response is required, in terms of both quickness and maintaining turn coordination. Weapon delivery requires rapid, precise control of the velocity vector for dropping unguided bombs, or of the pipper line of sight (and thus aircraft attitude) for gunnery.

While the original A-10 stability augmentation apparently met MIL-F-8785B requirements on lateral-directional dynamics, and pilots rated it satisfactory in "the originally planned tactical maneuvers… It was only as the maneuvers became very aggressive that the problems surfaced." For these aggressive maneuvers, the average maxima quoted are:

normal acceleration 4.5 g

roll rate	93º/sec
bank angle	93º
tracking time	2.33 sec .

To satisfy the requirements of the task outlined above, the aerodynamic configuration remained unchanged and the flight control system modifications were relatively minor. This will not necessarily be so in more sophisticated designs. The cost of fixing such deficiencies could be very high after a new aircraft has flown, and so it would obviously be beneficial to consider operational maneuvers as early as possible in the design phase. In the example cited, little more than figure 2 would be required as an additional maneuver to be assessed in the Combat Flight Phase in the requirement.

For this more severe maneuver, analysis and simulation may indicate the need for more stringent requirements than the minimums stated for Dutch roll damping and roll-yaw coupling (see figure 3 responses of lateral tracking to a roll-control doublet). The A-10 deficiency was indicated at high g's. Therefore, the lateral-directional characteristics for the GA Flight Phase must be investigated in pull-ups and turns—and roll-sideslip coupling in rapid rolls—as well as in straight flight. (While the requirements of MIL-F-8785B applied throughout the V-h-n Flight Envelopes, often the lateral-directional behavior has been evaluated primarily in 1-g flight.) Commonly it is observed that the amount of aileron-to-rudder crossfeed needed to coordinate turn entries varies considerably with angle-of-attack. Thus, one might find no single crossfeed gain suitable for all phases of the ground attack described.

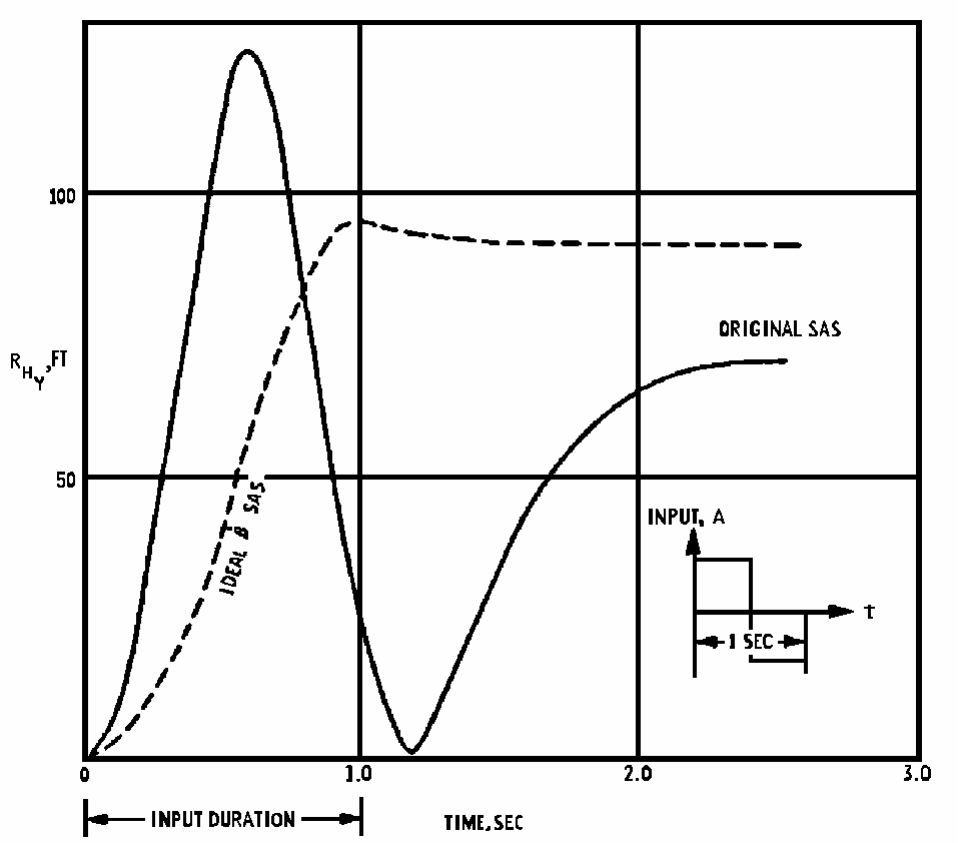


FIGURE 3. Response to a 1-sec, half-stick aileron doublet for tracking scenario.

In addition, advancing flight control technology has greatly increased the potential for tailoring the flying qualities for specific tasks within a Flight Phase Category without compromising other tasks. Truly task-oriented flying qualities would receive an impetus from the inclusion of requirements related to actual operational tasks into the design for a particular aircraft.

An example of the need for better flight characteristics for an added task is the low-altitude parachute extraction mission, in which the pilot must fly precisely at very low altitude. According to an Air Force test pilot, for this task the C-130 is "Level 2 at best, mostly Level 3." The lesson again is to account for the flying qualities implications of changes in operational usage.

To the extent feasible, the requirements have been tailored to particular tasks of the Flight Phases. A very important, but unstated, corollary is the need to avoid inconsistencies in flight control mechanization from one Flight Phase or configuration to another. Drastic or numerous changes in control mode have the potential to confuse the pilot, to the detriment of mission effectiveness or even flight safety. With few exceptions, a single flight technique shall suit all operations.

Vietnam experience showed that fighter aircraft often refuelled or landed with very low fuel states.

4.1.3 Defining Air Vehicle States.
The contractor shall define the Air Vehicle States in terms of internal and external store combinations, loadings, c.g. envelopes, moments and products of inertia, and air vehicle configurations in accordance with 4.1.3.1 through 4.1.3.4. Once defined, each Air Vehicle State shall be categorized as a Normal State, Extreme State, Failure State, or Special Failure State in accordance with 4.1.3.5 through 4.1.3.8.

REQUIREMENT RATIONALE

The requirement is intended to assure that all expected Air Vehicle States are defined and considered, and that the conditions for compliance are sufficiently called out. All Air Vehicle States either necessary or likely to be encountered shall be defined.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.6
MIL-STD-1797: 4.1.5.

The State of the air vehicle is defined by the selected configuration together with the functional status of each of the air vehicle components or systems, throttle setting, weight, moments of inertia, center-of-gravity position, and external stores complement. The trim setting and the position of the pitch, roll, and yaw controls are not included in the definition of Air Vehicle State since they are often specified in the requirements.

REQUIREMENT LESSONS LEARNED

Secondary effects of engine operation may have an important bearing on flying qualities and should not be overlooked in design. These considerations are: the influence of engine gyroscopic moments on airframe dynamic motions; the effects of engine operation (including flameout and intentional shutdown) on characteristics of flight at high angle of attack (5.2.5); and the reduction at low rpm of engine-derived power for operating the flight control system.

Since aeroelasticity, control equipment, and structural dynamics may exert an important influence on the air vehicle flying qualities, such effects should not be overlooked in calculations or analyses directed toward investigation of compliance with the requirements of this standard.

4.1.3.1 Determining internal and external stores.
The contractor shall determine the symmetric and asymmetric store combinations that are required for each mission Flight Phase. The effects of stores on the weight, moments of inertia, center-of-gravity position, and aerodynamic characteristics of the aircraft shall be determined for each store combination.

REQUIREMENT RATIONALE

Stores and store combinations affect the overall ability of the aircraft to meet its mission requirements. The c.g. and aerodynamic effects of internal and external stores need to be considered while verifying compliance with the flying qualities requirements.

REQUIREMENT GUIDANCE

This requirement corresponds to 4.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.4
MIL-STD-1797: 4.1.3.

The requirements of this standard shall apply for all symmetric and asymmetric store combinations required by the operational missions. When the stores contain expendable loads, the requirements of this standard shall apply throughout the range of store loadings, including sloshing and shifting. For external and internal fuel tanks, the critical fuel loading for each requirement shall apply (see 4.1.3.2).

The contractor shall assure that the store combinations evaluated cover all operational flight conditions. In determining the range of store loadings to be evaluated, the contractor shall consider such factors as store mixes, possible points of attachment, and asymmetries -- initial, after each pass, and the result of failure to release. The contractor may find it necessary to propose limitations on store loading to avoid excess design penalties. Since such limitations are operationally restrictive, the procuring activity may be reluctant to approve them.

The contractor shall attempt to assure that there are no restrictions on store placement on the aircraft within the range of design stores. However, it should be recognized that occasionally this goal will be impracticable on some designs. It may be impossible to avoid exceeding aircraft limits, or excessive design penalties may be incurred. Then, insofar as considerations such as standardized stores permit, it shall be made physically impossible to violate necessary store loading restrictions. If this should not be practicable, the contractor shall analyze the effects on flying qualities of violating the restrictions and estimate the likelihood that the restrictions will be exceeded.

While the aircraft is in service, new stores and store combinations will be added that will require clearance. Part of this clearance will be determination of the effect on flying qualities.

See MIL-HDBK-244 for additional guidance.

REQUIREMENT LESSONS LEARNED

High-angle-of-attack testing conducted on the F-15 (AFFTC-TR-75-32) showed a degradation in flying qualities and departure resistance with external stores. Asymmetric stores and internal fuel asymmetry affected departure and spin characteristics. Stores testing with the F-16 (AFFTC-TR-80-29) showed similar results, and serve to illustrate the importance of defining a comprehensive set of conditions for investigating stores effects. For example, during ground taxi of the F-16, "The pilot noticed a leaning or tip-over sensation especially during light weight taxi with a strong crosswind, tight turns, or with asymmetric store loadings."

Stores can have mass, inertial, and aerodynamic effects, typically decreasing both longitudinal and directional aerodynamic stability, increasing moments of inertia and the roll-mode time constant, and increasing susceptibility to departure from controlled flight and the difficulty of recovery. The available control power limits the amount of inertia increase and instability that can be tolerated. Store separation is a prime concern.

4.1.3.2 Defining air vehicle loadings.
The contractor shall define the longitudinal, lateral, and vertical envelopes of center-of-gravity and corresponding weights that will exist for each Flight Phase. These envelopes shall include the most forward and aft center-of-gravity positions. In addition the contractor shall determine the maximum center-of-gravity excursions attainable through failures in systems or components, such as fuel sequencing or hung stores, for each Flight Phase.

REQUIREMENT RATIONALE

Since aircraft characteristics vary with loading, limits must be defined and the loadings known at conditions for demonstration of compliance. The loading of an aircraft is determined by what is in (internal loading) and attached to (external loading) the aircraft. The loading parameters that normally define flying qualities are weight, center-of-gravity position, and moments and products of inertia. External stores affect all of these parameters and also affect aerodynamic coefficients.

REQUIREMENT GUIDANCE

This requirement corresponds to 4.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.2
MIL-STD-1797: 4.1.1.

This standard applies throughout the center-of-gravity envelopes, plus any growth margin, and for the excursions attainable through failures. For normal operation the allowable c.g. range is applicable. Certain failures may cause an adverse c.g. shift. In these cases, the abnormal c.g.'s so attained are applicable. Generally, the mass properties engineers use the SAWE Recommended Practice #7 to determine center-of-gravity positions. The flight control and flying qualities engineers should collaborate with the mass properties engineers in the determination of the center-of-gravity positions.

The requirements apply under all loading conditions associated with an aircraft's operational missions. Since there are an infinite number of possible internal and external loadings, each requirement generally is only examined at the critical loadings with respect to the requirement. Only possible center-of-gravity positions need to be considered for Air Vehicle Normal States. Fuel sequencing, transfer failures or malperformance, and mismanagement that might move the center-of-gravity outside of the established limits shall be considered as Air Vehicle Failure States. The worst possible cases that are not approved Special Failure States (4.1.3.8) must be examined.

The contractor may elect to specify a growth margin in c.g. travel to allow for uncertainties in weight distribution, stability level, and other design factors, and for possible future variations in operational loading and use. The peculiarities of configuration or possible alternative mission tasks may lead to a requirement for additional loadings. Fuel slosh and shift under acceleration may also need consideration.

It is fairly straightforward to determine those longitudinal flying qualities that set the longitudinal c.g. limits, but there are also cases where the aft c.g. limit may be set by lateral-directional flying qualities. Usually supersonic flight at high dynamic pressure is the most critical, because the level of directional stability is reduced due to Mach number and aeroelastic effects. Conditions to be investigated to insure acceptable lateral-directional characteristics at the aft limits include:

a. Roll performance/roll coupling

b. Abrupt engine loss or inlet unstart at one g

c. Abrupt engine loss or inlet unstart at high normal acceleration or angle-of-attack, especially for Class IV aircraft

d. Turbulence effects.

When evaluating aeroelastic effects and effects of hinge moments due to angle-of-attack, the pitch and yaw control surface deflection and sideslip must be taken into account. Pitching moment due to sideslip can also be significant on configurations using vortex lift or highly swept wings. The trend toward relaxed lateral-directional stability indicates that the effect of c.g. on low-speed lateral-directional characteristics shall be examined.

This requirement also requires the contractor to define the lateral c.g. limit. This is especially critical for Class IV aircraft. Conditions to evaluate with asymmetric loadings include:

a. Takeoff with and without crosswind

b. Roll performance/roll coupling

c. Abrupt engine loss at takeoff and in maneuvering flight

d. Dive pullout at high normal acceleration

e. Yaw departure at high angle-of-attack and spin resistance.

In defining this limit, the basic lateral asymmetry due to wing fuel system tolerances and equipment mounted off centerline, such as guns and ammunition, shall be taken into account.

Aircraft weight and balance are estimated during the design. Eventually, weight and balance measurements will be made on the actual vehicle to confirm the estimates. The requirements of this standard shall be applied to each update of the aircraft weight and balance data.

REQUIREMENT LESSONS LEARNED

Lateral asymmetries due to fuel loading can have important effects on trim, stall/post-stall characteristics, etc. Fuel system design has been known to promote such asymmetries, for example, at prolonged small sideslip in cruising flight.

Since the requirements apply over the full range of service loadings, effects of fuel slosh and shifting shall be taken into account in design. Balance, controllability, and airframe and structure dynamic characteristics may be affected. For example, takeoff acceleration has been known to shift the c.g. embarrassingly far aft. Aircraft attitude may also have an effect. Other factors to consider are fuel sequencing, in-flight refueling if applicable, and all arrangements of variable, disposable, and removable items required for each operational mission.

4.1.3.3 Determining moments and products of inertia.
The contractor shall determine the moments and products of inertia of the air vehicle associated with all of the loadings of 4.1.3.2.

REQUIREMENT RATIONALE

Inertial characteristics of the aircraft affect its flying qualities, so the contractor shall define the inertias for all expected loadings corresponding to possible distributions and the loadings for which flying qualities are evaluated. In any dynamic analysis of an aircraft, inertias must be known. The axis in which the values are given must be identified.

REQUIREMENT GUIDANCE

This requirement corresponds to 4.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.3
MIL-STD-1797: 4.1.2.

The requirements of this standard shall apply for all of the moments and products of inertia defined above. By meticulous accounting of weight and balance, inertia values can be estimated fairly accurately. Except for extremely large aircraft, inertias may be measured with equipment located at the Air Force Flight Test Center, Edwards AFB, CA. The procuring activity may, at its discretion, wish to review the methods used in estimating or measuring the inertial characteristics specified. If deemed necessary, checks of estimates can be made by ground tests (e.g., forced oscillation using equipment such as that at the Air Force Flight Test Center) or parameter estimation from flight test data.

REQUIREMENT LESSONS LEARNED

I_{xy} and I_{yz}, normally neglected, may be appreciable for asymmetric loadings.

4.1.3.4 Determining air vehicle configurations.
The contractor shall determine all configurations required or encountered in the applicable Flight Phases of 4.1.2 plus any additional configurations required for performance demonstration and mission accomplishment.

REQUIREMENT RATIONALE

The requirement is intended to assure that all expected aircraft configurations are defined and considered, and that the conditions for compliance are sufficiently called out. All aircraft configurations either necessary or likely to be encountered shall be evaluated.

REQUIREMENT GUIDANCE

This requirement corresponds to 4.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.5
MIL-STD-1797: 4.1.5.

A configuration is defined by the positions and adjustments of the various selectors and controls available to the crew, except for pitch, roll, yaw, throttle, and trim controls. Examples are: the flap control setting and the yaw damper ON or OFF. The requirements of this standard apply for all configurations required or encountered in the applicable Flight Phases of 4.1.2. Switches which activate stability augmentation necessary to meet the requirements of this standard are considered always to be ON unless otherwise specified.

The contractor shall define the configuration or configurations which the aircraft will have during each Flight Phase. This includes the settings of such controls as flaps, speed brakes, landing gear, wing sweep, high-lift devices, and wing incidence that are related uniquely to each aircraft design. The requirement specifies that the configurations to be examined shall include those required for performance demonstration and mission accomplishment. The position of yaw, roll, pitch, trim controls, and the thrust setting are not included in the definition of configuration since the positions of these controls are usually either specified in the individual requirements or determined by the specified flight conditions.

The requirements are stated for Air Vehicle States and Flight Phases, rather than for aircraft configurations. The flying qualities shall generally be a function of the job to be done rather than of the configuration of the aircraft. Special considerations or features may require investigation of additional configurations.

REQUIREMENT LESSONS LEARNED

4.1.3.5 Identifying Air Vehicle Normal States.
The contractor shall identify the Air Vehicle Normal States associated with each of the applicable Flight Phases and tabulate all pertinent parameters. This tabulation shall be in the format and use the nomenclature of table III. Certain parameters, such as weight, moments of inertia, center-of-gravity position, wing sweep, or thrust setting may vary continuously over a range of values during a Flight Phase. The contractor shall replace this continuous variation by a limited number of values of the parameter in question which shall be treated as specific States, and which shall include the most critical values and extremes encountered during the Flight Phase in question.

REQUIREMENT RATIONALE

Definition of Air Vehicle Normal States is basic to application of the flying qualities requirements.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.6.1
MIL-STD-1797: 4.1.6.

Air Vehicle Normal States are Air Vehicle States with no component or system failure. The contractor shall define the Air Vehicle Normal States for each applicable Flight Phase, in the format of table III. If the position of any particular design feature can affect flying qualities independently of the items in table III, its possible positions shall be tabulated as well. Bear in mind that items not normally considered such as setting or automatic operation of engine bypass doors, can affect flying qualities. Initially, the contractor shall tabulate the positions of such features in discrete steps small enough to allow accurate interpolation to find the most critical values or combinations for each requirement. Once these critical cases are found, they shall be added to the tabulation. As discussed under 4.1.3.2, center-of-gravity positions that can be attained only with prohibited, failed, or malfunctioning fuel sequencing need not be considered for Air Vehicle Normal States.

REQUIREMENT LESSONS LEARNED

TABLE III. Air Vehicle Normal States.

Flight Phase		Weight	C.G.	External Stores	Thrust	Thrust Vector Angle	High Lift Devices	Wing Sweep	Wing Incidence	Landing Gear	Speed Brakes	Bomb Bay or Cargo Doors	Stability Augmentation	Other
Takeoff	TO													
Climb	C													
Cruise	CR													
Loiter	LO													
Descent	D													
Emergency Descent	ED													
Emergency Deceleration	DE													
Approach	PA													
Wave-off/ Go-Around	WO													
Landing	L													
Air-to-Air Combat	CO													
Ground Attack	GA													
Weapon Delivery/ Launch	WD													
Aerial Delivery	AD													
Aerial Recovery	AR													
Reconnaissance	RC													
Refuel Receiver	RR													
Refuel Tanker	RT													
Terrain Following	TF													
Antisubmarine Search	AS													
Close Formation Flying	FF													
Catapult Takeoff	CT													

4.1.3.6 Identifying Air Vehicle Extreme States.
The contractor shall identify those Air Vehicle Normal States which shall be treated as Air Vehicle Extreme States. The contractor shall record the rationale for each case.

REQUIREMENT RATIONALE

Levels of flying qualities as defined in 3.2 are employed in this standard in realization of the possibility that the airplane may be required to operate under abnormal conditions. Such abnormalities that may occur as a result of extreme loadings are permitted to comply with a degraded Level of flying qualities. This requirement is a flight safety consideration.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1 and 4.3.11.1 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Air Vehicle Extreme States are Air Vehicle Normal States with extremely heavy loads or extreme asymmetries or unusual aerodynamic characteristics for which some Level 1 requirements may not be met.

REQUIREMENT LESSONS LEARNED

4.1.3.7 Identifying Air Vehicle Failure States.
The contractor shall identify and tabulate all Air Vehicle Failure States which can affect flying qualities. Those malfunctions that result in center-of-gravity positions outside the center-of-gravity envelope defined in 4.1.3.2 shall be included. Each mode of failure considered in a given Flight Phase shall be considered in all subsequent Flight Phases.

REQUIREMENT RATIONALE

Definition of Air Vehicle Failure States is basic to the application of the flying qualities requirements. This tabulation is the starting point for a failure modes and effects analysis, which is necessary in a complex aircraft to assure flying qualities adequate for mission effectiveness and flight safety.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1 and 4.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.6.2
MIL-STD-1797: 4.1.7.

Air Vehicle Failure States consist of Air Vehicle Normal States modified by one or more malfunctions in air vehicle components or systems. Because of the exhaustive work often involved and low confidence in the failure probability calculations, there is a tendency for the contractor to substitute an a priori list of specific failures. If a design is far enough along and not excessively complex, such an approach can work. See the guidance for 5.1.1.3. However, generally comprehensive reliability analysis will be required anyway.

Whether the approach to failure effects on flying qualities is probabilistic, specific, or a combination, failure possibilities of the specific aircraft must be catalogued thoroughly enough to assure adequate mission effectiveness and flight safety.

There is more to determining failure states than just considering each component failure in turn. Two other types of effects must be considered. First, failure of one component in a certain mode may induce other failures in the system, so failure propagation must be investigated. Second, one event may cause loss of more than one part of the system or can affect all channels: a broken bracket, a single crack, a fire, an electrical short, inadequate ground checkout, etc. The insidious nature of possible troubles emphasizes the need for caution in design applications.

Generally this task will amount to identifying pertinent items from the list required by the reliability requirements, and checking for completeness. Although the task may seem formidable, the alternative to a thorough review is a certainty that something important will be overlooked.

REQUIREMENT LESSONS LEARNED

4.1.3.8 Identifying Air Vehicle Special Failure States.
The contractor shall identify those Air Vehicle Failure States that shall be treated as Special Failure States. The contractor shall list all single failures and combinations of failures which are to be considered Special Failure States and record the rationale for each case. These Special Failure States need not be considered in complying with the requirements of this standard.

REQUIREMENT RATIONALE

Certain components, systems, or combinations thereof may have extremely remote probability of failure during a given flight. These failure probabilities may, in turn, be very difficult to predict with any degree of accuracy. Subject to approval by the procuring activity, such failures may be identified as Special Failure States and need not be considered in complying with the requirements of this standard. Definition of Air Vehicle Special Failure States is basic to application of the flying qualities requirements. Perfection is not a realistic expectation. This requirement is to determine the practical limits in each case.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1 and 4.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.6.2.1
MIL-STD-1797: 4.1.7.2.

Special Failure States are Air Vehicle Failure States which have extremely remote probabilities of failure during a given flight. In most cases, a considerable amount of engineering judgement will influence the selection of Air Vehicle Special Failure States. Probabilities that are extremely remote are exceptionally difficult to predict accurately. Judgements will weigh the consequences against the feasibility of improvements or alternatives, and against the projected ability to keep high standards throughout design, qualification, production, use, and maintenance. Meeting other pertinent requirements -- such as flight control and structural requirements -- shall be considered, as shall experience with similar items. Generally, Special Failure States shall be brought to the attention of those concerned with flight safety.

Several categories of Special Failure States can be distinguished. Certain items might be selected more or less categorically:

- Control-stick fracture
- Basic airframe or control-surface structural failure
- Dual mechanical failures in general.

Regardless of the degree of redundancy, there remains a finite probability that all redundant paths will fail. A point of diminishing returns will be reached, beyond which the gains of additional channels are not worth the associated penalties:

- Complete failure of hydraulic or electrical, etc., systems
- Complete or critical partial failure of stability augmentation that has been accepted as necessary to meet Level 3.

Some items might be excepted, if special requirements are met. For example, some limited control shall remain after failure of all engines, provided by accumulators or an auxiliary power source as appropriate.

When considering the admissibility of a Special Failure State on the basis of remote probability, the combined probability of having flying qualities worse than Level 3 -- not just each individual Failure State probability -- must be kept extremely remote.

In the last analysis the contractor is responsible for judging design tradeoffs that bear upon safety. Rather than inhibiting imaginative design, then, this paragraph should be construed as forcing examination of failure possibilities as they affect flight safety through deterioration of flying qualities. The present state of the art can support some properly implemented reliance on stability augmentation to maintain Level 3 flying qualities, but it must be done carefully and for good reason.

REQUIREMENT LESSONS LEARNED

4.1.4 Defining the Regions of Handling.
For each applicable Air Vehicle State, the contractor shall define three Regions of Handling (the ROSH, ROTH, and RORH) in accordance with 4.1.4.1 through 4.1.4.3.

REQUIREMENT RATIONALE

Definition of the Regions of Handling is basic to application of the flying qualities requirements.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1 and 6.4.6 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.7 through 3.1.9
MIL-STD-1797: 4.1.4 through 4.1.4.3.

For each Aircraft Normal State the Flight Envelope is divided into three regions defined in terms of speed, altitude, and load factor, plus any other parameters which may form flight limits (such as sideslip), within which the flying qualities requirements apply. The three regions are:

1. Region of Satisfactory Handling (ROSH)
The boundaries of the ROSH are determined by the speed, altitude, and load factor ranges which the aircraft must be able to achieve in order to accomplish the missions of 4.1.1 with Level 1 flying qualities for normal operation.

2. Region of Tolerable Handling (ROTH)
The boundaries of the ROTH are based on speed, altitude, and load factor limits which are derived from aircraft performance margins, such as service ceiling, at or beyond the limits of the ROSH and within which the aircraft must have at least Level 2 flying qualities for normal operation.

3. Region of Recoverable Handling (RORH)
The boundaries of the RORH encompass all regions of the Flight Envelope in which operation of the aircraft is both allowable and possible, and which the aircraft is capable of safely encountering.

The boundaries of these regions shall implicitly include the ranges of other parameters, such as sideslip, which can be expected within the speed, altitude, and load factor bounds. As a general policy, the contractor shall propose the boundaries and rationale for all regions. The contractor and the procuring agency may need to negotiate the boundaries of these regions.

At this stage the Flight Phases will also be known from 4.1.2. In response to these and other requirements the contractor can then design the aircraft and:

Further define the ROSH for each Flight Phase based on the associated Air Vehicle Normal States,

Construct the larger ROTH for the Air Vehicle Normal State associated with each Flight Phase, and

Similarly, construct portions of the RORH boundaries, beyond which operation is not allowed.

Each region must include the flight condition(s) related to any pertinent performance guarantees.

The regions are described by a two-dimensional figure representing the boundaries in terms of speed and load factor at a given altitude. An example is shown on figure 4. For each Flight Phase, a series of these figures for various altitudes up to the altitude limit forms a three-dimensional representation of the three regions for that Flight Phase. The load factor, n, denotes maneuverability without regard to thrust available; i.e., the flying qualities standard places no requirements on load factor capability in constant-speed level flight.

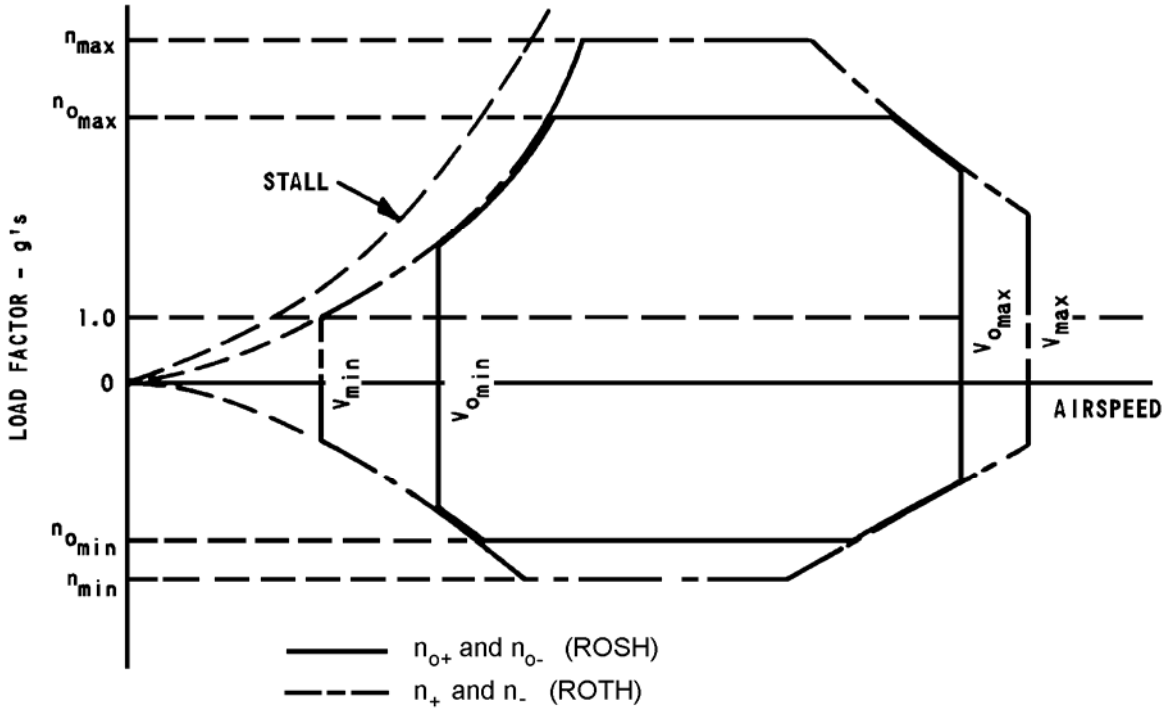


FIGURE 4. Two-dimensional Region of Handling boundaries (speed and load factor).

For a given design, angle-of-attack usually will be a more succinct bound than speed, altitude, and load factor boundaries which vary with gross weight. For relation to aircraft missions, the regions shall be kept in standard V-h-n form, but nominal weight can be used if it and the limit angle-of-attack are given. Flight testing is likely to use angle-of-attack directly.

Some Flight Phases of the same Category will involve the same, or very similar Air Vehicle Normal States; so one set of Regions may represent several Flight Phases. Each Flight Phase will involve a range of loadings. Generally, it will be convenient to represent this variation by

superimposing boundaries for the discrete loadings of 4.1.3.2, or possibly by bands denoting extremes. If different external store complements affect the Region boundaries significantly, it may be necessary for the contractor to construct several sets of Regions for each Flight Phase, each set representing a family of stores. A manageably small total number of Regions should result. It is apparent that the Regions can be and should be refined as the design is further analyzed and defined, by agreement between the contractor and the procuring activity.

Flight tests will be conducted to evaluate the aircraft against requirements in known (a priori) Regions. Generally, flight tests will cover the ROTH, with specific tests (stalls, dives, etc.) to the limits of the RORH. The same test procedures usually apply in both the ROSH and the ROTH; only the numerical requirements and qualitative Levels differ. If, for example, speed and altitude are within the ROSH but normal load factor is between the ROSH and ROTH boundaries, the requirements for the ROTH apply for Air Vehicle Normal States.

Ideally, the flight test program should also lead to definition of the ROSH and ROTH boundaries. These boundaries might aid the using commands in tactical employment, even long after the procurement contract has been closed out.

Separate Regions are not normally required for Air Vehicle Failure States. It is rational to consider most failures throughout the regions associated with Air Vehicle Normal States. There may be exceptions (such as wing sweep failure that necessitates a wing-aft landing, or a flap failure that requires a higher landing speed) that are peculiar to the specific design. In such cases the procuring activity may have to accept some smaller Regions for specific Failure States, making sure that these Regions are large enough for safe Level 2 or Level 3 operation.

Level 2 flying qualities are required in the ROTH. Note, however, that the minimum service speed is a function of stall speed, V_S, and the first item in the definition (see 4.1.4.3) is based on lift plus thrust component. For STOL or high thrust-to-weight ratio configurations, V_S by this definition can be significantly lower than aerodynamic or power-off stall speed. Other items in the definition of V_S and minimum service speed give a minimum usable speed which could be higher or lower than the aerodynamic stall speed. This applies in level flight and in maneuvers. It is doubtful that this interpretation has in fact been used; however, there are operational benefits to be gained from improving flying qualities at extreme flight conditions. The safe, usable attainment of more extreme flight conditions may be emphasized for missions in which maneuvering at high angle-of-attack is critical. The contractor could accomplish this by tailoring the requirements for determining the ROTH and RORH. As an example, the contractor could define the RORH to be consistent with operational maneuvers appropriate to the mission.

In the roll performance requirements we have felt the need to make a further distinction as a function of airspeed within the ROSHs and ROTHs. The relaxation close to the stall is a concession to aerodynamic realities for roll control via the usual means.

It should also be noted that the boundaries of these Regions should not be set by the ability to meet the flying qualities requirements. The flying qualities requirements should be met within the boundaries which are normally set by other factors, unless specific deviations are granted. The only exception is control power, which may set some boundaries for stable aircraft, if the

requirements on the ROSH are still met. The rationale for each type of Region is presented in the following discussions of each subparagraph.

High AOA shall be considered in determining the boundaries of the regions of handling (table I, table II, and table III). Stall and spin are related by their application at high AOA typically outside the ROTH. High AOA is considered to be at and above the AOA for stall warning (3.3), generally outside the ROTH. Thus the AOA value which is considered high will vary with the situation, such as: Air Vehicle Class, configuration, and Mach number.

Bear in mind that the purpose of the Regions of Handling is not to drive requirements on speed, altitude, and load factor. Those are driven by the performance requirements. The purpose of these regions is to define where the air vehicle will have Level 1 and Level 2 handling qualities, and the limits of permissible flight. Prior to contract award, the contractor may not be able to provide explicit numbers for all of the values in table I, table II, and table III. At this stage, it shall be sufficient to fill in the tables with formulas similar to those recommended in the tables. For example, prior to contract award, the stall speeds for different configurations will not be known, so the contractor would not be able to provide numbers for $V_{o_{min}}$ for these configurations. But it would be satisfactory for the contractor to put "$1.15V_S$" in the table. As the design matures and testing provides predictions for V_S, the formulas shall be replaced with actual numbers. The table shall be completely filled out with specific numbers by CDR or equivalent review. Because of the inherent uncertainties in these predictions, the values in this table should be treated with some tolerance. For the purpose of demonstrating compliance the following tolerances shall be applied: Minimum airspeeds shall be within a tolerance of ±1 kt or ±1%, whichever is less; Maximum airspeeds shall be within a tolerance of ±10 kts or ±10%, whichever is less; Maximum altitudes shall be within a tolerance of ±1000 ft or ±5%, whichever is less; Minimum load factors shall be within a tolerance of ±0.1 g or ±10%, whichever is less; and Maximum load factors shall be within a tolerance of ±0.5 g or ±10%, whichever is less.

REQUIREMENT LESSONS LEARNED

The Air Force Flight Test Center has expressed the desirability of more flight testing away from the middle of the Regions, where most of the testing is done.

4.1.4.1 Defining the Regions of Satisfactory Handling (ROSH).
The contractor shall define the boundaries of the ROSH for each applicable Air Vehicle State in terms of speed, altitude, and load factor within which the aircraft must be capable of operating in order to accomplish the missions of 4.1.1 and in which Level 1 flying qualities are required. These regions shall implicitly include the ranges of other parameters, such as sideslip, which may normally be encountered. The range of sideslip or lateral acceleration employed with direct side force control shall be stated explicitly.

REQUIREMENT RATIONALE

Definition of the ROSHs is basic to application of the flying qualities requirements. By bounding the regions in which the best flying qualities are desired, unnecessary cost, weight, complexity, etc. can be avoided while assuring capability to perform the intended missions. The boundaries of the ROSHs will be a competitive consideration, so the contractor may be held to them once they are established.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1 and 6.4.6 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.7
MIL-STD-1797: 4.1.4.1.

ROSHs are regions in speed-altitude-load factor space where it is necessary for an aircraft, in the configuration and loading associated with a given Flight Phase to have very good flying qualities—as opposed, for example, to regions where it is only necessary to insure the aircraft can be controlled without undue consideration. The required size of the ROSHs for a particular aircraft will have been defined in 4.1.1. The contractor may use the representative conditions of table I or may provide additional boundaries for the applicable Flight Phases.

Different Air Vehicle States (configuration, weight, center-of-gravity position, and external store combination) are appropriate for different flight phases. Therefore, to avoid confusion, a short-hand notation is used in table I, and throughout the rest of this document, to indicate which Flight Phase is associated with the specified speed, altitude, or load factor. For example, the notation V_{max}(TO) indicates that the speed intended is V_{max} for the Air Vehicle State associated with the Takeoff Flight Phase; the notation $n_{L_{max}}$(CO) indicates that the load factor intended is $n_{L_{max}}$ for the Air Vehicle State associated with the Air-to-Air Combat Flight Phase; etc. This notation is necessary to avoid confusion, since the configuration and Flight Phase will change throughout a mission.

It has not generally been found necessary to incorporate other flight parameters such as sideslip; the operationally encountered range of such parameters is implicit in the ROSHs. Also, certain requirements specify a sideslip range or capability.

In defining the speed-altitude-load factor combinations to be encompassed, the following factors shall be considered:

a. The ROSH for a given Flight Phase shall initially be considered to be as large a portion of the associated ROTH as possible, to permit the greatest freedom of use of the aircraft.

b. The detail specification shall be as specific as possible about the speed and altitude ranges over which stated load factor capabilities are required. Obviously, limit load factor can not be attained at a lift-limited combat ceiling; but normally it would be insufficient at a much lower altitude to have n_L capability at only one speed.

c. The ROSHs must encompass the flight conditions at which all appropriate performance guarantees will be demonstrated.

d. In setting the minimum approach speed, $V_{o_{min}}(PA)$, care shall be taken to allow sufficient stall margin. Commonly, 1.2 V_S has been used for military land-based aircraft and 1.15 V_S for carrier-based aircraft. For reference, FAR Part 25 specifies 1.3 V_S for landing-distance calculations; while FAR Part 23 specifies approach at 1.5 V_S for these calculations where required. Differences in method of determining stall speed tend to make the FAR Part 25 and military stall margins equivalent; herein V_S is defined at constant airspeed rather than with a slow deceleration.

e. If design tradeoffs indicate that significant penalties (in terms of performance, cost, system complexity, or reliability) are required to provide Level 1 flying qualities throughout the large region formed by items a-d above, consideration shall be given to restricting the ROSH toward the minimum consistent with the requirements of the Flight Phase of the operational mission under consideration.

When effective limiters are employed, Level 1 handling qualities shall be provided as close to the limits as practical. Without an effective limiter, graceful degradation is much preferred in order to preclude a flying qualities "cliff."

REQUIREMENT LESSONS LEARNED

Operational missions generally depart significantly from the design mission profile, even for the same type of mission. It is important to allow enough latitude to cover likely variations. Also, over the life of an aircraft its operational missions will likely change in both type and detail. There are, of course, tradeoffs with cost, weight, and the like. For a particular procurement the extent of the ROSH beyond minimum operational needs shall be as large, then, as these tradeoffs will reasonably permit. While stability and control augmentation can do wonders, such factors as basic control authority and rate, aeroelasticity, and stall speed are limiting at operational extremes and difficult and costly to change after the design freeze. Skimping on the ROSHs, then, can cause difficulties.

There is no connotation that operation is limited to the ROSHs. ROSH is a name, the best we could find, for the region in which the best flying qualities are required. Some pilots have objected that air combat routinely involves flight at lower speeds and higher angles-of-attack, even post-stall. It has never been intended to preclude such operational use where it is safe. For a particular procurement, requirements outside the ROSH may warrant strengthening where needs can be identified.

TABLE IV. Regions of Satisfactory Handling (ROSH).

Air Vehicle Normal State	Flight Phase	Min Airspeed $V_{o_{min}}$	Max Airspeed $V_{o_{max}}$	Min Altitude $h_{o_{min}}$	Max Altitude $h_{o_{max}}$	Min Load Factor $n_{o_{min}}$	Max Load Factor $n_{o_{max}}$
	Air-to-air combat	1.15 V_S(CO)	V_{MAT}(CO)	MSL	Combat ceiling(CO)	-1.0	$n_{L_{max}}$(CO)
	Ground attack	1.3 V_S(GA)	V_{MRT}(GA)	MSL	*	-1.0	$n_{L_{max}}$(GA)
	Weapon Delivery/ Launch	V_{range}(WD)	V_{MAT}(WD)	MSL	Combat ceiling(WD)	0.5	*
	Aerial Recovery	1.2 V_S(AR)	V_{MRT}(AR)	MSL	Combat ceiling(AR)	0.5	$n_{L_{max}}$(AR)
	Reconnaissance	1.3 V_S(RC)	V_{MAT}(RC)	MSL	Combat ceiling(RC)	*	*
	In-flight Refueling (Receiver)	1.2 V_S(RR)	V_{MRT}(RR)	MSL	Combat ceiling(RR)	0.5	2.0
	Terrain-following	V_{range}(TF)	V_{MAT}(TF)	MSL	10,000 ft	0.0	3.5
	Anti-submarine Search	1.2 V_S(AS)	V_{MRT}(AS)	MSL	*	0.0	2.0
	Close-formation flying	1.4 V_S(FF)	V_{MAT}(FF)	MSL	Combat ceiling(FF)	-1.0	$n_{L_{max}}$(FF)
	Climb	.85 $V_{R/C}$(C)	1.3 $V_{R/C}$(C)	MSL	Cruise ceiling(C)	0.5	2.0
	Cruise	V_{range}(CR)	V_{NRT}(CR)	MSL	Cruise ceiling(CR)	0.5	2.0
	Loiter	.85 V_{end}(LO)	1.3 V_{end}(LO)	MSL	Cruise ceiling(LO)	0.5	2.0
	In-flight Refueling (Tanker)	1.4 V_S(RT)	V_{MAT}(RT)	MSL	Cruise ceiling(RT)	0.5	2.0
	Descent	1.4 V_S(D)	V_{MAT}(D)	MSL	Cruise ceiling(D)	0.5	2.0
	Emergency Descent	1.4 V_S(ED)	V_{max}(ED)	MSL	Cruise ceiling(D)	0.5	2.0
	Emergency Deceleration	1.4 V_S(DE)	V_{max}(DE)	MSL	Cruise ceiling(DE)	0.5	2.0
	Aerial Delivery	1.2 V_S(AD)	200 kts	MSL	10,000 ft	0.0	2.0
	Takeoff	$V_{o_{min}}$(TO)	V_{max}(TO)	MSL	10,000 ft	0.5	2.0
	Catapult Takeoff	$V_{C_{min}}$(CT)	$V_{o_{min}}$(CT) + 30 kts	MSL	MSL	0.5	$n_{L_{max}}$(CT)
	Approach	$V_{o_{min}}$(PA)	V_{max}(PA)	MSL	10,000 ft	0.5	2.0
	Wave-off/Go-around	$V_{o_{min}}$(PA)	V_{max}(WO)	MSL	10,000 ft	0.5	2.0
	Landing	$V_{o_{min}}$(L)	V_{max}(L)	MSL	10,000 ft	0.5	2.0

* Appropriate to the operational mission

4.1.4.2 Defining the Regions of Tolerable Handling (ROTH).
For each Air Vehicle Normal State, the contractor shall establish ROTHs showing combinations of speed, altitude, and normal acceleration derived from aircraft performance margins (such as service ceiling) instead of from mission requirements. These regions shall implicitly include the ranges of other parameters, such as sideslip, which can be expected within the speed, altitude, and load factor bounds. For each applicable Flight Phase and Normal State, the boundaries of the ROTHs shall be coincident with or lie outside the corresponding ROSH boundaries.

REQUIREMENT RATIONALE

Definition of the ROTHs is basic to application of the flying qualities requirements. The size of the ROTHs may be a secondary competitive consideration.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1 and 6.4.6 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.8 through 3.1.8.4
MIL-STD-1797: 4.1.4.2.

The ROTH encompasses the ROSH for the same Flight Phase and Air Vehicle Normal State. Its larger volume denotes the extent of flight conditions that provide a margin of safety to avoid exceeding aircraft limitations (safe margins shall be determined by simulation and flight test). At least Level 2 handling qualities are required for normal operation. This allows the pilot to accomplish the mission Flight Phase associated with the Air Vehicle Normal State although mission effectiveness or pilot workload, or both, may suffer somewhat.

This region is also intended to help insure that any deterioration of handling qualities will be gradual as flight progresses beyond the limits of the ROSH. This serves two purposes. It provides some degree of mission effectiveness for possible unforeseen alternate uses of the aircraft, and it also allows for possible inadvertent flight outside the ROSH.

As with the ROSHs, a range of other flight parameters such as sideslip is implicit. For the ROTH this range shall be derived from aircraft performance margins as opposed to mission requirements.

The boundaries of the ROTH shall be based on the speed, altitude, and load factor considerations discussed below and sketched on figure 4.

1. Maximum Speed Boundary for the ROTH (maximum service speed)

The maximum service speed, V_{max} or M_{max}, for each altitude shall be the lowest of:

a. The maximum speed at which a safe margin exists from any potentially dangerous flight condition.

b. A speed which is a safe margin below the speed at which intolerable buffet or structural vibration is encountered.

In setting the maximum service speed, the contractor need not consider speed-altitude combinations that can only be reached in an attitude that would not permit recovery to level flight with a nominal 2000 ft clearance above sea level while remaining within the RORH.

2. Minimum Speed Boundary for the ROTH (minimum service speed)

The minimum service speed, V_{min} or M_{min}, for each altitude shall be the highest of:

a. 1.1 V_S

b. V_S + 10 kts equivalent airspeed

c. The speed below which full aircraft-nose-up pitch control power and trim are insufficient to maintain straight, steady flight.

d. The lowest speed at which level flight can be maintained with MRT.

e. A speed limited by reduced visibility or an extreme pitch attitude that would result in the tail or aft fuselage contacting the ground.

For engine failure during takeoff, the standard requires control at speeds down to V_{min}(TO); but requirements for engine-out climb capability are left to performance requirements.

3. Maximum Altitude Boundary for the ROTH (service ceiling)

The service ceiling, h_{max}, for a given speed shall be the maximum altitude at which a rate of climb of 100 ft per minute can be maintained in unaccelerated flight with maximum augmented thrust (MAT).

4. Load Factor Boundaries for the ROTH

The positive and negative load factor boundaries, n_+ and n_-, shall be established as a function of speed for several significant altitudes. The positive [negative] load factor boundaries, when trimmed for 1-g flight at a particular speed and altitude, shall be the lowest [highest] algebraically of:

a. The positive [negative] structural limit load factor, $n_{L_{max}}$ [$n_{L_{min}}$].

b. The steady load factor corresponding to the minimum allowable angle-of-attack for stall warning (5.2.5.2).

c. The steady load factor at which the pitch control is in full aircraft-nose-up [nose-down] position.

d. A safe margin below [above] the load factor at which intolerable buffet or structural vibration is encountered.

The maximum service load factor, n_{max}, is the highest value of n_+ for a given altitude. The minimum service load factor, n_{min}, is the lowest value of n_- for a given altitude.

The contractor may use the representative conditions of TABLE II or may provide additional boundaries for the applicable Flight Phases.

REQUIREMENT LESSONS LEARNED

Rarely has a military aircraft been used for only its design missions. Examples are plentiful: the P-47, designed as a high-altitude fighter but used more extensively in a ground support role; the B-47 and B-52, designed for high-altitude penetration and bomb-drop from level flight, but later assigned low-level penetration and toss delivery as well; the F-4, early assigned to a Navy interceptor role but then used as an all-purpose fighter, and by the Air Force as well; the F-15 and F-16, designed as air-superiority fighters but now also flown heavily loaded with external stores for dive bombing; the C-5 with largely unused capability for low-level penetration and forward-base operation.

In general, experience strongly indicates the significant benefits to be had from providing at least tolerable, if not satisfactory, flying qualities up to safe margins from stall, limit dive speed, etc. The capability to use all the performance fallout outside the design mission envelope can greatly enhance the worth of any military aircraft. A significant additional benefit is the promotion of graceful degradation rather than flying qualities “cliffs”, for which everything is fine right up to the boundary for departure from controlled flight.

TABLE V. Regions of Tolerable Handling (ROTH).

Air Vehicle Normal State	Flight Phase	Min Airspeed V_{min}	Max Airspeed V_{max}	Min Altitude	Max Altitude h_{max}	Min Load Factor n_{min}	Max Load Factor n_{max}
	Air-to-air combat	V_{min}(CO)	V_{max}(CO)	MSL	Service ceiling(CO)	$n_{L_{min}}$(CO)	$n_{L_{max}}$(CO)
	Ground attack	V_{min}(GA)	V_{max}(GA)	MSL	Service ceiling(GA)	$n_{L_{min}}$(GA)	$n_{L_{max}}$(GA)
	Weapon Delivery/ Launch	V_{min}(WD)	V_{max}(WD)	MSL	Service ceiling(WD)	n_{min}(WD)	n_{max}(WD)
	Aerial Recovery	V_{min}(AR)	V_{max}(AR)	MSL	Service ceiling(AR)	n_{min}(AR)	$n_{L_{max}}$(AR)
	Reconnaissance	V_{min}(RC)	V_{max}(RC)	MSL	Service ceiling(RC)	n_{min}(RC)	n_{max}(RC)
	In-flight Refueling (Receiver)	V_{min}(RR)	V_{max}(RR)	MSL	Service ceiling(RR)	n_{min}(RR)	n_{max}(RR)
	Terrain-following	V_{min}(TF)	V_{max}(TF)	MSL	Service ceiling(TF)	n_{min}(TF)	n_{max}(TF)
	Anti-submarine Search	V_{min}(AS)	V_{max}(AS)	MSL	Service ceiling(AS)	n_{min}(AS)	n_{max}(AS)
	Close-formation flying	V_{min}(FF)	V_{max}(FF)	MSL	Service ceiling(FF)	n_{min}(FF)	$n_{L_{max}}$(FF)
	Climb	V_{min}(C)	V_{max}(C)	MSL	Service ceiling(C)	n_{min}(C)	n_{max}(C)
	Cruise	V_{min}(CR)	V_{max}(CR)	MSL	Service ceiling(CR)	n_{min}(CR)	n_{max}(CR)
	Loiter	V_{min}(LO)	V_{max}(LO)	MSL	Service ceiling(LO)	n_{min}(LO)	n_{max}(LO)
	In-flight Refueling (Tanker)	V_{min}(RT)	V_{max}(RT)	MSL	Service ceiling(RT)	n_{min}(RT)	n_{max}(RT)
	Descent	V_{min}(D)	V_{max}(D)	MSL	Service ceiling(D)	n_{min}(D)	n_{max}(D)
	Emergency Descent	V_{min}(ED)	V_{max}(ED)	MSL	Service ceiling(ED)	n_{min}(ED)	n_{max}(ED)
	Emergency Deceleration	V_{min}(DE)	V_{max}(DE)	MSL	Service ceiling(DE)	n_{min}(DE)	n_{max}(DE)
	Aerial Delivery	V_{min}(AD)	V_{max}(AD)	MSL	Service ceiling(AD)	n_{min}(AD)	n_{max}(AD)
	Takeoff	V_{min}(TO)	V_{max}(TO)	MSL	Service ceiling(TO)	n_{min}(TO)	n_{max}(TO)
	Catapult Takeoff	V_{min}(CT)	V_{max}(CT)	MSL	Service ceiling(CT)	n_{min}(CT)	$n_{L_{max}}$(CT)
	Approach	V_{min}(PA)	V_{max}(PA)	MSL	Service ceiling(PA)	n_{min}(PA)	n_{max}(PA)
	Wave-off/Go-around	V_{min}(WO)	V_{max}(WO)	MSL	Service ceiling(WO)	n_{min}(WO)	n_{max}(WO)
	Landing	V_{min}(L)	V_{max}(L)	MSL	Service ceiling(L)	n_{min}(L)	n_{max}(L)

4.1.4.3 Defining the Regions of Recoverable Handling (RORH).
The contractor shall define the RORHs which encompass all regions in which operation of the air vehicle is both allowable and possible, and which the air vehicle is capable of safely encountering. The RORH boundaries shall be defined in terms of speed, altitude, load factor, and any other flight limits.

REQUIREMENT RATIONALE

Definition of the RORHs is basic to application of the flying qualities requirements. This is a flight safety consideration.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1 and 6.4.6 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.9
MIL-STD-1797: 4.1.4.3.

Basically the RORH shall be defined such that, from all points within it, ready and safe return to the ROTH is possible without exceptional pilot skill or technique, regardless of component or system failures. Structural limits on other flight parameters such as sideslip or lateral acceleration must also be observed and readily complied with. The requirements on stall, spin, and dive characteristics or dive recovery devices, and on approach to dangerous flight conditions also apply, although Level 2 flying qualities (for Air Vehicle Normal States) are not required there.

The maximum permissible speed in dives or level flight (the F-103 could reach V_{max} in a climb), and the minimum permissible speed in level flight, can and must be defined for pilots' information. Additionally, some minimum airspeed may need to be defined for zooms, to assure recoverability. For maneuvers such as spins, no minimum permissible speed is normally stated; one accepts the low airspeed attained in the maneuver, satisfactory recovery being the only criterion.

To specify these considerations the contractor shall, as a minimum, define the boundaries given below.

1. Maximum Speed Boundary for the RORH

The maximum permissible speed for each altitude shall be the lowest of:

a. Limit speed based on structural considerations.

b. Limit speed based on engine considerations.

c. The speed at which intolerable buffet or structural vibration is encountered.

In setting the maximum permissible speed, the contractor need not consider speed-altitude combinations that can only be reached in an attitude that would not permit recovery to level flight with a nominal 2000 ft clearance above sea level while remaining within the RORH. To allow for inadvertent excursions beyond placard speed, some margin shall be provided between the maximum permissible speed and the high-speed boundaries of the ROSH and ROTH. Such a margin is not specified because no satisfactory general requirement could be formulated. However, for specific designs, the contractor shall consider 1.1 V_H (commonly used for structural requirements) or the upset requirements of FAR Part 25 and Advisory Circular AC 25.253-1A.

2. 1-g Minimum Speed Boundary for the RORH

Where maximum lift determines minimum speed, the minimum permissible speed in 1-g flight shall be stall speed, V_S.

Stall speed (equivalent airspeed), at 1g normal to the flight path, is defined as the highest of:

a. Speed for steady straight flight at $C_{Lmax,}$ the first local maximum of the curve of lift coefficient ($L/\bar{q}S$) versus angle of attack which occurs as C_L is increased from zero.

b. Speed at which uncommanded pitching, rolling, or yawing occurs (5.2.5.3).

c. Speed at which intolerable buffet or structural vibration is encountered.

In general, this is fundamentally more an angle-of-attack limit than an airspeed limit. For some aircraft, considerations other than maximum lift determine the minimum permissible speed in 1-g flight [e.g., ability to perform altitude corrections, excessive sinking speed, ability to execute a waveoff (go-around), etc.]. In such cases an arbitrary angle-of-attack limit, or similar minimum speed and maximum load factor limits, shall be established for the RORH. This defined minimum permissible speed shall be used as V_S in all applicable requirements. V_S needs to be consistent with the stall speed used for performance, structural, and other requirements.

The contractor may use the representative conditions of table III or may provide additional boundaries for the applicable Flight Phases. The RORHs will be refined as better analysis, simulation, and test data become available.

REQUIREMENT LESSONS LEARNED

For both combat and training missions, flight outside the ROTH may well be routine. The flying qualities requirements that apply there are largely qualitative. Nevertheless these areas can be important parts of a useful flight envelope, or may be reached inadvertently, and so need careful consideration.

TABLE VI. Regions of Recoverable Handling (RORH).

Air Vehicle Normal State	Flight Phase	Min Airspeed	Max Airspeed	Min Altitude	Max Altitude h_L	Min Load Factor $n_{L_{min}}$	Max Load Factor $n_{L_{max}}$
	Air-to-air combat	V_S(CO)	V_L(CO)	MSL	h_L(CO)	$n_{L_{min}}$(CO)	$n_{L_{max}}$(CO)
	Ground attack	V_S(GA)	V_L(GA)	MSL	h_L(GA)	$n_{L_{min}}$(GA)	$n_{L_{max}}$(GA)
	Weapon Delivery/ Launch	V_S(WD)	V_L(WD)	MSL	h_L(WD)	$n_{L_{min}}$(WD)	$n_{L_{max}}$(WD)
	Aerial Recovery	V_S(AR)	V_L(AR)	MSL	h_L(AR)	$n_{L_{min}}$(AR)	$n_{L_{max}}$(AR)
	Reconnaissance	V_S(RC)	V_L(RC)	MSL	h_L(RC)	$n_{L_{min}}$(RC)	$n_{L_{max}}$(RC)
	In-flight Refueling (Receiver)	V_S(RR)	V_L(RR)	MSL	h_L(RR)	$n_{L_{min}}$(RR)	$n_{L_{max}}$(RR)
	Terrain-following	V_S(TF)	V_L(TF)	MSL	h_L(TF)	$n_{L_{min}}$(TF)	$n_{L_{max}}$(TF)
	Anti-submarine Search	V_S(AS)	V_L(AS)	MSL	h_L(AS)	$n_{L_{min}}$(AS)	$n_{L_{max}}$(AS)
	Close-formation flying	V_S(FF)	V_L(FF)	MSL	h_L(FF)	$n_{L_{min}}$(FF)	$n_{L_{max}}$(FF)
	Climb	V_S(C)	V_L(C)	MSL	h_L(C)	$n_{L_{min}}$(C)	$n_{L_{max}}$(C)
	Cruise	V_S(CR)	V_L(CR)	MSL	h_L(CR)	$n_{L_{min}}$(CR)	$n_{L_{max}}$(CR)
	Loiter	V_S(LO)	V_L(LO)	MSL	h_L(LO)	$n_{L_{min}}$(LO)	$n_{L_{max}}$(LO)
	In-flight Refueling (Tanker)	V_S(RT)	V_L(RT)	MSL	h_L(RT)	$n_{L_{min}}$(RT)	$n_{L_{max}}$(RT)
	Descent	V_S(D)	V_L(D)	MSL	h_L(D)	$n_{L_{min}}$(D)	$n_{L_{max}}$(D)
	Emergency Descent	V_S(ED)	V_L(ED)	MSL	h_L(ED)	$n_{L_{min}}$(ED)	$n_{L_{max}}$(ED)
	Emergency Deceleration	V_S(DE)	V_L(DE)	MSL	h_L(DE)	$n_{L_{min}}$(DE)	$n_{L_{max}}$(DE)
	Aerial Delivery	V_S(AD)	V_L(AD)	MSL	h_L(AD)	$n_{L_{min}}$(AD)	$n_{L_{max}}$(AD)
	Takeoff	V_S(TO)	V_L(TO)	MSL	h_L(TO)	$n_{L_{min}}$(TO)	$n_{L_{max}}$(TO)
	Catapult Takeoff	V_S(CT)	V_L(CT)	MSL	h_L(CT)	$n_{L_{min}}$(CT)	$n_{L_{max}}$(CT)
	Approach	V_S(PA)	V_L(PA)	MSL	h_L(PA)	$n_{L_{min}}$(PA)	$n_{L_{max}}$(PA)
	Wave-off/Go-around	V_S(WO)	V_L(WO)	MSL	h_L(WO)	$n_{L_{min}}$(WO)	$n_{L_{max}}$(WO)
	Landing	V_S(L)	V_L(L)	MSL	h_L(L)	$n_{L_{min}}$(L)	$n_{L_{max}}$(L)

5. DETAILED REQUIREMENTS

5.1 Primary requirements.

The flying qualities of the air vehicle shall meet the requirements specified herein. These requirements shall be met for Air Vehicle States (3.2) encountered in Flight Phases and tasks of the operational missions.

REQUIREMENT RATIONALE

These paragraphs contain the requirements for the flying and ground handling qualities of air vehicles. These requirements are intended to assure flying qualities that support adequate mission performance and flight safety regardless of the design implementation or flight control system augmentation. The requirements are to be applied during the design, construction, testing, and acceptance of the subject air vehicle.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1 in JSSG-2001.

The corresponding paragraphs in previous specifications are:
MIL-F-8785C: 1.1 and 1.2
MIL-STD-1797: 1.1 and 1.2.

The contractor will need to have accomplished certain steps in the development process to be able to apply the requirements of this standard. These steps were described in detail in 4.1.

Requirements 5 through 5.1.3 are the primary flying qualities requirements from JSSG-2001 (3.3.11.1 through 3.3.11.1.3). Detailed requirements and guidance to enable compliance with these top level requirements can be found in subsequent paragraphs of this standard. These subsequent requirements bridge the gap between a specified qualitative Level of flying qualities and the contractors' need to have a quantifiable, measurable set of parameters that will shape and size the resulting air vehicle. These requirements are intended to assure flying qualities for adequate mission performance and flight safety regardless of the design implementation or flight control system augmentation. The contractor and the procuring activity shall utilize appropriate requirements from this standard consistent with the specific acquisition. This standard provides performance requirements and guidance that flow down to other design areas such as aerial refueling, armament (weapons), crew systems, flight control, structures, subsystems, etc.

The requirements of this standard shall be met:

(a) For all configurations and Air Vehicle States required or encountered in all associated Flight Phases and tasks of the operational missions. Operational missions include the entire spectrum of intended usage, including aircrew upgrade and training.

(b) For all combinations of internal and external stores (both symmetric and asymmetric) required by the operational missions. When the stores contain expendable loads, the

requirements shall be met throughout the range of store loadings, including sloshing and shifting.

(c) Throughout the longitudinal, lateral, and vertical envelopes of center of gravity and the corresponding weights that exist for each Flight Phase. These envelopes include the most forward and aft center-of-gravity positions, as well as the maximum center of gravity excursions attainable through failures in systems or components, such as fuel sequencing, hung stores, etc., for each Flight Phase.

(d) For all moments and products of inertia of the air vehicle associated with all of the loadings above.

For those requirements which employ subjective terms (such as objectionable flight characteristics, realistic time delay, normal pilot technique, or excessive loss of altitude or build-up of speed), the Flying Qualities Working Group will make the final determination of compliance.

REQUIREMENT LESSONS LEARNED

A typical approach (but not the only one) for the system contractor is outlined below. The stages indicated are appropriate for the required calculations.

1. Initial Design

The basic airframe is designed for a Level 1 target in respect to most flying qualities in the ROSH. It may quickly become apparent that some design penalties would be inordinate (perhaps to provide sufficient aerodynamic damping of the short period and Dutch roll modes at high altitudes). In these cases the basic airframe target would be shifted to Level 2. In other cases it may be relatively painless to extend some Level 1 flying qualities over the wider range of the ROTH. Generally the design will result in Level 1 flying qualities in some regions, and perhaps, Level 2 or Level 3 in others. Augmentation of one form or another (aerodynamic configuration changes, response feedback, control feedforward, signal shaping, etc.) would be incorporated to bring flying qualities up to Level 1 in the ROSH and to Level 2 in the ROTH.

2. Initial Evaluation

The reliability and failure mode analyses are next performed to evaluate the nominal system design evolved above. All air vehicle subsystem failures that affect flying qualities are considered. Failure rate data for these analyses may be those specified in the related specifications, other data with supporting substantiation, or specific values provided by the procuring activity. Prediction methods used shall be in accordance with related specifications. The results of this evaluation will provide: a) a detailed outline of design points that are critical from a flying qualities/flight safety standpoint; b) quantitative predictions of the probability of encountering Level 2 in a single flight within the ROSH, Level 3 in the ROSH, and Level 2 in the ROTH; and c) recommended airframe/equipment changes to improve flying qualities or increase subsystem reliability to meet the specification requirements. Note that the flying qualities/flight safety requirements are concerned with failure mode effects, while the cancelled

MIL-STD-785 provided "basic" reliability requirements per se (all failures regardless of failure effects). In the event of a conflict, the most stringent requirement would apply.

3. Re-Evaluation

As the system design progresses, the initial evaluation is revised at intervals. This process continues throughout the design phase, with reviews by the procuring activity at times consonant with other reviewer activity.

The results of the analyses of air vehicle flying qualities/flight safety may be used directly to:
a) establish flight test points that are critical and shall be emphasized in the flight test program;
b) establish pilot training requirements for the most probable, and critical, flight conditions; and
c) provide guidance and requirements for other subsystem designs.

VERIFICATION

Selected configurations and loadings shall be analyzed and tested for each applicable Flight Phase. The selected configurations shall include:

a. All configurations required for mission accomplishment
b. All configurations required for performance demonstration
c. All configurations likely to be encountered
d. Any additional configurations the procuring agency feels necessary

Selected air vehicle loadings throughout the range of all possible loadings required or encountered shall be analyzed and tested for each applicable Flight Phase. The selected loadings for each Flight Phase shall include:

a. The most critical loading (in a flying qualities sense) for each individual requirement
b. The maximum and minimum permissible weights
c. The maximum and minimum weights attainable through failures in systems or components, such as fuel sequencing or hung stores
d. The maximum and minimum permissible c.g. positions (longitudinal, lateral, and vertical)
e. The maximum c.g. excursions (longitudinal, lateral, and vertical) attainable through failures in systems or components
f. The maximum and minimum permissible moments and products of inertia
g. The maximum and minimum moments and products of inertia attainable through failures in systems or components
h. The most critical stores combinations (internal and external, symmetric and asymmetric), including sloshing and shifting in stores containing expendable loads
i. The most critical stores combinations (internal and external, symmetric and asymmetric) attainable through failures in systems or components, including sloshing and shifting in stores containing expendable loads
j. The worst possible cases (in a flying qualities sense) that are not Special Failure States
k. Any additional loadings the procuring agency feels necessary

From a practical standpoint, only a few of the large number of possible store combinations can be flight tested. A careful analysis before flight testing will assure that the most critical combinations (from a flying qualities perspective) are being evaluated.

Flight testing goals are to identify the aircraft, evaluate its operational merits, and determine specification compliance. With limited flight time and test resources available, it is a challenge to get the data needed. Only a very limited amount of compliance will be demonstrated directly. For the rest, flight validation/correction of analytical models of the aircraft will make further checking possible.

5.1.1 Primary requirements for Air Vehicle States in Common environmental conditions. The Levels of flying qualities for Air Vehicle States in Common environmental conditions shall meet the requirements of 5.1.1.1 through 5.1.1.3.3.

REQUIREMENT RATIONALE

Air vehicle flying qualities vary as a function of task, air vehicle loading, air vehicle configuration, flight condition, and environmental condition. It is impractical to expect perfect flying qualities for all tasks, all loadings, all configurations, and all environmental conditions throughout the entire Flight Envelope. The purpose of specifying different Levels of flying qualities is to allow variation with these factors while still controlling the degree of variation. The requirements of 5.1.1 establish the expected Level of flying qualities for various combinations of Flight Phase Category (task), Air Vehicle State (loading and configuration), and Regions of Handling (flight condition) in Common environmental conditions. The requirements of 5.1.2 subsequently establish the degradation allowed for each of these combinations with more severe environmental conditions.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1.1 in JSSG-2001.

The corresponding paragraphs in previous specifications are:
MIL-F-8785C: 1.5
MIL-STD-1797: 3.3.

REQUIREMENT LESSONS LEARNED

VERIFICATION

Flying qualities Levels for Air Vehicle States in Common atmospheric conditions shall be verified by meeting the requirements of 5.1.1.1 through 5.1.1.3.3.

The handling characteristics described in this standard are specified in terms of qualitative degrees of suitability and Levels of flying qualities. These Levels and degrees of suitability are part of the structure of this standard, and are based on the Cooper-Harper Scale (see figure 5). The degrees of suitability are defined as:

Satisfactory	Flying qualities clearly adequate for the mission Flight Phase. Desired performance is achievable with no more than minimal pilot compensation.
Tolerable	Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists.
Controllable	Flying qualities such that the air vehicle can be controlled in the context of the mission Flight Phase, even though pilot workload is excessive or mission effectiveness is inadequate, or both. The pilot can transition from Category A Flight Phase tasks to Category B or C Flight Phases, and Category B and C Flight Phase tasks can be completed.
Recoverable	Flying qualities such that control can be regained following loss of control due to departure, failures, or atmospheric disturbances.

In Calm air, Level 1 is Satisfactory, Level 2 is Tolerable, and Level 3 is Controllable. In the presence of higher intensities of atmospheric disturbances, the relationship between Levels and qualitative degrees of suitability are given by 5.1.2.

The Levels and qualitative degrees of suitability are used to tailor the requirements to abnormal conditions such as flight outside the ROSH and Air Vehicle Failure States, as well as normal conditions. Abnormalities may also occur in the form of large atmospheric disturbances, or some combination of conditions. Where possible, the flying qualities requirements are stated for each Level in terms of limiting values of one or more parameters. Each value, or combination of values, represents a minimum condition necessary to meet one of the three Levels or degrees of suitability.

The Primary Requirements define the Level of flying qualities or qualitative degree of suitability required for each combination of Air Vehicle State, flight condition, and level of atmospheric disturbances. The Detail Requirements define the parameter values which shall be met to achieve each Level of flying qualities or degree of qualitative suitability.

a. To achieve Level 1, the relevant parameter value shall be better than or equal to the Level 1 boundary, or number, specified in the requirement

b. To achieve Level 2, the relevant parameter value shall be no worse than the Level 2 boundary, or number, specified in the requirement

c. To achieve Level 3, the relevant parameter value must be no worse than the Level 3 boundary, or number, specified in the requirement.

In cases where a given boundary, or number, is identified as Level 1 and Level 2, the relevant flying qualities parameter must be within this common boundary, or number, in both the ROSH and the ROTH for Air Vehicle Normal States (5.1.1.1). A parameter value outside the boundary conditions shown, or worse than the number given, will result in Level 3 flying qualities at best.

It should be noted that Level 3 is not necessarily defined as safe. This is consistent with the Cooper-Harper rating scale: for Cooper-Harper ratings of 8 and 9, controllability may be in question. If safe characteristics are required for Level 3, then action must be taken to improve aircraft flying qualities.

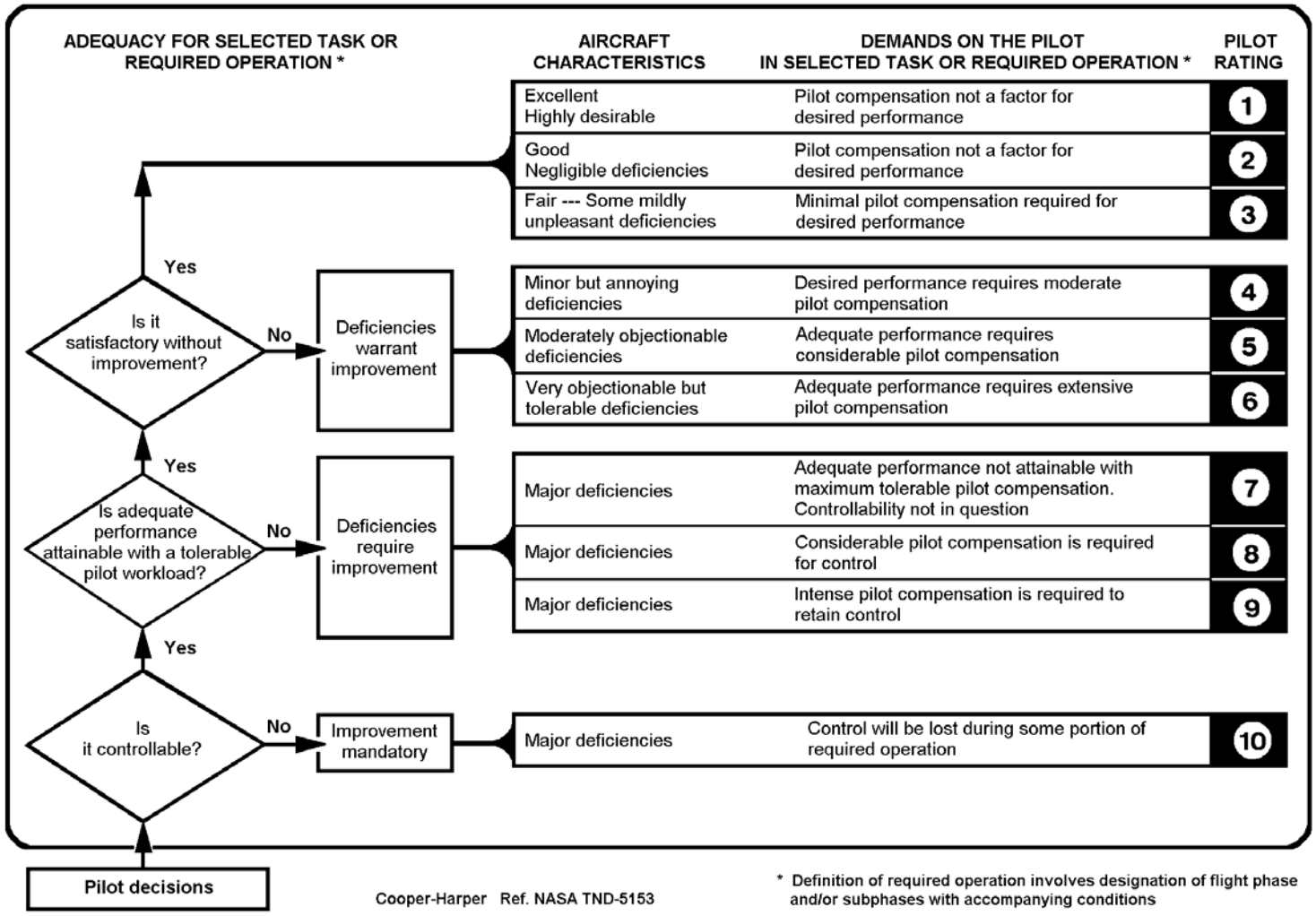


FIGURE 5. Cooper-Harper pilot rating scale.

In some cases sufficient data do not exist to allow specification of numerical values of a flying quality parameter. In such cases it is not possible to explicitly define a quantitative boundary for each Level, so the required Levels are then to be interpreted in terms of qualitative degrees of suitability for the piloting tasks appropriate for mission accomplishment.

Where possible, the requirements of this standard have been stated as allowable ranges of the stability or control parameter being specified. Each specified value is a minimum condition to meet one of three Levels of acceptability related to the ability to complete the operational missions for which the aircraft is designed. In actual practice, the flying quality boundaries referred to above were obtained by fairing lines of constant Cooper-Harper pilot rating. Hence it was necessary to define equivalent definitions between the Cooper-Harper scale shown on figure 5 and the Level definitions. Typically, a Cooper-Harper pilot rating of 1 to 3 defines Level 1, a Cooper-Harper rating from 4 through 6 defines Level 2, and a Cooper-Harper rating from 7 through 9 defines Level 3. In a few instances, as indicated in the associated Guidance, the boundaries have been further modified for other considerations. It should be noted that the Level 3 Controllable lower bound, like the other bounds, is sensitive to pilot workload. AFFDL-TR-69-72 shows the assumed correlation of the Cooper-Harper scale with earlier rating scales.

Fractional ratings are to be avoided: Cooper and Harper stress the need for clear choices, especially at the Level boundaries. Wildly scattered ratings shall not be averaged: the Cooper-Harper scale is nonlinear. Even for adjacent ratings, caution is needed: variations in pilot technique, disturbance time histories, subjective criteria, etc. may result in valid rating differences.

Relating the Cooper-Harper Pilot Rating Scale to the Levels of flying qualities has the added benefit of more precise definitions which are related to the operational considerations of pilot workload and task performance, as well as making the pilot rating correlations consistent with the Level 1, 2, and 3 criterion boundaries in the flying quality standard. It is especially important to note that “Controllable” is in the context of the Flight Phase: the pilot may have to attend to other duties in addition to flying the air vehicle.

Cooper-Harper ratings are not directly specified in the requirements of this standard for several reasons:

> Level applies to aircraft (which the requirements cover), doing design-mission tasks. Cooper-Harper (C-H) rating is given by the pilot doing the task with the aircraft in a given environment. Since C-H rating is expected to change with severity of the environment, Levels tied exclusively to C-H ratings would vary with intensity of disturbances—e.g., Level 1 in Common turbulence, Level 2 in Uncommon. That gets cumbersome to call-out in requirements, so it is necessary to ascertain the environment severity when determining Levels.
>
> Requirements needed to address ability to complete or terminate a Flight Phase, which the C-H ratings do not treat

Leeway for engineering-type input to requirements is necessary; e.g., increasing the Level 1 short-period damping boundaries to account for more severe turbulence and not allowing negative Dutch roll damping even for Level 3.

Some requirements are based on operational experience or need, rather than pilot evaluation; e.g., crosswind landing capability or two-engine-out controllability.

"Deficiencies warrant improvement" from the C-H ratings simply does not apply for normal operation between the ROSH and ROTH boundaries.

When compliance is assessed, as well as in design, the firmness of numerical values is preferred to the variability of evaluation-pilot ratings. Thus, even though the final decision to accept or reject will be made on the basis of pilot ratings (as these decisions have always been), more exactly-defined boundaries are needed for design guidance.

Having stated these caveats, it is noteworthy that given a well-defined task and "Calm air to Common turbulence" environment the Level definitions do closely correspond to 1 through 3, 4 through 6, and 7 through 9 C-H ratings. Thus, for the qualitative requirements in the proper environment there is a direct correspondence of C-H ratings to Levels. However it is done, lacking the "design" environment, the evaluation pilot must extrapolate. The Level definitions have been modified to be even closer to the C-H definitions.

Use of the C-H scale is accepted universally as a guide, but only by some as a way to state requirements. It is used herein as a principal way to relate flying qualities requirements to operational needs.

According to figure 5, piloting shall not require the pilot's attention to the exclusion of all other duties. If adequate performance requires "moderate [or greater] pilot compensation", the "deficiencies warrant improvement", etc.—all in the full context of the mission including hostile environment, weather, etc. The ratings are meant to apply to the most demanding tasks foreseen, aggressively performed.

5.1.1.1 Allowable Levels for Air Vehicle Normal States.
The boundaries of the Region of Satisfactory Handling (ROSH), Region of Tolerable Handling (ROTH), and Region of Recoverable Handling (RORH) for each Air Vehicle Normal State shall be as shown in table I, table II, and table III.

a) Flying qualities for Air Vehicle Normal States within the ROSH shall be Level 1 in Calm air and in Common environmental conditions.

b) Flying qualities for Air Vehicle Normal States within the ROTH but outside the ROSH shall be Level 2 or better in Calm air and in Common environmental conditions.

c) From all points in the RORH and outside the ROTH, it shall be possible readily and safely to return to the ROTH without exceptional pilot skill or technique in Calm air and in Common environmental conditions.

d) For requirements pertaining to ground operations and terminal flight phases such as taxiing, takeoffs, and landings which involve operation outside the ROSH, ROTH, and RORH (such as at or below V_S or on the ground), the Levels shall be applied as if these conditions were in the ROSH.

REQUIREMENT RATIONALE

These paragraphs are needed to guide the application of the rest of the requirements. Considered as a group, these requirements specify the minimum handling qualities to be attained for an air vehicle operating in a normal, i.e., unfailed, state in all regions of operation in Common environmental conditions. These Normal States represent the usual modes of piloted flight. Levels of flying qualities apply generally within the ROSHs and ROTHs. Some basic requirements, generally qualitative in nature, apply in both the ROSHs and ROTHs. Provision must also be made for expected and allowable operation outside of these Regions.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1.1.1 in JSSG-2001.

The corresponding paragraphs in previous specifications are:
MIL-F-8785C: 3.1.10.1, 3.1.10.3.1, and 3.1.10.3.3
MIL-STD-1797: 4.1.6.1 through 4.1.6.3.

Where Levels are not specified, care shall be taken in selection of requirements from this standard that will not overburden the design. The impact of 5.1.1.1 has influenced the development of these requirements, but qualitative words such as “objectionable” must be taken in the context of relevance to operational use.

Note that flying qualities which “warrant improvement” according to figure 5 nevertheless meet all of the requirements if they only occur outside the ROSH.

The requirements on flight at high-angle-of-attack, dive characteristics, dive recovery devices, and dangerous flight conditions shall also apply in all pertinent parts of the RORH.

REQUIREMENT LESSONS LEARNED

VERIFICATION

The air vehicle must be evaluated by pilots in high-gain, closed-loop tasks to insure the air vehicle has achieved the required flying qualities Levels. (In the context of this standard, high-gain task means a wide-bandwidth task, and closed-loop means pilot-in-the-loop.) For the most part, these tasks must be performed in actual flight. However, for conditions which are considered too dangerous to attempt in actual flight (e.g., certain flight conditions outside the ROTH, flight in Extraordinary atmospheric disturbances, flight with certain Failure States and Special Failure States, etc.), the closed-loop evaluation task can be performed in a simulator.

Selected Air Vehicle Normal States shall be analyzed and evaluated in simulation and flight test for each applicable Flight Phase. The selected Air Vehicle Normal States shall include the loadings and configurations of requirement 5 plus a limited number of values of those parameters such as weight, moments of inertia, c.g. position, wing sweep, and throttle setting which vary continuously over a range of values during the Flight Phase. The values selected shall include the most critical values and the extremes encountered during the Flight Phase in question.

Verification of 5.1.1.1 (a), (b), and (d) shall be by flight test in the performance of appropriate tasks from table VII. These tasks shall be flown by test pilots at specific flight conditions throughout the ROSH and ROTH. The specific flight conditions to be evaluated shall be the most common operating conditions, any operating conditions critical to the mission of the air vehicle, and any conditions predicted by analysis and simulation to be worse than the Level of flying qualities required. Proof of compliance in these demonstration tasks will consist of pilot comments and Cooper-Harper (C-H) ratings. For Level 1, pilot comments shall indicate satisfaction with air vehicle flying qualities, with no worse than “mildly unpleasant” deficiencies, and median C-H ratings shall be no worse than 3.5 in Common atmospheric disturbances. For Level 2, pilot comments shall indicate that, if any handling qualities deficiencies exist, air vehicle flying qualities are at least tolerable, and median C-H ratings for these tasks shall be no worse than 6.5 in Common atmospheric disturbances.

Verification of 5.1.1.1 (c) shall be by flight test to show that the air vehicle can be flown to particular flight conditions outside of the ROTH and return readily and safely to the ROTH or ROSH. The specific flight conditions to be evaluated shall be those flight conditions outside the ROTH but inside the ROHR which are predicted by analysis and simulation to have the worst handling qualities or to be unrecoverable. Obviously, an emergency recovery system shall be installed for these tests, regardless of whether the condition is predicted to be recoverable or not—the purpose of flight testing is to verify analytical predictions. Further discussion of emergency recovery systems can be found in 5.2.5 Verification. Examples of high angle-of-attack flight conditions that shall be tested can also be found in 5.2.5 Verification. Other parameters which may form ROTH limits, such as sideslip angle, shall also be considered in

selecting flight conditions to be tested. Proof of compliance in these demonstrations will consist of pilot comments. The pilot comments shall indicate that the air vehicle can be readily and safely returned to the ROTH or ROSH and that no exceptional pilot inputs are necessary to recover the air vehicle.

TABLE VII. Suggested handling qualities evaluation tasks.

Suggested Tasks	Suggested Performance Objectives
Air-to-Air and Air-to-Ground Tracking: Gross Acquisition	Desired Performance Time to acquire: TBD by CDR Overshoots: No more than one greater than 5 mils, none to exceed 10 mils No PIO
	Adequate Performance Time to acquire: TBD by CDR Overshoots: No more than two greater than 5 mils, none to exceed 20 mils
Air-to-Air and Air-to-Ground Tracking: Fine Tracking	Desired Performance Keep the pipper within 5 mils of the target point for 3 continuous sec No PIO
	Adequate Performance Keep the pipper within 10 mils of the target point for 3 continuous sec
Close Formation	Desired Performance Excursions no greater than ±2 ft from the formation position No PIO
	Adequate Performance Excursions no greater than ±4 ft from the formation position
Aerial Refueling: Boom Tracking	Desired Performance Keep pipper within 5 mils of the boom nozzle for at least 50% of the tracking time No PIO
	Adequate Performance Keep pipper within 10 mils of the boom nozzle for at least 50% of the tracking time
Aerial Refueling: Probe-and-Drogue	Desired Performance Hook-up without touching basket webbing in at least 50% of the attempts No PIO
	Adequate Performance Hook-up in at least 50% of attempts
Offset Precision Landing: Approach	Desired Performance Flightpath control: Remain within $\pm 1^{o}$ of glideslope angle or ± ½ dot on ILS Airspeed control: Maximum of 5 kts above approach speed, minimum TBD by CDR No PIO
	Adequate Performance Flightpath control: Remain within $\pm 2^{o}$ of glideslope angle or ±1 dot on ILS Airspeed control: Maximum of 10 kts above approach speed, minimum TBD (by CDR), but not less than V_S
Offset Precision Landing: Touchdown (Conventional Air vehicles)	Desired Performance Touchdown zone: Within ±25 ft of aim point laterally, within -100 to +400 ft of aim point longitudinally Speed at touchdown: Maximum of 5 kts above landing speed, minimum TBD by CDR Attitude at touchdown: TBD by CDR Sink rate at touchdown: TBD by CDR No PIO
	Adequate Performance Touchdown zone: Within ±50 ft of aim point laterally, within -250 to +750 ft of aim point longitudinally Speed at touchdown: Maximum of 10 kts above landing speed, minimum TBD by CDR Attitude at touchdown: TBD by CDR Sink rate at touchdown: TBD by CDR

TABLE VII. Suggested handling qualities evaluation tasks - Continued.

Suggested Tasks	Suggested Performance Objectives
Offset Precision Landing: Touchdown (STOL Air vehicles)	Desired Performance Touchdown zone: Within ±10 ft of aim point laterally, within -25 to +75 ft of aim point longitudinally Speed at touchdown: Maximum of 2 kts above landing speed, minimum TBD by CDR Attitude at touchdown: TBD by CDR Sink rate at touchdown: TBD by CDR No PIO
	Adequate Performance Touchdown zone: Within ±25 ft of aim point laterally, within -100 to +400 ft of aim point longitudinally Speed at touchdown: Maximum of 5 kts above landing speed, minimum TBD by CDR Attitude at touchdown: TBD by CDR Sink rate at touchdown: TBD by CDR
Offset Precision Landing: Rollout and Takeoff Roll	Desired Performance Keep the nosewheel within ±10 ft of the runway centerline No PIO
	Adequate Performance Keep the nosewheel within ±25 ft of the runway centerline
Takeoff Rotation	Desired Performance Attitude control: Keep within ±1° of takeoff attitude Overshoots: No more than one overshoot, not to exceed TBD° (by CDR) No PIO
	Adequate Performance Attitude control: Keep within ±2° of takeoff attitude Overshoots: No more than one overshoot, not to exceed TBD° (by CDR)
Takeoff Climbout	Desired Performance Flightpath control: Keep within ±1° of specified climbout angle Groundtrack: Keep air vehicle within ±10 ft of runway centerline or within ±2° of runway heading No PIO
	Adequate Performance Flightpath control: Keep within ±2° of specified climbout angle, but not less than 0° Groundtrack: Keep air vehicle within ±25 ft of runway centerline or within ±5° of runway heading

Selection of appropriate tasks from table VII shall include consideration of the following:

During the requirements definition process, the procuring activity together with the contractor(s) and the responsible test organization, shall select several closed-loop tasks with which to evaluate the air vehicle in flight test. During system development, ground-based and in-flight simulations shall be used to get an initial appraisal of how well the air vehicle will perform the tasks in flight. The simulations can also be used to train the test pilots and refine the tasks, performance objectives, and test procedures. Handling qualities evaluation during flight test shall consist of four parts: 1) "Open-loop" tasks such as steps, doublets, and frequency sweeps for parameter identification to compare air vehicle dynamic response to the open-loop design parameters, 2) capture tasks to familiarize the pilot with air vehicle response and capture characteristics, 3) handling qualities during tracking (HQDT) for initial closed-loop handling qualities evaluation, and 4) "operational" closed-loop tasks to obtain Cooper-Harper ratings. The distinction between HQDT and "operational" tasks is discussed in 5.2.1.6.

A wide variety of closed-loop tasks has been developed for the evaluation of air vehicle flying qualities. Recommended tasks for the evaluation of flying qualities for Flight Phase Category A include air-to-air and air-to-ground tracking, particularly the well-defined set of tasks known as HQDT, aerial refueling, close formation flying, and captures. Recommended tasks for the evaluation of flying qualities for Flight Phase Category C are tracking tasks, including HQDT, close formation flying, precision landings (with and without vertical and lateral offsets), takeoffs, and captures. More detailed discussions of each of these recommended tasks will be found in subsequent paragraphs of this section and in 5.2.1.6. Other tasks which are critical to the mission that the air vehicle is intended for may also be used as evaluation tasks, such as terrain-following, weapons delivery, or LAPES. If the air vehicle has any direct force controllers, the set of evaluation tasks shall include some tasks which utilize these controllers (see 5.2.8.3.7). Suggested evaluation tasks for carrier operations are discussed in 5.2.6. Suggested evaluation tasks for V/STOL operations are discussed in 5.2.7.

There are no recommended tasks for Flight Phase Category B because this Flight Phase Category generally consists of low-gain (low-bandwidth) tasks. Possible flying qualities problems in this Flight Phase Category will normally be exposed in the more demanding tasks for Flight Phase Categories A and C, and by normal operations during the flight test program. Thus, special tasks for this Flight Phase Category are not normally considered necessary. However, Flight Phase Category B tasks are normally of much longer duration than the tasks in the other Categories. Pilot fatigue may become a significant factor in certain mission critical Category B tasks, in which case an evaluation of this kind of task might be required.

Proof of compliance in these demonstration tasks will consist of pilot comments and C-H ratings. For Level 1, pilot comments must indicate satisfaction with air vehicle flying qualities, with no worse than "mildly unpleasant" deficiencies, and median C-H ratings must be no worse than 3.5 in Calm air or in Common atmospheric disturbances. For Level 2, pilot comments must indicate that whatever deficiencies may exist, air vehicle flying qualities are still tolerable, and median C-H ratings must be no worse than 6.5 in Calm air or Common atmospheric disturbances. For Level 3, pilot comments must indicate that the air vehicle is at least controllable despite any flying qualities deficiencies, and median C-H ratings must be no worse than 9 in Calm air or Common atmospheric disturbances. In atmospheric disturbances pilot comments and C-H ratings must comply with the requirements of 5.1.2.

Actual task performance is not recommended for use as proof of compliance because it is even more subject to pilot variability than pilot comments and C-H ratings. The performance objectives suggested in the tasks described below are not intended for use as proof of compliance, but, rather, for use with the C-H scale. Specific definitions of desired and adequate performance objectives reduce pilot variability by insuring that all of the evaluation pilots attempt to achieve the same level of performance. In the performance objectives suggested below, adequate performance is set at a level sufficient to successfully perform similar tasks in operational service. Desired performance is set at a more demanding level to insure that system deficiencies are exposed. Although task performance is not recommended as proof of compliance, task performance shall be recorded and analyzed by the flight test engineers to insure that pilot ratings are reasonably consistent with the level of performance achieved and that all pilots seem to be attempting to achieve the same level of performance.

The evaluation of air vehicle flying qualities is basically a subjective science, and human variability makes analysis of the results a difficult proposition. Nevertheless, there are steps that can be taken to reduce variability in the results and insure a good evaluation. First of all, it is absolutely necessary that more than one evaluation be made for each test condition. Studies of inter-pilot C-H rating variability have indicated that three pilots is the minimum number of pilots for an adequate evaluation (CAL Report No. TB-1444-F-1 and NADC-85130-60). More pilots will increase confidence in the results, but NADC-85130-60 further demonstrated that the point of diminishing returns was reached at about six. Therefore, the recommended number of pilots per test condition is three to six. Careful selection of the evaluation pilots will also reduce the variability in results. All of the evaluation pilots must be test pilots trained in the use of the C-H scale and they all must be experienced in the Class of air vehicle under evaluation. Furthermore, for acquisition purposes, it is highly desirable that at least half of the evaluation pilots be military-employed test pilots. (Use of operational pilots to evaluate the air vehicle during the development effort can often provide additional insights into the handling qualities. Such evaluations are strongly encouraged. However, for the purpose of demonstrating compliance with this requirement, the evaluation pilots must be trained test pilots.)

In order to insure that all of the pilots attempt to achieve the same level of performance, and thus insure consistency and reduce the effects of pilot variability, it is extremely important to explicitly define the desired and adequate levels of task performance to be used for the C-H ratings. Best results are achieved with task performance defined in terms of quantifiable objectives which the pilot can readily observe in real time. Furthermore, consideration must be given to defining objectives that can also be recorded on some medium so that the flight test engineer can confirm that pilot ratings are reasonably consistent with task performance. Defining quantifiable and recordable task performance objectives and setting appropriate levels of desired and adequate performance is the most difficult part of planning the flying qualities evaluation. During flight test planning, engineers shall discuss the task performance objectives with test pilots familiar with the type of task and air vehicle type. Guidance on task objectives and lessons learned from past experience are provided below for each of the recommended tasks.

Another method to reduce the effects of variability is to use the "long-look" evaluation technique. In this technique the pilot continues or repeats the task until he is confident of his evaluation before he assigns a C-H rating (as opposed to doing it once and assigning a rating). The "long-look" approach allows the pilot a more extensive appraisal of the test condition, it allows him to weed out the effects of unique events in a single run, it allows him to get over the learning curve, and it allows him to clear his memory of characteristics he may have observed in evaluation of a preceding condition. Although the C-H rating is given only after multiple runs, pilot comments must be provided during and after each run. In order to insure that variability is not introduced by pilots doing different numbers of repeat runs, the recommended procedure is to specify a minimum number of runs to be performed before a rating can be given, and allow the pilot to make additional runs if necessary.

Pilot comments shall be considered the most important data. A C-H rating is only a summary of observed flying qualities characteristics expressed as a single number. Pilot comments identify the specific deficiencies, if any, that must be corrected. Furthermore, the "long-look" technique

filters the effects of deficiencies on the C-H rating because, over several runs, the pilot learns to compensate for some deficiencies. Since pilot comments are given for every run, the comments will identify all observed deficiencies, even those which can be compensated for. Comments on succeeding runs provide guidance on the pilot's ability to compensate for the deficiencies and the final C-H rating indicates the relative significance of these deficiencies. Therefore, pilot comments must be recorded and analyzed for every test run.

Time and cost constraints prohibit piloted evaluation of every task in every possible air vehicle configuration at every possible point in the Flight Envelope. The conditions that must be evaluated are the most common operating conditions, operating conditions critical to the mission of the air vehicle, and the worst-case conditions; especially those where the quantitative, open-loop flying qualities requirements are violated by wide margins. For air vehicles with multiple flight control modes, all mode transitions shall be evaluated at common, mission critical, and worst-case conditions, especially mode switches which are done automatically, as opposed to those deliberately switched by the pilot. Furthermore, the degradation due to atmospheric disturbances shall be demonstrated by evaluation at different levels of disturbances. Evaluation of the effect of Extraordinary atmospheric disturbances may be performed in ground simulation. When using simulation to predict the degradation of flying qualities due to Extraordinary atmospheric disturbances, it will be necessary to correlate C-H ratings gathered from simulation sessions in lower levels of turbulence with C-H ratings obtained from flight test for the same task in similar levels of turbulence.

An important source of guidelines on the use of tracking techniques for handling qualities evaluation is AFFTC-TD-75-1. AFFTC-TD-75-1 discusses execution and analysis of results of both air-to-air and air-to-ground tracking techniques. Many of the recommendations in AFFTC-TD-75-1 are also applicable to other closed-loop handling qualities evaluation techniques. Further discussions of the design and conduct of handling qualities testing and the use of the C-H scale can be found in CAL Report TB-1444-F-1, NADC-85130-60, NASA TN D-5153, AIAA 89-3358, and AIAA 90-2822.

Some of the most detailed descriptions of closed-loop evaluation tasks which have been used in the past can be found in USAFTPS and AFFTC handling qualities test plans and flight test reports. Descriptions of several tasks taken from USAFTPS Letter Reports can be found in AFFDL-TR-77-34 and AFFDL-TR-79-3126. AFFTC handling qualities reports which contain descriptions of several closed-loop evaluation tasks include AFFTC-75-15 (the YF-16), AFFTC-TR-77-11 (the A-10), AFFTC-TR-77-23 (the YF-16 Control Configured Vehicle (CCV)), AFFTC-TR-83-45 (the Advanced Fighter Technology Integration AFTI/F-16), and AFFTC-TR-91-29 [the F-15 STOL and Maneuver Technology Demonstrator (S/MTD)].

Another source of closed-loop evaluation tasks is the Standard Evaluation Maneuver Set (STEMS). The results of this project are documented in WL-TR-93-3081, WL-TR-93-3082, and WL-TR-93-3083. The main products of this project were: 1) a process to develop handling qualities evaluation maneuvers, 2) an initial set of 20 evaluation maneuvers tested in ground simulation, and 3) guidelines to help users select appropriate maneuvers. WL-TR-93-3081 describes the maneuver development process. WL-TR-93-3082 provides descriptions of the initial set of evaluation maneuvers and a selection guide. WL-TR-93-3083 documents the results

of the ground simulation tests of the initial maneuver set. The maneuvers developed in this project were primarily aimed at evaluation of agility and high-AOA flying qualities; however, there were some conventional flying qualities evaluation maneuvers as well. AIAA-93-3645 provides a summary of the STEMS project.

The following paragraphs discuss some recommended tasks and some suggested performance objectives for each task. Bear in mind that the performance objectives are not required as proof of compliance. They are intended for use with the cooper-harper scale to reduce pilot variability. Most of these tasks are equally suitable for both the "operational" technique and the HQDT technique (see 5.2.1.6). The following list of tasks is not exclusive. Any closed-loop task, performed aggressively, may be used to evaluate an air vehicle's handling qualities and PIO characteristics. When developing a specification for a particular program, the procuring agency shall discuss possible tasks and performance objectives with other procuring agencies, laboratories, test centers, and potential contractors, and the operational community.

Capture Tasks

Capture tasks are intended to evaluate handling qualities for gross acquisition as opposed to continuous tracking. A wide variety of captures can be done provided that the necessary cues are available to the pilot. Pitch attitude, bank angle, heading, flightpath angle, angle of attack (AOA), and g captures have all been done in previous programs to evaluate different aspects of the air vehicle response. These capture tasks are done almost precognitively by the pilot and are usually over so quickly that they do not lend themselves well to use with the C-H scale. It can be done, of course, but it is not necessary. Qualitative comments are sufficient proof of compliance for these tasks.

These capture tasks can give the pilot a general impression of the handling qualities of the air vehicle, but because they do not involve closed-loop tracking they do not expose all of the problems that arise in continuous control tasks. Capture tasks shall not be used as the only evaluation tasks. As a minimum, an offset precision landing task and some form of tracking task shall also be used. Capture tasks are ideal as a pre-test before performing high-gain, closed-loop tasks because they serve to familiarize the pilot with the air vehicle response before attempting the more difficult (and sometimes more dangerous) high-gain, closed-loop tasks. If hazardous motions result from the capture tasks at any flight condition, closed-loop tasks shall not be attempted at that flight condition, and the air vehicle shall be considered to have failed this requirement at that condition.

For pitch captures, the air vehicle is trimmed about a specified flight condition. The pilot aggressively captures 5° pitch attitude (or 10° if the air vehicle is already trimmed above 5°). He then makes a series of aggressive pitch captures of 5° increments in both directions. He then continues this procedure with 10° increments in both directions, and then with 15° increments in both directions. Air vehicles with more capability can continue the procedure with larger pitch excursions. If possible, the initial conditions for each maneuver shall be such that the air vehicle will remain within ±1000 ft and ±10 kts of the specified flight condition during the maneuver, however, large angle captures at high-speed conditions will inevitably produce larger speed and

altitude changes. If the air vehicle should get too far from the specified condition during a task, it shall be retrimmed about the specified condition before starting the next maneuver.

The other kinds of captures are usually done in a similar manner, with some minor differences. G captures are usually done from a constant g turn and the increments are usually ±1g, ±2g, and ±3g, and larger increments if the air vehicle has greater capability. Heading captures can be used to evaluate the yaw controller alone (usually small heading changes of 5° or less) or to evaluate coordination of yaw and roll controllers (larger heading changes).

Bank angle captures are also commonly done using bank-to-bank rolls. Starting from a 15° bank angle, the pilot aggressively rolls and captures the opposite 15° bank angle (total bank angle change of 30°). He then rolls back and captures 15° bank in the original direction. This procedure shall continue for a few cycles. The procedure is then repeated using 30° bank angles, and then repeated again using 45° bank angles. Air vehicles with more capability can continue the procedure with larger bank angles. A variation of this is to capture wings-level from the initial banked condition. Four-point and eight-point rolls, standard aerobatics maneuvers, are also good tests of roll control.

Air-to-Air Tracking

Air-to-air tracking is one of the most commonly used handling qualities evaluation techniques. Examples of the use of this kind of task can be found in many handling qualities reports. The task descriptions and performance objectives recommended below stem largely from numerous USAFTPS projects conducted in the mid-1970s to the early 1980s using the variable-stability NT-33A. The task descriptions in these projects remained fairly similar throughout this period and are documented in AFFDL-TR-77-34 and AFFDL-TR-79-3126. In the earliest of these tests the performance objectives for C-H ratings were undefined. The performance objectives gradually became better defined with succeeding projects. The performance objectives suggested in table VII reflect the objectives used in the later projects. Similar performance objectives were used on McDonnell-Douglas and Wright Laboratory ground simulators during the development of the F-15 S/MTD (WRDC-TR-89-3036).

The air-to-air tracking task consists of two phases: gross acquisition and fine tracking. Gross acquisition evaluates the ability to point and capture with moderate-amplitude inputs. Fine tracking evaluates continuous closed-loop controllability. Two different kinds of targets have been used successfully in handling qualities evaluations: a real target air vehicle and a target generated by a HUD (Head-Up Display).

If a real target is used, there are several possible target maneuvers which have been used in handling qualities evaluations in the past. In all cases the target air vehicle begins the maneuver from straight and level flight in front of the evaluation air vehicle at a specific flight condition. Throughout the maneuver the evaluation air vehicle shall remain within ±1000 ft of the test altitude and within ±50 kts of the test airspeed.

The maneuver most commonly used is an S-turn. The target air vehicle initiates a level turn at a specified load factor. After a specified time period the target unloads, reverses, and begins a

level turn in the opposite direction at the specified load factor. After a specified time period, the maneuver is terminated.

For gross acquisition, the evaluation pilot shall allow the target air vehicle to achieve a certain amount of angular displacement before he initiates his maneuver to acquire the target. Some programs have stated the angular displacement explicitly (for example, 100 mils or 30°). Other programs have used the point at which the target crosses the canopy bow to initiate gross acquisition. Yet another option is to allow a specified amount of time between the target air vehicle turn and the evaluation air vehicle turn (3 to 4 sec). Commonly used performance objectives for gross acquisition are time to acquire, the number of overshoots, and the size of the overshoots. Acquisition is defined as bringing the pipper (or whatever the pilot is using to track with) within a certain radius of some specified point on the target (tail pipe, fuselage-wing junction, canopy, etc.). Time to acquire is the time it takes to bring the pipper within this radius and keep it there. Time to acquire is a difficult objective to recommend specific values for in a general standard because it is not only a function of handling qualities, but also a function of the size of the initial angular displacement between pipper and target point and of the maximum pitch rate performance of the air vehicle under evaluation. Programs must consider both of these when determining what time to acquire to specify. An overshoot is when the pipper moves past the target point by some amount of angular displacement. Some suggested performance objectives are given in table VII.

An illustration of overshoot is shown on figure 6, which shows time histories of three theoretical gross acquisitions. Assume the desired criteria are: no more than one overshoot greater than 5 mils and no overshoots greater than 10 mils. The thick continuous line would fail these criteria because it has one overshoot greater than 10 mils (at about 0.5 sec). The thin continuous line also fails these criteria because it has two overshoots greater than 5 mils (around 0.5 sec and 0.9 sec). The dashed line meets these criteria because it has only one overshoot that exceeds 5 mils but that overshoot is less than 10 mils.

For fine tracking, the time between reversals shall allow for time to acquire plus time for extended tracking. A minimum of at least 20 sec between reversals is recommended. The nominal range between the target air vehicle and the evaluation air vehicle shall be about 1500 ft, with excursions of no more than ±500 ft from nominal. The performance objective for fine tracking is to keep the pipper within a certain radius of the target point for a large percentage of the tracking time. Some suggested performance objectives are given in table VII.

For the long-look technique, this maneuver shall be repeated a few times before giving a C-H rating. On a ground-based simulator the sequence of turns can continue uninterrupted until the evaluation pilot is confident that he has a good evaluation of the air vehicle. This evaluation shall be conducted at different airspeeds, different altitudes, and with different load factors throughout the ROSH and ROTH.

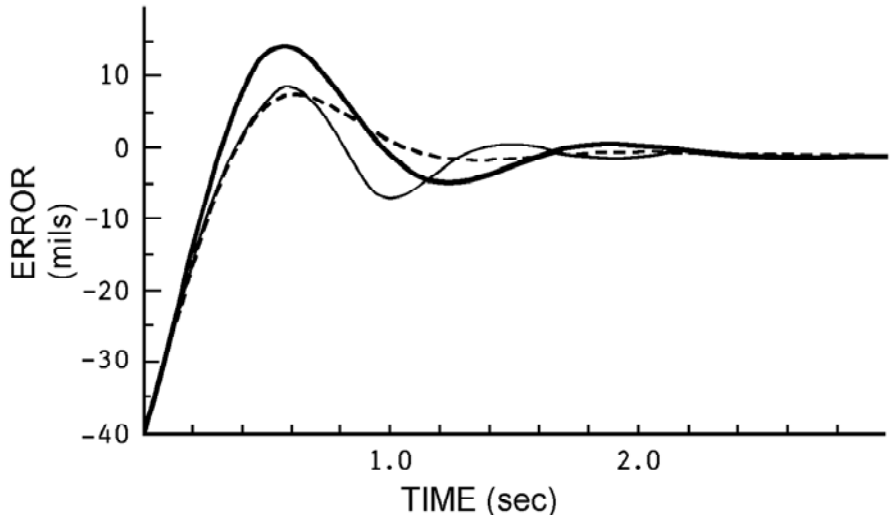


FIGURE 6. Illustration of overshoot.

Another common target maneuver is the wind-up turn. In the wind-up turn, the target air vehicle begins a turn and slowly and smoothly increases the load factor to a specified maximum load factor. The target air vehicle shall attempt to maintain a specified rate of g increase; about .2 g/sec is recommended. The maneuver is terminated shortly after the maximum load factor is reached. Gross acquisition in this maneuver is similar to that for the S-turns. For fine tracking, the rate of g increase shall allow sufficient time for gross acquisition and extended tracking time. For the long-look technique this maneuver shall be repeated a few times. This evaluation maneuver shall be initiated from various altitudes and airspeeds throughout the ROSH and ROTH.

Some other target maneuvers which have been used in air-to-air tracking are a modified Lazy-Eight maneuver, a constant-g barrel roll, and an unpredictable target maneuver. Discussions of the use of the modified Lazy-Eight and the barrel roll maneuver can be found in AFFDL-TR-79-3126. The unpredictable target is a target which is free to maneuver within certain restrictions. Normally it is restricted in airspeed (typically within ±50 kts of test condition), altitude (typically within ±1000 ft of test condition), load factor, and onset rate (typically restricted to no more than .5 g/sec).

One of the conventional evaluation tasks of the STEMS project was Tracking in Power Approach. The task was to track a target air vehicle from approximately 1500 ft range in power approach configuration at approach airspeed. The target performed gradual S-turns with periods of straight flight between turns. Constant altitude was maintained during the maneuver. The evaluation pilot selected specific aim points on the target and tracked them during the maneuver. In the simulation, different target profiles were required for different Classes of air vehicle. For fighter air vehicles, the target performed a 30° heading change every 20 sec. For transport air vehicles, a 15° heading change was performed every 15 sec. Desired performance was to keep the pipper within ±5 mils of the aim point for 50% of the task and within ±25 mils for the remainder of the task, with no PIO. Adequate performance was to keep the pipper within

±5 mils of the aim point for 10% of the task and within ±25 mils for the remainder of the task. This maneuver can be performed at a safe altitude before attempting precision landings.

An alternative to using a real target air vehicle is to do a HUD tracking task. In this task, a target symbol (tracking bars or a line-drawing of a target) is projected on the HUD which commands pitch and roll changes that the evaluation pilot must follow. The pitch and roll commands can be combinations of steps and ramps, a smoothly-varying function (such as a sum-of-sines), or a simulated target air vehicle maneuver (such as those described above). The sequence of pitch and roll commands shall be designed so as to keep the air vehicle within ±1000 ft of the test altitude and within ±50 kts of the test airspeed. The sequence shall be long enough and complex enough that the pilot cannot learn to anticipate the commands. Some example sequences that have been used before are discussed below.

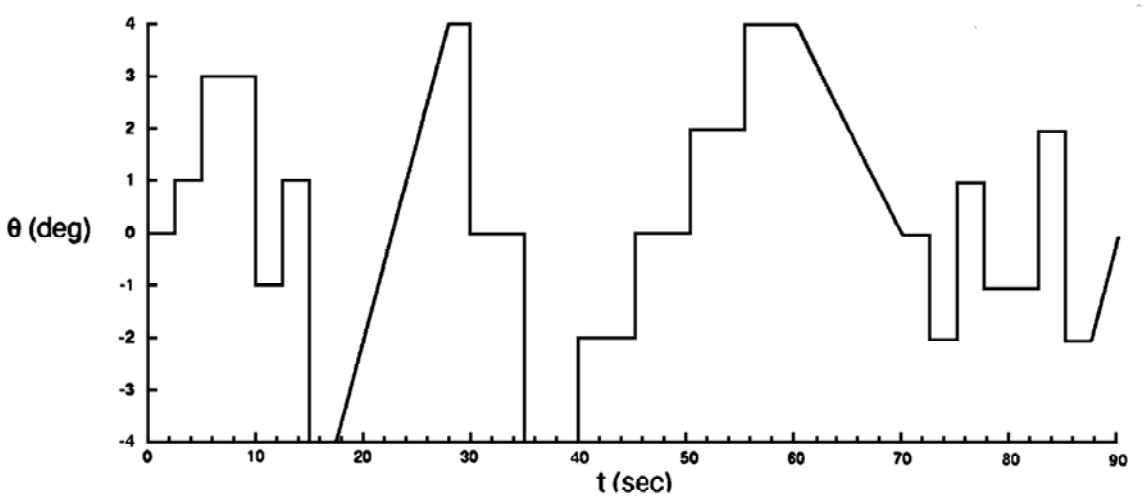
HUD tracking tasks have been used in a number of handling qualities research programs. Some example step-and-ramp tracking sequences are shown on figure 7a and figure 7b. The sequences shown in a) and b) are pitch tracking sequences that were used on the NT-33A in USAFTPS projects (USAFTPS Report 82B-4). The two sequences shown in c) are a combined pitch and roll tracking sequence used on the NT-33A and on Calspan's variable-stability LearJet in many recent projects (Calspan Report No. 7738-24). Another type of sequence in use is a sum-of-sines. This is a frequency-based function driven by an equation such as:

$$\theta_c = K\sum_{i=1}^{n} A_i \sin(\omega_i t + \phi_i)$$

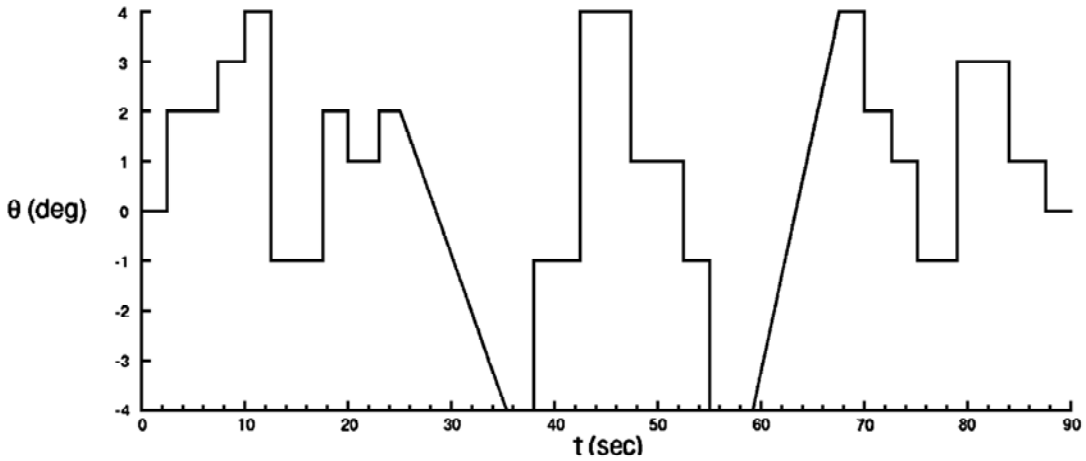
where θ_c = the commanded pitch attitude (or bank angle)
K = the task gain
n = the number of sine waves
A_i = the amplitude of each sine wave
ω_i = the frequency of each sine wave
ϕ_i = the phase of each sine wave.

Such a function was used in a project on the LearJet (AFFTC-TLR-93-41) using 13 sine waves evenly spaced in frequency between 0.1 and 30 rad/sec. The HUD symbology usually used in HUD tracking tasks are tracking bars, but with the computational power and electronic displays available today, it is worth considering special flight test software to provide a more definitive target.

One advantage of HUD tracking tasks is that, if the HUD update rate and HUD dynamic characteristics are duplicated in the simulators, the task itself can be identical between ground simulation, in-flight simulation, and flight test, providing a greater degree of commonality among these three stages of evaluation. Bear in mind, though, that HUD dynamic characteristics will affect handling qualities more in these tasks than in tasks such as formation flying or VFR (Visual Flight Rules) landings where the pilot is flying based on external visual references.

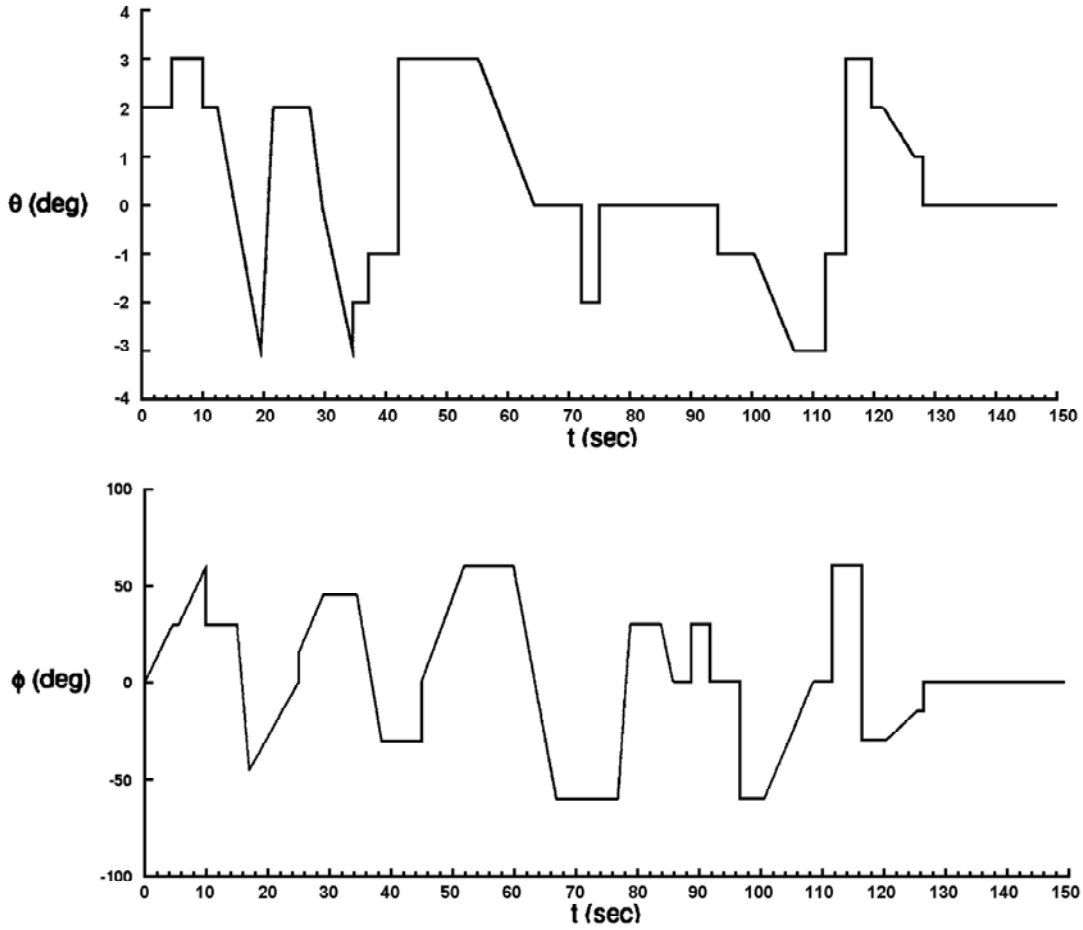


a) Pitch tracking task



b) Pitch tracking task

FIGURE 7a and b. Example step-and-ramp HUD tracking sequences.



c) Combined pitch and roll tracking task

FIGURE 7c. Example step-and-ramp HUD tracking sequences.

Air-to-Ground Tracking

The air-to-ground tracking task has two phases: gross acquisition and fine tracking. Gross acquisition evaluates the ability to switch from one target to another. Fine tracking evaluates the ability to continuously track a target.

For this task, the air vehicle flies at a specified glideslope and airspeed toward a group of widely-spaced targets on the ground. The airspeed and glideslope shall be representative of the intended operational application for the air vehicle. Initial range to the targets shall allow time for acquisition and tracking of several targets. The targets shall be from 60 to 180 ft apart perpendicular to the flightpath and anywhere from 90 to 360 ft apart parallel to the flightpath. The pilot aggressively captures the first target and tracks it for a specified period of time (4 sec is recommended), and then acquires and tracks succeeding targets. The sequence of targets to be tracked shall be specified in advance. As the air vehicle approaches the targets the angular displacement between the targets will increase. Therefore, at long range the sequence shall require switching between the more widely-spaced targets (from one end of the group to the

other end, for example). As the range closes the sequence shall require switching between targets which are closer together (adjacent targets). The last target switch shall require a pitch-up. A minimum recovery altitude shall be specified at which the pilot must abandon the task. This minimum altitude shall consider the airspeed and dive angle of the task and shall allow sufficient margin for the pilot to pull out. Suggested performance objectives are the same as those for air-to-air tracking. Some suggested performance objectives are given in table VII.

Previous flight test programs which have used air-to-ground tracking techniques include the A-10 (AFFTC-TR-77-11 and Brandeau, AFFDL-TR-78-171), YF-16 CCV (AFFTC-TR-77-23), and AFTI/F-16 (AFFTC-TR-83-45). A very promising system for an air-to-ground tracking task, called GRATE (Ground Attack Targeting Equipment), was developed and tested by the Deutsche Forschungs-und Versuchsanstalt fur Luft-und Raumfahrt (DFVLR) in the 1980s (Koehler, NASA CP 2428 and Koehler, AGARD-CP-452). This task used a pattern of lights on the ground as a target. The pilot acquires and tracks each light in turn as the lights are illuminated in a specific sequence. The Germans evaluated this task with an AlphaJet with great success. The task was also subsequently used successfully on a ground-based simulator (Biezad, AFWAL-TR-86-3093). In 1987 the Dryden Flight Research Center (DFRC) developed a derivative system known as the Adaptable Target Lighting Array System (ATLAS). This system has been tested with the NT-33 (USAFTPS-TR-88A-TM1), the X-29 (NASA TM 101700), and the F-15 S/MTD. An example of a typical ATLAS array is shown on figure 8. The ATLAS system is currently operational at DFRC at Edwards AFB.

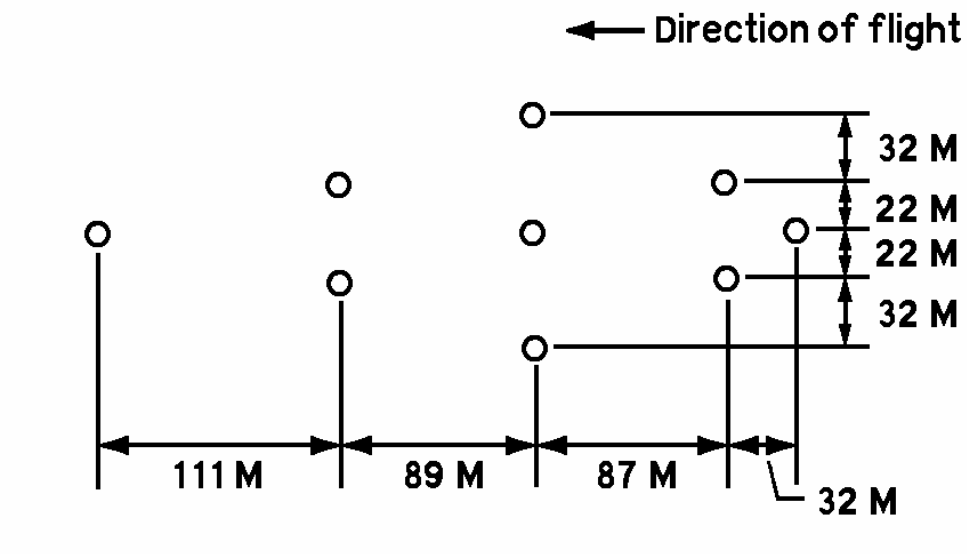


FIGURE 8. Typical ATLAS light pattern (from NASA TM 101700).

Close Formation Task

While the tracking tasks put a lot of emphasis on attitude control, close formation tasks put more emphasis on flightpath control. The task consists of holding close formation with a target air vehicle as it maneuvers. Both wing and trail formations are used. The performance objective is to maintain relative position between the target air vehicle and the air vehicle under evaluation.

Specific objectives are difficult to recommend because the pilot usually uses visual alignment of some part of his air vehicle against the target air vehicle to gauge his position, and, of course, this will vary with evaluation air vehicle and target air vehicle. Some suggested displacements to use for performance objectives are given in table VII; however, consideration shall be given to the nature of the target maneuvers. More relaxed performance objectives shall be used for the more extreme target maneuvers. A procuring agency will have to determine what these numbers mean in terms of visual alignment with the target air vehicle in their program.

The target maneuvers used in this task are usually the same or similar to the ones used in the air-to-air tracking task. For the long-look technique the task shall be done a few times in the wing formation and then done a few times in trail formation and then rated. The pilot's objectives in this task are outside visual references which are difficult to gauge and are usually not recorded by any medium (unless a video camera is specially mounted for this task or an observer is carried in one of the air vehicles).

Formation flying tasks were used in the evaluations of the YF-16 CCV (AFFTC-TR-77-23), the AFTI/F-16 (AFFTC-TR-83-45), and the F-15 S/MTD (AFFTC-TR-91-29 and WL-TR-92-3027). The YF-16 CCV evaluation included station keeping and position changes during straight and level flight and in "mild to moderate" lazy-eight maneuvers. Vertical and horizontal position changes of 500 ft and 50 ft were made.

AFTI/F-16 formation tasks were done in close trail and wingtip positions. The following descriptions are taken from AFFTC-TR-83-45.

> "The close trail evaluations were conducted with the AFTI/F-16 in a position ten feet below the lead aircraft with zero to ten feet nose-to-tail separation. The lead aircraft flew between 250 to 300 KCAS while making gentle maneuvers. This maneuvering included bank angle variations up to 30º and gentle climbs and descents."
>
> "The close formation evaluations were conducted with the AFTI/F-16 in a tight position off the wing of the lead aircraft. After a reasonable buildup, the AFTI/F-16 pilot flew as close to the lead aircraft as was comfortable. The lead aircraft flew either in level flight or through a series of lazy eight maneuvers. Bank angle changes ranged between ±90º, pitch attitude ranged between ±45º, airspeed between 200 and 500 KCAS and load factor between one and five g's. In this task the pilots attempted to maintain a precise position relative to the lead aircraft using fingertip formation techniques. The AFTI/F-16 would intentionally make lateral and vertical deviations in order to evaluate the aircraft's ability to return to the nominal position."

In the F-15 S/MTD program a formation task was used to evaluate PIO tendencies in power approach. The pilot attempted to maintain formation with an A-37 while the A-37 performed random 0.25-g step inputs (three- to five-ft vertical variations).

Aerial Refueling

There are two types of aerial refueling: probe-and-drogue and flying boom. To date, evaluation tasks associated with both types of aerial refueling have not been formalized as much as those for tracking and precision landing tasks. Some formalized tasks which have been used in previous programs are described below.

For boom-type aerial refueling, the most frequently used evaluation task is some type of boom tracking task. Two approaches have been tried. The first approach is to track the nozzle of the boom with a waterline symbol or a pipper on the HUD or windscreen. The evaluation air vehicle takes station in the pre-contact position about 50 ft aft and down on a 30º line from the tanker. The tanker maintains steady level flight with the boom extended. The boom may be held stationary or moved around slowly (no more than 1º/sec). The tracking time shall be extensive for a good evaluation: two to four minutes is recommended. The performance objective is to keep the pipper or waterline symbol within a certain radius of the nozzle of the boom for a large percentage of the tracking time. Some suggested performance objectives for this approach are given in table VII.

Boom tracking with a HUD pipper or a waterline symbol has been done in several programs at AFFTC. In every case, however, it was done using the HQDT technique (described in 5.2.1.6 Verification), where the objective was zero pipper error. At this time there is no data to support the suggested performance objectives for this task other than the fact that these objectives have been successfully applied in other kinds of tracking tasks.

The STEMS project also tested a boom tracking task. The task was to track the refueling probe of a tanker from the pre-contact position. The evaluation pilot can track a steady probe, periodically changing aim points on the boom (such as the boom wingtips), or the boom operator can make small random horizontal and vertical movements with the boom to create tracking errors. Desired performance in the STEMS project was to maintain the aim point within a 30-mil radius of the pipper for at least 50% of the task, with no objectionable PIOs. Adequate performance was to maintain the aim point within a 50-mil radius of the pipper for at least 50% of the task.

In the second approach, the evaluation pilot attempts to keep the end of the boom visually aligned with some point on the tanker air vehicle. The evaluation air vehicle maintains the pre-contact position within about ±10 ft. The boom is held stationary in this approach. The recommended tracking time is 2 to 4 min. The performance objective in this approach is to keep the end of the boom visually aligned within a clearly discernible area on the tanker for a specified percentage of the tracking time. It is difficult to recommend performance objectives for this approach because they will depend on the type of tanker in use. However, an example of a project which used this approach with a KC-135 is given below.

Boom tracking using visual alignment with the tanker was proposed as a handling qualities task by the 4950th Test Wing in a flight test project to develop an aerial refueling evaluation task for Class III air vehicles. The tanker air vehicle in their project was a KC-135. The visual desired zone for their task was between the rivet lines on the bottom of the KC-135. The adequate zone

was the edges of the fuselage. A diagram of these zones is shown on figure 9. In the test they used two different evaluation air vehicles: a C-135 and a C-18. The performance objective was the cumulative time the evaluation pilot could keep the boom nozzle aligned within the desired zone during 2 min of tracking. Four different levels of desired performance were tested: 30 sec, 45 sec, 60 sec, and 75 sec. Adequate performance was defined as keeping the nozzle within the adequate zone for the entire 2 minutes. Unfortunately, the pilots considered both air vehicles Level 2 for this task because of the amount of compensation required. The results for all four levels of desired performance gave Level 2 C-H ratings for both air vehicles, thus the C-H ratings give no indication of which is the best value to use for desired performance. The results do lend credence to their choice of adequate criteria, but this was not a variable in the test. Nevertheless, based on the performance achieved with both air vehicles throughout the project, the pilot consensus was that 60 sec in the defined desired zone (or about 50% of the tracking time) was "both attainable and realistic" and that the task was demanding enough to expose undesirable handling qualities. This project was documented in 4950-FTR-93-05.

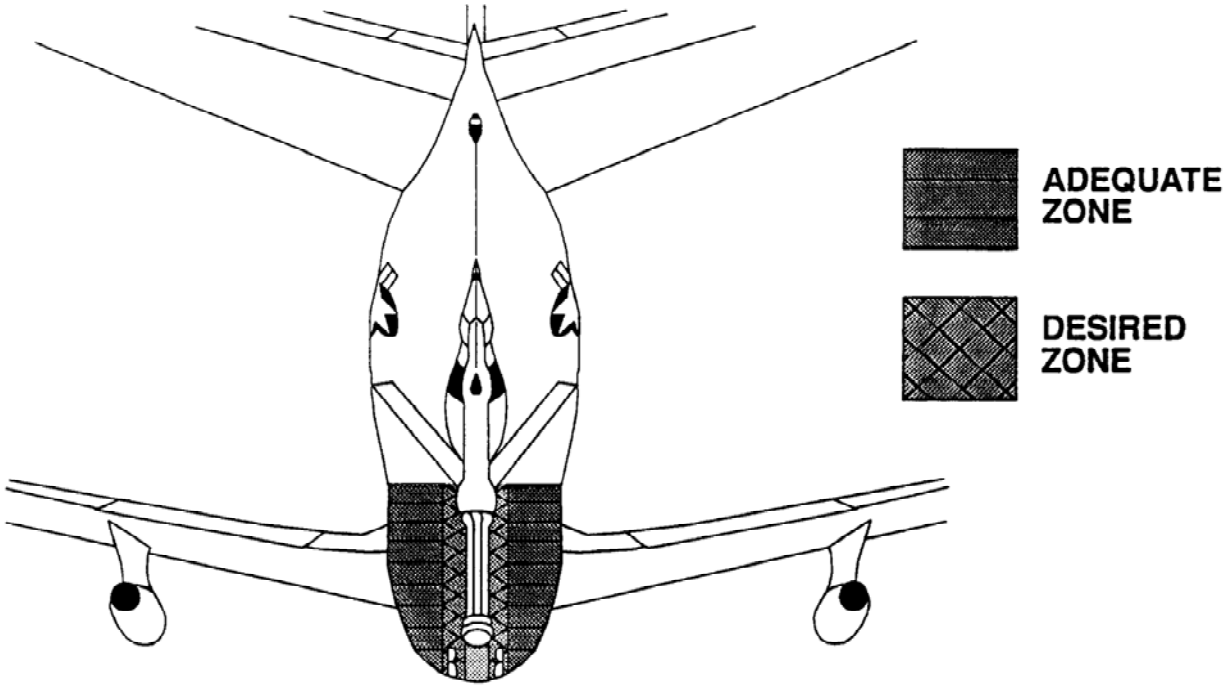


FIGURE 9. Adequate and desired performance for 4950th boom tracking task (from 4950-FTR-93-05).

For probe-and-drogue refueling actual hook-ups have been used as evaluation tasks. In one such program, the performance objective was the ratio of successful hook-ups to attempted hook-ups. The task starts from the standard pre-contact position. When cleared for contact, the evaluation pilot establishes a 3-5 kt closure rate towards the drogue and attempts to make contact. If the drogue is successfully engaged, the evaluation pilot stabilizes for approximately 30 sec, and then establishes a 3-5 kt separation rate to disconnect and return to the pre-contact position. If the closure rate exceeds 5 kts, the probe tip passes the outside edge of the drogue basket, the probe

tips the basket, or if a hazardous situation develops, the hook-up attempt is aborted and the evaluation pilot returns to the pre-contact position before making another attempt. The performance objective is a certain percentage of successful hook-ups out of a specific number of attempts. Six to twelve is the recommend number of attempts. Some suggested performance objectives are given in table VII.

Probe-and-drogue refueling was used as an evaluation task in an experiment with the NT-33A documented in AFFDL-TR-74-9. Standard probe-and-drogue refueling procedures were used. However, adequate and desired performance objectives were not explicitly defined for this project. The performance objectives suggested in table VII are taken from a USAFTPS study of response-types for probe-and-drogue refueling performed with the NT-33A in October 1993.

A position-keeping evaluation task has been used extensively for tanker evaluations including the S-3 with a buddy store, the KC-10, the KC-130, and the KC-135. A detailed description of this test technique and lessons learned from these test programs is provided in AGARD CP-519.

Offset, Precision Landing

The standard evaluation task for approach and landing is an offset, precision landing. There are up to three phases in this task: approach, touchdown, and, sometimes, rollout. The approach phase evaluates the ability to control flightpath, airspeed, and attitude, including flap and gear-down transients and large amplitude maneuvers. The touchdown phase evaluates the ability to control flightpath, airspeed, and attitude to a precise touchdown in the presence of ground effects and through touchdown transients. Rollout evaluates ground handling after touchdown.

The approach phase evaluation begins about a mile out on final approach, with gear and flaps up, at the required glideslope angle, but with a lateral offset from the runway centerline of about 150 to 300 ft, and a vertical offset from the glideslope of about 100 to 200 ft. Soon after the task begins, the pilot lowers gear and flaps to the landing configuration. The pilot maintains precise flightpath angle and airspeed control throughout the approach phase up to the offset correction point. Some suggested performance objectives are given in table VII. Determination of performance objectives for flightpath control shall consider what cues are available to the pilot. Determination of performance objectives for airspeed control must consider the margin between the recommended approach speed and V_S.

Lateral offsets of 200 ft or less shall be corrected at 150 ft Above Ground Level (AGL). Lateral offsets of more than 200 ft shall be corrected at 200 ft AGL. The pilot shall make an aggressive correction to the glideslope. The correction shall be completed with the wings approximately level by 50 ft AGL to avoid the possibility of striking the ground.

The touchdown phase begins at about 50 ft AGL. The pilot attempts to put the main wheels down inside a designated touchdown zone at a specified landing speed, attitude, and sink rate. The landing zone shall be clearly identifiable on the runway. Performance objectives are the touchdown location, landing speed, attitude, and sink rate at touchdown. Some suggested performance objectives for conventional air vehicles are given in table VII. Determination of performance objectives for airspeed control must consider the margin between the recommended

landing speed and V_S. Similarly, performance objectives for attitude at touchdown must consider air vehicle geometry (to preclude wingtip or tail strikes, etc.) and landing gear limitations (side force limits, etc.). Landing gear limitations must also be considered in the determination of performance objectives for sink rate at touchdown.

The rollout phase begins after touchdown. The pilot steers from the touchdown point to the runway centerline and thereafter stays on the runway centerline while bringing the air vehicle to a stop. For STOL air vehicles a stopping distance shall be specified (usually set by mission performance requirements). Some suggested performance objectives are given in table VII. Landing rollout shall be evaluated with and without crosswinds using rudder and nosewheel steering or differential braking. Landing rollout might also be evaluated under various runway conditions (dry, wet, icy, patched, etc.). Ground handling qualities should be expected to degrade with degraded runway conditions in a similar manner to the way handling qualities degrade with atmospheric disturbances, however, at this time there is no available guidance on how much degradation to allow under various runway conditions. The best that can be said at this time is that ground handling shall be Level 1 for normal runway conditions, and if the air vehicle cannot be kept on the runway under certain conditions, the air vehicle shall be considered uncontrollable for those conditions. As is the case in evaluations with atmospheric disturbances, dangerous runway conditions shall only be tested in ground simulation.

In flight test, bringing the air vehicle to a full stop on the runway on every run is inadvisable due to cost and time constraints. Therefore, for the long-look technique, approach and touchdown shall be evaluated by doing a few touch-and-goes before giving a C-H rating. Rollout shall be evaluated on the last landing, which is brought to a complete stop. On a ground-based simulator, the air vehicle shall be brought to a stop every time. On ground-based simulators, pilots tend to be "lower gain" than they are in flight. To counter this, Common random turbulence and fairly large discrete gusts shall be introduced throughout the task. In particular, a discrete gust shall be introduced after the offset correction. Degradation with atmospheric disturbances (5.1.2) shall be evaluated by increasing the turbulence and the gusts and by adding crosswinds and wind shears.

Because this task is done in close proximity to the ground, it shall not be attempted if other evaluations (analysis, ground simulation, or flight test) indicate a high probability of Level 3 handling qualities or hazardous PIO tendencies. Therefore offset precision landing tasks shall be performed on a ground-based simulator and approach handling qualities shall be evaluated with other in-flight tasks (such as HQDT) before attempting actual offset precision landings in flight test. Obviously, if this task is considered too dangerous to attempt, the air vehicle is considered to have failed this requirement.

This is another task which is so widely used for handling qualities evaluations that descriptions of it can be found in many handling qualities reports, particularly those that deal with approach and landing. The suggested performance objectives for the approach and touchdown phases are taken largely from Calspan experience with the NT-33A and the Total In-Flight Simulator (TIFS). Discussions of the use of this task can be found in NASA CR 172491, NASA CR 178188, AFWAL-TR-81-3118, and AIAA 93-3816.

Some data on the size of the suggested landing zone was provided in a USAFTPS project which studied the effect of different performance objectives on touchdown C-H ratings with three different Class IV air vehicles: the F-15D, the F-16D, and the F/A-18B. The experiment looked at three different desired landing zones: 25 ft wide by 200 ft long, 50 ft wide by 400 ft long, and 75 ft wide by 600 ft long. C-H ratings for each air vehicle and each landing zone were compared with the pilots' qualitative appraisal of each air vehicle. For the 75 by 600-ft zone, all three air vehicles received basically Level 1 C-H ratings. For the 50 by 400-ft zone, the F-18 received basically Level 1 C-H ratings, the F-15 had borderline Level 1/Level 2 C-H ratings, and the F-16 got basically Level 2 C-H ratings. For the 25 by 200-ft zone, all three air vehicles received basically Level 2 C-H ratings. The project report found the results from the 50 by 400-ft zone to be most representative of the pilots' qualitative opinions of each air vehicle.

Additional data on the size of the suggested landing zone was provided in a project by the 4950th Test Wing which studied the effect of different performance objectives on touchdown C-H ratings with three different Class III air vehicles: the C-141A, the C-135A/E, and the C-18B. The experiment looked at four different desired landing zones: 40 ft wide by 400 ft long, 20 ft wide by 200 ft long, 60 ft wide by 800 ft long, and 80 ft wide by 1000 ft long. C-H ratings for each air vehicle and each landing zone were compared with the pilots' qualitative appraisal of each air vehicle. All three air vehicles were considered Level 2 for the landing task. The final report (4950-FTR-93-05) recommended the 40 by 400-ft zone. For this zone, desired performance was met 6 out of 12 times, and the pilot ratings comprised two C-H ratings of 3 and four C-H ratings of 4. Interestingly, this suggested landing zone is very similar to the one recommended by USAFTPS for Class IV air vehicles.

Performance objectives for Short Take-Off and Landing (STOL) and carrier-based air vehicles shall be more demanding. Suggested performance objectives for STOL air vehicles are given in table VII and also in table XC. Suggested performance objectives for carrier operations are given in table LXXXVIII.

The performance objectives for STOL in table VII are taken from experience on the F-15 S/MTD program. Discussions of the tasks used in the S/MTD program can be found in AFWAL-TM-87-180-FIGC, WRDC-TR-89-3036, AFFTC-TR-91-29, and WL-TR-92-3027.

Takeoff

Takeoff tasks have not been done as often as landing and tracking tasks, so there is little practical experience on which to base the task recommended here. The task consists of three phases: takeoff roll, rotation, and climbout. The takeoff roll evaluates ground handling from brake release to takeoff rotation. The rotation phase evaluates ability to control attitude during takeoff. The climbout phase evaluates ability to control flightpath after takeoff, including leaving ground effect and flap and gear transients.

The takeoff roll begins from takeoff condition at the downwind end of the runway. The pilot advances the throttles to a specified setting and releases the brakes. The task is to track the runway centerline as the air vehicle accelerates. The suggested performance objectives for this

phase are the same as those for landing rollout. Some suggested performance objectives for the takeoff roll are given in table VII.

At a specified speed the pilot briskly rotates the air vehicle to takeoff attitude. Performance objectives in this phase are attitude control, number of overshoots, and size of overshoots. An overshoot in this case is defined as any deviation above the specified takeoff attitude. The purpose of the overshoot limit is to prohibit over-rotation. Some suggested performance objectives for takeoff rotation are given in table VII. Determination of performance objectives for overshoot are dependent on air vehicle geometry and the recommended takeoff attitude.

After main wheel liftoff, the pilot maintains a specific flightpath angle and groundtrack. He maintains this flightpath until the landing gear have been retracted and all transients have settled out. Some suggested performance objectives for takeoff climbout are given in table VII. Determination of performance objectives for flightpath control shall consider what cues are available to the pilot. In most cases pitch attitude is used as a substitute when flightpath angle is not an available cue. The tolerance for adequate flightpath control shall not allow a negative flightpath angle. For groundtrack control, heading angle may be used as a substitute for deviation from runway centerline.

It would be impractical to evaluate takeoff roll with a long-look technique in flight test because the air vehicle would have to land and taxi back to the end of the runway each time. However, rotation and climbout could be evaluated with a long-look technique by doing touch-and-goes. On a ground-based simulator the entire task could be done using a long-look technique. As with the landing tasks, Common random turbulence and Uncommon discrete gusts shall be used to increase the pilot's "gain" on a ground-based simulator. Degradation with atmospheric disturbances (5.1.2) can be evaluated on the ground simulator by increasing the turbulence and gusts and by adding crosswinds and wind shears.

The takeoff task described in table VII was taken from the F-15 S/MTD program. This task was only used as an evaluation task on the McDonnell-Douglas and Wright Laboratory simulators. It was not used as an evaluation task in the flight test program. Descriptions and results of the use of this task can be found in AFWAL-TM-87-180-FIGC and in WRDC-TR-89-3036.

5.1.1.2 Allowable Levels for Air Vehicle Extreme States.
The required Level of flying qualities for Air Vehicle Extreme States in Common environmental conditions shall be as shown in table VIII.

TABLE VIII. Allowable Levels for Air Vehicle Extreme States.

Air Vehicle Extreme State	Flight Phase	Region of the Flight Envelope	Required Level of Flying Qualities

REQUIREMENT RATIONALE

Levels of flying qualities as defined in 3.2 are employed in this standard in realization of the possibility that the air vehicle may be required to operate under abnormal conditions. Such abnormalities that may occur as a result of extreme loadings are permitted to comply with a degraded Level of flying qualities. This requirement is a flight safety consideration.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1.1.2 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommended values for table VIII:

Level 1 flying qualities shall be required for all Air Vehicle Normal States in the ROSH and Level 2 in the ROTH, however, this may not always be practical. If there are a few exceptions (and there should only be a few), they shall be identified as Air Vehicle Extreme States (4.1.3.6). The best practical Level of flying qualities attainable shall be required for each Air Vehicle Extreme State. Since the best practical Level of flying qualities attainable will probably depend on flight conditions, it may be necessary to require a different Level of flying qualities in each region of the flight envelope (ROSH, ROTH, and RORH). Once the best practical Level of flying qualities has been determined for each Extreme State in each applicable Region of Handling, the contractor shall complete table VIII.

REQUIREMENT LESSONS LEARNED

VERIFICATION

All Air Vehicle Extreme States shall be analyzed and evaluated by simulation and flight test for each applicable Flight Phase. Verification shall be by flight test in the performance of appropriate tasks from table VII for the applicable Flight Phases. These tasks will be flown by test pilots at specific flight conditions throughout the ROSH and ROTH. The specific flight conditions to be evaluated shall be any operating conditions critical to the mission of the air

vehicle and any conditions predicted by analysis or simulation to be worse than the Level required. Proof of compliance in these demonstration tasks will consist of pilot comments and C-H ratings. For Level 2, pilot comments shall indicate that, if any handling qualities deficiencies exist, air vehicle flying qualities are at least acceptable, and median C-H ratings for these tasks shall be no worse than 6.5 in Common atmospheric disturbances. For Level 3, pilot comments shall indicate that the air vehicle is at least controllable despite any flying qualities deficiencies, and median C-H ratings shall be no worse than 9 in Common atmospheric disturbances.

The contractor shall take care that the specific tasks and performance objectives selected for each Extreme State are appropriate for that Extreme State.

5.1.1.3 Primary requirements for Failure States.

The Levels of flying qualities for Failure States shall meet the requirements of 5.1.1.3.1 through 5.1.1.3.3.

REQUIREMENT RATIONALE

Higher performance of air vehicles has led to ever-expanding Flight Envelopes, increased control system complexity, and the necessity to face the problem of equipment failures in a realistic manner. The specification of Levels corresponding to Failure States is intended to achieve adequate flying qualities without imposing undue requirements that could lead to unwarranted system complexity or decreased flight safety. For example, an air vehicle with two separate pitch controllers is safer from the standpoint of controller jam, but the probability of such a failure is higher. Without actually requiring a good handling basic airframe, the specification requires:

high probability of good flying qualities where the air vehicle is expected to be used

acceptable flying qualities in reasonably likely, yet infrequently expected, conditions

a floor to assure, to the greatest extent possible, at least a flyable air vehicle no matter what single failures occur

a process to assure all the ramifications of reliance on powered controls, stability augmentation, etc., receive proper attention.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.10.2 and 3.1.10.2.1
MIL-STD-1797: 4.1.7.1.

Two procedures are presented to allow the contractor to quantify the allowable degradation in flying qualities due to Failure States.

The procedure in 5.1.1.3.1 is similar to that from MIL-F-8785C. It involves the following failure probability calculations:

Identify those Air Vehicle Failure States which have a significant effect on flying qualities (see 4.1.3.7).

Calculate the probability of encountering various Air Vehicle Failure States per flight hour. (This is one change from MIL-F-8785C, which calculated the probability per flight instead of per flight hour.)

Determine the degree of flying qualities degradation associated with each Failure State.

Compute the total probability of encountering Level 2 and 3 flying qualities in the ROSH. This total will be the sum of the probability of each failure if the failures are statistically independent.

The procedure in 5.1.1.3.2 assumes that certain listed failures and combinations of failures will occur sometime (with probability 1). Requirements are set on the degree of flying qualities degradation allowed for each of these Failure States. As with the first procedure, the degraded flying qualities for each selected Failure State are then evaluated and compared to the requirement for that Failure State. This approach is referred to as specific failure analysis.

Generally, the requirements consider only degradations in a single flying quality. The contractor should recognize that degradations in several flying qualities parameters can have an effect worse than any one of those degradations. However, data definitive enough for a specification are not available.

In addition to these requirements, there are further requirements on the flying qualities effects of certain failures, e.g., flight control system and engine failures (5.2.1.5 and 5.2.3.11, respectively), etc. Also, note that the factors called out in 5.1.3 include Air Vehicle Failure States and maneuvering flight appropriate to those Failure States.

REQUIREMENT LESSONS LEARNED

See 5.1.1.3.1 through 5.1.1.3.3.

VERIFICATION

Flying qualities Levels for Failure States shall be verified by meeting the requirements of 5.1.1.3.1 through 5.1.1.3.3.

5.1.1.3.1 Probability of encountering degraded levels of flying qualities due to failures. Assuming Calm air, the overall probability of encountering Level 2 flying qualities due to one or more failures shall be less than __(1)__ per flight hour within the ROSH, and the overall probability of encountering Level 3 flying qualities due to one or more failures shall be less than __(2)__ per flight hour within the ROSH and less than __(3)__ per flight hour within the ROTH. In no case shall a Failure State (except a Special Failure State) degrade flying qualities to worse than Level 3.

REQUIREMENT RATIONALE

For an Air Vehicle Failure State, a degradation in flying qualities is permitted only if the probability of encountering a lower Level than specified in 5.1.1.1 is sufficiently small. The probability of a degraded Level of flying qualities is related to, but not exactly the same as, mission or flight-safety reliability. A degraded flying qualities Level is allowed for some infrequently expected events: failure of air vehicle systems or flight outside the ROSH, near the air vehicle's limits (by definition, the ROSHs encompass the design missions). Degradation with environmental conditions is discussed in 5.1.2.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1.1.3.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.10.2 and 4.1.1.1
MIL-STD-1797: 4.1.7.3.

Recommended value for blank (1): 10^{-3}
Recommended value for blank (2): 10^{-4}
Recommended value for blank (3): 10^{-2}

In MIL-F-8785C and MIL-STD-1797A, failure probabilities were specified per flight, whereas here they have been specified per flight hour. The recommended values were chosen so that failures which degrade flying qualities would not contribute disproportionately to reduction or loss of mission effectiveness, or to flight safety problems. The form is consistent with failure rate data, which usually are presented per flight hour.

The numbers are given as orders of magnitude. When predicting the occurrence of events of such small probability, that is about the most accuracy that can be expected. For comparison, AFGS-87242A recommended the following probability limits for the entire flight control system, manual and automatic, with somewhat different ground rules:

Probability of mission failure < 10^{-3} per flight

Probability of loss of control < 10^{-7} per flight.

By comparison, Federal Aviation Regulation (FAR) Part 25 paragraph 25.671 for the flight control system states that:

> Probable malfunctions (malfunctions with a probability > 10^{-3} per hour) are allowed to have only minor effect.
>
> Extremely improbable failures (failures with a probability < 10^{-9} per hour) need not be considered.
>
> For all other failures/combinations continued safe flight and landing must be assured.

The probability of flying qualities degradation is influenced by a number of factors such as design implementation and complexity (including reconfiguration capability), computer reliability improvements, lightning protection, built-in test (BIT), maintenance practices, and dispatch rules. Peacetime versus wartime operation can be a necessary concern, although battle damage is a separate consideration.

REQUIREMENT LESSONS LEARNED

Based on experience with past air vehicles and current projected states of the art, the recommended values are reasonable. However, the numerical values shall reflect specific requirements for a given weapon system. The contractor shall, as a matter of course, confer with the procuring activity, the using-command, and the reliability engineers to assure that the probabilities associated with the Levels are consistent with the overall design goals.

VERIFICATION

The contractor shall determine, based on the most accurate available data, the probability of occurrence of each Air Vehicle Failure State per flight hour and the effect of that Failure State (assuming Calm air) on the flying qualities within the ROSHs and ROTHs. From these Failure State probabilities and effects, the contractor shall determine the overall probability, per flight hour, that one or more of the flying qualities are degraded to Level 2 because of one or more failures. The contractor shall also determine the probability that one or more flying qualities are degraded to Level 3. These probabilities shall be less than the recommended values. In no case shall a Failure State (except a Special Failure State) degrade flying qualities to worse than Level 3.

This approach addresses the reliability of flying qualities, rather than the reliability of hardware and software per se. This requirement provides a solid analytical method for accounting for the effects of failures. It serves to compel a detailed Failure Modes and Effects Analysis from the flying qualities standpoint. Such an analysis is vital as both system complexity and the number of design options increase.

Until some time downstream in the design process, the flight control and related subsystems will not be defined in enough detail to permit a comprehensive listing of failure possibilities, much less to estimate their likelihood. Initially, this requirement serves as guidance in selecting a design approach and components and redundancy levels which can potentially achieve or surpass the stated probabilities of not encountering the degraded Levels. As the design progresses, reliability analyses and failure modes and effects analyses will provide the means of determining compliance.

To determine theoretical compliance with the requirements of 5.1.1.3, the following steps shall be performed:

1. Identify those Air Vehicle Failure States which have a significant effect on flying qualities (4.1.3.7).

2. Determine the probability of encountering various Air Vehicle Failure States per flight hour (5.1.1.3.1).

3. Determine the degree of flying qualities degradation associated with each Air Vehicle Failure State in terms of Levels as defined in the specific requirements. These determinations shall be based on analysis of quantitative flying qualities criteria and on the assessment of the effects of those failures which are evaluated in simulation and flight test.

4. For each Air Vehicle Failure State, determine the points in the ROSH and in the ROTH where that failure is most critical (in a flying qualities sense).

5. Assuming the failures have occurred at the most critical point in the ROSH, compute the total probability of encountering Level 2 flying qualities in the ROSH due to equipment failures. Likewise compute the probability of encountering Level 3 flying qualities in the ROSH. Finally, compute the probability of encountering Level 3 flying qualities in the ROTH assuming that the failures have occurred at the most critical point in the ROTH. Special Failure States need not be considered in determining the probability of encountering degradation to Level 2 or Level 3.

6. Compare the computed values above with the recommended values in 5.1.1.3.1. An example which illustrates an appropriate estimate of the probabilities of encounter follows: if the failures are all statistically independent, determine the sum of the probabilities of encountering all Air Vehicle Failure States which degrade flying qualities to Level 2 in the ROSH. This sum must be less than 10^{-3} per flight hour.

If the requirements are not met, the contractor shall consider alternate courses such as:

a. Improve the aircraft flying qualities associated with the more probable failure states, or

b. Reduce the probability of encountering the more probable failure states through equipment redesign, redundancy, etc.

Regardless of the probability of encountering any given Air Vehicle Failure States (with the exception of Special Failure States) the flying qualities shall not degrade below Level 3.

The discussions that follow are from AFFDL-TR-69-72.

> "Implementation of the Level concept involves both reliability analyses (to predict failure probabilities) and failure effects analyses (to insure compliance with requirements). Both types of analyses are in direct accord with, and in the spirit of, MIL-STD-756B and MIL-STD-882B. Implementation of this flying qualities requirement is, for the most part, a union of the work that was required by these related specifications with normal stability and control analysis."

> "Failure States influence the air vehicle configurations, and even the mission Flight Phases, to be considered. All failures that could have occurred previously must be examined, as well as all failures which might occur during the flight phase being analyzed. For example, failure of the wings to sweep forward during descent would require consideration of a wings-aft landing that otherwise would never be encountered. There are failures that would always result in an aborted mission, even in a war emergency period. The pertinent Flight Phases after such failures would be those required to complete the aborted (rather than the planned) mission. For example, failure of the flaps to retract after takeoff might mean a landing with flaps at the takeoff setting, with certain unexpended external stores; but supersonic cruise would be impossible. If the mission might be either continued or aborted, both contingencies need to be considered."

The failure modes and effects analysis (MIL-STD-1629) will highlight items which need to be checked by piloted simulation and flight test—although safety considerations may limit flight test. Further, compliance is demonstrated on the basis of the probability calculations and checked as accumulated flight experience permits. All the assumptions, such as independence of failure modes, etc., shall be firmly established and mutually agreed upon by the contractor and procuring activity.

Proof of compliance is, for the most part, analytical in nature as far as probabilities of failure are concerned. However, some failure rate data on the actual flight equipment may become available during the final design phases and during flight test. Any data from these or other test programs shall be used to further demonstrate compliance. Stability and control data of the usual type (e.g., predictions, wind tunnel, flight test) will also be used to demonstrate compliance.

The determinations of probability of occurrence are provided by MIL-HDBK-338, except that each specific failure shall be assumed present at the most critical point (in a flying-qualities sense) in the relevant Region of Handling.

Reliability data for electronic components is provided by MIL-HDBK-217. There seems to be no standard source for reliability data on other components.

The requirements do not account for the expectation that degradation of more than one flying quality will be more severe than any of those degradations singly. In the absence of a definitive database, simulation is recommended. Multiple-axis failures shall also be simulated, especially where the flying qualities parameters result in pilot ratings near the applicable Level's lower limits.

Degradation in atmospheric disturbances is discussed in 5.1.2. The combined effects of failures and turbulence shall be validated in a manned simulation.

There are some special requirements pertaining to failure of the engines and the flight control system. For these special requirements the pertinent failure is assumed to occur (with a probability of 1), with other failures considered at their own probabilities. For all other requirements, the actual probabilities of engine and flight control system failure are to be accounted for in the same manner as for other failures.

5.1.1.3.2 Allowable Levels for specific Air Vehicle Failure States.

The allowable flying qualities Levels for specific Failure States in Common environmental conditions shall be as shown in table IX, regardless of the probability of occurrence.

TABLE IX. Levels of flying qualities for specific failure states.

Air Vehicle Failure State	Flight Phase	Region of the Flight Envelope	Flying Qualities Level
(1)	(2)	(3)	(4)

REQUIREMENT RATIONALE

The requirements on the effects of specific types of failures, for example propulsion or flight control system, etc., shall be met on the basis that the specific type of failure has occurred, regardless of its probability of occurrence. This approach assumes that a given component, or series of components, will fail. Based on the comments made by users of MIL-F-8785B, this approach is a common current practice. This paragraph has been included to provide a way to specify the allowable degradation in handling qualities due to failures without making detailed probability calculations.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1.1.3.2 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.10.2.1
MIL-STD-1797: 4.1.7.4.

Values for columns (1) through (4) are to be completed by the contractor and the procuring activity. Because the selection of failure modes is highly dependent on the details of the design, close coordination between the contractor, the procuring activity, and the using-command will be required when identifying failure modes to be analyzed and determining the allowable degradation.

Selection of failures to be considered shall be based on preliminary estimates of handling qualities degradations. For example, the loss of one to three channels of a quad-redundant SCAS may have no effect. Conversely the failure of a single-channel, limited-authority damper would warrant a complete analysis, possibly simulation too, to determine the resulting degradation in flying qualities. Requirements such as two-fail-operate assume a certain degree of reliability and so may penalize either the contractor or the user. In addition, the procuring activity may desire consideration of certain failures regardless of their probabilities.

When writing a specific failure requirement, it is best to associate the required Levels of flying qualities with the number of failures in the system, and the task that must be performed, as indicated in the following example.

Air Vehicle Failure State	Flight Phase	Region of the Flight Envelope	Flying Qualities Level
1. After two independent failures in the stability/control augmentation system	All	ROSH	1
2. After two independent failures in the air data system	RR, CR, PA, L	ROSH	2
3. After two independent failures in the electrical power system	RR, CR, PA, L	ROSH	2
4. After loss of all propulsion power	ED, L	ROSH	2 Means shall be provided to maintain stable and controlled flight for the time required to descend from cruise altitude to SL at the speed for best L/D, with a 5-minute reserve.
5. After two independent failures in a system (such as a fuel system) which can affect c.g. position	All	ROSH	2
6. Electric power interrupts or transients	All	ROSH	2 Shall not result in excessive air vehicle transients or loss of controlled flight.
7. After one failure in the hydraulic system	All	ROSH	2

It must be emphasized that this is only an example, not the recommended values for table IX. There will be failures that are not discussed in this example, such as generic software faults. Levels of flying qualities for these failure modes shall be coordinated with each of the technical disciplines involved and documented in table IX. Although this example may be safe with a simplex hydraulics system, no combat capability remains. Mission requirements shall be considered when writing these requirements.

REQUIREMENT LESSONS LEARNED

This approach has been utilized on the F-15 (a Level 2 basic airframe was specified), F-16 (a minor flight control system change from the prototype), A-10 (a simple flight control system design), and F-5E (a design evolution) with reasonable success. As an example of this process, in one case a failure requirement for the hydraulic system was stated as: "After loss of one hydraulic system, flying qualities sufficient to return to base and land shall remain (Level 2)." A flying qualities analysis was performed to determine the control capability with a reduced number of control surfaces. It was found that if one hydraulic system provided power to an aileron on one side of the vehicle, the elevator on the opposing side, and the rudder, sufficient control could be maintained to land the vehicle. This was confirmed on a piloted simulator. The hydraulic system was plumbed accordingly and demonstrated on the Iron Bird. As the flight test program progressed, modifications were made to the flight control laws and the aerodynamic package. Furthermore, there were also some slight changes made to the actuators. At the end of the test period the flying qualities were re-evaluated on the simulator to ensure that the vehicle could be landed. This data was then used to verify specification compliance.

VERIFICATION

Verification of specific failures is required to demonstrate that flying qualities meet specified Levels when those failures occur. A comprehensive failure modes and effects analysis has been found essential for all but very simple designs.

Initially, failure analysis will consist of determining that the flying qualities parameters in question fall within the prescribed boundaries for the specified Levels. Later, ground-based and/or in-flight simulation shall be used to demonstrate that the flying qualities meet the specific failure requirements. When evaluating the effects of failures, the failures shall be assumed to occur in the most critical flight condition; for example a yaw damper failure at the maximum service ceiling. Failures in more than one axis that cause the specified flying qualities parameters to fall near the boundaries of more than one flying qualities requirement shall also be simulated. Finally, the combined effects of failures and turbulence shall be investigated using a piloted simulation.

For each combination of Failure State and Flight Phase listed in 5.1.1.3.2, the resulting Level of flying qualities shall be evaluated by simulation or flight test during the Failure Modes and Effects Criticality Analysis (FMECA) and Failure Modes and Effects Test (FMET). Using the same tasks as those of 5.1.1.1 and 5.1.1.2, failures shall be introduced during the tasks and the resulting flying qualities evaluated by pilot comments and C-H ratings. The comments and ratings for each combination shall indicate that the flying qualities are no worse than the required Level of flying qualities for that Failure State and Flight Phase.

5.1.1.3.3 Failures outside the ROTH.
The air vehicle shall be capable of returning to the ROSH and ROTH after the following failures outside of the ROTH but within the RORH: ___(1)___.

REQUIREMENT RATIONALE

Air combat may require short excursions to AOAs beyond stall, or the normal limit AOA, frequently enough to warrant special consideration. Other tasks and ground operation may be pertinent. If the air vehicle experiences certain failures while operating within the RORH, it will be necessary to safely recover the vehicle to the ROSH and ROTH.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1.1.3.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.1.10.3.3
MIL-STD-1797: 4.1.7.6.

During development of the specification, if the program requires that the air vehicle return to the ROTH or ROSH without loss of control, this requirement shall be modified to so state. At this stage the only guidance possible is to raise the issue and to list some factors in avoiding loss of control:

Engine flame-out: duty cycle or throttle usage, compressor stall
Reaction controls
Fail-operate or fail-soft
Frequency of failure
Failure-warning reliability

Applicable Failure States shall be listed in blank (1).

Other post-stall requirements in this standard rely on the basic airframe, the flight-control system requirements, the presumed small chance of failure in the post-stall region, and avoiding the region if certain Air Vehicle Failure States exist. New emphasis on post-stall technology now gives added importance to this flight region for certain combat air vehicles.

REQUIREMENT LESSONS LEARNED

As for normal operation, stability augmentation failure modes can have completely different effects in the post-stall region from their action in the ROSH and ROTH.

VERIFICATION

Flight safety will be a prime factor in determining the means of verification. Outside the ROTH, wind tunnel tests, structural analyses, propulsion limits, etc., will need to be taken into account.

Verification shall be by simulation or flight test to show that the air vehicle can be flown to particular flight conditions outside the ROTH and return safely under the failure conditions listed in 5.1.1.3.3. The particular flight conditions to be evaluated shall be any conditions that are predicted by analysis or simulation to be worse than Level 3 under the relevant failure conditions. For conditions which are considered too dangerous to test in flight, verification can be shown in a manned simulation. Proof of compliance in these demonstrations shall consist of pilot comments. The pilot comments shall indicate that the air vehicle can be safely returned to the ROSH and ROTH.

5.1.2 Flying qualities in degraded environmental conditions.
The minimum required flying qualities for all Air Vehicle States (Normal, Extreme, and Failure States) when operated in degraded environmental conditions shall meet the Levels and qualitative degrees of suitability specified in table X.

TABLE X. Flying qualities in degraded environmental conditions.

Intensity of environmental conditions	Air Vehicle States which are required to be at least Level 1 in Calm air must meet:	Air Vehicle States which are required to be at least Level 2 in Calm air must meet:	Air Vehicle States which are required to be at least Level 3 in Calm air must meet:
Common			
Uncommon			
Extraordinary			

REQUIREMENT RATIONALE

The flying qualities requirements must incorporate the universal recognition that pilot workload or performance or both generally degrade as the intensity of atmospheric disturbances increases. This requirement provides a rational means for specifying the allowable degradation in handling qualities in the presence of increased intensities of atmospheric disturbances. It is especially important to stress applicability in atmospheric disturbances because most flight testing is done in Calm air. There is considerable evidence that atmospheric disturbances can expose handling qualities cliffs which are not apparent in Calm air (for example, see FAA-RD-75-123).

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1.2 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.8.1 and 3.8.3 through 3.8.3.2
MIL-STD-1797: 4.9 and 4.9.1.

Recommended values for completion of the blanks in table X:

Intensity of environmental conditions	Air Vehicle States which are required to be at least Level 1 in Calm air must meet:	Air Vehicle States which are required to be at least Level 2 in Calm air must meet:	Air Vehicle States which are required to be at least Level 3 in Calm air must meet:
Common	Level 1 quantitative requirements and Satisfactory qualitative requirements	Level 2 quantitative requirements and Tolerable or better qualitative requirements	Level 3 quantitative requirements and Controllable or better qualitative requirements
Uncommon	Level 1 quantitative requirements and Tolerable or better qualitative requirements	Level 2 quantitative requirements and Controllable or better qualitative requirements	Level 3 quantitative requirements and Recoverable or better qualitative requirements
Extraordinary	Controllable or better qualitative requirements	Recoverable or better qualitative requirements	

In table X, the qualitative requirements for each Air Vehicle State are a function of atmospheric disturbance intensity and the Level of flying qualities required for that State in Calm air. The Level of flying qualities required in Calm air is, in turn, determined from 5.1.1.1, 5.1.1.2, and 5.1.1.3. These qualitative requirements are a rough fit to the C-H scale. However, it is evident by direct comparison of adjacent blocks in the table that the quantitative requirements are not uniquely related to the qualitative descriptions or to the C-H scale. Indeed, for these environmental variations they cannot be.

Quantitative Level boundaries are applied in Calm air, Common, and Uncommon disturbances. While most of the data base includes consideration of these environmental disturbances, no basis in data or reason could be found for applying the quantitative requirements generally in Extraordinary conditions.

In some cases the expected motions due to turbulence are sufficiently extreme that pilot ratings are not appropriate. In these cases statements related to recoverability are used in table X.

REQUIREMENT LESSONS LEARNED

Experience has shown that aircraft with severe handling qualities deficiencies can receive Level 1 pilot ratings in Calm air. Addition of atmospheric disturbances to the problem forces the pilot to use aggressive control activity that tends to expose handling qualities deficiencies.

It is natural for pilot rating of flying qualities to degrade with increasing atmospheric disturbances. Since this standard is used to procure air vehicles, not pilots, a distinction between degradation of pilot rating and degradation of air vehicle characteristics must be made. This distinction is made in this requirement for all Air Vehicle States. These allowances, of course, should not be construed as a recommendation to degrade flying qualities with increasing intensities of atmospheric disturbances.

The concept of accounting for degradations in flying qualities in a flying qualities standard was introduced in AFFDL-TR-78-171 and discussed at some length in the Flying Qualities Workshops held in 1978 and 1979 at the USAF Flight Dynamics Laboratory (documented in AFFDL-TR-78-171 and AFWAL-TR-80-3067). A primary objection has been that it is

impractical to measure turbulence in flight test. A successful example of such measurements is given by NASA TN D-7703.

Accounting for the observed effects of atmospheric disturbances in a generally acceptable manner has been quite a problem. Although degraded pilot rating in Uncommon disturbances is expected and accepted, degraded air vehicle characteristics which result from saturation of stability augmentation or other nonlinearities are not acceptable. Acceptance of those characteristics would likely cause a further degradation in pilot rating. Therefore, a distinction between Levels and C-H ratings must somehow be made: their relationship must vary with the intensity of atmospheric disturbances.

Any description of disturbance intensity in terms of aircraft reaction misses the point: the reaction varies according to aircraft characteristics, and it is just that which must be specified. Defining flying qualities in terms of response intensity is a tautology, meaningless as a requirement. That leaves the question of how to determine the turbulence level in flight. Direct measurement should often be possible using the flight test instrumentation and parameter identification. Lacking that, one can estimate turbulence by using a chase plane with well-known gust response characteristics.

For the purposes of these requirements, an engineering model of atmospheric disturbances is required. This engineering model may be the simplest or minimum acceptable model that correctly identifies the primary parameters of particular interest. This is in contrast to the objectives of basic research into meteorological phenomena or the physics of atmospheric dynamics. Using an available more elaborate model, of course, is also acceptable and may be necessary to assess the effects of structural modes or saturation of the flight control system.

The philosophy in selecting this model is based on two fundamental precepts:

1) Keep the model form as simple as possible; and

2) Use parameters that have direct relationships to aircraft dynamics or flying qualities.

This requires a rational approach to the tradeoff between engineering convenience and physical correctness in disturbance models. The evaluation of the effects of atmospheric disturbances on aircraft flying qualities has been approached in diverse ways. The large volume of literature is evidence of this. It is far too easy to become bogged down in the ill-defined tradeoffs between Dryden and von Karman disturbance forms, the need for non-Gaussian or non-stationary characteristics, the debate over how and when to model wind shear effects, or whether shorter disturbance scale lengths are more realistic than longer ones. Aircraft designers and simulator researchers continually face such questions, and while they may find answers suitable for one situation, the same questions can and do reappear on subsequent occasions.

The approach taken herein is to define basic utility models that can be applied to most handling quality evaluations. For some applications a specialized model may be desired. For example, if a high-fidelity model is required to reproduce the effects of structural modes (generally at frequencies higher than those for piloted control), the von Karman model would be appropriate.

In general, variations in properties can be viewed in terms of their engineering convenience versus their physical correctness. For example, the well-known von Karman turbulence form yields more correct spectral characteristics, but it is not so easily realized computationally as the more appropriate Dryden form, and does not lend itself quite so well to statistical analysis. Nevertheless, where another subject which requires more rigor uses the same atmospheric model as in the flying qualities simulation—structural analysis may require the von Karman form—the more rigorous form can of course be used for flying qualities, too. The same kind of tradeoff between convenience and correctness is a dominant theme in several other respects, as discussed below.

a. Determinism versus randomness

Atmospheric disturbance models can be separated according to their degree of determinism or randomness. Characteristics such as mean wind and wind shear are normally handled on a deterministic basis, or as a set of deterministic cases presented in random order. Turbulence is usually modeled as a randomly occurring phenomenon, or as a time history with known statistical properties, started at a random time. However, wind velocity or wind shear can be described in strictly probabilistic terms; and turbulence, conversely, can be described in wholly deterministic terms (as composed of summed sinusoidal gusts). In addition, random and deterministic models are often combined to suit the needs of a particular application. Deterministic features are usually quantified directly using analytical functions or tables (e.g., mean wind speed and direction or wind shear gradients with respect to time or space). Random components, on the other hand, invoke random-variable sources having their own particular statistical properties of probability distribution and correlation. Appropriate partitioning of model determinism versus randomness figures greatly in the success of any given application.

b. Probability distribution

The probability distribution of gusts describes the frequency of occurrence of different amplitudes in terms of probability density, cumulative probability distribution, or a number of central moments (mean, variance, skewness, kurtosis, etc.). While the Gaussian distribution is mathematically convenient, several turbulence models having more realistic non-Gaussian distributions have been developed in order to address the characteristics of patchiness and intermittency. Patchiness is frequently considered as corresponding to proportionately higher rate of occurrence of very large-magnitude gusts found in the Gaussian distribution, and is reflected by the higher-order even central moments (4th, 6th, etc.) (NASA CR-2451). Intermittency is the counterpart to patchiness when applied to gust velocity differences over a given time or space interval (Delft Univ. of Tech., Dept. of Aerospace Engineering Memo M-304). The usefulness of these model features depends in part upon whether the specific application can accommodate a characteristic such as patchiness on a probabilistic basis. If the scenario involves a limited time frame and a limited number of sample runs (as in a manned simulation), then a more deterministic treatment of patchiness might be required. For example, Delft Univ. of Tech., Dept. of Aerospace Engineering Memo M-304 describes a direct modulation of turbulence intensity to obtain a patch gust field. This, in turn, could permit the

return to the more convenient Gaussian distribution for random gust components without undue sacrifice in correctness.

c. Correlation

Correlation is the measure of predictability of a gust component at some future time or point in space, based on the knowledge of a current gust. Since the modeling of a random process such as turbulence consists of developing techniques for emulating the behavior of that process in time, it can be seen that correct duplication of the correlation can be important. There are at least two ways of presenting correlation information, in the time or space domain (correlation functions) or in the frequency domain (spectral density functions).

The correlation function can be converted to the frequency domain via a Fourier transform, resulting in the power spectral density function. A frequency-domain representation is often useful because it permits comparison of the aircraft's spectral features with the spectral content of the disturbance. It is thereby possible to judge the degree to which the turbulence will affect the aircraft's motion, as described in AIAA Paper 77-1145. A time- or spatial-domain correlation function can be useful when generating gusts from multiple point sources (Stanford Univ. SUDAAR No. 489).

The two most common ways of describing gust correlation are the Dryden and von Karman power spectral density forms. As mentioned earlier, the significant aspect of a choice between the two lies in the engineering convenience versus the physical correctness. However, the correctness advantage of the von Karman form is not an issue unless the significant spectral content is centered in the microscale range about one decade or more above the integral scale break frequency.

d. Dimensionality

A gust field can be described using various orders of dimensionality. The simplest is a one-dimensional-field model that involves just the three orthogonal velocity components taken at a single point (usually the aircraft center of gravity). The Taylor hypothesis (frozen field) can be applied, however, in order to approximate gust gradients with respect to the x-axis of the aircraft without increasing dimensionality. The Taylor hypothesis (J. Atmos. Sciences, Vol 20) assumes a gust field frozen in space such that time and space dependencies along the relative wind are directly related by the airspeed. A two-dimensional field model used to define a gust field in the aircraft x-y plane can be modified for the size of the aircraft relative to gust scales. (A large aircraft relative to the gust scale attenuates gust gradient spectral power at high frequencies.) A two-dimensional field can lead to greatly increased mathematical complexity over that of a one-dimensional field (AGARD Rpt 372) but some turbulence models simply define one-dimensional uniform velocity components and then add two-dimensional forms for gust gradients that contain aircraft size effects (Stanford Univ SUDAAR No. 489). A third dimension can be introduced in the form of an altitude-dependent wind shear (e.g., FAA-RD-79-84 and NASA CR-152064).

e. Stationarity

A random gust is stationary if, for a collection of gust samples, the corresponding probability and correlation properties describe any additional gust sample that may be taken. Thus stationary implies an atmospheric disturbance having an invariant mean, variance, and correlation length (or time) along the flight path. There is no restriction on whether the probability distribution is Gaussian or not.

An alternative means of introducing patchiness or intermittency is to create a non-stationary turbulence field through direct modulation of intensity (AIAA Paper 80-0703). Thus the basic noise source can remain Gaussian.

f. A survey of existing models

The objective of surveying atmospheric disturbance models is to examine how various models make the tradeoff between convenience and correctness and to search for strengths or deficiencies that could be important to a flying qualities investigation. Table XI lists some of the models that offer some potential in flying qualities applications. A few summary remarks and a list of basic references are given for each table entry.

TABLE XI. A survey of atmospheric disturbance models.

Model	Key Features	Reference Documents
Dryden turbulence	A convenient spectral form based on an exponential autocorrelation function for the axial component	"A Review of the Statistical Theory of Turbulence"
von Karman turbulence	A spectral form for which the auto-correlation function includes a finite microscale, thus the relative proportion of spectral power at high frequencies exceeds that of the Dryden	"Atmospheric Turbulence," "Progress in the Statistical Theory of Turbulence," and NASA TR-R-199
Ornstein-Uhlenbeck turbulence	A spectral form with first-order longitudinal and transverse components	AIAA Paper 80-0703
Etkin one-dimensional turbulence power spectra	The local turbulent velocity field is approximated by a truncated Taylor series which yields uniform and gradient components. High-frequency spectral components eliminated on the basis of aircraft size. Based on Dryden form but gradient spectra are non-realizable unless simplified.	"Dynamics of Flight," AGARD Rpt 372, and "A Theory of the Response of Airplanes to Random Atmospheric Turbulence"
Versine gust	A discrete gust waveform	MIL-F-8785C
Lappe low-altitude turbulence model	Experimentally-obtained data of vertical gust spectra, mean wind speed, and lapse rate were used to develop a low-level turbulence model. The turbulence spectra are presented for different types of terrain, height, and meteorological conditions.	"Low-Altitude Turbulence Model for Estimating Gust Loads on Aircraft"

TABLE XI. A survey of atmospheric disturbance models - Continued.

Model	Key Features	Reference Documents
Multiple-point-source turbulence	A two-dimensional gust field generated from two or more noise sources having prescribed correlation functions and located spanwise or lengthwise on the vehicle	Stanford Univ SUDAAR No. 489, AIAA Paper 80-1836, and AFFDL-TR-68-85
Holley-Bryson random turbulence shaping filters	A matrix differential equation formulation of uniform and gradient components including aircraft size effects. Filter equation coefficients determined from least square fit to multi-point-source-derived correlation functions.	Stanford Univ SUDAAR No. 489
University of Washington non-Gaussian atmospheric turbulence model	Non-Gaussian model using modified Bessel functions to simulate the patchy characteristic of real-world turbulence. Spectral properties are Dryden and include gust gradients.	NASA CR-2451 and AFFDL-TR-69-67
Delft University of Technology non-Gaussian structure of the simulated turbulent environment	Non-Gaussian model similar in form to the University of Washington model but uses the Hilbert transform to model intermittency as well as patchiness. Include University of Washington model features extended to approximate transverse turbulence velocities and gradients.	Delft Univ of Tech Memo M-304
Royal Aeronautical Establishment model of non-Gaussian turbulence	Non-Gaussian turbulence model with a variable probability distribution function and a novel digital filtering technique to simulate intermittency. Spectral form approximately von Karman.	Royal Aircraft Establishment, Tech Memo FS46; NASA CR-152194; and “Modeling of Gusts and Wind Shear for Aircraft Assessment and Certification”
The Netherlands National Aerospace Laboratory model of non-Gaussian turbulence	Similar to the Royal Aeronautical Establishment model but extended to include patchiness and gust gradient components and transverse velocities	NLR Memo VS-77-024 and NLR Memo VS-77-025
University of Virginia turbulence model	Models patchiness by randomizing gust variance and integral scale length of basic Dryden turbulence	“Investigation of the Influence of Simulated Turbulence on Handling Qualities"
Indian Institute of Science nonstationary turbulence model	Nonstationary turbulence is obtained over finite time-windows by modulating a Gaussian process with either a deterministic or random process. The result is patchy-like turbulence similar to the University of Washington model—except the time-varying statistics of the turbulence are presented for the deterministic modulating functions.	AIAA Paper 80-0703

TABLE XI. A survey of atmospheric disturbance models - Continued.

Model	Key Features	Reference Documents
FAA wind shear models	Three-dimensional wind profiles for several weather system types including fronts, thunderstorms, and boundary layer. The profiles are available in table form.	FAA-RD-79-84 and FAA-RD-77-36
STI wind shear model	Time and space domain models of mean wind and wind shear—ramp wave forms—are combined with MIL-F-8785C Dryden turbulence to obtain the total atmospheric disturbance. The magnitudes of the mean wind and wind shear are evaluated in terms of the aircraft's acceleration capabilities.	NASA CR-152064 AGARD CP-249
Sinclair frontal-surface wind shear model	A generic model of frontal-surface wind shear derived from a reduced-order form of Navier-Stokes equations. Relatively simple to use and can match the overall characteristics of measured wind shears.	FAA-RD-79-59 and AGARD-CP-249
MIL-F-8785B atmospheric disturbance model	Intensities and scale lengths are functions of altitude and use either Dryden or von Karman spectral forms or a versine discrete gust. Also, spectral descriptions of rotary gusts.	AFFDL-TR-69-72
MIL-F-8785C atmospheric disturbance model	Same as MIL-F-8785B with the addition of a logarithmic planetary boundary layer wind, a vector shear, and a Nave carrier airwake model	MIL-F-8785C
ESDU atmospheric turbulence	Rather general but contains comprehensive descriptive data for turbulence intensity, spectra, and probability density	ESDU Item No. 74031 and ESDU Item No. 75001
Boeing atmospheric disturbance model	A comprehensive model of atmospheric disturbances that includes mean wind, wind shear, and random turbulence. Turbulence is Gaussian and uses filters that closely approximate the von Karman spectral form. Mean wind and turbulence intensity are functions of meteorological parameters.	NASA CP-2028/FAA-RD-77-173 and FAA-RD-74-206
Wasicko carrier airwake model	Includes mean wind profile, effect of ship motion, and turbulence	Systems Tech Inc Rpt No. 137-2
Nave ship airwake model	Includes free-air turbulence filters plus steady, periodic, and random components of airwake which are functions of time and space	NADC-78182-60
Vought airwake model for DD-963 class ships	Combined random and deterministic wind components for free-air and ship airwake regions. Based on wind-tunnel flow measurements.	Vought Corp Rpt No. 2-55800/8R-3500
STI wake vortex encounter model	A two-dimensional model of the flow field due to the wake vortex of an aircraft is presented. The parameters of the flow-field model are weight, size, and speed of the vortex-generating aircraft, and distance and orientation of the vortex-encountering aircraft. Strip theory is used to model the aerodynamics of the vortex-encountering aircraft.	"A Theory of the Response of Airplanes to Random Atmospheric Turbulence"

VERIFICATION

Verification shall be by simulation or flight test in the performance of the tasks of 5.1.1.1 through 5.1.1.3.3 under various levels of atmospheric disturbances. These tasks shall be flown by test pilots at specific flight conditions throughout the ROSH and ROTH. The specific flight conditions to be evaluated shall be the most common operating conditions, any operating conditions critical to the mission of the air vehicle, and any conditions determined by analysis or simulation to be worse than the Level of flying qualities required by table X. For conditions which are considered too dangerous to test in flight, verification can be shown in a manned simulation. Proof of compliance in these demonstration tasks shall consist of pilot comments and C-H ratings. The comments and ratings shall indicate that the flying qualities are no worse than the required Level of flying qualities for each combination of Air Vehicle State, Flight Phase, and environmental condition.

For the purposes of showing compliance with this requirement, atmospheric disturbance models are defined separately for low altitudes and medium/high altitudes. In addition to these two models (described below) there is also a carrier landing disturbance model described in 5.2.6. When compliance is to be shown using piloted simulation, an atmospheric disturbance model appropriate to the pilot task shall be utilized.

Low-altitude model

The low-altitude atmospheric disturbance model specifies the atmospheric disturbances to be used for all Category C and other low-altitude operations. The low-altitude atmospheric disturbance model defines the atmospheric disturbances to be used to show compliance with the requirements at altitudes below 2000 ft AGL. It consists of four parts: a steady wind, random turbulence, discrete gusts, and a wind shear.

a. Steady wind

The steady wind is the mean wind speed. In the absence of a wind shear, the mean wind speed and direction are constant. Different orientations of the mean wind relative to the runway for Category C, or relative to the aircraft flight path for other Flight Phases, shall be considered. The wind speed measured at 20 ft above the ground, u_{20}, is shown on figure 10 as a function of probability of occurrence. The values to be used for the three atmospheric disturbance intensities are indicated.

For those requirements specifically intended for ship-based operations (5.2.6 and 5.2.7, when applicable), the mean wind speeds for the three atmospheric disturbance intensities must include the ship's speed and are therefore significantly different from those shown on figure 10. See figure 203 in 5.2.6 Verification.

b. Random turbulence

Random turbulence is composed of three components: u_g, v_g, and w_g. These three components of random turbulence have Gaussian (normal) distributions. The spectra for the turbulence velocities can be either the von Karman form or the Dryden form. When showing compliance

analytically, the von Karman form shall be used for the continuous turbulence model so that the flying qualities analyses will be consistent with the comparable structural analyses. When showing compliance via real-time piloted simulation, use of the Dryden form is recommended.

The von Karman form of the spectra for random turbulence velocities is:

$$\Phi_{u_g}(\Omega) = \sigma_u^2 \frac{2L_u}{\pi} \frac{1}{\left[1+\left(1.339L_u\Omega\right)^2\right]^{5/6}}$$

$$\Phi_{v_g}(\Omega) = \sigma_v^2 \frac{2L_v}{\pi} \frac{1+\frac{8}{3}\left(2.678L_v\Omega\right)^2}{\left[1+\left(2.678L_v\Omega\right)^2\right]^{11/6}}$$

$$\Phi_{w_g}(\Omega) = \sigma_w^2 \frac{2L_w}{\pi} \frac{1+\frac{8}{3}\left(2.678L_w\Omega\right)^2}{\left[1+\left(2.678L_w\Omega\right)^2\right]^{11/6}}$$

The Dryden form of the spectra for random turbulence velocities is:

$$\Phi_{u_g}(\Omega) = \sigma_u^2 \frac{2L_u}{\pi} \frac{1}{1+\left(L_u\Omega\right)^2}$$

$$\Phi_{v_g}(\Omega) = \sigma_v^2 \frac{2L_v}{\pi} \frac{1+12\left(L_v\Omega\right)^2}{\left[1+4\left(L_v\Omega\right)^2\right]^2}$$

$$\Phi_{w_g}(\Omega) = \sigma_w^2 \frac{2L_w}{\pi} \frac{1+12\left(L_w\Omega\right)^2}{\left[1+4\left(L_w\Omega\right)^2\right]^2}$$

At low altitudes the recommended rms intensity of vertical turbulence, σ_w, is related to the mean wind as commonly measured 20 ft above ground level, u_{20}:

$$\sigma_w = 0.1\ u_{20}$$

where u_{20} is given on figure 10. The recommended horizontal components, σ_u and σ_v, are as defined on figure 11 as functions of σ_w and altitude. The appropriate scale lengths for the low-altitude model are shown on figure 12 as functions of altitude.

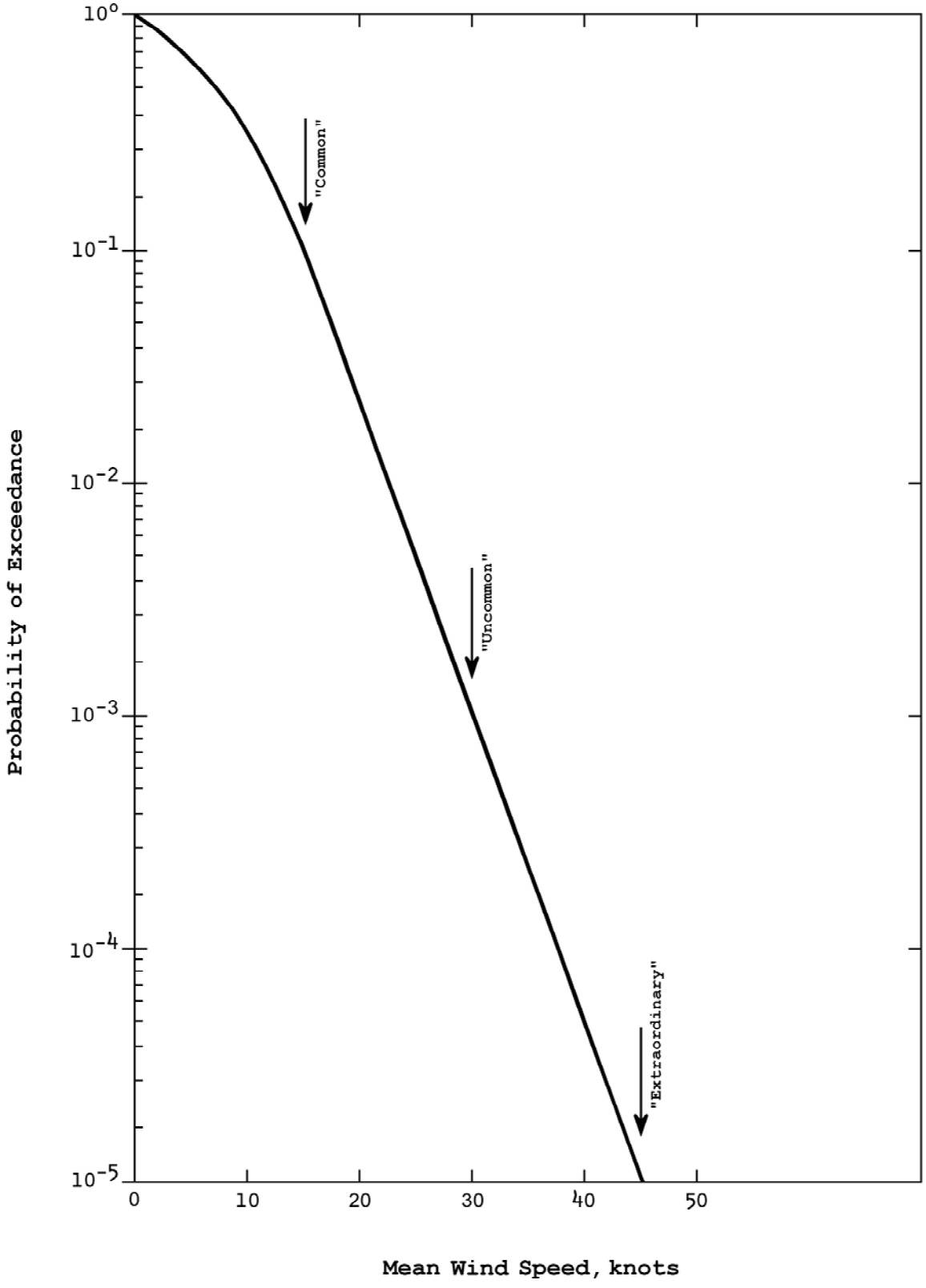


FIGURE 10. Probability of exceeding mean wind speed at 20 ft.

A random noise source (white noise) is required to mechanize the random turbulence model. This source shall be checked to insure adequate power at low frequencies. Experience with some simulations has shown that the noise source was deficient at low frequency. It is the low frequency component of the atmospheric disturbance that is important for handling qualities investigations. Finally, it seems nearly impossible to program a model without an error involving a factor of 2 or π. The lesson here is to measure the statistics of the output of the disturbance model before starting piloted evaluations.

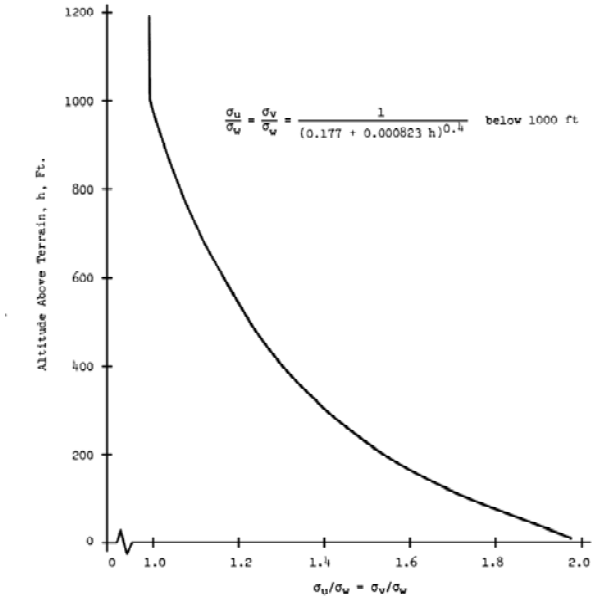


FIGURE 11. Horizontal turbulence intensities.

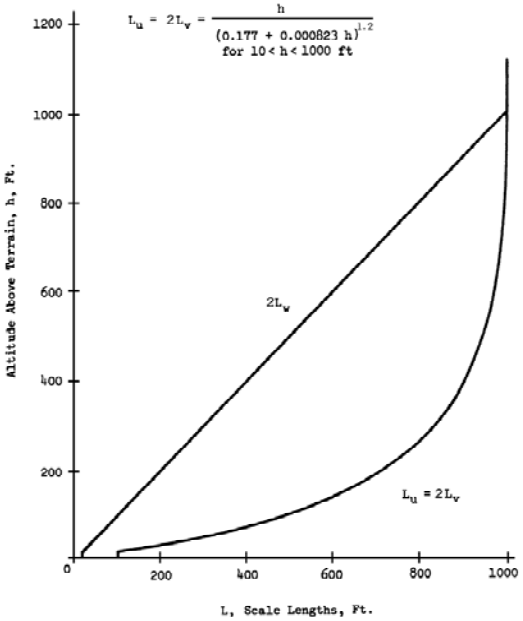


FIGURE 12. Low-altitude turbulence integral scales.

As piloted simulation has provided more experience in handling-qualities evaluations, a key factor has evolved: the separation of good and bad handling qualities. In fact, the complexity required to emulate real-world atmospheric disturbances often clouds the issue. For example, such items as boundary layer effects, patchiness, correlation of turbulence with steady wind velocity and terrain, and detailed wind shear characteristics associated with frontal activity can consume an inordinate amount of effort. To alleviate this problem, the simplified Dryden disturbance model is recommended whenever other considerations permit. This model retains all the essential features found to be useful in many piloted simulator handling quality investigations using complex disturbance models. The Dryden form has been chosen because it is simple to mechanize, as opposed to the von Karman form that must be approximated to become realizable.

Table XII gives a summary of the recommended digital filter implementation of the Dryden turbulence components. The Dryden turbulence power spectra given are the MIL-F-8785C spectra corrected for a factor of two (AFFDL-TR-72-41). The digital filter implementation of Table XII is for the Dryden form of this specification.

The application of atmospheric disturbance models can involve a number of practical implementation problems—many associated with digital computer programming. One role of this standard is to assist in answering some of the common implementation questions and to point out pitfalls frequently encountered. Some examples include:

Digital implementation of continuous spectral forms

Correct scaling of random noise sources

Evaluation of need for gradient components

Implementation of gust gradients, gust time derivatives, and gust transport lags.

Table XIII illustrates some of the practical implementation issues.

c. Discrete gusts

It is a recognized fact that the large-amplitude, low-frequency component of the disturbance model plays a dominant role in separating good and bad handling qualities. The problem with a random disturbance model is that the large low-frequency gusts (wind shears) occur at the critical point in only a few runs, resulting in discrepancies in pilot ratings and comments. In the FAA-RD-75-123 experiment one evaluation pilot received several large shears just prior to touchdown and rated the configuration a 7. Two other pilots only saw the large inputs several miles from touchdown and rated the same configuration between 3 and $4^{1}/_{2}$. The discrete gust and wind shear models are included to resolve this problem. Addition of discrete gusts and scalar and vector shears forces the pilot to act rather than waiting out the zero-mean random turbulence.

TABLE XII. Digital filter implementation of Dryden turbulence components.

Component	Intensity	Break Frequency	Digital Filter Finite Difference Equation
u_g	σ_u	$a_u = \frac{V}{L_u}$	$u_g = (1 - a_u T)\, u_g + \sqrt{2\, a_u\, T}\ \frac{\sigma_u}{\sigma_\eta}\ \eta_1$
v_g	σ_v	$a_v = \frac{V}{L_v}$	$v_g = (1 - a_v T)\, v_g + \sqrt{2\, a_v\, T}\ \frac{\sigma_v}{\sigma_\eta}\ \eta_2$
w_g	σ_w	$a_w = \frac{V}{L_w}$	$w_g = (1 - a_w T)\, w_g + \sqrt{2\, a_w\, T}\ \frac{\sigma_w}{\sigma_\eta}\ \eta_3$
p_g *	$\sigma_p = \frac{1.9}{\sqrt{2\, L_w\, b}}\, \sigma_w$	$a_p = \frac{2.6\, V}{\sqrt{2\, L_w\, b}}$	$p_g = (1 - a_p T)\, p_g + \sqrt{2\, a_p\, T}\ \frac{\sigma_p}{\sigma_\eta}\ \eta_4$
q_g **	$\sigma_q \approx \sqrt{\frac{\pi}{4 L_w\, b}}\, \sigma_w$	$a_q = \frac{\pi\, V}{4\, b}$	$q_g = (1 - a_q T)\, q_g + \frac{\pi}{4\, b}(w_g - w_{g_{PAST}})$
r_g ***	$\sigma_r \approx \sqrt{\frac{\pi}{3 L_v\, b}}\, \sigma_v$	$a_r = \frac{\pi\, V}{3\, b}$	$r_g = (1 - a_r T)\, r_g + \frac{\pi}{3\, b}(v_g - v_{g_{PAST}})$

* p_g is significant only if $\sqrt{\frac{b}{2\, L_w}}\, C_{\ell_p} > C_{\ell_\beta}$

** q_g is significant only if $\sqrt{\frac{\pi\, b}{16\, L_w}}\, C_{m_q} > C_{m_\alpha}$

*** r_g is significant only if $\sqrt{\frac{\pi\, b}{12\, L_v}}\, C_{n_r} > C_{n_\beta}$

TABLE XIII. Examples of practical implementation issues.

Implementation Item	Standard Method	Comments
Digital implementation of continuous filter forms. Example: first-order Dryden form (applicable u_g or p_g)	Spectral form: $\Phi_{u_g} = \sigma_u^2 \frac{2 L_u}{\pi} \frac{1}{1 + (L_u \Omega)^2}$ Discrete realization: $u_g = C_1 u_g + C_2 \eta$ where $C_1 = \begin{cases} \text{either } \exp(-a T) & (z - \text{transform}) \\ \text{or } 1 - a T & (\text{Euler integration}) \\ \text{or } \frac{2 - a T}{2 + a T} & (\text{Tustin transform}) \end{cases}$ $C_2 = \sqrt{1 - C_1^2} \left(\frac{\sigma_u}{\sigma_\eta} \right)$ where $a = V/L_u$ and η is a normally distributed random number with variance σ_η^2.	This matter can be particularly confusing because spectral forms are written in a number of ways—one-sided or two-sided, in terms of spatial or temporal frequency, or in terms of angular or cyclical frequency. Furthermore white noise in the continuous domain must be converted to random numbers in the discrete domain.
Determination of p_g importance	Criterion: p_g is significant relative to v_g if: $\sqrt{\frac{b}{L_w}} \left\lvert C_{\ell_p} \right\rvert > \left\lvert C_{\ell_\beta} \right\rvert$ or $\frac{2}{\sqrt{L_w b}} \left\lvert L_p \right\rvert > \left\lvert L_v \right\rvert$ where b is span and L_w is w-gust scale length	p_g can be an important disturbance component in the roll axis, especially if effective dihedral is small.

TABLE XIII. Examples of practical implementation issues - Continued.

Implementation Item	Standard Method	Comments
Determination of p_g intensity	Holley-Bryson model: $\sigma_p = \frac{2.15\,\sigma_w}{\sqrt{b\,L_w\,(1 + b/L_w)}}$ MIL-F-8785C: $\sigma_p = \frac{0.95}{\sqrt[3]{b^2\,L_w}}$ An approximate intensity averaged over several models: $\sigma_p \doteq \frac{1.9}{\sqrt{b\,L_w}}\,\sigma_w$	If the p_g component is considered important, one must determine the intensity in order to implement the filter. A specific, easy-to-compute value for intensity is seldom readily available. Also, the various p_g model forms all have different ways of expressing model parameters.
Realization of von Karman-like spectra	Boeing higher-order linear filter forms: $u_g: \quad \frac{\left(s + 4\frac{V}{L}\right)}{\left(s + 0.84\frac{V}{L}\right)\left(s + 6\frac{V}{L}\right)}$ $v_g, w_g: \quad \frac{\left(s + 0.38\frac{V}{L}\right)\left(s + 7.7\frac{V}{L}\right)}{\left(s + 0.48\frac{V}{L}\right)\left(s + 1.22\frac{V}{L}\right)\left(s + 11.1\frac{V}{L}\right)}$ STI reduced-order variation based on Boeing forms: $v_g, w_g: \quad \frac{\left(s + 8\frac{V}{L}\right)}{\left(s + 1.7\frac{V}{L}\right)\left(s + 12\frac{V}{L}\right)}$	An approximation to the increased correlation in the microscale range of the von Karman spectral form can be realized with second- and third-order linear filters. Digital implementation would involve finite difference equations of corresponding order. Correct spectral content above 100 (V/L) rad/sec would require matching von Karman spectra with even higher order filters.

The discrete gust model may be used for any of the three gust velocity components. Discrete gusts have the "1-cosine" shape given by:

$$v = 0 \qquad \text{for } x < 0$$

$$v = v_m [1 - \cos(\pi x/d_m)]/2 \qquad \text{for } 0 \le x \le d_m$$

$$v = v_m \qquad \text{for } x > d_m$$

where v is the gust velocity (u_g, v_g, or w_g), x is distance traveled measured from the beginning of the gust, v_m is the gust magnitude, and d_m is the gust length measured from the beginning of the gust. The resulting gust shape is shown on figure 10.

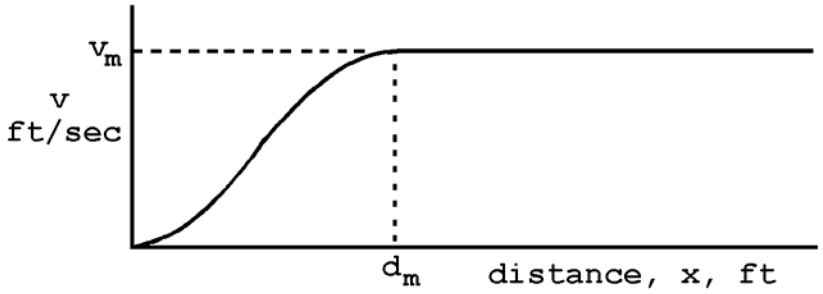


FIGURE 13. "1-cosine" shape for a discrete gust.

Discrete gusts may be used singly or in combinations. For example, the discrete gust of figure 13 might be coupled with an equal but opposite gust beginning at d_m. The two halves of a double gust do not have to be the same length or magnitude. Step function or linear ramp gusts may also be used. Several values of d_m shall be used, each chosen so that the gust is tuned to each of the natural frequencies of the air vehicle and its flight control system (higher frequency structural modes may be excepted). The gust magnitude for each value of d_m shall be determined from figure 14 using the values from figure 11 and figure 12 that are appropriate for the direction of the gust (i.e, for u_g, use σ_u and L_u). The scales and intensities used in determining the gust magnitudes for the discrete gust model shall be the same as those in the random turbulence model. For the Extraordinary intensities, modes with wavelengths less than the turbulence scale length may be excepted.

Alternatively, specific discrete gust data that have been extracted from gusts actually encountered during air vehicle flight tests are available. These may also be included in the definition of the atmospheric disturbance model for evaluation of air vehicle response. Figure 15 presents one such actual discrete gust profile.

In general, both the continuous turbulence model and the discrete gust model shall be used. Strict reliance upon a wholly random gust model for a small-sample short-term task evaluation is both impractical and improper. Investigators and evaluators desire to control disturbances well enough so that critical conditions and events can be staged, especially in the case of manned

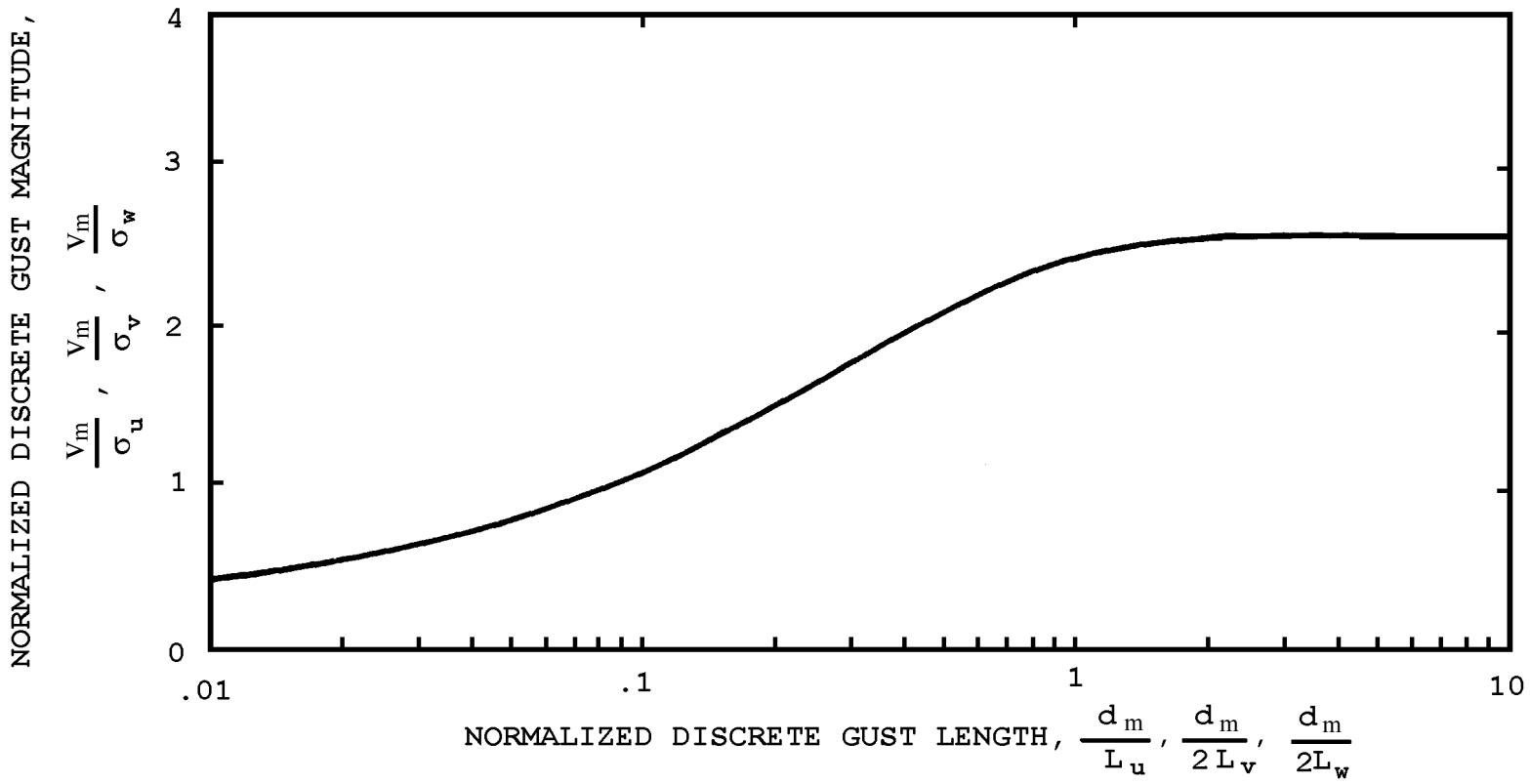


FIGURE 14. Magnitude of discrete gusts.

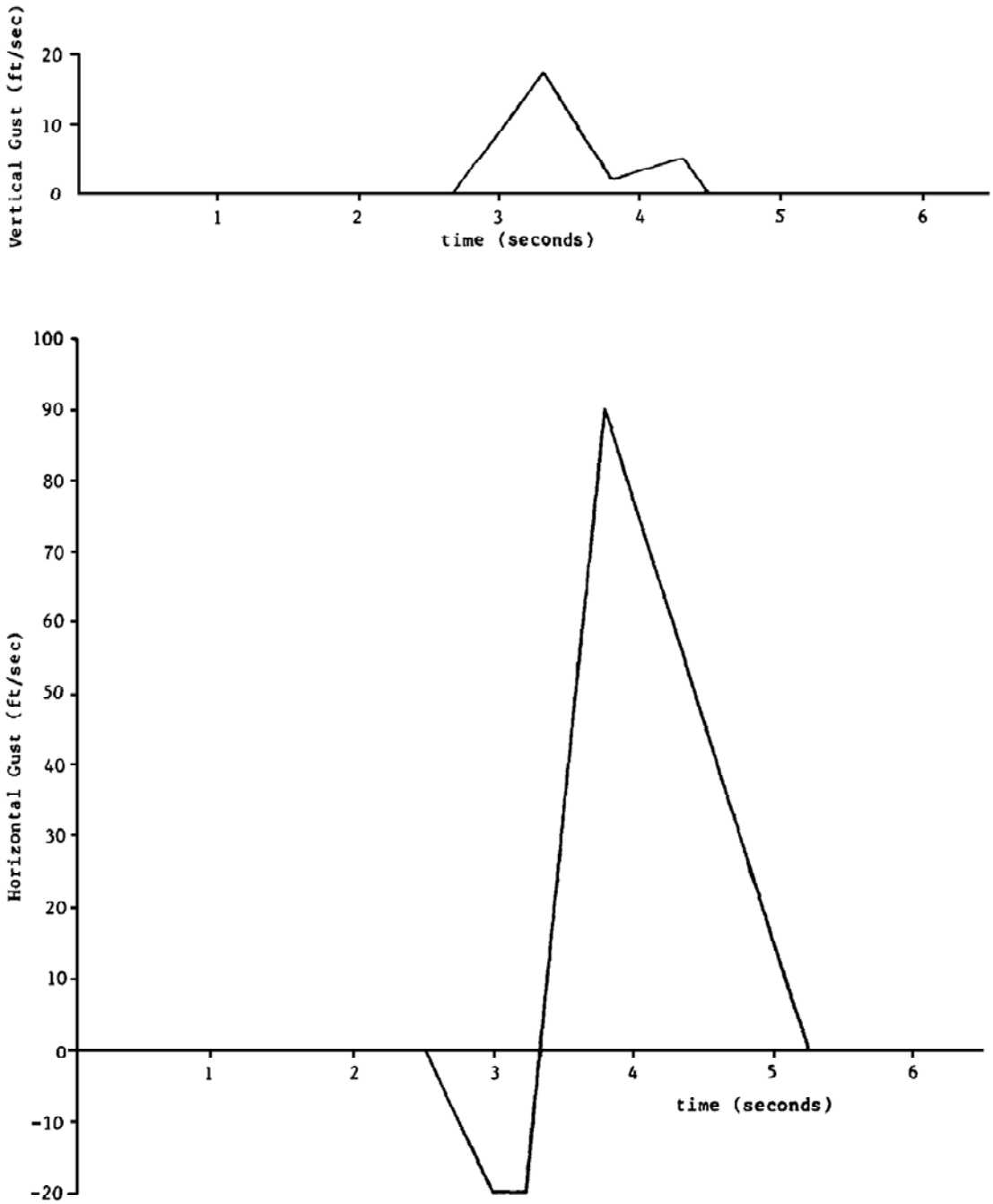


FIGURE 15. Example discrete gust profile (earth-axis winds).

simulation. This demands a fair degree of model determinism. On the other hand, pilot surprise and sensitivity to variation call for a degree of randomness. A compromise must therefore be reached. This blending deserves to be addressed in a systematic way, but sometimes solutions must be based upon experience more than clear rationale.

The discrete gust and random turbulence velocities shall be applied to the aircraft equations of motion through the aerodynamic terms only. The direct effect on the aerodynamic sensors shall be included when such sensors are part of the aircraft augmentation system.

Application of the disturbance model depends on the motion variables and the range of frequencies of concern in the analysis. When structural modes are within or close to this range, the exact distribution of turbulence velocities must be considered. For this purpose, it is acceptable to consider u_g and v_g as being one-dimensional for the evaluation of aerodynamic forces and moments. The w_g shall be considered two-dimensional, a function of both x and y.

When structural modes are not significant to the analysis or simulation, airframe rigid-body responses may be evaluated by considering uniform gust or turbulence immersion along with linear gradients of the disturbance velocities. The uniform immersion is accounted for by u_g, v_g, and w_g defined at the aircraft center of gravity. Angular velocities due to turbulence are equivalent in effect to aircraft angular velocities. Approximations for these angular velocities are defined (precise only at very low frequencies) as follow:

$$\dot{\alpha}_g = -q_g = \frac{\partial w_g}{\partial x}$$

$$p_g = \frac{\partial w_g}{\partial y}$$

$$r_g = -\frac{\partial v_g}{\partial x}$$

d. Wind shear

The use of wind shear in handling-quality evaluations is primarily to provide a disturbance that forces aggressive closed-loop pilot behavior. A discrete wind shear is defined for the low-altitude model to insure that all pilots will experience the critical input. The wind shear model has constant gradients of wind magnitude and allows for constant or changing direction. A great number of combinations of wind shear can be derived from such a model. However, only four limiting cases are recommended for verification of flying qualities: decreasing headwind, tailwind, and sidewind shears, and a constant-magnitude shear with changing azimuth.

The magnitude of the wind scalar shear is defined by the use of the following expression for the mean wind profile as a function of altitude:

$$u_w = u_{20} \frac{\ln(h/z_0)}{\ln(20/z_0)}$$

where z_0 = 0.15 ft for Category C Flight Phases
= 2.0 ft for other Flight Phases

The wind vector shear is defined by a change in direction of the mean wind speed over a given height change. The vector shears to be considered shall be as shown in table XIV.

TABLE XIV. Vector shear intensities.

Disturbance intensity	Change in mean wind heading (deg)	Height of vector shear (ft)
Common	0	
Uncommon	90	600
Extraordinary	90	300

A range of values for the initial wind orientation and the initial altitude for onset of the shear shall be considered. Relative to the runway, magnitudes of $u_{20} \sin \psi_w$ greater than the crosswind values in 5.2.3.10 or tailwind components at 20 ft greater than 10 kts need not be considered. At any altitude other than 20 ft these limits do not apply.

It is felt to be unnecessary to simulate vertical shears since the horizontal shear disturbs the aircraft both in airspeed and flight path. However, for some cases a more complex special-purpose may be required. For specialized applications, such as carrier landing burble or checking a downburst incident, a vertical shear shall be included.

Medium/high-altitude model

The medium/high-altitude atmospheric disturbance model defines the atmospheric disturbances to be used to show compliance with the requirements of this specification at altitudes above 2000 ft AGL. It consists of two parts: random turbulence and discrete gusts.

a. Random turbulence

Root-mean-square magnitudes of random turbulence for the medium/high-altitude model are shown on figure 16 as functions of altitude and probability of exceedance. These magnitudes apply to all axes. The dashed lines, labeled according to probability of encounter, are based on MIL-A-8861B and MIL-F-9490D. The solid lines define the recommended turbulence intensities and indicate simplified approximations to the data, except that a minimum rms magnitude of 3 ft/sec is specified at all altitudes in order to assure that air vehicle handling will always be evaluated in the presence of some disturbance.

The scales and intensities for the medium/high-altitude model are based on the assumption that turbulence above 2,000 ft is isotropic. Then

$$\sigma_u^2 = \sigma_v^2 = \sigma_w^2$$

and

$$L_u = 2L_v = 2L_w$$

The scales to be used for the medium/high-altitude model are

$L_u = 2L_v = 2L_w = 2{,}500$ ft using the von Karman form or

$L_u = 2L_v = 2L_w = 1{,}750$ ft using the Dryden form

b. Discrete gusts

For the discrete gusts, the Common and Uncommon gust magnitudes, u_g, v_g, and w_g, shall be determined from figure 14 using values of d_m determined according to the discussion in the low-altitude model, and the appropriate rms turbulence intensities from figure 16. Extraordinary gust magnitudes shall be as follow:

At or below 20,000 ft:

a) 66 ft/sec EAS at V_G, gust penetration limit speed
b) 50 ft/sec EAS at $V_{o_{max}}$
c) 25 ft/sec EAS at V_{max}
d) 50 ft/sec EAS at speeds up to V_{max}(PA) with landing gear and other devices which are open or extended in their max open or max extended positions

For altitudes between 20,000 ft and 50,000 ft, v_m may be reduced linearly with altitude from:

a) 66 ft/sec EAS at 20,000 ft to 38 ft/sec EAS at 50,000 ft for the V_G condition
b) 50 ft/sec EAS at 20,000 ft to 25 ft/sec EAS at 50,000 ft for the $V_{o_{max}}$ condition
c) 25 ft/sec EAS at 20,000 ft to 12.5 ft/sec EAS at 50,000 ft for the V_{max} condition

For altitudes above 50,000 ft, v_m is the appropriate value of v_m for the specified condition at 50,000 ft multiplied by the factor $\sqrt{\rho/\rho_{50}}$, the square root of the ratio of air density at altitude to the standard atmospheric density at 50,000 ft.

Summary

The suggested magnitudes for the elements of the two models of atmospheric disturbances are summarized in table XV, table XVI, and table XVII. The suggested magnitudes for Common atmospheric disturbances are shown in table XV. Suggested magnitudes for Uncommon atmospheric disturbances are shown in table XVI. Suggested magntudes for Extraordinary atmospheric disturbances are shown in table XVII.

When demonstrating compliance with these requirements, a complete environmental representation is required with the cumulative effects of wind shear, turbulence, and gusts. However, there may also be some value in analyzing the effects of wind shear, turbulence, and gusts separately. A non-Gaussian turbulence representation together with a wind model may also be used to represent the patchy, intermittent nature of actual measured turbulence.

When using simulation to verify compliance, each range of atmospheric disturbance shall be tested to its upper limits. For example, to show compliance with the requirements for Calm to Common disturbances, the wind speed at 20 ft shall be 15 kts (see figure 10).

Experience has shown that the atmospheric disturbance environment which has been labeled "Uncommon" is generally sufficient to force the pilot into the aggressive control activity expected to expose handling qualities deficiencies when they exist. Hence, it is recommended that in simulation the major effort be spent investigating the "Uncommon" disturbance level. During flight test there is no compelling reason to seek out mountain waves or thunderstorms to comply with the "Extraordinary" requirements. Of course, if the mission specifically dictates flight in extraordinary disturbances a significant portion of the time, these conditions shall be duly accounted for. In any case, some flying shall be done in turbulence as a general check of simulation.

The combined effects of failures and turbulence shall be investigated using piloted simulation. There may be aerodynamic and flight control system nonlinearities that are affected by very large disturbances. Such effects shall be investigated in manned simulation with the Extraordinary magnitudes of atmospheric disturbances defined above.

When demonstrating compliance with 5.2.3.10 and 5.2.3.11 via piloted simulation, the steady crosswind values may be invariant with time, position, or altitude. In flight test, it is only necessary that the crosswind component specified exist at altitudes high enough to require the pilot to establish a definite crosswind correction prior to touchdown.

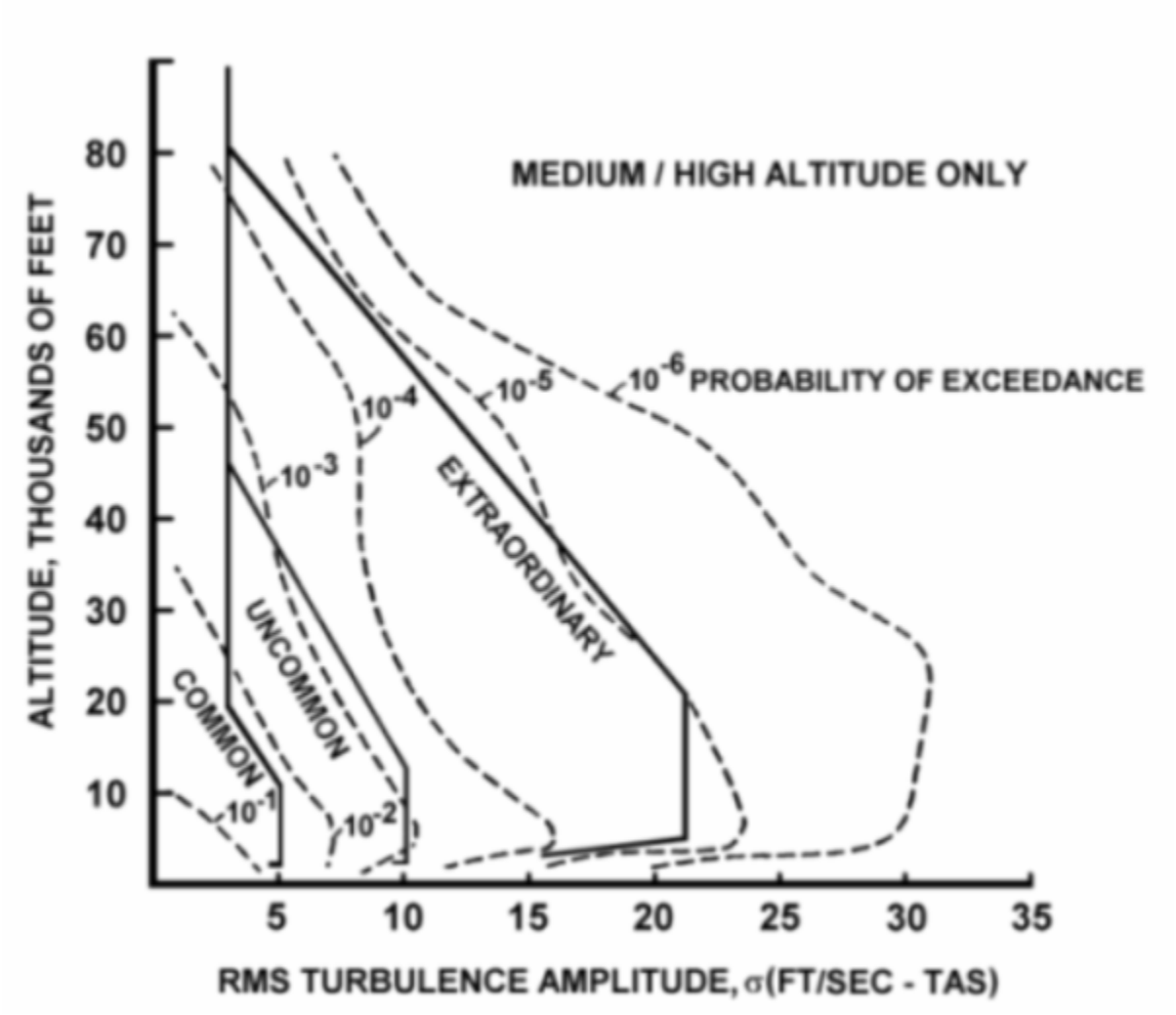


FIGURE 16. Turbulence intensities and probability of exceedance.

TABLE XV. Common atmospheric disturbances.

Model	Steady Wind Speed	Turbulence Scale Lengths and Intensities	Discrete Gust Length and Magnitude	Wind Shear
Low-altitude Model	u_{20} up to 15 kts; tailwind component no more than 10 kts at 20 ft	L_u, L_v, L_w as shown on figure 12 $\sigma_w = .1\ u_{20}$; σ_u, σ_v as given on figure 11	d_m as defined; v_m as determined from figure 14 using the appropriate values from figure 11 and figure 12	scalar shear as defined for u_{20}; no vector shear
High/Medium-Altitude Model		L_u, L_v, L_w as defined; rms turbulence intensities as shown on figure 16	d_m as defined; v_m as determined from figure 14 using appropriate rms turbulence intensities from figure 16	

TABLE XVI. Uncommon atmospheric disturbances.

Model	Steady Wind Speed	Turbulence Scale Lengths and Intensities	Discrete Gust Length and Magnitude	Wind Shear
Low-altitude Model	15 kts < $u_{20} \leq$ 30 kts; tailwind component no more than 10 kts at 20 ft; crosswind component no more than 20 kts at 20 ft for Class I air vehicles	L_u, L_v, L_w as shown on figure 12; $\sigma_w = .1\ u_{20}$; σ_u, σ_v as given on figure 11	d_m as defined; v_m as determined from figure 14 using the appropriate values from figure 11 and figure 12	scalar shear as defined for u_{20}; 90° change in mean wind heading in a height change of 600 ft
High/Medium-Altitude Model		L_u, L_v, L_w as defined; rms turbulence intensities as shown on figure 16	d_m as defined; v_m as determined from figure 14 using appropriate rms turbulence intensities from figure 16	

TABLE XVII. Extraordinary atmospheric disturbances.

Model	Steady Wind Speed	Turbulence Scale Lengths and Intensities	Discrete Gust Length and Magnitude	Wind Shear
Low-altitude Model	30 kts < u_{20} ≤ 45 kts; tailwind component no more than 10 kts at 20 ft; crosswind component no more than 20 kts at 20 ft for Class I, no more than 30 kts at 20 ft for Class II, III, or IV	L_u, L_v, L_w as shown on figure 12 σ_w = .1 u_{20}; σ_u, σ_v as given on figure 11	d_m as defined; v_m as determined from figure 14 using the appropriate values from figure 11 and figure 12	scalar shear as defined for u_{20}; 90° change in mean wind heading in a height change of 300 ft
High/Medium-Altitude Model		L_u, L_v, L_w as defined; rms turbulence intensities as shown on figure 16	d_m as defined (values of d_m less than the corresponding random turbulence scale length may be excepted); v_m as follows: At or below 20,000 ft: 66 ft/sec EAS at V_G 50 ft/sec EAS at $V_{o_{max}}$ 25 ft/sec EAS at V_{max} 50 ft/sec EAS at speeds up to V_{max}(PA) with landing gear and other devices which are open or extended in their max open or max extended positions For altitudes between 20,000 ft and 50,000 ft, v_m may be reduced linearly with altitude from: 66 to 38 ft/sec EAS at V_G 50 to 25 ft/sec EAS at $V_{o_{max}}$ 25 to 12.5 ft/sec EAS at V_{max} For altitudes above 50,000 ft, v_m is the appropriate value of v_m at 50,000 ft multiplied by the factor $\sqrt{\rho/\rho_{50}}$, the square root of the ratio of air density at altitude to the standard atmospheric density at 50,000 ft	

5.1.3 Control margin.
For all Normal and Extreme States, aerodynamic control power, control surface rate, and hinge moment capability shall provide adequate control margins for restoring moments throughout the combined range of all attainable AOAs (both positive and negative) and sideslip angles. For all Failure States within the ROTH, control margins shall be such that control can be maintained long enough to fly out of atmospheric disturbances, all Flight Phases can be terminated safely, and a waveoff (go-around) can be accomplished successfully.

REQUIREMENT RATIONALE

This overall requirement is intended to assure adequate control margins for safety in any situation not otherwise covered in this standard. It is intended to permit recovery from unusual situations in, and even beyond, the RORH—on the premise that if a condition is attainable, someday it will be attained. Experience has shown that to be a reasonable assumption. Control power, control effector rate, and hinge moment capability are essential in the establishment of adequate control margins.

REQUIREMENT GUIDANCE

This requirement corresponds to 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.10
MIL-STD-1797: 4.1.11.5.

This requirement applies to the prevention of loss of control and to recovery from any situation, including deep stall trim conditions, for all maneuvering, including pertinent effects of factors such as pilot strength, regions of control-surface-fixed instability, inertial coupling, fuel slosh, the influence of symmetric and asymmetric stores, stall/post-stall/spin characteristics (5.2.5), atmospheric disturbances (5.1.2), and Air Vehicle Failure States (5.1.1.3 through 5.1.1.3.3; failure transients and maneuvering flight appropriate to the Failure State are to be included). The contractor shall consider the degree of effectiveness and certainty of operation of limiters, c.g. control malfunction or mismanagement, and transients from failures in the propulsion, flight control, and other relevant systems.

For CTOL flight, the control margin requirements must be met with aerodynamic control power only, without the use of other effectors, such as thrust vectoring. This approach was chosen because experience to date with current technology inlets and engines operating at the distortion levels typical of high AOA at low speed dictates caution, due to the considerable uncertainty about reliability and dependability for use to stabilize and control the vehicle. Throttle usage is also a factor. While this requirement does not preclude the application of thrust vectoring for low-speed agility and supermaneuverability performance enhancements in the future, it does reinforce the position that current technology engines/inlets shall not be relied upon as the only means to assure flight safety, prevent loss of control, or provide recovery capability anywhere in the flight envelope. Should future technology advancements provide demonstrated engine/inlet reliability at low speeds and high AOAs, the procuring activity may allow this requirement to be

modified for multiple engine air vehicles such that thrust vectoring with one engine out may be used to meet it.

Control-surface-fixed stability is no longer required to attain performance benefits. Whatever the cause, control saturation can be catastrophic in a basically unstable aircraft. Then control deflection for recovery, whether commanded by the pilot or automatically, is just not available. This differs from the stable case, in which if the deflection limit is reached for trim, full control authority is available for recovery. Control rate limiting can also induce instability if the basic airframe is unstable. This requirement, together with related changes mentioned, is intended to require full consideration of all the implications of relaxed static stability and other Control-Configured Vehicle (CCV) concepts.

In considering how much margin of control shall be required there is no general quantitative answer, but it is possible to enumerate some cases to consider. Certainly there shall be sufficient control authority to pitch the aircraft out of any trim point to lower the AOA from any attainable value. That is, with full nose-down control the pitching moment shall be at least a little negative at the most critical attainable AOA, for a c.g. on the aft limit and nominal trim setting. Attainable AOA is another issue in itself; but lacking intolerable buffet or a limiter that is effective in every conceivable situation, angles to at least 90º shall be considered. Control margin is also necessary at negative AOA and large sideslips. In general, control margin shall be demonstrated at the most critical points for any axis of motion for which the unaugmented air vehicle is unstable.

The flight task will dictate some minimum amount of nose-down control capability. Air combat maneuvering certainly imposes such a requirement, and so do terminal-area operations including landing flare out. Then, there shall be some capability to counter atmospheric disturbances while maneuvering; counter c.g. movements due to fuel slosh while accelerating, diving, or climbing; stop rotation at the takeoff attitude; etc. Roll inertial coupling has been a critical factor for many slender aircraft.

In addition to conventional control modes, a CCV's direct-force controls can offer a number of new possibilities ranging from independent fuselage aiming to constant-attitude landing flares. The additional variables must be accounted for to assure adequate sizing of the control surfaces, and priorities may need to be established. The effectiveness of thrust vectoring varies with airspeed and altitude, and of course with the commanded thrust level; engine flameout or stall may be a consideration.

The instabilities and complications resulting from these factors can probably be rectified by stability augmentation if and only if control effectiveness is adequate. The controllability margin conventionally provided by static stability must be translated for CCV's into margins of controllability authority and rate. Control must be adequate for the combined tasks of trim (establishing the operating point), maneuvering, stabilization (regulation against disturbances), and handling of failures (flight control system, propulsion, etc.).

The requirements of 5.2.1.5 preclude dangerous single failures. After the first failure it may be advisable to constrict flight envelopes for some assurance of flight safety in case, say, a second

hydraulic system should fail. The contractor will need to weigh the expected frequency and operational consequences of such measures against predicted benefits.

Excessive stability, as well as excessive instability of the basic airframe, is of concern with respect to available control authority and rate. For example, large stable C_{ℓ_β} increases the roll control power needed to counteract gusts.

REQUIREMENT LESSONS LEARNED

It is well known that hinge moments can limit both deflection and rate of control surfaces. When using a surface for control in two axes, as with a horizontal stabilizer deflected symmetrically for pitching and differentially for rolling, priorities or combined limits must be set to assure safety (AIAA Paper 78-1500). Other demands on the hydraulic system can reduce control capability at times. Aeroelasticity can reduce control effectiveness directly, as well as alter the aircraft stability. For the F-16, full nose-down control put in by stability augmentation has to be overridden in order to rock out of the locked-in deep stall. Control surfaces stall at an incidence somewhat less than 90º; and if control is supplemented by thrust vectoring, for example, one must consider the control force or moment available in normal operation, the effect on forward thrust, and the possibility of flameout, as well as aerodynamic interference effects. All the possible interactions of active control must be taken into account.

Encountering the wake vortex of another aircraft can be an extremely upsetting experience. These encounters are not uncommon in practice or real combat, and also may occur in the terminal area and elsewhere; prediction is difficult. Other atmospheric disturbances can be severe, too; jetstreams, storms, wakes of buildings, etc., as well as gusts and wind shears.

The amount of control capability at extreme AOAs, positive and negative, must be enough to recover from situations that are not otherwise catastrophic. Avoidance of a locked-in deep stall has been known to limit the allowable relaxation of static stability. Also, control must be sufficient to counter the worst dynamic pitch-up tendency below stall or limit AOA. Propulsion and flight control system failure transients must be considered, along with possibly degraded control authority and rate after failure: spin/post-stall gyration susceptibility and characteristics may well be affected. The contractor must allow for fuel system failure or mismanagement.

The range of maneuvers considered shall account for both the stress of combat and the range of proficiency of service pilots. For example, in 1919 the British traced a number of losses of unstable airplanes to control authority insufficient to complete a loop that had flattened on top (ARC R&M No. 917). Thus nose-up capability at negative AOAs can also be important. Poorly-executed maneuvers may make greater demands on the flight control system for departure prevention or recovery. For CCVs as well as conventional aircraft, limiters can help greatly, but their effectiveness and certainty of operation need to be considered. Spins attained in the F-15 and F-16 attest to the possibility of defeating limiters. AFWAL-TR-81-3116 describes the A-7 departure boundary's closing in with increasing sideslip angle; angular rates also affect departure boundaries. Rapid rolling sometimes creates inertial coupling which can put great demands on pitch control; nose-down pitching seems to accentuate the divergence tendency.

External stores change both c.g. and pitching moment (C_{m_0} and C_{m_α}). Experience with past aircraft indicates a firm need to allow some margin to account for unforeseen store loadings. With relaxed static stability this can determine not only the safety but the possibility of flight with stores not considered in the design process.

Uncertainties exist in the design stage. Nonlinear aerodynamics, particularly hard to predict even from wind tunnel tests, are almost certain to determine the critical conditions. The c.g., too, may not come out as desired, and in service, the c.g. location is only known with limited accuracy. There are also possible malfunctions and mismanagement in fuel usage to consider. Recent cases (e.g., F-111 and F-16) of wind tunnel tests of basic static stability have been misleading. Aeroelasticity and dynamic control effectiveness (e.g., F-15) can also reduce control margins.

Asymmetric loadings need to be considered. A critical case for the L-19 (subsequently known as the O-1) was the addition of a wire-laying mission involving carriage of a large reel under one wing. Some aircraft—the F-15 is a recent example—have been prone to develop significant fuel asymmetries due to prolonged inadvertent small sideslipping. Dive pullouts (n greater than 1) will accentuate the effects of loading asymmetries. Some F-100s were lost from asymmetric operation of leading edge slats (non-powered, aerodynamically operated on their own, without pilot action) in dive-bombing pullouts.

Reconfigurable flight control systems add a new dimension to tracking and managing the available control power.

The control margin requirements must be met with aerodynamic control power only, without the use of other effectors, such as thrust vectoring. This approach was chosen because experience to date with current technology inlets and engines operating at the distortion levels typical of high AOA at low speed dictates caution, due to the considerable uncertainty about reliability and dependability for use to stabilize and control the vehicle. Throttle usage is also a factor. While this requirement does not preclude the application of thrust vectoring for low-speed agility and supermaneuverability performance enhancements in the future, it does reinforce the position that current technology engines/inlets shall not be relied upon as the only means to assure flight safety, prevent loss of control, or provide recovery capability anywhere in the flight envelope. Should future technology advancements provide demonstrated engine/inlet reliability at low speeds and high AOAs, the procuring activity may allow this requirement to be modified for multiple engine aircraft such that thrust vectoring with one engine out may be used to meet it.

VERIFICATION

Verification shall be by analysis and by simulation or flight test in the performance of the tasks of 5.1.1.1, 5.1.1.2, 5.1.1.3.2, and 5.1.1.3.3 under various levels of atmospheric disturbances. These tasks will be flown by test pilots at specific flight conditions throughout the ROSH and ROTH. The specific flight conditions to be evaluated shall be the most common operating conditions, any operating conditions critical to the mission of the air vehicle, and any conditions determined by analysis or simulation to be worse than the Level of flying qualities required. For conditions which are considered too dangerous to test in flight, verification can be shown in a manned simulation. Proof of compliance in these demonstration tasks will consist of pilot

comments and C-H ratings. The comments and ratings shall indicate that the flying qualities are no worse than the required Level of flying qualities for each combination of Air Vehicle State, Flight Phase, and level of atmospheric disturbance.

This is a flight safety item. Analysis and simulation shall proceed or accompany careful build-up to suspected critical flight conditions.

Compliance with this requirement through flight demonstration is not expected in dangerous cases. "Combined range of all attainable AOAs and sideslip" may even extend beyond the RORH, except for certain highly-maneuverable fighter and trainer air vehicles. Flight test bounds will be established according to such requirements as those contained in the cancelled MIL-F-83691. For extreme flight conditions a combination of model testing—wind tunnel, free-flight if necessary, and hardware—and analysis will often be adequate. These extremes shall be investigated in some way, whether or not the air vehicle incorporates a limiter. The scope of analysis, simulation, and testing needs careful consideration at the outset of a program. Then the progress must be monitored for possible additional troubles.

AFWAL-TR-87-3018 gives guidance on determining control deflection and rate margins and calculating the deflection-saturated departure boundary in the conceptual and preliminary design stages, based on a reduced-order system with full-state feedback. At high speed and high dynamic pressure, the system bandwidth required is high, increasing the importance of high-frequency control system modes, structural modes, and system noise amplification. At low speed and low dynamic pressure, design risks are related to the limited ability of aerodynamic control surfaces to generate control moments. The lack of stabilizing control moments beyond some AOA or control-surface rate limits, will compromise transient responsiveness. Describing function analysis treats control limiting as a gain reduction, which in general lessens the stabilizing effect of feedbacks. Statistically-based margins for gusts reduce, but do not eliminate, the possibility of inadequate control margin.

Figure 17 indicates some critical parameters in the response of an unstable system to a step command. Factors influencing some control margin increments may be seen in table XVIII. To these margins must be added another nose-down control increment to counter the pitch-out tendency while rolling about the x stability axis (flightpath). As a first cut,

$$\left|\Delta\delta_{pr}\right| = \left|\frac{I_{xx} - I_{zz}}{2M_{\delta}I_{yy}} \cdot p^2 \cdot \sin 2\alpha\right|$$

$$\left|\dot{\delta}_{pr}\right| = \left|\frac{I_{xx} - I_{zz}}{M_{\delta}I_{yy}} \cdot p \cdot \dot{p} \cdot \sin 2\alpha\right|$$

where p is the stability-axis roll rate (about the flightpath). Figure 18 shows the conceptual margins needed: $\Delta\delta_{marg}$ is the sum of turbulence and sensor noise components, $\Delta\delta_{tran}$ provides the pitching acceleration to meet the CAP requirement, and $\Delta\delta_{pr}$ can cancel the inertial pitching moment from rolling. Unless deactivated whenever saturation is encountered, an integrator in the flight control system tends to run away, leading to loss of control.

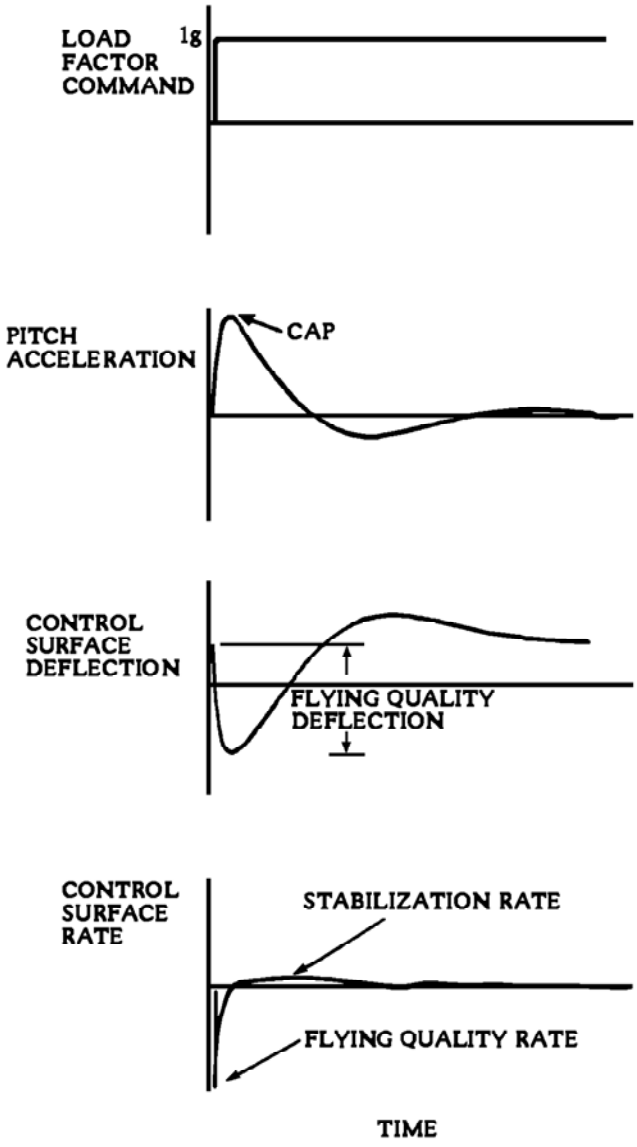


FIGURE 17. Control surface requirements.

Similar considerations, also treated in AFWAL-TR-87-3018, apply to any basic airframe having static directional instability.

While flight test risk must be bounded, it is necessary to assure by some means that any dangerous conditions are found and evaluated before Service pilots and air vehicles are lost through surprise encounters, with no known avoidance or recovery technique. Flight experience can be summarized by Murphy's Law: anything that can go wrong, will go wrong. Therefore, it is better for highly-skilled flight test pilots to find any serious glitches under controlled conditions rather than wait for some less-experienced operational pilots to find them in-service.

TABLE XVIII. Control margin increment.

Flying Quality	$\Delta\delta_{FQ}/\Delta n_c = 57.3\ CAP'/M_\delta$ °/g (for $T_{eff} \le 0.05$)
Stabilization	$\Delta\delta_{stab}/\Delta n_c = 57.3\frac{g}{U_0}\cdot\frac{1/T_{sp_1}\cdot 1/T_{sp_2}}{M_\delta \cdot 1/T_{\theta_2}}$ °/g (linear, 2DOF)
Turbulence	σ_δ/σ_w function of M_w, M_δ, $\omega_{sp_{cl}}$, $\zeta_{sp_{cl}}$, structural modes - most severe at low $\bar{q}$ - $3\sigma_\delta$ and σ_w for severe turbulence recommended
Sensor noise	σ_δ/σ_s function of K_s, K_F, ω_s, $1/T_a$, $\omega_{sp_{cl}}$, $\zeta_{sp_{cl}}$, $\omega^2_{sp_{ol}}$
Flying Quality	$\dot{\delta}_{FQ}/n_c = 57.3\ CAP/(M_\delta \cdot T_{eff})$ for desired CAP
Stabilization	$\dot{\delta}_{stab}/n_c < \dot{\delta}_{FQ}/n_c$ if FCS stability margins OK & $1/T_{eff} > \omega_c$ $\dot{\delta}_{stab}/n_c$ function of $1/T_{eff}$, $1/T_{sp_2}$, $\omega_{sp_{cl}}$, $\zeta_{sp_{cl}}$
Turbulence	$\sigma_{\dot{\delta}}/\sigma_w$ function of $1/T_a$, $\omega_{sp_{cl}}$, $\zeta_{sp_{cl}}$, M_δ - most severe at low $\bar{q}$ - $3\sigma_{\dot{\delta}}$ recommended for control margin
Sensor noise	$\sigma_{\dot{\delta}}/\sigma_s = K_s K_F \cdot fn(\omega_s, 1/T_a$ and, for low $\omega_{sp_{cl}}$: $\omega^2_{sp_{ol}}$, $\omega_{sp_{cl}}$, $\zeta_{sp_{cl}})$ - these parameters are not all independent - $3\sigma_{\dot{\delta}}$ recommended for control margin

Δn_c is the commanded increment of normal acceleration
$1/T_{sp_2}$ is the unstable pole of the transfer function (negative; 1/sec)
$\omega^2_{sp_{ol}}$ is the 2-deg-of-freedom product of the poles, $1/sec^2$
$\omega_{sp_{cl}}$ and $\zeta_{sp_{cl}}$ are the closed-loop frequency and damping ratio of the short-period mode
CAP is $\dot{q}_0/\Delta n_{z_{ss}}$, CAP′ is $\dot{q}_{max}/\Delta n_{z_{ss}}$
ω_s is the sensor bandwidth
K_s, K_F are the sensor and forward-loop gains
σ_s, σ_w are the rms intensities of sensor noise and vertical gusts
ω_c is the crossover frequency of the $\dot{\delta}/n_c$ transfer function
T_{eff} is the effective time constant of command-path plus forward-path control-loop elements (such as prefilters and actuators)
T_a is the time constant of the actuator ram

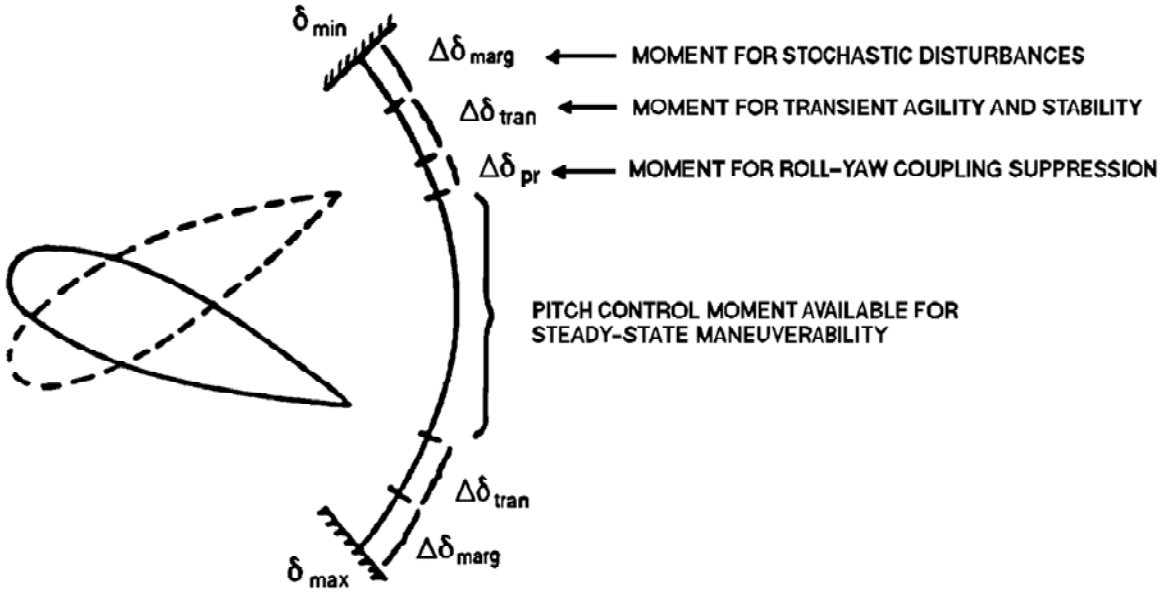


FIGURE 18. Control margin requirements.

During F/A-18 high-AOA/stall testing, an AOA hang-up phenomenon was observed (at 50-60º AOA) which was very similar to the F-16 deep stall. At operational aft c.g.'s and high AOA, delayed recoveries were experienced in the F/A-18 due to weak nose-down pitch restoring moments, even with full forward stick. Based upon the F/A-18 test experience a pitch restoring (nose-down) moment coefficient magnitude of at least 0.2 (i.e., $C_m \leq -0.2$) shall be available for the most longitudinally unstable loading and aft c.g. combination expected to exist on Class IV air vehicles. Analysis of the F/A-18 test data from high-AOA post-stall gyrations shows that the AOA hang-up phenomenon was further aggravated by uncommanded roll rate and yaw rate oscillations and resultant nose-up pitching moments. Flight test results indicate that these oscillations could generate pitching moments approximately equivalent to a change in pitching moment coefficient of +0.1, which significantly opposed natural aerodynamic pitch restoring moments. Occasionally F/A-18 recoveries from high-AOA hang-ups were significantly delayed because of accompanying roll/yaw rate oscillations when c.g./loading/AOA conditions caused C_m (with full nose-down control input) to be greater than approximately -0.2. This suggests that a pitch recovery criterion for Class IV air vehicles could be that the pitch recovery control produce a net pitch restoring moment coefficient magnitude not less than 0.1 ($C_m \leq -0.1$), which yields approximately 15-20º/sec^2 nose down at low airspeed.

5.2 Detail requirements

5.2.1 General flying qualities requirements

5.2.1.1 Approach to dangerous flight conditions.

When approaching dangerous flight conditions at which the air vehicle should not be flown, it shall be possible by clearly discernible means for the pilot to recognize the impending dangers and take preventive action.

5.2.1.1.1 Warning and indication.

Warning and indication of approach to a dangerous condition shall be clear and unambiguous.

5.2.1.1.2 Operation of devices for indication, warning, prevention, and recovery.

As a minimum, special devices incorporated to eliminate dangerous flight conditions shall perform their functions whenever needed but shall not limit flight within the ROSH. Neither normal nor inadvertent operation of such devices shall create a hazard to the air vehicle. For Levels 1 and 2, nuisance operation of these devices shall not be possible. Functional failure of these devices shall be indicated to the pilot.

REQUIREMENT RATIONALE

Approach to any dangerous flight condition must be clearly apparent to the pilot with sufficient margin (time, control power, etc.) to avoid loss of control. That, together with limiting the frequency of encounter, is the essence of flight safety as it involves flying qualities.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 and 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.1 through 3.4.1.2
MIL-STD-1797: 4.1.8 through 4.1.8.2.

REQUIREMENT LESSONS LEARNED

The designer shall endeavor to eliminate dangerous flight conditions by appropriate changes in aerodynamic, flight control, or mass distribution design. Dangerous flight conditions that can not be eliminated shall be identified as conditions at which the aircraft shall not be flown and appropriate devices for indication, warning, prevention, or recovery incorporated.

The need for warning may not become apparent until late in the development program (or after it), and each such device will generally have to be tailored to a specific set of conditions. These requirements clearly apply to stall warning and prevention devices, as well as to other types.

Certain failures may require restriction of the flight envelope in order to assure safety after the failure, or in the event of a subsequent failure. The flight crew needs to be made aware when such a situation exists.

One possible source of danger is the use of special, unconventional control modes for certain tasks. Under stress, a pilot may forget and revert to the normal technique or exceed a limit. For example, a yaw-pointing mode might generate excessive sideslip at low dynamic pressure or too much lateral acceleration at high dynamic pressure. Also, partially or fully automatic modes of flight need to be examined for possible hazards.

Normally, a reasonably reliable limiter or warning need not be redundant. If the pilot knows that the device is inoperative, he can stay well clear of the danger. Nuisance operation not only interferes with mission performance; it breeds disregard or disconnection. Reliance is placed on the flight control system requirements and other requirements to assure sufficient reliability of these devices, warning of their failure, and checkout provisions.

Requirement 5.2.1.1.2 is designed to discourage prevention devices that create more problems than they solve.

Stall limiters have proved to be of significant help, as with the F-101B Boundary Control System and the F-16 stall, load factor, and roll limiter, which allow carefree maneuvering up to the set limits. However, an undefeatable limiter is hard to design. If a dangerous pitch-up or locked-in deep stall lurks beyond, some pilot will encounter it. Indeed pilots would rather bend the wings than hit the ground, so a soft limiter may be in order. On the other hand, makers and flyers of air vehicles with no post-stall limitations (e.g., T-38/F-5, F-15) find these extreme AOAs useful occasionally in air combat.

Several C-133 losses over oceans are conjectured to have resulted from starting long-range cruise too close to stall, with no stall warning and a severe roll-off in a power-on stall. The artificial stall warning often was turned off because it was not reliable.

F-16s have been lost in ground attack runs because of deteriorated aerodynamics with external stores. The fix was to change the limits when air-to-ground stores are carried. Although the F-16 uses a pilot-operated switch, this can be done automatically, so that the pilot does not have to remember to switch.

Sensors critical to air vehicle safety shall be designed and located for adequate sensing of the appropriate parameters and minimization of exposure to conditions which could produce spurious indications. During development of a ground proximity warning system using an F/A-18 manned flight simulator, the system produced occasional spurious warnings, falsely indicating close proximity to the ground when the condition did not actually exist. The development philosophy was that the production system must not produce spurious indications since such false indications render warning systems useless. The development team recognized that even infrequent occurrences of false warnings would cause the pilot to ignore audible or visual warning cues and delay taking required action until the indication was verified to be genuine through secondary indications. Experience has shown that pilots are also inclined to

disable a warning system that emits frequent or even occasional false warning signals. Experience has also demonstrated that the pilot may inadvertently modify flying habits and techniques to prevent a system prone to false indications from enunciating a spurious warning. Any modification to air vehicle operations due to false warning signals will inherently degrade training and operations and may induce poor flying habits that hamper the completion of the mission or safety of the air vehicle. The development team on the ground proximity warning system chose to eliminate nuisance warnings at the expense of reducing protection since the system would be considered ineffective in preventing ground impact if the pilot spends several seconds assessing the validity of the indication.

VERIFICATION

Testing will be necessary to assure that the limiting or warning is satisfactory in all maneuvers and that functional failure of any such devices is indicated to the pilot. Ultimately, flight testing will be required (see, for example, the cancelled MIL-F-83691).

Verification shall be by analysis, simulation, and ground and flight test. Dangerous flight conditions at which the air vehicle should not be flown shall be identified. As part of envelope expansion, the air vehicle shall be flown toward these conditions. The pilot shall evaluate the warnings and indications of approach to dangerous flight conditions. The pilot comments shall indicate that these warnings and indications are clear and unambiguous and that they can recognize the impending dangers in time to take preventive action to avoid the dangerous conditions. Since it is not necessary to actually fly into the dangerous flight conditions, Air Vehicle Normal States and Extreme States shall be evaluated in flight. Failure States may be evaluated on the simulator if the failures are considered too dangerous to test in flight. During failure evaluations, the pilot, or the crew member concerned, shall evaluate the failure indications. Crew member comments shall indicate that the failure indication is immediate and easily interpreted. The pilot shall evaluate any special devices incorporated to eliminate dangerous flight conditions. Pilot comments shall indicate that these devices perform their functions as necessary but do not limit performance of required operational maneuvers. Pilot comments shall also indicate that operation of these devices is not hazardous to the air vehicle and, for those conditions at which Level 1 and Level 2 flying qualities are required, that operation of these devices does not create a nuisance in flight. Failure indication for each of these devices shall be verified by inspection.

The procuring and test activities must assess the degree of danger, and their test pilots must agree on the acceptability of devices, air vehicle characteristics, and flight manual warnings.

FTC-TIH-79-2, FTC-TD-73-2, and USNTPS-FTM-103 contain guidance on flight testing for stall/post-stall and other conditions of concern. Specification MIL-S-83691 was an Air Force flight test specification and MIL-D-8708 the corresponding Navy specification.

5.2.1.2 Buffet.
Within the boundaries of the __(1)__, there shall be no objectionable buffet which might detract from the effectiveness of the air vehicle in executing its intended missions. Within the __(2)__, vertical acceleration due to buffet shall not exceed __(3)__.

REQUIREMENT RATIONALE

The intent of this requirement is to prevent the occurrence of objectionable levels of buffet in the course of operational flight.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.5
MIL-STD-1797: 4.1.11.1.

The recommended value for both blanks (1) and (2) is: ROSH.

For a combat air vehicle the procuring activity may need to extend the requirement to apply throughout the RORH. The extension would apply for example to an air combat fighter intended to have high-AOA capability and to a trainer for spinning.

For blank (3), vertical acceleration shall not exceed: $\pm.2\ g_z$ about trim.

REQUIREMENT LESSONS LEARNED

In those cases where buffet is a signal to the pilot of approach to a dangerous flight condition (5.2.1.1) some buffet is desirable; but, there should be no need for that within the ROSH.

AGARD-AR-82 contains a concise discussion of buffet and offers some guidelines on the acceptability of various buffet levels:

> To the fighter pilot who knows his aircraft, buffet onset is a valuable source of information in moments of intense activity when he is not able to refer to his flight instruments. Of the many different buffet level criteria to be found...the following is a summary which smoothes out the variations. The "g" values quoted are maximum excursions about trim:
>
> | | | |
> |---|---|---|
> | Onset: | $\pm.035$ to .1 g_z | perception depends on workload/normal g |
> | Light: | $\pm.1$ to .2 g_z | definitely perceptible |
> | Moderate: | $\pm.2$ to .6 g_z | annoying |
> | Severe: | $\pm.6$ to 1.0 g_z | intolerable for more than a few seconds |

Provided that there are no other effects such as loss of full control or random aircraft motions, light buffet usually had no adverse effect on maneuvering, either coarsely or precisely. The average fighter pilot is so used to flying in this region that he may not even comment on it at the lower amplitudes. He will however feel annoyance and frustration when the buffet characteristics reach the level where his ability to track his target is affected; other effects on his performance may result from the arm-mass feedback to the stick and his ability to see the target on his cockpit controls and instruments. At the intolerable level the motion becomes physically punishing and full control is not possible as a result of the effect of the buffet on the pilot himself.

The significance of buffet in air combat depends upon the task. If flight in buffet gives a performance improvement then pilots will use this region during the tactical phase of combat. Tracking will also take place at high buffet levels, even with guns; but when the low-frequency, high-amplitude "bouncing" buffet occurs then there is no further advantage to be gained from operating in this region.

VERIFICATION

Wind tunnel tests can give early indication of buffet onset and intensity, but flight testing will be needed to determine the end effect with structural vibrations, noise, etc. included.

Verification shall be by flight test during the verification of 5.1.1.1 and 5.1.1.2. Pilot comments shall indicate that the buffet is not so objectionable as to detract from the effectiveness of the air vehicle in executing its intended missions. "Objectionable" is to be interpreted in the context of operational missions: a distraction, or discomfort so great as to interfere with the operational task. Judgement is subjective, so marginal or critical cases must be evaluated by a number of pilots.

Flight testing at elevated AOAs and load factors, and at lower angles transonically, will reveal any buffeting tendencies. A wind-up turn maneuver while tracking a target can be especially useful in identifying buffet regions.

The test loadings selected for the analysis of F/A-18E/F flying qualities during developmental testing were mission representative loadings that were expected to demonstrate the most degraded flying qualities characteristics induced by loading. The typical mission-representative loadings used in F/A-18 flying qualities flight testing were for the fighter escort mission and the interdiction mission. However, the operational test team selected a loading for test that was representative of an air vehicle that had completed weapons delivery. The operational test loading consisted of a centerline fuel tank with empty wing pylons. The particular loading had not previously been tested by the developmental test team. In preparation for operational testing, the developmental test team evaluated this loading in flight and discovered that the air vehicle had a tendency to buffet at transonic airspeeds in this mission-representative loading. This phenomena had not been encountered in the fighter escort or interdiction loadings. Discovering this critical problem with the air vehicle at a relatively late stage in the air vehicle development potentially compromised the entire program. Although it is not economical to flight test all

possible loadings, the test loadings must be selected to ensure all store-induced anomalies are discovered since dramatic problems, like those discovered during F/A-18E/F testing, may necessitate a design modification that could require repeating substantial flight testing and significantly affect other air vehicle characteristics.

5.2.1.3 Release of stores.
For Levels 1 and 2, the intentional release or ejection of any stores shall not result in objectionable flight characteristics or impair tactical effectiveness. The intentional release or ejection of stores shall never result in dangerous or intolerable flight characteristics.

REQUIREMENT RATIONALE

This requirement is included to insure that stores release or ejection will not have an adverse effect on flying qualities.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1.1 and 3.3.11.1.1.2 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.6
MIL-STD-1797: 4.1.11.2.

REQUIREMENT LESSONS LEARNED

Because of the variety of possibilities, this requirement must be left qualitative. All store loadings, internal and external, which are specified in the contract are covered. Also, see 3.4.1 in JSSG-2001 and MIL-HDBK-244 for additional guidance.

VERIFICATION

Evaluation of this requirement shall occur as a natural part of operational flight testing, usually preceded by analysis (e.g., AFFDL-TR-74-130, AFWAL-TR-80-3032, and Computational Fluid Dynamics (CFD) analysis) and wind tunnel testing. The wind tunnel tests may be guided on-line by trajectory calculations using a combination of currently generated and stored data.

Verification shall be by flight test during stores separation testing. Although 5.2.1.3 applies for all flight conditions and store loadings at which normal or emergency release or ejection of the store is permissible, testing of all flight conditions and store loadings is impractical, therefore selected Air Vehicle Normal States shall be tested at selected flight conditions throughout the envelope of permissible stores separation for each State. Pilot comments shall indicate that release or ejection of stores does not result in dangerous or intolerable flight characteristics. Where Levels 1 and 2 are required, pilot comments shall indicate that release or ejection of stores does not result in objectionable flight characteristics or impair tactical effectiveness.

There may be special flight envelopes in which store release or missile firing are not permitted. Generally such envelopes are ultimately cleared by flight testing.

Store motion after release is such a function of the local airflow field that few generalities can be made about the most critical conditions. At the same AOA the aerodynamic forces are greater at higher speed. Dive angle, normal acceleration, store location, and store and ejector configuration and loading are important to consider. The critical conditions will likely be at the boundaries of the release envelopes.

5.2.1.4 Effects of armament delivery and special equipment.
For Levels 1 and 2, operation of movable parts, such as weapons bay doors, cargo doors, armament pods, refueling devices, and rescue equipment, or firing of weapons, release of bombs, or delivery or pick-up of cargo shall not cause buffet, trim changes, or other characteristics which impair the tactical effectiveness of the air vehicle under any pertinent flight condition.

REQUIREMENT RATIONALE

This requirement is included to assure that armament delivery, etc., will not adversely affect flying qualities, impairing mission effectiveness.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1.1 and 3.3.11.1.1.2 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.7
MIL-STD-1797: 4.1.11.3.

REQUIREMENT LESSONS LEARNED

Because of the variety of possibilities, this requirement must be left qualitative. All armament and equipment for the design missions are covered.

VERIFICATION

Operational flight tests shall be required, preceded by suitable analyses and wind tunnel tests. Generally the critical conditions should thus be known before flight test.

Verification shall be by flight test during envelope expansion and stores separation testing. For those conditions for which Level 1 or Level 2 flying qualities are required, pilot comments shall indicate that operation of movable parts, firing of weapons, release of bombs, or delivery or pick-up of cargo, does not cause buffet, trim changes, or other characteristics which impair tactical effectiveness of the air vehicle.

Each movable part needs to be analyzed for its effect on stability and control.

Gun firing can cause deceleration and, depending on lateral and vertical location, attitude transients. It has also been known to interfere with engine-inlet airflow or pilot vision. Rigidity

and dynamics of local structure and items attached to it influence the air vehicle vibration resulting from gun firing.

5.2.1.5 Failures.
No single failure of any component or system or combination of single independent failures shall result in dangerous or intolerable flying qualities; Special Failure States (4.1.3.7) are excepted. The crew member concerned shall be given immediate and easily interpreted indications whenever failures occur that require or limit any flight crew action or decision. The air vehicle motions following sudden air vehicle system or component failures shall be such that dangerous conditions can be avoided by the pilot without requiring unusual or abnormal corrective action. Failure-induced transient motions and trim changes either immediately upon failure or upon subsequent transfer to alternate modes shall not cause dangerous or intolerable flying qualities. Configuration changes required or recommended following failure shall not cause dangerous or intolerable flying qualities. When determining compliance, a realistic time delay between the failure and initiation of pilot corrective action shall be incorporated in each case. This time delay shall include an interval between the occurrence of the failure and the occurrence of a cue such as acceleration, rate, displacement, or sound that will definitely indicate to the pilot that a failure has occurred, plus an additional interval which represents the time required for the pilot to diagnose the situation and initiate corrective action. While this time delay may be different for each combination of Failure State, flight condition, and Flight Phase, in no case shall this time delay be less than __(1)__.

REQUIREMENT RATIONALE

These requirements pertain directly to the flight control system and apply generally to all axes of control and response. These generally qualitative requirements result from experience and involve safety of flight. In addition to accounting for flying qualities after a failure, it is recognized the transient between the normal and the failed state could result in further flying qualities degradation.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1.2 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.8, 3.4.9, and 3.5.5
MIL-STD-1797: 4.1.7, 4.1.11.4, and 4.1.12.8.

Recommended value for blank (1): 1 sec

REQUIREMENT LESSONS LEARNED

A minimum realistic time delay value of 1 sec is consistent with requirement 3.3.9.3 in MIL-F-8785C. For civil operation the Federal Aviation Administration (FAA) is more conservative with hardover failures of autopilot servos, requiring 3 sec before pilot takeover is assumed. This time delay is to include an interval between the occurrence of the failure and the occurrence of a cue such as acceleration, rate, or sound that will definitely indicate to the pilot

that a failure has occurred, plus an additional interval which represents the time required for the pilot to diagnose the situation and initiate corrective action. The length of time shall correspond to the pilot's likely readiness to respond, for example longer during cruise than at takeoff.

NASA CR-177331 presents guidance on determining a realistic time delay that seems as applicable to winged air vehicles as it is to rotorcraft. Table XIX and the following paragraph are excerpts.

> "Pilot response time is especially critical in defining a reasonable minimum pilot intervention delay time to a failure. The status of the pilot in the overall task of controlling the rotorcraft can be described as active or attended control operation, divided attention control operation (both hands on the controls and hands off), or unattended control operations such as in autopilot mode (both hands on and hands off the control). For example, if the pilot is making a final approach to a landing, he would be considered to be in an attended operation mode of rotorcraft control with his hands on the controls. Should an automatic flight control failure occur, the minimum pilot response time for corrective control input following recognition of the failure would be quite small, approximately half a second. Therefore, for testing the acceptability of failures in this mode of flight it would be unreasonable to require testing (or specification) of a minimum allowable response time any greater than ½ second. However, for cross-country flight at cruise speeds, it is very possible that the pilot will not have his hands on the control if an autopilot is engaged. For failures which have a significant probability of occurrence in this flight mode, the specification of ½ second pilot response time for test purposes would be unreasonable and unsafe. In this standard, therefore, the minimum allowable pilot response time would be adjusted to 2½ seconds following any single failure."

Following some failures, a pilot's lack of adaptation, or inappropriate adaptation, can result in a pilot/airframe closed-loop instability, even if the aircraft itself remains stable. Examples are the B-58 yaw damper and NASA TN D-1552. The required failure indications depend on operational rules. Consistent maintenance and checkout capability and rules are needed.

Air vehicles have been lost from runaway trim. That possibility needs careful consideration for every powered trim system.

VERIFICATION

Verification shall be by analysis during the FMECA and FMET and by evaluation in simulation and flight test. For conditions which are considered too dangerous to test in flight, verification can be shown in a manned simulation. Proof of compliance in these demonstration tasks shall consist of pilot comments. Pilot comments shall indicate that no single failure of any component or system results in dangerous or intolerable flying qualities and that transients following sudden failures do not require unusual or abnormal corrective action to avoid dangerous conditions. Crew member comments shall indicate that failure indications are immediate and easily interpreted.

TABLE XIX. Examples of alternate minimum intervention times for system failures.

Phase of Flight	Rotorcraft Response $t_1 - t_0$*	Pilot Response $t_2 - t_1$**	Minimum Allowable Intervention Delay Time and Method of Test
Attended Operation	Time for rotorcraft to achieve change of rate about any axis of 3^o/sec or The time to reach a change of "g" in any axis of 0.2 or For an attention getter to function	½ second	System failures will be injected without warning to the pilot. His ability to recover as rapidly as possible without a dangerous situation developing will be used to assess system failure mode acceptability.
Divided Attention Operation Hands On	As above	1½ seconds (Decision 1 + reaction ½)	The pilot will be warned of the system failure. Demonstration of compliance must show that an intervention delay time equal to 1½ seconds plus ($t_1 - t_0$) can be tolerated.
Divided Attention Operation Hands Off	As above	2½ seconds (Decision 1½ + reaction 1)	As above but intervention delay time 2½ seconds plus ($t_1 - t_0$)
Unattended Operation Hands On	As above but the threshold rates and "g" values are 5^o/sec and 0.25 respectively	2 ½ seconds (Decision 2 + reaction ½)	As above
Unattended Operation Hands Off	As above	4 seconds (Decision 3 + reaction 1)	As above but intervention delay time 4 seconds plus ($t_1 - t_0$)

*ROTORCRAFT RESPONSE TIME INTERVAL ($t_1 - t_0$). This is the period between the failure occurring and the pilot being alerted to it by a suitable cue. The cue may take the form of an adequate tactical, audio, or visual warning. (The eye cannot be relied upon to distinguish abnormal instrument indications sufficiently early for these to be regarded as an adequate cue.) In the absence of the adequate cues listed above, it can be assumed that a pilot will be alerted when the rotorcraft meets or exceeds the responses listed for unattended operation.

**PILOT RESPONSE TIME INTERVAL ($t_2 - t_1$). The period commences at the time the pilot is alerted that something abnormal is happening and terminates when the controls are moved to commence the recovery maneuver. The period consists of the recognition time, decision time, and reaction time. As shown above, the recognition and decision times are assumed to increase as the pilot relaxes his level of involvement, i.e., the transition from"attended operation" to "unattended operation" and also from "hands on" to "hands off." The reaction time is longer "hands off" than "hands on" as the pilot has to locate the controls before he can move them.

For those failures and flight conditions judged to be too hazardous to evaluate in flight, demonstration likely will be by simulation. Validated models of the air vehicle and its flight control system will be needed for this.

"Dangerous flying qualities" need to be interpreted in a rational manner, so that it applies to feasible design options. All the danger of flying cannot be removed.

5.2.1.5.1 Transients following failures.
With controls free, the air vehicle motions due to partial or complete failure of any subsystem of the air vehicle shall not exceed the limits of table XX for at least __(1)__ following the failure.

TABLE XX. Transients following failures.

Required Level After Failure	Flight Phase Category	Transient Limits

REQUIREMENT RATIONALE

The general failure requirement of 5.2.1.5 governs allowable air vehicle motion when corrective action is taken. This requirement limits the severity of failure transients before corrective action is taken, avoiding a possible flight safety problem. The transients due to the failure must be small enough to allow the pilot to regain control; and, having done so, to operate at least adequately to terminate the mission (this is implied in 5.2.1.5 by requiring Level 3 or better flying qualities following any single failure). Adequate protection for failure transients must be provided in each axis, in case corrective action is delayed even slightly. Quantitative limits are needed to avoid pilot workload increases and flight safety problems.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1.2 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.5.5.1
MIL-STD-1797: 4.2.6.1, 4.5.7.1, and 4.6.5.2.

Recommended value for blank (1): 2 sec

Recommended values for table XX:

Required Level After Failure	Flight Phase Category	Transient Limits
1 and 2	B and C	No more than ±0.5 g incremental normal and lateral acceleration at the pilot's station; No more than ±10°/sec roll rate; Neither stall AOA nor structural limits shall be exceeded
	A	No more than ±0.5 g incremental normal and lateral acceleration at the pilot's station; No more than ±10°/sec roll rate; Vertical and lateral excursions no more than ±5 ft; No more than ±2º of bank angle; Neither stall AOA nor structural limits shall be exceeded
3	All	No dangerous attitude or structural limit is reached, and no dangerous alteration of the flightpath results from which recovery is impossible

REQUIREMENT LESSONS LEARNED

The transient motion limits were taken from requirement 3.5.5.1 of MIL-F-8785C. Although the intent of the requirement is to insure that dangerous flying qualities never result, there may be some benefit to a noticeable transient after a failure, or after transfer to an alternate control mode, in order to alert the pilot to the change. That possibility is left to the designer without explicit direction to minimize transients. This requirement also places quantitative limits on the altitude change, effectively restricting the 2-second average acceleration in addition to the peak value. The 2-second interval is to account for crew distraction and the possible need for time to determine and accomplish corrective action.

The revision to MIL-F-8785C followed the recommendations of Systems Technology Inc. TR-189-1: the authors noted that the allowable transient levels of MIL-F-8785B were consistent with failure probability considerations, but not with flying qualities considerations. Level 2 had a lower probability of occurrence than Level 1 and was permitted to have larger transient responses; however, Level 2 is a poorer handling qualities state and cannot accept the larger responses as readily. It was felt that the values in MIL-F-8785C were representative of transients which could be handled with Level 1 flying qualities. Conversely, the low allowable transients of MIL-F-8785B were conducive to soft failures which could lead to catastrophic situations if undetected by the pilot. This comment applied to the B-58 in particular, and lead General Dynamics/Fort Worth to suggest a minimum allowable transient (according to Systems

Technology Inc. TR-189-1). This has not been incorporated into this document, but shall be a consideration in the design process.

VERIFICATION

Verification shall be by analysis during the FMECA and FMET and by evaluation in simulation and flight test. For conditions which are considered too dangerous to test in flight, verification can be shown in a manned simulation. Proof of compliance in these demonstration tasks shall consist of pilot comments. Pilot comments shall indicate that no single failure of any component or system results in dangerous or intolerable flying qualities and that transients following sudden failures do not require unusual or abnormal corrective action to avoid dangerous conditions. Crew member comments shall indicate that failure indications are immediate and easily interpreted.

Worst-case flight conditions shall be identified and tested. High control effectiveness, authority, and gain; low or negative static stability or damping; low weight and inertia will tend to make the transients larger. Generally a dynamic analysis is needed, but constant speed can be assumed for the two-second period of time.

Any failures specified under 5.1.1.3.2 shall be evaluated in flight testing at the most critical flight conditions. Other failures and worst-case flight conditions will be found from the Failure Modes and Effects Analyses of 5.1.1.3.1.

Some failures may be considered too dangerous to flight test. Where final demonstration is by simulation, flight-validated aerodynamic data shall be used with actual flight hardware and software loaded as necessary to replicate the response in flight. Adequate motion cues shall be available to simulate the acceleration environment with one-to-one fidelity for at least 2 sec following the failure.

For flight testing, the test aircraft may need special provisions for introducing specific failures in flight. Analysis or simulation shall precede such flight tests in order to avoid excessively risky combinations of failure and flight condition.

5.2.1.5.2 Trim changes due to failures.
Without retrimming, the change in control force required to maintain constant attitude and sideslip following complete or partial failure of the flight control system shall not exceed the limits of table XXI for at least __(1)__ following the failure.

TABLE XXI. Control force limits following failures.

Axis of Response	Maximum Control Force Change
Pitch	
Roll	
Yaw	

REQUIREMENT RATIONALE

These requirements are intended to assure that the short-term response of the air vehicle to a flight control system failure does not cause loss of control before the pilot can react, and that a large effort to maintain control is not required of the pilot. The requirements of 5.2.1.5.1 describe allowable transient responses. However, there shall also be limits on the control forces required to minimize these transient responses. Flight control system failures shall not cause abrupt or severe changes in the trim state of the air vehicle. The ability to retain reasonable control is measured in terms of demands on the pilot to maintain trim conditions. Limits must be placed on the maximum force to counter trim changes after a failure of any portion of the primary flight control system.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1.2 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.5.5.2
MIL-STD-1797: 4.2.8.6.5, 4.5.9.5.6, and 4.6.7.9.

Recommended values for table XXI:

Axis of Response	Maximum Control Force Change
Pitch	20 lbs*
Roll	10 lbs
Yaw	50 lbs

*While 20 lbs is within the capability of all pilots, a lower value may better accommodate females.

Recommended value for blank (1): 5 sec

REQUIREMENT LESSONS LEARNED

The control force limits are consistent with 5.2.2.1.8, 5.2.3.6, and the data of AFAMRL-TR-81-39. It seems reasonable to state a time limit during which this requirement applies. Zero to 2 sec generally shall be a rational range of times for a pilot to react (whether he is set for the failure or taken unawares), and it shall be possible to retrim after 5 sec.

VERIFICATION

Verification shall be by analysis during the FMECA and FMET and by evaluation in simulation and flight test. For conditions which are considered too dangerous to test in flight, verification can be shown in a manned simulation. Proof of compliance in these demonstration tasks shall consist of pilot comments. Pilot comments shall indicate that no single failure of any component or system results in dangerous or intolerable flying qualities and that transients following sudden failures do not require unusual or abnormal corrective action to avoid dangerous conditions. Crew member comments shall indicate that failure indications are immediate and easily interpreted.

For flight testing, it may be necessary to provide the test air vehicle with special means of introducing failures. Testing of failure modes—in flight or simulation—shall always include consideration of demands on the pilot to retrim manually. The location of critical points in the flight envelope is highly dependent on the aerodynamic and flight control system configurations.

5.2.1.6 Pilot-in-the-loop oscillations.
There shall be no tendency for pilot-in-the-loop oscillations, that is, unintentional sustained or uncontrollable oscillations which result from the efforts of the pilot to control the air vehicle.

REQUIREMENT RATIONALE

The purpose of this requirement is to insure that abrupt maneuvers or aggressive tracking behavior will not result in instabilities of the closed-loop pilot-vehicle system. Any such tendency will degrade or even destroy mission effectiveness and likely will be dangerous.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 through 3.3.11.1.2 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.2.3 and 3.3.3
MIL-STD-1797: 4.1.11.6, 4.2.2, 4.5.2, and 4.6.3.

REQUIREMENT LESSONS LEARNED

Pilot-in-the-loop oscillations are associated with abrupt maneuvers and precise tracking as in air-to-air gunnery, formation flying, flare, and touchdown. Tight, aggressive pilot control action will tend to bring on any PIO tendency. High sensitivity to control inputs is often a factor.

Some pilots are better at exposing PIO tendencies than others, depending upon piloting technique.

Likely causes are equivalent time delay, control system friction, inappropriately-located zeros of air vehicle transfer functions, or nonlinearities such as rate limiting, hysteresis, and abrupt control system gain changes. NORAIR Rpt No. NOR-64-143 discusses a number of possible PIO mechanisms.

This requirement precludes PIO tendencies or general handling qualities deficiencies resulting from inadequate pilot-vehicle closed-loop gain and phase margins. Pilot-In-the-loop Oscillations have occurred in the T-38A, A-4D, and YF-12 due to abrupt amplitude-dependent changes in air vehicle dynamic response to pilot control inputs. These effects can be of mechanical origin, e.g., bobweights coupled with static friction, or due to saturation of elements within the control system, or due to compensation added to the automatic control system. Other known sources are short-period dynamics (e.g., large ω_{sp} T_{θ_2}), feel system phasing (e.g., effective bobweight location not far enough forward), and sensitive control force and motion gradients. AFFDL-TR-69-72 and NORAIR Rpt NOR-64-143 can furnish some insight.

A very good summary report on PIOs is given in NOR-64-143. Table XXII, from NOR-64-143, lists some known PIO cases and their probable causes for then-current (early 1960s) air vehicles. The causes are equally relevant for modern air vehicles and the lessons learned from these cases are valuable in the prevention of PIOs.

TABLE XXII. Classification of some known PIO cases (from NOR-64-143).

Examples shown as: SPECIES (Air vehicle); Critical Subsystem*; Critical Flight Condition**: Remarks

Class	Type		
	I. Linear	II. Series Nonlinear Elements	III. Subsidiary Feedback Nonlinear Elements
Pitch	IMPROPER SIMULATION; D, V; a: Abnormally high value of $1/T_{\theta_2}$ and low $\zeta_{sp}\omega_{sp}$ led to zero ζ_{sp} when regulating large disturbances. GCA-INDUCED PHUGOID (C-97); D; c, b: Lag from radar-detected error to voice command led to unstable closed-loop phugoid mode. ARM ON STICK (A4D-1, T-38A); F; a: Arm mass increases feel system inertia, leads via B feedback to unstable coupling with short-period dynamics if pilot merely hangs loosely onto stick after a large input.	PORPOISING (SB2C-1); F; c: Hysteresis in stick versus elevator deflection resulted in low-frequency speed and climb oscillations. J. C. MANEUVER (F-86D, F-100C); F, S; a: Valve friction plus compliant cabling resulted in large oscillations at short period. PITCH-UP (XF-104, F-101B, F-102A); V; c: Unstable kink in M_α curve led to moderate-period oscillations of varying amplitudes (depending on extent and nature of the kink) during maneuvers near the critical angle-of-attack. LANDING PIO (X-15); S; b: Closed loop around elevator rate limiting caused moderate oscillations at short-period.	BOBWEIGHT BREAKOUT (A4D-1, T-38A): F, B; a: At high-g maneuvers the bobweight overcomes system friction and reduces apparent damping of the air vehicle in response to force inputs, resulting in large oscillations at short-period. LOSS OF PITCH DAMPER
Lateral-Directional	ω_ϕ/ω_d EFFECT (X-15, NT-33A, F-101B, F-106A, KC-135A, B-58); V; c: Zeros of roll/aileron transfer function are higher than Dutch roll frequency, $\lvert \omega_\phi/\omega_d \rvert > 1.0$, leading to closed-loop instability at low ζ_d conditions. BORESIGHT OSCILLATIONS (F-5A); D, V; c: Spiral roll mode driven unstable if roll information is degraded during gunnery.		LOSS OF YAW DAMPER
Yaw	FUEL SLOSH SNAKING (KC-135A, T-37A); V; c: Fuel slosh mode couples with Dutch roll mode when rudder used to stop yaw oscillation.	TRANSONIC SNAKING (A3D); V, F; a, c: Separation over rudder causes control reversal for small deflections, leading to limit cycle if rudder used to damp yaw oscillations.	
Roll	NONE KNOWN	PILOT-INDUCED CHATTER (F-104B); A; c: Small limit cycle due to damper aggravated whenever pilot attempted to control it.	

* Critical Subsystems:
D = Display
F = Feel system (except B)
B = Bobweight
S = Power servo actuator
V = Vehicle (airframe)
A = Augmentor (damper)

** Critical Flight Conditions:
a = Low altitude, near-sonic Mach
b = Landing approach and takeoff
c = Cruise

VERIFICATION

The need to use high-gain, closed-loop tasks to evaluate handling qualities is fully discussed in the Verification of 5.1.1.1. An additional reason, if any more are needed, is that most of the open-loop design criteria assume a linear system. Pilot evaluation in high-gain, closed-loop tasks is at this time the best evaluation of the effects of nonlinearities. This is particularly important in the evaluation of PIO tendencies because nonlinearities, such as rate limiting, hysteresis, abrupt gain changes, and aerodynamic nonlinearities are some of the common causes of PIO.

Verification shall be by analysis, simulation, and flight test in the performance of the tasks of 5.1.1.1, 5.1.1.2, 5.1.1.3.2, and 5.1.1.3.3 under various levels of atmospheric disturbances. The recommended tasks to demonstrate compliance with this requirement are the tasks described in Verification of 5.1.1.1 using the HQDT technique. AFFTC makes a distinction between HQDT and "operational" closed-loop evaluation tasks. The key element of the HQDT technique is that the pilot must attempt to totally eliminate any error in the performance of the task; he adopts the most aggressive control strategy that he can. Adequate and desired performance objectives are not defined and Cooper-Harper ratings are not recommended. The reason for this is that, in the "operational" tasks, definition of adequate and desired performance encourages the pilot to adopt a control strategy which best meets these performance objectives. In the case of a PIO-prone air vehicle, attempting to totally eliminate any deviation may induce oscillations which reduce task performance, but by accepting small errors (reducing pilot gain) the pilot may be able to avoid these oscillations and still meet the performance objectives (which, by their definition, allow such a tactic). The HQDT technique does not allow the pilot to do this, thus exposing any possible handling qualities deficiencies. HQDT could be considered a "stress test" of handling qualities. For this reason, the HQDT technique is considered the best test of PIO tendencies.

HQDT is not exclusively a PIO evaluation technique. It is a general handling qualities evaluation technique. It is discussed in more detail here in the PIO requirement because it is a better PIO evaluation technique than the "operational" technique. On the other hand, the "operational" technique uses performance objectives more representative of operational use, and the C-H ratings provide a quantitative measure of flying qualities which can be related to the required Levels. Therefore, use of both techniques is recommended in the flight test evaluation, as well as parameter identification techniques and capture tasks. As mentioned in the Verification of 5.1.1.1, the recommended parts of the handling qualities evaluation are: 1) steps, doublets, and frequency sweeps for parameter identification and comparison to open-loop requirements, 2) capture tasks for pilot familiarization with air vehicle dynamic response and evaluation of gross acquisition, 3) HQDT for initial handling qualities and PIO evaluation (HQDT may also provide good inputs for frequency-domain analysis), and 4) “operational” tasks for handling qualities evaluation with C-H ratings.

The PIO tendency classification scale shown on figure 19 has been developed specifically for evaluation of PIO tendencies. It can be used with either the HQDT or the "operational" techniques. Comparing the PIO rating descriptions with descriptions of Levels of flying qualities, a rough approximation would be: Pilot-In-the-loop Oscillations ratings of 1 or 2 would be Level 1, PIO ratings of 3 or 4 would be Level 2, and a PIO rating of 5 would be Level 3. A PIO rating of 6 would be extremely dangerous.

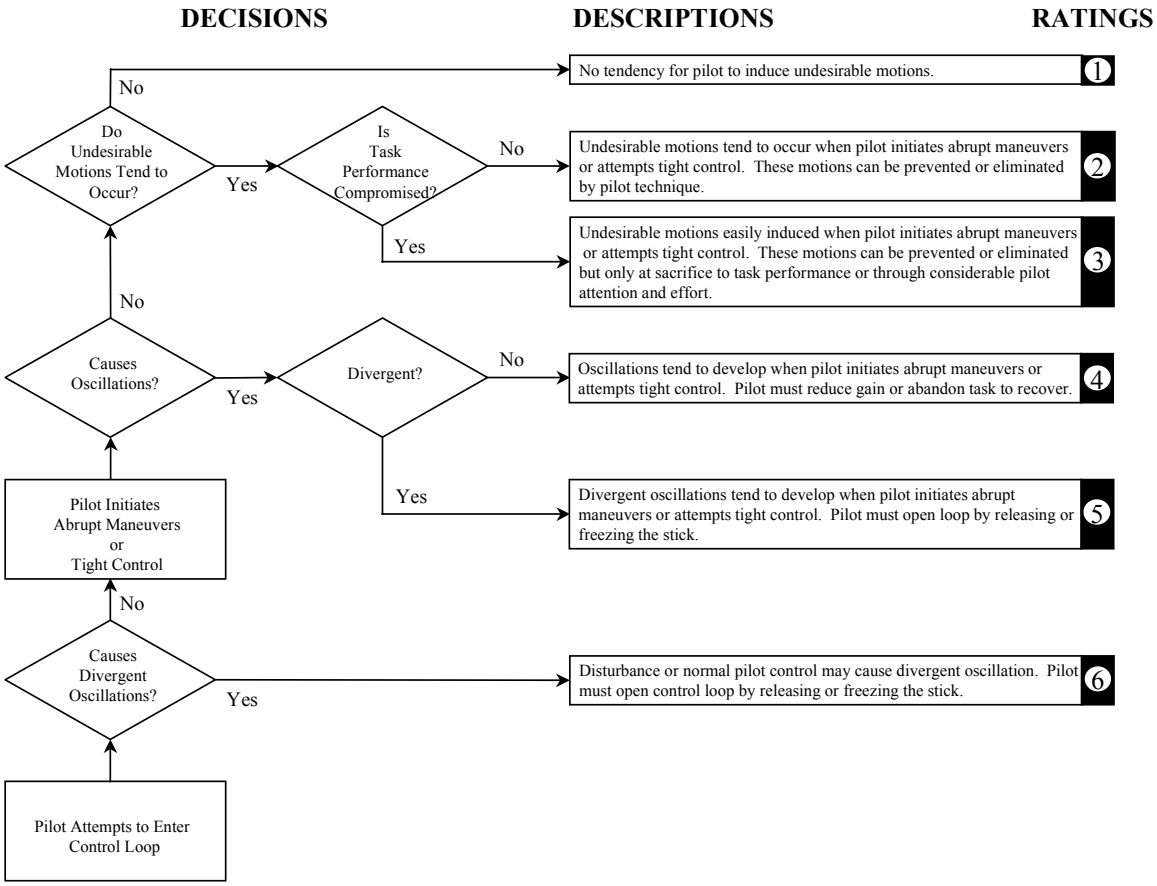


FIGURE 19. PIO tendency classification.

Ground-based simulation may or may not show up any PIO tendencies. Flight evaluation in variable-stability air vehicles is a valuable tool. Final determination will come from flight test of the actual vehicle.

Tom Twisdale provides some guidance on possible HQDT tasks:

> "Probably any test maneuver that allows the evaluation pilot to aggressively and assiduously track a precision aim point is a suitable HQDT test maneuver. In HQDT testing, the test maneuver is not nearly as important as the piloting technique. It is the piloting technique that increases the evaluation pilot's bandwidth and makes possible a good handling qualities evaluation. For this reason there is no exclusive catalog of HQDT maneuvers. The ones discussed below have worked well, but others, perhaps better suited to a particular airplane, may be invented as the need arises.

Air-to-Air HQDT

Air-to-air HQDT involves tracking a precision aim point on a target airplane while using a fixed, or non-computing gunsight. There are three main variations of air-to-air HQDT: constant load factor HQDT at a constant range of about 1500 ft; wind-up turn HQDT at a constant range of about 1500 ft; and HQDT while closing on the target.

The purpose of a constant load factor air-to-air HQDT maneuver is to evaluate handling qualities at a specific angle of attack. The maneuver begins with the test airplane positioned 1500 ft behind and offset above, or below, or to the inside of the target. The offset position is helpful in avoiding jet-wake encounters. At the evaluation pilot's signal the target pilot rolls smoothly into a turn and slowly increases load factor until the test load factor is attained. A g onset rate of two seconds or so per g is satisfactory. When the test load factor has been attained the target pilot calls "on condition" and maintains the turn and the test conditions for the specified period of time, which will depend on the test and analysis objectives. During the load factor build-up the evaluation pilot turns on the airborne instrumentation system and positions the target airplane 50 mils or so from the pipper or aiming index at a clock position of 1:30, 4:30, 7:30, or 10:30. After the target pilot calls "on condition" the evaluation pilot calls "tracking" and drives the pipper toward the precision aim point to initiate the evaluation. The evaluation pilot continues to track while using the HQDT piloting technique, until the target pilot or other aircrew or the control room calls “time.” However, the maneuver is not concluded until the evaluation pilot calls “end tracking.” At that time the target pilot rolls out of the turn.

The constant load factor air-to-air HQDT maneuver may be a constant turn to the left or right, or turn reversals may be included. When reversals are included they shall be performed at constant load factor. The evaluation pilot continues to track the precision aim point throughout the reversal, always using the HQDT piloting technique.

The evaluation pilot shall maintain a 1500-ft separation from the target airplane. Variations of a few hundred feet either way are permissible, but range to the target shall not be allowed to exceed 2000 ft. Range may be determined stadiometrically with adequate accuracy.

The purpose of a wind-up turn air-to-air HQDT maneuver is to quickly explore handling qualities across a range of angle of attack. The maneuver gets under way when the target pilot establishes the test conditions and calls "on condition.” The evaluation pilot positions the test airplane 1500 ft behind and offset above, or below, or to the inside of the target. The offset position is helpful in avoiding jet-wake encounters. The evaluation pilot turns on the airborne instrumentation system and positions the target airplane 50 mils or so from the pipper or aiming index at a clock position of 1:30, 4:30, 7:30, or 10:30. The evaluation pilot then

signals the target pilot to begin the maneuver. The target pilot rolls smoothly into a turn and slowly increases load factor at a g onset rate of five or six seconds per g. As the target airplane begins rolling into the wind-up turn, the evaluation pilot calls "tracking" and drives the pipper toward the precision aim point to initiate the evaluation. The evaluation pilot continues to track while using the HQDT technique, until the target pilot attains the target load factor and calls "target g." The target load factor is maintained until the evaluation pilot calls "end tracking." At that time the target pilot may unload and roll out of the turn.

The evaluation pilot shall maintain a 1500-ft separation from the target airplane. Variations of a few hundred feet either way are permissible, but range to the target shall not be allowed to exceed 2000 ft. Range may be determined stadiometrically with adequate accuracy.

In HQDT with closure, the evaluation pilot slowly closes on the target airplane while tracking. The purpose of the closing HQDT maneuver is to help the evaluation pilot distinguish attitude dynamics from normal and lateral acceleration dynamics. Attitude dynamics are evident at any tracking range, but translation caused by normal and lateral acceleration become more noticeable as the evaluation pilot closes on the target.

In a closing HQDT maneuver the target airplane may either fly straight and level, maneuver gently in pitch and roll, or perform a constant load factor turn. Gently maneuvering or a constant load factor turn is often preferred because it helps to increase the evaluation pilot's bandwidth. In all other respects the closing maneuver is similar to a constant load factor or wind-up tracking turn.

The closing HQDT maneuver can begin once the target pilot has established the test conditions and calls "on condition." The evaluation pilot positions the test airplane 1500 ft behind and above, below, or to the inside of the target; turns on the airborne instrumentation system; and positions the target airplane 50 mils or so from the pipper or aiming index at a clock position of 1:30, 4:30, 7:30, or 10:30. The evaluation pilot then signals the target pilot to begin the maneuver. The target pilot flies straight and level; or begins to maneuver gently and randomly in pitch and roll; or performs a constant load factor turn. The evaluation pilot calls "tracking" and drives the pipper toward the precision aim point to initiate the evaluation. The evaluation pilot continues to track, using the HQDT technique, while slowly closing on the target airplane. The rate of closure will depend on the desired tracking time (which will depend on the test and analysis objectives). The evaluation pilot may find it easier to control the rate of closure if the control room or the target pilot or other aircrew announce the elapsed time in five second increments. At the end of the specified tracking time, the target pilot or other aircrew or the control room calls "time." However the maneuver is not concluded until the evaluation pilot calls "end tracking."

Power Approach HQDT

Power approach HQDT is air-to-air HQDT performed with the test airplane configured for power approach. This maneuver is designed to evaluate approach and landing handling qualities at a safe altitude (10,000 to 15,000 ft), rather than a few feet above the ground during a real landing. Power approach HQDT may be flown with or without closure, however closure is a desirable feature because it helps the evaluation pilot distinguish between attitude and translation dynamics.

The target airplane may either fly straight and level or maneuver gently in pitch and roll. Maneuvering gently is often preferred because it helps to increase the evaluation pilot's bandwidth. In all other respects the power approach HQDT maneuver is similar to a closing HQDT maneuver.

Closure during the maneuver is useful for distinguishing attitude dynamics from normal and lateral acceleration dynamics. Attitude dynamics are evident at any tracking range, but translation caused by normal and lateral acceleration become more noticeable as the evaluation pilot closes on the target.

Jet-wake encounters are a frequent source of difficulty during power approach HQDT testing. Simple geometry, together with a maneuvering target airplane, make jet-wake encounters difficult to avoid. The slow speeds introduce the risk that a jet-wake encounter will precipitate a stall or departure, although this has never occurred. There are two solutions to the problem of jet-wake encounters. One is to use a small propeller-driven airplane as a target. Excellent candidates are the T-34C or Beechcraft Bonanza, or similar airplanes. These airplanes can easily match the slowest speeds of most military airplanes, and they produce very little propwash. The second solution is to use a target that is programmed into a flight test head-up display, similar to the head-up display used on the Calspan NT-33.

Air-to-Ground HQDT

Air-to-ground HQDT involves tracking a precision aim point on the ground with a fixed, or non-computing gunsight. Shallow or steep dive angles may be used. Shallow dive angles approximate strafing attack profiles and steeper angles approximate ballistic weapons delivery profiles.

The evaluation pilot trims the airplane at the specified dive entry altitude and airspeed, turns on the airborne instrumentation system, calls "on condition", and rolls or pitches to the specified dive angle. When the outer ring of the gunsight reticle crosses the precision aim point, the evaluation pilot calls "tracking" and commences to track the precision aim point using the HQDT piloting technique. The evaluation pilot continues to track until the recovery altitude is reached, then calls "end tracking" and recovers from the dive.

A useful variation, on the basic maneuver is to track two or more precision aim points, instead of one. For example, precision aim points may be positioned at each apex of an imaginary isosceles triangle laid out on the ground. This triangle has a base of 100 ft and a height of 375 ft (for 15º dive angles) or a height of 100 ft (for 45º dive angles). During the maneuver the evaluation pilot randomly switches from one precision aim point to another, perhaps at a signal from the control room.

Boom Tracking HQDT

In boom tracking, the evaluation pilot tracks the nozzle on an aerial refueling boom. This maneuver is designed to explore aerial refueling handling qualities without the risk of close proximity to a tanker and a refueling boom.

The tanker airplane establishes the test conditions of Mach number (or airspeed) and altitude and maintains them during the test maneuver. The boom operator positions the refueling boom at 0º azimuth and a midrange elevation angle. When the test conditions have been established the tanker pilot or the boom operator call "on condition." The evaluation pilot moves the test airplane into position a short distance behind the nozzle (20 to 50 ft) and positions the nozzle about 50 mils from the pipper or aiming index at a clock position of 1:30, 4:30, 7:30, or 10:30. To begin the maneuver, the evaluation pilot turns on the airborne instrumentation system, calls "tracking", and drives the pipper toward the nozzle. The evaluation pilot continues to track the nozzle, using the HQDT piloting technique, while the boom operator randomly maneuvers the refueling boom in azimuth and elevation. The boom motion shall be a combination of gentle and abrupt changes in rate and position. After the specified period of tracking time (which will depend on the test and analysis objectives) has elapsed, the control room or another crew member calls "time." The maneuver is not concluded, however, until the evaluation pilot calls "end tracking."

Formation HQDT

In formation HQDT, the evaluation pilot attempts to maintain a precisely defined position relative to the lead airplane during a series of gentle maneuvers. Properly done, formation HQDT can highlight for the evaluation pilot the vertical and lateral translation dynamics of the test airplane. This maneuver is also useful for evaluating the throttle response of the airplane. Care must be taken not to force the evaluation pilot to fly too close to the lead airplane. Close proximity can increase bandwidth, but too close proximity can reduce it. As the separation between airplanes narrows, good and prudent pilots will reduce their bandwidth to reduce the risk of collision."

The existence of a PIO tendency is difficult to assess. Attention to flying qualities per se during flight control design will take care of many potential problems. Pilot-In-the-loop Oscillations may occur early in the air vehicle life as on the YF-16 high-speed taxi test that became airborne

before its first scheduled flight, or later in service, as with the T-38 as more pilots flew it. If PIO is not found readily, it shall be sought during the flight test program. High-stress tasks such as approach and landing with a lateral offset, air-to-air tracking, air-to-ground tracking, or terrain-following may reveal PIO proneness.

Depending upon the cause, ground-based simulation may or may not prove a useful investigation technique—often it does not. PIOs observed in flight are often not obtained in ground-based simulators, even simulators with some motion. In a number of cases, optimization of p/F_{as} in a fixed-base simulator has resulted in gross oversensitivity in actual flight.

5.2.1.7 Residual oscillations.
Any sustained air vehicle residual oscillations in Calm air shall not interfere with the pilot's ability to perform all flight phase tasks required in service use of the air vehicle. For Levels 1 and 2, with pitch control fixed and with it free, any sustained residual oscillations in Calm air shall not exceed __(1)__ in normal acceleration at the pilot station for any Flight Phase.

REQUIREMENT RATIONALE

The requirement prohibits limit cycles in the control system or structural oscillations that might compromise tactical effectiveness, cause pilot discomfort, etc.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 through 3.3.11.1.2 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.2.1.3 and 3.3.1.1
MIL-STD-1797: 4.1.11.7 and 4.2.3.

The recommended value for blank (1) is ±.02 g. Given the proper data, this threshold could be made a function of frequency in order to correspond more closely with human perception.

REQUIREMENT LESSONS LEARNED

Former MIL-STD-1797 decreased the allowable normal acceleration oscillations to ±0.02 g from the ±0.05 g of MIL-F-8785C. This is based on flight test experience with the B-1 (AFFTC-TR-79-2), which encountered limit cycle oscillations during aerial refueling, subsonic and supersonic cruise. A primary contributor was identified to be mechanical hysteresis in the pitch system. According to AFFTC-TR-79-2, "Flying qualities were initially undesirable due to this limit cycle." Normal acceleration transients were about 0.05 to 0.12 g, as figure 20 shows. The limit cycle was eliminated by installation of a mechanical shaker (dither) vibrating at 20 Hz.

Residual oscillations are limit cycles resulting from nonlinearities such as friction and poor resolution. Negative static stability will contribute and low damping may augment the amplitude. Thus high speed, high dynamic pressure, or high altitude may be critical. Residual oscillations are most bothersome in precision tasks.

Likely causes are flight control system nonlinearities such as valve friction or, especially in unpowered flight control systems, control system friction, or hinge-moment nonlinearities. The X-29A, an unstable basic air vehicle, exhibited very noticeable control-surface activity during ground roll. This was a result of a compromise which kept the stability and control augmentation active on the ground in order to assure flight safety in the event of bouncing or early lift-off.

VERIFICATION

Verification shall be by analysis and by simulation or flight test in the performance of the tasks of 5.1.1.1, 5.1.1.2, 5.1.1.3.2, and 5.1.1.3.3 in Calm air. These tasks shall be flown by test pilots at specific flight conditions throughout the ROSH and ROTH. The specific flight conditions to be evaluated shall be the most common operating conditions, any operating conditions critical to the mission of the air vehicle, and any conditions determined by analysis or simulation to be worse than the Level of flying qualities required in 5.1.1. Proof of compliance in these demonstration tasks shall consist of time histories of normal acceleration, pilot comments, and C-H ratings. The comments and ratings shall indicate that the flying qualities are no worse than the required Level of flying qualities for each combination of Air Vehicle State and Flight Phase.

Limit cycle amplitude depends on characteristics of the actual hardware and software, and so may be different in simulations than in actual flight. Measurements of normal acceleration at the pilot's station shall be made in the course of test flight.

Any residual oscillations should become manifest during expansion of the flight envelope, if they have not already been discovered during ground simulations with flight hardware. Flight or ground-roll conditions with high stability-augmentation gains may be critical.

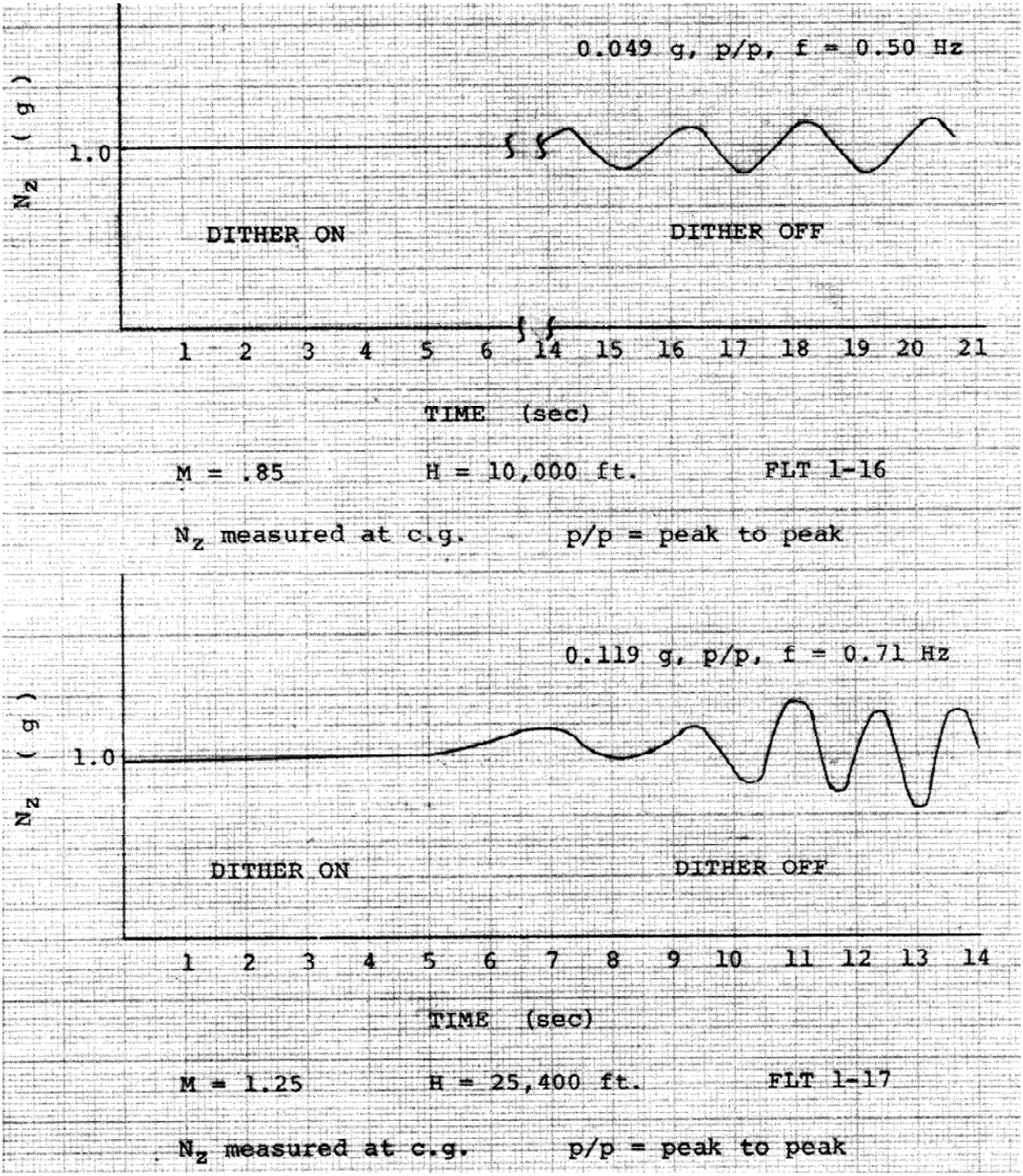


FIGURE 20. Effect of dither on B-1 limit cycle oscillations (from AFFTC-TR-79-2).

5.2.1.8 Ride qualities.
The following short-term and applicable long-term vertical and lateral axis Ride Discomfort Index levels shall not be exceeded at any crew station during flight in the turbulence levels specified in table XXIII.

TABLE XXIII. Ride Discomfort Index limits.

Flight Phase	RMS Turbulence Intensity	Exposure Time	Maximum Ride Discomfort Index
Terrain Following	Up to 1,000 ft AGL σ_w = 3.5 ft/sec, σ_v = 4.0 ft/sec, σ_u = _____	Over 3 hours	
		From 1.5 to 3 hours	
		From 0.5 to 1.5 hours	
	Up to 1,000 ft AGL σ_w = 7.0 ft/sec, σ_v = 8.0 ft/sec, σ_u = _____	Less than 0.5 hours	
Normal Flight, Climb, Cruise, and Descent	Up to 7,000 ft $\sigma_w = \sigma_v = (1.75\text{x}10^{-8})h^2 - (6.05\text{x}10^{-4})h + 3.40$ σ_u = _____ Above 7,000 ft $\sigma_w = \sigma_v = 0.0$, σ_u = _____	Over 3 hours	
		From 1.5 to 3 hours	
		From 0.5 to 1.5 hours	
	Up to 10,000 ft $\sigma_w = \sigma_v = (-4.96\text{x}10^{-8})h^2 - (4.26\text{x}10^{-4})h + 6.40$ σ_u = _____ From 10,000 ft to 37,000 ft $\sigma_w = \sigma_v = (-2.10\text{x}10^{-4})h + 7.82$ σ_u = _____ Above 37,000 ft $\sigma_w = \sigma_v = 0.0$, σ_u = _____	Less than 0.5 hours	

REQUIREMENT RATIONALE

The intent of this requirement is to specify the ride experienced by the crew. Soft seats or other isolation techniques used should be considered in meeting this requirement. Care must be taken that relative motion between the crew member and his controls and instruments, resulting from isolation techniques, does not degrade crew performance. Visual problems, for example, can be aggravated by relative motion.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A. This requirement corresponds to 3.1.2.12 and 3.1.2.12.1 in MIL-F-9490D.

Recommended values for table XXIII follow.

Flight Phase	RMS Turbulence Intensity	Exposure Time	Maximum Ride Discomfort Index
Terrain following	Up to 1,000 ft AGL σ_w = 3.5 ft/sec, σ_v = 4.0 ft/sec, σ_u = 0.0	Over 3 hours	0.10
		From 1.5 to 3 hours	0.13
		From 0.5 to 1.5 hours	0.20
	Up to 1,000 ft AGL σ_w = 7.0 ft/sec, σ_v = 8.0 ft/sec, σ_u = 0.0	Less than 0.5 hours	0.28
Normal flight, climb, cruise, and descent	Up to 7,000 ft $\sigma_w = \sigma_v = (1.75x10^{-8})h^2 - (6.05x10\text{-}4)h + 3.40$ σ_u = 0.0 Above 7,000 ft $\sigma_w = \sigma_v = \sigma_u = 0.0$	Over 3 hours	0.10
		From 1.5 to 3 hours	0.13
		From 0.5 to 1.5 hours	0.20
	Up to 10,000 ft $\sigma_w = \sigma_v = (-4.96x10^{-8})h^2 - (4.26x10^{-4})h + 6.40$ σ_u = 0.0 From 10,000 ft to 37,000 ft $\sigma_w = \sigma_v = (-2.10x10^{-4})h + 7.82$, σ_u = 0.0 Above 37,000 ft $\sigma_w = \sigma_v = \sigma_v = 0.0$	Less than 0.5 hours	0.28

Longitudinal gusts should be considered for some STOL applications.

REQUIREMENT LESSONS LEARNED

There are several sources of literature available on ride smoothing. AIAA 72-772 describes development of the B-1 ride smoothing system which was synthesized using the Identical Location of Accelerometer and Force (ILAF) concept. AFFDL-TR-65-190, AIAA 66-999, AIAA 68-1067, Wykes, J. H., Proceedings of AGARD 34th Flight Mechanics Panel, and Mori, A., Proceedings of NAECON 1972, describe the development of the ILAF concept. NASA CR-2158 describes a study of modal suppression on the YF-12A air vehicle.

The production B-1 ride smoothing system used a vertical long-term index near 0.10. The lateral B-1 requirement was more stringent. Commercial feasibility studies have used much more conservative design goals. NASA CR-2276, for example, used an unweighted index of 0.03 in 0.01 turbulence. This is equivalent to an unweighted index of 0.015 in 0.20 turbulence at low level, and is roughly a factor of 10 more stringent than the criterion recommended here. Of course, for specific procurements it may be necessary to specify different values of the Ride Discomfort Index based on unique mission requirements.

The B-52 was known for its marginal ride during low-level penetrations. When compared to this long-term criteria (3 hours = 0.10), the B-52 exceeds the criterion for medium and light gross

weights and satisfies the criterion for heavier gross weights. Thus, for the initial penetration flight phase the B-52 ride is acceptable. For later phases the ride is unacceptable if the remaining low-level phase exceeds 3 hours.

The figure 21 acceleration weighting functions are based on the MIL-STD-1472 human sensitivity curves, ISO 2631, as extrapolated to lower frequencies by an Aerospace Medical Research Laboratory (AMRL) Memo, 8 July 1974. The extrapolations below 1.0 Hz, especially for lateral vibration, are supported by a minimum of data. However, the values defined represent the best consensus of experts within the 6750th Aerospace Medical Research Laboratory, and reflect the current US recommendation to the International Organization for Standardization for human exposure to vibration from 0.1 to 1.0 Hz. The weighting functions defined are truncated at 0.1 Hz and at high frequencies.

The reason for weighting function truncation is the limitations of test equipment used to generate data upon which these curves are based. Moving-base simulators can be used to simulate air vehicles at low frequencies; however, the data obtained below 0.1 to 0.2 Hz is of questionable value since continuous oscillations at these frequencies do not normally occur in flight. In many cases, the pilot or Automatic Flight Control System (AFCS) will control low frequency motions, effectively smoothing these oscillations and reducing the truncation error resulting from this approach (see Hoblet, F. M., 37th Meeting of AGARD Structures and Materials Panel, 1973). Note that attitude-hold or other pertinent modes are to be simulated to satisfy this requirement. These modes shall approximate pilot or AFCS suppression of low-frequency responses. Truncation at higher frequencies is permitted since the gust spectrum has very little power beyond 30 Hz. Since structural modes seldom extend to 30 Hz, in practice integration is normally stopped near the frequency of the highest aeroelastic mode modeled. The requirement is to include significant effects to the truncation frequency. Due to gust filter roll-off, integration beyond 15 Hz seldom affects the integral value significantly.

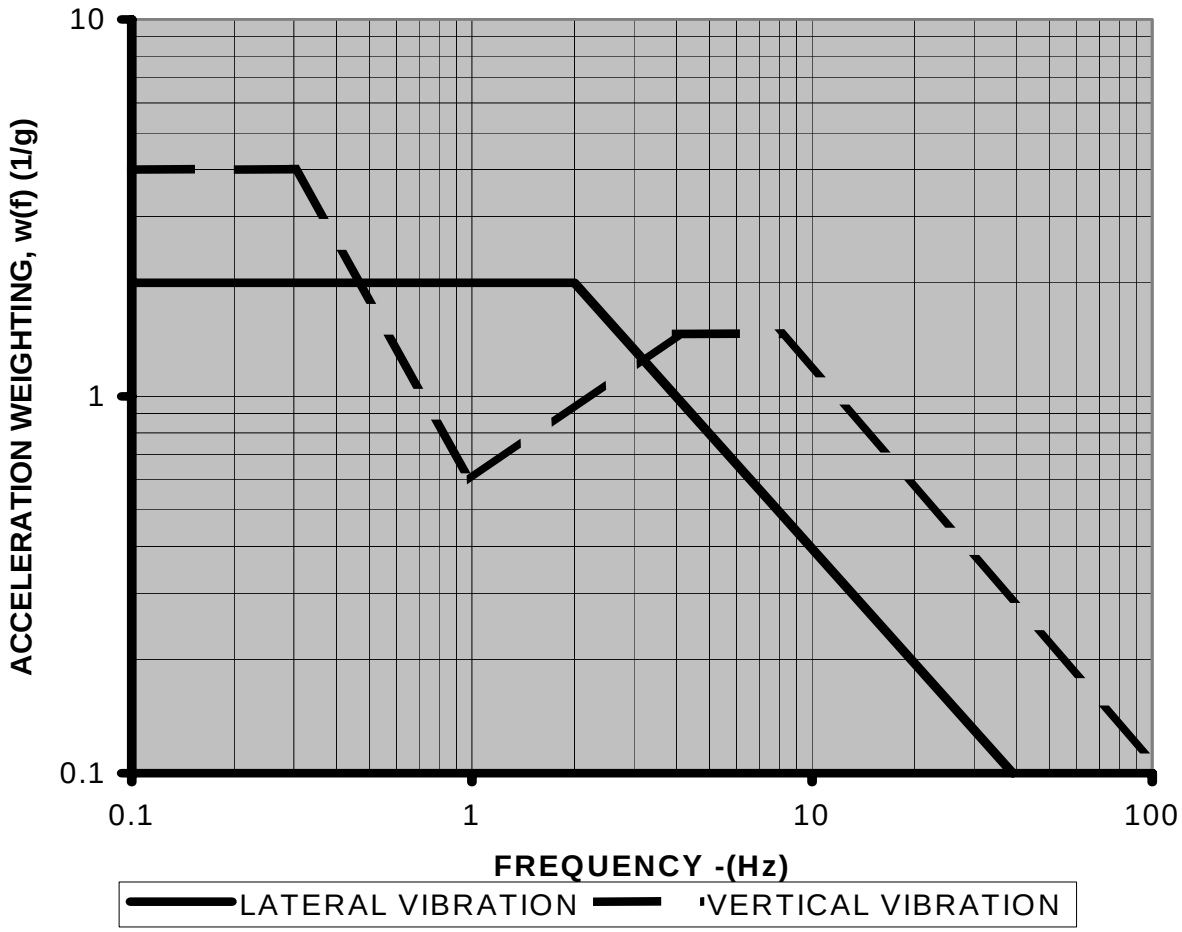


FIGURE 21. Acceleration weighting functions.

VERIFICATION

Ride Discomfort Index is defined as:

$$D_i = \left[\int_{0.1}^{f_t} |w(f)|^2 \, |T_{cs}(f)|^2 \, \Phi_{u_g}(f) \, df \right]^{1/2}$$

where:

$w(f)$ = Acceleration weighting function (vertical or lateral) 1/g

$T_{cs}(f)$ = Transmissibility, at crew station, g/ft/sec

$\Phi_{u_g}(f)$ = Von Karman gust power spectral density of intensity (form specified in 5.1.2)

f = Frequency, Hz

f_t = Truncation frequency beyond which aeroelastic responses are no longer significant in turbulence

Acceleration weighting functions are defined for vertical and lateral acceleration by figure 21. Probability of exceedance versus turbulence intensity is specified in the recommended values for table XXIV.

TABLE XXIV. RMS gust intensities for selected cumulative exceedance probabilities, ft/sec TAS.

Flight Segment	Altitude (FT-AGL)	Probability of Exceedance		
		2×10^{-1}		10^{-2}
Terrain Following	Up to 1000 (lateral)	4		8
	Up to 1000 (vertical)	3.5		7
Normal Flight Climb Cruise and Descent	500	3.2		6.6
	1,750	2.2		6.9
	3,750	1.5		7.4
	7,500	0		6.7
	15,000	0		4.6
	25,000	0		2.7
	35,000	0		0.4
	45,000	0		0
	55,000	0		0
	65,000	0		0
	75,000	0		0
	Over 80,000	0		0

The requirements apply, separately, to each of the vertical and lateral axes. For the lateral axis requirement only lateral gusts apply and for vertical acceleration only vertical gusts apply. Effects of altitude hold or other pertinent AFCS modes shall be included where used.

Ride requirements are stated in terms of probabilities, since the ride discomfort addressed by this requirement is generated by random turbulence. The exceedance probabilities and corresponding Ride Discomfort Index values specified are based on the recommendations of ASD-TR-70-18

and ASD-TR-72-64. Generally these requirements should provide ride quality equal to or better than that existing in currently operating air vehicles within the USAF inventory.

The turbulence intensities to be used are determined by the exceedance probabilities specified for Ride Discomfort Index. Generally, the system is required to reduce ride discomfort to the levels specified while flying in turbulence with a cumulative exceedance probability equal to or less than the probability specified. System nonlinearities must be considered. System dead zone and other nonlinearities must not be so large that ride discomfort exceeds the 0.10 or other pertinent long-term limits in Common turbulence. System saturation must not be so severe in turbulence at the 0.01 exceedance level that the 0.28 ride discomfort limit is exceeded. The designer should note that the cumulative exceedance probabilities for turbulence are stated in terms of stationary probabilities rather than the nonstationary probabilities used in reliability work. Turbulence exceedance probabilities are tabulated in table XXIV.

A stationary probability or cumulative probability of exceedance for turbulence encounter means that at a randomly selected time during flight, the probability of being in turbulence at or above the stated intensity is of a given value. This does not define the probability of exceeding a given level of turbulence during a given flight or flight segment. On a fleet lifetime basis, this probability can be interpreted as the portion of total flight time to be spent above the stated intensity. Since the statistics upon which these probabilities are based were measured over extended operating times, the temptation to convert these values to hours per hour or hours per individual flight should be resisted.

The levels of ride discomfort specified are based on short-term tolerance and long-term tolerance. Data from ASD-TR-70-18, ISO 2631, and NASA TM-X-2620, indicate that below a ride discomfort index of 0.07, little or no degradation in crew performance or passenger comfort is expected. Above a ride discomfort index of 0.28 the USAF references indicate crew action must be initiated to reduce the acceleration environment by changing flightpath, altitude, and/or airspeed.

There is disagreement in the literature on the proper approach for evaluating combined axis accelerations. ASD-TR-70-18 recommends a method for evaluating combined axis accelerations based on USAF experience. ISO 2631, the ISO standard, recommends that accelerations in separate axes be considered separately; and Boeing D3-7600-14, a commercial air vehicle study, recommended another method for combined axis acceleration evaluation.

Due to the lack of agreement on method and limited test data available on combined axis accelerations, this requirement follows the ISO recommendation and places requirements only on vertical and lateral axis accelerations, separately. The designer should note that vertical ride discomfort is to be evaluated due to vertical axis turbulence only and lateral ride evaluated due to lateral turbulence only. No requirement is specified for roll gusts or longitudinal gusts, although longitudinal gusts should be considered for some STOL applications.

5.2.2 Longitudinal flying qualities requirements

5.2.2.1 Longitudinal response to the pitch controller.

The longitudinal response of the air vehicle to the pitch controller shall meet the requirements of 5.2.2.1.1 through 5.2.2.1.8.8.

REQUIREMENT RATIONALE

Pitch control of air vehicles is a vital element of flying qualities, both as a primary control axis (for example, in pointing the aircraft during gunnery) and as an indirect way of controlling the aircraft flight path (for example, in glide path control for landing). Even though control of the flight path may be the pilot's ultimate aim, pitch attitude control is commonly used as a surrogate.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 through 3.3.11.1.2 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2
MIL-STD-1797: 4.2.

It is strongly suggested several or all of the longitudinal response requirements of 5.2.2.1 are used in an aircraft design.

REQUIREMENT LESSONS LEARNED

While normally force is considered the primary cue of pilot feel, both control force and control position are significant. With a deflection control system, force feel will lag deflection because of the finite bandwidth of the feel system. Where a pilot force pick-off (say, a strain gauge) is used, any significant deflection of the pilot control is effected through a follow-up. It has been found that a displacement control should not be penalized unduly for the feel system lag, since pilots sense both stick input and stick output and can compensate. The response requirements are to be applied to pilot control inputs accordingly.

VERIFICATION

Verification shall be via verification of requirements 5.2.2.1.1 through 5.2.2.1.8.8. As the design progresses, analysis, simulation, and flight test will be appropriate demonstration means. In the end, flight test will measure some parameters and collect data to validate analytical and simulator results for the rest.

Construction of frequency responses for matching or plotting is conveniently performed by linearizing the high-order system (for all possible input amplitudes, if necessary). The linearized high-order model is almost always available because it is used in the design process. If it is not, and, for example, a flight control element is to be changed on an existing system, and relinearization is not feasible, then fast Fourier analysis of a nonlinear simulation model of the

system works well. The exact linearized model must account for the lower-frequency effects of structural filters, aliasing, etc. in order to faithfully represent the response in the frequency range of primary interest to flying qualities. Fast Fourier analysis of real-time or non-real-time simulations of the aircraft is best performed using responses to a stick input with wide frequency content. Background can be found, for example, in "Applied Time Series Analysis, Vol. I, Basic Techniques."

AIAA Paper 80-1611-CP, AFFTC-TR-77-27, AIAA Paper 80-1633, and AFWAL-TR-81-3027 describe fast Fourier reduction of flight data. AFFTC-TR-77-27 describes AFFTC experience with the method. AIAA Paper 80-1633 discusses use of an electronically-generated frequency sweep which worked adequately, and AIAA Paper 80-1611-CP shows that FFT can work adequately even when the test condition is theoretically least suited to the method. AFWAL-TR-81-3027 shows a pilot-generated frequency sweep that worked very well.

5.2.2.1.1 Longitudinal lower-order equivalent system dynamics.
The dynamics of a lower-order equivalent system match of the air vehicle response to the pitch controller shall meet the requirements of 5.2.2.1.1.1 and 5.2.2.1.1.2.

REQUIREMENT RATIONALE

The advent of high-order feedback control systems has required specification of flying qualities parameters for systems of, for example, twentieth order or more. The use of lower-order equivalent systems allows us to extend to many higher-order systems the application of well-established boundaries generated from classical airplane data; i.e., MIL-F-8785C.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 through 3.3.11.1.2 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.1.1, 3.2.1.2, 3.2.2.1 through 3.2.2.1.2, and 3.5.3
MIL-STD-1797: 4.2.1 through 4.2.1.2.

REQUIREMENT LESSONS LEARNED

This concept, originally developed about thirty years ago (AFFDL-70-74 and Systems Technology Inc. TR-190-1), has been refined considerably (for example, see AIAA Paper 77-1122, AIAA Paper 79-1783, MDC Rpt A6792, AIAA Paper 80-1611-CP, AIAA Paper 80-1627-CP, "Analysis of High Order Aircraft/Flight Control System Dynamics Using an Equivalent System Approach", "Handling Qualities of Aircraft with Stability and Control Augmentation Systems – A Fundamental Approach", "Equivalent Systems Approach for Flying Qualities Specification", and MDC Report A5596). This approach has the advantage of including the effects of adjacent modes, it is easily related to previous specifications, and identifies important delay effects. In AFWAL-TR-82-3064, Gentry describes matching techniques, Hodgkinson and Wood discuss the effects of mismatch, and Bischoff shows how to allow for the higher-order system effect of reducing the maximum pitching acceleration for a step input.

Although experience has found that the existing data base may be extended by means of the equivalent system concept to cover a number of more elaborate stability augmentation applications, problems remain. Of course there are bound to exist some configurations that just don't fit. But even among the ones that do, several questions remain unresolved, including:

- Effects of pilot location and blended direct lift control have been observed and need to be accounted for
- A continuing controversy over the merits of n/α versus $1/T_{\theta_2}$ has not yet been resolved
- Various forms of Birhle's Control Anticipation Parameter (CAP) relating initial pitch acceleration to steady-state normal acceleration have been tendered
- Tolerable values of effective time delay need to be pinned down more fully
- Task dependence needs to be explored more fully
- There are those who prefer, or insist, upon a time domain form of criteria (in the world of linear systems a duality exists between time and frequency responses, so the question becomes just what details the requirements should emphasize).

Nevertheless, the lower-order equivalent system approach not only fits the data of MIL-F-8785B/C but has demonstrated its effectiveness for a number of highly-augmented aircraft as well as for classical response. For cases with no good equivalent system match, frequency-response criteria involving bandwidth and frequency response envelopes of the actual higher-order system should be used (5.2.2.1.3).

There is currently insufficient data to place definitive requirements on mismatch (cost) between the HOS and LOES. A brief NT-33 landing approach simulation tackled the question of mismatch (AIAA Paper 79-1783 and AFWAL-TR-81-3116). High-order systems and simulations of their low-order equivalents were flown. The experiment indicated that very large mismatch proved unnoticeable to pilots (a sum of squares mismatch around 200 in the frequency range of $0.1 \leq \omega \leq 10$ rad/sec compared to the previous arbitrary limit of 10). MDC Rpt A6792 offers a theory to explain the adequacy of such large mismatches. By examining pilot rating differences between pairs of configurations in previous NT-33 data (AFFDL-TR-78-122 and AFFDL-TR-70-74), frequency response envelopes were derived. Each pair of configurations consisted of an unaugmented, low-order aircraft response and a corresponding high-order system formed by adding terms to the low-order response. The basic aircraft dynamics were modified via equalization networks. Modifications that resulted in a pilot rating difference no greater than 1 between the low-order system and the corresponding high-order system were defined as maximum unnoticeable added dynamics. Figure 22 shows the envelopes that were derived. Some rough guidance is available from these envelopes, which are an approximate measure of maximum unnoticeable added dynamics.

As would be expected, the pilots were most sensitive to changes in the dynamics in the region of crossover (1-4 rad/sec). Mismatches between the HOS and LOES in excess of the values shown in the figure 22 envelopes would be cause to suspect that the equivalent parameters may not accurately predict pilot opinion. In such cases it is recommended that criteria applicable directly to the actual system, such as some of those in 5.2.2.1.3, be used in place of the equivalent system form.

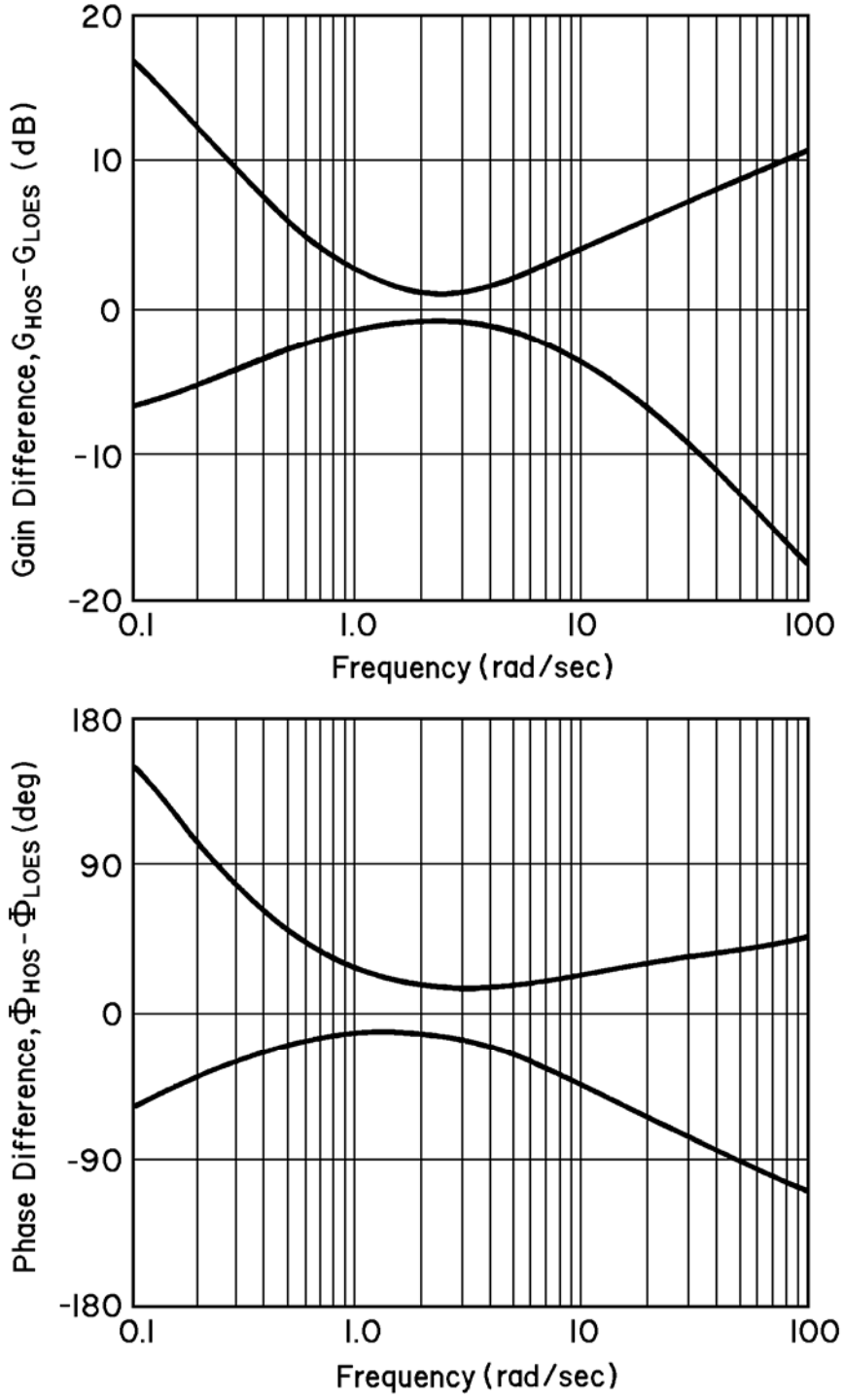


FIGURE 22. **Envelopes of maximum unnoticeable added dynamics (AFWAL-TR-82-3064).**

The mismatch envelopes are shown as a guide to determining whether a match is acceptable. The envelopes are defined as functions of the Laplace variable, s, as follow:

Upper Gain Envelope: $\dfrac{3.16\,s^2 + 31.61\,s + 22.79}{s^2 + 27.14\,s + 1.84}$

Lower Gain Envelope: $\dfrac{0.095\,s^2 + 9.92\,s + 2.15}{s^2 + 11.6\,s + 4.95}$

Upper Phase Envelope: $\dfrac{68.89\,s^2 + 1100.12\,s - 275.22}{s^2 + 39.94\,s + 9.99}\,e^{0.006\,s}$

Lower Phase Envelope: $\dfrac{475.32\,s^2 + 184100\,s + 29460}{s^2 + 11.66\,s + 0.039}\,e^{-0.0072\,s}$

These envelopes should be used only after the matching process has been performed. Normally, mismatches will be far smaller than those the envelopes allow.

Additional comments on the use of equivalent systems may be found in AGARD-CP-260 and "Low-Order Approaches to High-Order Systems: Problems and Promises." The influence of mid-frequency added dynamics on LOES was discussed in "Low-Order Approaches to High-Order Systems: Problems and Promises", where it was shown that a series of (possibly unrealistic) lead-lag combinations evaluated in the Neal-Smith in-flight evaluation (AFFDL-TR-70-74) produced LOES parameters which are not necessarily equivalent to their classical counterparts. Of the ten configurations with added lead-lag dynamics, only five are predicted accurately. For the five that failed, the equivalent dynamics (ζ_{sp_e}, ω_{sp_e}, τ_{θ_e}) were predicted to have Level 1 flying qualities but were rated Level 2 by the pilots. Table XXV lists the dynamics of the HOS and LOES for these configurations. With the exception of Configuration 1C, all have $\zeta_{sp_e} < 0.5$ (though still greater than 0.35). Three have $\tau_{\theta_e} = 0$. All but 1C have a first-order lag near the short period frequency; 1C has a first-order lead near ω_{sp}.

TABLE XXV. Lead-lag configurations with Level 1 LOES and Level 2 pilot ratings.

Conf	HOS						LOES			Pilot Rating	
	$1/T_{\theta_2}$	ζ_{sp}	ω_{sp}	$1/T_1$	$1/T_2$	ω_3	ζ_{sp_e}	ω_{sp_e}	τ_{θ_e}	M	W
1A	1.25	0.69	2.2	0.5	2.0	63	0.39	3.14	0	6, 4	5
1C	1.25	0.69	2.2	2.0	5.0	16	0.67	3.02	0.079	3, 5, 5	4
2A	1.25	0.70	4.9	2.0	5.0	63	0.46	5.96	0	4.5	4
2B	1.25	0.70	4.9	2.0	5.0	16	0.42	5.67	0.059	6, 6	4, 5
7A	2.5	0.79	7.3	3.3	8.0	63	0.44	8.23	0	4, 5	2

NOTES:
1. HOS from Neal-Smith (AFFDL-TR-70-74); LOES from McDonnell Aircraft Company (MDC Rpt A6792)
2. Equivalent dynamics are Level 1 on MIL-F-8785C limits.
3. T_1 is an added lead; T_2 a lag; ω_3 an actuator.

figure 23 and figure 24 show the effects of the added lead-lag combinations on these configurations. The net effect is an apparent hump around ω_{sp} characterized in the LOES match by a low equivalent damping ratio (table XXV). The low-order form has no other way to match a hump in the amplitude plot. Similar effects are seen in the phase angle: on figure 24, the humps appear as phase lead (since, for the basic configurations, $\tau_{\theta_e} = 0$). In fact, figure 24 shows that an LOES match over the frequency range of 0.1-10 rad/sec would produce $\tau_{\theta_e} < 0$ (if negative time delays were allowed) for Configurations 1A, 2A, and 7A. The small positive τ_{θ_e} for Configurations 1C and 2B results from the relatively low frequency of the second-order lag (ω_3) for these cases, 16 rad/sec as opposed to 63 rad/sec.

There are two potential methods to accommodate lead-lag systems like those of table XXV; unfortunately, neither is physically very appealing. And in each, there is an underlying question as to the universality of the equivalent systems approach.

a. Redefine limits on ζ_{sp_e}

If the minimum ζ_{sp_e} for Level 1 were increased from 0.35 to 0.50, four configurations in table XXV would fit the requirements (ignoring Configurations 1C, for which none of this discussion is applicable). But restricting unaugmented vehicles as well is not appealing as lower ζ_{sp_e} is very well supported by flight test data for classical aircraft. The alternative to specify two sets of requirements—one for unaugmented aircraft and another for augmented aircraft is especially unattractive, since this is tacit admission that equivalent systems is a misnomer. Additionally, it presents the problem of defining the specific level of augmentation at which the requirements would change over. For example, should addition of a simple high-frequency stick filter (whose only major effect is to increase τ_{θ_e}, figure 25) change the requirement? In fact, the problem with Configuration 1A, 2A, 2B, and 7A is directly traceable from pilot commentary to over-abruptness and apparently has nothing to do with damping ratio at all. The high gain at high frequency depicted on figure 23 would indeed be expected to lead to an abrupt response. If for some unforeseen reason a flight control system designer would ever suggest equalization that would produce such humps in the frequency response, use of a frequency-response criterion is suggested. Four of the five configurations of table XXV fit the bandwidth requirement (see figure 43).

b. Redefine τ_{θ_e}

As mentioned above, three of the four low-ζ_{sp_e} violators of table XXV also have $\tau_{\theta_e} = 0$. As figure 24 suggests, a better LOES fit is obtained for these three cases if τ_{θ_e} is allowed to be less than zero. Specifically, negative time delays found for these cases are as follow:

Configuration 1A -- τ_{θ_e} = -0.004 sec
Configuration 2A -- τ_{θ_e} = -0.008 sec
Configuration 7A -- τ_{θ_e} = -0.014 sec .

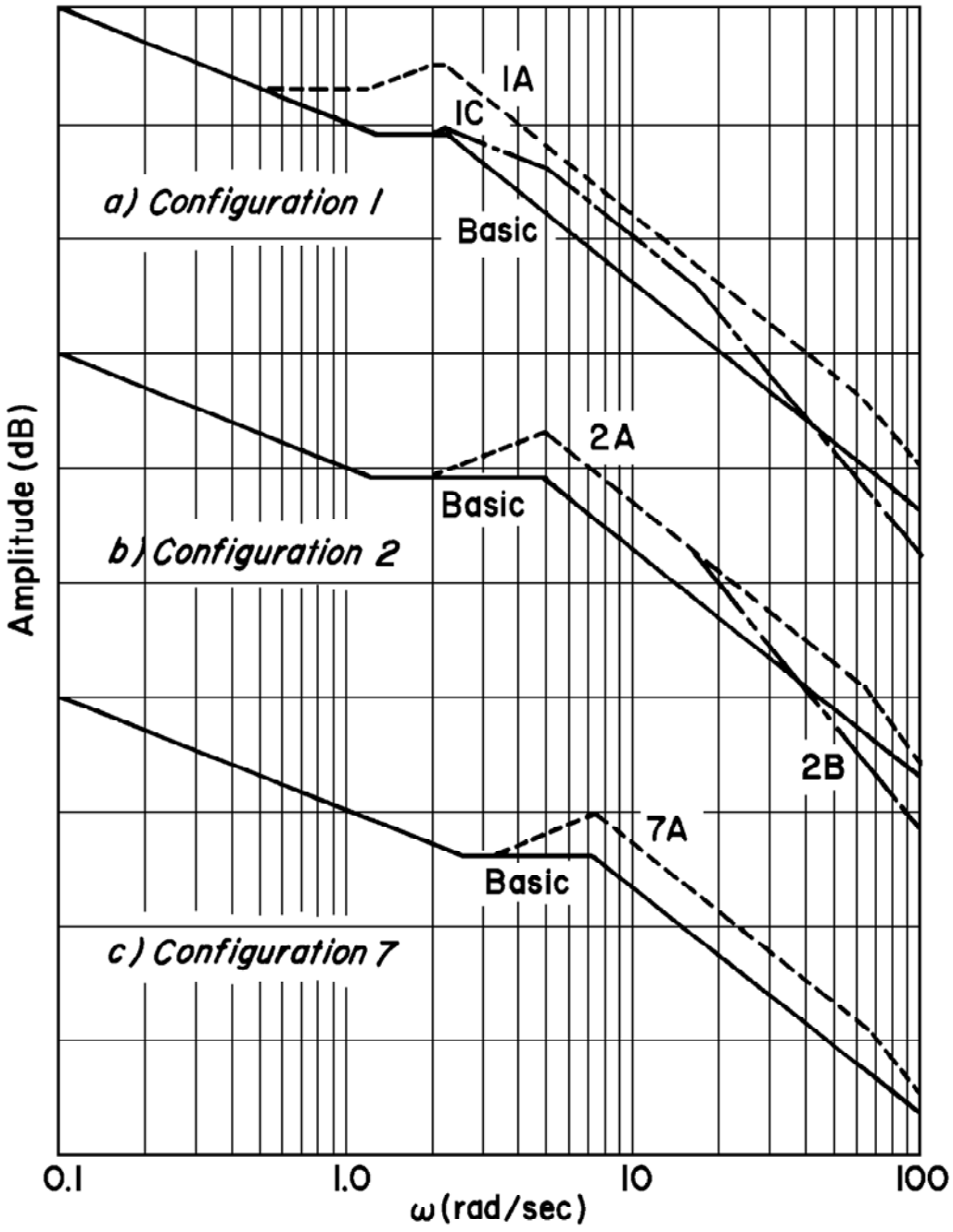


FIGURE 23. **Comparison of Bode amplitude plots for basic and augmented configurations of table XXV.**

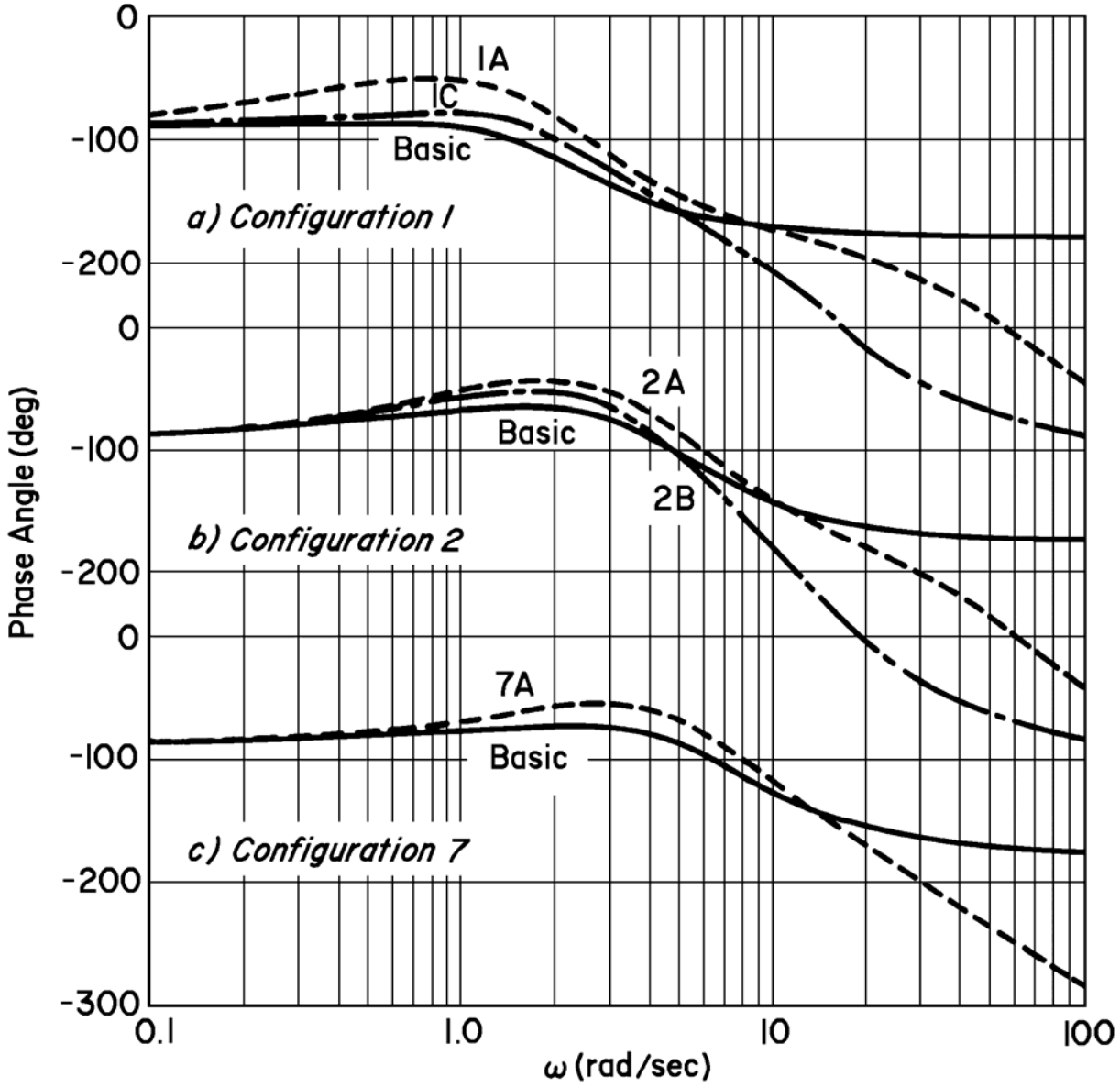
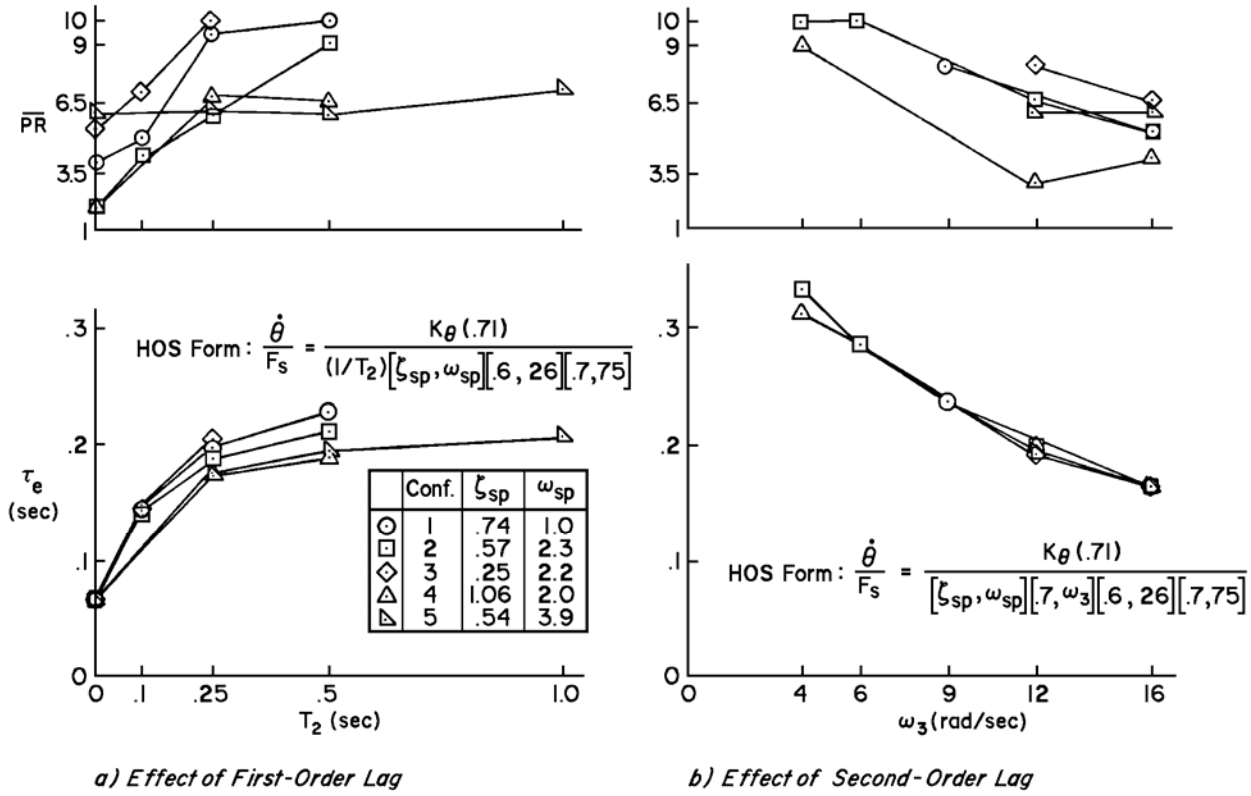


FIGURE 24. **Comparison of phase angles for basic and augmented configurations of table XXV.**

Physically, unrealizable negative time delay, or time lead, might be considered to represent an HOS which is too abrupt (i.e., if $\tau_{\theta_e} < 0$, the system responds to an input τ_{θ_e} sec before the input is made or has finite magnitude at zero time) more or less in keeping with the above-noted pilot commentary.



a) Effect of First-Order Lag *b) Effect of Second-Order Lag*

FIGURE 25. Effect of first- and second-order lags on equivalent time delay and pilot rating: LAHOS configurations (AFFDL-TR-78-122).

VERIFICATION

A. Equivalent system match for a single aerodynamic pitch control surface.

If there is one and only one pitch aerodynamic control surface and no other pitch control effectors (e.g., a thrust vectoring nozzle), then the steps are as follow:

A1. Determine if the pitch response of the aircraft has any lightly damped (damping ratio less than or equal to 0.2), low frequency oscillatory modes (period of oscillation greater than 15 sec). This can be determined through analysis of analytical models of the aircraft in preliminary design, time history analysis from simulation models in preliminary or final design, or time history analysis from flight test data as the design progresses from preliminary design to flight test. If the aircraft has any lightly damped, low frequency oscillatory modes, then the equivalent system match must include both phugoid and short-period modes (proceed to step A2). If there are no lightly damped, low frequency oscillatory modes, then the equivalent system match need only include the short-period mode (proceed to step A5). The absence of any lightly damped, low-frequency oscillatory modes (assuming a stable system) guarantees compliance with the phugoid requirement (5.2.2.1.1.1).

A2. Steps A2 and A3 describe the process for determining the equivalent system match using both phugoid and short-period modes. Estimate the values of the two numerator factors, $1/T_{\theta_1}$ and $1/T_{\theta_2}$, that appear in the transfer function from airframe pitch rate to pitch control surface deflection. These factors can be estimated from the nondimensional or dimensional stability derivatives as derived in numerous references on aircraft flight dynamics (Roskam, McRuer, etc.). Also, $1/T_{\theta_2}$ can be estimated one of two ways. First, $1/T_{\theta_2}$ is approximately equal to the dimensional stability derivative L_w with units of sec^{-1}. Second, $1/T_{\theta_2}$ can be estimated using the following equation if n_z/α for the aircraft is known:

$$\frac{1}{T_{\theta_2}} = \frac{g\,(n_z/\alpha)}{V}$$

where: g = acceleration due to gravity (ft/sec^2)
V = aircraft speed (ft/sec)
n_z/α = change in steady-state normal acceleration per unit change in angle of attack (g/rad)

A3. Perform a fourth-order equivalent system match of aircraft pitch rate to pilot input. The pilot input shall be force for a force controller or displacement for a displacement controller. The form of the equivalent system shall be as follows:

$$\frac{q}{F_s \text{ or } \delta_s} = \frac{K_{q_e}\, s\,(s + \frac{1}{T_{\theta_1}})\,(s + \frac{1}{T_{\theta_2}})\, e^{-\tau_{\theta_e} s}}{(s^2 + 2\zeta_{p_e}\omega_{p_e} s + \omega_{p_e}^2)\,(s^2 + 2\zeta_{sp_e}\omega_{sp_e} s + \omega_{sp_e}^2)}$$

where: q = aircraft pitch rate (rad/sec or deg/sec)
F_s = controller force (lbs)
δ_s = controller deflection (in or deg)
K_{q_e} = gain factor (appropriate units)
s = Laplace operator (sec^{-1})
$1/T_{\theta_1}$, $1/T_{\theta_2}$ = pitch rate numerator factors (sec^{-1})
τ_{θ_e} = equivalent time delay (sec)
ζ_{p_e} = equivalent phugoid damping ratio
ω_{p_e} = equivalent phugoid natural frequency (rad/sec)
ζ_{sp_e} = equivalent short-period damping ratio
ω_{sp_e} = equivalent short-period natural frequency (rad/sec)

The three factors at s = 0, $-1/T_{\theta_1}$, and $-1/T_{\theta_2}$ shall be fixed during the match. The values for $1/T_{\theta_1}$ and $1/T_{\theta_2}$ shall be fixed at the estimated values from step A2. All other parameters in the match shall be free to move to minimize the weighted square error in frequency response gain and phase angle between the aircraft (analytical model, simulator model, or actual aircraft depending upon the stage of aircraft development) and the equivalent system transfer function. The cost function to be minimized in the match shall be:

$$\text{COST} = \frac{20}{n} \sum_{\omega_1}^{\omega_n} [(G_{HOS} - G_{LOES})^2 + 0.01745\,(\Phi_{HOS} - \Phi_{LOES})^2]$$

where: G_{HOS} = the aircraft value of gain at each input frequency (dB)
G_{LOES} = the equivalent system value of gain at each input frequency (dB)
Φ_{HOS} = aircraft value of phase at each input frequency (deg)
Φ_{LOES} = equivalent system value of phase at each input frequency (deg)
n = the number of discrete input frequencies used in the match
$\omega_1 \ldots \omega_n$ = the discrete input frequencies (rad/sec)

The lowest frequency of the match shall not be higher than 0.01 rad/sec and shall be lower than all lightly-damped, low-frequency oscillatory modes present in the aircraft response to pilot pitch input to make sure the equivalent phugoid characteristics are properly identified in the match. The highest frequency of the match shall be 10 rad/sec. At least ten frequency values shall be used per decade of frequency for a total of at least 30 frequencies for the match. When analytical data are used, the frequencies shall be equi-spaced on the log scale. When experimental data are used the lower-order system shall be matched as closely as possible to the data at the experimental frequencies.

A4. The resulting equivalent phugoid damping ratio shall be compared to the appropriate phugoid requirement (5.2.2.1.1.1). Proceed to step A7.

A5. Steps A5 and A6 describe the process to determine the equivalent system match using only the short-period mode. Estimate the value of the numerator factor $1/T_{\theta_2}$ that appears in the transfer function of airframe pitch rate to pitch control surface deflection. See step A2 on ways to estimate this factor.

A6. Perform a second-order equivalent system match of aircraft pitch rate to pilot input. The pilot input shall be force for a force controller or displacement for a displacement controller. The form of the equivalent system shall be as follows:

$$\frac{q}{F_s \text{ or } \delta_s} = \frac{K_{q_e}\left(s + \frac{1}{T_{\theta_2}}\right) e^{-\tau_{\theta_e} s}}{(s^2 + 2\zeta_{sp_e}\omega_{sp_e} s + \omega_{sp_e}^2)}$$

where: q = aircraft pitch rate (rad/sec or deg/sec)
F_s = controller force (lbs)
δ_s = controller deflection (in or deg)
K_{q_e} = pitch rate gain factor (appropriate units)
s = Laplace operator (sec^{-1})
$1/T_{\theta_2}$ = pitch rate numerator factor (sec^{-1})
τ_{θ_e} = pitch rate equivalent time delay (sec)
ζ_{sp_e} = equivalent short-period damping ratio
ω_{sp_e} = equivalent short-period natural frequency (rad/sec)

The numerator factor at $s = -1/T_{\theta_2}$ shall be fixed at the estimated value from step A5. All other parameters in the match shall be free to move to minimize the weighted squared error in frequency response gain and phase angle between the aircraft (analytical model, simulator model, or actual aircraft depending upon the stage of aircraft development) and the equivalent system transfer function. The cost function to be minimized in the match shall be the same as that shown in step A3. The lowest frequency of the match shall be 0.1 rad/sec and the highest frequency shall be 10 rad/sec. At least ten frequency values shall be used per decade of frequency for a total of at least 20 frequencies for the match. When analytical data are used, the frequencies shall be equi-spaced on the log scale. When experimental data are used the lower-order system shall be matched as closely as possible to the data at the experimental frequencies.

A7. The resulting values of $1/T_{\theta_2}$, equivalent time delay, equivalent short-period damping ratio, and equivalent short-period natural frequency shall be used to verify compliance with the appropriate short-period response requirements (5.2.2.1.1.2).

B. Equivalent system match for multiple aerodynamic pitch control surfaces moving in unison.

If there are two or more pitch aerodynamic control surfaces and no other non-aerodynamic control effectors, and the aerodynamic control surfaces move in unison, as if they are geared to each other, follow steps B1 through B9.

B1. Determine if the pitch response of the aircraft has any lightly damped (damping ratio less than or equal to 0.2), low frequency oscillatory modes (period of oscillation greater than 15 sec). This can be determined through analysis of analytical models of the aircraft in preliminary design, time history analysis from simulation models in preliminary or final design, or time history analysis from flight test data as the design progresses from preliminary design to flight test. If the aircraft has any lightly damped, low frequency oscillatory modes, then the equivalent system match must include both phugoid and short-period modes (proceed to step B2). If there are no lightly damped, low frequency oscillatory

modes, then the equivalent system match need only include the short-period mode (proceed to step B6). The absence of any lightly damped, low-frequency oscillatory modes (assuming a stable system) guarantees compliance with the phugoid requirement (5.2.2.1.1.1).

B2. Steps B2 through B4 describe the process for determining the equivalent system match using both phugoid and short-period modes. Estimate the values of the two numerator factors, $1/T_{\theta_1}$ and $1/T_{\theta_2}$, that appear in the transfer functions for airframe pitch rate to each pitch control surface deflection. These factors can be estimated from the nondimensional or dimensional stability derivatives as derived in numerous references on aircraft flight dynamics (Roskam, McRuer, etc.).

B3. Using block diagram manipulation methods, combine the parallel paths through each surface to the airframe pitch rate to determine values of $1/T_{\theta_1}$ and $1/T_{\theta_2}$ to be used in the equivalent system match. Again remember that this assumes that the surfaces move in unison, as if they are geared to each other. The following example for two surfaces will help explain the procedure.

In this example the airframe has two pitch aerodynamic control surfaces, an elevator and a flap. The values for $1/T_{\theta_1}$ and $1/T_{\theta_2}$ for the elevator are 0.0259 and 1.081 sec^{-1}, respectively. The values for $1/T_{\theta_1}$ and $1/T_{\theta_2}$ for the flap are 0.0227 and 1.394 sec^{-1}, respectively. Assume the gain on the input to the elevator is 2 and the gain on the input to the flap is 3. The resulting block diagram for the parallel paths through both surfaces should look like figure 26.

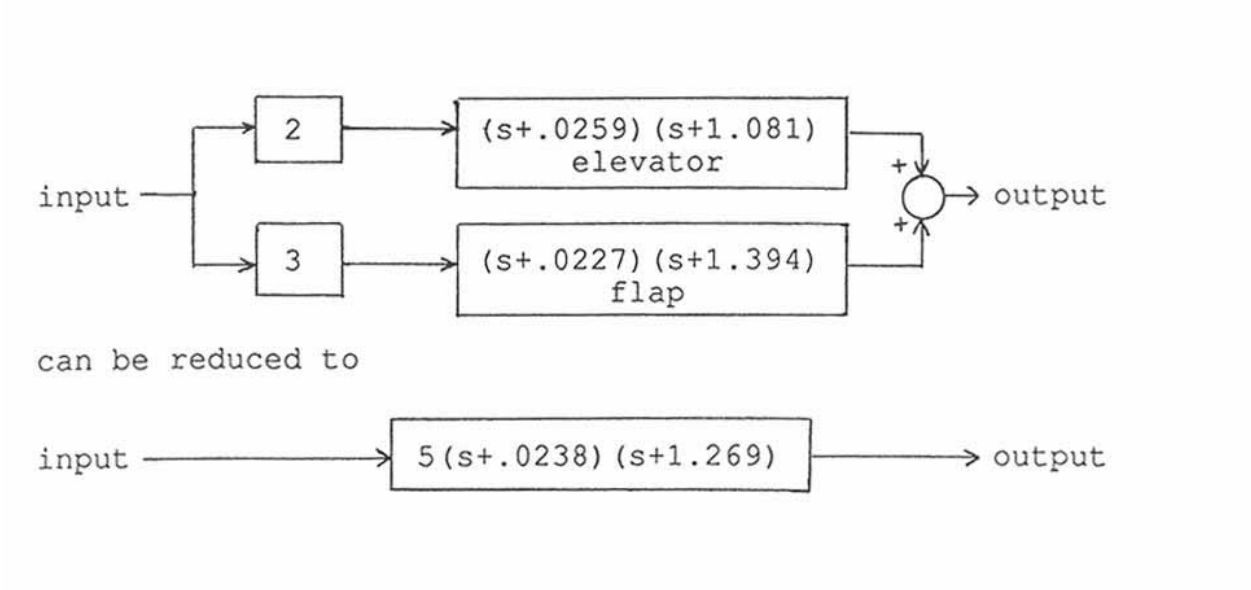


FIGURE 26. Block diagram for two pitch control surfaces moving in unison.

The result of the parallel combination of both surfaces is:

$$5\ (s + 0.0238)\ (s + 1.269)$$

Therefore, the values of $1/T_{\theta_1}$ and $1/T_{\theta_2}$ for the equivalent system match are 0.0238 and 1.269 sec^{-1}, respectively.

B4. Perform a fourth-order equivalent system match of aircraft pitch rate to pilot input in the same manner as described in step A3. The fixed values for $1/T_{\theta_1}$ and $1/T_{\theta_2}$ shall be equal to the values from step B3.

B5. The resulting equivalent phugoid damping ratio shall be compared to the appropriate phugoid requirement (5.2.2.1.1.1). Proceed to step B9.

B6. Steps B6 through B9 describe the process to determine the equivalent system match using only the short-period mode. Estimate the values of the $1/T_{\theta_2}$ numerator factors that appear in the transfer functions of airframe pitch rate to each pitch control surface deflection.

B7. Using block diagram manipulation methods, combine the parallel paths through each surface to the airframe pitch rate to determine the value of $1/T_{\theta_2}$ to be used in the equivalent system match. Again remember that this assumes that the surfaces move in unison, as if they are geared to each other. The following example for two surfaces will help explain the procedure.

In this example the airframe has two pitch aerodynamic control surfaces, an elevator and a flap. The value for $1/T_{\theta_2}$ for the elevator is 1.081 sec^{-1} and the value for $1/T_{\theta_2}$ for the flap is 1.394 sec^{-1}. Assume the gain on the input to the elevator is 2 and the gain on the input to the flap is 3. The resulting block diagram for the parallel paths through both surfaces look like figure 27.

The result of the parallel combination of both surfaces is:

$$5\ (s + 1.269)$$

Therefore, the value of $1/T_{\theta_2}$ for the equivalent system match is 1.269 sec^{-1}.

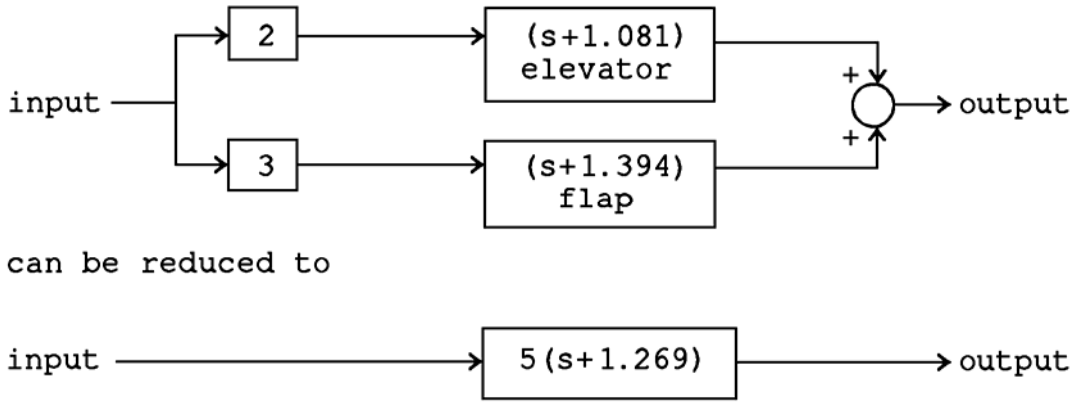


FIGURE 27. Block diagram for two pitch control surfaces moving in unison (high frequency modes only).

B8. Perform a second-order equivalent system match of aircraft pitch rate to pilot input in the same manner as described in step A6. The fixed value of $1/T_{\theta_2}$ shall be equal to the value from step B7.

B9. The resulting values of $1/T_{\theta_2}$, equivalent time delay, equivalent short-period damping ratio, and equivalent short-period natural frequency shall be used to verify compliance with the appropriate short-period response requirements (5.2.2.1.1.2).

C. Equivalent system match for multiple pitch control surfaces which do not move in unison and for non-aerodynamic pitch control effectors.

If there are two or more aerodynamic pitch control surfaces which do not move in unison, or, if there are non-aerodynamic pitch control effectors in addition to the aerodynamic pitch control surfaces, follow steps C1 through C5.

C1. Determine if the pitch response of the aircraft has any lightly damped (damping ratio less than or equal to 0.2), low frequency oscillatory modes (period of oscillation greater than 15 sec). This can be determined through analysis of analytical models of the aircraft in preliminary design, time history analysis from simulation models in preliminary or final design, or time history analysis from flight test data as the design progresses from preliminary design to flight test. If the aircraft has any lightly damped, low frequency oscillatory modes, then the equivalent system match must include both phugoid and short-period modes (proceed to step C2). If there are no lightly damped, low frequency oscillatory modes, then the equivalent system match need only include the short-period mode (proceed to step C4). The absence of any lightly damped, low-frequency

oscillatory modes (assuming a stable system) guarantees compliance with the phugoid requirement (5.2.2.1.1.1).

C2. Step C2 describes the process for determining the equivalent system match using both phugoid and short-period modes. Perform a simultaneous equivalent system match of aircraft pitch rate to pilot input and aircraft angle of attack to pilot input. The pilot input shall be force for a force controller or displacement for a displacement controller. The forms of the two simultaneous equivalent systems shall be as follow:

$$\frac{q}{F_s \text{ or } \delta_s} = \frac{K_{q_e}\, s\,(s + \frac{1}{T_{\theta_{1_e}}})\,(s + \frac{1}{T_{\theta_{2_e}}})\, e^{-\tau_{\theta_e} s}}{(s^2 + 2\zeta_{p_e}\omega_{p_e} s + \omega_{p_e}^2)\,(s^2 + 2\zeta_{sp_e}\omega_{sp_e} s + \omega_{sp_e}^2)}$$

$$\frac{\alpha}{F_s \text{ or } \delta_s} = \frac{K_{\alpha_e}\,(s^2 + 2\zeta_{\alpha_e}\omega_{\alpha_e} s + \omega_{\alpha_e}^2)\, e^{-\tau_{\alpha_e} s}}{(s^2 + 2\zeta_{p_e}\omega_{p_e} s + \omega_{p_e}^2)\,(s^2 + 2\zeta_{sp_e}\omega_{sp_e} s + \omega_{sp_e}^2)}$$

where:
- q = aircraft pitch rate (rad/sec or deg/sec)
- α = aircraft angle of attack (rad or deg)
- F_s = controller force (lbs)
- δ_s = controller deflection (in or deg)
- K_{q_e} = pitch rate gain factor (appropriate units)
- K_{α_e} = angle of attack gain factor (appropriate units)
- s = Laplace operator (sec^{-1})
- $1/T_{\theta_{1_e}}$, $1/T_{\theta_{2_e}}$ = pitch rate numerator factors (sec^{-1})
- ζ_{α_e} = angle of attack numerator damping ratio
- ω_{α_e} = angle of attack numerator natural frequency (rad/sec)
- τ_{θ_e} = pitch rate equivalent time delay (sec)
- τ_{α_e} = angle of attack equivalent time delay (sec)
- ζ_{p_e} = equivalent phugoid damping ratio
- ω_{p_e} = equivalent phugoid natural frequency (rad/sec)
- ζ_{sp_e} = equivalent short-period damping ratio
- ω_{sp_e} = equivalent short-period natural frequency (rad/sec)

All parameters in the match shall be free to move to minimize the weighted squared error over the same recommended frequency range as given in step A3.

The overall cost function to be minimized for the simultaneous match shall be the average of the two individual weighted squared errors.

$$COST = \frac{20}{2n}\sum_{\omega_1}^{\omega_n}[(G_{HOS_q} - G_{LOES_q})^2 + 0.01745\,(\Phi_{HOS_q} - \Phi_{LOES_q})^2 + (G_{HOS_\alpha} - G_{LOES_\alpha})^2 + 0.01745\,(\Phi_{HOS_\alpha} - \Phi_{LOES_\alpha})^2]$$

where:
- G_{HOS_q} = the aircraft value of pitch rate gain at each input frequency (dB)
- G_{HOS_α} = the aircraft value of angle of attack gain at each input frequency (dB)
- G_{LOES_q} = the equivalent system value of pitch rate gain at each input frequency (dB)
- G_{LOES_α} = the equivalent system value of angle of attack gain at each input frequency (dB)
- Φ_{HOS_q} = aircraft value of pitch rate phase at each input frequency (deg)
- Φ_{HOS_α} = aircraft value of angle of attack phase at each input frequency (deg)
- Φ_{LOES_q} = equivalent system value of pitch rate phase at each input frequency (deg)
- Φ_{LOES_α} = equivalent system value of angle of attack phase at each input frequency (deg)
- n = the number of discrete input frequencies used in the match
- $\omega_1 \ldots \omega_n$ = the discrete input frequencies (rad/sec)

When analytical data are used, the frequencies shall be equi-spaced on the log scale. When experimental data are used the lower-order system shall be matched as closely as possible to the data at the experimental frequencies.

C3. The resulting equivalent phugoid damping ratio shall be compared to the appropriate phugoid requirement (5.2.2.1.1.1). Proceed to step C5.

C4. Step C4 describes the process for determining the equivalent system match using only the short-period mode. Perform a simultaneous equivalent system match of aircraft pitch rate to pilot input and aircraft angle of attack to pilot input. The pilot input shall be force for a force controller or displacement for a displacement controller. The forms of the two simultaneous equivalent systems shall be as follow:

$$\frac{q}{F_s \text{ or } \delta_s} = \frac{K_{q_e}\ (s + \frac{1}{T_{\theta_{2_e}}})\, e^{-\tau_{\theta_e} s}}{(s^2 + 2\zeta_{sp_e}\omega_{sp_e} s + \omega_{sp_e}^2)}$$

$$\frac{\alpha}{F_s \text{ or } \delta_s} = \frac{K_{\alpha_e}\, e^{-\tau_{\alpha_e} s}}{(s^2 + 2\zeta_{sp_e}\omega_{sp_e} s + \omega_{sp_e}^2)}$$

where: q = aircraft pitch rate (rad/sec or deg/sec)
α = aircraft angle of attack (rad or deg)
F_s = controller force (lbs)
δ_s = controller deflection (in or deg)
K_{q_e} = pitch rate gain factor (appropriate units)
K_{α_e} = angle of attack gain factor (appropriate units)
s = Laplace operator (sec^{-1})
$1/T_{\theta_{2_e}}$ = pitch rate numerator factors (sec^{-1})
τ_{θ_e} = pitch rate equivalent time delay (sec)
τ_{α_e} = angle of attack equivalent time delay (sec)
ζ_{sp_e} = equivalent short-period damping ratio
ω_{sp_e} = equivalent short-period natural frequency (rad/sec)

The cost function to be minimized in the match shall be the same as that shown in step C2. All parameters in the match shall be free to move to minimize the weighted squared error over the same recommended frequency range as given in step A6. When analytical data are used, the frequencies shall be equi-spaced on the log scale. When experimental data are used the lower-order system shall be matched as closely as possible to the data at the experimental frequencies.

C5. The resulting values of $1/T_{\theta_{2_e}}$, equivalent time delay, equivalent short-period damping ratio, and equivalent short-period natural frequency shall be used to verify compliance with the appropriate short-period response requirements (5.2.2.1.1.2).

The weighting between gain and phase is such that 1 dB of gain mismatch and 7º of phase mismatch have equal significance in the total cost function. In practice the results have been found to be very insensitive to this weighting. For example, the value of 0.01745 used here (which appears to be a degrees-to-radians correction factor but actually is not) will not produce materially different results than will 0.02. Experience with one matching program is reproduced in the following excerpt from a McDonnell paper in Journal R.Ae.S., Feb 1976:

> The nominal value of W [weighting] in the cost function was 0.01745. Some runs were made with $W = (0.01745)^2$, so that equal weight could be assigned to gain (dB) and phase (rad) mismatches. This produced matches which were

qualitatively judged to weight gain too heavily, and using 0.01745 instead produced matches which were a good balance between gain and phase for Neal and Smith's configurations. In fact, the parameter values were fairly insensitive to weighting coefficient choice, as indicated in the following table, which illustrates the effects of W changes for configuration 2-H.

W	τ_{θ_e}	ω_{sp_e}	ζ_{sp_e}	$1/T_{\theta_{2_e}}$	K_{q_e}
$(0.01745)^2$	0.092	3.89	0.50	4.67	3.37
0.01745	0.095	3.95	0.51	4.67	3.44
0.05	0.104	3.75	0.54	3.71	3.88
0.1	0.110	3.70	0.56	3.34	4.13

The factor 20/n does not affect the equivalent parameters. It is included as a convenience to allow the cost function value to be compared with similarly defined cost functions in the literature.

Both theoretically and in practice, equivalent system parameters determined by matching q/F_s or q/δ_s with fixed $1/T_{\theta_2}$ shall be about the same as those determined by simultaneously matching q/F_s (or q/δ_s) and α/F_s (or α/δ_s) with free $1/T_{\theta_{2_e}}$.

When different responses are matched simultaneously (i.e., pitch rate to stick force and angle of attack to stick force), each response shall have equal significance (magnitude different in dB). However, note that the minimum value of the total cost function will usually occur with numerically unequal gain and phase mismatches and unequal mismatches for different responses. Modal parameters common to the two responses shall be constrained to be identical. For example, the short-period mode shall have the same frequency and damping in the pitch rate and angle of attack responses.

5.2.2.1.1.1 Phugoid dynamics.
Any oscillatory mode with a frequency of 0.42 rad/sec or less shall meet the damping requirements shown in table XXVI. There shall be no unstable aperiodic modes within the ROTH for any Level of flying qualities, except as may be allowed in 5.2.2.1.1.2, 5.2.2.1.2.4, and 5.2.2.1.2.4.2. These requirements apply with cockpit controls fixed and with them free.

TABLE XXVI. Damping requirements for low-frequency oscillatory modes.

Level	Damping Requirement
1	
2	
3	

REQUIREMENT RATIONALE

The long-term response, characteristically the phugoid mode involving airspeed and pitch attitude, is important in unattended or divided-attention operation, as well as being the means through which speed is regulated.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:

MIL-F-8785C: 3.2.1.1 and 3.2.1.2
MIL-STD-1797: 4.2.1.1.

The recommended values for table XXVI are:

Level	Damping Requirement
1	$\zeta > .04$
2	$\zeta > 0$
3	$T_2 \geq 55$ sec

REQUIREMENT LESSONS LEARNED

Control force and deflection have both been found universally to be important pilot cues, so both controls-fixed and controls-free (control deflection and force) characteristics have been specified. The recommended values are based on the data and analysis presented in AFFDL-TR-69-72.

An additional Level 3 requirement might be considered to cover the case of two unstable phugoid roots, which has been encountered. Such a limit is probably a good idea, but data are insufficient to establish a firm value.

Two other factors have been observed to alter the phugoid mode: thrust offset and compressibility; a stability derivative, M_u, can result from vertical placement of the thrust line relative to the center of mass; too much M_u of either sign can be destabilizing and cause increased trim changes and gust response. The transonic trim change, or tuck tendency, is restricted in 5.2.2.1.2.4.2. The common, slight long-term instability from lift-curve slope decrease with increasing Mach number in supersonic flight has not generally been a noticeable flying qualities problem. However, the XB-70 experienced a troublesome sensitivity of flight path to pitch attitude at Mach 3, a different but perhaps related problem.

Simple approximations for classical frequency and damping (valid only at subsonic speeds, see AFFDL-TR-65-218) involve only airspeed, L/D, and the static and maneuver margins (Etkin, "Dynamics of Atmospheric Flight"):

$$\omega_p^2 \doteq 2\left(\frac{g}{V}\right)^2\left(\frac{-C_{m_\alpha}/C_{N_\alpha}}{N_m - x_{cg}}\right) = 2\left(\frac{g}{V}\right)^2\frac{h_n}{h_m} \doteq 2\left(\frac{g}{V}\right)^2$$

$$2\zeta_p\omega_p \doteq 2\left(\frac{g}{V}\right)\frac{C_D}{C_L}$$

in an exchange of kinetic and potential energy at constant AOA. Combinations of stability augmentation, center-of-gravity variation with loading, thrust, and compressibility can cause an unstable phugoid oscillation, its decomposition into two aperiodic roots (with a possible divergence), or a restructuring which involves the short-period roots as well.

Since the effectiveness of attitude or integrated angular rate stabilization depends upon the lift-curve slope (stabilization drives a transfer functions poles towards its zeros, and the zero at $-1/T_{\theta_1}$ changes sign as stall is approached), angle of attack limiting or inhibition may be necessary with pitch feedback (WADC-TR-58-82).

Note that for the entire database, all other aircraft response modes are stable. Although little data exists on multiple Level 3 flying qualities, some such combinations can be unflyable.

Although AFWAL-TR-83-3015 indicates a damping ratio somewhat less than 0.04 to be satisfactory for the L-1011 transport, the MIL-STD-1797A values have been retained on the basis that: a) they are derived from systematic flight evaluations, and b) with relaxed static stability seeming to become the norm, stability augmentation shall not frequently be required only in order to meet the phugoid requirement. When employed, the augmentation shall improve rather than degrade the long-term response.

While stability augmentation can easily improve the phugoid damping, the reverse has also been observed. AFWAL-TR-81-3118 reports a simulation of two stability augmentation schemes for a basically unstable airframe. Pitch rate feedback with forward-loop integration eliminated the phugoid mode. AOA feedback, however, at high gain actually reduced phugoid damping. Evidently the small AOA contribution had improved the damping, so reducing the excursion affected it adversely.

Pitch attitude stabilization, or integration of pitch rate feedback, improves the phugoid damping, but at least for high gain, restricts the maximum steady normal acceleration and makes $d\delta/dV$, the static stability indicator, zero. Even for terminal flight phases, where required maneuverability is not great, some adjustments are helpful. Several investigators have found that pilots appreciate the further addition of a turn coordination feature that eliminates the steady control force in coordinated turns, thus removing an undesirable feature of such augmentation, the need to push forward to recover. Similarly, Calspan (NASA CR-172491) has found it beneficial to insert a 0.2 rad/sec washout so that the nose drops normally as speed bleeds off in a flare or landing. Without the washout, a nose-down correction to lose altitude requires pushing on the stick—an unnatural action which pilots are reluctant to do near the ground, at least until they learn the technique well. However, that same washout, it was found, would saturate the Shuttle cockpit controller when the pilot pushes over to capture the glide slope.

VERIFICATION

The equivalent phugoid damping ratio, ζ_p, is to be determined from the three-degree-of-freedom equivalent classical longitudinal response; i.e., the transfer function

$$\frac{\theta}{\delta_{es}} \text{ or } \frac{\theta}{F_{es}} = \frac{K_\theta (s + 1/T_{\theta_1})(s + 1/T_{\theta_2}) e^{-\tau_{\theta_e} s}}{[s^2 + 2\zeta_{p_e}\omega_{p_e} s + \omega_{p_e}^2][s^2 + 2\zeta_{sp_e}\omega_{sp_e} s + \omega_{sp_e}^2]}$$

is matched to the actual frequency response to determine the "best fit" of the equivalent-system parameters. The Level 3 requirement, stated in terms of time to double amplitude, shall be checked for both nose-up and nose-down control impulses.

In practice, this matching should seldom be necessary: either a single lightly damped mode will stand out at low frequency or no more than a glance will be needed to see that the requirement has been more than met. In unaugmented aircraft, the phugoid is the lightly damped oscillation at which this requirement is aimed. If stability augmentation suppresses the phugoid, the requirement obviously has been met and exceeded; then the only concern will be that the augmentation has not introduced other difficulties.

T_2 is the time to double amplitude based on the unstable root [$T_2 = (\ln 2)/\lambda$ where λ is the value of the unstable root].

5.2.2.1.1.2 Short-period dynamics.
The Control Anticipation Parameter (CAP) and equivalent short-period dynamics shall meet the requirements of table XXVII. These requirements apply with cockpit controls fixed and with them free.

TABLE XXVII. Short-period dynamic requirements.

Flight Phase Category	Equivalent Short-period Dynamics	Level 1	Level 2	Level 3
A	CAP_e ($g^{-1}sec^{-2}$)			
	ζ_{sp_e}			
	ω_{sp_e} (rad/sec)			
	τ_{θ_e} (sec)			
B	CAP_e ($g^{-1}sec^{-2}$)			
	ζ_{sp_e}			
	τ_{θ_e} (sec)			
C	CAP_e ($g^{-1}sec^{-2}$)			
	ζ_{sp_e}			
	ω_{sp_e} (rad/sec)			
	n/α (g/rad)			
	τ_{θ_e} (sec)			

REQUIREMENT RATIONALE

The short-term pitch response characteristics are universally regarded to be extremely important.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.2.1 through 3.2.2.1.2 and 3.5.3
MIL-STD-1797: 4.2.1.2.

Recommended values for table XXVII:

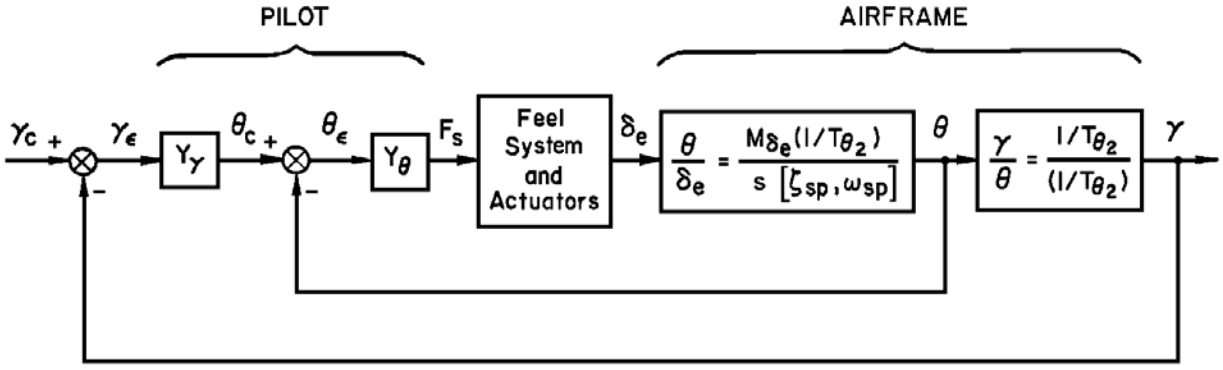
Flight Phase Category	Equivalent Short-period Dynamics	Level 1	Level 2	Level 3
A	CAP_e ($g^{-1}sec^{-2}$)	$0.28 \le CAP_e \le 3.6$	$0.16 \le CAP_e \le 10$	
	ζ_{sp_e}	$0.35 \le \zeta_{sp_e} \le 1.3$	$0.25 \le \zeta_{sp_e} \le 2$	$\zeta_{sp_e} \ge 0.15$ (may be relaxed above 20,000 ft)
	ω_{sp_e} (rad/sec)	$\omega_{sp_e} \ge 1$	$\omega_{sp_e} \ge 0.6$	
	τ_{θ_e} (sec)	$\tau_{\theta_e} \le 0.1$	$\tau_{\theta_e} \le 0.2$	$\tau_{\theta_e} \le 0.25$
B	CAP_e ($g^{-1}sec^{-2}$)	$0.085 \le CAP_e \le 3.6$	$0.038 \le CAP_e \le 10$	
	ζ_{sp_e}	$0.3 \le \zeta_{sp_e} \le 2$	$0.2 \le \zeta_{sp_e} \le 2$	$T_2 \ge 6$ sec* or, in the presence of any other Level 3 flying qualities, $\zeta_{sp_e} \ge 0.05$
	τ_{θ_e} (sec)	$\tau_{\theta_e} \le 0.1$	$\tau_{\theta_e} \le 0.2$	$\tau_{\theta_e} \le 0.25$
C	CAP_e ($g^{-1}sec^{-2}$)	$0.16 \le CAP_e \le 3.6$	$0.05 \le CAP_e \le 10$	
	ζ_{sp_e}	$0.35 \le \zeta_{sp_e} \le 1.3$	$0.25 \le \zeta_{sp_e} \le 2$	$\zeta_{sp_e} \ge 0.15$
	ω_{sp_e} (rad/sec)	Class I, II-C, & IV: $\omega_{sp_e} \ge 0.87$ Class II-L & III: $\omega_{sp_e} \ge 0.7$	Class I, II-C, & IV: $\omega_{sp_e} \ge 0.6$ Class II-L & III: $\omega_{sp_e} \ge 0.4$	Class I, II-C, & IV: $\omega_{sp_e} \ge 0.6$
	n/α (g/rad)	Class I, II-C, & IV: $n/\alpha \ge 2.7$ Class II-L & III: $n/\alpha \ge 2.0$	Class I, II-C, & IV: $n/\alpha \ge 1.8$ Class II-L & III: $n/\alpha \ge 1.0$	
	τ_{θ_e} (sec)	$\tau_{\theta_e} \le 0.1$	$\tau_{\theta_e} \le 0.2$	$\tau_{\theta_e} \le 0.25$

*T_2 is the time to double amplitude based on the unstable root [$T_2 = (\ln 2)/\lambda$ where λ is the value of the unstable root].

REQUIREMENT LESSONS LEARNED

For conventional aircraft, without direct lift control, the piloting task can be viewed as an inner- and outer-loop, single-controller task as indicated for short-period dynamics on figure 28. This figure illustrates how $1/T_{\theta_2}$ influences the short-term attitude dynamics and also defines the short-term flight path (or load factor) response. The limits in table XXVII are based on a combination of load factor response and the quickness of the pitch attitude response to a control input, i.e., $\omega_{sp}^2/(n/\alpha)$. Some interpretations of this parameter are given in the following discussion. Note on figure 28 that γ/θ, being the ratio of bare-airframe numerators, incorporates

the actual value of $1/T_{\theta_2}$, not necessarily the same as the equivalent system's $1/T_{\theta_{2e}}$ if equalization is employed in feedback or feedforward loops.



$$\left.\frac{\theta}{F_s}\right|_e = \frac{A_\theta (1/T_{\theta_2})\, e^{-\tau_{\theta_e} s}}{(0)\left[\zeta_{sp_e}, \omega_{sp_e}\right]}$$

$$\left.\frac{\gamma}{F_s}\right|_e = \frac{A_\gamma e^{-\tau_{\gamma_e} s}}{(0)\left[\zeta_{sp_e}, \omega_{sp_e}\right]}$$

where

$(a) = (s + a)$ example $(0) = s + 0 = s$

$\left[\zeta, \omega\right] = \left[s^2 + 2\zeta\omega s + \omega^2\right]$

FIGURE 28. Pilot control of pitch attitude and flight path.

DEFINITION OF CONTROL ANTICIPATION PARAMETER (CAP)

Bihrle in AFFDL-TR-65-198 defines CAP as the ratio of initial pitching acceleration to steady-state normal acceleration (the pseudo steady-state corresponding to the two-degree-of-freedom short-period approximation),

$$\text{CAP} \stackrel{\Delta}{=} \frac{\ddot{\theta}_0}{\Delta n_{z_{ss}}} g^{-1} \sec^{-2} \doteq \frac{\omega_{sp}^2}{\frac{V}{g}\frac{1}{T_{\theta_2}}} g^{-1} \sec^{-2}$$

The last expression is an approximation that generally holds for aircraft with negligible control system dynamics and tail lift effect (circumstances which are not uncommon). Under these circumstances, $(V/g)\,(1/T_{\theta_2})$ can also be approximated by n/α. Note that $\ddot{\theta}_0$ and $\Delta n_{z_{ss}}$ apply to the time response to a step input for an actual system of any order.

Equivalently, in the frequency domain the high-frequency gain of pitch acceleration (thought to be important in fine tracking tasks) is given by M_{F_s} and the steady-state gain of normal load factor (thought to be important in gross or outer loop tasks) by $M_{F_s}(n/\alpha)/\omega_{sp}^2$. Hence their ratio is CAP (see figure 29).

A closely-related analysis in AFFDL-TR-69-72 arrives at $(F_{es}/n)\, M_{F_s}$ equal to $\omega_{sp}^2/(n/\alpha)$ where M_{F_s} is the initial pitch acceleration per pound. This can be seen in the asymptotic n_z/F_{es} frequency response of figure 29.

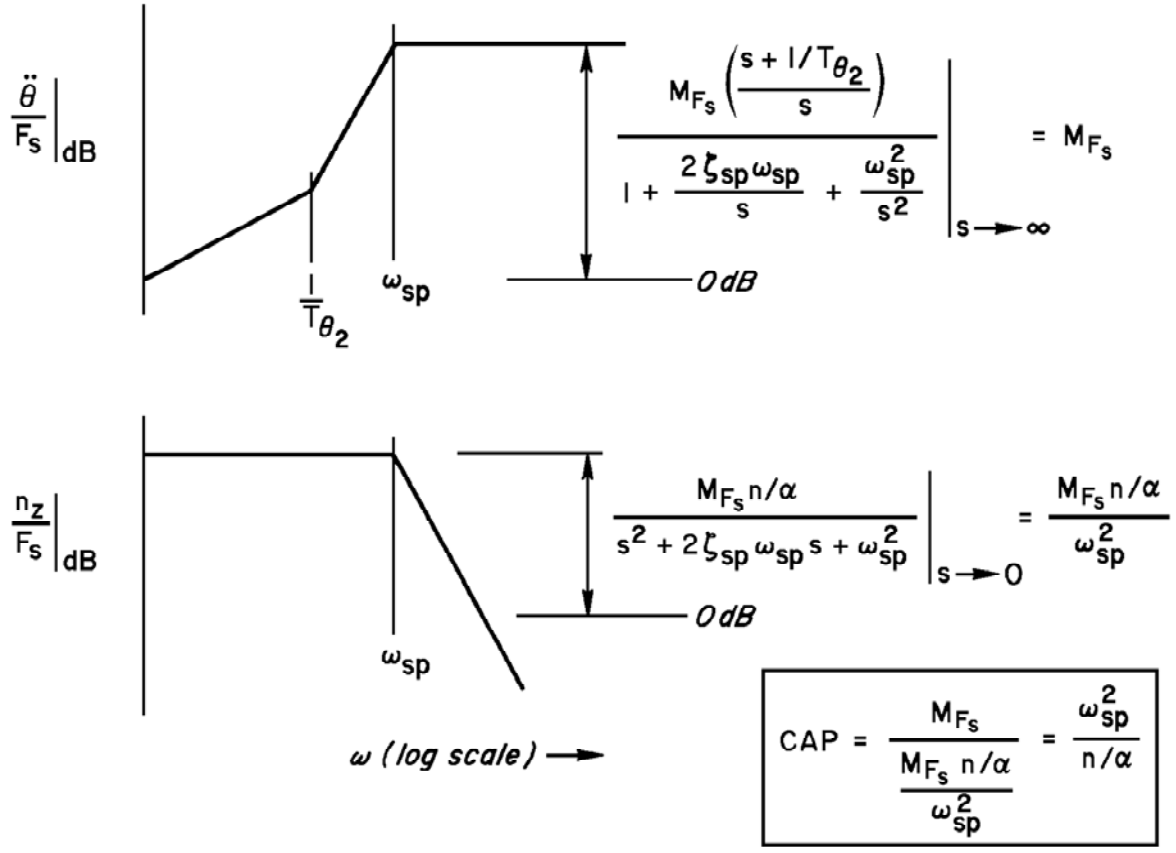


FIGURE 29. Definition of CAP from frequency-response asymptotes.

As noted by Bischoff (NADC-81186-60), the Control Anticipation Parameter must be redefined for aircraft with effective time delay since $\ddot{\theta}_0 = 0$ in this case. Following DiFranco (AFFDL-TR-66-163), Bischoff defines, on the basis of a unit step stick force input, a more general Control Anticipation Parameter, CAP′, as

$$CAP' \stackrel{\Delta}{=} \frac{\ddot{\theta}_{max_{HOS}}}{\Delta n_{z_{ss}}}$$

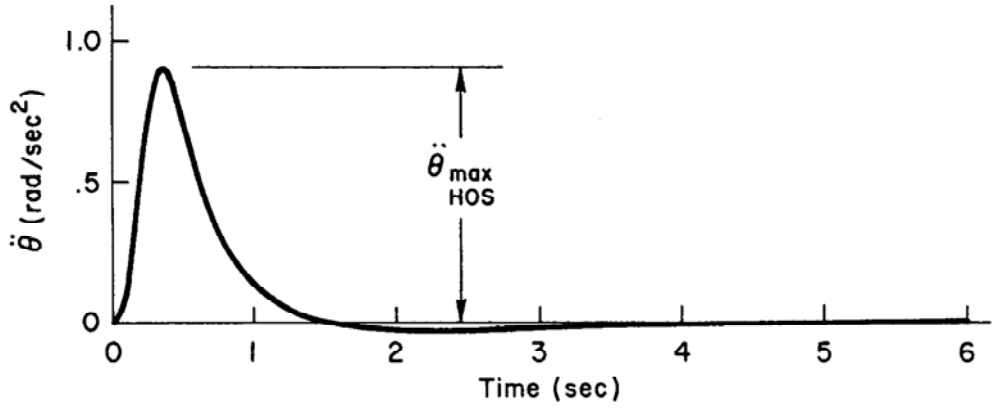


FIGURE 30. Pitch acceleration response to a unit step force input.

The maximum pitch acceleration, $\ddot{\theta}_{max_{HOS}}$, will occur sometime after the force input as shown on figure 30.

The parameter CAP is further extended to the short-period lower-order equivalent system model by defining

$$CAP_e \doteq \left(\frac{\omega_{sp}^2}{n/\alpha}\right)_e \triangleq \frac{\omega_{sp_e}^2}{\frac{V}{g}\frac{1}{T_{\theta_{2_e}}}} g^{-1}\,sec^{-2}$$

where e denotes LOES parameters.

ADDITIONAL SHORT-PERIOD DYNAMIC REQUIREMENTS

Rationally, lower limits on satisfactory short-period frequency and on normal acceleration sensitivity to pitch control should exist. This concept is consistent with the in-flight and ground-based simulator experiments of AFWAL-TR-81-3118 and Mooij & van Gool and Wilhelm & Lange in AGARD-CP-333 (Gibson opts, as have some others, for bounding $1/T_{\theta_2}$ instead of n/α). Although the numerical values of MIL-STD-1797 have been retained, for which there still are few data points, Mooij & van Gool indicate more stringent limits.

A first-order divergence (T_2 = 6 sec) is allowed for the Category B Level 3 pitch attitude dynamics. This is consistent with the Level 3 static stability requirement in 5.2.2.1.2.4. The 6-second limit on instability was derived from in-flight and ground-based simulator studies which have documented the degree of instability that is safely flyable. AFFDL-TR-72-143, for example, indicated a Level 2 boundary with T_2 (based on the unstable aperiodic root) of 2.5 sec in light turbulence and 4.25 sec in moderate turbulence. Pilot ratings were fairly constant at 5 to 6 until the time to double amplitude was reduced below 6 sec, when significant deterioration

began. Some margin is allowed in order to account for pilot distraction, design uncertainties, etc. In this region near neutral stability, the root locations generally are extremely sensitive to static margin. On the other hand, Schuler (AFWAL-TR-82-3014) concludes from his ground-based simulations that for light total damping (small value of the other stable short-term pole) the 6-second limit can be unconservative.

The data base consists of airplanes with classical flying qualities as well as highly augmented airplanes which are treated in this section by reduction to lower-order equivalent systems. The supporting data for classical airplanes and highly augmented airplanes are presented separately in the following two sub-sections, which are further sub-divided according to Flight Phase Category of the data. Lastly, Level 3 substantiation is presented.

Supporting data - classical airplanes

Most of the available data are for Category A Flight Phases only. A small amount of Category C data is available, while data for Category B are extremely sparse. AFFDL-TR-66-163, AFFDL-TR-70-74, AFFDL-TR-68-91, and AFFDL-TR-69-3 provide good short-period data for Category A. NASA TM-X-1584, NASA CR-159236, and AFWAL-TR-83-3015 contain usable Category B data. FDL-TDR-64-60 and NASA TN D-3971 contain Category C information. For large aircraft, AFFDL-TR-72-41, NASA CR-159236, AFWAL-TR-81-3118, AFWAL-TR-83-3015, and AGARD-CP-333 furnish additional insight.

Category A

Figure 31 shows the applicable data (with Level 1 F_s/n) from AFFDL-TR-66-163, AFFDL-TR-68-91, and AFFDL-TR-69-3, and the baseline cases of AFFDL-TR-70-74 compared with the short-period frequency boundaries for the Category A Flight Phases. These data represent 58 separate ω_{sp}-n/α combinations flown and rated by eight pilots. The boundaries correctly predicted pilot ratings about 80 percent of the time—an adequate success rate given the variability of flight tests and pilot ratings. Eight configurations which fell within the Level 1 boundaries were rated Level 2 or worse. Note that most of the violations occurred at large n/α (as at high speed). The Level 1 boundaries are slightly more lax than the best fit to the data presented. The data on figure 31 represent those cases for which ω_{sp} and F_s/n were within the present Level 1 boundaries. Therefore, the ratings shown can be assumed to be due to short-period frequency and n/α influences solely.

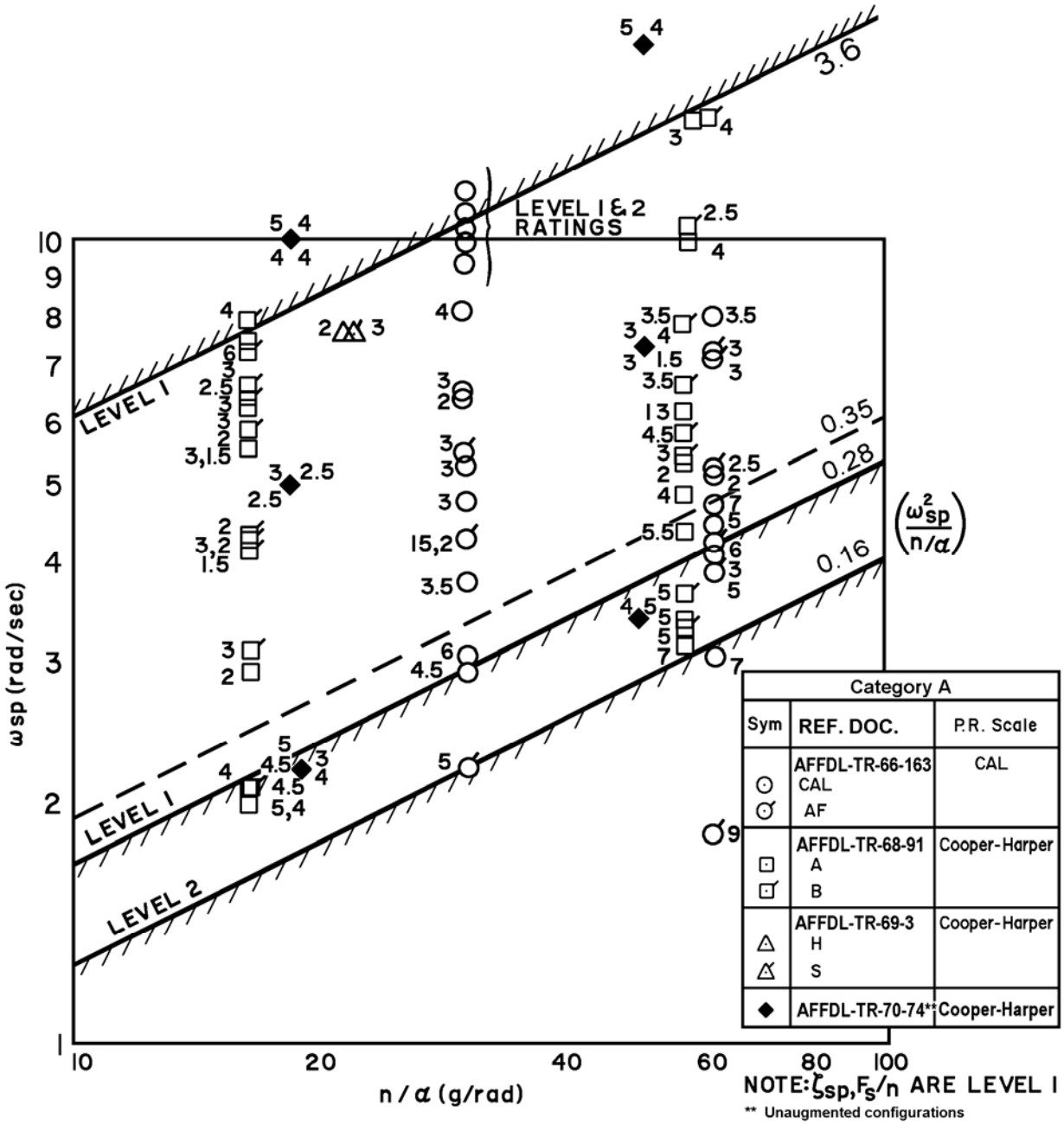


FIGURE 31. **Comparison of pilot ratings with Category A short-period frequency requirements.**

VERIFICATION

The values of ζ_{sp_e}, ω_{sp_e}, and τ_{θ_e} from the lower-order equivalent system match shall be compared with the requirements of table XXVII.

To determine CAP_e, recall that CAP can be approximated by:

$$CAP \doteq \frac{\omega_{sp}^2}{\frac{V}{g}\frac{1}{T_{\theta_2}}} g^{-1} sec^{-2}$$

Using the equivalent system values of $1/T_{\theta_2}$ and ω_{sp} yields:

$$CAP_e \doteq \frac{\omega_{sp_e}^2}{\frac{V}{g}\frac{1}{T_{\theta_{2_e}}}} g^{-1} sec^{-2}$$

For Level 3, the time to double amplitude must be obtained using the perturbation values of pitch attitude (or airspeed) away from trim. A step input establishes a new equilibrium airspeed, but an impulse or doublet does not. Thus, the difference between stability and instability may not be discernible for some seconds after a step control input.

5.2.2.1.2 Longitudinal time responses to the pitch controller.
The longitudinal time responses shall meet the requirements of 5.2.2.1.2.1 through 5.2.2.1.2.4.2.

REQUIREMENT RATIONALE

The time response design criteria limit characteristics of the pitch rate response to pilot commands. This format avoids explicit identification of dominant roots or equivalent system models by working directly with the pitch rate response. Nevertheless, by virtue of assuming a conventional-appearing response these criteria do rely on a form of equivalent system. The criteria are applicable to aircraft exhibiting conventional aircraft dynamic modes and to most pitch augmentation systems, but not to high-gain pitch attitude command systems.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 through 3.3.11.1.2 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.1.1 through 3.2.1.3, 3.2.2.1 through 3.2.2.1.2, and 3.5.3
MIL-STD-1797: 4.2.1 through 4.2.1.2, 4.2.8.3, 4.3.1.1, 4.3.1.2, 4.4.1, 4.4.1.1.

REQUIREMENT LESSONS LEARNED

VERIFICATION

Verification shall be via verification of requirements 5.2.2.1.2.1 through 5.2.2.1.2.4.2. As the design progresses, analysis, simulation, and flight test will be appropriate demonstration means. In the end, flight test will measure some parameters and collect data to validate analytical and simulator results for the rest.

5.2.2.1.2.1 Long-term longitudinal response.
Any oscillation with a period of 15 sec or longer shall meet the damping requirements shown in table XXVIII. There shall be no aperiodic divergence within the ROTH for any Level of flying qualities, except as may be allowed in 5.2.2.1.1.2, 5.2.2.1.2.4, and 5.2.2.1.2.4.2. These requirements apply with cockpit controls fixed and with them free.

TABLE XXVIII. Damping requirements for long-term oscillations.

Level	Damping Requirement
1	
2	
3	

REQUIREMENT RATIONALE

The long-term response, characteristically the phugoid mode involving airspeed and pitch attitude, is important in unattended or divided-attention operation, as well as being the means through which speed is regulated.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.1.1 and 3.2.1.2
MIL-STD-1797: 4.2.1.1.

The recommended values for table XXVIII are:

Level	Damping Requirement
1	$\zeta > .04$
2	$\zeta > 0$
3	$T_2 \geq 55$ sec

REQUIREMENT LESSONS LEARNED

Control force and deflection have both been found universally to be important pilot cues, so both controls-fixed and controls-free (control deflection and force) characteristics have been specified.

VERIFICATION

The Level 3 requirement, stated in terms of time to double amplitude, T_2, shall be checked directly from the time response of the actual aircraft, for both nose-up and nose-down control

impulses. T_2 applies to the value of the unstable root: $T_2 = (\ln 2)/\lambda$ where λ is the value of the unstable root.

Testing specifically for long-term flight characteristics is time consuming. However, most long-term deficiencies will surface during attempts to stabilize at a new flight condition, for example establishing a steady climb or turn, changing airspeed, or flaring for landing.

Some mechanizations of stability augmentation may greatly enhance phugoid damping, but at the same time alter some steady-state flight characteristics: for example, holding a new pitch attitude after removal of a pilot command. Test pilots will need to evaluate any unusual characteristics qualitatively (Supersonic Transport Aeroplane Flying Qualities).

5.2.2.1.2.2 Short-term pitch response to pitch controller.
The pitch rate response to a step input of pitch controller shall meet the requirements of table XXIX for step inputs of cockpit controller force and also for step inputs of cockpit controller deflection, of magnitudes to produce small changes typical of fine tracking. In addition, for Level 1, the pitch response to a pulse input of pitch controller force, and also a pulse controller deflection, shall meet the requirements of table XXX.

TABLE XXIX. Pitch rate time response characteristics to a step input.

Level	Effective Time Delay	Transient Peak Ratio	Effective Rise Time	
			Flight Phase Categories A and B	Flight Phase Category C
1				
2				
3				

TABLE XXX. Pitch time response characteristics to a pulse input.

Flight Phase Category	Pitch Attitude Dropback Ratio (sec)	Pitch Rate Overshoot Ratio	Pitch Rate Peak Time	Pitch Acceleration Peak Time
A				
B				
C				

REQUIREMENT RATIONALE

The short-term pitch response characteristics are universally regarded to be extremely important. This requirement and the discussion that follows is adapted from NASA CR-159236.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.2.1 through 3.2.2.1.2 and 3.5.3
MIL-STD-1797: 4.2.1.2 and 4.2.2.

Recommended values for table XXIX:

Level	Effective Time Delay	Transient Peak Ratio	Effective Rise Time*	
			Flight Phase Categories A and B	Flight Phase Category C
1	$t_1 \leq .12$ sec	TPR $\leq .3$	$9/V_T \leq \Delta t \leq 500/V_T$	$9/V_T \leq \Delta t \leq 200/V_T$
2	$t_1 \leq .17$ sec	TPR $\leq .6$	$3.2/V_T \leq \Delta t \leq 1600/V_T$	$3.2/V_T \leq \Delta t \leq 645/V_T$
3	$t_1 \leq .21$ sec	TPR $\leq .85$		

*V_T is true airspeed in ft/sec

Recommended values for table XXX:

Flight Phase Category	Pitch Attitude Dropback Ratio (sec)	Pitch Rate Overshoot Ratio	Pitch Rate Peak Time	Pitch Acceleration Peak Time
A	$\frac{\theta_{DB}}{q_{ss}} < (3 - \frac{q_{max}}{q_{ss}})/0.7$	$q_{max}/q_{ss} < 3$	$.3 \leq t_q \leq .9$	$t_{\dot{q}} \leq .3$
C			Class III: $.5 \leq t_q \leq 2$ Class IV: $.5 \leq t_q \leq 1.1$	

REQUIREMENT LESSONS LEARNED

The flight experiments reported in AFFDL-TR-68-90, AFFDL-TR-70-74, AFWAL-TR-81-3116, and AFFDL-TR-78-122 have established the critical, detrimental nature of transport time delay and effective time delay resulting from cascaded dynamic elements in the control system. The limits of effective time delay are based on interpretation of the data in these documents.

The effective time delay used in this time domain requirement can be uniquely defined (unless the response is dominated by higher-frequency modes or an aperiodic instability) and easily evaluated graphically or analytically (see figure 32). Values of t_1 are generally smaller, for example, than the time delay determined by the method of AFFDL-TR-70-155. It was necessary, therefore, to evaluate t_1 for the configurations evaluated in various experiments and to correlate these values with pilot ratings.

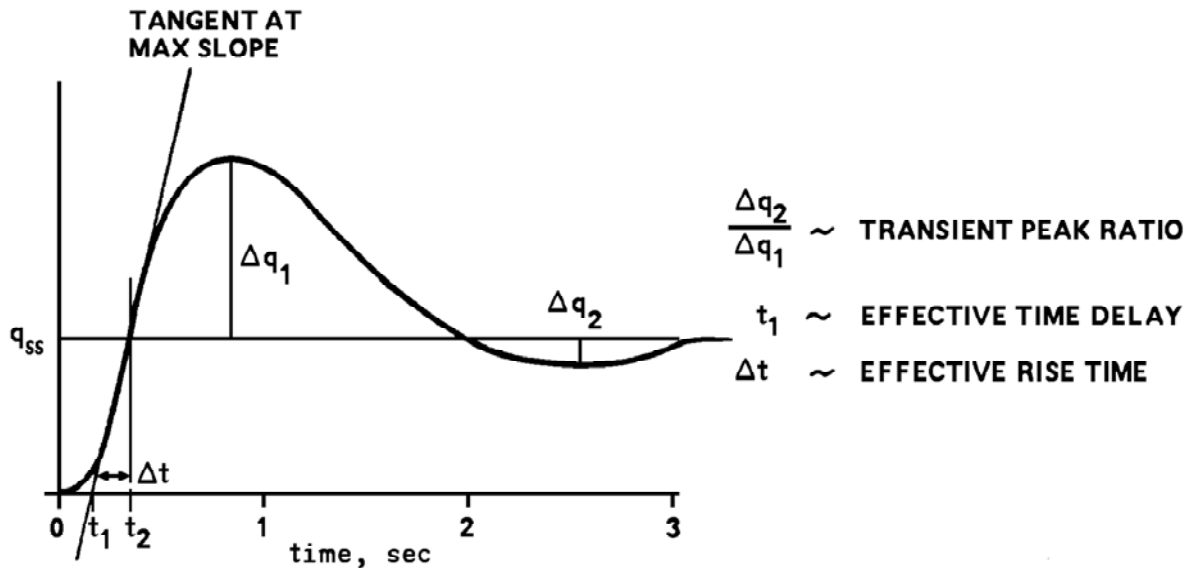


FIGURE 32. Pitch rate response to step input of pitch controller force or deflection.

The intent of the transient peak ratio criteria is to insure adequate damping of the short-period or dominant mode of the pitch response. The specified values are based on interpretation of short-period data in AFFDL-TR-72-41 and AFFDL-TR-69-72. For a classical airplane response, the transient peak ratios would correspond to the following damping ratio values:

Level 1:	TPR = 0.30	$\zeta = 0.36$
Level 2:	TPR = 0.60	$\zeta = 0.16$
Level 3:	TPR = 0.85	$\zeta = 0.052$

The criteria of table XXX also limit the effective rise time, Δt, of the pitch rate response to step pilot command. The effective rise time is related to $\omega_{sp}^2/(n/\alpha)$ by the following:

$$\frac{\omega_{sp}^2}{n/\alpha} = \frac{\ddot{\theta}_0}{n_{z_{ss}}} \cong \frac{q_{ss}/\Delta t}{q_{ss}\, V_T/g} - \frac{g}{V_T\, \Delta t}$$

Limits on $\omega_{sp}^2/(n/\alpha)$ were defined in MIL-F-8785C as a function of Flight Phase Category. For example, the Level 1 limits for Flight Phase Category C were:

$$0.16 \le \omega_{sp}^2/(n/\alpha) \le 3.6 \ \text{rad}^3/\text{g-sec}^2$$

These limits can be related to the effective rise time, as follows, by substitution:

$$0.16 \leq g/(V_T \, \Delta t) \leq 3.6$$

By taking the reciprocals, reversing inequality signs, and rearranging, the following limits for Δt result, in commensurate units:

$$g/(3.6 \, V_T) \leq \Delta t \leq g/(0.16 \, V_T)$$

This development indicates that limits on the effective rise time, Δt, expressed as constants divided by the true speed are analogous to the constant limits on $\omega_{sp}^2/(n/\alpha)$ used in MIL-F-8785C to specify short-period frequency as a function of n/α. Separate Δt requirements are stated for terminal and non-terminal Flight Phases for Level 1 and Level 2. The Level 3 limit is again 6 sec to double amplitude.

The dropback criterion was developed by Gibson (AGARD-CP-333) as part of a set of design guidelines for highly-augmented fighter aircraft. The pitch attitude dropback ratio is a measure of the mid-frequency response to attitude changes. Excessive dropback results in pilot complaints of abruptness and lack of precision in pitch control. The following discussion by Gibson is extracted from AGARD-CP-333 and ICAS-86-5.3.4. Figure 33 illustrates these concepts and defines his parameters. It is important to point out that this discussion is applicable only to conventional and rate response-types: a criterion involving steady-state pitch rate cannot be applied to attitude response-types.

USAF/Calspan in-flight simulations of pitch handling with high-order flight control systems, using their NT-33 aircraft, have furnished a source of information about acceptable attitude and flight path characteristics. In AFFDL-TR-74-9 the tasks of air combat maneuvering, air-to-air tracking, in-flight refueling, and in AFFDL-TR-78-122 the approach and landing tasks were assessed for wide ranges of basic frequency and damping modified by stick prefilters to simulate high-order effects.

Step responses were calculated for all of these configurations and their features compared with pilot rating and comment data. Some quite clear results were obtained which can be summarized as follow:

- Negative pitch attitude dropback (i.e. attitude overshoot) was usually associated with sluggish, unpredictable response both in flight path control and in tracking, leading sometimes to overdriving PIO.

- Pitch attitude dropback ratio from 0 to about 0.25 sec was excellent for fine tracking and was associated with comments typified by “the nose follows the stick”.

- Increasing pitch attitude dropback with large pitch rate overshoot ratio led to abrupt response and bobbling, from a slight tendency to continuous oscillations in tracking tasks. Sometimes this was called PIO, but it did not cause concern for safety.

- Within the range of parameters tested, pitch attitude dropback had little effect upon landing approach, flight refueling, or gross maneuvering without a target, provided the dropback was not negative.

- The pitch rate overshoot ratio seems to qualify the attitude dropback behavior, with a value greater than 3.0 resulting in unacceptable pitch attitude dropback ratio as small as 0.25 sec. The trends are indicated on figure 34.

- Small values of flight path time delay were associated with excellent flight refueling control, but were not essential for good gross maneuvering and did not on their own ensure predictable behavior.

One result stands out clearly. This is to specify that pitch attitude dropback, as defined by Gibson, shall be zero or positive, as negative values were always rated sluggish and unsatisfactory in these experiments. This leads also to the result that some pitch rate overshoot is always necessary for optimum handling, with the function of minimizing pitch attitude dropback for tracking inputs or of rapid generation of the angle of attack increment required for crisp flight path response in gross maneuvers, landing flare, etc. It has no significance otherwise for the pilot unless it becomes too large. However, it has been shown in other results that a small pitch attitude overshoot can be satisfactory for basic attitude control, e.g., in aerobatics, cruise, etc., and for attitude tracking with rigid or small displacement sticks.

The principal time response features of table XXX can be identified in the $\omega_{sp}T_{\theta_2} - \zeta_{sp}$ format as shown on figure 35. Some are constant for all T_{θ_2} and the others vary in proportion to T_{θ_2}.

The constant features are:

- Zero attitude dropback line, along which flight path delay, t_γ, equals T_{θ_2}.

- Lines of constant pitch rate overshoot ratio.

- PIO line where $2\,\zeta_{sp}\,\omega_{sp} = 1/T_{\theta_2}$. Serious PIO and extreme pilot nonlinearities occur below this boundary.

Variable features are:

- Lines of constant pitch rate peak time, t_q.
- Lines of non-zero attitude dropback, along which $t_\gamma = (T_{\theta_2} - DB_\theta/q_{ss})$. Note that $t_\gamma = 2\,\zeta_{sp}/\omega_{sp}$ also, see figure 36.

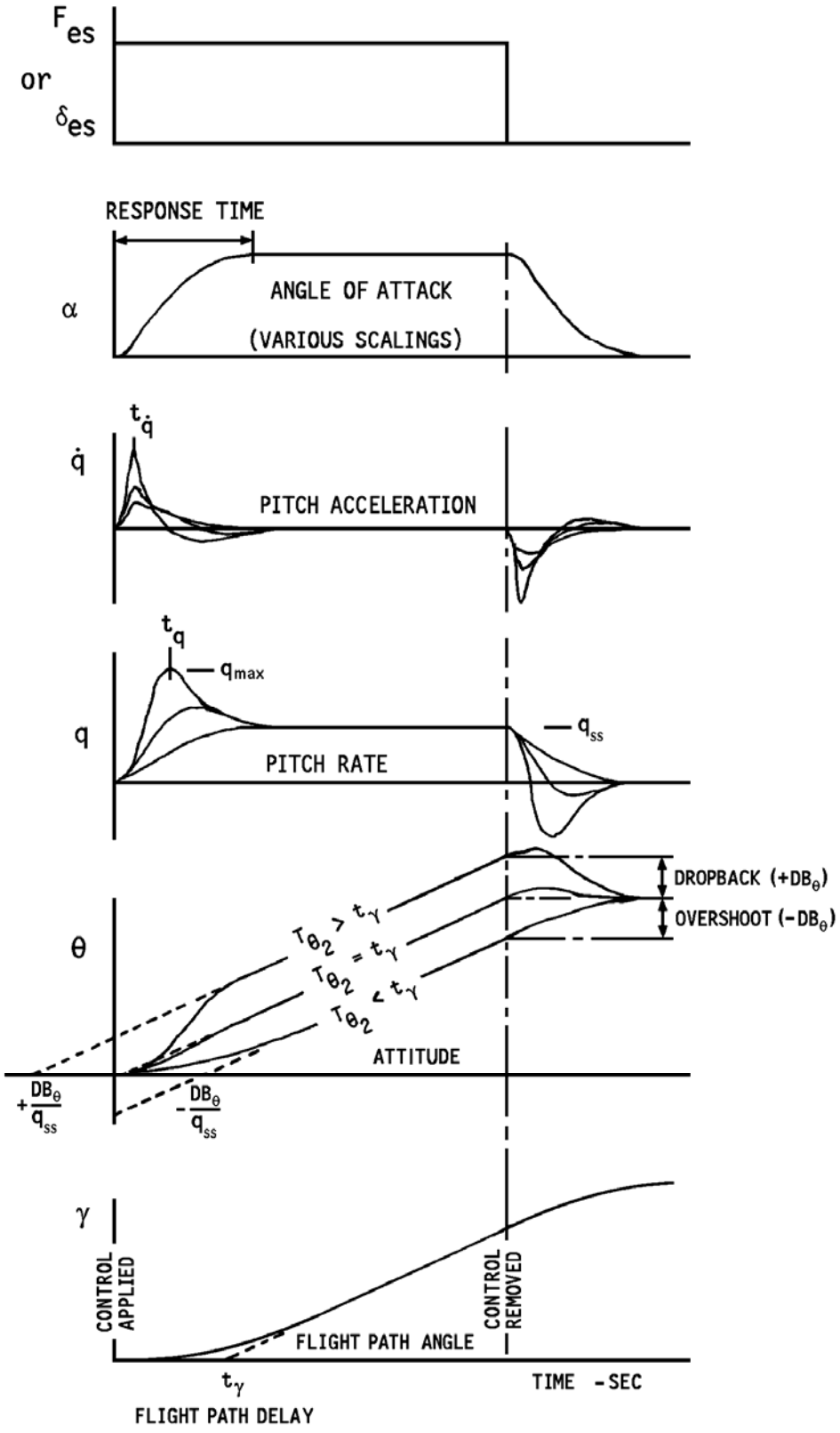


FIGURE 33. Short-term longitudinal time responses.

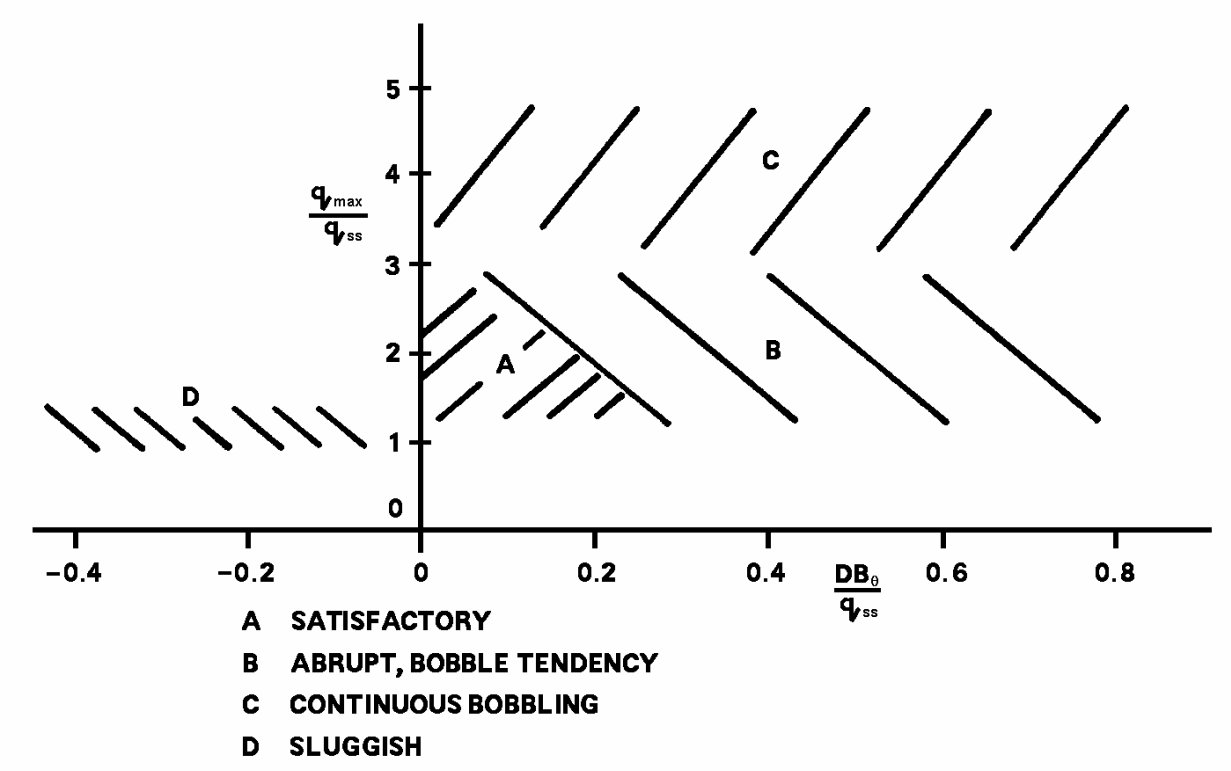


FIGURE 34. Trends with pitch rate overshoot ratio during precision tracking.

Analyses of low-order aircraft indicate that the satisfactory range of Category A short-period frequency is limited by values of t_q between about 0.3 and 0.9 sec. The area of satisfactory handling therefore lies in the upper part of figure 35 for large T_{θ_2} and moves down to the lower part with reducing T_{θ_2}. Larger values of t_q are satisfactory for Category C, e.g., from 0.5 to 1.1 sec for Class IV aircraft up to possibly 2.0 sec for large aircraft. With typical T_{θ_2} of 1.5 to 2.0 at low speed this is in a region of large pitch attitude dropback ratio and pitch rate overshoot ratio, but these are acceptable up to at least 1.5 and 3.0 respectively.

The PIO line forms a lower limit which should not normally be approached if conventional damping augmentation practices are followed, a practical limit slightly above it being expected. The zero pitch attitude dropback line should normally be respected, but in some cases the limit may be above it; e.g., in Category A Flight Phases with T_{θ_2} = 2.75 sec, the recommended Class IV short-period damping and short-period frequency requirements of 5.2.2.1.1.2 for Level 1 (1.3 and 1.0 rad/sec, respectively) form a t_γ limit of 2.6 sec, which sets the minimum pitch attitude dropback ratio (as defined by Gibson) at 0.15 sec. With smaller T_{θ_2}, a lower short-period frequency can make a nominal zero pitch attitude dropback look like a large overshoot in the first few seconds of the landing flare, but this is avoided by observing the t_q limit.

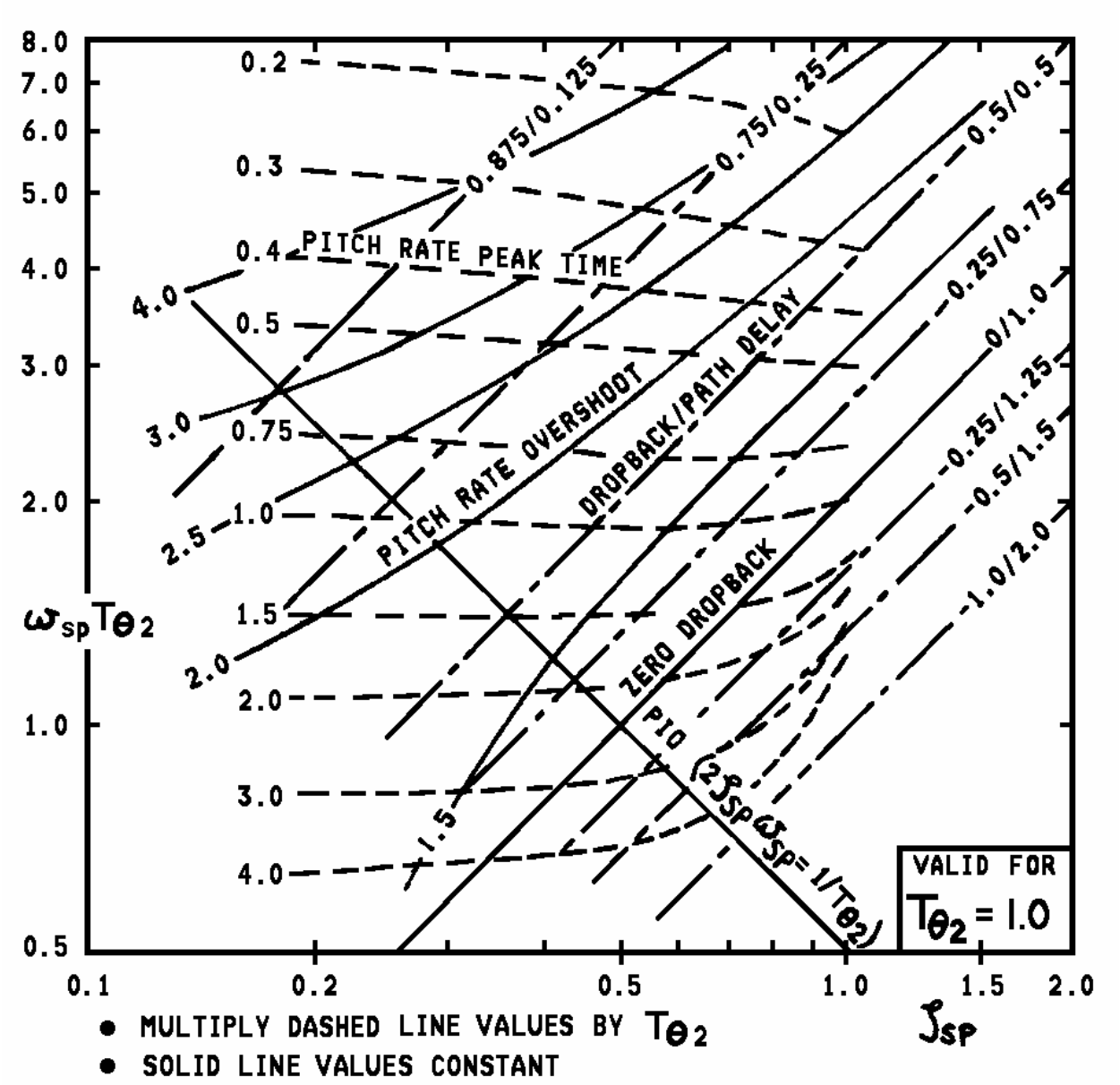


FIGURE 35. Relationships between time response parameters and modal parameters.

These time response carpets cannot be used for plotting high-order aircraft results, as they will contain a mixture of aerodynamic and FCS modes. Although the pitch attitude dropback ratio and path delay remain connected by T_{θ_2}, they and the pitch rate overshoot ratio and time to peak no longer have a unique relationship to $\omega_{sp} T_{\theta_2}$ and ζ_{sp}. For the analysis or design of such systems the time response features are considered individually. Additional high-order effects will be evident most importantly in t_q, the delay in reaching the pitch acceleration peak which is a strong indicator of PIO and handling problems when greater than 0.3 sec. The elimination of this defect is achieved by the frequency response shaping discussed in 5.2.2.1.3.2.

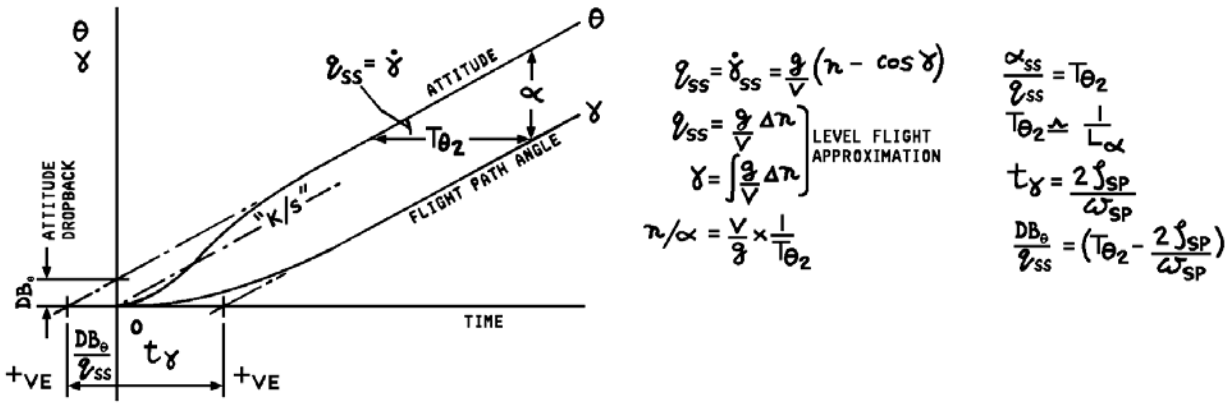


FIGURE 36. Relationship between pitch attitude and flight path responses.

A conclusion to be drawn is that for low-order aircraft with elementary pitch damper augmentation, a low maneuver margin should be aimed for, with its inherently high natural damping, if precision pitch handling is required. This is completely consistent with the excellent BAC Lightning low-altitude, high-speed (LAHS) pitch handling characteristics where in fact a frequency lower than the MIL-F-8785C minimum was satisfactory, together with only 2.0 lbs/g stick forces. The much larger TSR-2 prototype also had a low maneuver margin with good damping in the LAHS region and was taken on only its twentieth flight to 550 kts at 250 ft over hilly terrain without any stability augmentation. It was rated as having control and response well-matched to this task.

Note that pitch attitude overshoot (negative pitch attitude dropback, figure 33) and normal acceleration hang-off tendencies are sometimes found in rate-command/attitude-hold systems. One solution then is to quicken the sluggish response by addition of a feedforward directly to the control surface.

As noted above, excessive dropback results in pilot complaints of abruptness and lack of precision in pitch control—complaints common also to aircraft with excessive values of pitch attitude bandwidth. This commonality of piloting problems led to an analysis of handling qualities data to determine the applicability of the dropback criterion. When the bandwidth criteria were developed (Hoh, AGARD-CP-333), it was found that several aircraft configurations with very high levels of bandwidth frequency, ω_{BW}, received Level 2 Handling Qualities Ratings (HQRs). Because the reasons for this were not fully understood at that time, the limits on pitch attitude bandwidth as published in Appendix A of MIL-STD-1797A showed upper limits as dashed lines (figure 43).

Based on the discussion that follows, for this standard it was recommended that the upper limits on bandwidth be deleted and dropback be added as an additional element. However, the definition of pitch attitude dropback, θ_{DB}, in this standard differs from Gibson's original definition (figure 33). Figure 37 illustrates the definition of pitch attitude dropback adopted in this military standard. The boundaries on the pitch attitude dropback ratio recommended for table XXX are based on this definition. It can be seen from figure 33 that Gibson's definition of dropback, while less susceptible to low-frequency responses than the definition used in this standard, is strongly influenced by time delay: for the attitude response shown on figure 33

(where $DB_\theta/q_{ss} > 0$), addition of time delay will reduce the value of dropback without affecting pitch rate overshoot at all, while dropback as shown on figure 37 is unaffected by time delay. Since pitch attitude dropback is hypothesized as a limit on excessive mid-frequency abruptness, it is desirable to use a parameter that is not strongly affected by time delay—which is separately accounted for by the effective time delay requirement in table XXX or the phase delay requirement in the bandwidth requirements (5.2.2.1.3.1).

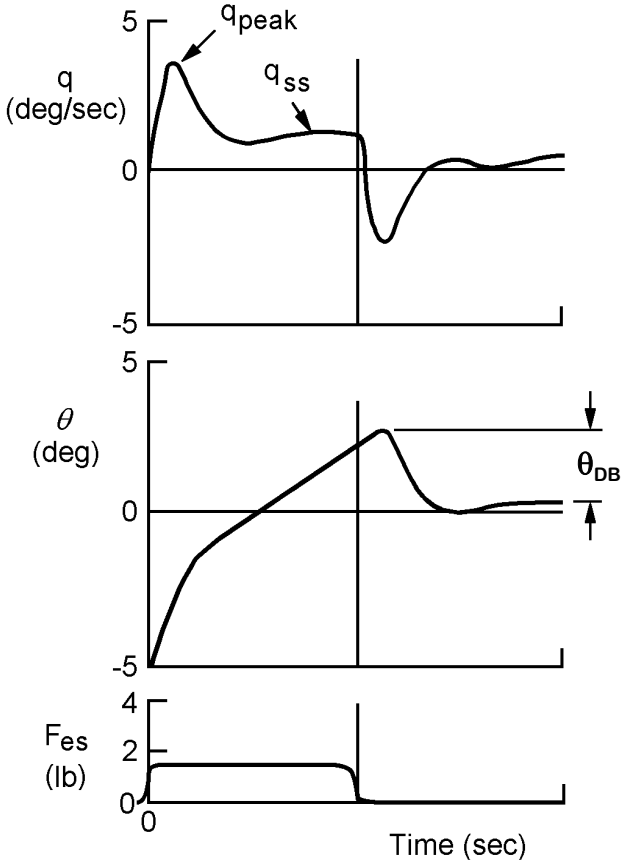


FIGURE 37. Definition of pitch attitude dropback.

An additional distinction between the two definitions of dropback is that Gibson's definition can have negative values, as shown on figure 33, while the dropback of figure 37 cannot. The trends shown by Gibson on figure 34 suggest that negative dropback is associated with sluggish aircraft—as it must be, since the only way to get negative dropback by his definition is with a large time delay, which produces a sluggish response.

The pitch tracking data of the Neal-Smith experiment (AFFDL-TR-70-74) were the basis for the Category A pitch attitude bandwidth requirements in MIL-STD-1797A (figure 43 in this standard). The dropback characteristics for those configurations evaluated in the Neal-Smith experiment with Level 1 bandwidth (i.e., greater than 6.5 rad/sec) are plotted on figure 38. figure 38 shows only one of the data points above the dropback limit with an average HQR better than 3.5 (i.e., Level 1). Dropback alone is not sufficiently discriminating, since HQRs below the

limit may be Level 1, 2, or 3. Thus, dropback is not a stand-alone criterion for good handling qualities—some other criteria must also be applied—but it does expose bad handling qualities. If the characteristics of a specific airplane are otherwise Level 1 (i.e., pitch attitude and flight path bandwidths), dropback must also be low to assure overall Level 1 handling qualities.

- **Plots Show Average HQRs for Configurations With Level 1 Bandwidths**
- **15 out of 19 Points Below the line are Level 1**
- **12 out of 14 Points Above the Line are Level 2**

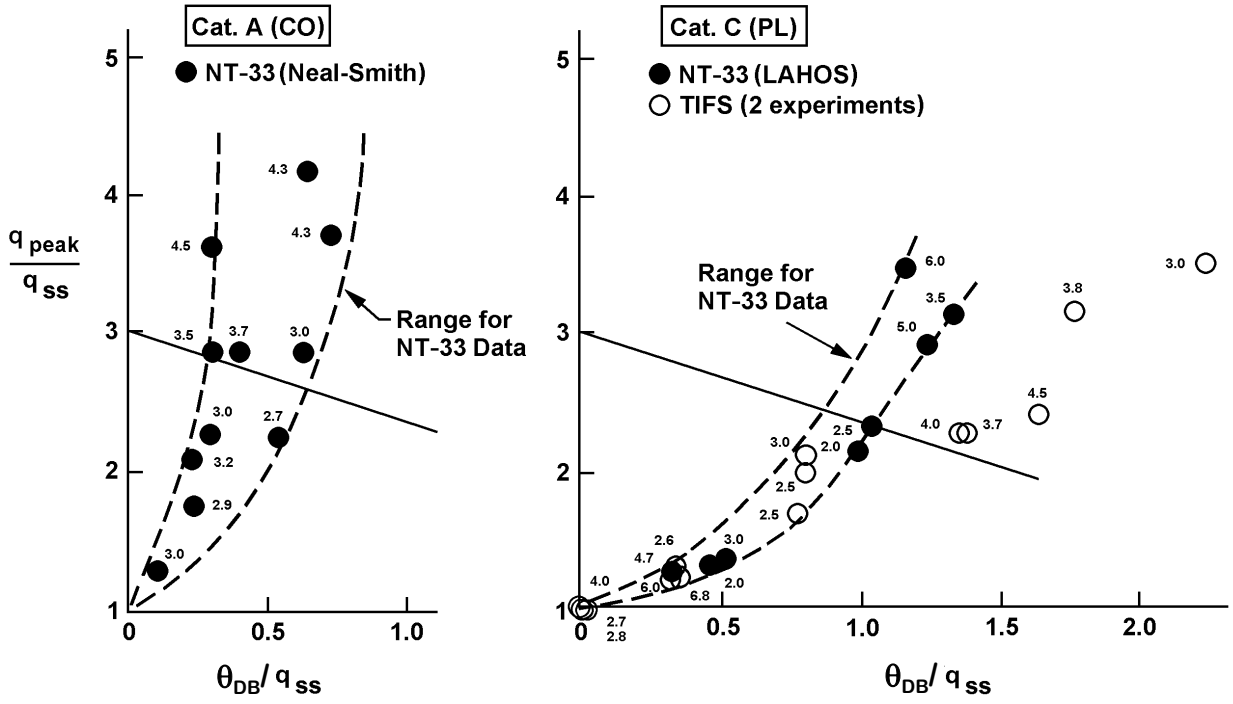


FIGURE 38. Dropback characteristics of high bandwidth configurations.

The approach and landing data of AFFDL-TR-78-122 formed the basis for the Category C bandwidth limits of MIL-STD-1797A. The dropback characteristics for the cases with Level 1 bandwidths are plotted on figure 38. Of these eight cases, two of the three with excessive dropback received Level 2 average HQRs and the third was rated 2 and 5, for an average of 3.5; four of the five below the dropback limit received Level 1 average ratings, and the fifth—which was rated only once—received an HQR of 6. (The HQR 6 case appears to have inadequate flight path bandwidth, as is shown in 5.2.2.1.3.3.) This supports the removal of an upper limit on bandwidth (5.2.2.1.3.1), as long as dropback is added as supplementary criterion to prevent excessively abrupt pitch responses.

Two flight experiments conducted on the USAF/Calspan TIFS provide new data for Class III approach and landing operations (NASA CR-172491 in 1984 and NASA CR-178188 in 1986). These data are not in Appendix A of MIL-STD-1797A, since the experiments were conducted after the writing of the 1982 draft MIL Standard and Handbook. Figure 38 includes the dropback characteristics of the otherwise Level 1 configurations.

For the TIFS data on figure 38, four of the five cases above the dropback limit received Level 2 average ratings. The fifth case, with an average HQR of 3.0, was rated by two pilots who assigned HQRs of 4 and 2. Hence even this case was considered to be Level 2 by one of the pilots.

VERIFICATION

The pitch rate response parameters of table XXX are defined below. For the step input, use controller deflection for a deflection control system (pilot controller deflections commands the control effectors) and controller force for a force control system (pilot controller force commands the control effectors).

Two straight lines are drawn on the pitch rate time history and the following measurements are defined (see figure 32):

A horizontal line defining the steady-state pitch rate.

A sloping straight line tangent to the pitch rate time history at the point of maximum slope. This line is extended to intersect the steady-state line and the time axis.

Time t_1 (effective time delay) is the time at which the maximum slope tangent line intersects the time axis (with time measured from the instant of the step controller input).

Time t_2 is the time at which the maximum slope tangent line intersects the steady-state pitch rate line (with time measured from the instant of the step controller input).

The rise time (Δt) is $t_2 - t_1$.

Δq_1 is defined as the maximum pitch rate minus the steady-state pitch rate.

Δq_2 is defined as the steady-state pitch rate minus the first minimum pitch rate

The transient peak ratio (TPR) is equal to the ratio $\Delta q_2/\Delta q_1$.

The measurements defined above shall meet the requirements of table XXX. If the residue of the phugoid or other low-frequency modes prohibits defining a constant speed, short-term, steady-state pitch rate, it may be necessary to apply the criteria to the pitch rate transient computed from constant-speed equations of motion (i.e., two-degree-of-freedom equations of motion with speed constrained).

The criteria of table XXX are based on the pulse input time response features shown on figure 33, except that pitch attitude dropback is as defined on figure 37. Satisfactory values depend on the task and on the pitch rate transients.

In practice a true step is not usually feasible. Time may be measured from the midpoint of the control input transient, as shown on figure 39, for the most abrupt input feasible. Good instrumentation will be needed to measure the time delay accurately.

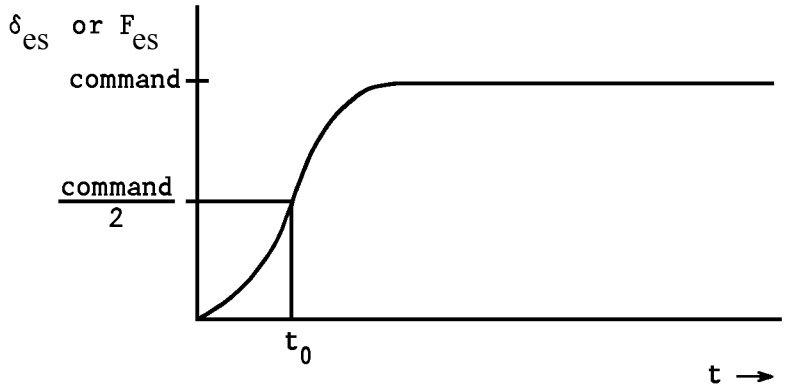


FIGURE 39. Measurement of control input time.

5.2.2.1.2.3 Steady-state flight-path response to pitch controller.
In the landing approach flight phase, the steady-state flight path and airspeed response to attitude change via the pitch controller shall meet the following requirements:

a. For flight control modes in which flight path is controlled primarily through the pitch controller, with thrust at the setting required for the normal approach glide path at $V_{o_{min}}$, the curve of flight path angle versus true airspeed shall have a local slope at $V_{o_{min}}$ which is negative or less positive than the limits of table XXXI. The slope of the curve of flight path angle versus airspeed at 5 kts slower than $V_{o_{min}}$ shall not be more than __(1)__ more positive than the slope at $V_{o_{min}}$ as illustrated by figure 40.

b. For flight control modes in which flight path is controlled using some designated flight path controller other than the pitch controller, the Level 1 requirement for flight path response to attitude change via the pitch controller may be relaxed to the Level 2 limits of table XXXI.

TABLE XXXI. Maximum slope of γ versus V_T at $V_{o_{min}}$.

Level	Maximum Slope of γ versus V_T
1	
2	
3	

REQUIREMENT RATIONALE

The accepted piloting technique for conventional flight is to adjust flight path via pitch attitude control. This requirement is included to insure that the long-term flight path response to pitch attitude changes is acceptable to the pilot. For aircraft using another specific flight path controller for primary control of flight path, relaxation of this requirement is warranted when use of such a piloting technique is deemed acceptable. Examples of when such techniques might be acceptable could be some shipboard and STOL operations.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.1.3
MIL-STD-1797: 4.3.1.2.

Recommended value for Blank (1): 0.05º/kt

Recommended values for table XXXI:

Level	Maximum Slope of γ versus V_T
1	0.06°/kt
2	0.15°/kt
3	0.24°/kt

This requirement addresses flight path response to the pitch controller. For aircraft which have modes in which flight path is meant to be commanded by some controller other than the pitch controller, see 5.2.2.2 for requirements on the response to that controller.

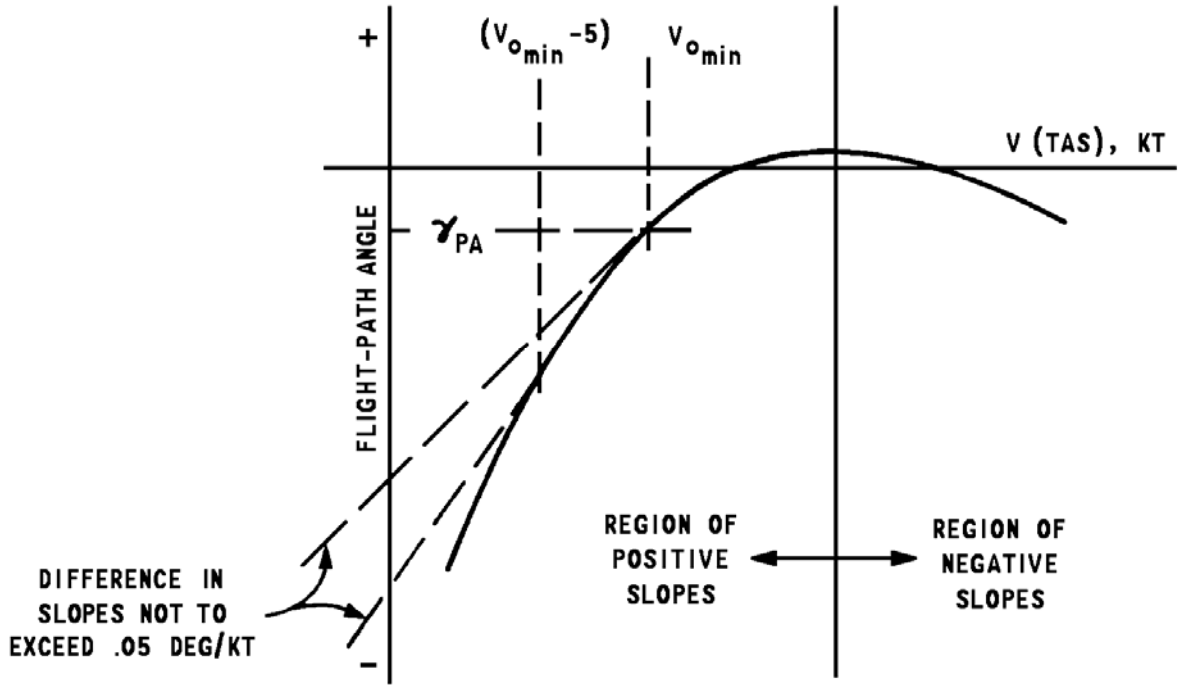


FIGURE 40. Flight path angle versus true airspeed.

REQUIREMENT LESSONS LEARNED

The first part of this requirement is applicable for most conventional aircraft Aircraft in which primary control of flight path is not with pitch attitude, such as STOLs, a relaxation of 5.2.2.1.2.3 (that is, to allow operation well on the back side of the thrust-required versus airspeed curve) shall be allowed. Discussions for this section are taken from AFFDL-TR-69-72.

Operation on the backside of the drag curve [negative d(T-D)/dV] in the landing approach leads to problems in airspeed and flight path control. Systems Technology Inc. TR-124-1, AGARD Report 122, RAE Aero. 2504, and AGARD Rpt 357 show that airspeed behavior, when elevator is used to control attitude and altitude, is characterized by a first-order root that becomes unstable at speeds below minimum drag speed. This closed-loop, constrained flight path instability, even

when the open-loop (unattended) phugoid motion is stable, is caused by an unstable zero in the $\dot{h}/\delta_e$ aircraft transfer function. Specifically, Systems Technology Inc. TR-124-1 used closed-loop analysis to show the importance of the factor $1/T_{h_1}$ as an indicator of closed-loop system stability and throttle activity required. A useful measure of the quantity $1/T_{h_1}$ is needed.

Working from the altitude-to-elevator transfer function, FDL-TDR-64-60 shows that $1/T_{h_1}$ is closely approximated (the other two zeroes generally are much larger) by the ratio D/C, where D and C are from the expression:

$$\frac{\dot{h}(s)}{\delta_e(s)} = \frac{A\,s^3 + B\,s^2 + C\,s + D}{[s^2 + 2\zeta_p\omega_p s + \omega_p^2][s^2 + 2\zeta_{sp}\omega_{sp} s + \omega_{sp}^2]}$$

The additional assumption that C is approximately equal to $V\,(Z_{\delta_e}\,M_w - M_{\delta_e}\,Z_w)$ is generally valid so that (WADC-TR-58-82):

$$\frac{1}{T_{h_1}} = \frac{D}{V\,(Z_{\delta_e}\,M_w - M_{\delta_e}\,Z_w)}$$

The climb angle, γ, is $\dot{h}/V$. Applying the limit value theorem to $\gamma(s)/\delta_e(s)$, for a step δ_e [$\delta_e(s) = |\delta_e|/s$] then

$$\frac{d\gamma}{d\delta_e} = \left.\frac{\gamma(s)}{\delta_e(s)}\right|_{ss} = \frac{1}{V}\,\frac{D}{\omega_p^2\,\omega_{sp}^2}$$

In a similar manner, the slope of steady-state u versus δ_e curve is obtained.

$$\frac{du}{d\delta_e} = \frac{u(s)}{\delta_e(s)} = -\frac{g\,(Z_{\delta_e}\,M_w - M_{\delta_e}\,Z_w)}{\omega_p^2\,\omega_{sp}^2}$$

Then the slope of the steady-state γ versus u curve for elevator inputs can be written

$$\frac{d\gamma}{du} = \frac{d\gamma/d\delta_e}{du/d\delta_e} = -\frac{1}{g}\,\frac{D}{V\,(Z_{\delta_e}\,M_w - M_{\delta_e}\,Z_w)}$$

$$= -\frac{1}{g}\,\frac{1}{T_{h_1}}$$

The $d\gamma/du$ limits, therefore, set limits on $1/T_{h_1}$.

The limit on dγ/du at 5 kts slower than $V_{o_{min}}$ was added to assure that the aircraft remains tractable at commonly encountered off-nominal speeds.

For design purposes, dγ/du can be estimated from the dimensional stability derivatives (which must include any important thrust effects) as follows:

$$\frac{d\gamma}{du} = \frac{1}{g}\left[X_u - \left(X_w - \frac{g}{V} \right)\left(\frac{Z_u - M_u\, Z_{\delta_e}/M_{\delta_e}}{Z_w - M_w\, Z_{\delta_e}/M_{\delta_e}} \right) - \frac{X_{\delta_e}}{M_{\delta_e}}\left(\frac{M_w\, Z_u - Z_w\, M_u}{-Z_w + M_w\, Z_{\delta_e}/M_{\delta_e}} \right) \right]$$

or

$$\frac{d\gamma}{du} = \frac{1}{g}\left[X_u - \left(X_w - \frac{g}{V} \right)\left(\frac{Z_u - M_u\, Z_{\delta_e}/M_{\delta_e}}{1/T_{\theta_2}} \right) - \frac{X_{\delta_e}}{M_{\delta_e}}\left(\frac{\omega_p^2\, \omega_{sp}^2}{g/T_{\theta_2}} \right) \right]$$

For small M_u and X_{δ_e}, the following approximation is valid except for very-short-tailed aircraft:

$$\frac{d\gamma}{du} \doteq \frac{1}{g}\left[X_u - \left(X_w - \frac{g}{V} \right)\frac{Z_u}{Z_w} \right]$$

It is possible to violate this requirement by operating well on the backside of the power-required curve (dγ/du > 0) and still have a Level 1 aircraft as long as some other means of controlling flight path is provided (usually power or thrust). Naturally this other controller must have satisfactory characteristics. For example, if the throttle is designated as the flight path controller, good dynamic and steady-state flight path response to throttle changes (γ/δ_T) must be assured. Although there are no quantitative data to support this, it seems logical that progressively degraded γ/θ can be compensated with incremental improvements in $(\gamma/\delta_T)_{ss}$. Examples of aircraft that have poor $(\gamma/\theta)_{ss}$ characteristics but are acceptable because flight path control is augmented with thrust are the de Havilland Twin Otter, the DHC-7, and many carrier-based fighters (e.g., Systems Technology Inc. TR-124-1). But Pinsker (RAE-TR-71021) found that an autothrottle to hold constant airspeed can be quite destabilizing if the thrust line passes below the c.g. of a statically stable aircraft. Requirements on γ/δ_T are specified in 5.2.2.2.1 based on STOL aircraft research.

Since backside operation (defined as having dγ/du > 0) is most critical during landing approach, this requirement is oriented toward that Flight Phase. To improve dγ/du requires increasing the airspeed, which has obvious performance implications. Backside operation is also troublesome for takeoff, cruise, and high-altitude maneuvering, but it will probably not be as critical as for landing approach, and there are virtually no data to define numerical limits for these Flight Phases.

In the event the aircraft is operated with a continuous flight path controller (e.g. DLC on the YC-15), which serves (one hopes) to improve the flight path response, the relaxation for aircraft with a designated flight path controller shall be allowed. In those cases the pilots shall be trained appropriately.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. The climb-angle-versus-airspeed used to demonstrate compliance can be obtained during the stabilized-airspeed tests for static stability at low airspeeds. By its nature, the climb angle to be measured is relative to the air, not the ground. When using Doppler radar or ground-based tracking equipment to obtain the data, the wind must be calm, or at least constant and accurately measured. Still air is necessary in any case, to minimize data scatter. Because of thrust and density variation it has been found necessary to keep altitude excursions small (less than 1000 ft) to get an acceptably accurate curve of flight path angle versus airspeed. The trim flight path angle can have a marked effect on the results; the range of glide slopes expected in the operational and training missions shall be tested.

In terms of non-dimensional quantities, for small γ, neglecting C_{D_δ}, C_{D_u}, C_{L_u}, C_{m_u}, and $\partial T/\partial\alpha$,

$$\frac{d\gamma}{dV} \approx -\frac{2}{U_0}\left\{\left[\left(\frac{T}{W}-\frac{V}{2}\frac{\partial(T/W)}{\partial u}\right)\cos(\alpha+i_T)-\gamma\right]\left[1+\frac{\gamma_0\, C_L/C_{N_\alpha}}{1-\frac{C_{L_\delta}\, C_{m_\alpha}}{C_{m_\delta}\, C_{N_\alpha}}}\right]\right.$$

$$\left.-\frac{C_{L_1}}{C_{N_\alpha}}\left[\frac{\frac{T}{W}\sin(\alpha+i_T)+C_{D_\alpha}/C_{L_1}}{1-\frac{C_{L_\delta}\, C_{m_\alpha}}{C_{m_\delta}\, C_{N_\alpha}}}\right]\left[1+\left(\frac{z_T}{\bar{c}}\frac{C_{L_\delta}}{C_{m_\delta}}-\sin(\alpha+i_T)\right)\left(\frac{T}{W}-\frac{V}{2}\frac{\partial(T/W)}{\partial u}\right)\right]\right\}$$

showing the effects of flight path angle, thrust offset, and thrust variation with airspeed at constant throttle.

The most straight forward measurement is probably to use a well-calibrated airspeed indicator and an accurate measure of vertical speed, such as a radar altimeter. The climb angle is then

$$\gamma = \sin^{-1}\left|\frac{\text{Vertical speed}}{\text{True airspeed}}\right|$$

5.2.2.1.2.4 Speed response to attitude change.
With cockpit controls fixed and with them free, for Levels 1 and 2, there shall be no tendency for the airspeed to diverge aperiodically when the air vehicle pitch attitude is disturbed from trim. For Level 3, the time for the airspeed perturbation to double amplitude following a disturbance from trim shall not be less than __(1)__, with controls fixed and with them free. However, in the presence of one or more other Level 3 flying qualities, no airspeed divergence shall be allowed.

5.2.2.1.2.4.1 Direction of speed change.
For Levels 1 and 2, over an equilibrium speed range of ±15% or ±50 kts about trim, whichever is less, within the ROTH, the direction of the airspeed response to pitch attitude change shall meet the following requirements:

a. For a fixed positive change in commanded pitch attitude from trim, the steady-state airspeed shall not increase. For a fixed negative change in commanded pitch attitude from trim, the steady-state airspeed shall not decrease.

b. For rapid commanded pitch attitude changes the short-term airspeed change shall be in the same direction as the steady-state airspeed change.

REQUIREMENT RATIONALE

This requirement is intended to assure that there is no aperiodic instability in attitude and airspeed during either piloted control or periods of unattended pilot operation. The first part of this requirement insures positive static stability for unattended operation (cockpit controls fixed or free) for Levels 1 and 2 and limits the amount of negative static stability for Level 3. The second requires that airspeed track pitch attitude in the conventional way (airspeed decreases with increasing pitch attitude) both in the long term and short term, in order to assure ease of piloting.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.1.1
MIL-STD-1797: 4.4.1.

Recommended value for Blank (1): 6 sec

REQUIREMENT LESSONS LEARNED

For either augmented or unaugmented aircraft, static stability means that restoring pitching moments are generated when the airspeed is disturbed from trim. Airspeed is easily measured—it is always available as a cockpit display. Furthermore, in most circumstances airspeed is a more meaningful cue to most pilots than angle of attack, which is not displayed at all in some aircraft, and not much used except at low speed by some pilots.

From a piloting standpoint, pitch attitude is a primary longitudinal control variable. Pilots quickly learn that good control of pitch attitude generally leads to good control of airspeed and flight path. Hence it would seem possible that the desire for angle of attack stability is more related to pitch attitude than airspeed. Evidence to support this point of view can be found in good pilot acceptance of some systems with pitch rate command/attitude hold augmentation, which has zero static stability in the classical sense (control force or deflection versus airspeed at constant throttle) but is allowed by this requirement. This static stability requirement supplements 5.2.2.1.1.1 and 5.2.2.1.2.1, which require a positive phugoid damping ratio.

For all but the most unconventional aircraft, the requirement on airspeed/attitude consonance is satisfied automatically throughout the ROTH, in which the flying qualities requirements generally apply. The ratio is determined by the ratio of transfer function numerators

$$\frac{u(s)}{\theta(s)} = \frac{N_\delta^u}{N_\delta^\theta} = \frac{(X_\alpha - g)\,(s + 1/T_{u_1})}{(1 + M_{\dot{w}}\, Z_\delta / M_\delta)\,(s + 1/T_{\theta_1})\,(s + 1/T_{\theta_2})}$$

For $X_\delta = 0$ (expression adapted from WADC-TR-58-82), where δ generically represents either pitch control force or deflection, $1/T_{u_1}$ and $1/T_{\theta_2}$ are generally positive (stable). From WADC-TR-58-82,

$$\frac{1}{T_{u_1}} \doteq \frac{\zeta_{sp}\,\omega_{sp}}{\frac{\partial C_D}{\partial C_L^2}\, C_{N_\alpha} \left[1 - \frac{C_{m_q} + C_{m_{\dot{\alpha}}}}{2\,k_y^2\, C_{N_\alpha}}\right]}$$

will always be roughly the same as $\zeta_{sp}\omega_{sp}$ but can be either greater or smaller. Since

$$\frac{1}{T_{h_1}} = \frac{1}{T_{\theta_1}} - g\,\frac{Z_u}{Z_\alpha}\left[\frac{1 - (Z_\delta / M_\delta)\,(M_u / Z_u)}{1 - (Z_\delta / M_\delta)\,(M_\alpha / Z_\alpha)}\right]$$

$1/T_{\theta_1}$ goes through zero at an airspeed below that for zero $1/T_{h_1}$ (see 5.2.2.1.2.3). As airspeed is decreased, $1/T_{\theta_1}$ becomes negative (unstable) somewhere between the speed for zero $d\gamma/dV$ and stall speed, and usually below V_{min}. A pilot tightly closing an inner attitude loop will tend to drive one of the closed-loop characteristic loops (of the transfer function denominator) into $1/T_{\theta_1}$. This consonance requirement is incorporated in this standard primarily to insure acceptable airspeed response characteristics for augmented aircraft that may have unconventional airspeed response to changes in attitude, or aircraft with an unusually broad range of negative $1/T_{\theta_1}$.

Most aircraft that meet the equivalent phugoid and short-period requirements of 5.2.2.1.1 through 5.2.2.1.2.2 should automatically meet the requirements of this section because of the inherent relationships between pitch attitude and airspeed. However, aircraft with some form of direct force control (such as DLC or autothrottles) may modify the classical attitude/airspeed

relationship significantly. For example, a tight autothrottle loop will result in essentially zero airspeed change with changes in pitch attitude. In some flight conditions, with an offset or a tilted thrust line it is conceivable that the autothrottle could result in increasing airspeed with increasing pitch attitude. Such undesirable characteristics would be disallowed by this requirement.

In accordance with classical definitions of static stability the prohibition of airspeed divergence will be considered satisfied if the gradients of pitch control force and deflection with airspeed are negative, that is, if the aircraft will return toward its trim airspeed after a speed disturbance, controls fixed or free. Insisting on a stable control position gradient can result in significant restriction in the aircraft c.g. limit if no more than downsprings and bobweights are used to improve controls-free stability. That limits aircraft utility. Downsprings and bobweights have been utilized on many successful aircraft—particularly commercial transports to augment the stick force gradient when the stick position gradient is unstable, although overdoing that can induce a dynamic instability. The requirement on deflection gradient has been kept here as a reflection of the potentially greater demands of military missions and the tendency of an arm supported by a thigh or armrest to restrict stick motion when the pilot is grasping the control. With the use "response feel", through feedback of normal acceleration and pitch rate in place of bobweights, downsprings, etc. and a fully-powered system, controls-fixed and controls-free stability become the same.

It is well recognized that the stick force gradient with airspeed is an important flying qualities metric. Unfortunately, there seems to be no general agreement on what constitutes the lower boundary short of neutral stability. Hence, it is required only that the gradient not be unstable. For Level 3, refer to Lessons Learned for 5.2.2.1.1.2. The data supporting this requirement are the same as the data supporting the phugoid requirements in 5.2.2.1.1.1 and 5.2.2.1.2.1 and the Level 3 requirement in 5.2.2.1.1.2.

The need for a convergent airspeed response for acceptable flying qualities is well recognized. Recent experience in testing modified control laws on the F-16 has shown that some level of speed stability is helpful in the approach flight condition. The rest of the augmentation insures good attitude stability with or without angle of attack feedback. However, pilots indicated that the speed cue (provided by the angle of attack feedback) was necessary at high angle of attack to avoid inadvertent stalls near the ground: an angle of attack signal comes just below approach speed as a cue of stall approach, though at normal approach speed F-16 pilots prefer a pitch rate command.

The DFVLR variable-stability Hansa Jet used spoilers for direct lift control. It was found that without further compensation, the spoiler drag more than negated the improvements in flight path control due to DLC.

The allowance of a divergent airspeed response for Level 3 is based on ground-based and in-flight simulation studies related to the Boeing SST, the B-1, and other configurations that have shown the apparent feasibility even of instrument landing with instabilities as great as 6 sec to double amplitude. For prolonged flight with other high pilot workload, the less acute attention to piloting may further limit the allowable divergence. Actually, with complete pilot attention

somewhat quicker divergences have appeared controllable. Some allowances for workload, design and loading uncertainties, pilot unfamiliarity with instability, and a range of possibilities for total damping have been made. With regard to the last factor, Schuler (AFWAL-TR-82-3014) has shown that the 6-second limit is not necessarily conservative.

VERIFICATION

A succinct description of measurement techniques for compliance with this requirement appears as Appendix IVA of AFFDL-TR-69-72 and can be summarized as follows:

The obvious method of determining the stick force gradient is to first trim the aircraft then use the pitch attitude control alone to change and restabilize airspeed, leaving the throttle and trimmer controls at their trim settings. The altitude, of course, will vary constantly during this test; but careful programming of the test sequence can keep the altitude within reasonable bounds, for subsonic speeds. At low speed this test gives an excellent indication of phugoid stiffness or any divergent roots, though it is time-consuming. But at higher speeds, larger altitude changes are encountered during the runs. Then the lack of any unique relation between altitude and speed can cause difficulty, because compressibility effects are functions of both h and V. For example, gross differences in apparent stability are common transonically between results of airspeed variation at constant Mach number and results of Mach variation at constant airspeed. Neither of these latter two tests gives the desired result, which is an indication of long-term stability.

The acceleration-deceleration method is popular because it is the quickest. After retrimming, the aircraft is decelerated to the specified lower limit of the speed range by reducing power and holding altitude constant with the elevator. The aircraft is next accelerated to the maximum specified speed and then decelerated to the trim speed. All this is done at constant altitude. The method is fast and data can be recorded continuously during the maneuver. One practical problem, however, is that unless the pilot changes power slowly and moves the elevator smoothly so that normal acceleration is held very close to 1.0, the data will include unwanted contributions from δ_e/n (constant speed).

At low speeds, the control force versus airspeed gradients obtained by the two methods will be essentially equal. At higher speed, air density and the speed of sound will vary appreciably during static stability testing using the stabilized method (not during acceleration-deceleration tests, in which altitude is constant). It is not obvious which type of test most accurately measures static stability. It is obvious, however, that the stabilized method is very time-consuming and exhibits poor repeatability at high speed. For this reason, the acceleration-deceleration method is generally preferable for testing at high speeds.

A possible source of error with the acceleration-deceleration tests should be mentioned. The tests are usually conducted using off-trim throttle settings. The pitching moment and vertical force changes with speed at an off-trim throttle setting may be significantly different from those obtained at the trim throttle setting. Thrust line offset from the c.g. and inlet airflow turning are sources of the difference. This difference can be accounted for by a priori knowledge of engine thrust and slipstream effects, or a small difference can be averaged out.

In view of the above discussion, the following techniques are recommended as a reasonable compromise between accuracy and practicality. At low speeds, where the altitude changes associated with constant-throttle airspeed changes are small and where operation near the stall speed is required, the constant-power stabilized airspeed method works very nicely. At high speeds (M > 0.4), where altitude excursions associated with the stabilized-airspeed method become larger, economy considerations dictate that some form of the acceleration-deceleration method be employed. To insure that the results of the test give reasonable indication of throttle-fixed stability, hold normal acceleration as close to 1.0 g as is possible without use of abrupt control movements; data shall only be taken during the parts with the throttle at the trim setting. The combined effect of thrust and acceleration can be seen by comparing acceleration and deceleration data, but for showing compliance with this requirement only the data for the trim throttle setting are pertinent. For climbing or descending Flight Phases, other appropriate throttle settings shall be used; but the acceleration-deceleration runs are still to be conducted in level flight.

In testing for compliance, if the control gradients obtained for a number of trim points are stable over the specified speed range, relatively few trim points will be needed. If an unstable region is found far from the trim point, however, the test shall be repeated with the aircraft trimmed closer to the unstable region; the aircraft may or may not be stable within the specified speed range about the new trim point.

Aircraft having certain types of stability augmentation, such as rate-command/attitude-hold or maneuver-command systems, will have zero stick force gradients with airspeed. For these aircraft, the flight tests conducted to satisfy the phugoid stability requirements of 5.2.2.1.2.1 shall be utilized to show compliance with this requirement.

In terms of effective stability derivatives, speed stability is neutral when the dynamic characteristic equation's coefficient of s^0 is zero, indicating one real root at the origin. From AFFDL-TR-65-218, this coefficient is known as E and:

$$E \propto \cos\gamma_0 \left[(C_{L_\alpha} + C_D)\left(\frac{V}{2}\frac{\partial C_m}{\partial u} - \frac{T}{\bar{q}\,S}\frac{z_T}{\bar{c}} + \frac{z_T}{\rho\,V\,S\,\bar{c}}\frac{\partial T}{\partial u} \right) - C_{m_\alpha}\left(C_L + \frac{V}{2}\frac{\partial C_L}{\partial u} + \frac{\sin\xi}{\rho\,V\,S}\frac{\partial T}{\partial u} \right) \right]$$
$$+ \sin\gamma_0 \left[(C_L - C_{D_\alpha})\left(\frac{V}{2}\frac{\partial C_m}{\partial u} - \frac{T}{\bar{q}\,S}\frac{z_T}{\bar{c}} + \frac{z_T}{\rho\,V\,S\,\bar{c}}\frac{\partial T}{\partial u} \right) + C_{m_\alpha}\left(C_D + \frac{V}{2}\frac{\partial C_D}{\partial u} - \frac{\cos\xi}{\rho\,V\,S}\frac{\partial T}{\partial u} \right) \right]$$
$$= 0$$

where $\bar{q} = \rho\,V_0^2/2$, z_T is the normal distance from the aircraft center of gravity to the thrust line (positive for thrust line below c.g.), and $\xi = \alpha + i_T$ is the inclination of the thrust line to the direction of motion. The expression is only half as complicated for level flight ($\gamma = 0$), and in cases where the u derivatives are negligible:

$$\frac{C_{m_\alpha}}{C_{L_\alpha} + C_D} = -\frac{T\,z_T}{W\,\bar{c}}$$

as the condition for neutral stability. C_{m_α} of course varies with c.g. location, being more stable (negative) at forward c.g.'s.

AFFDL-TR-65-218 also gives methods to analyze the effects of spring tabs, downsprings, etc.

5.2.2.1.2.4.2 Relaxation in transonic flight.
The requirements of 5.2.2.1.2.4 and 5.2.2.1.2.4.1 may be relaxed in the transonic speed range provided that any divergent air vehicle motions or reversals in slope of pitch control force and position with speed are gradual. In no case, however, shall the requirements of 5.2.2.1.2.4 and 5.2.2.1.2.4.1 be relaxed more than the following:

a. Levels 1 and 2: In 1-g flight, no local gradient of pitch control force versus speed shall be more unstable than __(1)__ nor shall the change in pitch control force exceed __(2)__ in the unstable direction. This relaxation does not apply to Level 1 for any Flight Phase which requires prolonged transonic operation.

b. Level 3: In 1-g flight, no local gradient of pitch control force versus speed shall be more unstable than __(3)__ nor shall the pitch control force ever exceed __(4)__ in the unstable direction.

The contractor shall include in the detailed system definition a statement delineating if the relaxation is applicable and for which Flight Phases it is to be applied.

REQUIREMENT RATIONALE

Aircraft naturally exhibit local instabilities in the transonic region. Operation in this region is often transient in nature, and limited amounts of static instability have not been troublesome for the short term.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.1.1.1
MIL-STD-1797: 4.4.1.1.

Recommended values for Blanks (1) through (4):

Type of Controller	Blank (1)	Blank (2)	Blank (3)	Blank (4)
Centerstick	3 lbs per 0.01 M	10 lbs	6 lbs per 0.01 M	20 lbs
Wheel	5 lbs per 0.01 M	15 lbs	10 lbs per 0.01 M	30 lbs
Sidestick				

Also, see 5.2.2.1.8.3.

REQUIREMENT LESSONS LEARNED

The extent of the region that may be considered transonic has been left unspecified because of the difficulty in stating a definition that can be applied with generality. It is not the intent to define the transonic region at that where a relaxation is necessary; such a definition would leave essentially no requirement for stability. The lower end of the transonic region might be taken as the drag-rise Mach number, but the intent is not to penalize a design for minor speed instability in a region of only transient flight. The upper bound might be the Mach number at which the lift and drag approach the classical $(M^2 \cos^2\Lambda - 1)^{-1/2}$ variation with freestream Mach number, where Λ is the sweepback angle. A key point is that the relaxation shall not apply at any flight condition at which an operational or training mission requires operation for any length of time. Transonic for the A-10, defined by the onset of compressibility effects, is Mach > 0.5, and is encountered routinely for protracted periods, in tactical operations. The lesson is that low is relative.

Since phugoid oscillations involve speed changes, all speeds at which operational missions might require prolonged flight shall be reasonably far removed from the region of transonic trim changes. Otherwise, normally encountered disturbances could cause divergence.

VERIFICATION

See Verification for 5.2.2.1.2.4. The u-derivatives can be interpreted as having

$$\frac{V_0}{2}\frac{\partial C_m}{\partial u} = \frac{M}{2}\frac{\partial C_m}{\partial M}$$

etc., where M is Mach number.

Transition from subsonic to supersonic flight makes C_{m_α} more negative. Then, for a given Mach number change, the ΔC_m will be larger at higher angle of attack. Thus, a larger transonic trim change would be expected at high altitude, where the region is traversed at higher angle of attack.

With increasing speed, C_{m_α} may first become less stable before starting to increase at transonic speeds. But, the stable shift as flow becomes supersonic causes a tuck-under tendency. In the same Mach number range a flap-type elevator will decrease in effectiveness as the local flow becomes supersonic.

5.2.2.1.3 Longitudinal frequency response to the pitch controller.
The longitudinal frequency response to the pitch controller shall meet the requirements of 5.2.2.1.3.1 through 5.2.2.1.3.3.

REQUIREMENT RATIONALE

For cases with no good equivalent system match, frequency response criteria involving Nichols charts or the bandwidth of the actual higher-order system shall be used.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.2.1.2 and 4.3.1.1.

REQUIREMENT LESSONS LEARNED

VERIFICATION

Verification shall be via verification of requirements 5.2.2.1.3.1 through 5.2.2.1.3.3. As the design progresses, analysis, simulation, and flight test will be appropriate demonstration means. In the end, flight test will measure some parameters and collect data to validate analytical and simulator results for the rest.

5.2.2.1.3.1 Pitch attitude bandwidth.
The bandwidth and phase delay of the pitch attitude response to pilot pitch control input shall be within __(1)__.

REQUIREMENT RATIONALE

A measure of the handling qualities of an aircraft is its stability margin when operated in a closed-loop compensatory tracking task. The maximum frequency at which such closed-loop tracking can take place without threatening stability is referred to as bandwidth (ω_{BW}). It follows that aircraft capable of operating at a large enough value of bandwidth will have superior performance when regulating against disturbances. A bandwidth criterion is especially useful for highly augmented aircraft in which the response characteristics are non-classical in form (i.e., have large mismatch in equivalent system fits). Although not restricted to such cases, this requirement shall be utilized when the mismatch between the lower-order and higher-order systems exceed the values defined on figure 22. No assumption of pilot dynamics is necessary in applying this requirement since such assumption would simply shift the boundaries. Also, for Level 1 minimal pilot compensation shall be necessary.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.2.1.2.

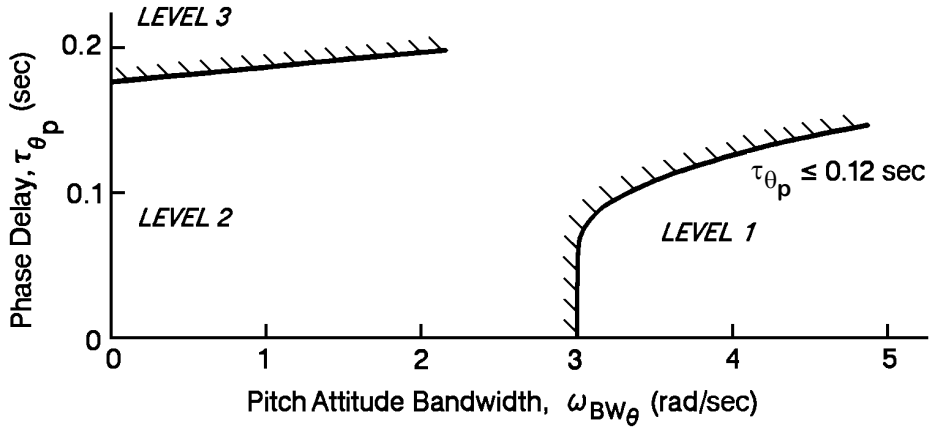
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REQUIREMENT LESSONS LEARNED

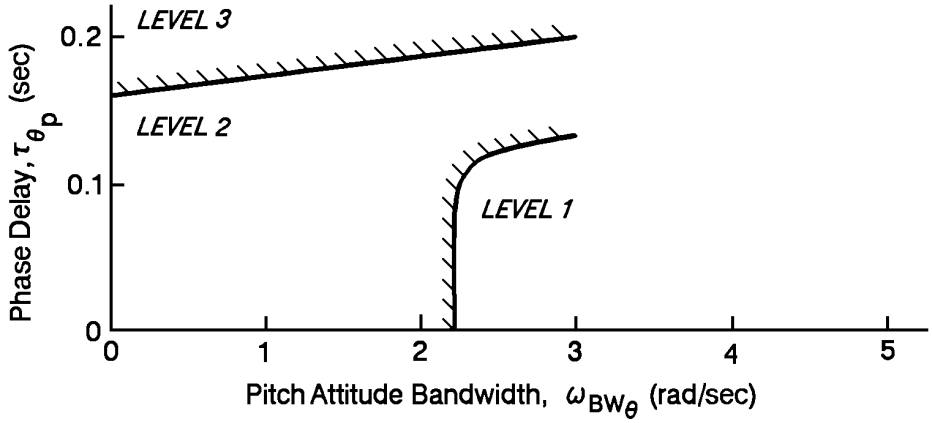
The following discussion is adapted from WL-TR-94-3162:

The pitch attitude bandwidth criteria were developed for highly-augmented aircraft (Hoh, et. al.; AGARD-CP-333) and were included as alternatives in MIL-STD-1797. The requirements have continued to evolve since the publication of that military standard, and the current Army helicopter flying qualities specification, ADS-33C, uses bandwidth as the basic criterion for pitch, roll, and yaw. In their present form, the bandwidth criteria—augmented with a pitch attitude dropback criterion (5.2.2.1.2.2)—are highly effective in defining flying qualities for both pitch attitude and flight path control for all airplane Classes and all Flight Phase Categories.

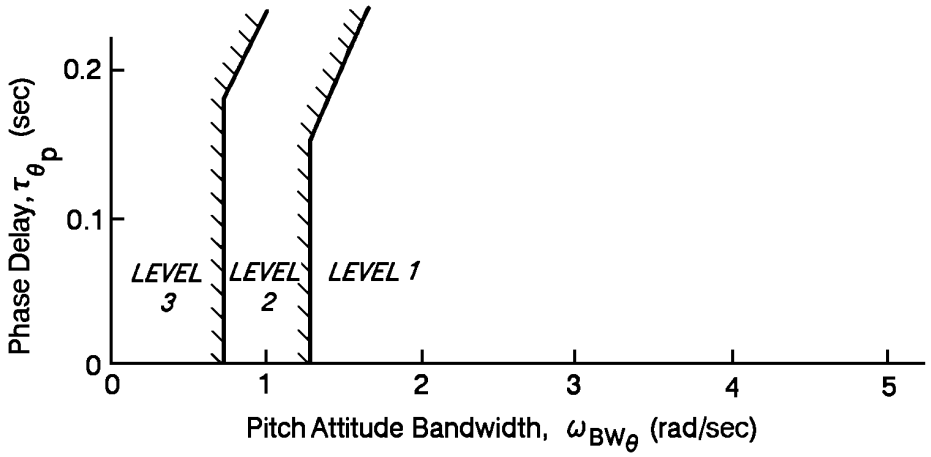
Most, if not all, familiar handling qualities metrics are in fact related to bandwidth. However, these metrics are generally tailored to classical aircraft which can be characterized by lower-order systems—for example, the q/F_s and α/F_s transfer functions of 5.2.2.1.1.



a) Categories A, all Classes



b) Categories B and C, Class IV



c) Categories B and C, Classes I, II, and III

FIGURE 41. Bandwidth requirements.

It is easily shown for these (and similar) transfer functions that the quality of closed-loop error regulation depends on the pilot's ability to increase the short-period root (ω_{sp}) without driving it into the right-half plane (unstable). As illustrated by the generic sketches on figure 42 for an idealized pilot supplying only gain and pure delay, aircraft with low short-period damping (ζ_{sp}), frequency (ω_{sp}), or both, tend to become unstable at low values of frequency [compare figure 42 a) and b)]. The aircraft of figure 42 is represented as a simple short-period vehicle to simplify the example; for real highly-augmented aircraft, many more roots are involved.

Consider the bandwidth frequency as occurring at some (for now) arbitrary margin below the frequency of instability (see boxes on root locus on figure 42). It can be seen from figure 42 that ω_{BW} depends uniquely on ω_{sp}, ζ_{sp}, $1/T_{\theta_2}$, and τ_{θ_p} (the delay τ_{θ_p}, draws the locus to the right as gain increases). Hence, these familiar flying qualities metrics are, in fact, a measure of bandwidth. Again, it is evident the flying qualities application of bandwidth has roots in familiar metrics.

The present impetus for using ω_{BW} as a criterion evolved from attempts to develop a flying qualities standard for aircraft utilizing unconventional response modes with direct force controls (wings-level turns, pitch pointing, etc.), documented in AFWAL-TR-81-3027. The infinite variety of responses that could occur due to coupling within and between axes made it necessary to retreat to a more fundamental metric, which turned out to be bandwidth. Strictly speaking, bandwidth in pitch involved θ/θ_c, a closed-loop describing function of pilot-vehicle response. Here, however, a pilot model is merely a unity-gain feedback and so bandwidth is specified in terms of the aircraft-alone gain and phase margins only.

When the Bandwidth criteria were originally developed (AGARD-CP-333), it was found that several aircraft configurations with very high levels of Bandwidth frequency, ω_{BW}, received Level 2 Handling Qualities Ratings (HQRs). Because the reasons for this were not fully understood at that time, the limits on pitch attitude bandwidth as published in Appendix A of MIL-STD-1797A show upper limits as dashed lines (figure 43).

The data from Neal-Smith (AFFDL-TR-70-74) on figure 43 (from MIL-STD-1797A) are compared with the original bandwidth Category A requirements. These results were reasonable, though there were a number of Level 2 ratings at high values of bandwidth. The abbreviated pilot comments (taken from AFFDL-TR-70-74 and AFFDL-TR-74-9) indicate that abruptness and oversensitivity become a problem when ω_{BW} is large. This is especially true of the AFFDL-TR-74-9 pilot ratings (given in parentheses on figure 43). A possible upper boundary on ω_{BW} was shown on figure 43 to account for this problem. This boundary was considered tentative at the time because the issue of over-responsiveness was not completely understood. Complaints of abruptness and lack of precision are also common to aircraft with excessive pitch attitude dropback. This commonality of piloting problems led to an analysis of handling qualities data to determine the applicability of the dropback criterion. Based on the evidence that follows, it was recommended upper limits on bandwidth be deleted in the next version of MIL-STD-1797 (as indicated by the lack of an upper boundary on ω_{BW} on figure 41) and pitch attitude dropback be added as an additional requirement (5.2.2.1.2.2).

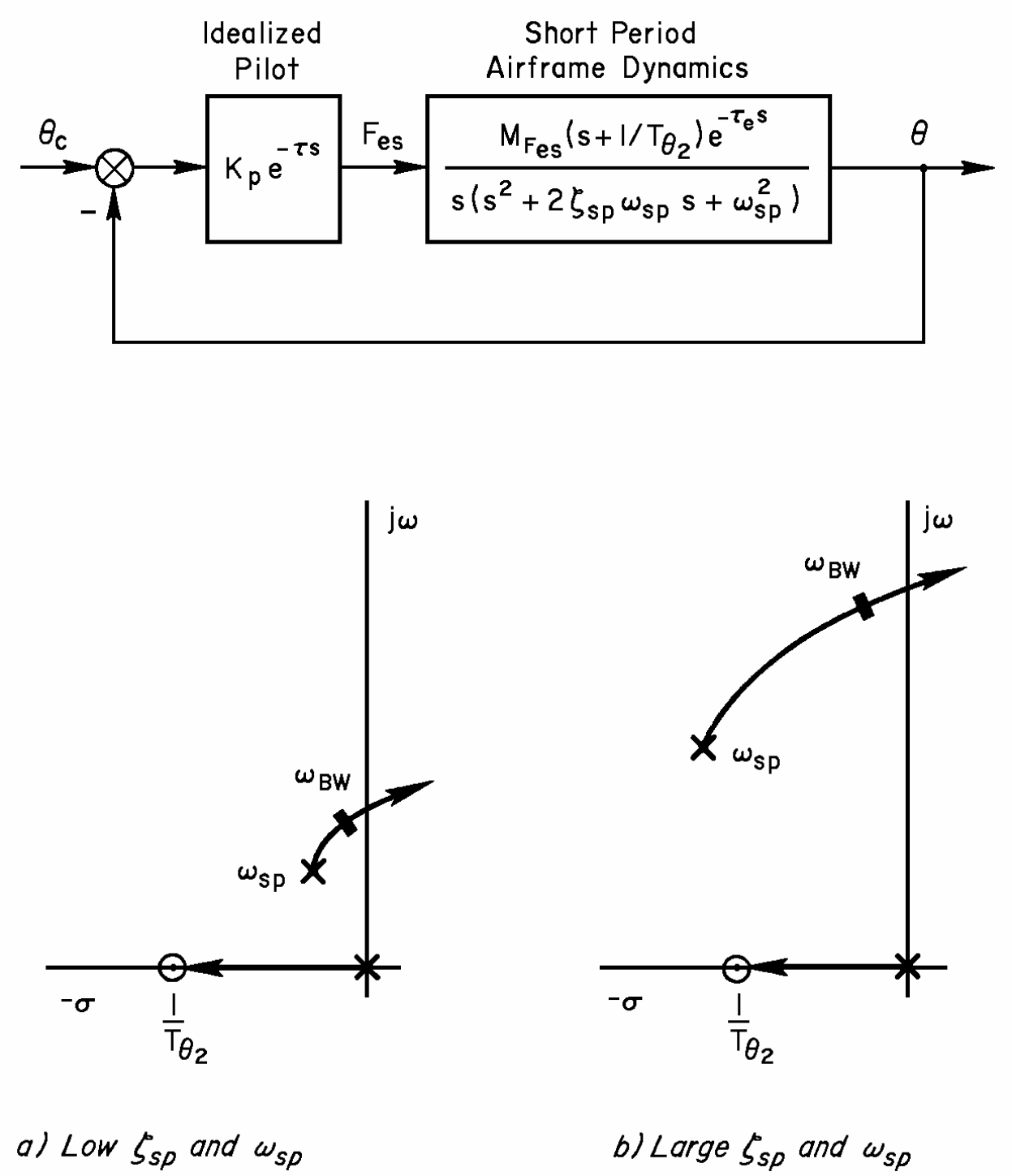


FIGURE 42. Simplified pilot-vehicle closure for pitch control.

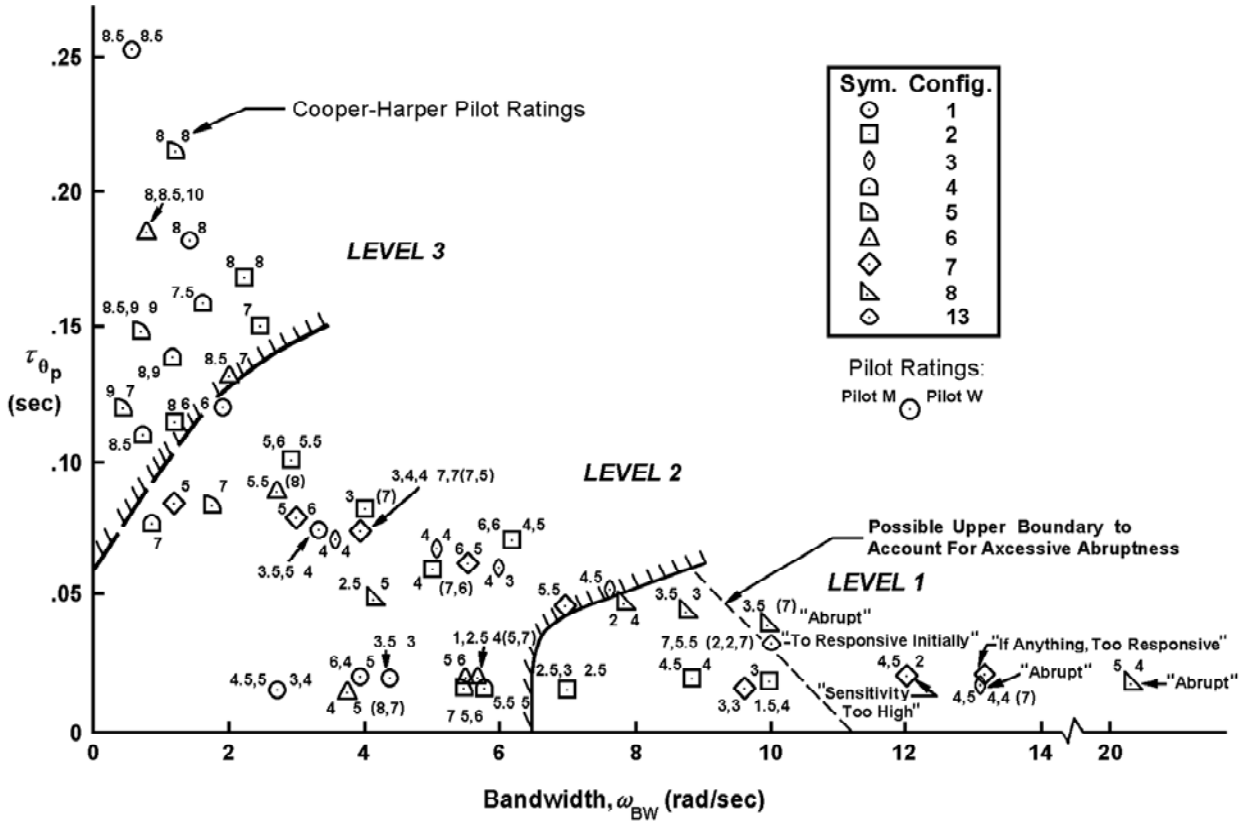


FIGURE 43. Correlation of pilot ratings with original ω_{BW} and τ_{θ_p} requirements for Neal-Smith data (Category A) (Data from AFFDL-TR-70-74, ratings in parentheses from AFFDL-TR-74-9).

Class III, Category C boundaries

The bandwidth and phase delay boundaries for transports (Class III) in precision landing (Category C) are based primarily on two NASA-sponsored studies using in-flight simulation to investigate the flying qualities of transport aircraft in the approach and landing task (NASA CR-172491 and NASA CR-178188). The aircraft used in both experiments was the Air Force / Calspan TIFS. The 1984 experiment explored the effect of different flight control system gains and feedback architecture applied to seven different aerodynamic models. The approach in the 1986 experiment was to define the exact model pole/zero placement directly to achieve a desired command response.

The aircraft in the 1984 experiment was defined to be a generic 193,000-lb transport with relaxed static stability. Seven different bare aircraft models were implemented, studying the effects of variations in $1/T_{\theta_2}$ (0.4,0.7, and 1.0 rad/sec) and ω_{sp} (1.8 and 2.8 rad/sec), neutral static ability, and location of the center of rotation. Several flight control configurations were applied to the aircraft models in various combinations.

The task in the 1984 experiment was a precision approach, flare, and landing. After being given control of the aircraft on the downwind leg, the evaluation pilot made a visual turning approach to intercept the ILS Glideslope 1.5 to 2 miles from touchdown. The final approach began with a 200 ft lateral offset. The pilot was required to track the ILS Glideslope beam to a point 3500 ft from touchdown to disallow "duck under." During the approach, a computer-generated (1-cos ωt) angle-of-attack gust was applied. The pilot was required to align with the runway centerline and land the aircraft, after which the safety pilot initiated a climbout.

The touchdown area was partitioned into "desired" and "adequate" areas (20 ft x 500 ft and 40 ft by 1000 ft, respectively). In addition, the pilots were instructed to hold airspeed as close as possible to 132 KIAS during the approach. Holding the airspeed to within ± 3 KIAS as the altitude "barrier" passage was considered "desired" and a variation of ± 5 KIAS was considered "adequate." Furthermore, the sink rate at touchdown was rated as "desired" if 0-3 fps; and adequate if 3-6 fps.

In the 1986 experiment, the simulated generic transport model was the same as that used in the 1984 experiment. Rather than wrapping control loops around a predefined airframe model, however, the poles, zeros, and sensitivity of the closed loop system were directly set a priori. Specifically, the values of ζ_{sp}, ω_p, ω_α, $1/T_{\theta_2}$, and n_Z/α were perturbed in various combinations about a known Level 1 base system. The force-feel and lateral-directional systems, as well as the flight controls, were essentially those of the 1984 experiment. Thrust was designed to respond in the speed degree of freedom only (for an idealized frontside configuration), and had a first-order time constant of 0.2 sec. The task in the 1986 experiment was exactly the same as that for the 1984 experiment. This set of TIFS data has the advantage that more pilots—as many as six, compared to two in the 1984 study—participated in the experiment, which justifies a higher confidence level for certain ratings.

A plot of τ_{θ_p} versus ω_{BW} for the TIFS data is shown on figure 44. The identifiers used in the experiment, as well as the HQRs (or converted HQRs) for each configuration, are indicated, as is the general response-type. Pilot HQRs separated by comma were given by the same pilot; a "/" indicates a change of pilots. For example "3,3,4/5/2" would mean that three different evaluation pilots rated the configuration, and one of them rated it three times.

There were many more points in the database than are shown on figure 44. Only those points relevant to defining the τ_{θ_p} versus ω_{BW} Level 1 and Level 2 boundaries are included on figure 44. That is, points with excessive dropback, and points that are not Level 1 due to excessively low or high ω_{BW_γ}, are not included in these plots, because the flying qualities of these configurations suffer for reasons unrelated to pitch attitude bandwidth or phase delay.

Although the placement of the phase delay and attitude bandwidth boundaries was primarily based on the TIFS data, additional large aircraft data (AFFDL-TR-65-210, Boeing D6-10725, NASA CR-182066, and NASA TN D-3971) were considered as supporting information, and used to refine boundary locations where there were no TIFS cases for precise definition.

Nevertheless, the degree of correlation of this additional large aircraft data with the recommended bandwidth requirements is comparable to that for the TIFS data.

Two in-flight simulation studies were performed during the design phase of the proposed Boeing version of the C-5A heavy military transport, around 1965. The first was conducted by the Cornell Aeronautical Laboratory (using a variable-stability B-26) to gain an understanding of the control problems faced by pilots of the new aircraft during the landing task (AFFDL-TR-65-210). This was later expanded upon by Boeing in a parallel series of ground-based simulations and flight tests using a Boeing 367-80 (prototype of the 707) modified for in-flight simulation of their C-5A design (Boeing D6-10725).

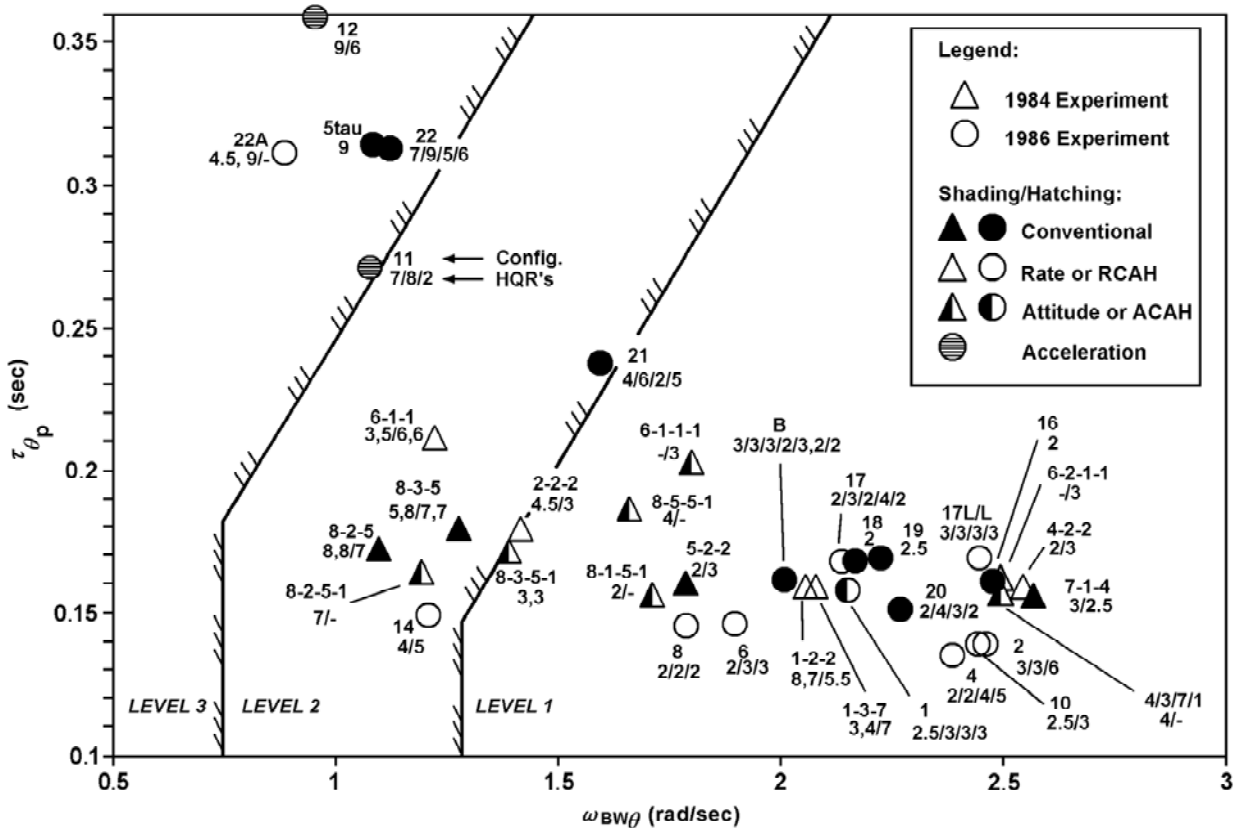


FIGURE 44. Pitch attitude bandwidth versus phase delay (TIFS experiments).

The Boeing simulations considered a matrix of variations in longitudinal short-period frequency and damping ratio. The data included here from the Cornell investigation cover short-period frequency variations only.

The pilots participating in the Cornell test program were presented the following task, quoted from AFFDL-TR-65-210:

The evaluation task for this program was an ILS approach to low altitude at the airfield. The evaluation pilot flew under the hood for some time prior to intercepting the outer marker. He

went under the hood while on a dogleg outside the outer marker and approximately 1000 ft above outer-marker intercept altitude. Thus, the pilot began the ILS interception by having to reduce his altitude and to acquire the localizer prior to glide slope interception at ILS approach down to 200 ft above runway altitude. At the 200-ft point, he removed the hood and proceeded VFR to the airfield and a very low approach. The approach was carried sufficiently low that the pilot had to initiate a flare, and, in some instances, the flare was continued to touchdown or almost to touchdown.

The following is taken from Boeing D6-10725 and describes the task the evaluation pilots were required to perform during the Boeing experiment:

1. Establish a 500 fpm rate of descent and perform a flare (repeated for 1000 fpm). (This maneuver evaluated the flight path and speed control characteristics and the flare capabilities of each configuration.)

2. Perform a 'roller coaster' maneuver. (This maneuver evaluated the pitch rate response, pitch damping, and capabilities of each configuration.)

3. ILS approach. (This maneuver evaluated the precision control characteristics. A longitudinal offset was performed by flaring to level flight at 1000 ft altitude and then pushing over and recapturing the glide slope. This maneuver evaluated the large amplitude correction maneuver capabilities.)

Clearly, both sets of tasks in the C-5A experiments were much less stringent than those performed by the evaluation pilots in the TIFS experiments. There were no supplementary lateral-correction, gust-alleviation, or airspeed-hold tasks, all of which were present in the TIFS experiments. There were no "desired" and "adequate" touchdown point areas specified; the pilots were not necessarily even required to land. While the C-5A results have proven useful in establishing the correlations between bandwidth and flying qualities, the nature of the experiments warrants caution in interpreting the results when comparing across databases.

The test pilots in both the Cornell and Boeing programs rated the configurations using the CAL ten-point rating scale, which is similar to, but not the same as, the Cooper-Harper HQR scale (AIAA 90-2823). It has been common in the past to simply plot CAL and Cooper-Harper ratings on the same plot, but this is not correct. A procedure was developed for this effort to convert from the CAL and Cooper rating scales to equivalent HQRs. This procedure is described in WL-TR-94-3162. This difficulty, combined with the aforementioned lack of specificity of the task, lowers the credibility of the C-5A data as compared to the TIFS data.

The NASA CR-182066 simulation evaluated candidate response-types for a Task-Tailored Flight Control System for the landing of a large, advanced, hypersonic transport. A general set of vehicle dynamics typical of this class of aircraft was utilized, and variations of three basic advanced response-type categories were studied. In addition to Rate Command/Attitude Hold, the relatively untested Flightpath Command/Flightpath Hold and Flightpath Rate Command/Flightpath Hold categories were explored in various ways.

All of the evaluations took place using the moving-base (hexapod) Visual/Motion Simulator at NASA Langley Research Center. Initial tests were conducted using a transport-style column and wheel as the primary flight control device, but most of the evaluations used a centerstick. The longitudinal force-feel system for the column was a second-order actuator:

$$\frac{\text{control position}}{\text{control force}} = \frac{50.41}{[1.15,\ 7.1]}$$

and that for the centerstick was:

$$\frac{\text{control position}}{\text{control force}} = \frac{3.75}{(3.75)}$$

The pilot was provided with high-fidelity sound cues (i.e., engine noise), as well as motion and visual information. The task was a precision manual flare and landing with touchdown position and sink rate requirements: desired performance was touchdown within a 54 ft x 1500 ft region with a sink rate less than 6 ft/sec, adequate performance was touchdown within a 174 ft x 5000 ft region with a sink rate less than 8 ft/sec. Additionally, the pilots were required to maintain airspeed at 178 ± 10 kts during the approach, and were not allowed to "duck under," or fly an approach flatter than the 3º ILS glideslope. A light-to-moderate level of turbulence was applied, and runs were made in conditions of no wind, high head- and tailwind, and heavy windshears.

The data from NASA TN D-3971 are the results of a NASA investigation of flying characteristics of fixed and variable-geometry SST designs, simulated in-flight on a Boeing 367-80. Added to the basic aircraft configuration were various combinations of variable geometry, pitch rate augmentation, aft center of gravity location, and supplementary angle of attack feedback.

The following excerpt from NASA TN D-3971 summarizes the approach and landing task performed by the evaluation pilots:

For this task, an intercept of the localizer was made with landing gear down approximately 8 miles from the runway at an altitude of 1500 ft. The flaps and airspeed were then adjusted for the landing approach as required by the simulation. At the intercept of the glide slope, approximately 5 miles from the runway, a descent was initiated and the pilot attempted to fly that prescribed flight path as closely as possible down to approximately 200 ft and, if conditions were favorable, continue visually to touchdown. Some tests were made with the localizer offset 200 ft from the runway center line during the approach to evaluate the lateral maneuverability. Following the simulated IFR breakout to line up with the runway. Other tests were also made with square-wave vertical offsets of the glide slope approximately halfway down the glide slope to study the speed-thrust stability and longitudinal maneuverability while the pilot was under the hood.

The test sequence also included a series of basic VFR maneuvers at altitude, which unfortunately were factored into the test pilots' evaluations, casting some doubt on their applicability to

Category C flight phases exclusively. Again, there were no specific performance requirements for landing, and landings were performed only "if conditions were favorable." This experiment had notable elements in common with the TIFS experiments; for example, the lateral offset correction as a secondary task, and the flare was continued all the way to touchdown. The data are considered inferior to the TIFS data for the purposes of this analysis, however, for at least two reasons: the Cooper scale was used by the evaluation pilots to rate the configurations, which necessitates a translation to Cooper-Harper HQRs; and the approach and landing task was not as stringent and not rated separately from the other VFR maneuvers.

A plot of τ_{θ_p} versus ω_{BW} for the additional large aircraft data is shown on figure 45. Again, there were many more points in the database than are shown on figure 45. Only those points relevant to defining the τ_{θ_p} versus ω_{BW} Level 1 and Level 2 boundaries are included on figure 45. That is, points with excessive dropback, and points that are not Level 1 due to excessively low or high ω_{BW_γ}, are not included in these plots, because the flying qualities of these configurations suffer for reasons unrelated to pitch attitude bandwidth or phase delay.

Class IV, Category A boundaries

The valid database for these boundaries is the familiar Neal-Smith experiment (AFFDL-TR-70-74). Plots of pitch attitude bandwidth versus phase delay for the Neal-Smith data are presented on figure 46. Figure 46 includes all cases. This figure forms the basis for the proposed boundaries on figure 41. The feel system has been included for calculation of parameters, even though this experiment used force commands. The addition of dropback as a requirement, along with the inclusion of the feel system, has resulted in pitch attitude Bandwidth limits that are well below those that were published in MIL-STD-1797A: the Level 1 limit, for example, has dropped from 6.5 to 3 rad/sec.

The proposed boundaries correlate the Neal-Smith data extremely well. With one exception, every case in the Level 1 region on pitch attitude bandwidth that is not Level 1 fails either the flight path bandwidth (5.2.2.1.3.3) or dropback requirements (5.2.2.1.2.2). The single exception is a case rated a 4 by one pilot. Similarly, there is only one case with a Level 1 rating in the Level 2 region on figure 46 (a configuration rated a 3 by one pilot), and all level 3 cases are correctly explained by either excessive phase delay or excessive dropback.

Class IV, Category C boundaries

The LAHOS data (AFFDL-TR-78-122) were used for these boundaries. The data are shown on figure 47. The pitch attitude bandwidth boundaries are well supported by the data. Again, most of those cases that are not correctly predicted by the boundaries are cases of either excessive pitch attitude dropback (5.2.2.1.2.2) or insufficient flight path bandwidth (5.2.2.1.3.3).

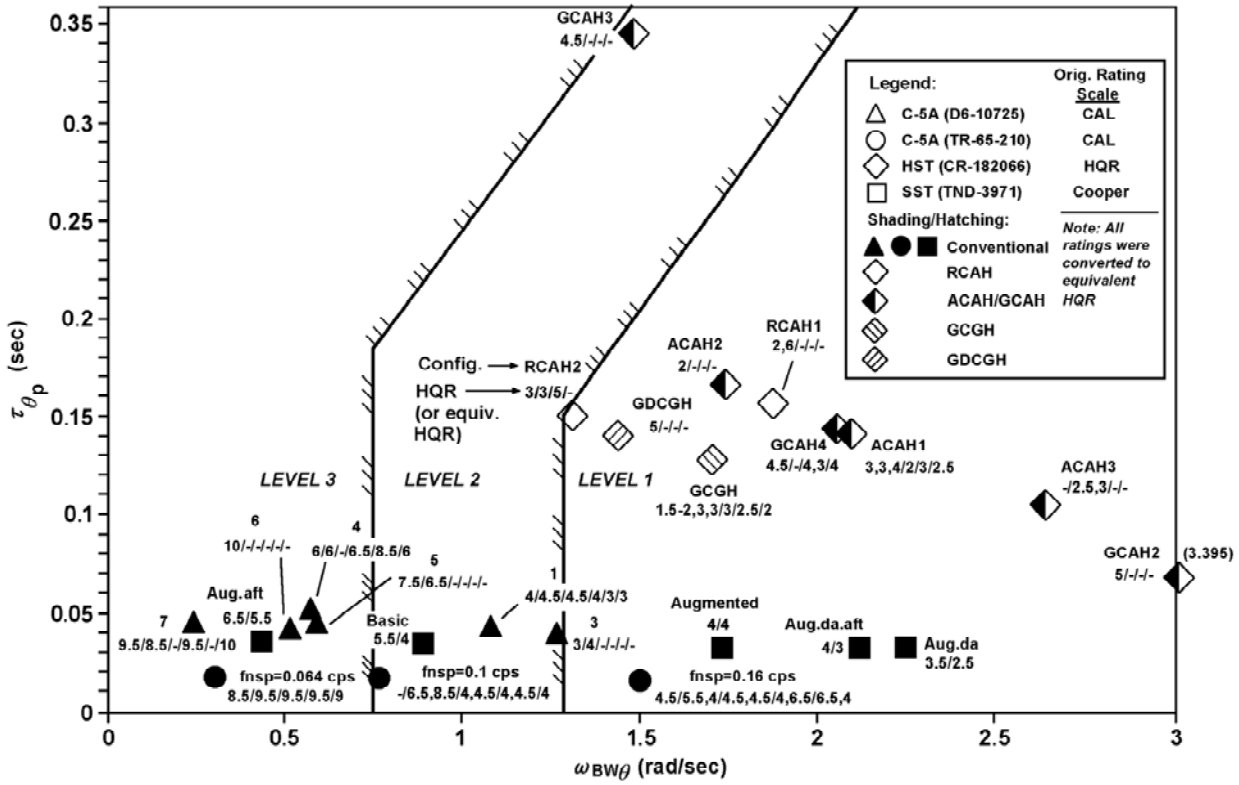


FIGURE 45. Pitch attitude bandwidth versus phase delay (additional large aircraft data).

VERIFICATION

For handling qualities criterion purposes, the bandwidth, ω_{BW}, is defined as the highest frequency at which the frequency response of aircraft pitch attitude to pilot control inputs has both 45º or more of phase margin and 6 dB or more of gain margin (see figure 48). In reference to figure 48, this describes the pilot's ability to double his gain or to add a time delay or phase lag without causing an instability ($\Phi \leq -180º$ at the ω for 0 dB gain indicates instability). In order to apply this definition, one first determines the frequency for natural stability, ω_{180}, from the phase portion of the Bode plot. The next step is to note the frequency at which the phase margin is 45º, ω_{135}. This is the bandwidth frequency as defined by phase, $\omega_{BW_{phase}}$. Finally, note the amplitude corresponding to ω_{180} and add 6 dB. Find the frequency at which this value of response magnitude occurs; call it $\omega_{BW_{gain}}$. The bandwidth, ω_{BW}, is the lesser of $\omega_{BW_{phase}}$ and $\omega_{BW_{gain}}$.

If $\omega_{BW} = \omega_{BW_{gain}}$, the system is said to be gain-margin limited; that is the aircraft is driven to neutral stability when the pilot increases his gain by 6 dB (a factor of 2). Gain-margin-limited aircraft may have a great deal of phase margin, Φ_M, but then increasing the gain slightly causes a large decrease in Φ_M. Such systems are characterized by frequency-response amplitude plots that are flat, combined with phase plots that roll off rapidly, such as shown on figure 48.

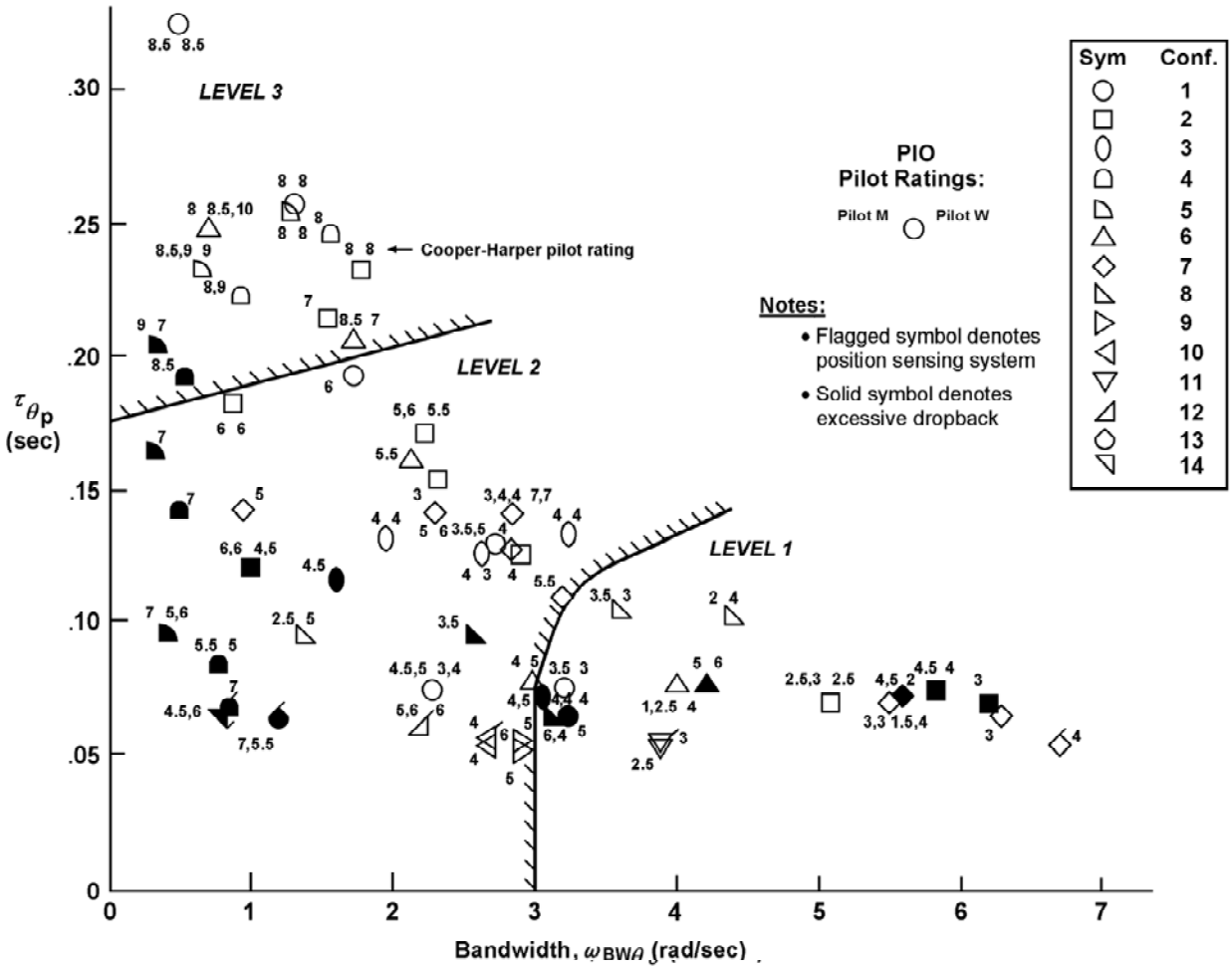


FIGURE 46. Neal-Smith data on pitch attitude bandwidth boundaries.

A steep phase dropoff means less ability to increase bandwidth unless lead is added, or less increase in bandwidth for a given increase in pilot lead. Hence, the ability of the pilot to attain good closed-loop regulation without threatening stability depends not only on the value of the bandwidth, ω_{BW}, but also on the shape of the phase curve at frequencies above ω_{BW}. Phase delay, τ_{θ_p}, is defined as

$$\tau_{\theta_p} = -(\phi_{2\omega_{180}} + 180°)/(57.3 \times 2 \times \omega_{180})$$

where ω_{180} is the frequency corresponding to -180º phase and $\phi_{2\omega_{180}}$ is the phase angle at twice that frequency.

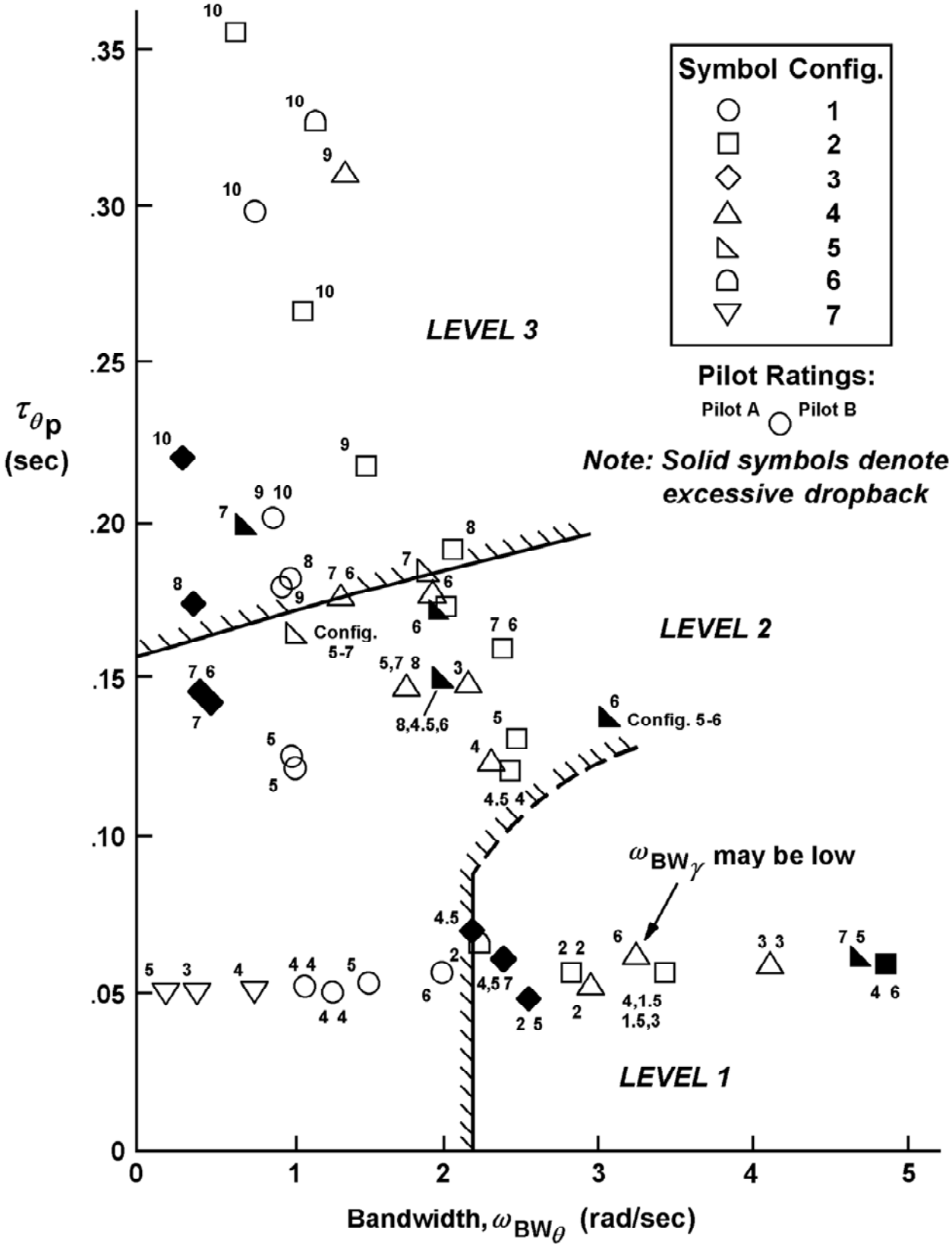


FIGURE 47. LAHOS data on pitch attitude bandwidth boundaries.

The values of τ_{θ_p} and ω_{BW} required to demonstrate compliance with the figure 41 boundaries are obtained from open-loop frequency responses of pitch attitude. These plots may be obtained from analysis (figure 48) or from Fourier-transformed flight test or simulator data. The Air Force Flight Test Center (AFFTC) has had considerable success in Fourier-transforming flight test data taken during operational tasks (as opposed to specially-tailored frequency sweeps). This saves flight test time and allows configuration identification at the flight condition to be utilized operationally.

If significant nonlinearities are present in the system, the open-loop frequency response will depend on the size of the input used in the identification process. When such nonlinearities are suspected, several frequency sweeps shall be accomplished with different input magnitudes. Data taken during operational tasks will implicitly account for nonlinear effects if technically-sound data can be obtained.

Phase Delay:

$$\tau_{\theta_p} = \frac{\Delta\Phi 2\omega_{180}}{57.3\,(2\omega_{180})}$$

Note: if phase is nonlinear between ω_{180} and $2\omega_{180}$, τ_{θ_p} shall be determined from a linear least squares fit to phase curve between ω_{180} and $2\omega_{180}$

Rate Response-Types:

ω_{BW} is lesser of $\omega_{BW_{gain}}$ and $\omega_{BW_{phase}}$

Attitude Response-Types:

$\omega_{BW} \equiv \omega_{BW_{phase}}$

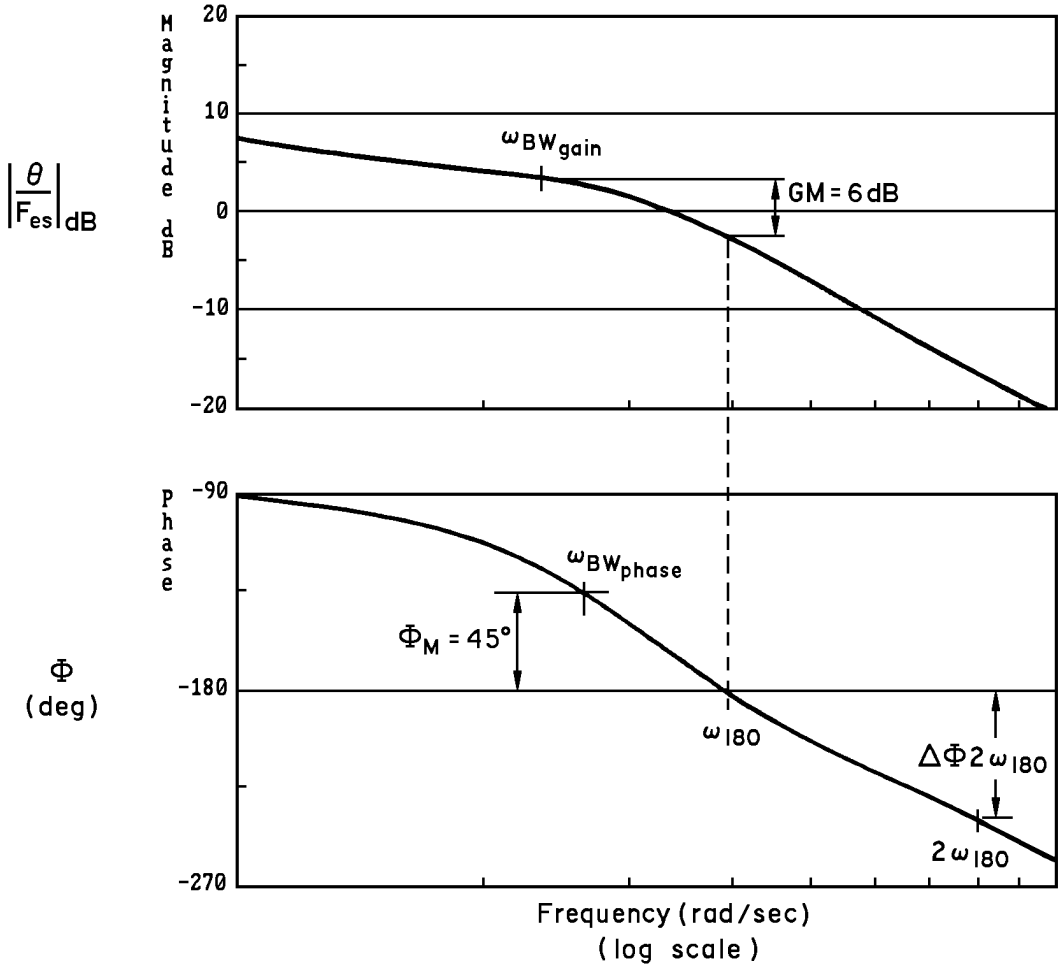


FIGURE 48. Definition of bandwidth and phase delay from open-loop frequency response.

5.2.2.1.3.2 Pitch attitude frequency response envelopes.
The pitch attitude frequency response shall lie within __(1)__.

REQUIREMENT RATIONALE

Frequency-response criteria involving Nichols charts shall be used for cases with no good equivalent system match.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.2.1.2.

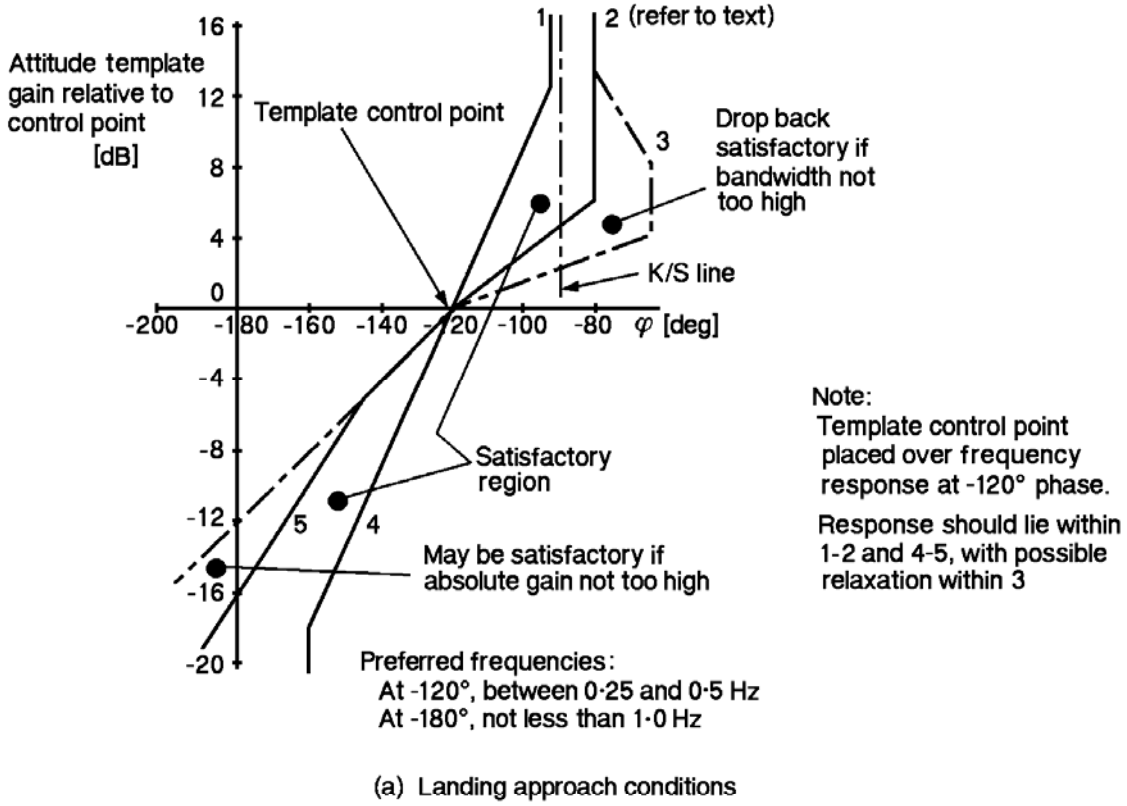
Recommended value for Blank (1): "the envelopes shown on figure 49"

REQUIREMENT LESSONS LEARNED

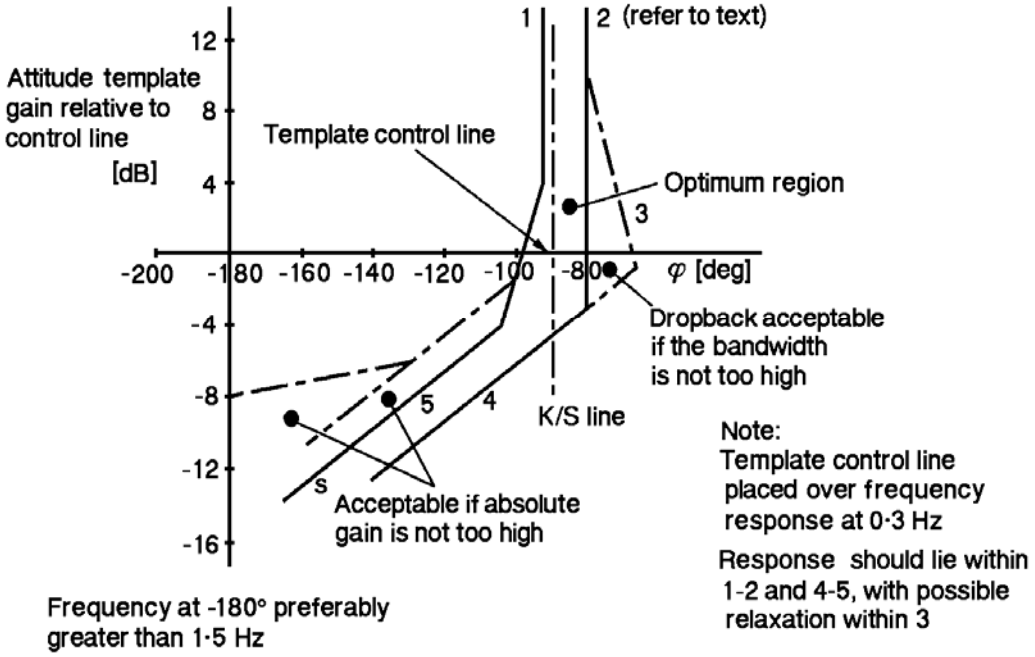
The following discussion by Gibson is extracted from AGARD-CP-333.

Much work was done in attempting to predict pilot opinion from closed-loop analysis, though this does not seem to have been followed in more recent years with the exception of the Neal-Smith criterion (5.2.2.1.4). With the advent of the computing power potential of digital fly-by-wire it is now much more useful to the flight control system designer to attempt to shape the aircraft frequency response into a form known to be attractive to the pilot, with which he can perform both well and easily and hence will result in a good opinion rating. The pilot model which achieves this aim is well known to be the simple gain and time delay, the latter always being present in random error tracking. It is possible to define an envelope of aircraft attitude response which is very robust, in the sense that the pilot can achieve good closed-loop control with a wide range of gain and delay only. In this approach it is unnecessary to define a pilot-vehicle bandwidth since he has a wide choice according to the needs of the task.

The classical aircraft dynamics which have always been shown to achieve the best ratings in simple tracking experiments are a pure-gain pitch rate response and the resulting attitude response of K/s. Real aircraft have inertia, control power limits, and pilots who dislike excessive acceleration, and can only be represented by this model at low frequencies. These attitude responses are indicated on figure 50 using the Nichols' chart form on which open-loop and closed-loop responses are related. Because of this facility, these charts are often more useful to the FCS control law designer than the more usual Bode plots, even where no pilot model is being added to the aircraft response. On figure 50 a pilot gain and delay model is added to a pure low-order attitude response to show good closed-loop performance with negligible droop or resonance. The gain is chosen to give the pilot-vehicle open-loop crossover frequency of 0.3 Hz, and a small delay typical of simulation results is selected. A K/s response with the same crossover frequency is included for comparison with the aircraft response.



(a) Landing approach conditions



(b) Up-and-away flight conditions

FIGURE 49. Pitch attitude frequency response envelopes.

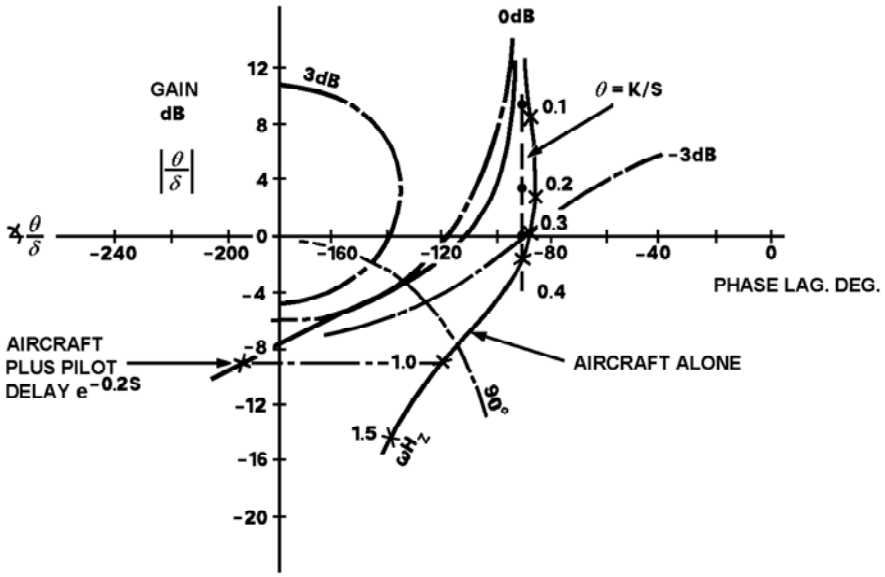


FIGURE 50. Low-order pilot-aircraft attitude frequency response.

This basic pilot model assumption underlies the aircraft response boundary limits used as a design criterion. The crossover frequency typifies the upper end of the 1 to 2 rad/sec range and the 0.2-sec time delay typifies the pilot delay noted generally in the literature in simulation experiments. WADC-TR-57-520 measured the difference between flight and simulation to show that, while the lead or lag equalization did not change, the pilot gain was lower and the time delay was larger in flight.

Choice of these values therefore represents an upper limit on pilot performance in the definition of aircraft response boundaries. The choice of frequency in Hz rather than rad/sec is deliberate since the pilot sees frequency behavior in terms of its period or cycles per second, and this serves to present a more obvious view of the effect of such boundaries.

Figure 49b shows optimum aircraft pitch attitude response boundaries for precision tracking tasks, in which a crossover frequency of 0.3 Hz is inherently achieved with a low pilot gain appropriate to an aircraft with low stick force per g. If this criterion is satisfied, the allowable pilot phase lag for optimum tracking can be attributed to his time delay and no further equalization is required from him. If the pilot chooses a lower crossover frequency, the allowable lag increases and he can adopt a larger time delay without departing from a good closed-loop performance.

These boundaries do not represent overall Level 1 limits. Depending on the task, responses outside them can be very satisfactory. General characteristics associated with areas outside the boundaries are indicated, and were derived from correlation with comment data from AFFDL-TR-74-9 and up-and-away flight configurations. While the boundaries represent the smaller number of cases which were optimum for all tasks, some cases attained Level 1 ratings for flight refueling despite bobbles severe enough to degrade pitch tracking to Level 2, and the best flight refueling case (really excellent) was close to the boundary confines but was a poor Level 2 for

gross maneuver because of unpredictable g response. Despite this it had excellent flight path control with small delay, provided that aggressive control was not attempted. More generally, attenuation greater than 5 or 6 dB/octave near the crossover frequencies is associated with sluggish, unpredictable flight path control.

The overall open-loop boundaries on figure 49a were derived from the AFFDL-TR-78-122 cases. The satisfactory range of bandwidth limit frequencies lies between about 0.25 and 0.5 Hz.

VERIFICATION

To apply this criterion the aircraft attitude response to stick input is plotted so that its open-loop amplitude at 0.3 Hz is 0 dB. The phase difference between this and the 0 dB closed-loop line represents the allowable pilot phase lag for optimum tracking. Boundaries of satisfactory frequency response for Category A precision attitude tracking are shown on figure 49b. Responses outside the low-frequency limits with the handling tendencies indicated may be satisfactory for other tasks, as determined by time response requirements. An envelope of satisfactory Category C landing approach response for frequencies below the required bandwidth of 0.25 to 0.5 Hz at 120º phase lag is shown on figure 49a.

5.2.2.1.3.3 Transient flight-path response to pitch attitude change.
The relationship between flight path response and pitch attitude response, for pilot control inputs shall have the following characteristics:

a. The flight path bandwidth frequency shall be within __(1)__.

b. For flight control modes in which a designated controller other than the pitch controller is the primary means of controlling flight path, then the Level 1 requirement on flight path bandwidth may be relaxed to coincide with the Level 2 boundaries.

c. In all cases, the pitch attitude response shall lead the flight path angle by the margins given in table XXXII, and shall have a magnitude equal to or greater than the flight path angle.

TABLE XXXII. Phase difference between pitch attitude and flight path angle responses.

Flight Phase Category	Level	Minimum phase difference between pitch attitude and flight path angle at $\omega=\omega_{sp}$
A	1	
	2	
B	1	
	2	
C	1	
	2	

REQUIREMENT RATIONALE

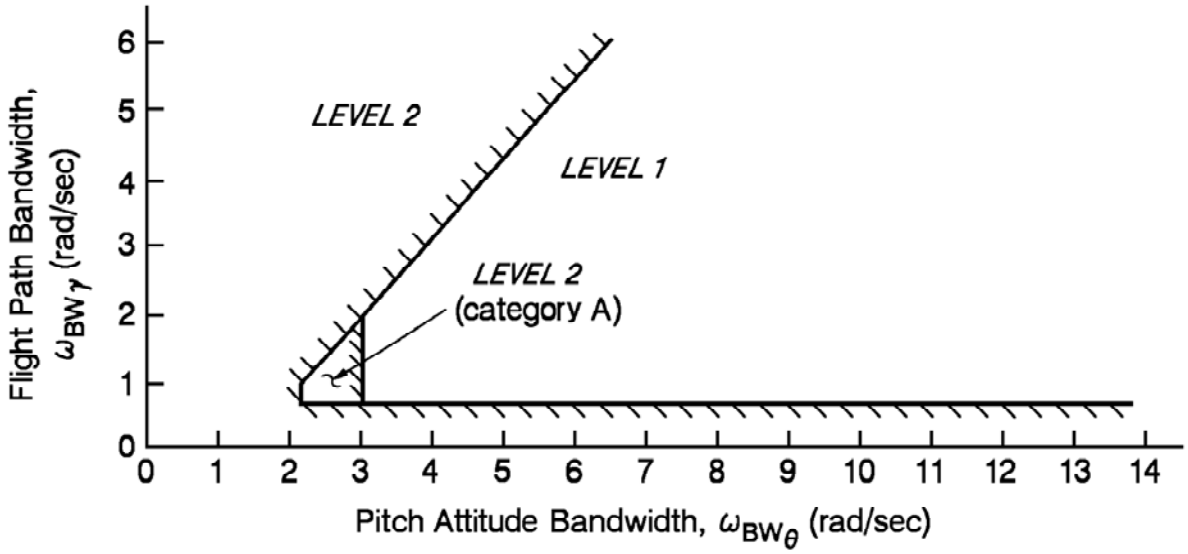
This requirement is included to provide a separate and independent criterion for flight path response to pitch attitude changes. It is intended to insure that the consonance between flight path and pitch attitude is consistent with the pilot's expectations. When pitch attitude is the primary short-term controller of flight path, excessively abrupt or sluggish flight path response to attitude changes will cause problems for precise control and possibly lead to pilot-in-the-loop oscillations. If the flight path response to attitude is not adequate a separate controller shall be provided. Different criteria are necessary for conventional aircraft (for which pitch attitude is the primary means for flight path control) and for STOL aircraft (for which pitch attitude plays a secondary role in path control or is used only to control speed).

REQUIREMENT GUIDANCE

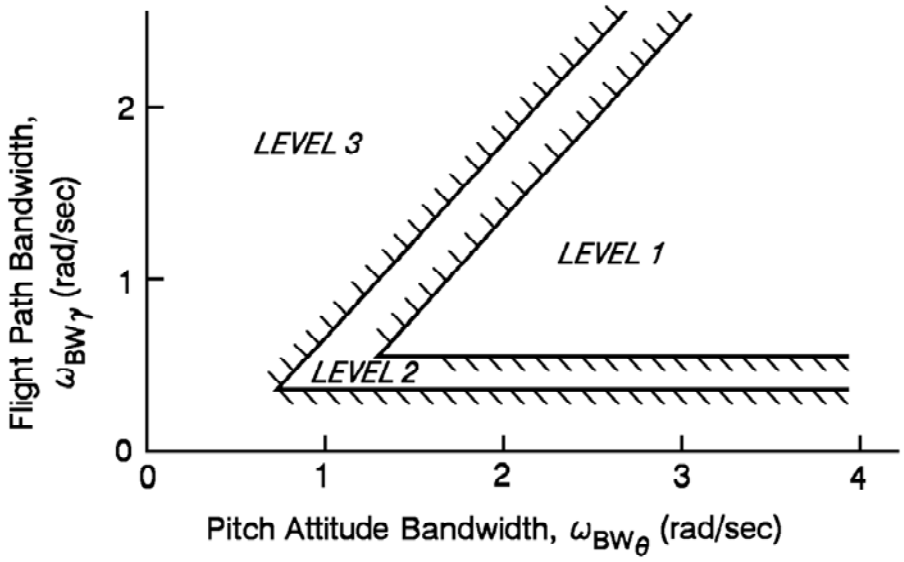
These requirements flow down from 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.3.1.1.

Recommended value for Blank (1): "the boundaries of figure 51"



a) Classes I and IV, All Categories



b) Classes II and III, Categories B and C

FIGURE 51. Relationship between pitch attitude and flight path bandwidths.

Recommended values for table XXXII:

Flight Phase Category	Level	Minimum Phase Difference between Pitch Attitude and Flight Path Angle at $\omega = \omega_{sp}$
A	1	58°
	2	45°
B	1	45°
	2	45°
C	1	52°
	2	45°

REQUIREMENT LESSONS LEARNED

Aircraft operating on the front side of the power-required curve are capable of flight path control via control of pitch attitude. In fact, the primary motivation for the limits set in 5.2.2.1.1.2 is to provide the required inner loop which will allow aggressive precision outer-loop (path) tracking characteristics. A block diagram depicting the pilot-vehicle loop structure for this situation is shown on figure 28. As shown in that block diagram, the short-term flight path response is related kinematically to the aircraft pitch attitude change by the ratio of numerators of the open-loop transfer functions (using $1/T_{\theta_2}$ of the basic aircraft, $\approx -Z_w$):

$$\frac{\gamma}{\theta} = \frac{N_\delta^\gamma}{N_\delta^\theta} \doteq \frac{1}{T_{\theta_2}\, s + 1}$$

No feedbacks to a single control surface can alter this ratio in form or dynamics. The long-term response is related to $d\gamma/dV$, which of course depends on the position of the operating point on the power-required curve (see 5.2.2.1.2.3).

The equivalent-system requirements for pitch attitude control (see 5.2.2.1.1.2) involve equivalent $1/T_{\theta_2}$ indirectly [ω_{sp} versus n/α where $n/\alpha \doteq (V/g)\,(1/T_{\theta_2})$]. Hence these requirements appear to involve pitch and path control in a single criterion. However, because the experimental data used to develop correlations for the criteria do not contain sufficient independent variation of speed and T_{θ_2} (primarily NT-33 data), it is not possible to determine unequivocally whether the boundaries do indeed account for path as well as pitch.

An aircraft designed to operate well on the backside of the power-required curve ($d\gamma/dV$ positive) must rely on thrust or thrust vectoring for flight path control. Such STOL aircraft without autothrottle usually have sufficiently fast engine response characteristics (or some type of blended DLC) to allow precision flight path tracking with the throttles (or other designated flight path controller). Pitch attitude is then used to control airspeed, which varies only slowly. The control of flight path via pitch attitude becomes much less critical and hence some relaxation

in the flight path bandwidth Level 1 limits can be tolerated. Such a control technique obviously requires special pilot training.

It may be argued that with an effective means of directly controlling flight path (i.e., a controller that meets the requirements of 5.2.2.2), there is no need to have any flight path bandwidth. There is no real evidence that this is true, however, and it seems that a more conservative approach is warranted for two reasons: 1) Sometimes, even with direct control of flight path, pitch attitude will be used by the pilot for minor corrections in flight path. In such a case, there shall still be adequate (i.e., Level 2) consonance between attitude and flight path. 2) In the event the designated flight path controller is not Level 1 by 5.2.2.2, the pilot is certain to attempt to control flight path with pitch at some time during the operational lifetime of the air vehicle. Level 2 flight path bandwidth will provide some response, and will hopefully protect against more severe consequences of incorrect control technique.

Experience has shown that the path response bandwidth should be well separated from the pitch response bandwidth. Evidence to support this result is given in the analysis and flight test results obtained by DFVLR (using an HFB 320 in-flight simulator) and reported by Hanke et al. in AGARD-CP-333. These results indicate that an appropriate criterion parameter would be the phase angle between path and attitude at the short-period frequency, i.e., $\phi(\gamma/\theta)\big|_{\omega=\omega_{sp}}$. Noting that the phase angle $\phi(\gamma/\theta)$ evaluated at $\omega = \omega_{sp}$ is given by $\tan^{-1}(\omega_{sp} T_{\theta_2})$, the criterion of 5.2.2.1.1.2 on equivalent CAP can be easily converted to $\phi(\gamma/\theta)\big|_{\omega=\omega_{sp}}$, with the results shown in table XXXII. The advantage of using $\phi(\gamma/\theta)\big|_{\omega=\omega_{sp}}$ is that it does not require a LOES fit to identify ω_{sp} and T_{θ_2} when the bandwidth criterion is utilized. It should be recognized that the values in table XXXII are based on the same NT-33 flight test data as the LOES boundaries in 5.2.2.1.1.2. Until more data can be obtained to indicate pilot rating trends and T_{θ_2} is varied at constant ω_{sp}, it is felt that table XXXII should be kept in the category of guidance.

Notice that $\omega_{sp} T_{\theta_2} = 1$ means that $\omega_{sp} = 1/T_{\theta_2}$, that is, there is no frequency separation between these two roots. A γ/θ phase angle less than 45° at ω_{sp} means that $1/T_{\theta_2}$ is greater than ω_{sp}, that is, a high-frequency γ/θ response. Low $1/T_{\theta_2}$ with respect to ω_{sp} results in a large pitch rate overshoot or, looked at another way, a large lag in the response of γ to θ. A number of writers, including Gibson in AGARD-CP-333, have commented on the need to avoid these excesses, too.

It is important to note that these requirements do not specifically address the magnitude of the flight path response to attitude changes. It is possible to have good flight path bandwidth but inadequate long-term power.

Recent ground-based and in-flight simulations have indicated that too much direct lift control blended with the pitch controller may actually be harmful (see 5.2.2.1.6 Lessons Learned).

Class III aircraft

Requirements on flight path bandwidth for transports have been under development for several years. The initial work was performed in the mid-1980's as part of a development effort for STOL handling qualities criteria (AFWAL-TR-86-3050). The requirements have evolved to their present state where, in combination with criteria on pitch attitude bandwidth and on pitch rate overshoot/pitch attitude dropback, they are extremely effective at predicting handling qualities. Supporting data for these requirements comes from the experiments described in 5.2.2.1.3.1 Lessons Learned.

The flight path bandwidth boundaries are based primarily on two NASA-sponsored studies using in-flight simulation to investigate the flying qualities of transport aircraft in the approach and landing task (NASA CR-172491 and NASA CR-178188). The aircraft used in both experiments was the Air Force/Calspan TIFS. The 1984 experiment explored the effect of different flight control system gains and feedback architecture applied to seven different aerodynamic models. The approach in the 1986 experiment was to define the exact model pole/zero placement directly to achieve a desired command response. The details of both experiments are described in 5.2.2.1.3.1 Lessons Learned. A plot of ω_{BW_γ} versus ω_{BW_θ} for the TIFS data is shown on figure 52.

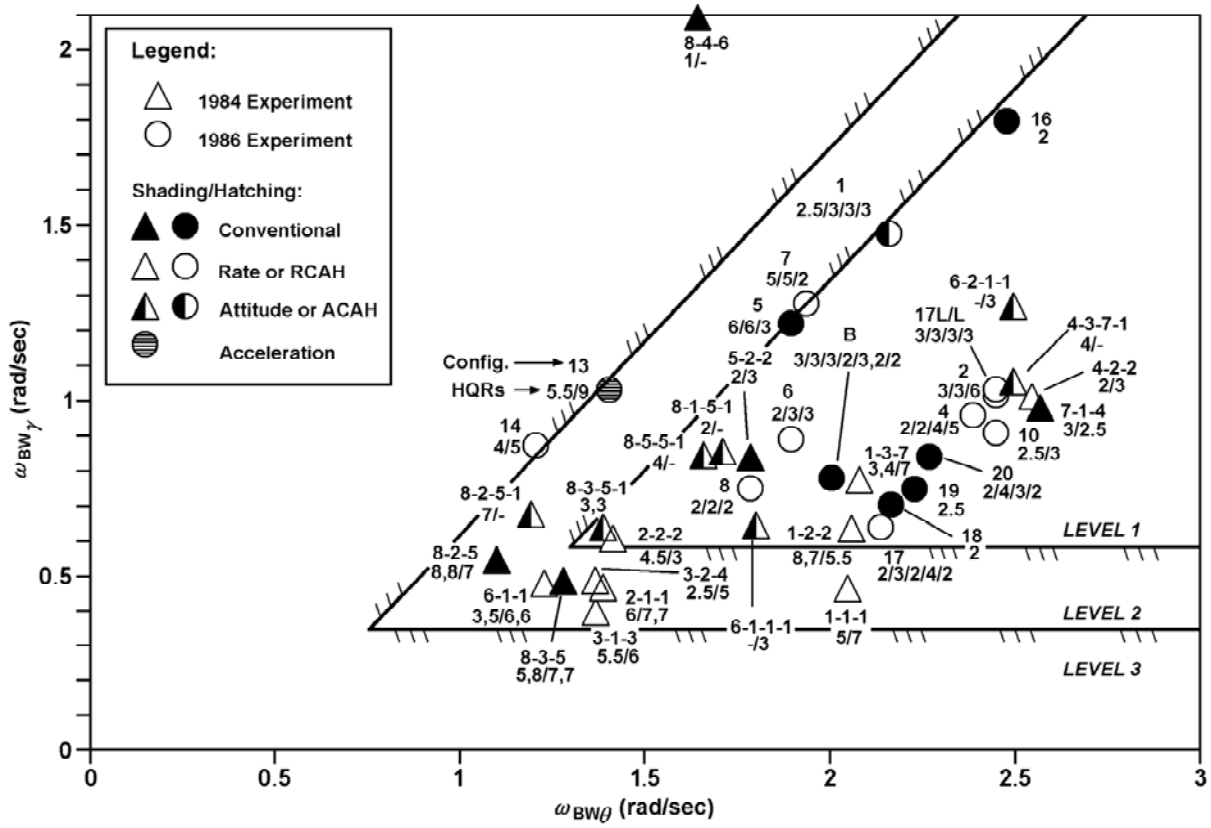


FIGURE 52. Pitch attitude bandwidth versus flight path bandwidth (TIFS experiments).

The identifiers used in the experiment, as well as the HQRs (or converted HQRs) for each configuration, are indicated, as is the general response-type. Pilot HQRs separated by a comma were given by the same pilot; a "/" indicates a change of pilots. For example "3,3,4/5/2" would mean that three different evaluation pilots rated the configuration, and one of them rated it three times.

There were many more points in the database than are shown on figure 52. Only those points relevant to defining the ω_{BW_γ} versus ω_{BW_θ} Level 1 and Level 2 boundaries are included on figure 52. That is, points with excessive dropback, and points that have a τ_{θ_p} of about 0.23 or greater, are considered to have dominant adverse characteristics other than flight path bandwidth that affect the pilot ratings, and do not appear on figure 52.

Although the placement of the attitude bandwidth and flight path bandwidth boundaries was primarily based on the TIFS data, additional large aircraft data (AFFDL-TR-65-210, Boeing D6-10725, NASA CR-182066, and NASA TN D-3971) were considered as supporting information, and used to refine boundary locations where there were no TIFS cases for precise definition. Nevertheless, the degree of correlation of this additional large aircraft data with the recommended bandwidth requirements is comparable to that for the TIFS data.

The AFFDL-TR-65-210 and Boeing D6-10725 in-flight simulation studies were performed during the design phase of the proposed Boeing version of the C-5A heavy military transport, around 1965. The NASA CR-182066 simulation evaluated candidate response-types for a Task-Tailored Flight Control System for the landing of a large, advanced, hypersonic transport. The data from NASA TN D-3971 are the results of a NASA investigation of flying characteristics of fixed- and variable-geometry SST designs, simulated in flight on a Boeing 367-80. The details of these experiments are described in 5.2.2.1.3.1 Lessons Learned.

A plot of ω_{BW_γ} versus ω_{BW_θ} for the additional large aircraft data is shown on figure 53. Again, there were many more points in the database than are shown on figure 53. Only those points relevant to defining the ω_{BW_γ} versus ω_{BW_θ} Level 1 and Level 2 boundaries are included on figure 53. That is, points with excessive dropback, and points that have a τ_{p_θ} of about 0.23 sec or greater, are considered to have dominant adverse characteristics other than flight path bandwidth that affect the pilot ratings, and do not appear on figure 53.

Class IV, Category A boundaries

The valid database for these boundaries is the familiar Neal-Smith experiment (AFFDL-TR-70-74). Plots of flight path bandwidth versus pitch attitude bandwidth for the Neal-Smith data are presented on figure 54. Figure 54 only those cases with relatively low phase delay (less than 0.10 sec) and acceptable dropback are shown. This figure forms the basis for the proposed boundaries on figure 51. The feel system has been included for calculation of parameters, even though this experiment used force commands. There is not strong support for the flight path bandwidth limits of figure 54. These limits are drawn to be consistent with the Category C boundaries discussed below.

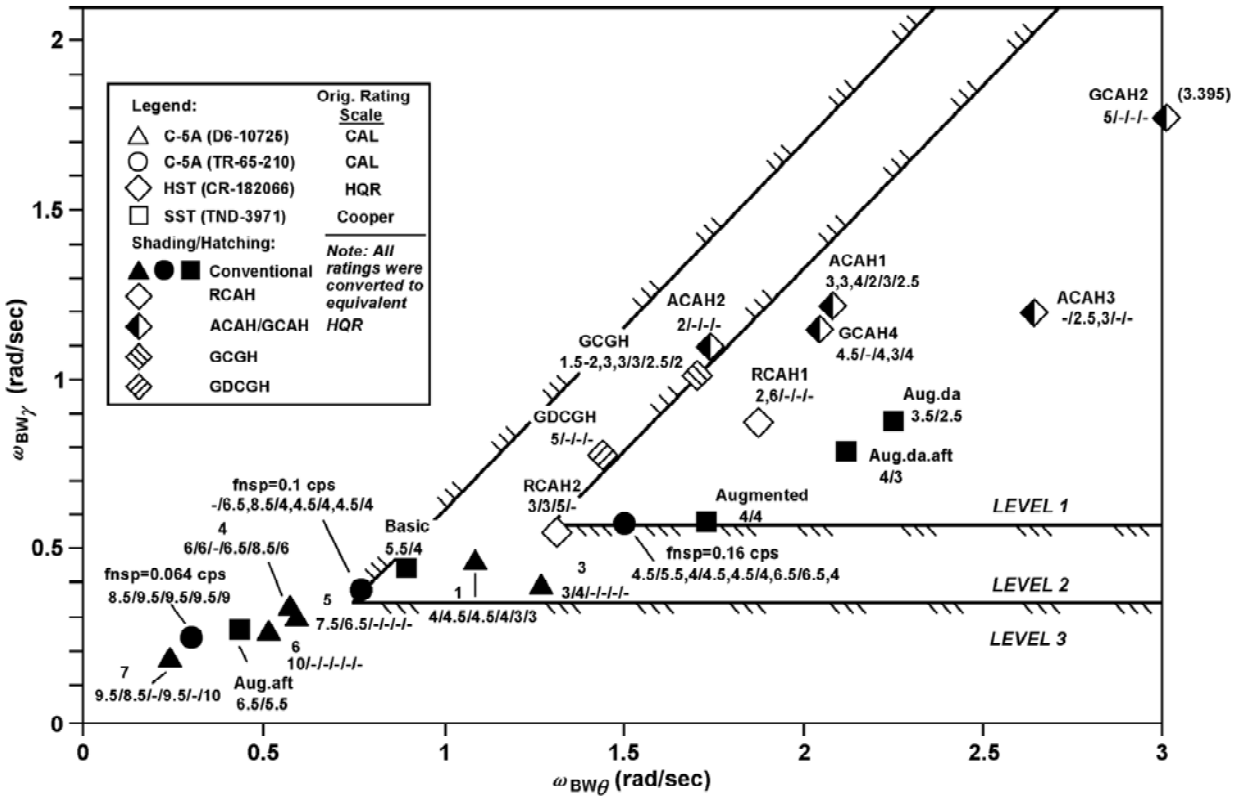


FIGURE 53. Pitch attitude bandwidth versus flight path bandwidth (large aircraft data).

Class IV, Category C boundaries

The LAHOS data (AFFDL-TR-78-122) were used for these boundaries. The data are shown on figure 55. Flight path bandwidth is not well supported by the data, due primarily to a lack of data. The boundaries are drawn, in fact, based on only a single point (the case with an HQR of 6); they were sketched to resemble the limits for transports discussed above. A single case does not seem to be strong support for the lower limit as sketched. More data are clearly needed.

VERIFICATION

Flight path bandwidth is defined as that frequency at which the response of flight path angle, measured at the aircraft center of gravity, lags the cockpit control input, including the feel system, by 135°. Verification will involve frequency response measurements.

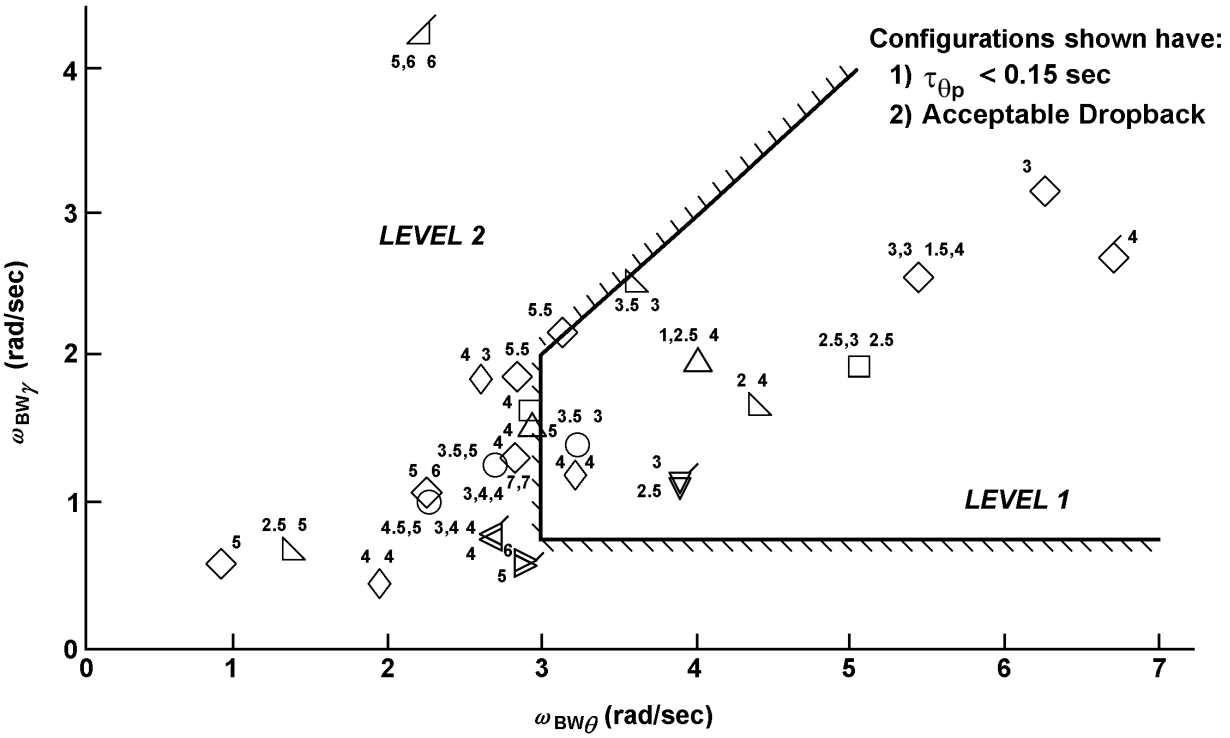


FIGURE 54. Flight path bandwidth versus pitch attitude bandwidth for selected Neal-Smith configurations.

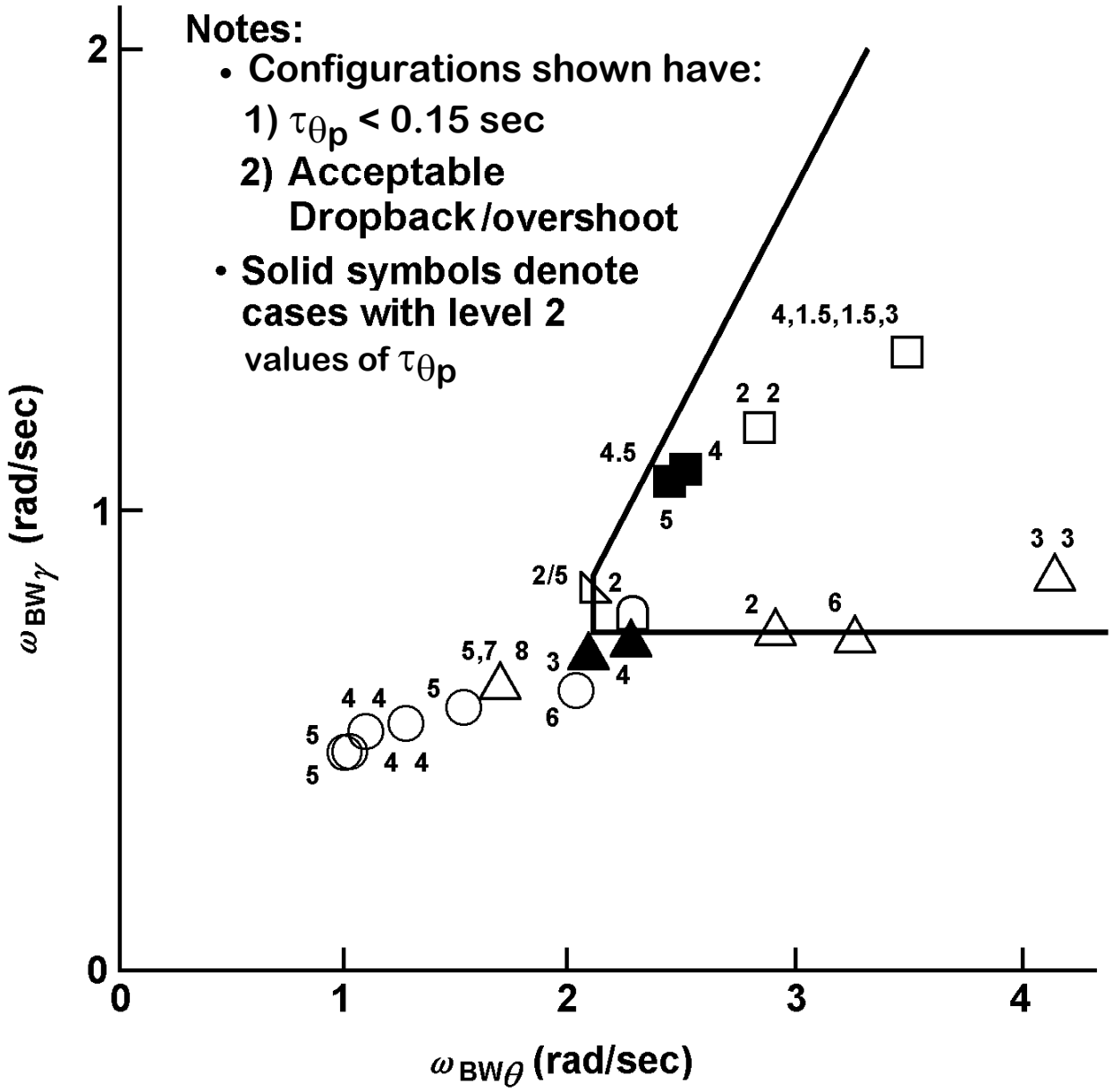
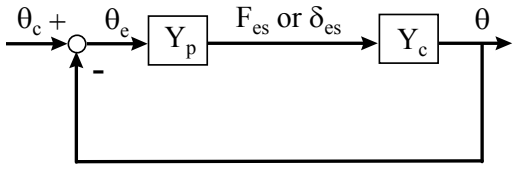


FIGURE 55. **Flight path bandwidth versus pitch attitude bandwidth for selected LAHOS configurations.**

5.2.2.1.4 Closed-loop analysis with a pilot model.
For pilot control of pitch attitude with the pilot model of figure 56, the closed-loop bandwidths of table XXXIII shall be attainable with closed-loop droop of -3 dB, and with closed-loop resonance and pilot model compensation within __(1)__.



where: Y_c is the transfer function of the aircraft and flight control system
θ is pitch attitude
θ_c is an external pitch command
θ_e is pitch attitude error
F_{es} is pilot force on the pitch controller
δ_{es} is pilot deflection of the pitch controller
Y_p is pilot compensation in one of the following forms:

	Pilot needs to provide low-frequency integration	Pilot does not need to provide low-frequency integration
Pilot needs to provide lead	$\frac{K_p(5s+1)(T_{p_1}s+1)e^{-\tau_p s}}{s}$	$K_p(T_{p_1}s+1)e^{-\tau_p s}$
Pilot needs to provide lag	$\frac{K_p(5s+1)(T_{p_1}s+1)e^{-\tau_p s}}{s(T_{p_2}s+1)}$	$\frac{K_p(5s+1)(T_{p_1}s+1)e^{-\tau_p s}}{s(T_{p_2}s+1)}$

FIGURE 56. Pilot model to be used for pilot model analysis.

TABLE XXXIII. Closed-loop bandwidths for pilot control of pitch attitude.

Flight Phase	Closed-loop Bandwidth
Category A	
Category B	
Landing	
Other Category C	

REQUIREMENT RATIONALE

The criteria contained in this paragraph are intended to insure good dynamic performance capability of the pilot-vehicle dynamic system. The form in which criteria are stated was selected to permit accommodation of highly augmented aircraft and systems with transport time delay or cascaded dynamic elements. Through application of describing function techniques, the

criteria may also permit investigation of the effects of certain nonlinearities typically encountered in flight control systems.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.2.1.2.

Recommended value for Blank (1): "the boundaries of figure 57"

Recommended values for table XXXIII:

Flight Phase	Closed-loop Bandwidth
Category A	3.5 rad/sec
Category B	1.5 rad/sec
Landing	2.5 rad/sec
Other Category C	1.5 rad/sec

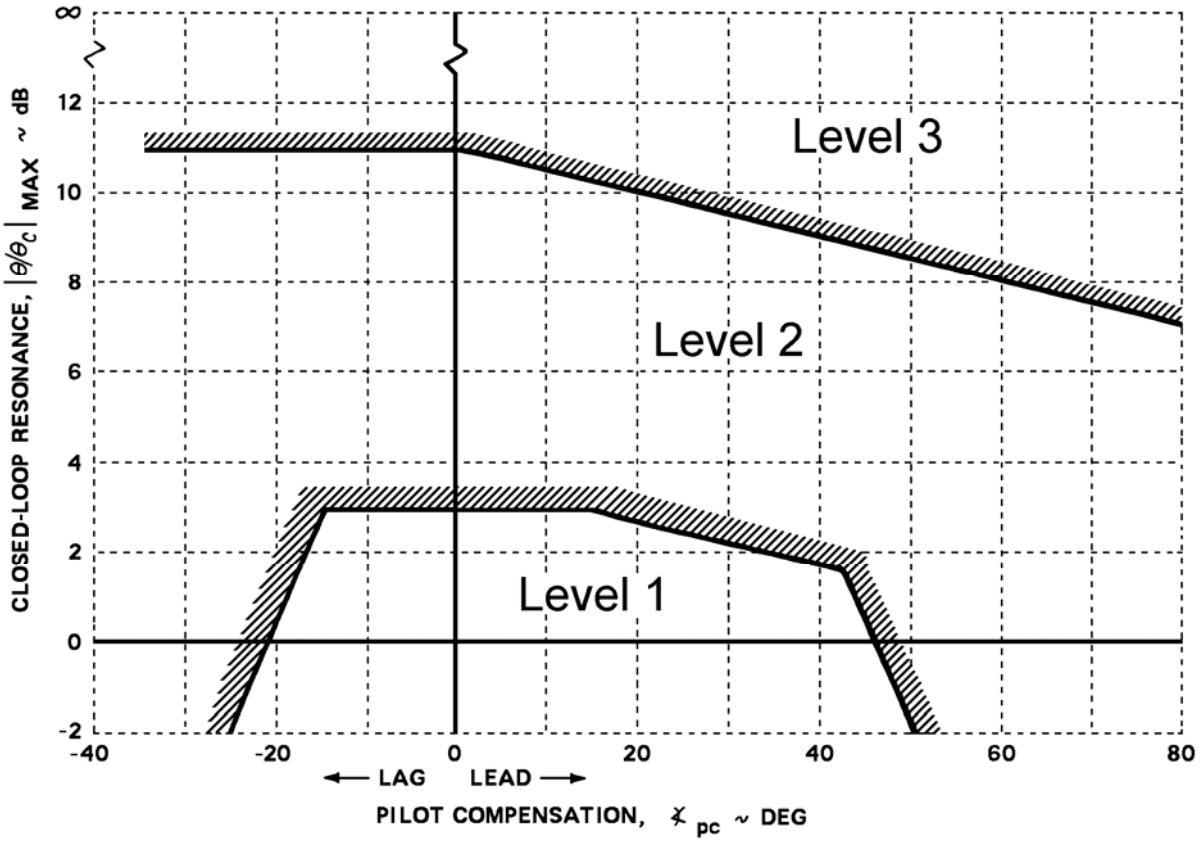


FIGURE 57. Closed-loop resonance and pilot compensation requirements.

REQUIREMENT LESSONS LEARNED

The criteria are derived from the work by Neal and Smith reported in AFFDL-TR-70-74. The basic approach is to model the pilot-vehicle pitch attitude control loop as a unity-feedback system with a pilot model of an assumed form in the forward loop. The form of the assumed pilot model permits accounting for the following characteristics exhibited by pilots when controlling dynamic systems:

Adjustable gain
- Time delay
- Ability to develop lead, or to operate on derivative or rate information
- Ability to develop lag, or to smooth inputs
- Ability to provide low-frequency integration

The two forms of the pilot model account for the observed capabilities and limitations of the pilot with sufficient accuracy to permit approximate analysis of the dynamics of the closed-loop pilot-vehicle system in pitch. It should be emphasized that the pilot model need not be an exact

analog of the human pilot in order to be useful in the context of design criteria. The criteria are based on the hypothesis that if good closed-loop dynamic performance can be achieved with an autopilot with the characteristics described by the assumed pilot model, then the human pilot will also be able to achieve good closed-loop dynamic performance with acceptable workload.

These two forms differ by the (5 s + 1)/s term, the low-frequency integration capability. It is intended that the form of the model without this term will be used when constant speed or two-degree-of-freedom equations are used to represent the aircraft. In this case the aircraft transfer function shall have a free s in the denominator and low-frequency integration by the pilot will not be necessary. When three-degree-of-freedom equations are used or when the flight control system uses attitude stabilization ("Aircraft Dynamics and Automatic Control"), it may be necessary for the pilot model to perform low-frequency integration to avoid droop at frequencies less than the closed-loop bandwidth.

Although the Neal-Smith criteria account explicitly for pilot compensation, they may not always adequately account for sensitivity of closed-loop performance to the pilot-chosen bandwidth (see Radford and Smith in AFWAL-TR-80-3067). Also note that the Neal-Smith criteria do not address path control at all. Sarafian (NASA TM-86728) used another variant of the Neal-Smith technique to correlate approach and landing data from two TIFS variable-stability airplane evaluation programs. He closed the inner pitch attitude loop, for all configurations, with the same amount of pilot phase compensation. Then closing an outer flight path loop, he found that the achievable bandwidth correlated with the pilot ratings.

VERIFICATION

The pilot model is constrained to the given forms but there are no limits on K_p, T_{p_1}, or T_{p_2}. The requirements apply for both force and deflection control inputs.

The $e^{-\tau_p s}$ term in the pilot model accounts for time delay in the pilot's neuromuscular system. Records for the discrete tracking task in AFFDL-TR-70-74 and AFFDL-TR-78-122 exhibited delays ranging from 0.20 to 0.40 sec. Based on cut-and-try correlation with the data from these records, the best value to use in this criteria is either 0.25 or 0.30 sec. For the Landing Flight Phase, a time delay value of 0.25 sec and a bandwidth frequency of 2.50 rad/sec have been determined to give the best empirical correlation with data from AFFDL-TR-72-143 and AFFDL-TR-78-122.

In order to meet the Neal-Smith pilot-in-the-loop criteria, the designer must succeed in finding a combination of K_p, T_{p_1}, and T_{p_2} which will cause the closed-loop resonance and pilot compensation to plot in the applicable region of figure 57. It is necessary, therefore, to perform a parameter search. This search procedure is not difficult; it can be performed graphically using aids described in AFFDL-TR-70-74 or the process can be mechanized on a digital computer. Because the calculations involved in evaluating the closed-loop resonance and pilot compensation as a function of frequency are simple to perform, it is feasible to use a simple trial-and-error approach to test whether or not a proposed aircraft design meets the criteria.

5.2.2.1.5 Pitch PIOs.
There shall be no tendency for sustained or uncontrollable pitch oscillations resulting from efforts of the pilot to control the aircraft. The phase angle of the pitch attitude frequency response to pitch stick force at the criterion frequency, ω_c, shall be greater than or equal to __(1)__. If this phase angle is less than -160°, the phase parameter of normal acceleration at the pilot station, at the same criterion frequency, $\phi_{a_{z_p}}(\omega_c)$, shall be greater than or equal to __(2)__.

Furthermore, the requirements of 5.2.2.1.1.2, 5.2.2.1.2.2, 5.2.2.1.3.1, 5.2.2.1.3.2, 5.2.2.1.3.3, 5.2.2.1.4, 5.2.2.1.6, 5.2.2.1.8.1, 5.2.2.1.8.2, 5.2.2.1.8.4, 5.2.8.2.1, 5.2.8.3.1, 5.2.8.3.2, and 5.2.8.3.5 shall be met.

REQUIREMENT RATIONALE

The purpose of this requirement is to insure that abrupt maneuvers or aggressive tracking behavior will not result in instabilities of the closed-loop pilot-vehicle system. Any such tendency will degrade or even destroy mission effectiveness and likely will be dangerous. Also, see 5.2.1.6 for a general PIO requirement.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.2.3
MIL-STD-1797: 4.2.2.

Recommended value for Blank (1) and Blank (2): -180º

REQUIREMENT LESSONS LEARNED

The qualitative requirement is generalized in view of uncertainties in the state of the art of flight control system design, a tacit recognition of the complexity of the PIO problem. No detailed specification is, at this time, a guarantee against building a PIO-prone airframe/flight-control-system combination.

The quantitative requirement is popularly known as the Smith-Geddes PIO criteria. It was proposed in its original form in AFFDL-TR-77-57. It was more fully-developed as a general longitudinal response requirement in AFFDL-TR-78-154, and further developed and extended to the lateral-directional axis in AFWAL-TR-81-3090. This requirement is an attempt to catch and correct any PIO tendencies as early as possible in the design, when changes are easiest and least costly to make. It also has been found helpful in identifying PIO tendencies in flight and determining fixes.

The criterion frequency, ω_c, is an approximation of the crossover frequency of the pilot-vehicle system during pitch attitude tracking. This approximation is based on data from AFFDL-TR-65-15 shown on figure 58. This figure shows crossover frequency as a function of forcing function bandwidth for different controlled elements. The equation for ω_c was derived from this data as

shown on figure 59, taken from AFFDL-TR-78-154. This equation was altered slightly in AFWAL-TR-81-3090 to the form recommended in Verification. This modification was made in conjunction with the modifications in the calculation of $\overline{S}$ to fit better the LAHOS data and F-15 CAS-off, supersonic PIO experiences.

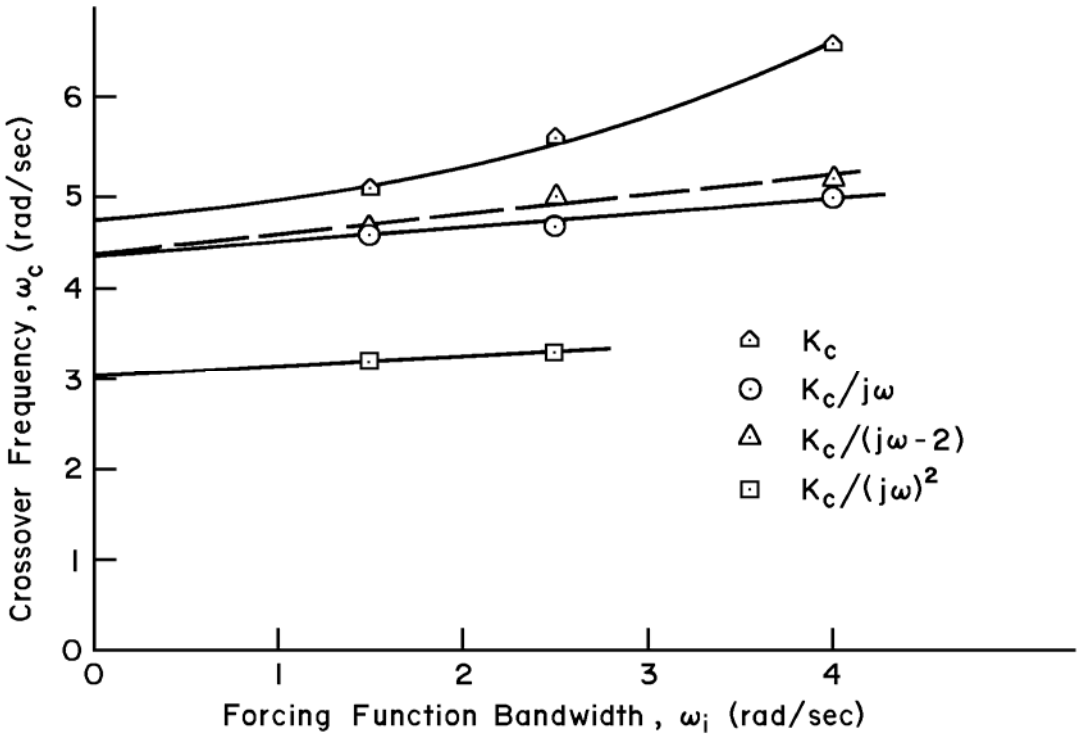


FIGURE 58. Variation of crossover frequency with pitch attitude dynamics (from AFFDL-TR-65-15).

The other parameter in this requirement, $\phi_{a_{z_p}}(\omega_c)$, is a phase parameter associated with the normal acceleration sensed at the pilot's station. It consists of the aircraft phase angle of normal acceleration frequency response plus a phase angle due to an assumed pilot response delay at the pitch attitude criterion frequency, ω_c. This parameter becomes important if there is too much phase lag in the pitch attitude response to stick force; thus the condition that this parameter be considered when the phase angle of the pitch attitude response to stick force is less than -160°. This is a fundamental element of the Smith-Geddes criteria, originally derived in AFFDL-TR-77-57, and was included in its original form in MIL-F-8785C and MIL-STD-1797. In the original form, $\phi_{a_{z_p}}$ was evaluated at ω_R, where ω_R was defined at any frequency between the range of 1 to 10 rad/sec at which lightly damped (resonant) oscillations in pitch attitude could result from turbulence inputs or from pilot control of the aircraft when used in the intended manner. In AFFDL-TR-78-154, ω_R was replaced by ω_c as defined in Verification. The concept behind this part of the requirement is that, if the pitch attitude (θ) loop is resonant at ω_c, then the pilot may attempt to control normal acceleration, a_{z_p}, instead of θ. The aircraft will be

PIO-prone if there is too much phase lag in this response. The criteria for this requirement are based largely on correlation with the Neal-Smith database (AFFDL-TR-70-74).

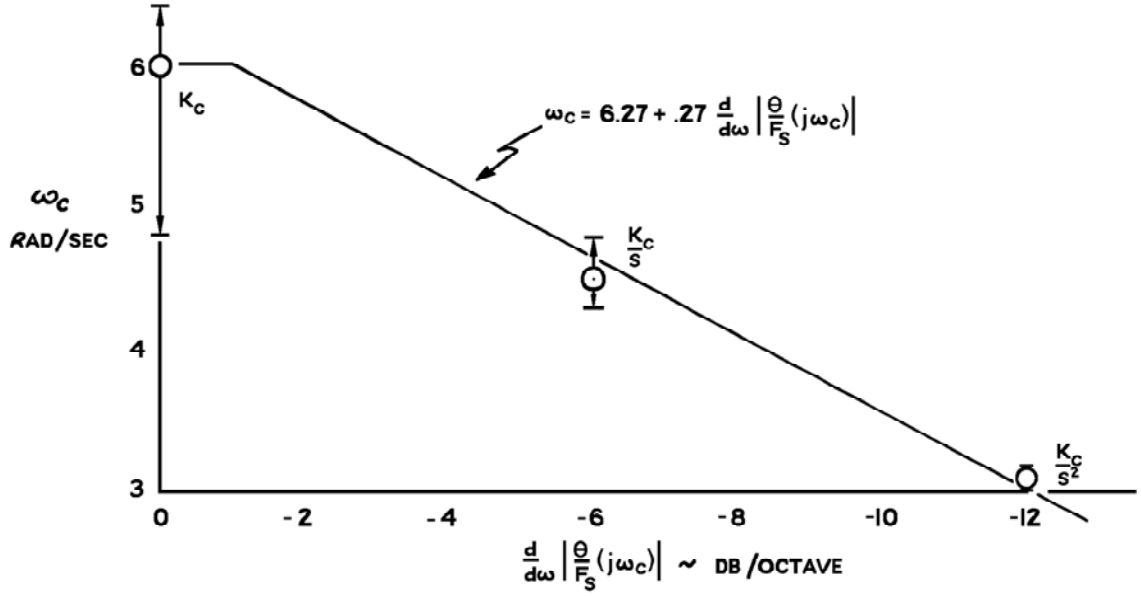


FIGURE 59. Specification of the criterion frequency.

The Smith-Geddes criteria has been used by AFFTC with considerable success for several years. The criteria has been used to analyze PIOs in the Space Shuttle, the F-15 with CAS-off, the AFTI/F-16, the AFTI/F-111, the F-15 S/MTD, the YF-22, and the C-17. Application of the criteria to the Space Shuttle was documented in "Prediction and Occurrence of Pilot-Induced Oscillations in a Flight Test Aircraft" by Twisdale and Kirsten. In an analysis of three PIOs in the Space Shuttle, the Smith-Geddes criteria correctly predicted the PIO tendency and closely predicted the frequency of the PIO. For a PIO in landing flare the criteria predicted a frequency of 3.5 rad/sec and the frequency of the observed PIO was 3.6 rad/sec. In another PIO at an altitude of 18,000 ft and a speed of 610 ft/sec, the criteria predicted a frequency of 3.3 rad/sec and the observed frequency was 3.1 rad/sec. In the final example, at a similar flight condition, a PIO occurred while tracking a cockpit display. When the display dynamics were added to the analysis, the criteria accurate predicted the observed PIO frequency of 2.0 rad/sec.

A valuable lesson learned in the determination of the criterion frequency, ω_c, is found in a Northrop white paper, "Evaluation of B-2 Susceptibility to Pilot-Induced Oscillations" by Margo L. Givens and Frank L. George, presented at the Flying Qualities Workshop at the 1994 AIAA Atmospheric Flight Mechanics Conference.

> "For the most part, the approach taken was as recommended by Ralph Smith in [AFFDL-TR-78-154] which presented a straight forward process of evaluation. Exceptions were made for criterion frequency selection. The recommended method [in AFFDL-TR-78-154] for criterion frequency selection is based on calculating an average slope of the pitch attitude-to-controller Bode magnitude plot in the range of 2.0 to 6.0 rad/sec and then applying this value to the [ω_c]

formula. This frequency range stipulation was often inappropriate for the B-2 which has higher break frequencies than those systems described in the [Smith-Geddes] documentation. Because the validity of a [Smith-Geddes] analysis is dependent on the correct selection of the criterion frequency, three other methods of the criterion frequency selection were evaluated.

The first method, which had been used by Ralph Smith in the past, involved selecting parameters for a pilot model, closing the pitch attitude loop, and obtaining an n_z to α_{gust} response power spectral density (PSD) in search of resonant frequencies which would be defined as the criterion frequencies. The implementation of this method was unproductive because no resonant frequencies were found in the B-2 n_z PSDs.

The second method adapted the recommended ω_c derivation formula to use the slope calculated after the short-period break rather than the average slope in the 2.0 to 6.0 rad/sec frequency range. This method worked quite well for most of the cases. Because of the occasional case which produced questionable results, a third method of frequency determination was devised for use in a validity check on criterion frequencies derived using method 2.

The third approach used typical B-2 pilot pitch stick input frequencies as the criterion frequencies. These frequencies were determined by calculating PSDs from stick, n_z, and θ time histories of landings and refuellings extracted from flight test data. It was found that the ω_c values calculated with the second method were consistent with these flight data pilot stick input frequency ranges."

The statement that requirements 5.2.2.1.1.2, 5.2.2.1.2.2, 5.2.2.1.3.1, 5.2.2.1.3.2, 5.2.2.1.3.3, 5.2.2.1.4, 5.2.2.1.6, 5.2.2.1.8.1, 5.2.2.1.8.2, 5.2.2.1.8.4, 5.2.8.2.1, 5.2.8.3.1, 5.2.8.3.2, and 5.2.8.3.5 shall also be met would seem to be redundant, since these are already requirements. However, recent history would seem to indicate that because the term PIO does not appear in the titles of these requirements, the importance of these requirements in precluding PIO is not appreciated. Pilot-In-the-loop Oscillations were a consideration in setting the boundaries of these requirements. Many recent PIO incidents can be traced directly to problems addressed by these requirements. Therefore, these requirements are repeated here in the PIO requirement to insure that their significance in precluding PIO tendencies is understood. Pilot-vehicle analysis in the manner described in these paragraphs shall help in the design stage.

Gibson's views of PIOs are taken directly from ICAS-86-5.3.4:

"High order characteristics are associated with pilot-vehicle closed-loop handling problems or PIO. As this term has been used to describe low-order problems the differences should be clearly understood. The abrupt pitch bobble type is discontinuous, consisting of repeated tracking correction. The sluggish pitch overdriving type is also discontinuous with input pulses to stop the unpredictable excess in response. Although the aircraft is not under complete control, it is not out of control.

High-order PIO is a continuous out of control attitude instability, the amplitude ranging from small to large and potentially destructive. Because the problem is due to inadequate pilot-vehicle closed-loop gain and phase margins, examination of the pitch attitude frequency response identifies the cause of the solution.

Figure 60 shows the features which separate low- and high-order pitch handling. The area of interest can be confined to the region of phase lags between 180 and 200º, which determines the PIO frequency. This arises from the success of the synchronous pilot (NOR-64-143) in PIO analysis, assuming that any pre-PIO equalization is abandoned for a pure gain behavior in the undamped or divergent oscillation. The correct frequency is adopted instantaneously with the stick in phase with the pitch attitude error and 180º out of phase with the attitude. The stick is not always moved so purely in practice, but very often the pilot can be seen to apply the stick a little too quickly and then hold it while waiting for the pitch rate reversal before also reversing the stick.

The tendency of a configuration to PIO can therefore be assessed without using a pilot model by empirically establishing the range of characteristics found in actual PIO examples. Enough have now been published to do this with considerable accuracy. An important feature at the PIO frequency is the response gain. If this is small enough, dangerous oscillation amplitudes cannot occur, and PIO has not been found where this is less than 0.1º/lb of stick force. This is not a completely necessary condition, but it is a highly desirable design aim.

Pilot-In-the-loop Oscillations have occurred most frequently, though not exclusively, in the landing flare. The connection with the common place stick pumping is well established. This subconscious excitation of pitch acceleration in the landing flare occurs near the same frequency as a PIO. If the attitude in the oscillation suddenly intrudes into the pilot's awareness, a ready-made PIO is already in existence. The lower the frequency, the larger is the attitude oscillation at the usual acceleration amplitude of about 6º/sec^2, and the more likely the conversion becomes. This indicates strongly the desirability of a high crossover frequency through the PIO region.

While an oscillation amplitude of less than 0.5º in the flare will not usually be noticed, the one significantly more than 1º is very likely to. This, or the corresponding pumping/PIO frequency is not an ideal parameter for correlation. The most successful has proved to be the rate at which the pitch attitude phase lag increases with frequency in the PIO lag crossover region, equally applicable to the landing or to target tracking tasks. By the nature of the attitude frequency response, if the crossover frequency is low and the attitude attenuates only slowly towards the crossover region, the phase rate is large. If the frequency is high and there is substantial attenuation, the phase rate is low. The gain margin is increased, the stick pumping amplitude is reduced, and the tendency is decreased automatically by designing a low phase rate into the control laws.

This simple attitude parameter alone is almost sufficient to quantify the tendency to high-order PIO, and it correlates well with available examples of high-order PIO. Figure 61 shows the trends with an accuracy good enough to allow Level 1, 2, and 3 boundaries to be drawn if desired. For the control law designer it is enough to aim for a phase rate of less than 100º/cps and attitude response smaller than 0.1º/lb at the crossover. These characteristics are a natural feature of low-order aircraft whose attitude phase lag exceeds 180º due to the power control and so could in principle suffer from PIO, yet do not. Early examples of bobweight PIO were high-order in kind and are found to have had very large phase rates with the stick free.

For most combat aircraft configurations, consideration of normal acceleration effects does not improve the PIO analysis. The g at the cockpit is usually attenuated and phase advanced relative to the c.g. and will often not reach the 180º-lag necessary for piloted instability. Human sensing of the g response is poor and at the initiation of the PIO the g may be undetectable. In large aircraft with the cockpit far ahead of the c.g., the heave can have a significant effect and has to be taken into account in the dominant requirement to optimize the pitch attitude behaviour.

Although the attitude to stick force response gain is significant in PIO, there is little evidence that a damper modifies the pilot's stick phasing in a PIO and only the stiffness component shall be used. Where PIO tendencies exist, they will be exacerbated by a high stick stiffness. Gradients of 5 to 8 lbs/in. with forces of 2 to 2.5 lbs/g have proved to be extremely satisfactory [fly-by-wire] aircraft. Designed to the phase rate and gain margin discussed above, the attitude gain phase rate and gain at the PIO frequency is only some 0.5º/in. In AFFDL-TR-74-9, case 4D had high phase rate and low PIO gain margin. With a gradient of 22 lbs/in. and 6.7 lbs/g it had an attitude gain of 7º/in. at the PIO frequency. Not surprisingly it suffered from continuous pitch oscillations and severe tracking PIO, earning ratings of 9 and 10.

The boundaries in the frequency response criteria of Figure 62 are based directly on these considerations and will eliminate high-order PIO. Low-order PIO will also be eliminated by the optimization criteria given above."

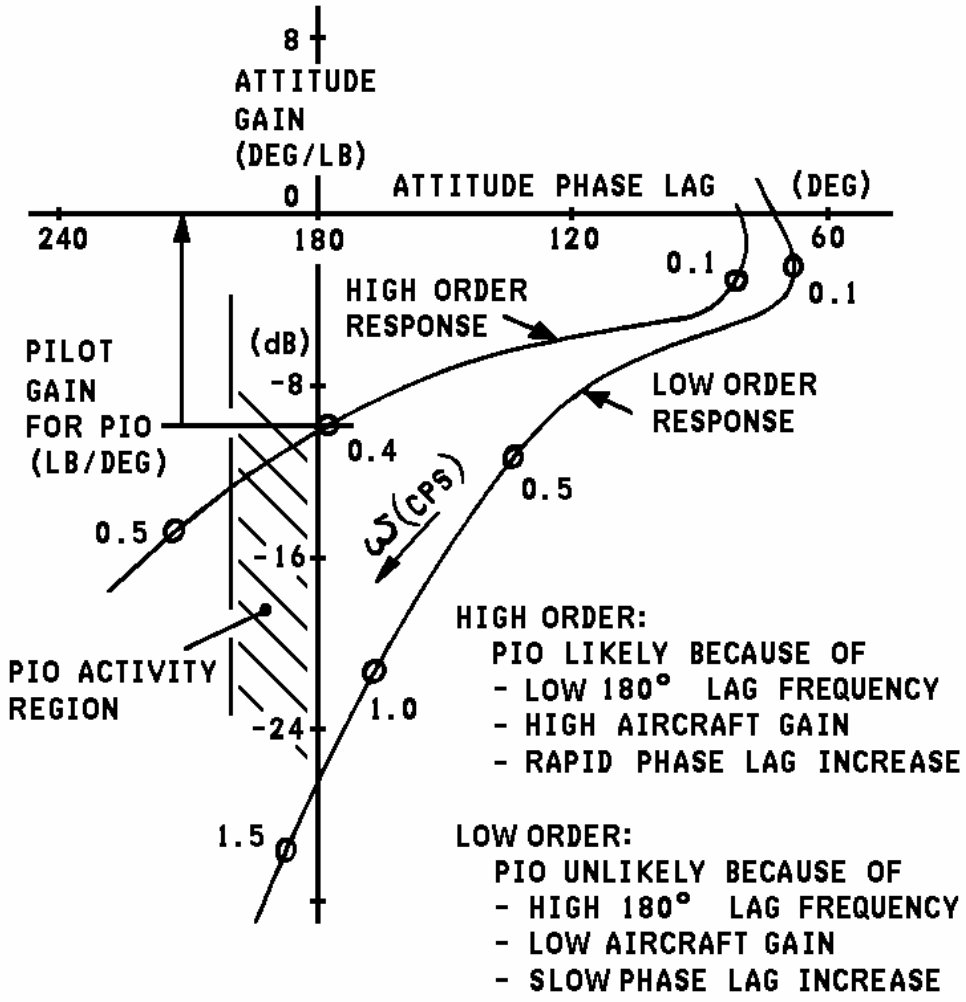


FIGURE 60. High- and low-order frequency response.

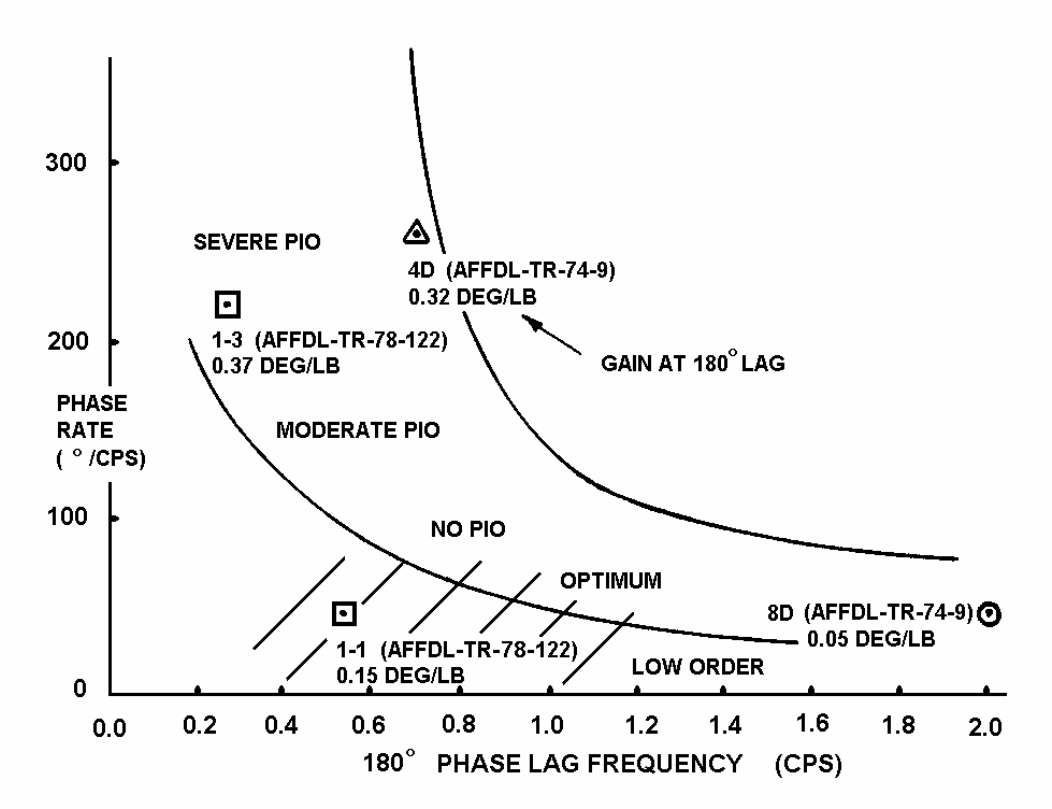


FIGURE 61. Trends of high-order phase rate.

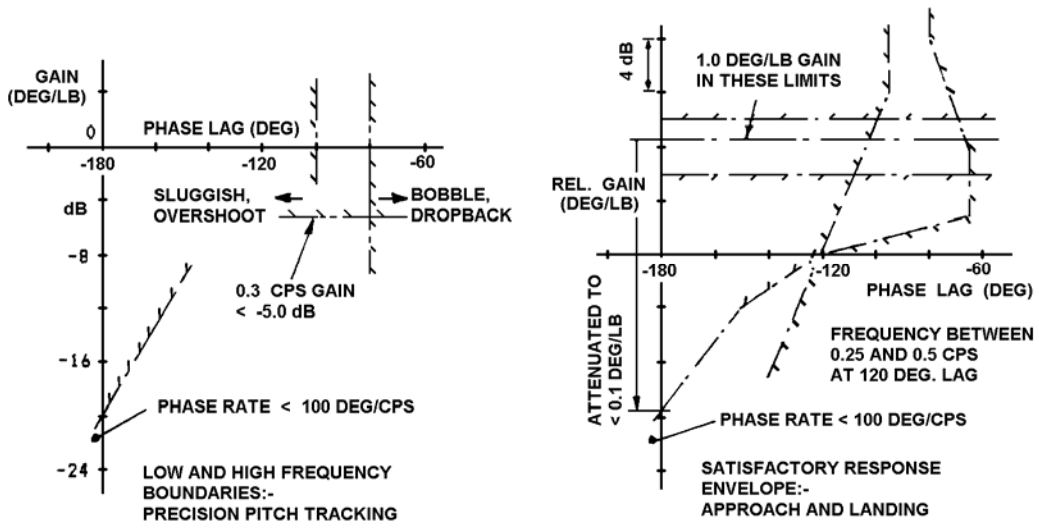


FIGURE 62. Design aim criteria for pitch attitude frequency response.

VERIFICATION

The value of ω_c is determined as follows:

$$\omega_c = (0.24 \text{ rad-oct/dB-sec})\ \overline{S} + 6.0 \text{ rad/sec}$$

where $\overline{S}$ is the average slope of $\left|\frac{\theta(s)}{F_s(s)}\right|$ in dB/oct over the interval from 1 to 6 rad/sec.

The phase parameter of normal acceleration at the pilot's stations, $\phi_{a_{z_p}}(\omega_c)$, is defined by:

$$\phi_{a_{z_p}}(\omega_c) = \angle \frac{a_{z_p}(j\omega_c)}{F_s(j\omega_c)} - (14.3 \text{ deg-sec/rad})\ \omega_c$$

where a_{z_p} is normal acceleration at the pilot's station.

No specific method for determining $\overline{S}$ was required in AFFDL-TR-78-154. Smith originally recommended a range of 2 to 6 rad/sec and computed $\overline{S}$ as the average of three linear approximations of slopes between pairs of points one octave apart, i.e.:

$$\overline{S} = \frac{1}{3}\left[\left|\frac{\theta}{F_s}(4j)\right| - \left|\frac{\theta}{F_s}(2j)\right| + \left|\frac{\theta}{F_s}(5j)\right| - \left|\frac{\theta}{F_s}(2.5j)\right| + \left|\frac{\theta}{F_s}(6j)\right| - \left|\frac{\theta}{F_s}(3j)\right|\right]$$

In AFWAL-TR-81-3090 the frequency range was extended to 1 to 6 rad/sec, and a similar formula was used to compute $\overline{S}$ using five slopes instead of three, i.e.:

$$\overline{S} = \frac{1}{5}\left[\left|\frac{\theta}{F_s}(2j)\right| - \left|\frac{\theta}{F_s}(1j)\right| + \left|\frac{\theta}{F_s}(3j)\right| - \left|\frac{\theta}{F_s}(1.5j)\right| + \left|\frac{\theta}{F_s}(4j)\right| - \left|\frac{\theta}{F_s}(2j)\right| + \left|\frac{\theta}{F_s}(5j)\right| - \left|\frac{\theta}{F_s}(2.5j)\right| + \left|\frac{\theta}{F_s}(6j)\right| - \left|\frac{\theta}{F_s}(3j)\right|\right]$$

This modification brought predictions of ω_c more in line with observations based on Landing Approach Higher-Order System (LAHOS) data. AFFDL-TR-78-154 advises using a consistent method to calculate $\overline{S}$, even when the slope of $\left|\frac{\theta(s)}{F_s(s)}\right|$ varies considerably in the 1 to 6 rad/sec range.

It would be an easy matter for the contractor to ascertain compliance with this paragraph without relying on pilot-vehicle analysis methods. For example, ω_c and the specified

phase lag can easily be obtained from simulator or in-flight time histories (ground-base simulations will not show up acceleration-dependent PIO tendencies). Nonetheless, analytical estimates can and should be made by the contractor as part of the design evolution. The user shall refer to AFFDL-TR-77-57, AFFDL-TR-78-154, and AFWAL-TR-81-3090 when applying the quantitative requirement.

For flight evaluation the PIO tendency classification scale of figure 19 will be helpful. Pilot-In-the-loop Oscillations are associated with abrupt maneuvers and precise tracking as in air-to-air gunnery, formation flying, flare, and touchdown. Pilot-In-the-loop Oscillations observed in flight are often not obtained in ground-based simulators, even those with some motion. Tight, aggressive pilot control action will tend to bring on any PIO tendencies. High sensitivity to control inputs is often a factor. Some pilots are better at exposing PIO tendencies than others, depending upon piloting technique.

5.2.2.1.6 Normal acceleration at the pilot station.
The dynamic normal acceleration response at the pilot station to the pitch controller shall not be objectionably large or of a confusing nature.

REQUIREMENT RATIONALE

Unusual pilot locations can adversely affect handling qualities in terms of the pilot's perception of pitch rate response to a pitch controller input.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.2.4.

REQUIREMENT LESSONS LEARNED

This is a new requirement whose need is apparent though insufficient information exists to formulate recommended criteria. A prominent example is the Space Shuttle, in which confusing acceleration cues played a part in pilot-in-the-loop oscillations encountered during approach and landing tests (NASA CR-163108). In that vehicle the pilot's station is noticeably aft of the instantaneous center of rotation for elevator inputs, so when he pulls back on the controller he first sinks noticeably before rising. In most cases, the pilot station is forward of the instantaneous center of rotation for pilot pitch control inputs, giving an initial impulse $\ell_x \dot{q}_0$ in the correct direction and a transfer function (short-term approximation)

$$\frac{a_{z_p}}{F_s} = \frac{K\,(s^2 + 2\,\zeta_a\,\omega_a\,s + \omega_a^2)}{s^2 + 2\,\zeta_{sp}\,\omega_{sp}\,s + \omega_{sp}^2}$$

Generally $\omega_a > \omega_{sp}$, but it is conceivable that this may not hold universally. Figure 63 shows no ill effects from $\omega_a = 1.25$, $\omega_{sp} \leq 1$ for simulated approach and landing of a very large transport. The initial a_z seems to help overcome any extent of adverse time delay effects, but a limit might be reached where the initial response is too abrupt.

Blended direct-lift control has not always produced the expected gains. The F-16 CCV and AFTI maneuver enhancement modes were well-liked for both their quickened path response and their gust alleviation. Hanke, Wilhelm, and Lange (AGARD CP-333) found deteriorating ratings with increasing DLC gain when they evaluated washed-out spoiler blended with elevator control. Evidently, if the pilot must close an inner attitude loop, he needs some phase separation between pitch and path responses in order to distinguish the two during the approach phase of landing. For the classical case, this separation was given by $\tan^{-1}(\omega_{sp} T_{\theta_2})$. Another in-flight evaluation of approaches and actual landings with a rate-command/attitude-hold system failed to realize the improved ratings that DLC had produced in ground-based simulations. It has been argued whether the disturbing heave motion associated with stick motions was attributable to the basic DLC or was an artifact of the simulation (Mooij and Van Gool, AGARD-CP-333).

With the USAF/Calspan TIFS airplane, attempts to control the value of $1/T_{\theta_2}$ via direct-lift control have excited a structural mode which was very evident in the evaluation cockpit at high DLC gain. In one recent case the result was a long time delay followed by a very abrupt response at the cockpit, which the pilot rated unacceptable or uncontrollable. Since all aircraft have some flexibility, the lesson is to be alert for aeroelastic effects on flying qualities as well as on the structure and the flight control system.

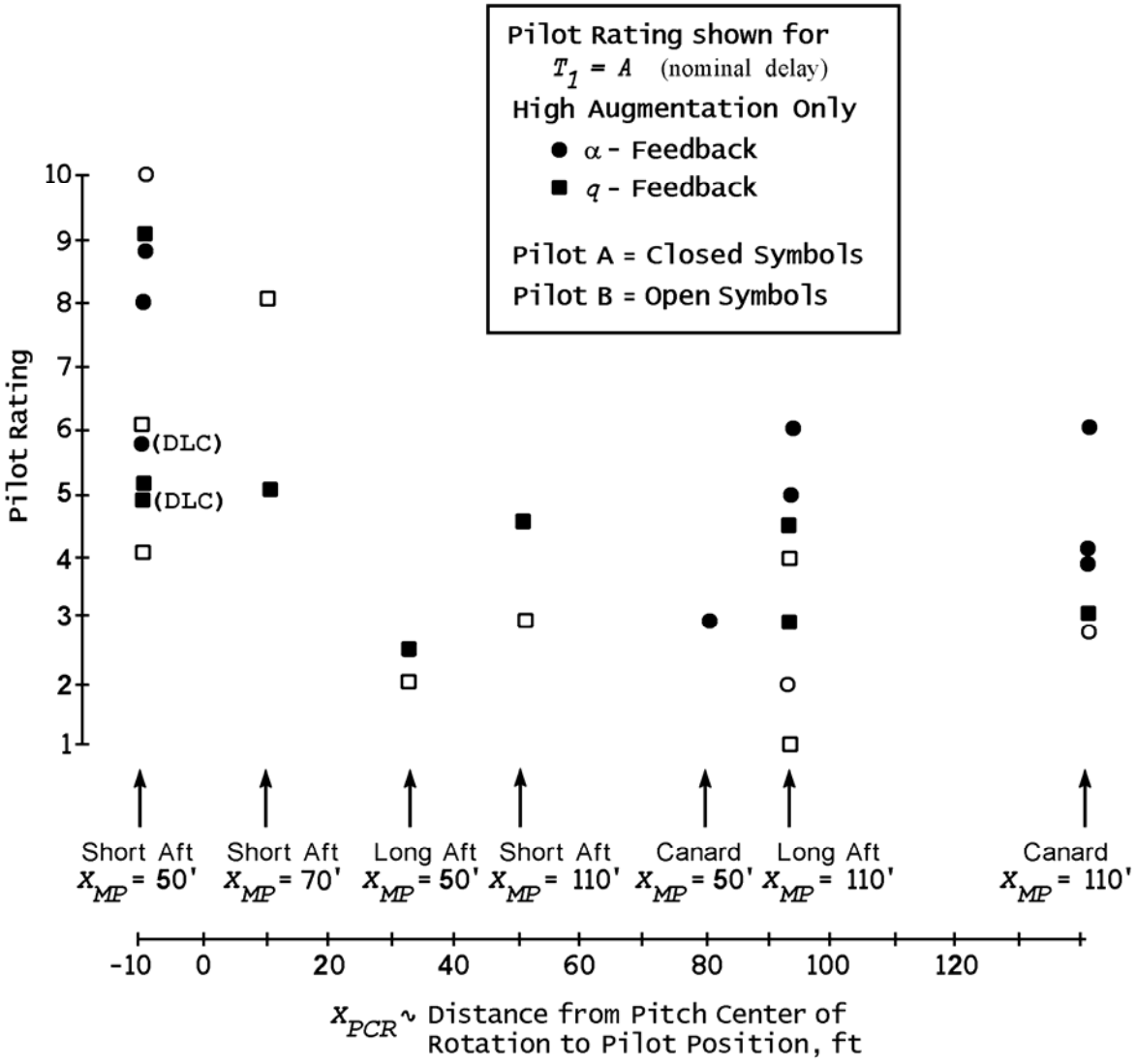


FIGURE 63. Pilot rating versus pilot position—center of rotation (from AFWAL-TR-81-3118).

VERIFICATION

Any difficulty will be apparent in tasks requiring very tight control of flight path. The effective control arm is given by $\ell_c = \bar{c}\, C_{m_\delta} / C_{L_\delta}$, positive forward from the c.g. (negative for an aft tail). The instantaneous center of rotation for control inputs is at $\ell_{CR} = Z_\delta / (M_\delta + Z_\delta\, M_{\dot{w}}) \doteq Z_\delta / M_\delta$, positive forward of the c.g. ($Z_\delta = -\bar{q}\, S\, C_{L_\delta} / m$, $M_\delta = \bar{q}\, S\, \bar{c}\, C_{m_\delta} / I_{yy}$).

For the two-degree-of-freedom approximation (assuming constant speed) the normal-acceleration transfer-function numerator for control inputs is

$$N_{\delta}^{a_z} = -g\frac{C_{L_\delta}}{C_{L_1}}\left(1-\frac{\ell_x}{\ell_{CR}}\right)\left[s^2-\frac{g/V}{2k_y^2 C_{L_1}}\left(\frac{C_{m_q}+C_{m_{\dot\alpha}}-\frac{2\ell_x\ell_c}{\bar c^2}\left(1-\frac{\bar c\, C_{m_\alpha}}{\ell_c C_{N_\alpha}}\right)C_{N_\alpha}}{1-\ell_x/\ell_{CR}}\right)s\right.$$
$$\left.-\frac{g\ell_c}{k_y^2\bar c^2}\frac{C_{N_\alpha}}{C_{L_1}}\left(\frac{1-\frac{\bar c\, C_{m_\alpha}}{\ell_c C_{N_\alpha}}}{1-\ell_x/\ell_{CR}}\right)\right]$$

at a distance ℓ_x forward of the c.g., where $k_y^2 = I_{yy}/(m\,\bar c^2)$. Fuselage bending might also affect the response.

5.2.2.1.7 Longitudinal control power.
Longitudinal control power shall meet the requirements of 5.2.2.1.7.1 through 5.2.2.1.7.6.

5.2.2.1.7.1 Longitudinal control margin.
Aerodynamic control power, control surface rate, and hinge moment capability shall be sufficient for safe recovery throughout the range of attainable angles of attack to prevent the occurrence of deep stall and enhance maneuverability for tactical utility. For all Normal States, the air vehicle shall exhibit no deep stall trim point within the center of gravity limits, and shall have no objectionable nose-up or nose-down recovery characteristics, such as positive or negative AOA hang-up.

REQUIREMENT RATIONALE

The amount of pitching moment available is critical to tactical maneuverability and safe recovery from high angle of attack flight conditions. The vortex lift augmentation and relaxed static stability design features have led to improved performance, maneuverability, and agility but have also resulted in inadequate nose-down pitch control power for high AOA recovery at some conditions. The Navy and USAF have experienced numerous incidents and accidents in air vehicles that have developed high angle of attack hang-up or deep stall and were unable to recover because the nose-down pitching moment available was insufficient to overcome the established conditions.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.10
MIL-STD-1797: 4.1.11.5.

REQUIREMENT LESSONS LEARNED

An overly stringent requirement for longitudinal control margin may result in excessive weight and supersonic performance penalties, whereas one which is too lax could lead to low-speed high angle of attack controllability problems and degraded maneuvering capability. The level of static pitching moment coefficient is important throughout the angle of attack range but especially at the pinch point where pitching moment is at a minimum. The effects of factors such as pilot strength, regions of control-surface-fixed instability, inertial and kinematic coupling, symmetric and asymmetric stores, atmospheric disturbances, failure transients due to center of gravity control malfunction, or failures in the propulsion, flight control and other integrated systems, shall also be factored into the guideline.

Extensive simulation and flight testing by NASA and the Navy resulted in the development of figures of merit to guide the design for nose-down control capability. This joint effort determined the desired values for nose down performance with gear and flaps retracted, in 1g flight, idle power, and dynamic pressure for stall.

Parameter	Tactical Utility	Safety Minimum
Pitch acceleration within the first second	Greater of: .07 rad/sec/sec + pitch acceleration corrected for inertial coupling or .25 rad/sec/sec	.07 rad/sec/sec + pitch acceleration corrected for inertial coupling
Pitch rate at 2 sec	24º/sec	5º/sec

Results of the Navy and NASA Langley Research Center pitch control margin simulation studies were validated in flight. The figures of merit that relate pilot cueing and qualitative assessment of high angle of attack nose-down recovery to quantitative measures were confirmed to be short term response in the form of pitch acceleration and pitch rate. During the NASA simulation studies, it was determined by the evaluation pilots that pitch acceleration was the most strongly perceived nose-down response cue. In the absence of significant angular rates, pitch acceleration is strongly related to the pitch control power due to the direct proportionality to static pitching moment. Pitch acceleration was considered to be the most important figure of merit in quantifying longitudinal control margin requirements since acceleration was readily perceived by the pilots during a pushover recovery from high angle of attack within the first second of the recovery. Pilot comments also indicated that in addition to initial pitch acceleration, pitch rate approximately 2 sec after the recovery input was useful in the pitch recovery rating process.

For the F/A-18E/F at flight idle power, the pitch acceleration was required to be 0.14 rad/sec/sec in cruise configuration. Minimum nose down pitch rate achieved within the first 2 sec from initial pilot-commanded forward longitudinal stick force was required to be at least 8.5º/sec at the maximum power setting and 12.0º/sec at an idle power setting. The F/A-18E/F gear and flaps extended requirement was a minimum of -0.28 rad/sec/sec pitch acceleration from 20º AOA within the first second of pilot-commanded longitudinal stick force command.

VERIFICATION

Pitch control margin shall be demonstrated during flight test. Nose down acceleration and pitch rate capability shall be measured at the minimum pitching moment point with full forward longitudinal control stick command, with normal trim, at aft c.g. locations, flight conditions for low fuel state, and controls deflected to maintain stability. Maneuvers that also have some utility for assessing pitch control margin include pushovers with initial nose up pitch rate and recoveries from zoom climb and rolling conditions.

During the joint NASA and Navy test program, fundamental types of maneuvers were performed and evaluated during piloted simulation and flight test. The maneuvers were open-loop such that specific recovery conditions were not targeted for capture. The pushover from 1g stabilized, trimmed, wings-level, high angle of attack allows for direct assessment of the nose-down control moment available over an angle of attack range, while minimizing the thrust and performance effects.

Use of advanced propulsive and aerodynamic control effectors will aid in achieving the control moments to meet maneuvering requirements. Control power for verification may be derived from aerodynamic control surfaces, thrust vectoring and reaction-type control devices. Additionally, control power/control margin shall be verified, in environmentally-calm conditions, adequate to maintain or recover control following any failure and to ensure that the flight phase can be terminated. Results of the Navy and NASA Langley Research Center pitch control margin simulation and flight test studies proved that high angle of attack pitch control margin can be demonstrated using a stabilized pushover method from 1g stabilized trim wings-level flight at high angles of attack, which minimizes the thrust and performance effects. A nose-down command applied at initial conditions at which the pitch attitude or the flight path angle is changing results in changes in angle of attack that are not due solely to the nose-down moment generated by the application of controls. With residual pitch attitude or flight path angle rates, the pilot technique and resulting motion are more complex, which reduces the repeatability of the maneuver and complicates the analysis.

During the NASA/Navy studies, simulation and flight test results for linear pitch acceleration were compared and found to correlate well. However, at aft center of gravity locations with nonlinearities in pitch acceleration, qualitative flight test data diverged from predicted simulation trends due to motion cue effects and increased pilot sensitivity to degraded pitch response. Flight tests also revealed that extensive workload was required to stabilize the air vehicle at the specified initial conditions of 1g, stabilized, trim at high angles of attack. Since the simulation tests used preset conditions, the pilots were not required to perform the maneuver setups during the piloted simulation portion of the evaluation. When the maneuvers were duplicated during flight tests, the pilots discovered establishing the required angle of attack and 15º pitch attitude was very difficult because the pilot had to "close the loop" on trim airspeed with the throttles to stabilize the flight path angle.

5.2.2.1.7.2 Longitudinal control power in unaccelerated flight.
Longitudinal control effectiveness shall not limit the ability to attain and hold any speed from V_S to V_{max} in steady 1-g flight at any altitude within the ROTH.

REQUIREMENT RATIONALE

This requirement is intended to insure the pilot can maintain equilibrium level flight throughout the flight envelope by normal means.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.3.1
MIL-STD-1797: 4.2.7.1.

REQUIREMENT LESSONS LEARNED

Controllability at speeds down to the 1-g stall speed is generally deemed necessary for safety, as well as full utilization, of such maneuvering air vehicles as those used by the military. V_{max}, the high-speed boundary of the ROTH, must be at least $V_{o_{max}}$; beyond that, it may be set by the contractor.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. Operational flight test will help reveal any deficiencies in pitch control power. Compliance may be demonstrated measurement of the gradient of pitch control force with airspeed (5.2.2.1.2.4).

The controls are to be used in their normal manner, and sideslip minimized. It is important to explore all speed-altitude corners of the ROTH. For example, a transonic tuck or high-speed dives can be critical due to combined aeroelastic and Mach number effects. Extremes of static stability or instability (Mach number, AOA, c.g.) will be critical. Also, hinge moments may limit control deflection and aeroelastic deformations may affect controllability.

5.2.2.1.7.3 Longitudinal control power in maneuvering flight.
Within the ROSH, it shall be possible, by use of the pitch control alone, to achieve the load factors of table XXXIV. This maneuvering capability is required at constant altitude at the 1-g trim speed and, with trim and throttle settings not changed by the crew, over a range about the trim speed the lesser of ±15% or ±50 kts EAS (except where limited by the boundaries of the ROSH).

TABLE XXXIV. Required load factor ranges for maneuvering flight.

Level	Load Factor Range
1	
2	
3	

REQUIREMENT RATIONALE

The pitch axis controller must be sufficiently powerful to produce an adequate range of load factors for maneuvering. Fixed-wing air vehicles generally use the pitch controller to affect flight-path changes.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.3.2
MIL-STD-1797: 4.2.7.2.

Recommended values for table XXXIV:

Level	Load Factor Range
Levels 1 and 2	n_{o-} to n_{o+}
Level 3	From n = 0.5 g to the lower of: a) n_{o+}, or b) n = 2.0 g for $n_{o+} \leq 3$ g = $0.5(n_{o+} + 1)$ for $n_{o+} > 3$ g

REQUIREMENT LESSONS LEARNED

The Level 1 and 2 values stem from mission performance needs; while the Level 3 values, which vary to an extent with maneuverability of the aircraft, are related to flight safety. The requirements for control effectiveness over a ±15 percent range about the trim speed assure that excessive amounts of pitch-surface-fixed static stability or instability will not limit maneuver

capability unduly, for any possible mechanization of the trim system. Where pitch control authority limits normal-acceleration capability, the requirement at off-trim speeds often will be the designing consideration for pitch control effectiveness. At most flight conditions, however, it would be expected that investigation at the nominal airspeed would be sufficient.

This requirement is restricted in application to the ROSH, with relaxed requirements for infrequent Failure States. Outside the ROSH, whatever falls out of the design is acceptable, as long as the other control requirements are met; in particular, note the dive pullout (5.2.2.1.8.7 and 5.2.2.1.8.8) and control margin (5.1.3 and 5.2.2.1.7.1) requirements. High supersonic speed and aeroelasticity and large, high hinge moments at high dynamic pressure tend to restrict the control capability. The Level 3 requirement assures modest nose-down and nose-up control capability for stabilization as well as for altering equilibrium and maneuvering. AFWAL-TR-83-3015 recommends requiring +1.5 g capability for Level 3 at the design dive speed, against the most adverse stabilizer (pitch) trim. That is consistent with some civil aircraft requirements for a 2.5 g transport: 0.5 (n_L + 1) is 1.75 g.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall proceed or accompany careful build-up to suspected critical flight conditions.

Flight test techniques are discussed under 5.2.2.1.8.1. Operational flight test shall be required for final demonstration, in order to assure that any problems are uncovered before introduction into service. Since demonstration is at constant throttle setting for 1-g equilibrium, these turns will involve loss of either speed or altitude. While in a real sense sustained turn performance is a flying quality, its specification has traditionally been a performance matter. The flying qualities requirement is on steady-state turn capability, with throttle setting for 1-g flight, in order to avoid trim-change effects which are specified separately. Another term sometimes used is instantaneous turn capability; properly that refers to a dynamic overshoot capability.

Critical flight conditions are indicated by the expression

$$C_m(\alpha, q, \delta, M, \bar{q}, x_{cg}) + (z_T / \bar{c})[T/(\bar{q}S)] = 0$$

where T is the 1-g trim thrust (which normally varies little with n_z) and δ is a generic pitch control deflection. Throttle setting is constant.

Measuring α to the principal axis and n_z normal to the flight path:

$$\delta = \frac{-C_{L_1}}{C_{m_\delta} - \frac{C_{m_\alpha}}{C_{L_\alpha}} C_{L_\delta}} \left\{ \frac{C_{m_\alpha}}{C_{L_\alpha}} (n_z - 1) + \left[\frac{C_{m_q}}{4\mu} - \frac{I_{z_p} - I_{x_p}}{n_z\, m\, \bar{c}^2} \frac{g\, \bar{c}}{V^2} \sin\alpha \cos\alpha \right] \left(n_z - \frac{1}{n_z} \right) \right.$$

$$\left. \mp \frac{H_e}{m\, V\, \bar{c}} \cos(\alpha + \varepsilon) \sqrt{1 - 1/n_z^2} \right\}$$

where $C_{L_1} = W/(\bar{q}\, S)$, $\mu = (W/S)/(g\, \rho\, \bar{c})$, sub p indicates principal axes, H_e is the engine angular momentum $I_e\omega_e$, $n_z = \cos\gamma/\cos\phi$ and ε is the inclination of the engine rotor axis to the principal x axis. In the last term, the "-" is for right turns and the "+" is for left turns.

Initial flight safety restrictions to 80 percent of limit load may well dictate that the required Level 1 and 2 capability first be demonstrated in the course of the flight loads demonstration program. But the often critical effects of angle of attack can still be investigated earlier, at higher altitude.

5.2.2.1.7.4 Load factor onset and peak pitch rate.
Following an abrupt maximum pitch control step input from a steady level turn, the air vehicle shall achieve the normal load factors of table XXXV and the peak body-axis pitch rates of table XXXVI within the specified time interval for the given initial conditions. The integral of the body-axis pitch rate over this time interval shall yield the specified change in attitude. For Normal States, the air vehicle shall not develop yaw or roll rates in excess of __(1)__ during this maneuver. For Extreme States, the air vehicle shall not develop yaw or roll rates which cause departure during this maneuver. Overshoots that result from abrupt pullups into the lift- or control-system-limited region of the load factor boundary shall not result in departure or exceedance of load factor limits. The air vehicle shall immediately return to level flight by relaxing the stick force.

TABLE XXXV. Load factor achieved following max pitch control step inputs.

Air Vehicle States	Initial Conditions			Load Factor Achieved (g)	Time Interval (sec)
	Altitude (ft)	Airspeed (KEAS)	Load Factor (g)		
Normal					
Extreme					

TABLE XXXVI. Peak pitch rates following max pitch control step inputs.

Air Vehicle State	Initial Conditions			Peak Body-Axis Pitch Rate (deg/sec)	Time Interval (sec)	Integral of Body-Axis Pitch Rate Over Time Interval (deg)
	Altitude (ft)	Airspeed (KEAS)	Load Factor (g)			
Normal						
Extreme						

REQUIREMENT RATIONALE

Stating the operational load factor boundaries is not enough to ensure satisfactory maneuvering response. Recent experience with F-5, F-15, and F-16 has shown that for aircraft that are departure resistant, abrupt pitch commands for point and shoot capability or rapid deceleration provide increased mission effectiveness. Use of relaxed static stability can require angle-of-attack/load factor limiters and pitch-rate or angle-of-attack-rate anticipation features to be incorporated into the flight control system to ensure that the limits are not exceeded. The effects of these features must be evaluated to ensure that load factor response is adequate.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.2.7.2.1.

Recommended value for blank (1): 10º/sec

For recommended values for table XXXV and table XXXVI, see Requirement Lessons Learned.

REQUIREMENT LESSONS LEARNED

There has been little work done to define satisfactory load factor onset rates. The following, derived from F-15 flight test data, can be used as a basis for comparison. The F-15 is considered to have good response characteristics.

F-15 LOAD FACTOR RESPONSE			
Altitude (ft)	Speed Range (kts eas)	Pilot Control Input from Trimmed Initial Conditions	Response
20,000	V = 250	Step pitch control input from 1.5-g turn to max	Integral under body axis pitch rate curve equals a change in attitude of 32° in 1.1 sec
20,000	300 < V < 400*	Ramp pitch control input (≈10% max control surface rate) from a trimmed 2-g turn	From 2-g to 7-g in 1.8 sec

*NOTE: Between 300 and 400 kts, flight test data on the aircraft with a g-limiter indicates that it took 3 sec to go from 2 g's to 6 g's with a step input. The report indicated that this was inadequate for air-to-ground operations. All load factors are measured at the c.g.

VERIFICATION

Operational tests shall be required for final demonstration, to assure any problems are uncovered before introduction into service. The critical flight conditions are dependent upon the characteristics of the particular vehicle.

5.2.2.1.7.5 Longitudinal control power for takeoff.
The effectiveness of the pitch control shall not restrict the takeoff performance of the air vehicle and shall be sufficient to prevent overrotation to undesirable attitudes during all types of takeoff. During the takeoff roll, at __(1)__, pitch control effectiveness shall be sufficient to __(2)__. Satisfactory takeoffs shall not depend upon use of the trimmer control during takeoff or on complicated control manipulation by the pilot. These requirements shall be met on __(3)__.

5.2.2.1.7.5.1 Longitudinal control power in catapult takeoff.
Furthermore, in aircraft designed for catapult takeoff, the effectiveness of the pitch control shall be sufficient to prevent the air vehicle from pitching up or down to undesirable attitudes in catapult takeoffs at speeds ranging from __(4)__ to __(5)__. Satisfactory catapult takeoffs shall not depend upon complicated control manipulation by the pilot.

REQUIREMENT RATIONALE

The requirement is intended to regulate against air vehicles that exhibit no apparent pitch response to commands during the takeoff roll until flying speed is reached (V_{min}). These air vehicles give no assurance that rotation will be forthcoming, but then tend to "pop off", resulting in overrotation and a necessity for immediate control reversal to avoid stall.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.3.3 and 3.2.3.3.1
MIL-STD-1797: 4.2.7.3.

Recommended values for Blanks (1) and (2):

Air Vehicle Class	Blank (1)	Blank (2)
Nosewheel air vehicles, all Classes	0.9 V_{min}	obtain the pitch attitude which will result in lift-off at $V_{min}(TO)$
Tailwheel air vehicles, Class I	0.5 $V_S(TO)$	maintain any pitch attitude up to that for a level thrust-line
Tailwheel air vehicles, Classes II, III, and IV	$V_S(TO)$	maintain any pitch attitude up to that for a level thrust-line

Recommended values for Blank (3): Normally, Blank (3) shall be "hard-surface runways." In the event that the air vehicle has a mission requirement for operation from unprepared fields, Blank (3) shall be "hard-surface runways and unprepared fields."

Recommended value for Blank (4): $V_{C_{min}}(CT)$

Recommended value for Blank (5): $V_{C_{min}}(CT)$ + 30 kts

Note that Requirement 5.2.2.1.7.5.1 applies only to aircraft designed for catapult takeoffs.

The term takeoff includes the ground run, rotation, and liftoff, the ensuing acceleration to $V_{max}(TO)$, and the transient caused by assist cessation. Takeoff encompasses operation both in and out of ground effect.

REQUIREMENT LESSONS LEARNED

This requirement is based on operational experience, which has shown that the ability to control pitch attitude to achieve the proper attitude for liftoff before V_{min} is necessary for acceptable liftoff at V_{min}. The requirement is especially important for turbojet aircraft for which relatively large pitch attitudes are required for liftoff. A related requirement is 5.2.2.1.8.6.

The requirement on takeoff control power is more important for single-engine aircraft or any multi-engine aircraft which have V_{MCA} (the airborne minimum control speed for a propulsion failure) equal to or less than V_{min}. For multi-engine aircraft where V_{MCA} is greater than V_{min}, the requirement could be relaxed to 0.9 V_{MCA}.

All aircraft will have to operate from hard-surface runways, and therefore hard surfaces were used as the basic requirement. An increased coefficient of friction, however, such as occurs with unprepared fields, increases the elevator effectiveness required for nosewheel aircraft, but

decreases the effectiveness required of tailwheel aircraft. The increased rolling friction force on the main wheels gives a nose-down pitching moment about the aircraft c.g. Nose-wheel liftoff speed will increase monotonically with increasing main-gear coefficient of rolling friction, approaching the speed for takeoff in the ground attitude. But tailwheel liftoff speed will decrease with the increasing main-gear coefficient of rolling friction until just the application of takeoff thrust will rotate the aircraft at zero speed. Then a different technique would be required. The value of 0.9 V_{min} is a compromise between early enough indication of controllability and minimization of any tendency to overrotate.

Single-engine propeller-driven airplanes with a T-tail have been deficient in terms of nosewheel rotation prior to liftoff. As a result, pilot acceptance is very poor. Takeoff performance over an obstacle has been demonstrated to be considerably worse in one T-tailed aircraft than in an identical aircraft with a conventional horizontal tail. Delayed liftoff has been attributed to inability to rotate to the takeoff attitude prior to V_{min}. The root cause is that the horizontal tail is out of the propeller wake. Multi-engine aircraft which are not normally lifted off until V_{MCA} (which is usually above V_{min}) do not have as strong a requirement for nose rotation at 0.9 V_{min}. As an indication, multi-engine airplanes with T-tails have generally been found to be acceptable to pilots.

On the F-18, the pitch control-surface deflection range of ±20º moved with pitch trim position, but could never exceed 24º trailing-edge-up (-) stabilizer. It was possible to mis-set trim for takeoff so that -24º deflection could not be obtained when needed during takeoff. The pilot shall always have full deflection available.

The requirement for control in catapult takeoffs could also be applied to ski-jump takeoffs or use of a jump-strut.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall proceed or accompany careful build-up to suspected critical flight conditions.

Normally the critical test will be to start the rotation, i.e., to at least balance the aircraft with the nosewheel or tailwheel just touching the runway but bearing no load. The main gear extension at this condition, which determines the attitude there, can be found from the lift and load stroke equations. Control to maintain the takeoff attitude shall also be checked for compliance with the limitation of 5.2.2.1.8.6 on control travel during takeoff. In flight test, takeoff performance tests are monitored for flying qualities.

The ability to comply with this requirement should be obvious during operational flight test. Special emphasis shall be placed on short-field takeoffs at the maximum forward and aft center-of-gravity limits for liftoff and overrotation, respectively.

From the equation

$$\bar{q}\, S\, \bar{c}\, C_m + T\, z_T - (W - \bar{q}\, S\, C_L - T \sin\xi)\,(X + \mu\, Y) = 0$$

the liftoff speed is given by

$$\frac{1}{2}\rho\, V_{LO}^2 = \bar{q}_{LO} = \frac{W}{S}\left[\frac{1 - \frac{T}{W}\left(\sin\xi + \frac{z_T/\bar{c}}{X/\bar{c} + \mu\, Y/\bar{c}}\right)}{C_L + \frac{C_m}{X/\bar{c} + \mu\, Y/\bar{c}}}\right]$$

where $\xi = i_T + \alpha$; C_m and C_L are the aerodynamic pitching moment and lift coefficients, in ground effect, of the entire aircraft with takeoff thrust and fully deflected pitch controller; X and Y are the horizontal and vertical distances between the c.g. and the main gear axle; and μ is the main gear coefficient of rolling friction.

A balance must be struck between adequate control and overrotation tendency, e.g., the DC-9-80 takeoff performance tests. The test pilot should be alert to avoid tail strike.

5.2.2.1.7.6 Longitudinal control power in landing.
For Levels 1 and 2, the pitch control shall be sufficiently effective in the Landing Flight Phase in close proximity to the ground that, in Calm air, during landing flare and rollout with the air vehicle trimmed for the Approach Flight Phase at the minimum recommended approach speed [not to exceed 1.3 $V_S(L)$]:

a. The geometry-limited touchdown attitude can be achieved at touchdown, or alternatively

b. The lower of $V_S(L)$ or the guaranteed minimum landing speed [$V_{min}(L)$] can be achieved when flaring from shallow (γ = -3º) and steep (γ = -6º) approaches,

and the __(1)__ can be gently lowered to the ground at speeds down to __(2)__. The pitch control forces required to meet these requirements shall be pull forces and shall not exceed __(3)__. This limit applies both in and out of ground effect.

REQUIREMENT RATIONALE

This requirement is to assure adequate pitch control during flare and rollout in ground effect. It insures that the air vehicle can be pitched up sufficiently, in ground effect, to achieve the guaranteed minimum landing speed. It also insures that the nosewheel or tailwheel can be gently lowered to the ground during landing rollout. The forces required in landing shall always be natural (that is, pull for flare) and shall be small enough to allow one-handed operation without placing excessive demands on the pilot.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.3.4 and 3.2.3.4.1
MIL-STD-1797: 4.2.7.4 and 4.2.8.6.2.

Recommended values for Blanks (1) and (2):

Landing Gear Configuration	Air Vehicle Class	Blank (1)	Blank (2)
Nosewheel	All	Nosewheel	0.9 $V_{min}(L)$
Tailwheel	Class I	Tailwheel	0.5 $V_{min}(L)$
	Classes II, III, and IV	Tailwheel	0.75 $V_{min}(L)$

Recommended values for Blank (3):

Controller	Air Vehicle Class	Blank (3)
Centerstick and Wheel	Classes I, II-C, and IV	35 lbs
	Classes II-L and III	50 lbs
Sidestick	All	The forces shall not be objectionable to the pilot.

REQUIREMENT LESSONS LEARNED

Elevator effectiveness can be severely degraded in ground effect due to a decrease in downwash caused by the presence of the ground plane. A shallow approach is specified to eliminate the possibility of performing most of the flare out of ground effect. Steep approaches are also required, as they tend to result in firmer touchdowns, making it difficult to keep the nosewheel from slamming to the ground (at forward c.g.). The -6º glide path is generally attainable, but could be moderated where the mission usage warrants.

The Mitsubishi Mu-2 (a twin-engine turboprop) is well known for a rapid pitchover immediately at touchdown. Service difficulties with flight instruments and avionics are felt to result from this high shock environment.

The T-46 had no guaranteed landing speed; since its geometry-limited touchdown attitude was greater than its stall angle of attack, $V_S(L)$ governs. For vehicles augmented to counter degraded static stability, the change in center of rotation from c.g. to main landing gear at touchdown might result in an uncontrollable situation even if ample control power exists. The original Space Shuttle in-flight control configuration did not provide a stable control loop or fast enough trim capability after main gear touchdown due to the change in control geometry. It was difficult to lower the nose wheel gently. Control system parameter switching at touchdown was required to correct this. For the Shuttle, then, control effectiveness was more than just moment-producing capability. Some augmented aircraft, the X-29 for one, put up with a great deal of control

surface movement while taxiing in order to assure safe transition between ground and air. In any case the stability and control augmentation must function effectively and safely during bounces, hard landings, etc., though the aircraft response characteristics are quite different on the ground than when airborne. A canard control surface must not be allowed to stall: a gust at touchdown would be hard on the nose gear.

The force limits are intended to prevent unreasonable demands on the pilot at landing, where one-handed operation is almost mandatory and corrections are continually made. Although some pilots hold a control force, trim in approach (after extension of gear/flaps, etc., but before landing flare) is a generally valid standard reference condition. The recommendations explicitly include landing conditions in and out of ground (taken to include carrier deck, sea, etc.) effect. Crosswind also may reduce or negate ground effect.

These force limits are more restrictive than the civil requirements: FAA FAR Part 25 allows 75 lbs and FAA FAR Part 23 60 lbs for stick or 75 lbs for wheel controllers (the quoted values for short-term application). However, operational experience shows support for lower limits for continuous maneuvering. For one thing, this allows keeping some pull force at equilibrium conditions on approach, as mentioned in the paragraph above. Furthermore, the recommended maximum force again be too high for the average female pilot.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall proceed or accompany careful build-up to suspected critical flight conditions. Ultimately, compliance must be proven through flight test at forward c.g.

The requirement is structured so that verification may largely, if not entirely, be a part of maximum performance landing tests. No particular guidance is deemed required except to note that the maximum forward c.g. (generally regardless of weight) defines the critical flight condition for nose-up control of a stable airframe. Crosswind may reduce ground effect; flight tests shall be in Calm air. In general, ground effects often do not correlate too well between wind tunnel and flight.

Analysis and simulation are subject to the accuracy of aerodynamic estimates in ground effect. The flight test shall encompass approach or that part of preparation for landing that involves gear/flap extensions, power adjustment, and pitch trim over velocities from 1.3 V_S, at the most forward center of gravity position. Both power-on and -off landings, if applicable, shall be performed, to assure that the power setting does not affect the force characteristics too greatly. Generally, the most stable loading and configuration will be critical.

5.2.2.1.8 Longitudinal control forces and displacements.
The longitudinal control forces and displacements shall meet the requirements of 5.2.2.1.8.1 through 5.2.2.1.8.8. When a cockpit control is adjustable for pilot physical dimensions or comfort, the control force limits in these requirements refer to the mean adjustment. A force referred to any other adjustment shall not differ by more than 10% from the force referred to the mean adjustment.

REQUIREMENT RATIONALE

The 10% force allowance for control adjustments is intended to assure keeping within the threshold of pilot perception.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.5.2.4
MIL-STD-1797: 4.1.12.3.

REQUIREMENT LESSONS LEARNED

This section contains the control force gradients and limits to be applied to the pitch controller. As a word of introduction, several points must be made which are applicable to all of the following force requirements:

a. In general, the force requirements of MIL-F-8785C are unchanged. This is due in part to lack of good, valid test data to justify changes, even if anecdotal information should suggest such a change is warranted.

b. The requirements should be considered to be stringently applicable for male pilots only. There is almost no available data for setting requirements for female pilots. A limited amount of data, reviewed below, suggests considerably lower limits would be needed. This of course presents a dilemma in setting limits for aircraft expected to be routinely operated by both male and female pilots.

c. Maximum forces specified appear in most cases to be quite large for the weaker male pilots for continuous operation.

d. Effects of stick (or wheel) geometry and position on maximum force capabilities are not explicitly covered in any of the requirements, though it is obvious that control location will affect maximum attainable forces. This can be seen in the discussion that follows.

In a review of past research, Lockenour (AFFDL-TR-78-171) discussed the effect of stick location on push and pull capability (figure 64), the effect of upper arm angle on push and pull

strength for the 5th and 95th percentile male (figure 65). The data shown in these figures are from Human Engineering Guide to Equipment Design for male Air Force personnel in the sitting position. As described in AFFDL-TR-78-171 these data show that:

> "…one's maximum force capability is not symmetric left and right and varies about a factor of two for forward and aft stick positions….pull and push strength differ significantly and…the 5th and 95th percentile male strengths differ by as much as a factor of three…Certainly a given stick force at the grip will feel heavier to the pilot for aft stick positions. Also one must be very careful in correlating the acceptability of stick forces for various aircraft to include the effect of stick location and maximum stick deflection. For example, the F-5A stick deflection is greater than that of the A-7D by more than a factor of two. This places the stick in a different location in the cockpit for maximum deflection."

A more recent study, AFAMRL-TR-81-39, presents a comparison between male and female strength characteristics for operating an aircraft control stick. Table XXXVII summarizes the percentiles for maximum forces exerted by 61 men and 61 women on an aircraft control stick during a 4-second static exertion with the right hand only. The 5th percentile values of men and women from these tests are also shown for comparison on figure 64. The figure shows a 95th percentile woman has approximately the same performance as a 5th percentile man. As AFAMRL-TR-81-39 observes, the force limits in this standard may not be consistent with the capabilities of pilots.

Figure 66 shows data from Human Engineering Guide to Equipment Design illustrating the effect of wheel angle on maximum push and pull capability for the 5th percentile male. The data are again for male Air Force personnel, using the right arm only; the wheel grips are 18 inches above the seat reference point (SRP) and 15 inches apart. figure 66a shows the various wheel angles and positions from the SRP. The greatest push and pull capability occurs at the furthest position of the wheel where the pilot's entire arm is used. This can be seen on figure 66b where the push capability at 23.25 inches from the SRP is approximately twice that obtained when the control wheel is at its closest at 10.75 inches from the SRP. Similarly, the maximum pull capability varies almost by a factor of two on figure 66c depending on the control wheel angle and position.

It must be stressed that these are maximum forces in single applications; clearly, continuous operation (such as would be expected in meeting any of the force requirements) would produce much lower maximum forces. In a discussion of some general principles of control design, in one- versus two-handed operations. Human Engineering Guide to Equipment Design states that:

> "For controls requiring single applications of force, or short periods of continuous force, a reasonable maximum resistance is half of the operator's greatest strength. For controls operated continuously, or for long periods, resistances should be much lower…Controls requiring large forces should be operated with two hands (which, for most controls, about doubles the amount of force that can be applied),

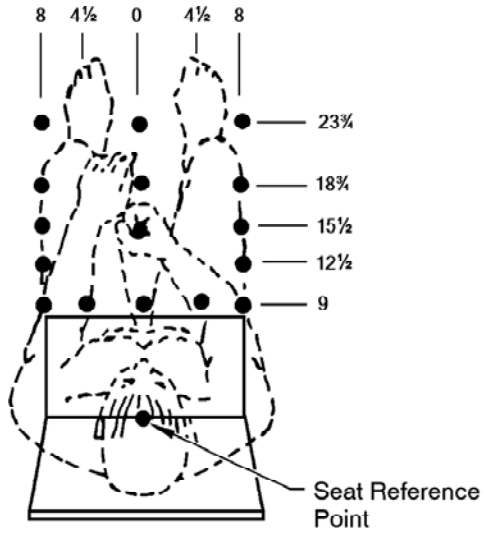
> depending on the control type and location and on the kind and direction of movement as follows:
>
> When two hands are used on a stick…located along the body midline, pull is generally almost doubled. Push is doubled near the body but is only slightly stronger at distances away from the body…
>
> When two hands are used on stick…controls located on either side of the body midline, at or beyond the shaller, pull is approximately doubled, push is not greatly increased except at close distances…"

Since this standard will likely be used for few aircraft with unpowered controls, little design penalty would result from lowering the maximum allowable forces. Stability and control augmentation, and response feel systems, could still keep the lightest force gradients within the presently allowable limits. Concerned has been expressed, however, that with lighter maximum forces some heavy-handed pilots might be inclined to overstress the vehicle. In particular that might be the fate of trainers with new student pilots. Then, especially in a simple aircraft, a lower gradient at forward c.g. might make the gradient too low at aft c.g.

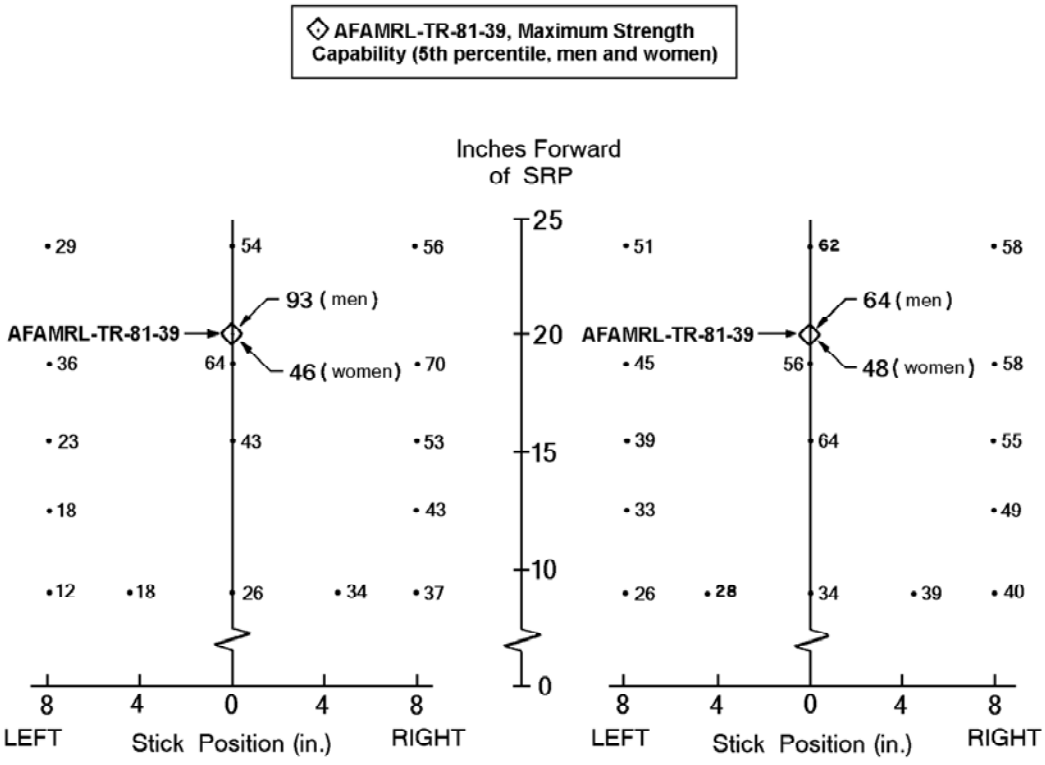
VERIFICATION

Verification shall be via verification of requirements 5.2.2.1.8.1 through 5.2.2.1.8.8.

The longitudinal control force is that component of applied force exerted by the pilot on the cockpit control in the direction to command pitch, acting at the center of the stick grip or wheel in a direction perpindicular to a line between the center of the stick grip and the control column pivot.



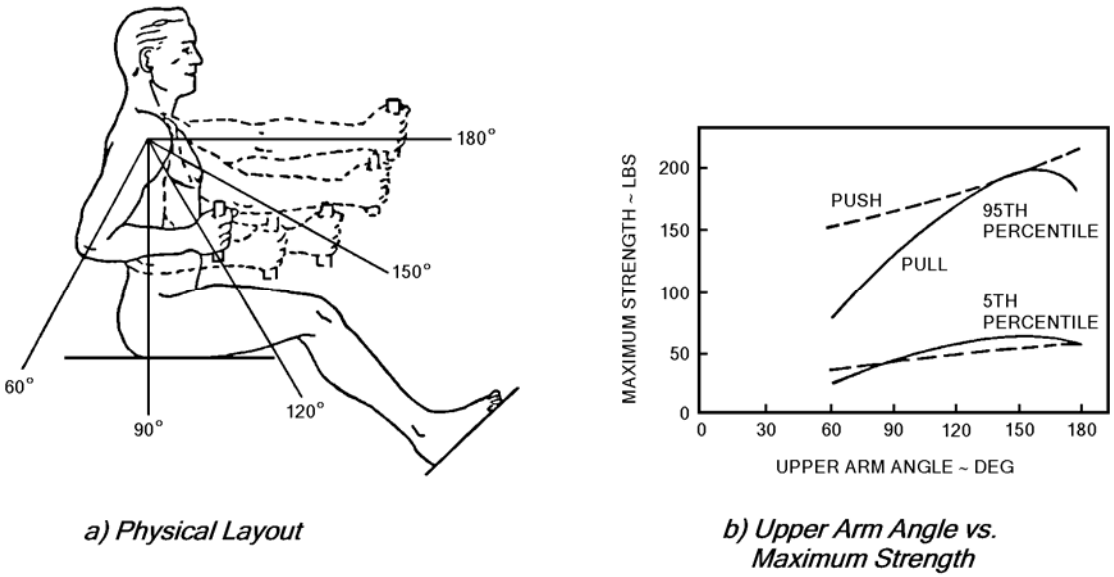
a) Physical Layout for Stick Force Tests of "Human Engineering Guide to Equipment Design" (stick 13½ inches above Seat Reference Point)



b) Maximum Push Capability (lb)

c) Maximum Pull Capability (lb)

FIGURE 64. Effect of arm/stick geometry on maximum push and pull capability by the right arm for the 5th percentile male (Human Engineering Guide to Equipment Design).



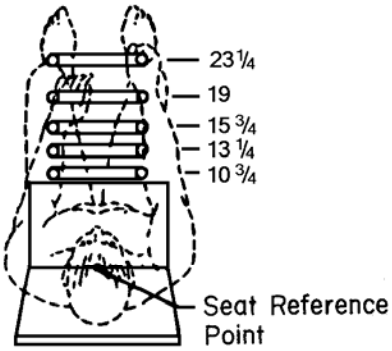
a) Physical Layout

b) Upper Arm Angle vs. Maximum Strength

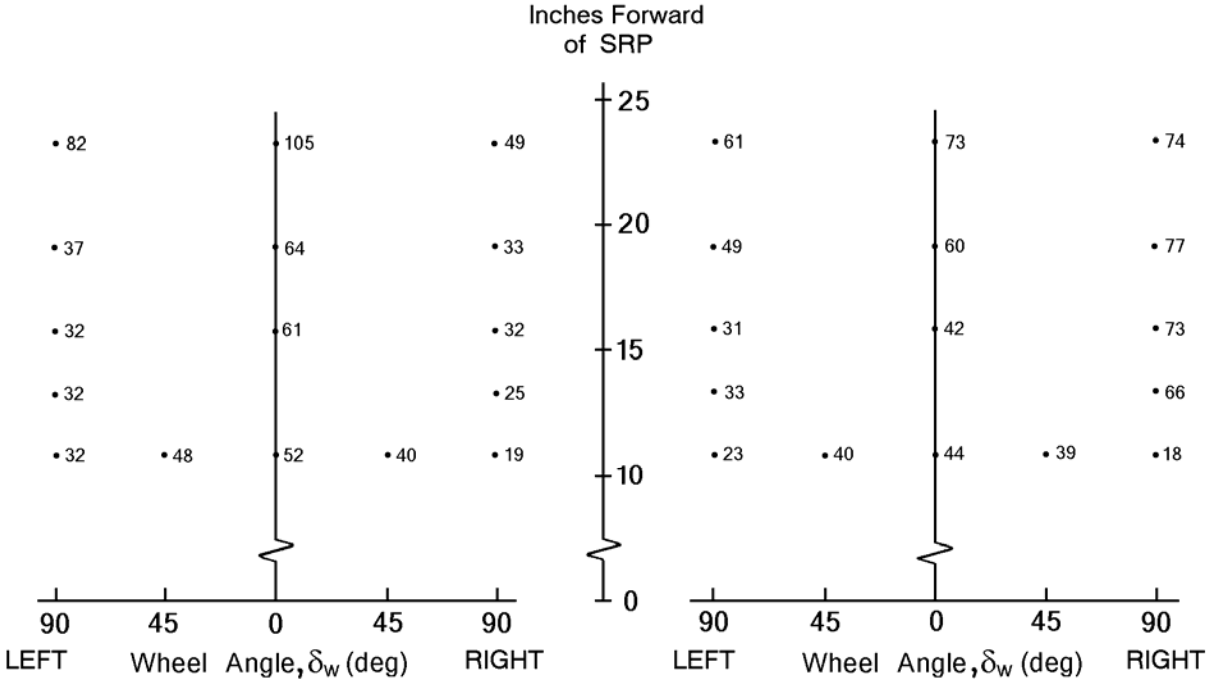
FIGURE 65. Effect of upper arm angle on pull and push strength for the 5th and 95th percentile male (Human Engineering Guide to Equipment Design).

TABLE XXXVII. Maximum forces exerted on aircraft control stick (lbs) by men and women (AFAMRL-TR-81-39).

Control Stick Direction	Men			Women		
	Percentile			Percentile		
	5th	50th	95th	5th	50th	95th
Stick Forward (Push)	93	123	165	46	87	109
Stick Back (Pull)	64	85	106	48	52	64



a) Physical Layout for Wheel Force Tests of "Human Engineering Guide to Equipment Design" (wheel 18 inches above seat reference point)



b) Maximum Push Capability (lb)

c) Maximum Pull Capability (lb)

FIGURE 66. **Effect of arm position and wheel angle on maximum push and pull capability by the right arm for the 5th percentile male (Human Engineering Guide to Equipment Design).**

5.2.2.1.8.1 Steady-state control force and deflection per g.
In steady turning flight and in pullups and pushovers at constant speed, for all local gradients throughout the ROTH, the incremental control force and control deflection required to maintain a change in normal load factor and pitch rate shall be in the same sense (aft - more positive, forward - more negative) as that required to initiate the change. At constant speed in steady turning flight, pullups, and pushovers the variations in pitch controller force with steady-state normal acceleration shall have no objectionable nonlinearities within the load factor range from __(1)__ to __(2)__. Outside of this range a departure from linearity resulting in a local gradient which differs from the average gradient for the maneuver by more than 50% is considered excessive, except that larger increases in force gradient are permissible at load factors greater than 0.85 n_L. All local force gradients shall be within the limits of table XXXVIII. In addition, for Level 1, F_s/n_z shall be near the upper boundaries of table XXXVIII for combinations of high frequency and low damping. Also, for Air Vehicle Normal States with high frequency and low damping, F_s/n_z for Level 1 shall be near the maximum gradient limit. For the purposes of these requirements, the term gradient does not include that portion of the force-versus-n_z curve within the breakout force.

TABLE XXXVIII. Pitch maneuvering force gradient limits.

Level	Maximum Gradient $(F_s/n)_{max}$ (lb/g)	Minimum Gradient $(F_s/n)_{min}$ (lb/g)
1		
2		
3		

REQUIREMENT RATIONALE

These requirements relate to the classical stick-free maneuvering stability (stick force per g, F_s/n) at constant speed. The basic premise is that F_s/n represents a necessary tactile cue for pilot regulation of load factor. Low values of F_s/n result in excessive sensitivity with a tendency toward exceeding the aircraft structural limits. High values lead to pilot fatigue during maneuvering. The above requirements are also intended to assure that control motion and aircraft motion are in harmony, with positive maneuvering stability throughout the ROTH.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.2.2 and 3.2.2.2.1
MIL-STD-1797: 4.2.8.1 and 4.2.8.4.

Recommended value for Blank (1): For Classes I, II, and III: 0.5 g
For Class IV: 0 g

Recommended value for Blank (2): 0.5 (n_{o+} + 1) or 3 g whichever is less

Recommended values for table XXXVIII:

Type of Controller	Level	Maximum Gradient $(F_s/n)_{max}$ (lb/g)	Minimum Gradient $(F_s/n)_{min}$ (lb/g)
Centerstick	1	240/(n/α) but not more than 28.0 nor less than 56/(n_L – 1)*	the greater of 21/(n_L – 1) and 3.0
	2	360/(n/α) but not more than 42.5 nor less than 85/(n_L – 1)*	the greater of 18/(n_L – 1) and 3.0
	3	56.0	the greater of 12/(n_L – 1) and 2.0
Wheel	1	500/(n/α) but not more than 120.0 nor less than 120/(n_L – 1)	the greater of 35/(n_L – 1) and 6.0
	2	775/(n/α) but not more than 182.0 nor less than 182/(n_L – 1)	the greater of 30/(n_L – 1) and 6.0
	3	240.0	5.0

* for $n_L < 3$, $(F_s/n)_{max}$ is 28.0 for Level 1, 42.5 for Level 2

REQUIREMENT LESSONS LEARNED

It was decided that the major differences in the desired maneuvering forces between fighter aircraft and transports are due to the type of controller in addition to Aircraft Class. The effects of Aircraft Class (really a grouping of types of missions) seem to be described adequately by limit load factor, through the K/(n_L – 1) formulas of MIL-F-8785C. In addition, however, there are several arguments for having different maneuvering forces for centerstick and wheel controllers. For example, the lower limits on maneuvering forces must be higher with a wheel control because the pilot's arm is usually unsupported; whereas the pilot has very good vernier control with a centerstick even with light forces because his forearm is partially supported on his thigh. In any case, pilots seem to agree that they cannot maintain the precision of control with a wheel that they can with a stick, and that the maneuvering control forces should be higher for the wheel.

There is some evidence (ASD-TDR-63-399 and AFFDL-TR-67-51) that F_s/n at very low n/α can or should be higher than at n/α. This is possibly due to a gradual change from concern with load factor and structural protection at high n/α to concern with control of pitch attitude alone at low

n/α. Specification of forces in the form of limits on F_s/α at low n/α can be accomplished by making the F_s/n limits vary inversely with n/α: at constant speed, $F_s/\alpha = (F_s/n)\,(n/\alpha)$.

On the basis of these considerations, the upper limits on F_s/n were expressed in the form $K/(n/\alpha)$ at low n/α and $K/(n_L - 1)$ at high n/α, with separate requirements for stick and wheel controllers. On the basis of long experience with unpowered-control airplanes, which tend to have F_s/n invariant with airspeed, the lower boundaries do not vary with n/α.

Effects of stick/wheel position on acceptability of F_s/n are not covered by these requirements. But with the variation in force capability shown in the discussion of 5.2.2.1.8 it seems intuitively obvious that there must be an interrelationship of force and deflection gradients with controller location.

Another item for consideration is the allowance in table XXXVIII for considerably higher values of F_s/n when n/α is low. For example, figure 67 (from AFFDL-TR-78-171) illustrates the variation of F_s/n with airspeed for the OV-10A aircraft, due to an elevator spring tab. The apparently large and rapid change in F_s/n actually results in fairly constant maximum stick force at stall (figure 68) or n_L, whichever comes first. Perhaps that becomes the important factor at speeds below maneuvering speed (V_A, see figure 67).

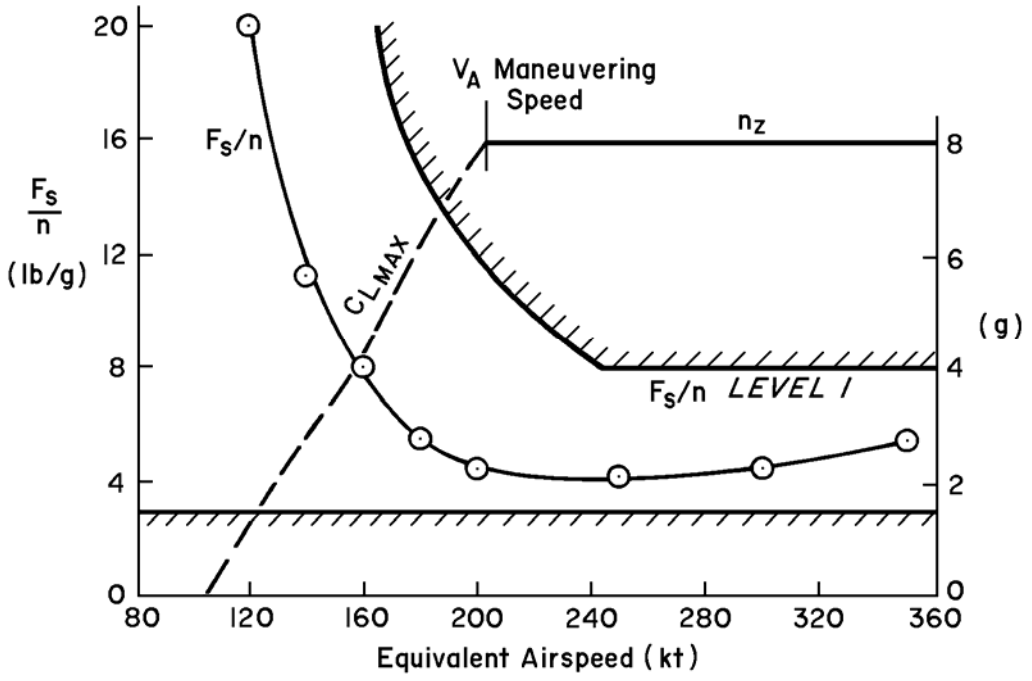


FIGURE 67. OV-10A maneuvering control (AFFDL-TR-78-171).

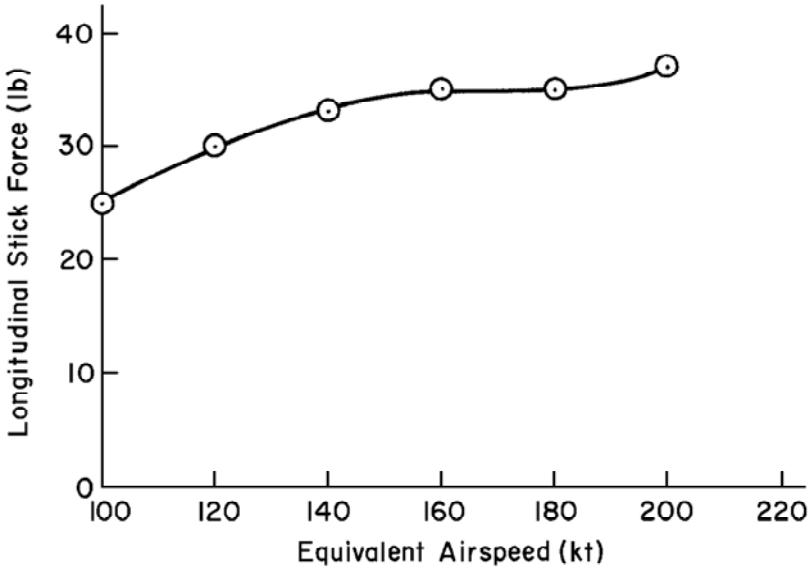


FIGURE 68. Longitudinal stick force at stall (AFFDL-TR-78-171).

Much of the available information on existing vehicles suggests that the lower limits of F_s/n for wheel controllers may be too high. For example, figure 69 (from AFFDL-TR-78-171) summarizes elevator control force gradient characteristics of three large cargo or transport airplanes, for Category B at forward and aft center-of-gravity conditions. According to this summary the L-1011 complies with the maximum and minimum control force gradient requirements. The C-5A at forward c.g., pitch dampers off, compares favorably with the Level 1 maximum values, however, some gradients at aft c.g. fall below the Level 1 and Level 2 boundaries (only the SAS off data were presented for comparison). The C-141A/YC-141B data slightly exceed the Level 1 maximum limit at forward c.g. Pilot comments for the C-141A/YC-141B support the maximum boundary. According to Silvers and Withers (AFFDL-TR-75-3), "The C-5A control force gradients are rated satisfactory and acceptable", so the minimum boundary for Level 1 requirements appears to be too high in this instance.

Results from simulation and flight test of different stick force gradients on the B-1 bomber (AFFDL-TR-78-171) are summarized below:

Consolidated pilot comments—based on flight simulator tests:

- 21 lbs/g Level 1 minimum is too high for 2-g aircraft
- 7-8 lbs/g Level 1 is a good design guide for 3-g aircraft
- 3 lbs/g minimum shall be maintained for failure modes

B-1 terrain following flight pilot comments:

- 17 lbs/g too high based on fatigue:
  - Meets $n_L = 3$ requirements
  - Below $n_L = 2$ requirements
- Over rugged terrain: 10 minutes is tiring
- Composite terrain: two pilots sharing task
  - Short task: 30 minutes
  - Medium task: 1 hour
  - Long task: 2 hours

The pertinent findings were: 1) that a minimum F_s/n of 7-8 lbs/g for Level 1 was acceptable for an aircraft with $n_L = 3$ g [where, by table XXXVIII $(F_s/n)_{max} = 10.5$ lbs/g]; and 2) that F_s/n as low as 17 lbs/g was too high for terrain-following flight, based upon pilot fatigue. The conclusion suggests that acceptable values of F_s/n are task-dependent (or time-dependent), though a relaxed lower limit alone might suffice. The acceptability of both minimum and maximum values of F_s/n may be directly related to workload: e.g., high gradients may be undesirable if the pilot is required by the task to divert his attention or to track tightly in the presence of atmospheric disturbances. Low gradients in an emergency situation may lead to overcontrol.

Especially for fighter aircraft, nonlinear gearing and force gradients are commonly used to get a gradient good for fine tracking without a fatiguing force level for gross maneuvering. Carried too far, however, these nonlinearities can promote pitch-up like an aerodynamic pitch-up. The F-15 design, which barely meets the specified linearity requirement has been well accepted.

A pitch rate command/attitude-hold system, compensated for bank angle, (AFWAL-TR-81-3118) seems well accepted for approach and landing of large aircraft. The initial gradient can meet this requirement, but an integrator zeros the control force in steady turns (a g-limiter was incorporated).

Many Class I (general aviation) aircraft tend to fall around and below the minimum F_s/n requirements. Figure 70 compares various aircraft in landing configurations (gear and flaps extended) with the wheel controller requirements of table XXXVIII. The data were obtained from NASA TN D-3726 and FAA-RD-77-25. F_s/n limits are drawn for $n_L = 3.8$ (Federal Aviation Regulations requirements for Normal category operations) and $n_L = 2.0$ (the limit specified for most of the aircraft in landing configuration). With the single exception of the Cessna 177, none of the aircraft of figure 70 at aft c.g. meet the $(F_s/n)_{min}$ requirement for

n_L = 2.0 [the Citabria is an aerobatic category (6-g) stick-controlled design]. The seven-aircraft of NASA TN D-3726 resulted in a range of 19 < F_s/n < 45 lbs/g for acceptable gradients for Normal category operations (see figure 70). No information is available on what was considered unsatisfactory, though 5 lbs/g was considered to be too light; these low gradients allowed the limit load factor to be attained too easily.

During F-14A flight tests, a reduced stick force gradient was noticed at load factors greater than 4 g. In simulated defensive break turns and rolling pullouts, using full lateral stick and rapid aft stick inputs lead to an overshoot of target load factor severe enough to increase the likelihood of exceeding established limits in aggressive transonic maneuvering. This is a deficiency which should be avoided in future designs, according to the Naval Air Test Center.

A more basic consideration relative to table XXXVIII is the complete absence of a force gradient requirement for sidestick controllers, reflecting in part the limited data base. However, AFFDL-TR-79-3126 and Class 79A Final Report of USAF Test Pilot School, based on a series of flight tests conducted by students of the USAF Test Pilot School, gives some insight into preferred gradients. These data, figure 71, are for an air-to-air tracking task. This test aircraft was the USAF/Calspan variable-stability NT-33, with a T-38A utilized as the target. The ratings shown are the average over three pilots, all with fighter experience. An approximate Level 1 boundary is suggested on figure 71. In general, the F_s/n range is comparable to that of table XXXVIII; i.e., 2-14 lbs/g. The relatively-weak frequency dependence may also exist for centerstick controllers, although there are no data to support this. More data from these evaluations are given in Requirement Lessons Learned for 5.2.2.1.8.4.

The F_s/n gradients on figure 71 are the initial values; as mechanized, the slope at larger deflections was half the initial slope. The breakout forces were 0.5 lb. AFFDL-TR-79-3126 suggests that the sidestick neutral position

> "…be oriented so that in wings-level unaccelerated flight the pilot need never move his wrist further aft than 5-7° forward of vertical to command maximum permissible load factor…
>
> Available data would tend to support a neutral position of 10° to 17° forward of vertical and 8° to 12° left (inboard) of vertical…. A pilot adjustable armrest is absolutely mandatory, and its design can influence pilot acceptability as much as any other parameter."

A simple, conventional control system requires more control deflection per g at low speed than at high speed, even though stick force per g remains fairly constant. The change in deflection gradient is a useful cue to airspeed, one of the few available cues in maneuvering with head out of cockpit.

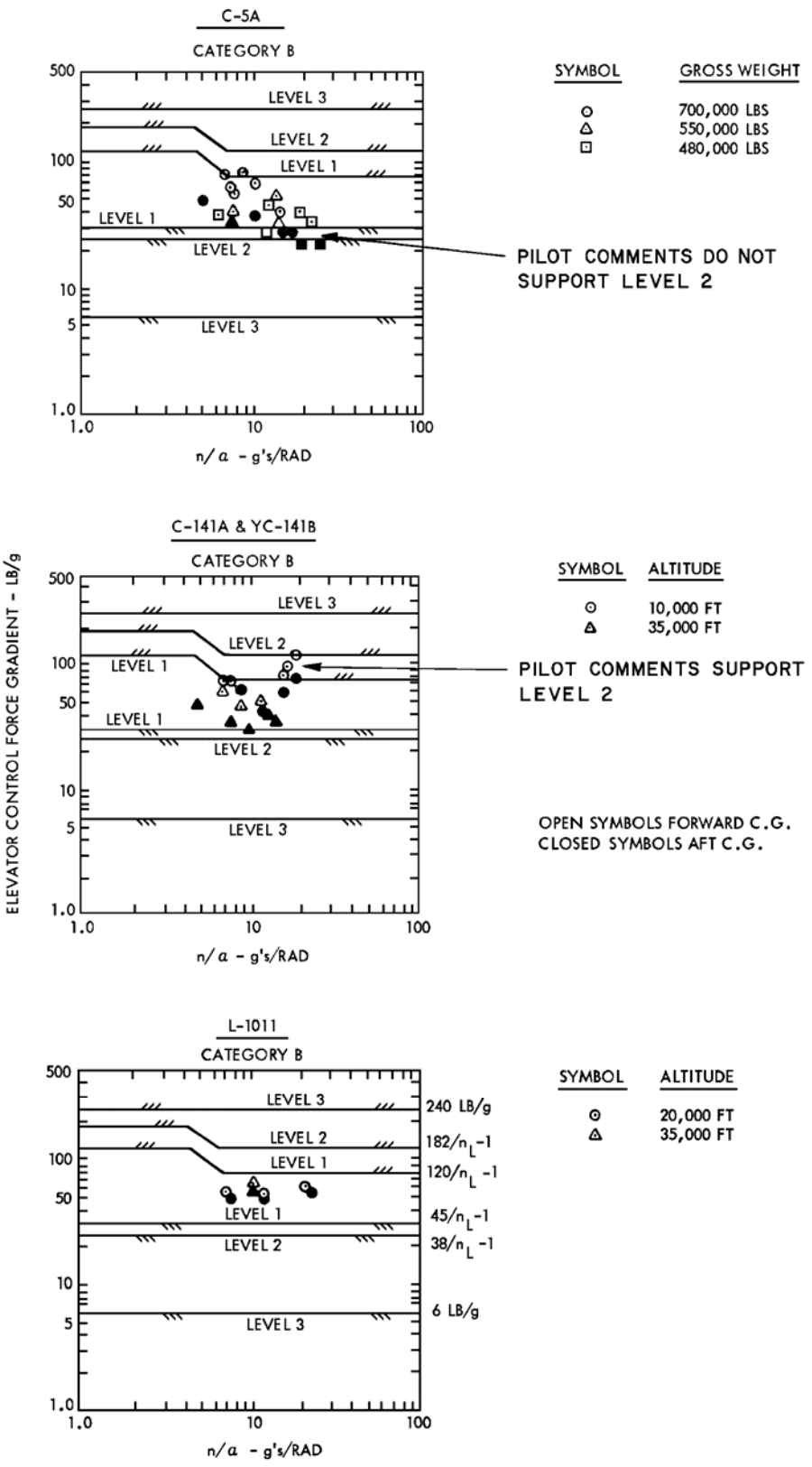


FIGURE 69. Elevator control force gradients for transport aircraft (from AFFDL-TR-78-171).

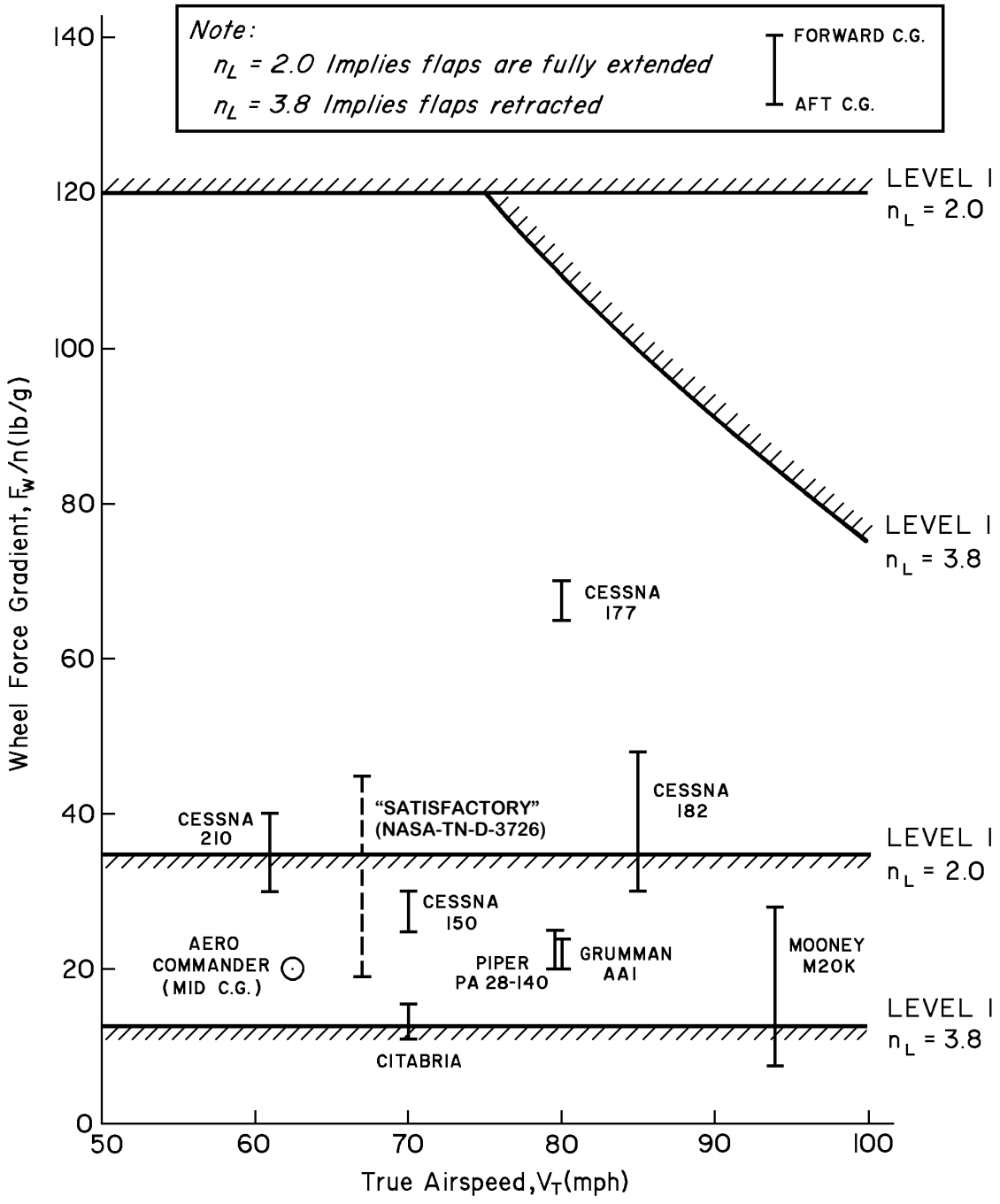


FIGURE 70. **Comparison of F_s/n for various Class I airplanes in landing approach (Category C) with limits of table XXXVIII.**

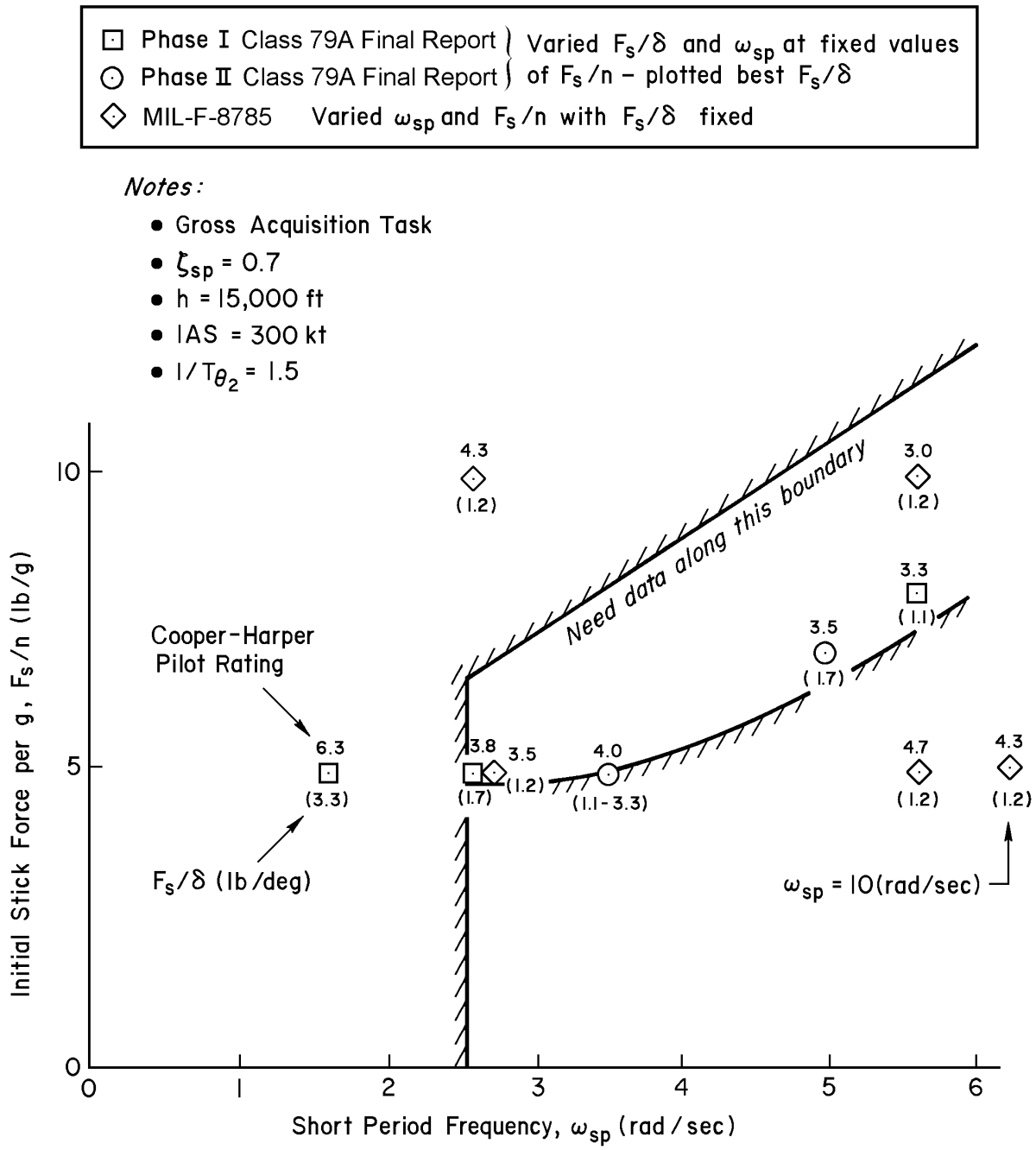


FIGURE 71. Short-period frequency versus longitudinal stick force per g (F_s/δ separately optimized) for a sidestick controller.

VERIFICATION

The following discussion is concerned with the flight testing aspects of determining control force characteristics. A valid moving-base piloted simulation may be used in proof-of-concept stages to assess the suitability of force and gearing nonlinearities.

There are several methods for obtaining the required control force data. The best method to use depends primarily on the speed range under consideration. The major factor in determining the appropriate method for a given speed range is that load-factor control gradients are defined for constant speed. The method selected must therefore result in zero or small speed changes with n, or at least include a means for eliminating the effects of any speed changes. At speeds where characteristics vary significantly with Mach number, speed shall be interpreted as Mach number, since that is the primary pilot reference.

One method is to use a series of alternating symmetric pullups and pushovers, sequenced so as to minimize the airspeed and altitude changes. The control is held fixed after each input until the short-term motion becomes steady state, and measurements are taken at a near-level attitude.

Another method is to perform a series of stabilized turns after trimming the aircraft in level flight. The load factor can be changed by changing the bank angle, and the airspeed held constant by using a different rate of descent for each load factor. The throttle and trim controller shall be left at their trim settings throughout the maneuver to minimize the possibility of introducing extraneous pitching moments. The gradients obtained in this manner will not be quite as linear as the symmetric pullup method because of the difference in pitch rate between pullups and turns (see, e.g., Airplane Performance Stability and Control). But, with the possible exception of a more stable slope near 1 g in the turns, the differences are generally small and can easily be accounted for, if necessary, knowing pitch rate. Of course load factors between ±1 g cannot be obtained in near-level turns. A progressively tightened turn of this sort, at constant airspeed, is a wind-up turn, Navy style (USNTPS-FTM-103).

A third method that is sometimes used is a wind-up turn, Air Force style (FTC-TIH-79-2). After trimming in level flight, a turn of a certain number of g's is initiated, and the speed is allowed to decrease slowly as the g-level and altitude are held constant. The test is then repeated at several other g-levels until the complete range is covered. In this way, control gradient data can be obtained rapidly for several speeds. Again, the trimmer and throttle shall be left at the trim settings and the rate of change of airspeed controlled by changing the rate of descent. The major disadvantage of this method is that it is less accurate because more careful pilot technique is required.

In general, the symmetric pullup method will work well at high speeds, but the airspeed changes will be excessive if the method is used at low speeds. On the other hand, the turn methods work well at low speeds, but can cause excessive altitude changes at high speeds. For the X-29, constant-altitude (±2000 ft) wind-up turns worked well.

5.2.2.1.8.2 Transient control force per g.
The peak pitch control forces developed during abrupt maneuvers shall not be objectionably light. The buildup of control forces during maneuver entry shall not lag the buildup of normal acceleration at the pilot's location. The frequency response of normal acceleration at the pilot station to pitch control force input shall be such that the inverse amplitude is greater than the limits of table XXXIX for all frequencies greater than 1.0 rad/sec.

TABLE XXXIX. Minimum F_s/n (at frequencies above 1 rad/sec).

Level	Minimum F_s/n (lb/g)
1	
2	
3	

REQUIREMENT RATIONALE

This requirement accounts for the possibility that stick force per g for high-frequency inputs may be reduced considerably below the steady-state limits set in 5.2.2.1.8.1. Such reduced values of F_s/n may lead to pilot-in-the-loop oscillations. This requirement is intended to help prevent such an occurrence.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.2.3.1 and 3.2.2.3.2
MIL-STD-1797: 4.2.8.2.

Recommended values for table XXXIX:

Type of Controller	Minimum F_s/n (lb/g)		
	Level 1	Level 2	Level 3
Centerstick controllers and other one-handed controllers	$\frac{14}{n_L - 1}$	$\frac{12}{n_L - 1}$	$\frac{8}{n_L - 1}$
Wheel controllers and other two-handed controllers	$\frac{30}{n_L - 1}$	$\frac{25}{n_L - 1}$	3.3
Sidestick controllers	The minimum F_s/n shall not be objectionable to the pilot.		

Lacking any other data, use of the centerstick values for one-handed controllers and the wheel-controller values for two-handed controllers of other types is suggested.

REQUIREMENT LESSONS LEARNED

Bobweights tend to alter the phasing between the pilot's force inputs to the stick and the resulting stick motion. For an aircraft which obtains all of its F_s/n (steady-state) from a bobweight, for instance, the stick can be moved with essentially zero force to initiate a rapid pull-up. As the end response develops, the bobweight then tries to pull the stick back to neutral, thus requiring that the pilot add increasing forces to hold the control input. This means that the stick position leads the control force by a considerable amount at moderately high frequencies. Also, if the damping of the control system itself is low, the pilot will feel the stick constantly slapping against his hand during rapid maneuvering. Requirements have thus been set on controls-free damping in terms of dynamic F_s/n and feel system phasing.

The requirements concerning ζ_{sp}, ω_{sp}, and F_s/n will normally be sufficient to insure adequate maneuvering characteristics. In certain situations, however, these requirements alone will not insure against PIOs. Consider, for example, an aircraft that meets the Level 2 requirements on ζ_{sp}, ω_{sp}, and F_s/n for Category A Flight Phases. If both ζ_{sp} and F_s/n are near the lower limits, the aircraft can have PIO tendencies serious enough to make it unacceptable. The requirements listed above are designed to prevent this situation by setting an upper limit on frequency-response amplitude of $n(s)/F_s(s)$ [expressed as a lower limit of $F_s(s)/n(s)$]. This has the effect of increasing the minimum F_s/n requirements of 5.2.2.1.8.1 for low values of ζ_{sp} (stick-free), as can be seen by examination of figure 72. The dip in amplitude corresponds to the short-period resonance; and the size of the dip, expressed as the ratio $(F_s/n)/(F_s/n)_{min}$ is a unique function of ζ_{sp} (stick-free) (assuming that the control-system natural frequency is appreciably higher than ω_{sp} and that the pilot location is not extreme), figure 73. It can be seen that F_s/n must increase rapidly with decreasing values of stick-free ζ_{sp} in order to maintain a given value of $(F_s/n)_{min}$.

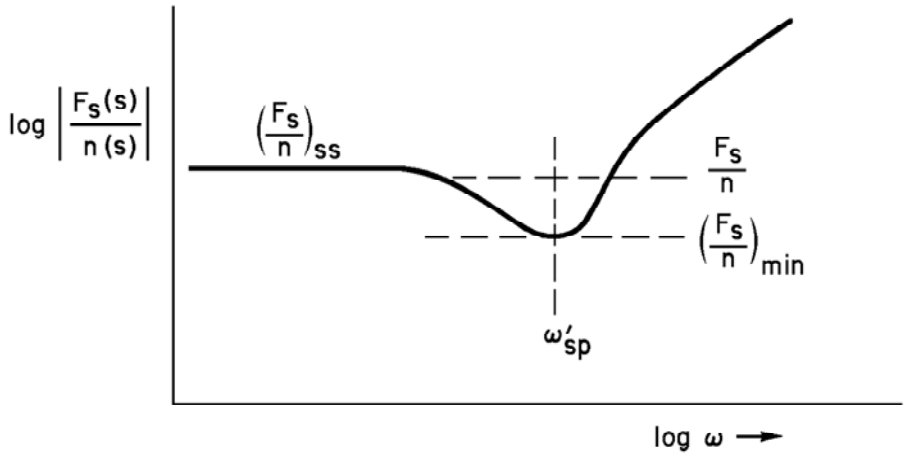


FIGURE 72. Illustration of resonance dip in F_s/n due to low ζ_{sp}.

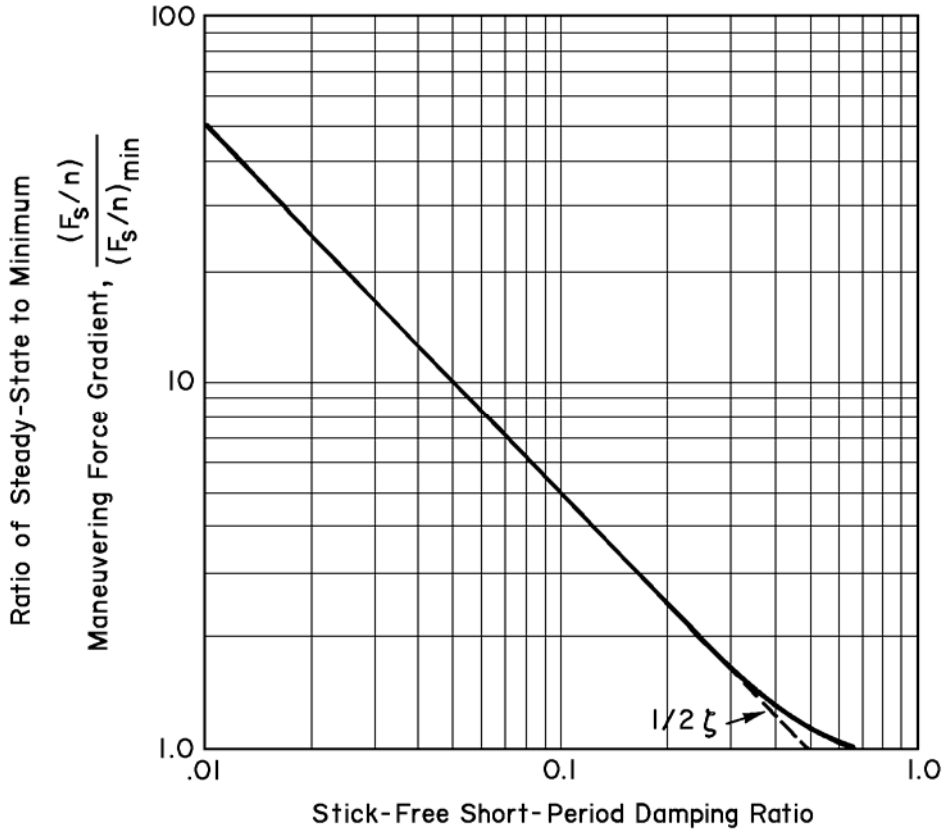


FIGURE 73. The ratio $(F_s/n)/(F_s/n)_{min}$ versus ζ_{sp}.

It should be understood that if the control system natural frequency is not appreciably higher than ω_{sp} (stick-free), the frequency response $F_s(s)/n(s)$ will not be entirely second-order in the region of $(F_s/n)_{min}$. If the control system damping is not very high, as is often the case, the resonant dip can be accentuated by the control system mode, as can be seen from figure 74. In

this situation, an equivalent ζ_{sp} (stick-free) can be obtained from figure 73 by measurement of $(F_s/n)/(F_s/n)_{min}$. That value may not be the same as obtained by fitting a lower-order system to the actual frequency response. For this requirement the actual $(F_s/n)_{min}$ shall be used. The effect of cockpit distance from the instantaneous center of rotation for control inputs, and of structural modes, will usually, though not necessarily, be negligible.

The requirements stated above are intended to inhibit development of longitudinal PIOs. However, to the extent that $(F_s/n)_{min}$ is defined by ζ_{sp}, the requirement is redundant; i.e., figure 73 is defined by F_s/n and ζ_{sp}. Multiplying $(F_s/n)/(F_s/n)_{min}$ from figure 73 by the required minimum value of F_s/n gives curves which can be compared with the Level 1, 2, and 3 boundaries of steady-state F_s/n and ζ_{sp} from 5.2.2.1.1.2 and 5.2.2.1.8.1 for a 7-g aircraft. The shaded areas on figure 75 indicate regions where the F_s/n requirement is not redundant. It is noteworthy that dynamic F_s/n is not a consideration for Level 1 but has increasing influence for Levels 2 and 3, respectively. Lightly damped controls system or structural modes that occur near the short-period frequency will increase the influence of the $(F_s/n)_{min}$ requirement; i.e., the $(F_s/n)_{min}$ boundaries on figure 75 will have a tendency to shift to the right in terms of ζ_{sp} (though not necessarily in terms of the equivalent ζ_{sp} of 5.2.2.1.1.2). The NT-33 test program of AFFDL-TR-66-163 involved pilot assessments of various short-period configurations in Category A Flight Phases.

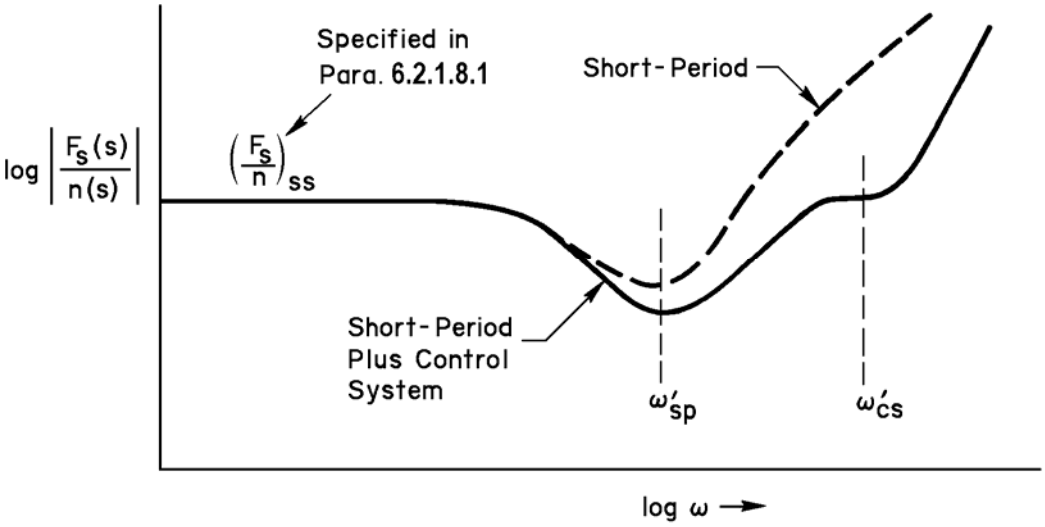


FIGURE 74. Effect of control system on resonant dip.

VERIFICATION

To obtain flight data, parameter identification methods can be used or the pilot can pump the stick sinusoidally at various frequencies. Several techniques have been employed to aid the pilot in this task. One method is described in FTC-TR-66-24, where the pilot visually follows an oscillating spot on the instrument panel. In other studies, oscillating aural tones have been fed to the pilot through earphones. Response to non-sinusoidal input can be analyzed readily via Fast Fourier Transform; e.g., Applied Time Series Analysis, Vol. I, Basic Techniques.

If the frequencies desired are not too high or too low, pilots can do an amazingly good job of moving the stick sinusoidally with no aids whatsoever. In addition, if the damping ratio is not too high, the pilot can find the resonant dip in the F_s/n versus frequency curve fairly accurately, by pumping at the frequency that gives the most aircraft response for the least effort.

Testing is required with the most aft c.g., since for a given configuration and flight condition this is the condition for lowest F_s/n. For meaningful analysis or simulation, a linear approximation of the flight control system and aircraft must be accurate. In the end, however, PIO tendencies need to be evaluated in flight. Ground-based simulator evaluations may be of little value.

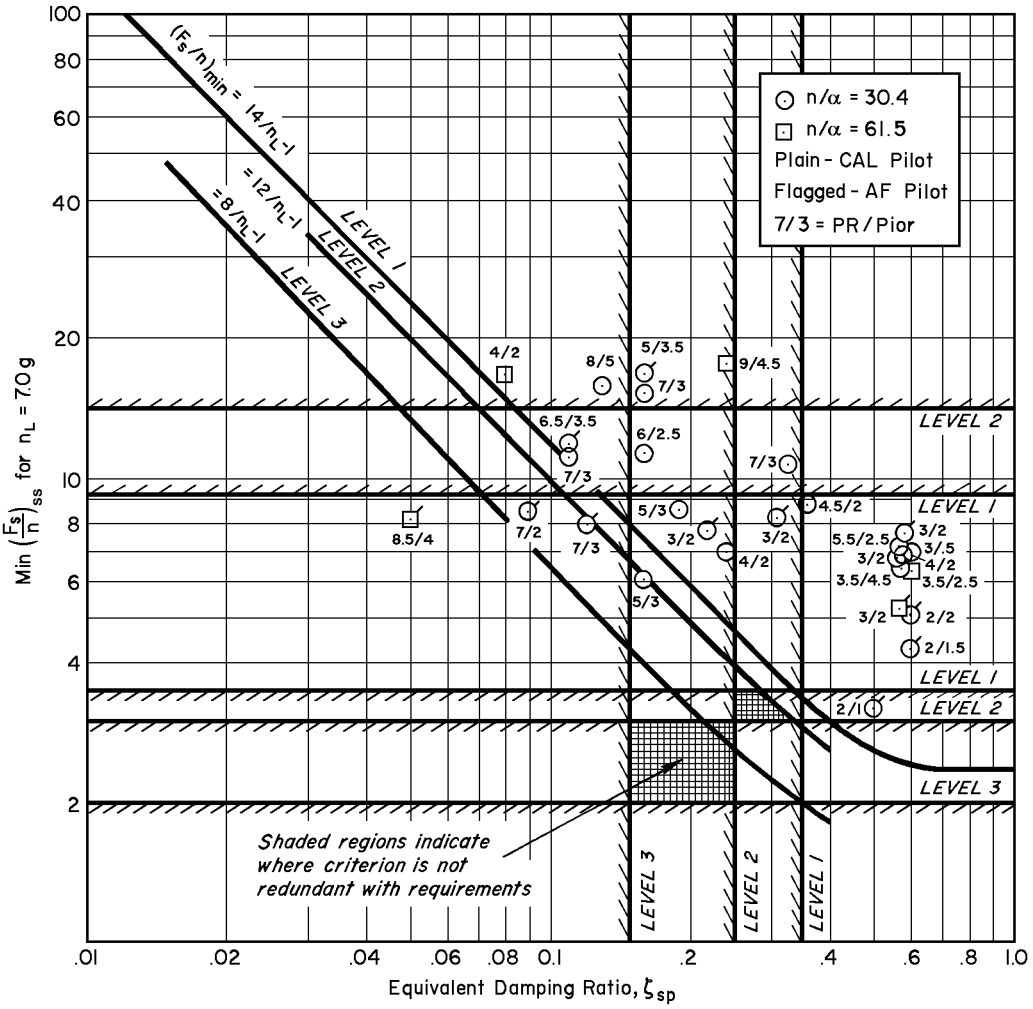


FIGURE 75. Comparison of $(F_s/n)_{min}$ boundaries with $(F_s/n)_{ss}$ and ζ_{sp} for cases where $\omega_{sp} << \omega_{cs}$ (AFFDL-TR-66-163).

5.2.2.1.8.3 Control force variations during rapid speed changes.
When the aircraft is accelerated and decelerated rapidly through the operational speed range and through the transonic speed range by the most critical combination of changes in power, actuation of deceleration devices, steep turns and pullups, the magnitude and rate of the associated trim change shall not be so great as to cause difficulty in maintaining the desired load factor by normal pilot techniques.

REQUIREMENT RATIONALE

This is intended to prevent unduly large pitch control force gradients with speed, which require excessive trimming or high steady control force. There are two kinds of problems for which this requirement is primarily intended. First, aircraft can have stick force and position gradients with speed which are so stable that considerable pilot effort is required during rapid speed-change maneuvers. Second, in the transonic region the local gradients may change so rapidly with Mach number that it is difficult for the pilot to maintain the desired pitch attitude or normal acceleration during rapid speed changes.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.1.1.2
MIL-STD-1797: 4.2.8.3.

REQUIREMENT LESSONS LEARNED

The requirement is the result of operational experience with early supersonic airplanes. Although difficulties were experienced, enough data have never been collected for more than a qualitative requirement. The AFTI/F-16 experienced a pitch-up tendency due to a rapid slow-down while performing high-g rolls at transonic speeds.

VERIFICATION

Application of the requirement should be straightforward, except that its qualitative wording leaves demonstration of compliance purely subjective. There is no better way to apply it than simply performing acceleration and deceleration maneuvers typical of extreme task demands, including emergency decelerations, and asking the pilot about difficulties.

Early analysis can determine transonic pitching and control forces to maintain 1-g flight as well normal acceleration with fixed controls. The flight profile of the critical test for this requirement will be a function of gearing and feel mechanization, as well as of the normal and emergency maneuvers to be expected. Generally, the transonic trim change increases with lift coefficient, and so will be more pronounced at high load factors and high altitude.

Simulator and eventually flight evaluations, covering the operational speed range (and the transonic speed range, if applicable) shall be conducted. Forward c.g. gives the largest

magnitude of C_{m_α} for stable aircraft; an unstable one undoubtedly will be highly augmented and thus likely to experience less difficulty.

5.2.2.1.8.4 Control force versus control deflection.
For all types of pitch controllers, control deflections in maneuvering flight shall not be so large or so small as to be objectionable. For Category A Flight Phases, the average gradient of pitch control force per unit of incremental pitch control deflection at constant speed shall not be less than __(1)__ nor greater than __(2)__ for Levels 1 and 2.

REQUIREMENT RATIONALE

Both control force and control deflection provide pilot cues. This requirement is intended to assure consonance between the two and adequate control deflection cues where only control force is specified. There is evidence that stick deflection characteristics, while secondary to force characteristics, are still important to the pilot when maneuvering. For maneuvering flight it is necessary that control displacements be comfortable and natural to the pilot.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.2.2.2 and 3.2.2.3.2
MIL-STD-1797: 4.2.8.4 and 4.2.9.2.

Recommended value for Blank (1): For wheel and centerstick controllers: 5 lbs/in.
For sidestick controllers: 1 lb/deg

Recommended value for Blank (2): For wheel and centerstick controllers: no data
For sidestick controllers: 2.5 lbs/deg

REQUIREMENT LESSONS LEARNED

Proper design of the pitch control power (5.2.2.1.7.3) and forces (5.2.2.1.8.1) for maneuvering flight should result in the above requirements being met. However, these requirements function as final tests of the adequacy of the controller specified by 5.2.2.1.7.3 and 5.2.2.1.8.1.

Requirements 5.2.2.1.8.1 and 5.2.2.1.8.2 set limits on pitch control force. This requirement sets limits on the controller deflection corresponding to a given force. The Lessons Learned show a need for such a requirement, and intuitive logic indicates some interrelationship among control forces, deflections, and motions. As with control force, any control position reversal in maneuvering flight (stick-fixed maneuvering instability) requires the pilot's concentration and promotes loss of control.

Limited flight test data strongly support a lower limit on F_s/δ_s (indeed, WADC-TR-55-299 suggests that F_s/δ_s should be 25 lbs/in. or higher for centerstick controllers). The specific limits

are directly related to F_s/n and control location (see 5.2.2.1.8.1). However, they are not well defined at this time.

Some guidance for designing sidestick controllers may be gained from figure 76 (reproduced with minor changes from AFFDL-TR-79-3126) a sidestick evaluation by the USAF Test Pilot School using the variable-stability NT-33 (with Level 1 short-period characteristics) produced a series of ratings and comments for varying F_s/n and F_s/δ_s. The lateral force/deflection characteristics were varied as shown in table XL to maintain control harmony for the air-to-air evaluations (2-g bank-to-bank and 3.5-g wind-up turns). If the pilots commented that control harmony detracted from the rating given, variations in control harmony were evaluated. With the exception of the light F_s/n and large F_s/δ_s (configurations 1 and 2), there is not a substantial variation in pilot ratings over the test matrix.

In general, pilots preferred increased control stick motion with decreased control force gradients and decreased control stick motion with increased control force gradients. Control Configurations 13, 14, and 15 of figure 76 yielded the best results, both in pilot ratings and comments. Pilots indicated that control motions were noticeably large, but not uncontrollable. These configurations were on the edge of the test matrix; thus, the extent of this favorable region was not determined and additional testing is warranted.

Configurations 4 and 7 were found to be good, but slightly inferior to Configurations 13, 14, and 15. Pilot comments indicated that the stick forces for Configuration 4 were tiring and uncomfortable. Though the boundaries were not completely determined, these comments imply that even heavier force gradients would be unacceptable. Configurations 1 and 2 were rated the poorest, characterized by longitudinal and lateral oversensitivity.

All of the remaining control configurations indicate that with medium control stick motion the control force gradient selected had essentially no effect on pilot ratings. However, pilot comments show a trend from oversensitivity to sluggishness as the control force gradient increased from very light to heavy.

The effect of breakout force on pilot ratings was investigated by increasing the breakout force from 0.5 lb to 1 lb for control Configurations 7 and 11. For Configuration 7 the average pilot ratings increased from 3.8 to 5, whereas for Configuration 11 the ratings remained essentially unchanged. Pilot comments indicated that the effect of increasing breakout was to increase the pitch sensitivity in an unfavorable way.

Recent USAF Test Pilot School experiments with the NT-33 sidearm controller varied the force/deflection gradient, F_s/δ_s, and short-period frequency, ω_{sp}. F_s/n was 7 or 8 lbs/g for the high ω_{sp} and 5 lbs/g for the lower values of ω_{sp} in accordance with earlier results [FDL-TDR-64-60] as shown in table XLI. A summary of average pilot ratings (three pilots) and commentary is given on figure 77 for the gross acquisition task and on figure 78 for the fine tracking maneuver.

The poor pilot ratings for the low short-period frequency cases, $\omega_{sp} = 1.6$ rad/sec, are expected based on 5.2.2.1.1.2. However, the ratings for the lower values of F_s/δ_s are worse than Level 2, indicating that failure modes should be a consideration when contemplating light force/deflection

gradients. For ω_{sp} in the Level 1 region ($\omega_{sp} \geq 2.55$ rad/sec), larger values of F_s/δ_s (i.e., approaching a force stick) result in rapidly degraded pilot opinion in the fine tracking task (see figure 78). The optimum value of F_s/δ_s is seen to be about 1.7 lbs/deg until $\omega_{sp} \geq 5$ rad/sec, at which point the data indicate that decreasing F_s/δ_s is desirable (see figure 77).

There is little or no recent data available for analysis. The following discussion is taken directly from AFFDL-TR-69-72, with a few words added at the end. Data for sidestick controllers has been presented above.

The flying qualities investigations of WADC-TR-55-299, AFFDL-TR-67-51, AFFDL-TR-65-210, WADC-TR-57-719, WADC-TR-56-258, and AFFDL-TR-68-91 all include variations of control position per g as well as control force per g. AFFDL-TR-67-51 and AFFDL-TR-65-210 deal with the Landing Approach Flight Phase, while all the others are for Category A Flight Phases.

Both AFFDL-TR-67-51 and AFFDL-TR-65-210 indicate unfavorable pilot comments when the control motions required to maneuver the airplane become too large. Since these investigations were specific simulations of the some earlier C-5A configurations, the short-period frequency was below the Level 1 limit for Category C Flight Phases. When the short-period frequency is low, the pilots tend to overdrive the airplane with large, pulse-like inputs to speed up the response. Therefore the pilots might not have disliked the control motion gradients as much if the short-period response had been faster. Because of the uncertainties caused by the low short-period frequencies, and because of the limited amount of data, no attempt was made to place quantitative limits on control motion gradients for Category C Flight Phases.

Working under the assumption that there are lower limits on F_s/δ_s (upper limits on δ_s/F_s), the Level 1 and Level 2 boundaries were initially drawn as a best fit to the data of figure 79. There are not sufficient data to define a Level 3 limit. Although the only data plotted were those having Level 1 values of F_s/n, there are poorly rated configurations from WADC-TR-55-299 and WADC-TR-57-719 which lie inside the Level 1 F_s/δ_s boundary.

Because of strong objections from the manufacturers, the Level 1 and 2 limits shown on figure 79 were reduced to 5 lbs/in. Examples of "good" operational aircraft were produced with indicated gradients as low as 5 lbs/in. The requirement for a force/deflection gradient of at least 5 lbs/in. has been retained as a recommended lower limit from requirement 3.2.2.2.2 of MIL-F-8785C. This number seems to have originated more from a rule of thumb based on experience than from hard data. Hence more experimental data are deemed highly desirable.

The isometric sidearm controller used in the YF-16 and early F-16 was unsatisfactory to many pilots (see AFFTC-TR-75-15 and AFFTC-TR-79-10) and a change was deemed necessary. Their primary objection was the lack of tactile cues as to when the controller was at its limit. A small amount of motion, as permitted by the cockpit contours, was incorporated into the controller that was finally adopted. Although this sidearm controller clearly would not meet the requirements of this section, F-16 pilots seem to have adapted to this essentially fixed stick (F_s/δ_s very high). However, before accepting such a controller as being Level 1, it should be noted that these pilots had no alternative but to adapt. The NT-33 subject pilots (figure 76) did not feel that extreme force gradients were desirable when given the opportunity to compare with gradients across the spectrum.

Very Light (3.0)	13 No pitch bobble tendency but imprecise positioning. Avg of C-H 3.7	9 Pitch and lateral are both too sensitive. Avg of C-H 4.4	5 Pitch and lateral both a little too sensitive. Avg of C-H 5.1	1 Pitch extremely sensitive. Lateral fair. Avg of C-H 6.7
Light (4.0)	14 Pitch and lateral steady and responsive. Motion noticeably large. Avg of C-H 2.9	10 Pitch a little sensitive. Lateral bobble. Avg of C-H 4.3	6 Slight pitch bobble. Better at higher g's. Lateral sluggish (cont. harmony). Avg of C-H 4.5	2 Pitch too sensitive. Lateral wandering and sensitive. Avg of C-H 6.0
Medium (5.0)	15 Motion noticeably large. No pitch bobble, slightly sluggish. Avg of C-H 3.3	11 Very slight pitch bobble tendency, but good. Large lateral corrections difficult. Forces high & bobble. Avg of C-H 4.4	7 Pitch steady once on tgt. Lateral forces high (cont. harmony). Avg of C-H 3.85	3 Pitch a little sensitive. Lateral slow to respond. Avg of C-H 5.0
Heavy (8.6)	16 A/C very sluggish and forces uncomfortable. Avg of C-H 5.0	12 A/C sluggish but stable. Forces heavy. Avg of C-H 4.5	8 Pitch steady but forces too heavy. Lateral forces too heavy. Tiring. Avg of C-H 4.3	4 Pitch very stable at higher g's, but forces tiring. Avg of C-H 4.1
	1.1	1.4	2.0	5.0

Control Force/Deflection Gradient (lb/deg)

FIGURE 76. Pilot comments for air-to-air tasks with standard harmony (from AFFDL-TR-79-3126).

TABLE XL. Category A control configurations for NT-33 sidestick evaluations (AFFDL-TR-79-3126).

Configuration Number	Longitudinal		Lateral	
	F_{es}/n	δ_{es}/ F_{es} (deg/lb)	F_{as}/p	δ_{as}/ F_{as} (deg/lb)
1	Very Light	.2	Very Light	.3
2	Light	.2	Light	.3
3	Medium	.2	Medium	.3
4	Heavy	.2	Heavy	.3
5	Very Light	.5	Very Light	.77
6	Light	.5	Light	.77
7	Medium	.5	Medium	.77
8	Heavy	.5	Heavy	.77
9	Very Light	.7	Very Light	1.08
10	Light	.7	Light	1.08
11	Medium	.7	Medium	1.08
12	Heavy	.7	Heavy	1.08
13	Very Light	.91	Very Light	1.43
14	Light	.91	Light	1.43
15	Medium	.91	Medium	1.43
16	Heavy	.91	Heavy	1.43

The X-29 was able to attain limit load factor at all speeds with full aft stick. Combined with sensitive roll control, this resulted in poor control harmony at all speeds.

VERIFICATION

Control displacements shall be measured and assessed during the verification of 5.2.2.1.7.3 and 5.2.2.1.8.1. Flight testing shall provide proof of compliance with this requirement, at the most forward and most aft c.g., over the load factor range of the ROTH. The subjective requirement for control motions to be not objectionable may also be interpreted as "not cause a degradation in flying quality Level." For example, control motions alone shall not degrade a Level 1 aircraft to Level 2.

Flight testing performed to demonstrate compliance with control force per g (5.2.2.1.8.1) shall include measurements of δ_s/n. For a discussion of flight test techniques, see 5.2.2.1.8.1 Verification. When the force gradient, gearing, or both are nonlinear, the critical deflection should be found. Qualitatively, pilot comments relating to control forces which are initially too light even though $(F_s/n)_{min}$ and ζ_{sp} are Level 1 provide a clue to the fact that this requirement is being violated.

TABLE XLI. Experimental test points used in sidearm controller evaluations.

Phase I: Air-to-Air Test Points
(13,000 ft MSL, 300 KIAS)

Configuration	ω_{sp} (rad/sec)	Longitudinal Force/Deflection Gradient (lbs/deg)	Longitudinal Force/Response Gradient (lbs/g)	Lateral Force/Deflection Gradient (lbs/deg)
A (Baseline)	5.6	1.7	8	1.8
B	5.6	3.3	8	1.8
C	5.6	1.1	8	1.8
D	2.6	3.3	5	0.95
E	2.6	1.7	5	0.95
F	2.6	1.1	5	0.95
G	1.8	3.3	5	0.95
H	1.8	1.7	5	0.95
I	1.8	1.1	5	0.95

Phase II: Air-to-Air Test Points
(13,000 ft MSL, 300 KIAS)

Configuration	ω_{sp} (rad/sec)	Longitudinal Force/Deflection Gradient (lbs/deg)	Longitudinal Force/Response Gradient (lbs/g)	Lateral Force/Deflection Gradient (lbs/deg)
A	5.6	1.7	8	1.8
B	5	3.3	7	1.8
C	5	1.7	7	1.8
D	5	1.1	7	1.8
E	3.5	3.3	5	1.8
F	3.5	1.7	5	1.8
G	3.5	1.1	5	1.8
H	2.6	1.7	5	0.95

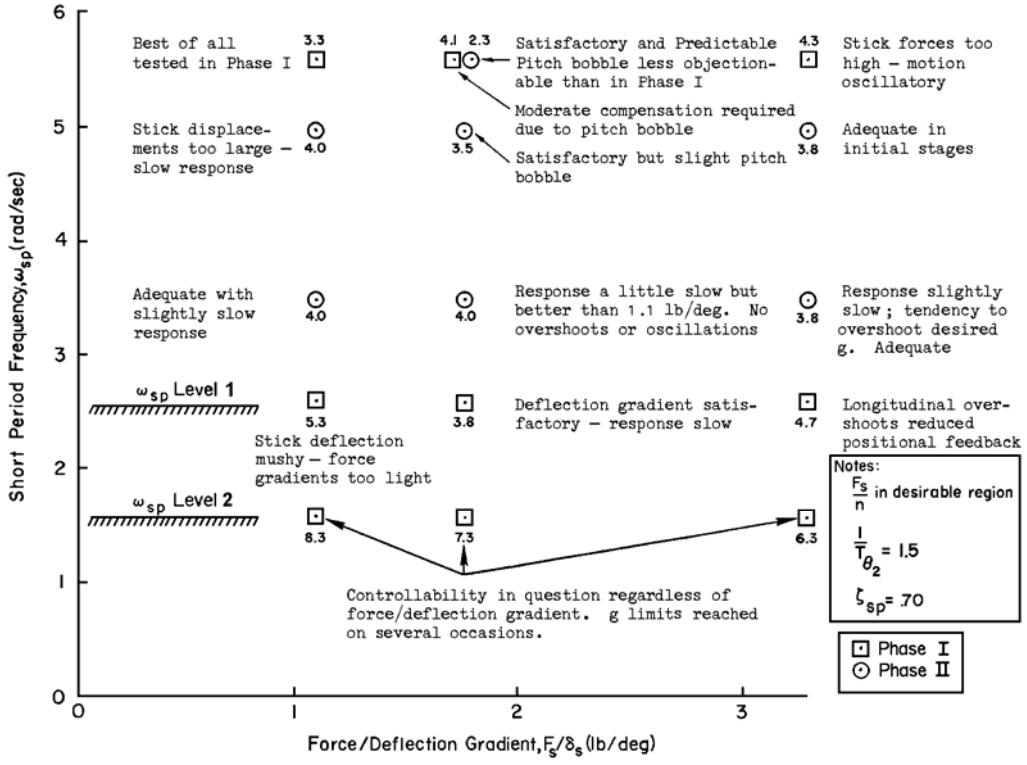


FIGURE 77. Average pilot ratings for gross acquisition task.

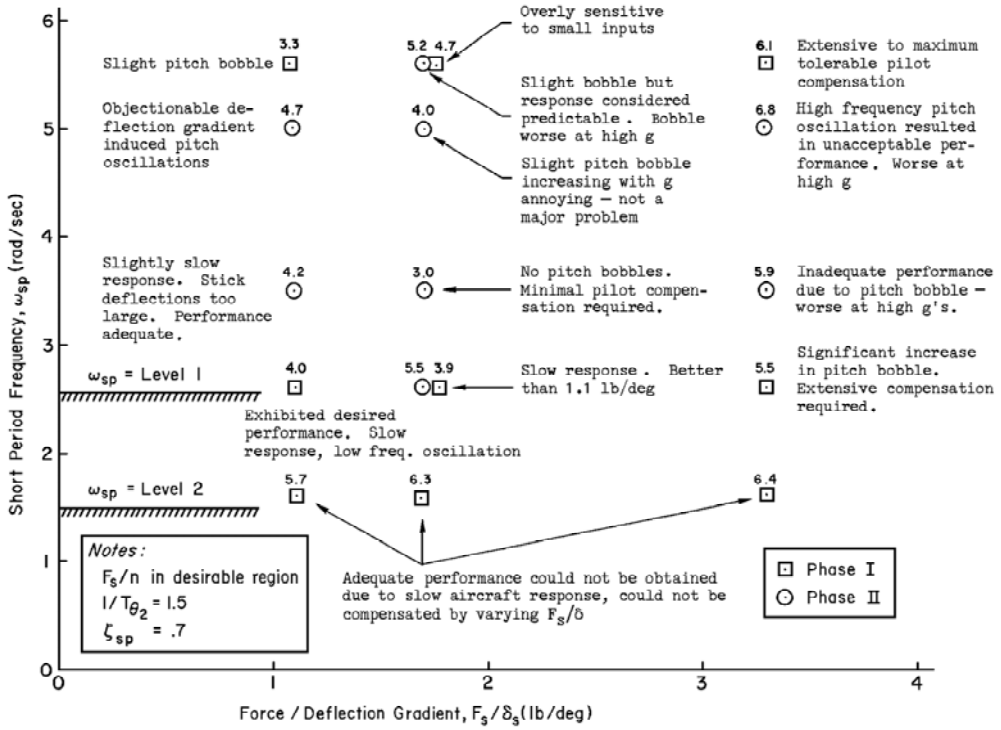


FIGURE 78. Average pilot ratings for fine tracking task.

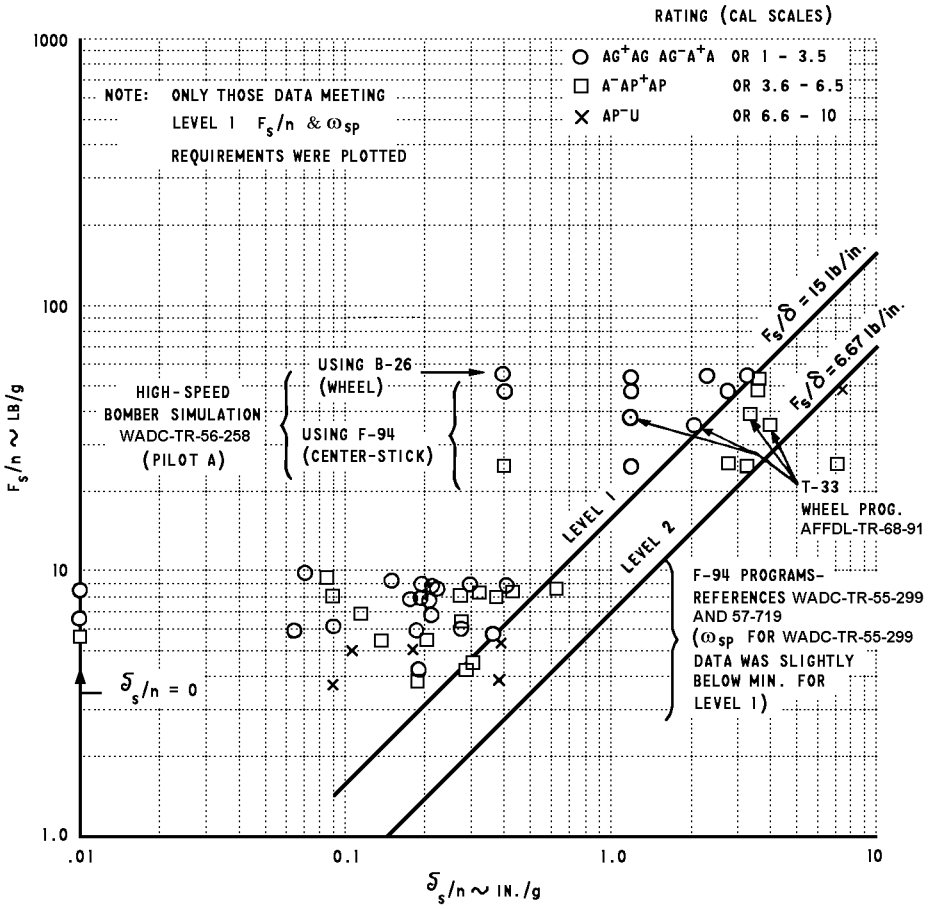


FIGURE 79. Control force per control displacement, Category A Flight Phases—centerstick (from AFFDL-TR-69-72).[1]

[1] The F-94 data used an early pilot rating scale. The letters stand for Acceptable Good Plus, Acceptable Good, Acceptable Good Minus, Acceptable Plus, Acceptable, Acceptable Minus, Acceptable Poor Plus, Acceptable Poor, Acceptable Poor Minus, and Unacceptable.

5.2.2.1.8.5 Pitch controller breakout forces.
Pitch controller breakout forces (the cockpit control force required to affect movement of the control surface in flight), including friction, preload, etc., shall be within the limits of table XLII.

TABLE XLII. Allowable pitch controller breakout forces.

Level	Minimum (lb)	Maximum (lb)
1		
2		
3		

REQUIREMENT RATIONALE

These values refer to the cockpit control force required to start movement of the control surface. Some small effort should be necessary to deflect a control out of its neutral position. This, in combination with centering, serves as the major initial pilot cue of control motion.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.5.2.1
MIL-STD-1797: 4.2.8.5.

Recommended values for table XLII:

Type of Controller	Level	Classes I, II-C, and IV		Classes II-L and III	
		Minimum (lb)	Maximum (lbs)	Minimum (lb)	Maximum (lbs)
Centerstick	1 and 2	0.5	3	0.5	5
	3		6		10
Wheel	1 and 2	0.5	4	0.5	7
	3		8		14
Sidestick	1 and 2	0.5	1	0.5	1
	3		2		2

REQUIREMENT LESSONS LEARNED

Several studies (NADC-ED-6282 and AFFDL-TR-72-141) have substantiated the recommended upper limits on breakout forces in Requirement Guidance. Design towards the lower limits is recommended, since operation with large breakout forces could contribute to pilot fatigue.

VERIFICATION

Measurement of breakout forces in the aircraft on the ground will ordinarily suffice in lieu of actual flight measurement, provided that qualitative agreement between ground measurement and flight observation can be established. Factors such as temperature and vibration could affect breakout forces. Flight control system failures shall also be considered.

5.2.2.1.8.6 Longitudinal control force and travel in takeoff.
With the trim setting optional but fixed, the pitch control forces required during all types of takeoffs for which the air vehicle is designed (including short-field takeoffs and assisted takeoffs, such as catapult or rocket-augmented, if applicable) shall not exceed __(1)__ in push nor __(2)__ in pull. The maximum pitch control displacements during these takeoffs shall not exceed __(3)__ of the total available travel in either direction. Takeoff power shall be maintained until $V_{max}(TO)$ is reached, with the landing gear and high-lift devices retracted in the normal manner at speeds from $V_{o_{min}}(TO)$ to $V_{max}(TO)$.

REQUIREMENT RATIONALE

Limits on maximum push and pull forces required for takeoff shall be lower than those allowed for other operations by the F_s/n limits of 5.2.2.1.8.1. This is to assure that the pitch control input for takeoff need not be abrupt or require excessive effort or two-handed operation. In addition, this requirement insures that an excessive amount of cockpit-control movement does not add to the pilot's workload during this critical period. It also serves to help assure that there is reserve pitch control power during takeoff to allow regulation against gusts or pilot abuses.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.3.3.2
MIL-STD-1797: 4.2.8.6.1 and 4.2.9.1.

Recommended values for Blanks (1) and (2):

Type of Controller	Type of Landing Gear	Aircraft Class	Control Force Limits (lbs)	
			Push	Pull
Centerstick and Wheel Controllers	Nose-wheel and Bicycle-gear	Classes I and IV-C	10	20
		Class II-C and IV-L	10	30
		Class II-L and III	20	50
	Tail-wheel	Classes I, II-C, and IV	20	10
		Classes II-L and III	35	15
Sidestick Controllers	All	All	The forces shall not be objectionable to the pilot.	

Recommended value for Blank (3) for all controller types: 75%

The term takeoff includes the ground run, rotation, and liftoff, the ensuing acceleration to $V_{max}(TO)$, and the transient caused by assist cessation. Takeoff encompasses operation both in and out of ground effect. The procuring activity may wish to include additional procedures, such as for fuel conservation or noise abatement.

REQUIREMENT LESSONS LEARNED

The force limits for this paragraph are intended to assure adequate control force characteristics at takeoff. Pitch forces in takeoff should not normally place great demands on the pilot, either in the form of large pull forces (requiring two-handed operation and possibly causing abrupt responses) or large unnatural push forces. The limits are strictest for small (Class I) aircraft and relaxed for large (Class III) aircraft, but 50 lbs may well be too high for the majority of female pilots. For tail-wheel aircraft the push forces required to raise the tail may be larger than pull forces. At this time there are insufficient data to suggest limits for sidestick controllers, but it is clear that acceptable values will be quite small when compared to limits on the centerstick.

For comparison, FAR Part 23 lists 60 lbs as the maximum stick force for temporary application. There are no known systematic flight tests, but these requirements are an outgrowth of service experience. Little information can be found for existing flight vehicles. AFFDL-TR-72-141, a validation of MIL-F-8785B, using a Class III-L airplane (P-3B), indicates pilot support for the limits for Class III, nose-wheel-equipped aircraft.

This requirement insures that under no condition will it be necessary to use the full range of nose-up and nose-down control to meet the operational takeoff performance requirements. An obvious motivation for this is to provide adequate control margin during takeoff for regulation against external disturbances or abuses of the aircraft. It should be noted that the use of full nose-up pitch control during the early phases of the takeoff roll is usually necessary to raise the nosewheel on soft fields. For a conventional centerstick or control column, having to reverse control travel fully would demand additional concentration at a time when assuring safety requires full pilot attention.

The T-46 main gear location with respect to the c.g. resulted in an overrotation tendency at nosewheel lift-off, while moving the main gear aft would have required more pitch control power. At forward c.g., the pilot had to push forward a great deal in order to avoid a tail strike.

A related requirement in 5.2.2.1.7.5 is intended to provide adequate control of any overrotation tendency without complicated control manipulation.

VERIFICATION

Analysis and simulation may be used for early guidance, but these will only be as good as initial estimates of elevator control power and aircraft performance characteristics. Because of the well-recognized problems in modeling aerodynamics in ground effect, simulation results generally will not be adequate for demonstrating compliance with this requirement. The final proof of compliance shall be flight testing. Flight testing performed to demonstrate compliance shall include measurement of control displacements as well as control forces.

Verification shall include not only normal takeoffs, but demonstration of performance guarantees and any takeoffs which may be peculiar to the aircraft mission requirements. Tests shall be conducted at the conditions for most forward and most aft c.g. positions and will cover the velocity range from 0 to V_{max}(TO). Forward c.g. is critical for initiating the takeoff attitude, while aft c.g. will accentuate any overrotation tendency that requires a control reversal. Control effectiveness varies directly with dynamic pressure, so that effectiveness is minimum at low gross weight (low takeoff speed) and maximum at high gross weight. Light weight will give more rapid acceleration and lower takeoff speed, which may be factors pro or con (refer to the 5.2.2.1.7.5 discussions).

5.2.2.1.8.7 Longitudinal control force limits in dives within the ROTH.
With the air vehicle trimmed for level flight at speeds throughout the ROTH, the control forces in dives to all attainable speeds within the ROTH shall not exceed __(1)__ in push nor __(2)__ in pull. In similar dives, but with use of trim optional following the dive entry, it shall be possible with normal piloting techniques to maintain the forces within __(3)__ push and __(4)__ pull. The force gradients during recovery from these dives shall be within the limits of table XXXVIII, although speed may vary during the pullout.

REQUIREMENT RATIONALE

As a frequently-used but peculiar flight operation, the dive is subject to separate requirements to keep control forces within reason.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.3.5
MIL-STD-1797: 4.2.8.6.3.

Recommended values for Blanks (1) and (2):

Type of Controller	Force Limits	
	Push	Pull
Centerstick	50	10
Wheel	75	15
Sidestick	The forces must be acceptable to the pilot.	

Recommended values for Blanks (3) and (4):

Type of Controller	Force Limits	
	Push	Pull
Centerstick	10	10
Wheel*	20	20
Sidestick	The forces must be acceptable to the pilot.	

* If operation of the trim system requires removal of one hand from the wheel controller, the force limits shall be the same as for the centerstick controller.

REQUIREMENT LESSONS LEARNED

Operation of manual trim systems implies that the dive can be a sustained (rather than momentary) maneuver, so the trim should be effective in substantially reducing pitch forces. The force limit without retrimming effectively limits the gradient of control force with speed while maneuvering throughout the ROSH and ROTH, and further provides a measure of protection against trim failures in dives. Some of the limits may be too much for the average female pilot.

VERIFICATION

Analysis may be used for initial verification, but compliance must ultimately be proven through flight test. The analyses and tests shall be conducted over the following range of Air Vehicle States and flight conditions: a center of gravity range from most forward (combined with heaviest aircraft weight) to most aft (combined with lightest aircraft weight), altitudes from 2000 ft above MSL to the maximum service altitude, and a speed range from minimum to maximum service speeds.

5.2.2.1.8.8 Longitudinal control force limits in dives within the RORH.
With the air vehicle trimmed for level flight at V_{MAT}, but with use of trim optional in the dive, it shall be possible to maintain the longitudinal control forces within the limits of __(1)__ in push and __(2)__ in pull in dives to all attainable speeds within the RORH. The force required for recovery from these dives shall not exceed __(3)__ push and __(4)__ pull. Trim and deceleration devices, etc., may be used to assist in recovery if no unusual pilot technique is required.

REQUIREMENT RATIONALE

Outside the ROTH, flight safety requires some force limit for control and recovery throughout the RORH.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.3.6
MIL-STD-1797: 4.2.8.6.3.

Recommended values for Blanks (1) and (2):

Type of Controller	Force Limits	
	Push	Pull
Centerstick	50	35
Wheel	50	35
Sidestick	The forces must be acceptable to the pilot.	

Recommended values for Blanks (3) and (4):

Type of Controller	Force Limits	
	Push	Pull
Centerstick	120	120
Wheel	120	120
Sidestick	The forces must be acceptable to the pilot.	

REQUIREMENT LESSONS LEARNED

Besides limiting the permissible amount of static stability, this requirement serves to limit instabilities such as transonic tuck in the ROHR, outside the ROTH, where most of the other flying qualities requirements do not apply. See maximum service speed (4.1.4.2) and maximum permissible speed (4.1.4.3). For trim system stalling, see 5.2.9.1.3.

The limited dive testing of AFFDL-TR-72-141, using a Class III (P-3B), supports the one-handed wheel push force limit of 50 lb.

VERIFICATION

Analysis may be used for initial verification, but compliance must ultimately be proven through flight test. The analyses and tests shall be conducted over the following range of Air Vehicle States and flight conditions: a center of gravity range from most forward (combined with heaviest aircraft weight) to most aft (combined with lightest aircraft weight), altitudes over the range of the ROHR, and over a speed range from V_{MAT} to the maximum permissible speed.

5.2.2.2 Longitudinal response to the designated flight path controller.
When a designated flight path controller (other than the pitch controller) is used as a primary flight path controller, the air vehicle response shall meet the requirements of 5.2.2.2.1 through 5.2.2.2.3.

REQUIREMENT RATIONALE

This set of requirements, oriented toward STOL aircraft, defines the effectiveness and cockpit characteristics of the designated flight path controller. A designated flight path controller might be a throttle. These requirements are grouped together because of their similar purposes.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.3.2 through 4.3.4.

REQUIREMENT LESSONS LEARNED

These requirements are applicable only to aircraft equipped with a designated flight path controller other than the pitch controller. The form of the flight path control is irrelevant; STOL designs have used spoilers, flaps, nozzle vectoring, and throttles to provide flight path control. Throughout these requirements the controller will often be described as “throttle” for convenience, since “designated flight path controller” is unwieldy. The use of the term “throttle” to represent the designated flight path controller should not be construed to indicate any preconceptions as far as specific design.

It would be expected that a designated flight path controller will be required for most powered-lift aircraft because: 1) a significant component of the thrust vector is vertical, and/or 2) the aircraft operates well on the backside of the power-required curve.

In some cases the contractor may choose to augment the STOL aircraft so that it has frontside path control characteristics. Such augmentation requires a feedback of airspeed to some auxiliary force producer in the X direction. For example, the Boeing version of the AMST utilized the Coanda flap. The requirements of 5.2.2.1.3.3 would be appropriate for such an aircraft.

VERIFICATION

Verification shall be via verification of requirements 5.2.2.2.1 through 5.2.2.2.3.

5.2.2.2.1 Flight path response to designated flight path controller.
For Level 1, the effective flight path rise time, $t_{R_{\gamma_T}}$, and the flight path overshoot ratio, $\Delta\gamma_{max}/\Delta\gamma_{ss}$, following a step change in the designated flight path controller, shall be within __(1)__ for the indicated Flight Phases.

REQUIREMENT RATIONALE

For STOL aircraft, this is the primary requirement to assure good short-term flight path control. These aircraft operate well on the backside of the power-required curve and therefore use a designated controller other than the pitch controller (such as throttle) to control flight path.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.3.3.1.

Recommended value for Blank (1): "the Level 1 boundaries of figure 80"

Aircraft which fall outside the Level 1 boundaries on figure 80 shall be required to have Level 1 flight path response to attitude changes; i.e., they shall meet the Level 1 requirements of 5.2.2.1.3.3. Any aircraft with inadequate path response to pitch attitude for landing approach or for flare (i.e., does not meet the Level 1 requirement of 5.2.2.1.3.3) shall meet this requirement.

REQUIREMENT LESSONS LEARNED

There was an intense flurry of activity on STOL handling qualities and airworthiness in the 1970's. Most of this was focused towards civil STOL transports. In the mid-1980's a second, smaller effort was undertaken in parallel with the initial ground work for the F-15 S/MTD

aircraft. The recommended requirement was originally developed in the early STOL activity, then refined to its present state in the later work. Supporting data can be found in AFWAL-TR-83-3059 and AFWAL-TR-86-3050.

There are insufficient data to define the boundary between Level 2 and Level 3. Separate criterion boundaries are specified for Landing and Approach on figure 80. Aircraft with flight-path-to-throttle characteristics which meet the Approach boundaries, but not the Landing criterion, are expected to meet the Level 1 criteria of 5.2.2.1.3.3.

The most important short-term requirements for the designated flight path controller are rapidity of response and effectiveness in changing the flight path. Rapidity is defined here in terms of flight path rise time, $t_{R_{\gamma_T}}$, and flight path overshoot ratio, $\Delta\gamma_{max}/\Delta\gamma_{ss}$, determines how well the commanded flight path change stabilizes in the short term. Figure 81 illustrates how $t_{R_{\gamma_T}}$ and $\Delta\gamma_{max}/\Delta\gamma_{ss}$ are defined. Note that $t_{R_{\gamma_T}}$ is identical to the parameter $t_{0.5\Delta\gamma_{max}}$ from NASA

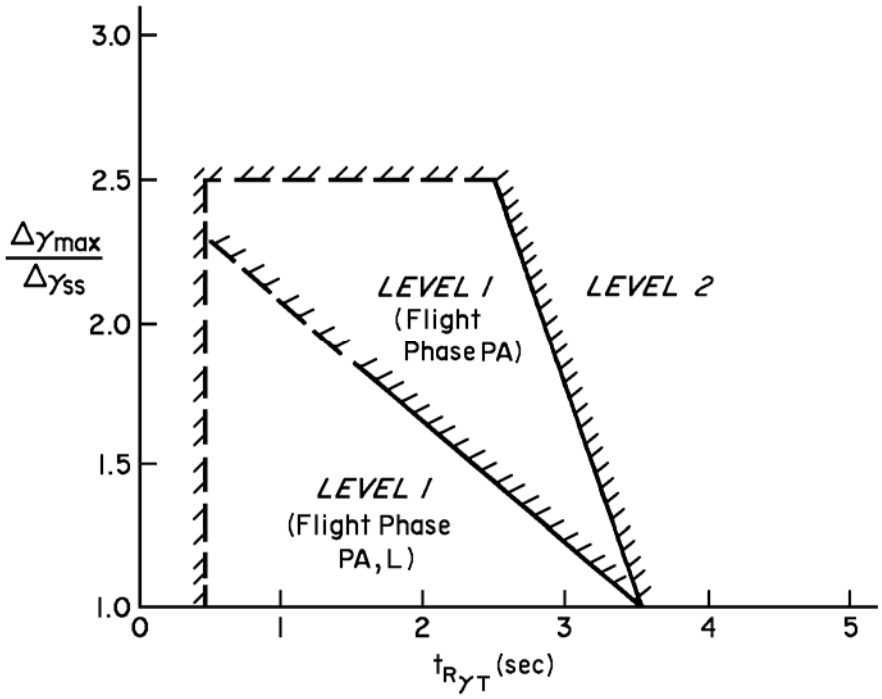
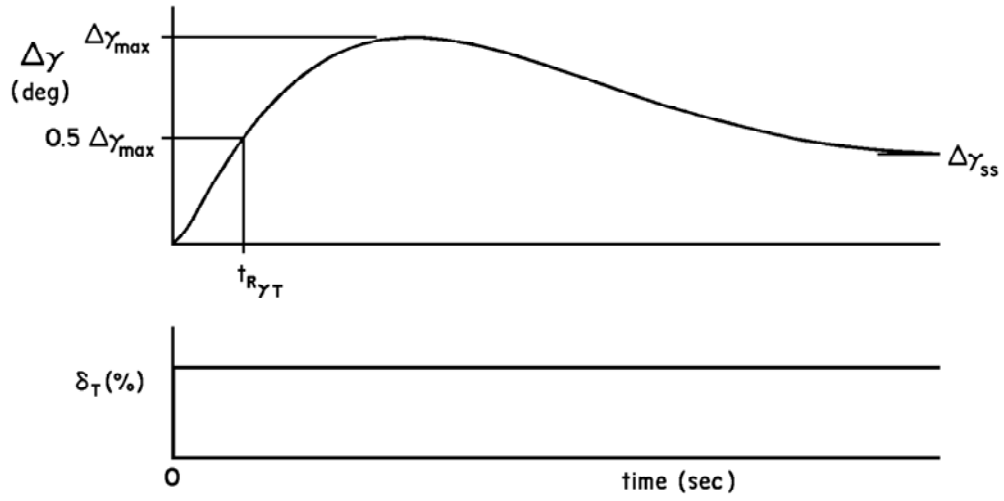


FIGURE 80. Requirement for short-term flight path response to step input of flight path controller.

TP-1911, and that it is related to the bandwidth of h/δ_T (normal pitch SAS on) as defined on figure 82. Figure 83 shows the relationship between $\omega_{BW_{hT}}$ and $t_{R_{\gamma_T}}$ for the data of FAA-RD-75-123 and FAA-RD-75-197. This figure may be used to convert the figure 80 requirement to $\omega_{BW_{hT}}$ versus $\Delta\gamma_{max}/\Delta\gamma_{ss}$, if desired.

The scatter in the available data base (discussed in AFWAL-TR-83-3059) makes it difficult to accurately define response limits. Therefore, the boundaries of figure 80 are very tentative and shall be investigated through further analysis and testing. Additionally, a boundary between Levels 2 and 3 is needed on figure 80.

The limits of figure 80 reflect pilot acceptance of less precise flight path control (i.e., more overshoot) during the approach than for flare and landing. For flare, large path overshoots generally lead to high workload and touchdown dispersions. The dashed lines on the Level 1 boundaries reflect uncertainty (primarily due to a lack of data) in setting a lower limit on $t_{R_{\gamma_T}}$. It is certain that the excessive abruptness consistent with $t_{R_{\gamma_T}}$ near zero would be unacceptable to the pilot. However, the lower limit on $t_{R_{\gamma_T}}$ on figure 80 is not based on any existing data and shall be the subject of piloted simulation or flight test experimentation.



Note: Pitch attitude controller is free during response

FIGURE 81. Definition of γ/δ_T time response parameters (pitch SAS active).

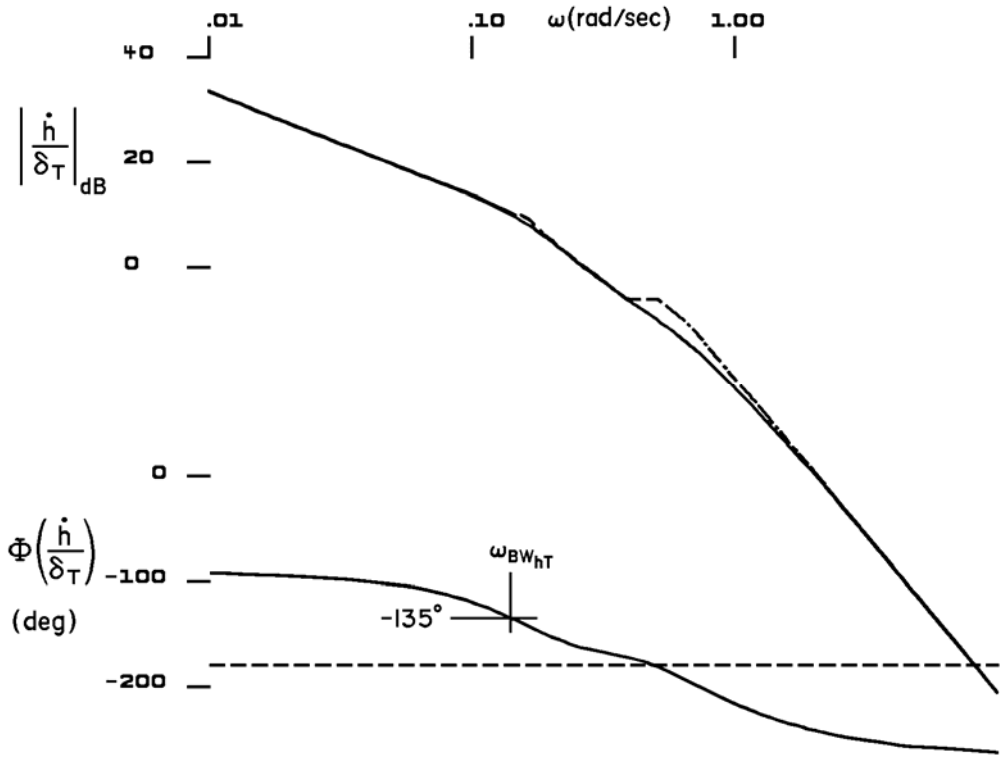


FIGURE 82. Definition of $\dot{h}/\delta_T$ frequency response parameter (pitch SAS active).

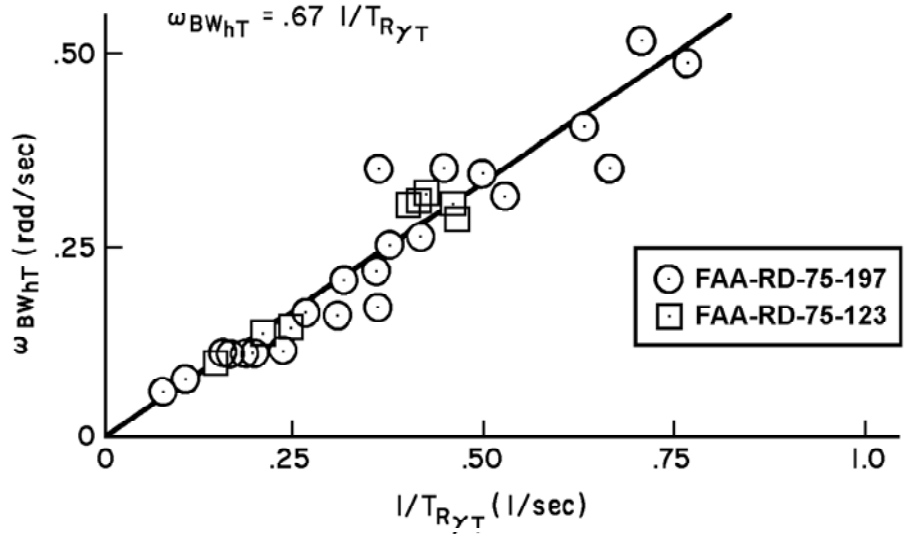


FIGURE 83. Relationship between throttle bandwidth and rise time for typical powered-lift STOL aircraft.

VERIFICATION

Verification shall be by analysis, simulation, and flight test.

5.2.2.2.2 Flight path control power.
The designated flight path controller shall be capable of producing the steady-state flight path angle changes given in table XLIII following full actuation of the controller. It shall be possible to achieve these flight path angle changes without changing the trim airspeed for the approach flight condition and without reconfiguring the air vehicle. Any airspeed bleedoff associated with the $\Delta\gamma_{ss_{up}}$ required for flare and landing shall not be excessive.

TABLE XLIII. Minimum flight path control power.

Flight Phase	Level	Minimum flight path angle changes (measured from γ_{trim})	
		$\Delta\gamma_{ss_{up}}$	$\Delta\gamma_{ss_{down}}$
PA and CA	1		
	2		
	3		
L and CL	1		
	2		
	3		

REQUIREMENT RATIONALE

Powered-lift STOL aircraft typically are quite limited in terms of flight path control power in the power approach (PA) configuration. This requirement is intended to insure adequate authority for glide slope tracking and landing, rather than a discrete event such as go-around. Adequate control power does not tend to be a problem for such discrete events when the pilot has time to reconfigure the aircraft.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.3.3.1.

Recommend values for table XLIII:

Flight Phase	Level	Minimum Flight Path Angle Changes (measured from γ_{trim})	
		$\Delta\gamma_{ss_{up}}$	$\Delta\gamma_{ss_{down}}$
PA and CA	1	4°	-4°
	2	2°	-2°
	3	2°	-2°
L and CL	1	6.5° or $\Delta\gamma_{ss}$ which gives γ = 1.5°, whichever is greater	-4°
	2	4° or $\Delta\gamma_{ss}$ which gives γ = -1°, whichever is greater	-2°
	3	4° or $\Delta\gamma_{ss}$ which gives γ = -1°, whichever is greater	-2°

REQUIREMENT LESSONS LEARNED

For most current STOL designs, flight path is primarily controlled with throttle. For such cases, this requirement applies directly to the limits of travel for the thrust controller. For configurations which are augmented so that flight path is controlled exclusively with attitude (such as the Boeing YC-14), the requirements of this section apply except that the limits apply to attitude control rather than throttle. The use of a separate auxiliary cockpit controller (such as spoilers) is considered to be a way of reconfiguring the air vehicle and therefore does not apply.

The C-X specification required the following flight path change capability for maximum-effort landing:

Speed	$\Delta\gamma_{ss}$ (deg), All Engines Operating
$V_{o_{min}}$	Level flight and -4.0
$V_{o_{min}}$ ± 5 kt	±2.0

The recommended values for $\Delta\gamma_{ss}$ are based on moving-base simulations (FAA-RD-74-143, FAA-RD-74-179, FAA-RD-75-197). The data for these simulations are reviewed below. Values for $\Delta\gamma_{ss}$ must include consideration of control in turbulence and ability to counter horizontal windshears.

A requirement on $\Delta\gamma_{ss}$, as opposed to $\Delta\dot{h}$ or Δn_z, has the advantage of being independent of speed (since $\Delta\gamma_{ss} = \Delta\dot{h}/U_0$). Both the Air Force AMST specification (ASD-TR-78-13) and the AGARD V/STOL document (AGARD Report No. 577) set requirements on minimum

incremental normal acceleration—the former at ±0.1 g for all STOL operations, the latter at +0.2 g for flare. For typical STOL approach speeds and flight path angles, these values are reasonably consistent with the table XLIII requirement to achieve $\gamma = 1.5^{\circ}$. This can be illustrated by assuming the exponential flare model in AFWAL-TR-83-3059. For ideal path control:

$$\dot{h} = \dot{h}_0 \, e^{-t/T_F}$$

where T_F is the flare-mode time constant defined in AFWAL-TR-83-3059 and $\dot{h}_0$ is the sink rate at flare initiation. For an ideal path-mode response ($t_{R_{\gamma_T}}$ very small) the peak normal acceleration is:

$$n_{z_{peak}} = \frac{1}{g} \frac{\dot{h}_0}{T_F}$$

For typical STOL flares T_F varies between 3 and 5 sec. If it is assumed $\dot{h}_0$ is approximately 18 ft/sec to account for initial sink rate plus the required 1.5° up capability (e.g., if $\gamma_0 = -6^{\circ}$ and $V_{ref} = 85$ kts) then $n_{z_{peak}}$ varies from 0.11 g to 0.19 g for values of T_F of 5 and 3 sec, respectively. This is consistent with the 0.10 to 0.20 g requirements specified in ASD-TR-78-13 and AGARD Report No. 577.

Large discrete horizontal windshears near touchdown represent a limiting condition on flight path control power. From the results of NASA CR-152064 it is found that unless $(1/T_{\theta_2})_{eff}$ is quite large (greater than about 0.9 rad/sec), a powered-lift STOL cannot safely negotiate horizontal windshears which correspond to the aircraft performance limits; i.e., $\dot{V}_w = g\,\gamma_{max}$ or $g\,\gamma_{min}$. It follows that to insure the capability for negotiating a design windshear, one can either augment $(1/T_{\theta_2})_{eff}$ or provide excess control power over the design windshear. If it is unpermissible for a powered-lift STOL to be more vulnerable to windshear than the low-wing-loading STOL (Twin Otter), the required flight path control power can be approximated from figure 84 and figure 85 with the following result:

$$\gamma_{max} > 1.5^{\circ}$$

$$\Delta\gamma_{min} < -6.5^{\circ} .$$

These values were obtained by noting the values of γ_{max} where the lines for ISO-accident potential rating of 4 become asymptotic, and subtracting 6° (nominal approach angle) from this value.

The table XLIII γ_{max} requirement for landing ($\gamma = 1.5^{o}$) is consistent with the windshear requirement. However, the $\Delta\gamma_{min}$ requirement is substantially greater than the recommended -4º requirement in table XLIII, a result of the fact that the FAA-RD-75-197 experiment upon which table XLIII is based did not include large discrete decreasing tailwind windshears.

The possibility of relaxing the flight path control power as a function of $(1/T_{\theta_2})_{eff}$ shall be considered in future handling quality experiments. For the present, the question of whether or not the $\Delta\gamma_{min}$ requirement in table XLIII shall be made more stringent based on the above-noted windshear considerations must be addressed when developing the final requirements.

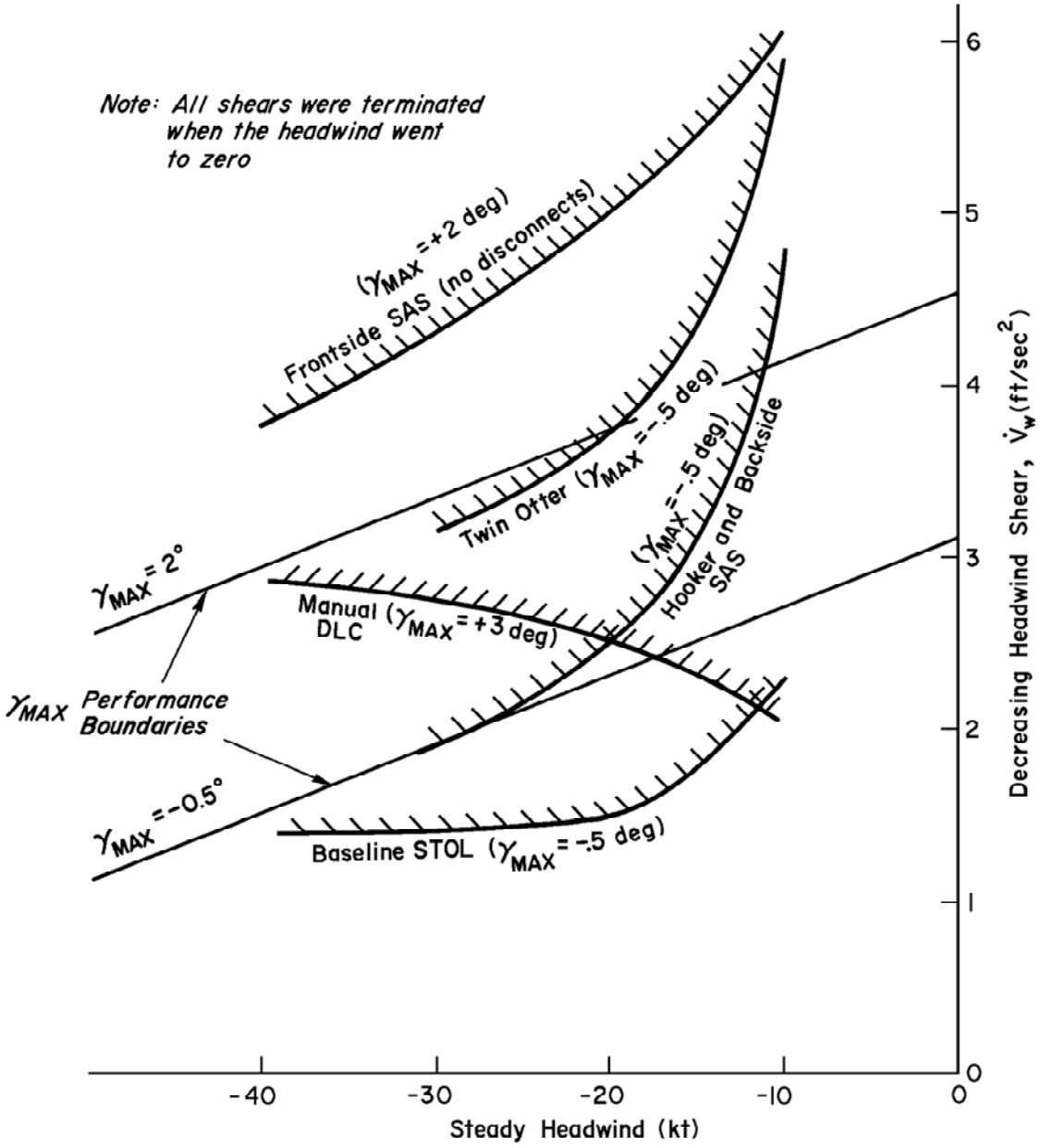


FIGURE 84. Approximate boundaries where accident potential ratings equal 4 for decreasing headwind shears (from NASA CR-152064).

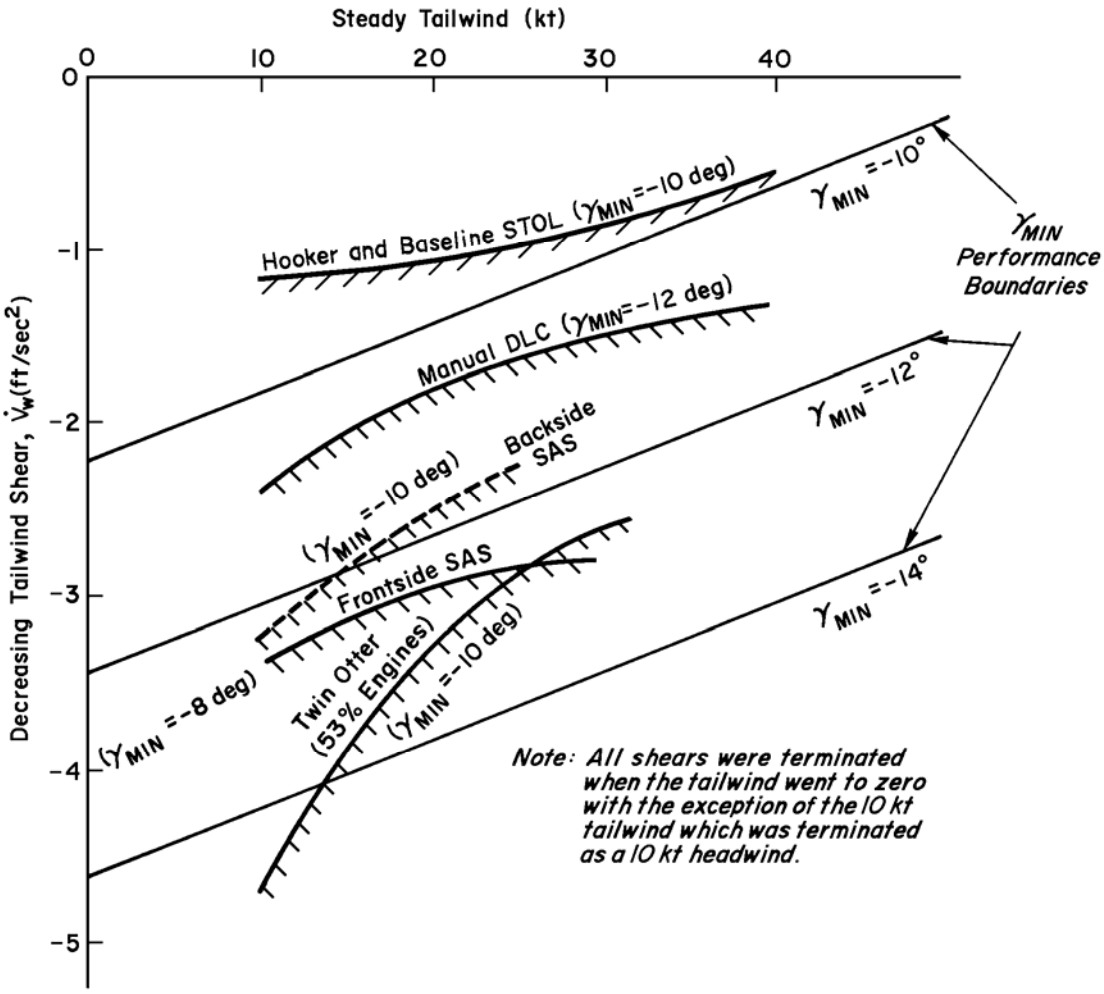


FIGURE 85. Approximate boundaries where accident potential ratings equal 4 for decreasing tailwind shears (from NASA CR-152064).

The primary source of data for developing the table XLIII requirements was a flight path margin study in FAA-RD-75-197. A series of approaches and landings was flown on the NASA Ames FSAA simulator. Tasks covered VFR approaches and ILS approaches to break out on a 6º glideslope, and flare and landing. Aircraft coupling, engine response time, and turbulence conditions were varied. Winds include calm air, headwinds, tailwinds, and crosswinds, and some shears. Turbulence levels were $\sigma_{u_g} = 0$ and 4.5 ft/sec (Uncommon). Both up- and down-γ capability were limited systematically. The pilots were told to accomplish all flight path control, including flare, with the throttle in order to avoid the "contaminating" effects of $(1/T_{\theta_2})_{eff}$ noted above.

Figure 86 shows the results of the flight path margin investigation. Cooper-Harper pilot ratings for two pilots are shown as a function of maximum $\Delta\gamma_{ss}$ available. The pilots rated the aircraft after flying a full profile of ILS tracking to break out visual approach, flare, and landing. The

ratings may therefore be considered composites for the entire task. Figure 86 indicates that pilot ratings did not degrade until maximum $\Delta\gamma_{ss}$ was 4º or less, up or down.

Another simulation experiment, also reported in FAA-RD-75-197, provides additional data. The task and initial conditions were identical to those for figure 86 with the exception that a steady headwind and decreasing headwind shear were added. The wind velocity was 20 kts at altitudes over 200 ft above ground, and varied down to 12 kts at touchdown. Therefore, while the inertial approach angle, $\gamma_{inertial}$, was -6º, the aerodynamic angle, γ_{aero}, varied from -4.4º at 200 ft and above to -5º at touchdown.

For this simulation, pilot ratings are available for three separate mission segments: ILS approach, breakout (at 220-300 ft) to pre-flare (about 35 ft), and flare and landing. However, only upside $\Delta\gamma_{ss}$ was limited. Figure 87 shows the ratings for this simulation.

Ratings for the approach portion (figure 87a and figure 87b) show trends consistent with figure 86a: pilot ratings degrade to Level 2 at around $\Delta\gamma_{ss} = 4º$ for $t_{R_{\gamma_T}}$ near the Level 1 limit of 3.5 sec (see 5.2.2.2.1). The data indicate a possible relaxation in the $\Delta\gamma_{ss}$ requirement for faster path response characteristics (lower $t_{R_{\gamma_T}}$). However this has not been reflected in the recommended requirement in table XLIII because of the limited amount of data available to support this trend. The data on figure 86 and figure 87a and figure 87b indicate $\Delta\gamma_{ss}$ can be decreased to ±2º for Level 2 flying qualities in the approach flight condition (PA).

Considerably more control power is required in the upside direction ($\Delta\gamma_{ss_{up}}$) for flare and landing (L) than is required in power approach (PA), according to the data on figure 87c. This is not surprising since the flare ideally involves a change in flight path angle equal to the inertial approach angle corrected for wind (5º in FAA-RD-75-197). Based on fairing the few available data points on figure 87c, the recommended Level 1 and Level 2 limits are 6.5º and 4º, respectively. For the purpose of formulating the requirement (table XLIII), this has been interpreted as $\gamma = 1.5º$ for Level 1 and $\gamma = -1º$ for Level 2. This Level 1 requirement seems reasonable when compared to practical STOL designs. For example, the YC-15 had a maximum γ of about 4º at a 75-kt approach speed. The downside limits for flare are unchanged from the approach values; i.e., Level 1 is -4º and Level 2 is -2º. It must be emphasized that the table XLIII values are based on fairing a few widely spread data points and that more data are highly desirable to refine these limits.

The FAA, in developing tentative airworthiness criteria for STOLs (FAA-RD-76-100), specified a $\Delta\gamma_{ss}$ of ±4º. Most subject pilots in the FAA STOL simulations generally agreed that "airworthy" corresponded to a Cooper-Harper rating of about 5, i.e., slightly better than Level 3. With this interpretation, the Level 2 limit on $\Delta\gamma_{ss}$ in table XLIII for the PA flight condition would increase from 2º to about 3.5º.

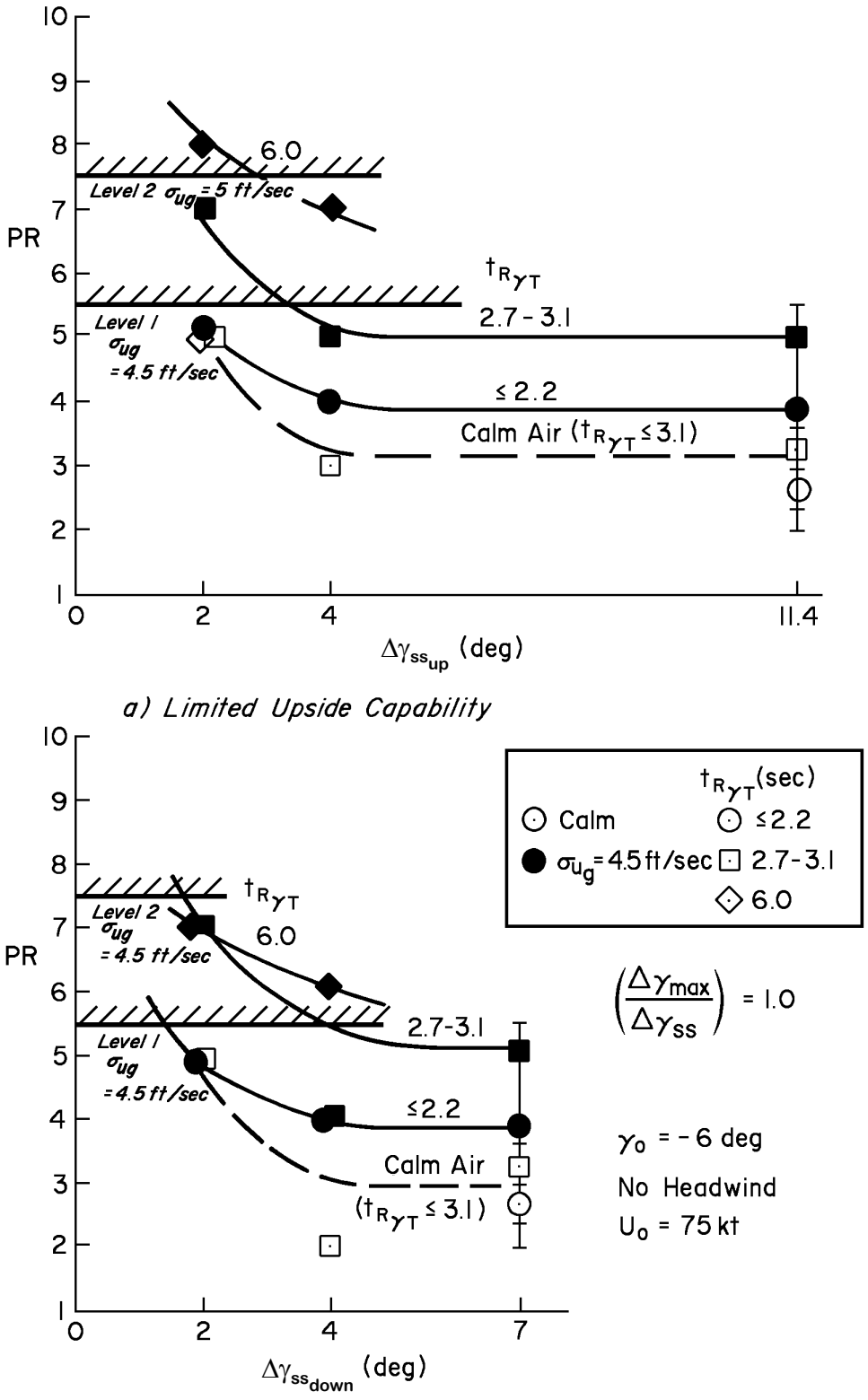


FIGURE 86. Effect of flight path margin on pilot ratings, ILS approach and visual landing, FAA-RD-75-197 (200- and 300-series configurations).

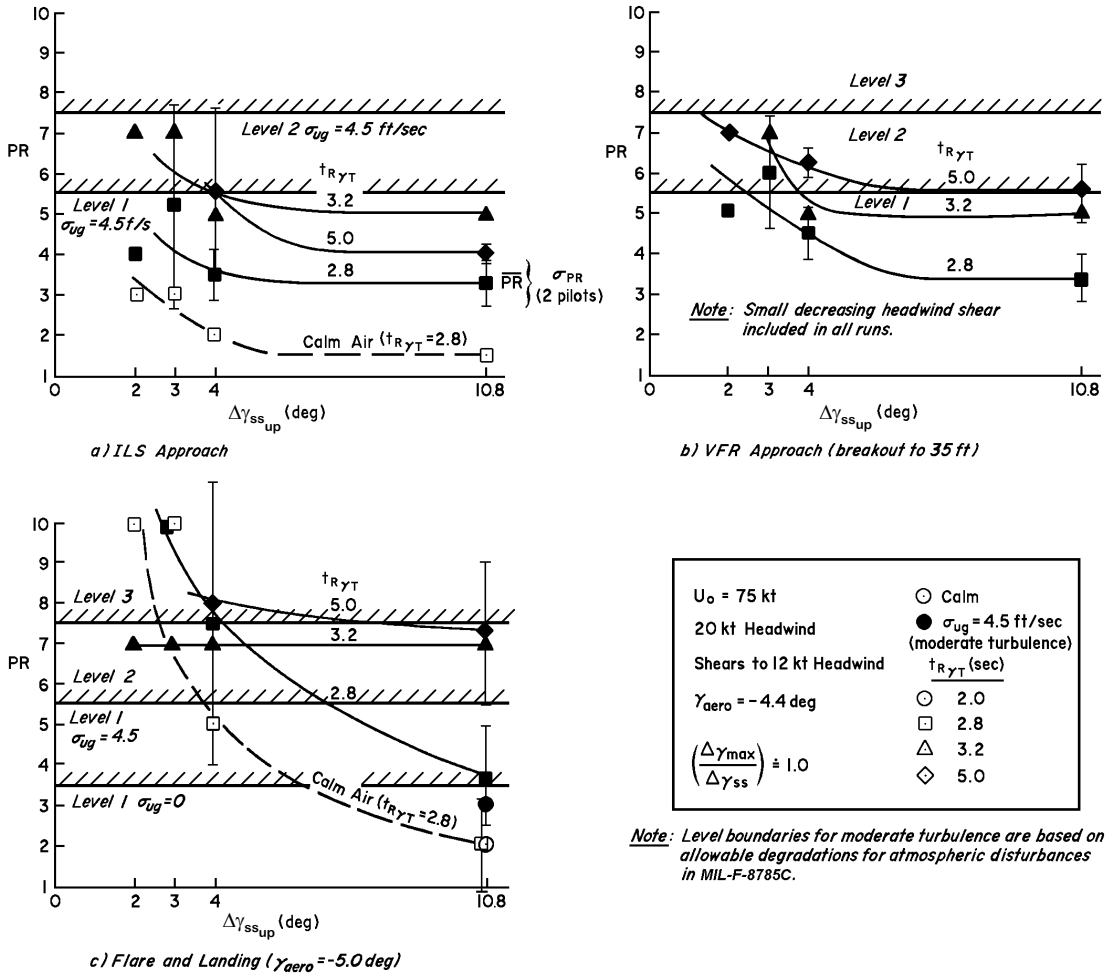


FIGURE 87. Effects of upside flight path margin on pilot ratings of FAA-RD-75-197 (1200-series configurations).

All of the data on figure 86 and figure 87 are based on flight path control with power and the pilots were specifically requested to avoid the use of attitude. Therefore, the data may be somewhat conservative. As discussed at length in AFWAL-TR-83-3059, most powered-lift STOLs must be flown using the backside technique wherein flight path is controlled primarily with power on the approach. However, it is quite common to flare with attitude. It was felt that the path control power required to flare is probably not dependent on whether the flare controller is throttle or attitude and that the figure 87c data are applicable in both cases. The requirement allows a "reasonable" airspeed bleedoff when meeting the $\Delta\gamma_{ss}$ requirements for landing to account for the use of attitude to flare. It should be noted that pure attitude flares require reasonably large values of $(1/T_{\theta_2})_{eff}$ which essentially guarantees meeting the $\Delta\gamma_{ss_{up}}$ requirement for flare and landing.

See AFWAL-TR-83-3059.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall proceed or accompany careful build-up to suspected critical flight conditions.

5.2.2.2.3 Flight path controller characteristics.
At all flight conditions the pilot-applied force and deflection required to maintain a change in flight path shall be in the same sense as those required to initiate the change.

REQUIREMENT RATIONALE

This requirement is intended to assure that control motion and aircraft motion are in harmony.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.3.3.1.

REQUIREMENT LESSONS LEARNED

None.

VERIFICATION

Verification will depend on the characteristics of the particular controller.

5.2.3 Lateral-directional flying qualities.
The lateral-directional responses of the air vehicle shall meet the requirements of 5.2.3.1 through 5.2.3.13.4.

REQUIREMENT RATIONALE

Lateral-directional control of air vehicles is a vital element of flying qualities, both as primary control axes (for example, holding the wings level during landing) and as an indirect way of controlling the aircraft flight path (for example, banking to initiate a turn).

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3 through 3.3.9.5
MIL-STD-1797: 4.5 through 4.6.7.11.

REQUIREMENT LESSONS LEARNED

This section was difficult to organize since, primarily because of coupling between lateral and directional motions, each requirement has implications in many areas of flying qualities. Conversely, each flying qualities area is generally a function of many different parameters. For example, the rolling moment acting on an aircraft following an aileron input is a function of the directional stability, the yaw due to aileron, and the dihedral characteristics of the aircraft in addition to the aileron roll effectiveness characteristics. Directional stability, however, also influences the Dutch roll characteristics and the flying qualities in steady sideslips, with asymmetric thrust, and during takeoff and landing. Thus, to place a requirement dealing with the Dutch roll mode under a section titled "Flying qualities requirements for the yaw axis", as was done in MIL-STD-1797, is misleading since this mode pertains to roll response as well as yaw response.

To compound the problem, the paragraphs in MIL-STD-1797 that dealt with roll and yaw control power and roll and yaw control forces in crosswinds were treated as separate requirements. A similar situation existed with requirements for steady sideslips and for asymmetric thrust. The original intent of these requirements in MIL-F-8785 was that the control power and control force requirements in both roll and yaw be met simultaneously in these situations. Thus, these paragraphs needed to be recombined in this document.

A major reorganization of the lateral-directional section was required. In this reorganization the paragraphs were arranged as much as possible so that like things are grouped together, and headings were chosen to reflect this grouping. The first four requirements (5.2.3.1 Lateral-directional modal characteristics., 5.2.3.2 Lateral-directional dynamic response characteristics., 5.2.3.3 Roll PIO., and 5.2.3.4 Yaw PIO.) deal with complex coupled lateral-directional parameters which cannot be treated independently. The next four requirements (5.2.3.5 Roll control effectiveness., 5.2.3.6 Lateral-directional control with speed changes., 5.2.3.7 Yaw control forces in waveoff (go-around)., and 5.2.3.8 Lateral-directional control forces and displacements.) deal with, hopefully, relatively uncoupled motions or control considerations that can be identified as primarily dealing with either roll or directional control characteristics. The next three requirements (5.2.3.9 Steady sideslips., 5.2.3.10 Lateral-directional control in crosswinds., and 5.2.3.11 Lateral-directional control with asymmetric thrust.) all deal with control under specific operating conditions that have implications to both roll and directional control characteristics. The last two requirements (5.2.3.12 Wings-level turn. and 5.2.3.13 Lateral translation.) deal with direct-force control modes.

The term "roll control" is used in lieu of "lateral control" since lateral motions are literally motions along the y axis. Also, sometimes "lateral control" is used to describe both rolling and yawing commands. On the other hand, the meaning of the term "directional control" is not ambiguous, is accepted practice, and so is used throughout. "Roll control" means simply control of roll. The term "controller" refers to the pilot's input device in the cockpit which commands the vehicle motions, and the term "control surface" refers to the aerodynamic surface which generates the forces and moments.

VERIFICATION

Verification shall be via verification of requirements 5.2.3.1 through 5.2.3.13.4.

5.2.3.1 Lateral-directional modal characteristics.
The lateral-directional modal characteristics (roll mode, spiral mode, Dutch roll, etc.) shall meet the requirements of 5.2.3.1.1 through 5.2.3.1.6.

REQUIREMENT RATIONALE

In order to most clearly and completely take into account the many complex phenomena that are associated with the lateral-directional axes, requirements 5.2.3.1.1 through 5.2.3.1.6 are basically specified in terms of such conventional modal parameters as roll mode time constant, Dutch roll natural frequency, Dutch roll damping ratio, and time to double amplitude for the spiral. The form of the requirements is dictated by the extent of our knowledge of aircraft motions.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3 through 3.3.9.5
MIL-STD-1797: 4.5 through 4.5.1.5 and 4.6.1.1.

REQUIREMENT LESSONS LEARNED

See Requirement Lessons Learned in 5.2.3.1.1 through 5.2.3.1.6.

VERIFICATION

The advent of high-order feedback control systems has required specification of flying qualities parameters for systems of, for example, twentieth order or more. The use of lower-order equivalent systems allows us to extend to many higher-order systems the application of well-established boundaries generated from classical airplane data; i.e., MIL-F-8785C.

Normally, the equivalent lateral-directional system would be taken as the classical three-degree-of-freedom roll response to a stick- or wheel-force input, with the provision for an added time delay. For roll, this is of the form:

$$\frac{\phi(s)}{\delta_{as}(s) \text{ or } F_{as}(s)} = \frac{K_{\phi_e} (s^2 + 2\zeta_{\phi_e} \omega_{\phi_e} s + \omega_{\phi_e}^2) e^{-\tau_{\phi_e} s}}{(s + 1/T_{S_e})(s + 1/T_{R_e})(s^2 + 2\zeta_{d_e} \omega_{d_e} s + \omega_{d_e}^2)}$$

and for sideslip:

$$\frac{\beta(s)}{\delta_{rp}(s) \text{ or } F_{rp}(s)} = \frac{K_{\beta_e}\,(s+1/T_{\beta_{1_e}})(s+1/T_{\beta_{2_e}})\,(s+1/T_{\beta_{3_e}})\,e^{-\tau_{\beta_e}s}}{(s+1/T_{S_e})\,(s+1/T_{R_e})\,(s^2+2\,\zeta_{d_e}\,\omega_{d_e}\,s+\omega_{d_e}^2)}\ .$$

Equivalent system parameter values obtained by matching the responses of such transfer functions to the actual high-order responses would then be used for comparison with modal requirements. The actual high-order responses, however, would be used for comparison with time-domain requirements, such as ϕ_t, p_{osc}/p_{av}, and $\Delta\beta$. The equivalent time delays, τ_{ϕ_e} and τ_{β_e}, are included to represent assorted effects of higher-order modes, lags, and time delays in the flight control system.

In general, the "best" match is expected to be obtained by matching both lateral and directional transfer functions simultaneously, as discussed in AIAA Paper 77-1122. It is also obvious that many configurations, modes, and flight conditions would be amenable to the use of even lower-order transfer functions, such as the conventional approximation of the first-order roll subsidence or second-order Dutch roll. At present, no rules for application are available. One possible approach, however, would be to use the parameters resulting from the best match to justify further reduction in the order of equivalent representation. The order reduction may be complicated by the introduction of additional poles and zeros by the flight control system in the relevant frequency range. The "dominant modes" or simplified representations shall not be used until validated by analysis of the transfer function forms above.

5.2.3.1.1 Roll mode.
The equivalent roll mode time constant, T_{R_e}, shall be no greater than the appropriate value in table XLIV.

TABLE XLIV. Maximum roll mode time constant (seconds).

Flight Phase Category	Level 1	Level 2	Level 3
A			
B			
C			

REQUIREMENT RATIONALE

This requirement is directed at precision of roll control. For aircraft that exhibit classical spiral (large spiral time constant) and Dutch roll (low $|\phi/\beta|_d$) characteristics, the equivalent roll mode time constant, T_{R_e}, describes the aircraft roll damping.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.1.2
MIL-STD-1797: 4.5.1.1.

Recommended values for table XLIV:

Flight Phase Category	Class	Level 1	Level 2	Level 3
A	I and IV	1.0	1.4	10
	II and III	1.4	3.0	10
B	All	1.4	3.0	10
C	I, II-C, and IV	1.0	1.4	10
	II-L and III	1.4	3.0	10

For aircraft with a roll rate response that is not easily approximated by a first-order equation, an alternative specification such as bandwidth may be in order.

REQUIREMENT LESSONS LEARNED

The p/F_{as} response function can be transformed to the time domain. If it is assumed τ_{ϕ_e} is small, the roll rate time history following a step roll control input is given by:

$$\frac{p(t)}{\delta_{a_{step}}} = K_S\, e^{-t/T_S} + K_R\, e^{-t/T_R} + K_d\, e^{-\zeta_d \omega_d t} \cos\left[\omega_d\, t \sqrt{1-\zeta_d^2} + \psi_p\right]$$

For a normal aircraft the roll mode, characterized by the first-order time constant T_R, takes on the form shown on figure 88 for a step aileron input.

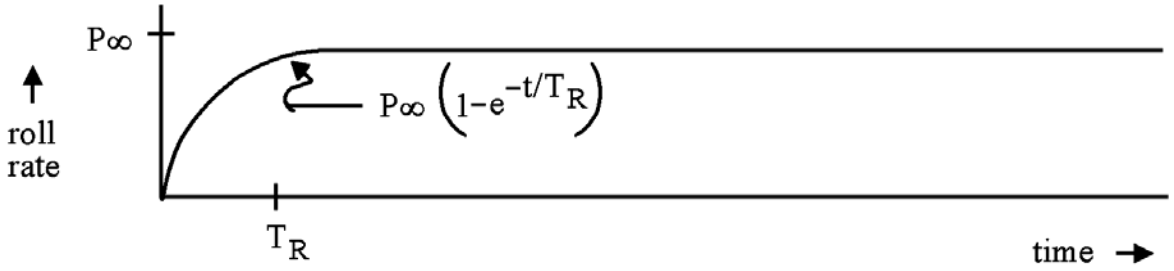


FIGURE 88. Normal aircraft roll rate response following a step aileron input.

where

$$\frac{1}{T_R} \approx \frac{-\rho V g}{4 (W/S) k_x^2} \left[C'_{\ell_p} - \frac{C'_{\ell_\beta}}{C'_{n_\beta}} \left(C'_{n_p} - 2 k_z^2 C_{L_1} \right) \right]$$

$$C'_{\ell_i} = \frac{C_{\ell_i} + C_{n_i} I_{xz} / I_{zz}}{1 - I_{xz}^2 / (I_{xx} I_{zz})}$$

$$C'_{n_i} = \frac{C_{n_i} + C_{\ell_i} I_{xz} / I_{xx}}{1 - I_{xz}^2 / (I_{xx} I_{zz})}$$

$$C_{L_1} = W / (\bar{q} S)$$

$$k_x^2 = I_{xx} / (m b^2)$$

$$k_z^2 = I_{zz} / (m b^2)$$

where i = p, r, β, δ_a, δ_r, etc.

There are considerable data to show that pilot rating is a function of roll damping; for example, figure 89 (from AFFDL-TR-65-138). Roll damping is generally expressed in terms of the first-order roll mode time constant, T_R, of the roll rate response following a step rolling moment input. Therefore, a direct requirement on T_R has been specified. Limits for the other parameters in the ϕ/F_{as} transfer function of 5.2.3.1 are given in 5.2.3.1.2, 5.2.3.1.3, 5.2.3.1.4, and 5.2.3.1.5.

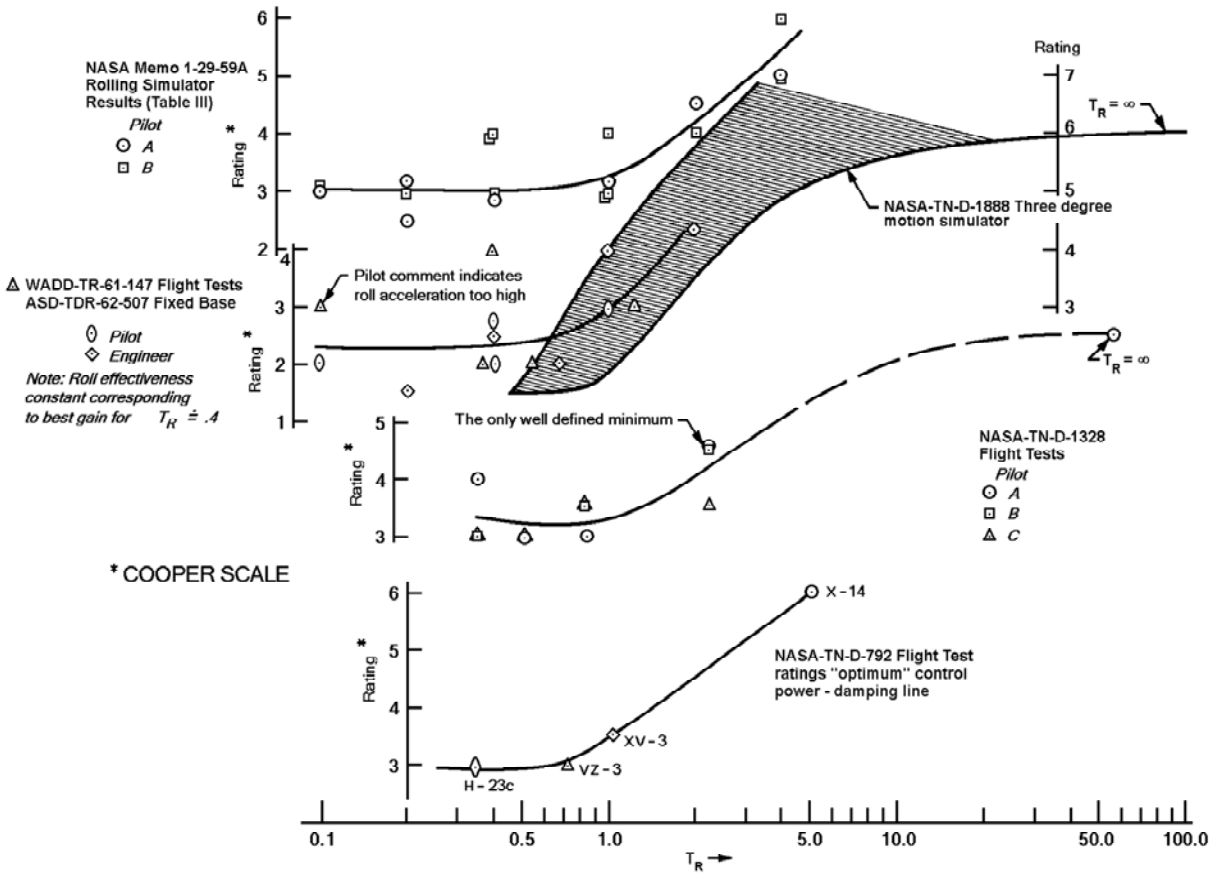


FIGURE 89. Ratings versus roll damping—flight test, moving-base, fixed-base with random input (AFFDL-TR-65-138).

AFFDL-TR-69-72 contains a concise description of data available for development and support of the recommended values for table XLIV. The following discussion is primarily taken from AFFDL-TR-69-72.

Level 1 Requirements

The starting point for specification of the criteria was the recommendation pertaining to roll mode time constant given in AFFDL-TR-65-138 and NASA Memo 1-29-59A. Both report on extensive surveys of roll flying qualities and so are directly applicable to this effort. NASA Memo 1-29-59A proposes a maximum $T_R = 1.3$ sec for Class IV aircraft and $T_R = 1.5$ sec for all other Classes (figure 90). From theoretical considerations and from analysis, AFFDL-TR-65-138 concluded that "The maximum value of T_R considered satisfactory is about 1.3 to 1.5; and there is no strong evidence in existing data or theory for allowing this value to increase with airplane size or mission." While there is still no strong evidence to indicate that the requirements can be relaxed, several reports on in-flight evaluations (NASA TN D-3726, Princeton Univ Rpt 727, and AFFDL-TR-67-98) indicate that, for Class I and IV aircraft performing precision tasks, even shorter values of T_R are required to obtain satisfactory flying qualities.

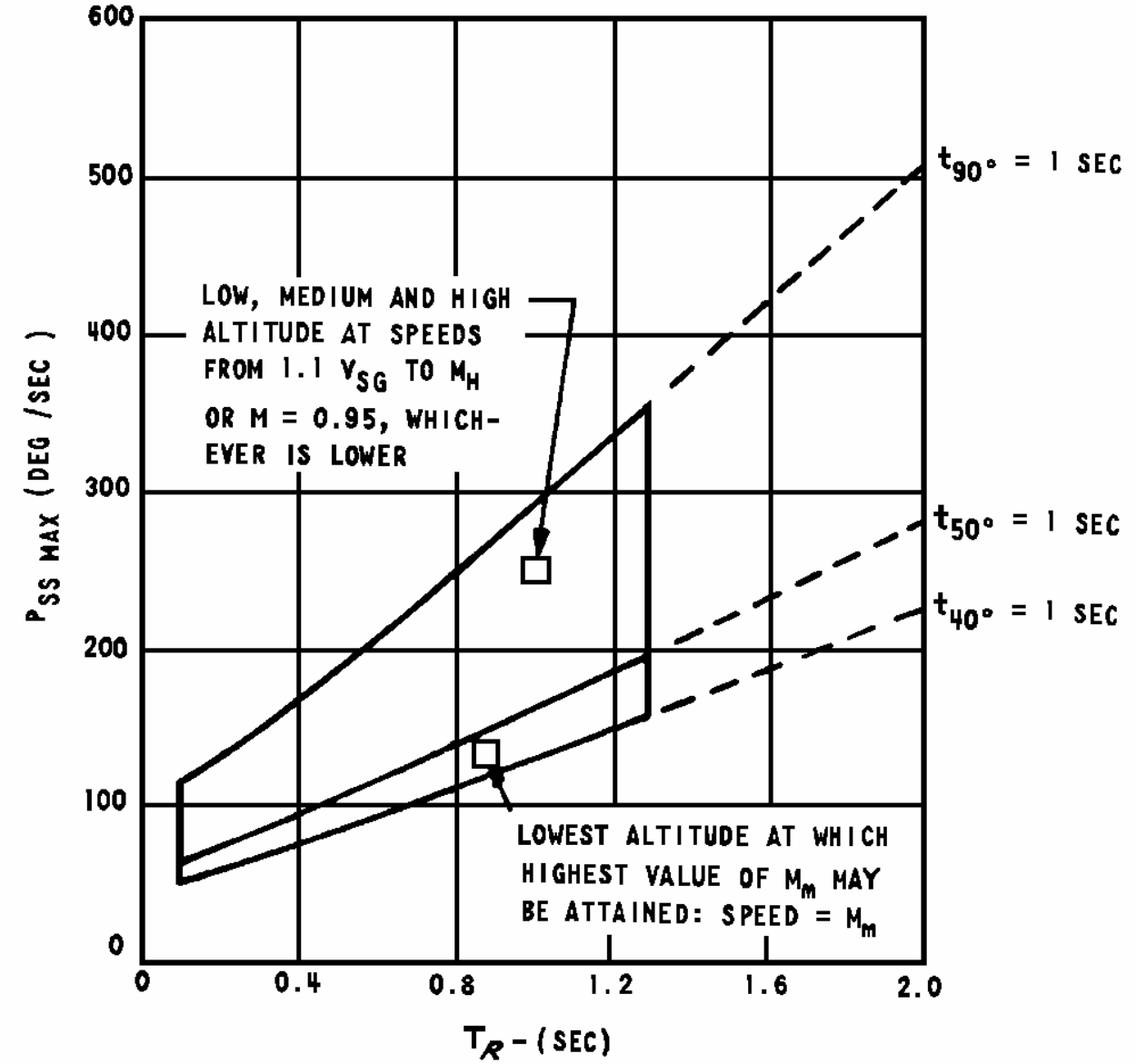


FIGURE 90. Proposed roll performance requirements (MIL-F-8785) for Class III aircraft (from NADC-ED-6282).

Princeton Univ Rpt 727 (figure 91, figure 92, and figure 93) shows that maximum satisfactory T_R for fighter aircraft for a carrier approach is approximately 1 sec. AFFDL-TR-67-98 (figure 94) shows that with a T_R of 1.3 sec, the best pilot rating obtained was 5 and in conclusion stated, "Because of the roll control difficulties the pilot experienced with the long roll mode time constant configuration, it was concluded that a roll mode time constant of 1.3 seconds or greater is unsatisfactory for a fighter mission." One prominent manufacturer of fighter aircraft stated that the fighter aircraft shall have a T_R of 0.6 to 0.8 sec. NASA TN D-3726 indicates, from consideration of time required to reach maximum roll rate, that Class I and small Class II aircraft require reasonably short roll mode time constants as well.

The data of NASA Memo 1-29-59A (figure 95 through figure 99) have been widely referenced and interpreted, as for example in AFFDL-TR-65-138 and NADC-ED-6282. It should be noted, however, that the in-flight evaluations in NASA Memo 1-29-59A were all for T_R less than 0.8 sec (figure 98) and any conclusions about roll mode time constants longer than 0.8 sec should be based on the ground simulation data only. In general, the in-flight ratings of NASA Memo 1-29-59A were worse than for the single-degree-of-freedom ground simulation ratings (figure 99). This indicates that the presented one-degree-of-freedom data (figure 97) may be a little optimistic. This difference in pilot ratings was discussed in NASA Memo 1-29-59A:

> "The principle argument is that the pilots' opinion of roll performance was adversely influenced by the coupling between the modes of motion which exist to some degree in all airplanes, but which for airplane D and for the low speed range of airplane F [see figure 89] were excessive, and which the single-degree-of-freedom analysis used herein obviously does not take into account. However, for airplane F, as the speed was increased the rolling motions approached those described by a single-degree-of-freedom system and correspondingly the actual pilot rating approached the predicted rating. Secondary factors which may have contributed to the above trend, wherein the actual rating was greater than the predicted, were objectionable control system dynamics and control system forces which may have been present."

So the simulator data (figure 95 and figure 96) may be considered to represent ideal aircraft.

Since, in general, a knee occurs in most of the data at approximately $T_R = 1$ sec (figure 89), and since $T_R = 1$ sec is at least consistent with all pertinent data, this value has been selected as the recommended Level 1 limit for Class I, II-C, and IV aircraft for Flight Phase Category C, and for Class I and IV aircraft for Flight Phase Category A.

For Class I and IV aircraft performing Flight Phase Category B tasks, and for Class II-L and III aircraft performing all tasks, available data support a maximum value of $T_R = 1.3$ to 1.5 sec; an average value of $T_R = 1.4$ sec was selected. Ground simulator data in NASA TN D-1888 tend to support this value for large aircraft (cross-hatched curves on figure 89); and in-flight data in WADD-TR-61-147 for small Class II airplanes, Flight Phase Category B (figure 100), support a T_R at least greater than 1.2 sec.

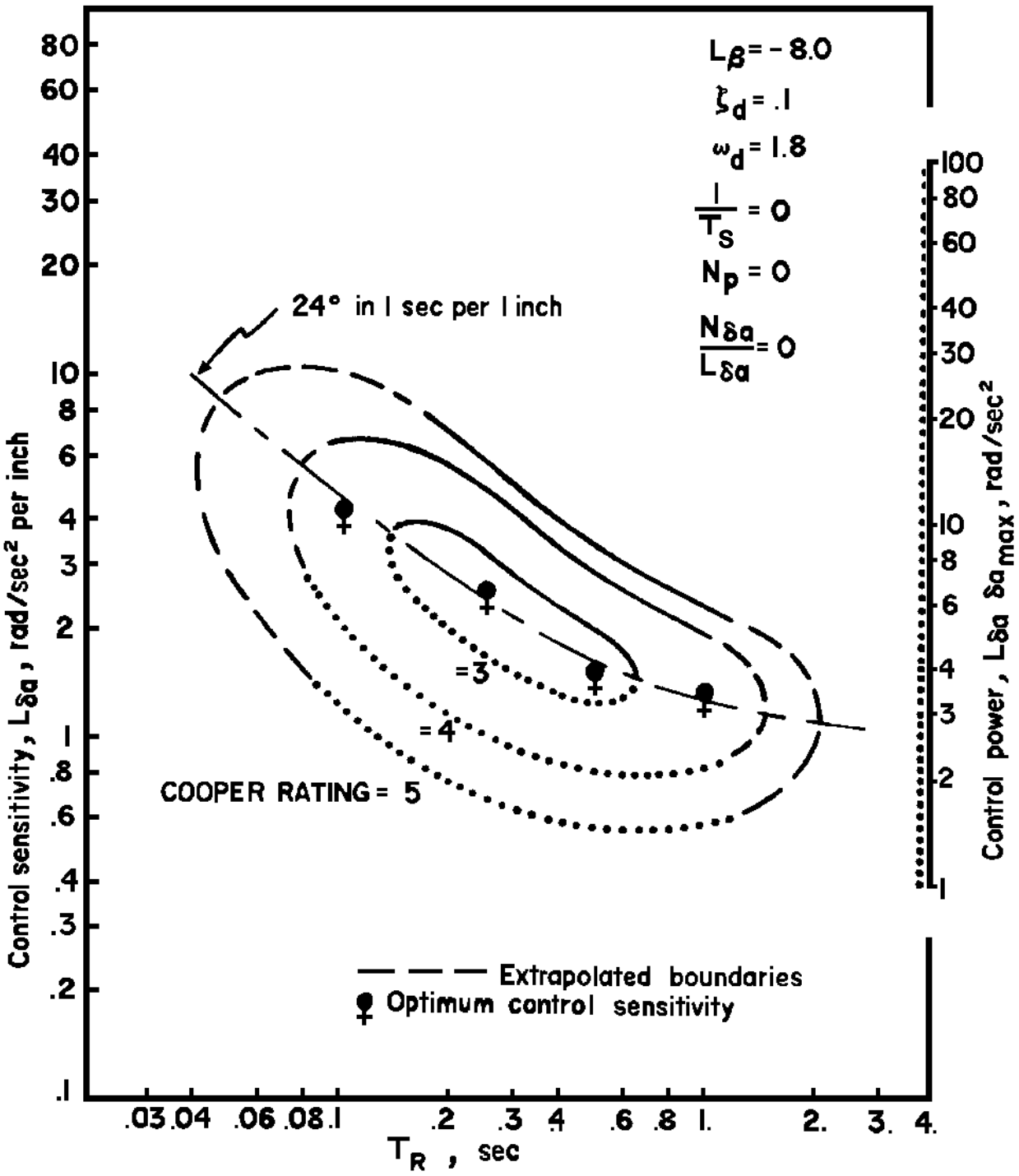


FIGURE 91. Lateral control boundaries (from Princeton Univ Rpt 727).

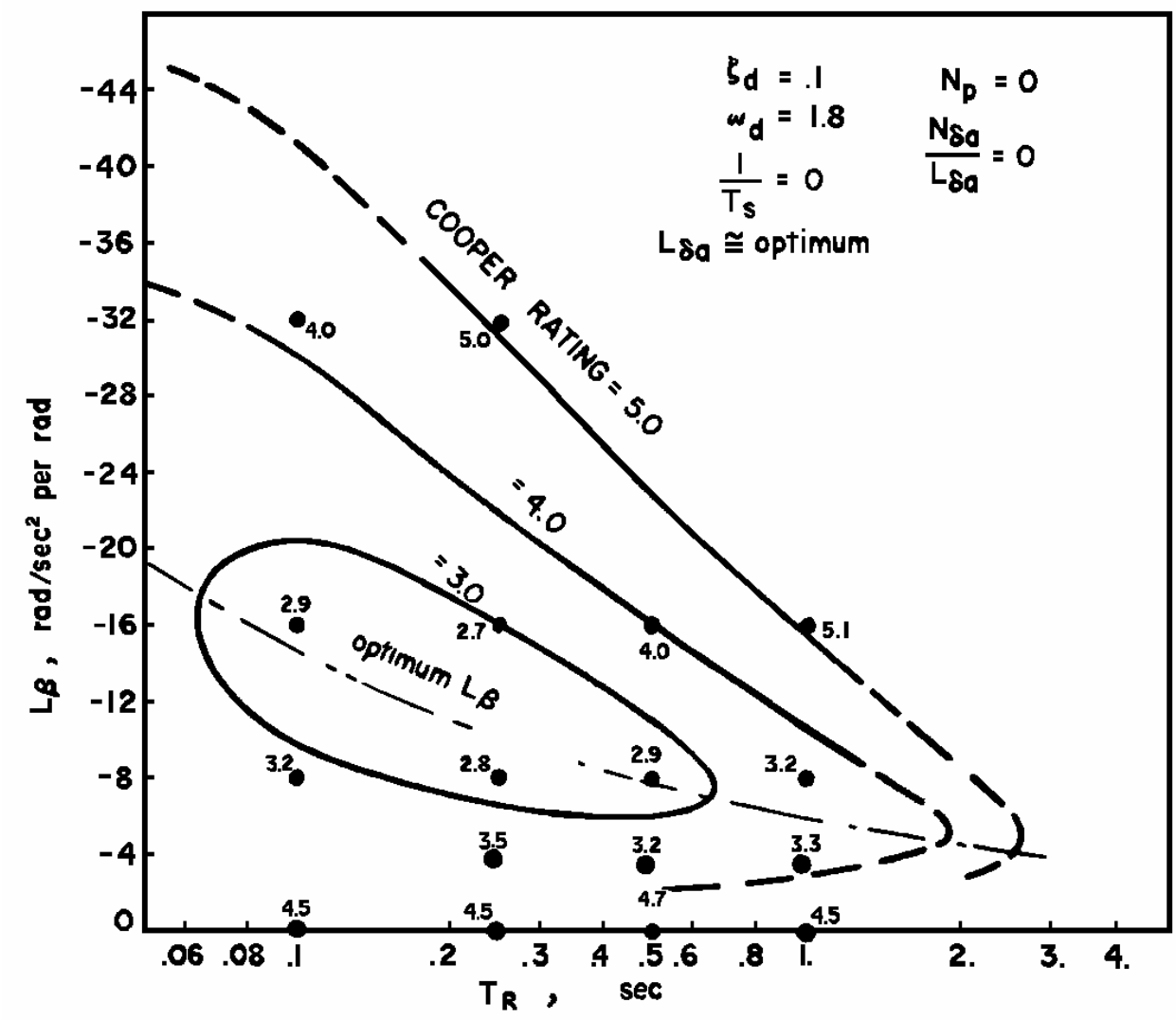


FIGURE 92. Lateral flying qualities boundaries (L_β versus T_R, ζ_d = 0.1) (from Princeton Univ Rpt 727).

An additional consideration demonstrated by much of the data, for example Princeton Univ Rpt 727 and AFFDL-TR-65-39, is that the required T_R is, to a degree, determined by the value of L_β or $|\phi/\beta|_d$. The in-flight data of Princeton Univ Rpt 727 (figure 92 and figure 93) show this dependence directly. In the opinion of the author of AFFDL-TR-65-39, the main reason for the differences between the data of AFFDL-TR-65-39 and the data to which it is compared (see figure 101) is that the AFFDL-TR-65-39 ground simulator data were based on a much larger value of $|\phi/\beta|_d$. In addition, the lack of adequate flight path display for the simulated high-speed condition (M = 1.2) of AFFDL-TR-65-39 may have contributed. The pilot ratings of both the AFFDL-TR-65-39 and Princeton Univ Rpt 727 data are degraded because of the response to atmospheric disturbances. This phenomenon is discussed in the substantiation for the paragraph which addresses the response to atmospheric disturbances (5.1.2).

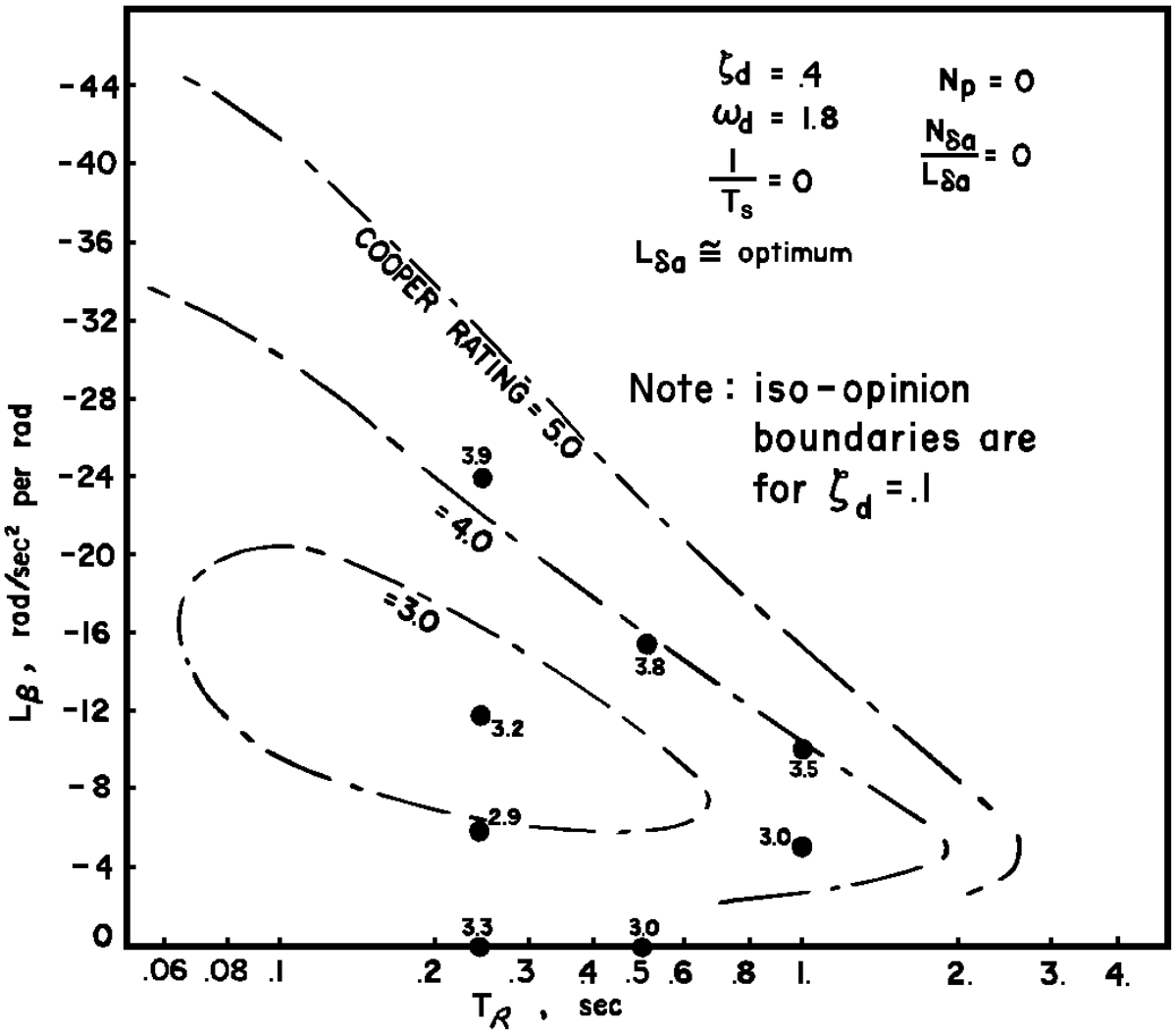


FIGURE 93. Lateral flying qualities boundaries (L_β versus T_R, $\zeta_d = 0.4$) (from Princeton Univ Rpt 727).

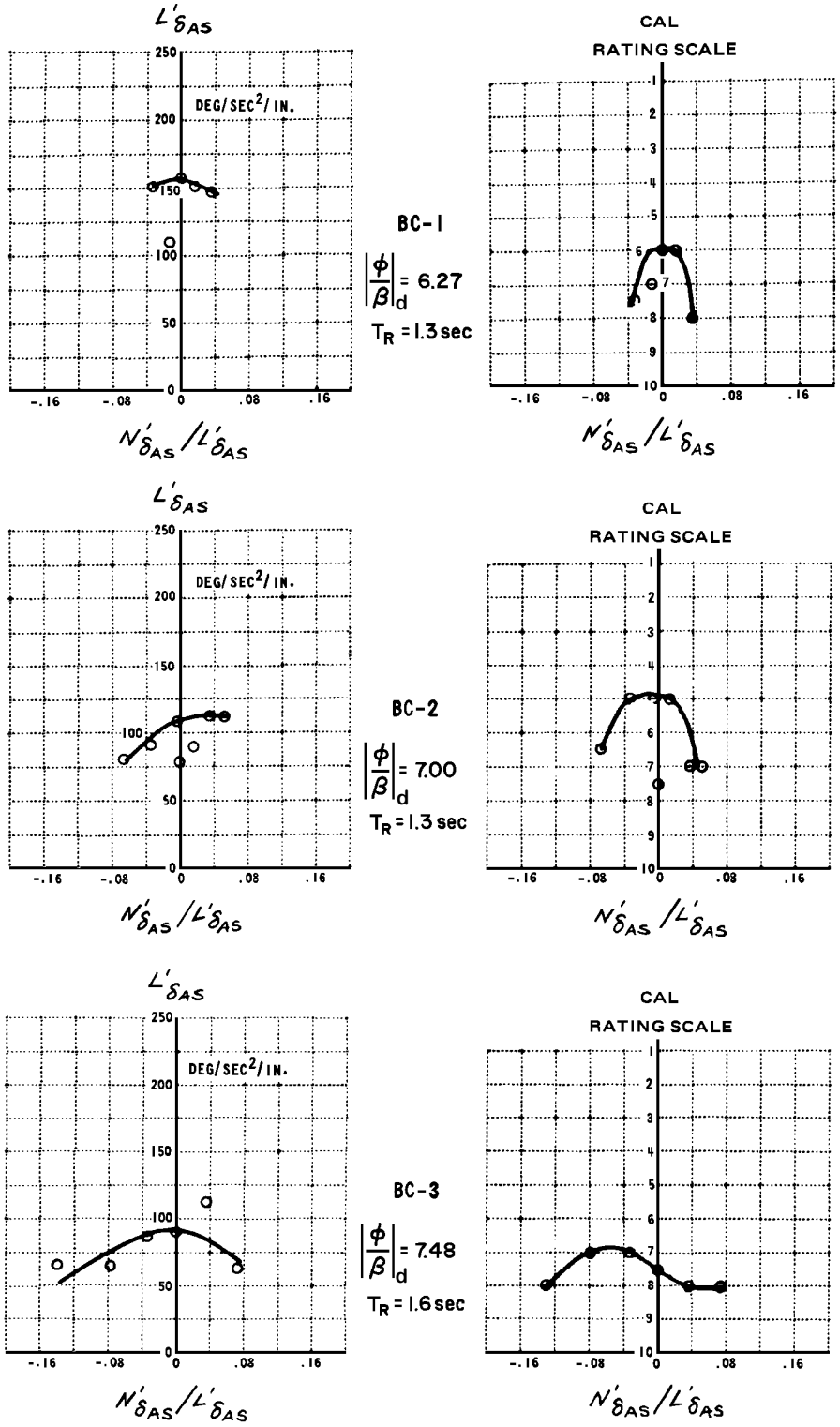


FIGURE 94. **Pilot ratings and optimum aileron sensitivity (medium $|\phi/\beta|_d$, long T_R) (from AFFDL-TR-67-98).**

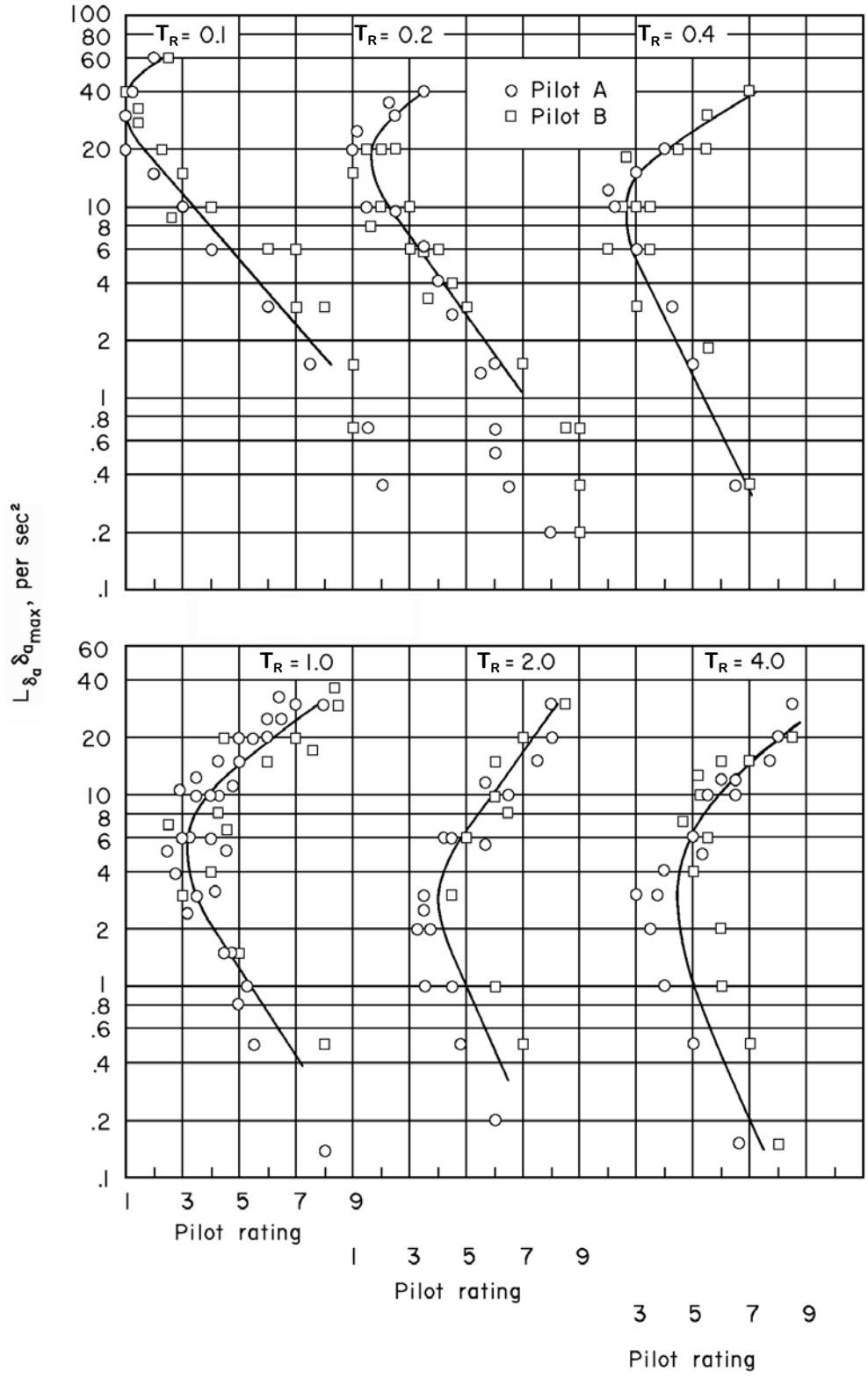


FIGURE 95. Variation of pilot opinion with $L_{\delta_a} \delta_{a_{max}}$, for constant values of T_R as obtained from the stationary flight simulator (from NASA Memo 1-29-59A).

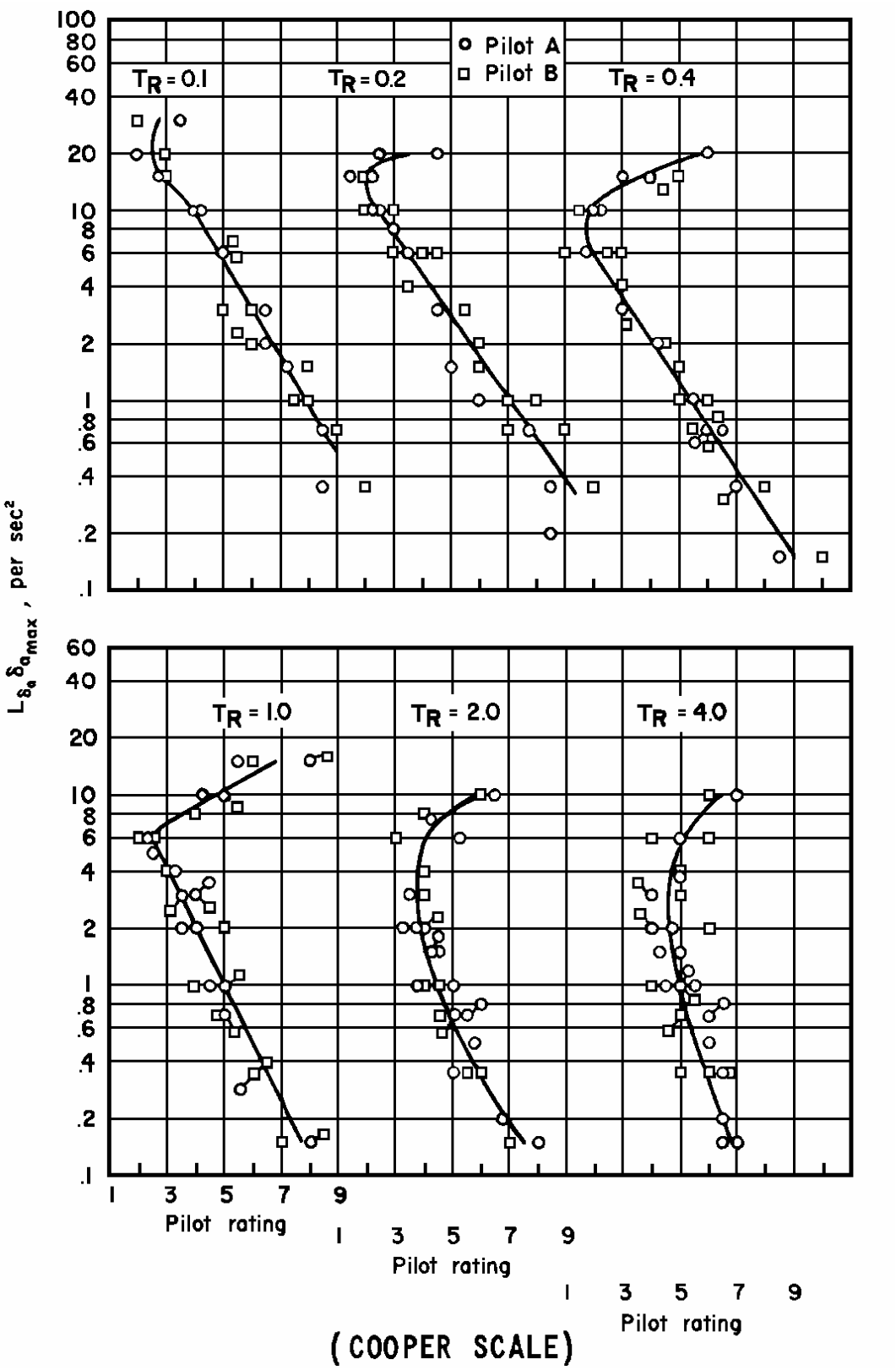


FIGURE 96. Variation of pilot opinion with $L_{\delta_a} \delta_{a_{max}}$, for constant values of T_R as obtained from the moving flight simulator (from NASA Memo 1-29-59A).

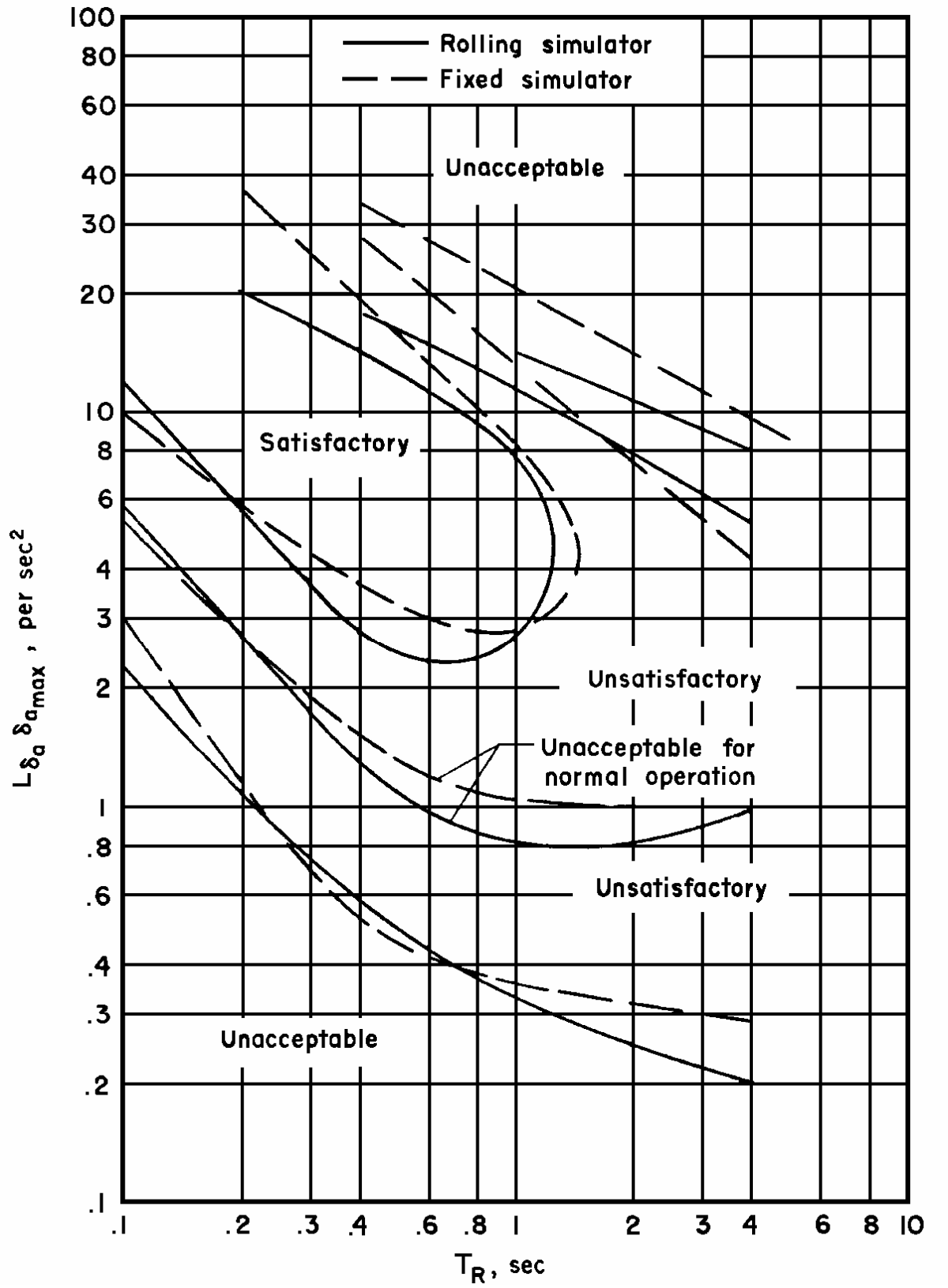


FIGURE 97. **Comparison of pilot opinion boundaries obtained from the fixed and moving flight simulators (from NASA Memo 1-29-59A).**

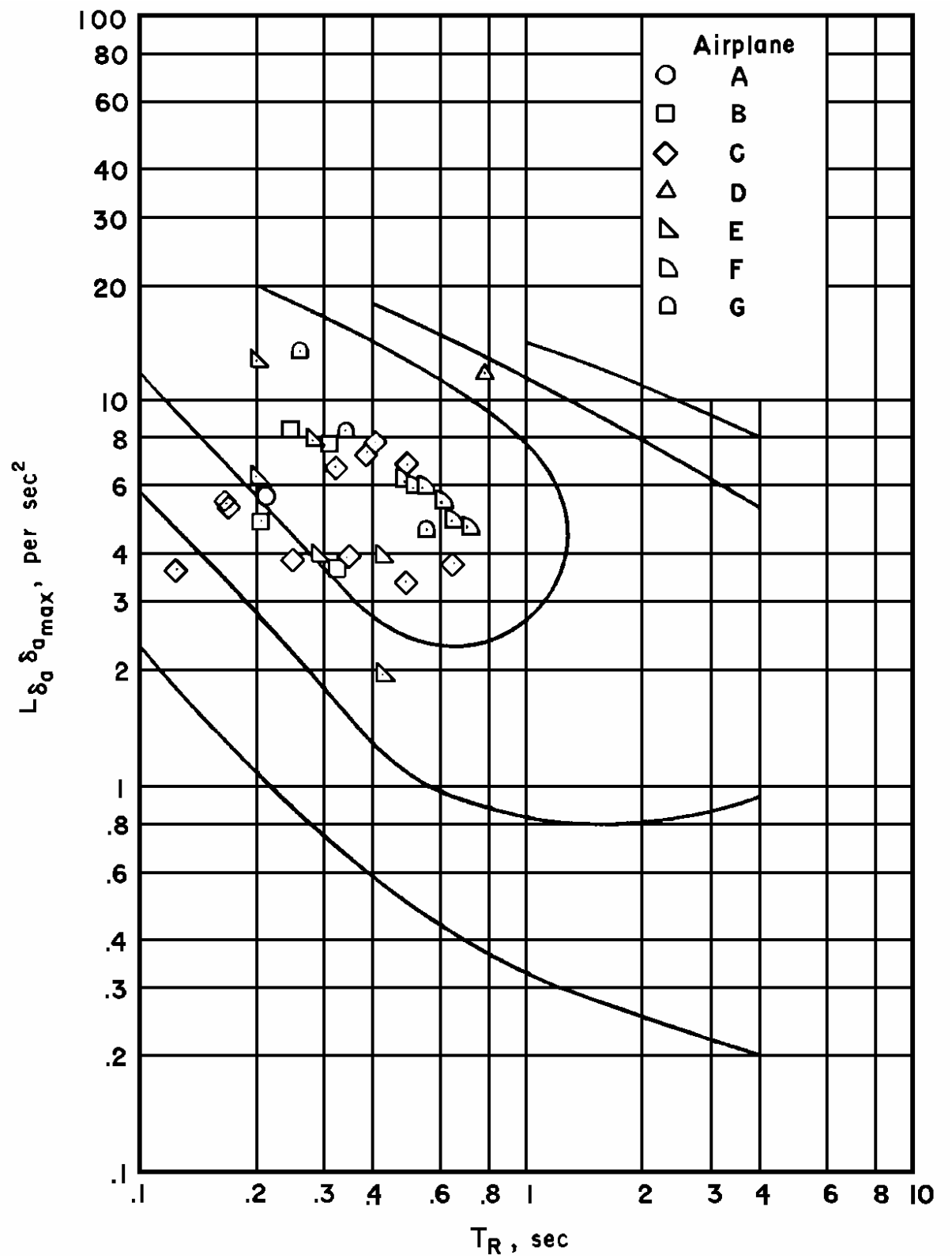


FIGURE 98. **Range of parameters $L_{\delta_a}\delta_{a_{max}}$ and T_R covered in the flight investigation shown in comparison with the motion-simulator-derived boundaries (from NASA Memo 1-29-59A).**

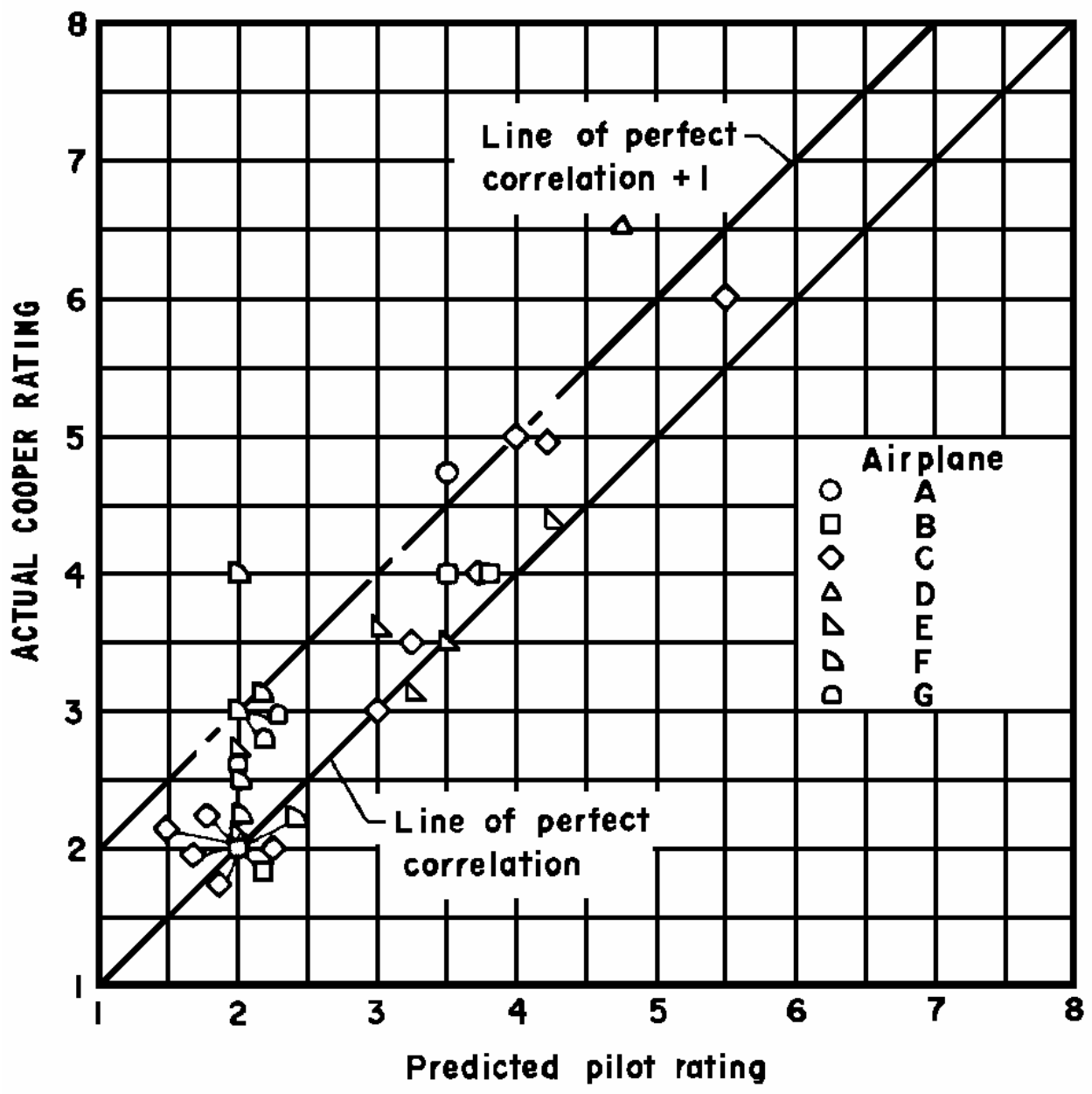


FIGURE 99. **Comparison of in-flight pilot-opinion rating with those predicted from flight simulator boundaries (from NASA 1-29-59A).**

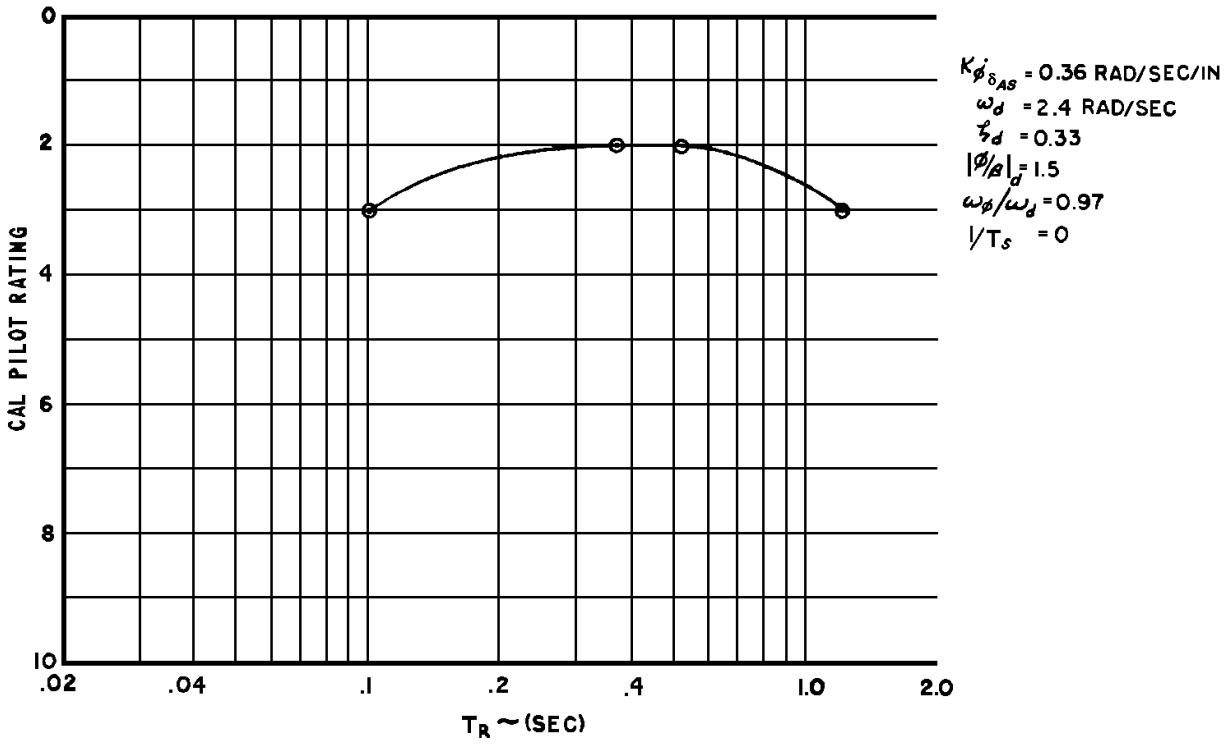


FIGURE 100. Pilot rating versus roll mode time constant (from WADD-TR-61-147).

Level 2 Requirements

AFFDL-TR-65-138 and NASA TN D-1888 do not make recommendations for Level 2 criteria as they did for Level 1. However, using available pilot rating data it is possible to select values of T_R that are consistent with available data.

Examination of figure 89 (from AFFDL-TR-65-138), which summarized data from NASA Memo 1-29-59A and NASA TN D-1888, shows that for a change of pilot rating from 3.5 to 5 or 5.5, T_R goes from approximately 1.3 to 3 sec. Thus, even though the NASA Memo 1-29-59A data are based on a fighter mission, the data do indicate that the gradient of pilot rating with T_R over the noted ranges. AFFDL-TR-67-98 indicates, from in-flight evaluations, that for fighter aircraft performing precision and maneuvering tasks, the pilot ratings degraded to marginally acceptable for T_R of 1.3 to 1.6 sec. For large aircraft, RAE Aero 2688 suggests T_R values of 2.3 sec for the satisfactory boundary and 6 sec for acceptable; however, these levels are probably associated with somewhat poorer flying qualities than the Level 1 and 2 defined in 3.2. The Level 2 recommendations were selected from these considerations.

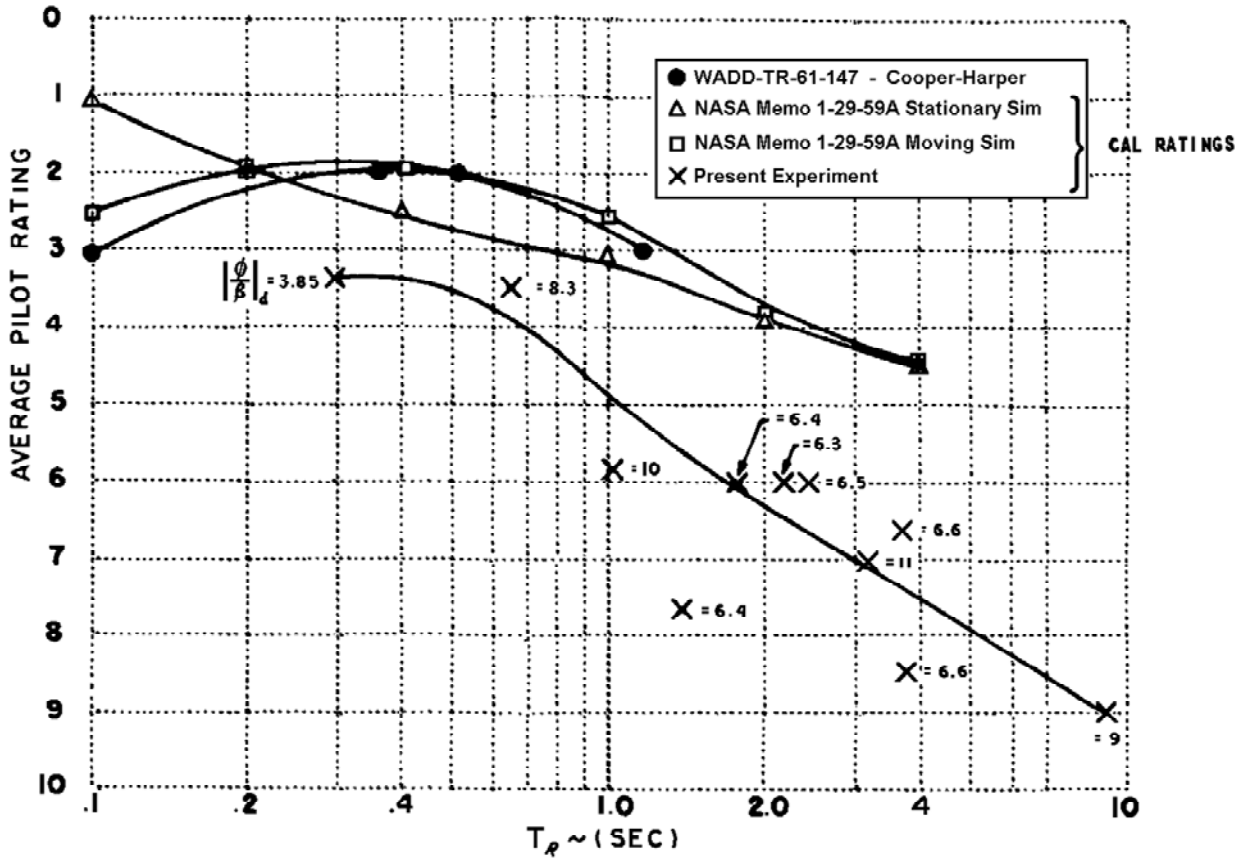


FIGURE 101. Average pilot rating of roll mode time constant (from AFFDL-TR-65-39).

Level 3 Requirement

A Level 3 value of T_R = 10 sec is relatively arbitrary but is based on data of AFFDL-TR-65-39 (figure 101) for fighter aircraft. While the selected value cannot be vigorously defended, it does legislate against unstable roll modes while permitting effective acceleration-like responses to control inputs such as can be obtained from wingless vehicles.

A comprehensive series of flight tests was conducted on the variable-stability NT-33 to investigate the effects of higher-order system dynamics on lateral handling qualities (LATHOS, for Lateral High-Order Systems, AFWAL-TR-81-3171), figure 102. Values of T_R within the existing Level 1 boundary were tested. However, the data for $1/T_R$ greater than 1.0 supports the current boundary (T_R < 1.0 sec) up to a value of $1/T_R$ = 3.0 (T_R = 0.33). For $1/T_R$ greater than 3.0, the pilot ratings show a consistent degradation, a trend that is not included in the current requirement. The pilot comments for these cases center about excessive lateral abruptness and roll ratcheting. These results are supported by the fact that some modern airplanes equipped with high-gain command augmentation systems (CAS) have short T_R and also experience excessive lateral sensitivity which has been described as roll ratcheting. CAS characteristics which may alleviate roll ratcheting are discussed at length in 5.2.3.8.2. The following will examine the effects of low values of T_R.

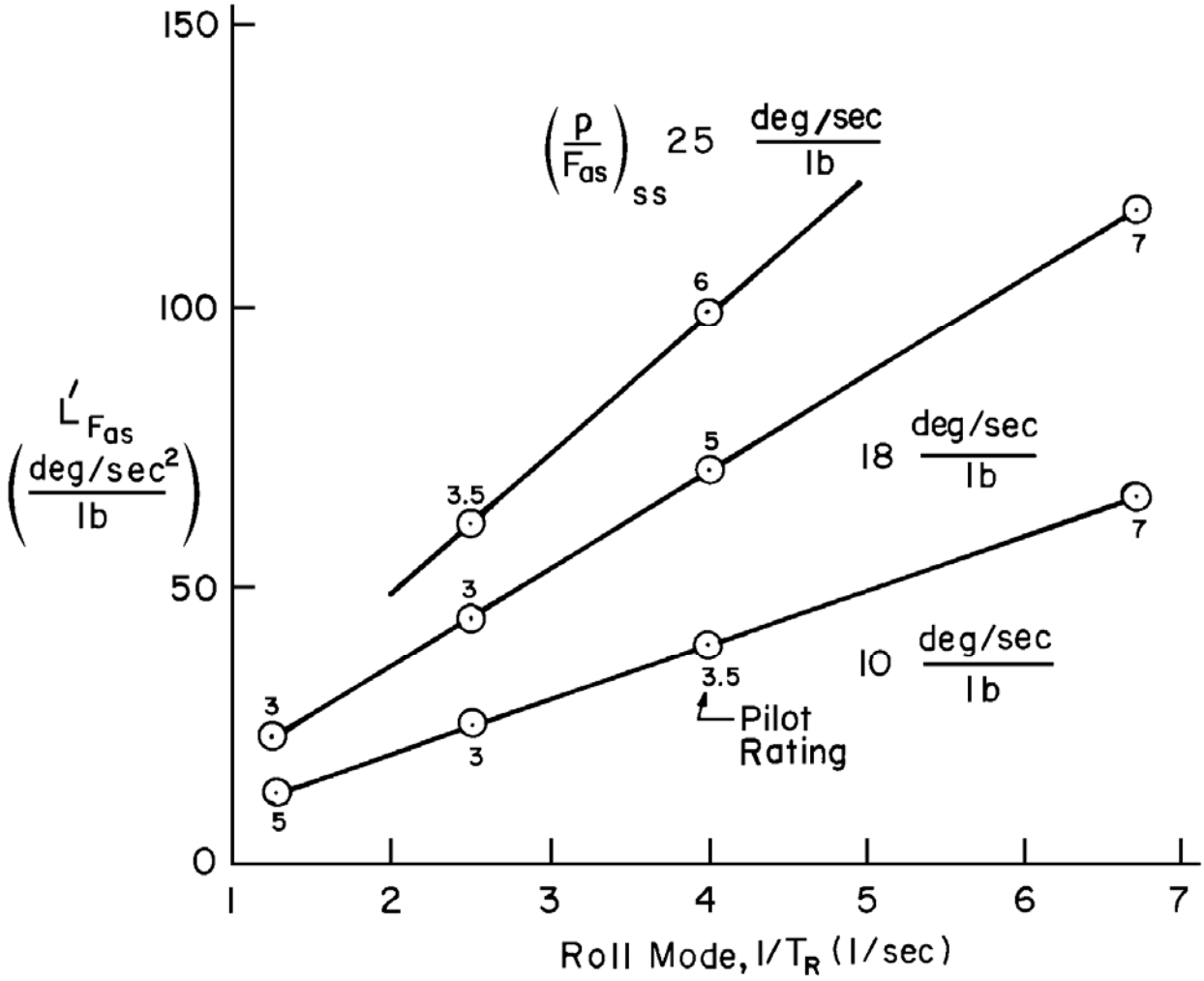


FIGURE 102. Effect of roll mode—LATHOS (AFWAL-TR-81-3171), Category A.

Several examples of ratcheting are shown on figure 98, figure 100, and figure 101. The phenomenon is characterized by near-limit-cycle oscillations at frequencies between 2 and 3 cycles per sec (12 to 18 rad/sec), well above the frequency of pilot control in the roll axis. The apparent dominant factor in ratcheting is excessive control gain (i.e., stick sensitivity) at these high frequencies. It has been suggested (CALSPAN FRM No. 554) that the root cause of ratcheting involves pilot closed-loop response to lateral acceleration cues: with a reasonable pilot lag, a closed-loop instability can exist when T_R is too short.

A related possible explanation of ratcheting is physiological in nature. That is, since the mass combination of pilot hand/arm and control stick are subjected to abrupt lateral accelerations, the effect would be a bobweight which would feed back to the aircraft motion. This phenomenon has been related to longitudinal pilot-in-the-loop oscillations (NORAIR Rpt No. NOR-64-143). Experiments conducted at the Air Force Aerospace Medical Research Laboratory (AMRL) investigated pilot control performance while experiencing sinusoidal lateral vibrations (AMRL-TR-73-78). A simple roll-bar-tracking maneuver with a well-behaved controlled element was utilized. Figure 103 compares results of this experiment with an analytical model for stick

deflection response to lateral accelerations. Pilot closed-loop tracking was at around 5 rad/sec, while an oscillatory arm/stick bobweight mode occurred at about 2 cycles per sec (12 rad/sec)—near the frequencies of the observed ratcheting oscillations in the LATHOS experiment.

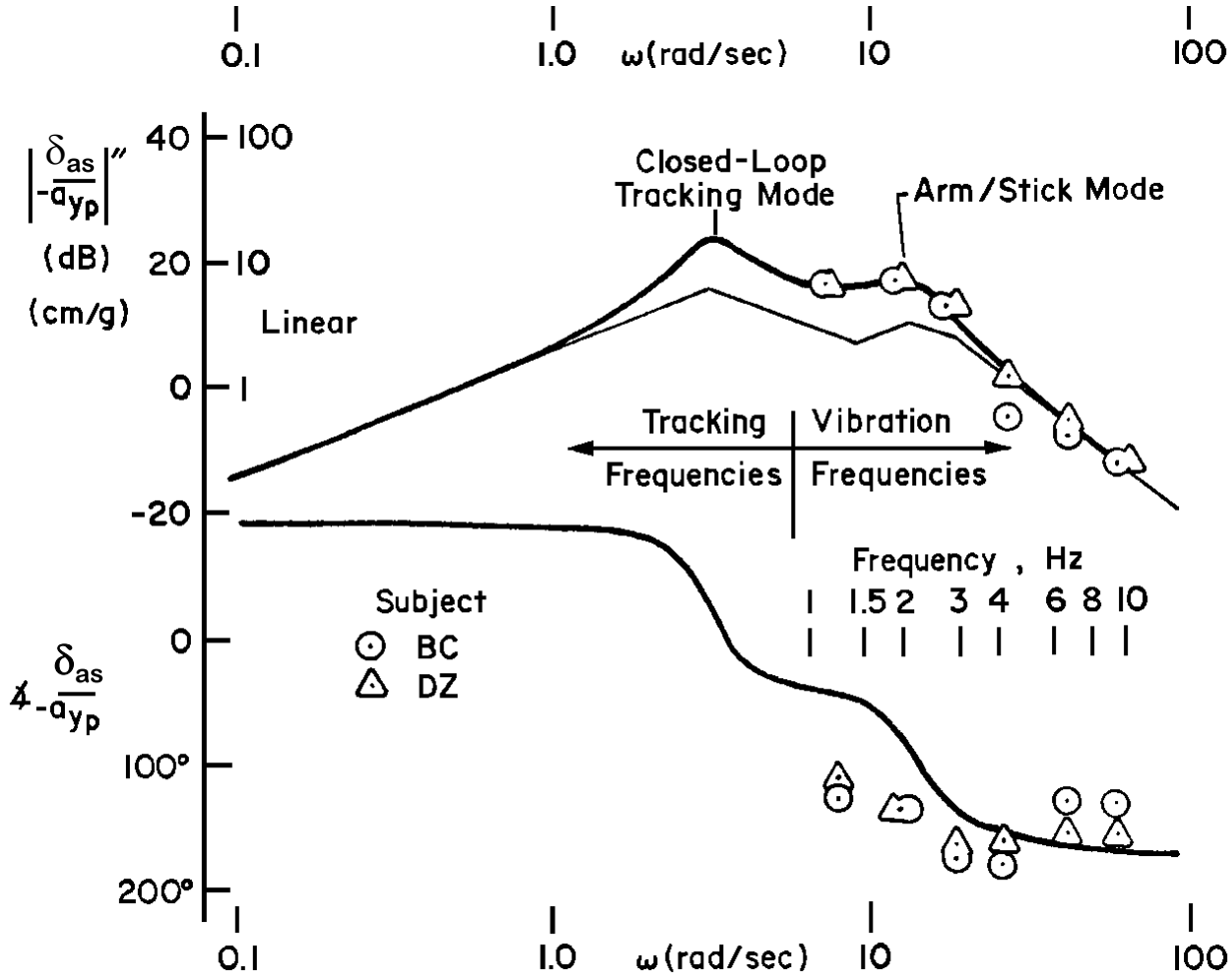


FIGURE 103. Comparison of models and data for closed-loop stick deflection responses under lateral vibration (AMRL-TR-73-78).

Several solutions to the problem of excessive sensitivity have been found. These are discussed in conjunction with the roll sensitivity discussion in 5.2.3.8.2. Those solutions are to: 1) decrease the stick sensitivity around neutral; 2) avoid too-low augmented-aircraft values of $1/T_R$; and 3) add a low-frequency stick prefilter with a break frequency of at least 10 rad/sec.

Reduction of stick sensitivity for CAS-equipped aircraft is relatively straightforward. Use of nonlinear gradients (ϕ/F_{as}) on LATHOS reduced the sensitivity only slightly but improved pilot ratings from 7 to 4 (see figure 102 and figure 103). It is clear that low sensitivity around neutral is essential for acceptable flying qualities.

Prefilters in the forward path were found to alleviate ratcheting on both LATHOS and the YA-7D DIGITAC (AFFTC-TR-76-15). The time constants of the filters were well into the range of

pilot crossover ($1/T_F$ around 3 rad/sec), and their effect as observed by the pilot was to smooth aircraft response (i.e., increase T_R). However, this should not be considered as a universal fix to the problem of sensitivity, since the aircraft response to outside disturbances might still be unacceptably abrupt. More importantly, prefilters can add considerable equivalent time delay to the system (see 5.2.3.1.5).

In summary, a large value of $1/T_R$ appears to result in excessive gain at high frequencies (see figure 104), which seems to be the root cause of roll ratcheting. The resulting lateral acceleration on the pilot would seem to account for the observed difference between in-flight and ground-based simulator results. The ratcheting can be alleviated to some extent by reducing the stick gain for small inputs; i.e., most high-frequency control activity occurs close to neutral roll command (see figure 173). However, resisting the temptation to overaugment $1/T_R$ seems to be the best overall solution. Even then, some nonlinear stick shaping will most likely be required (see figure 173).

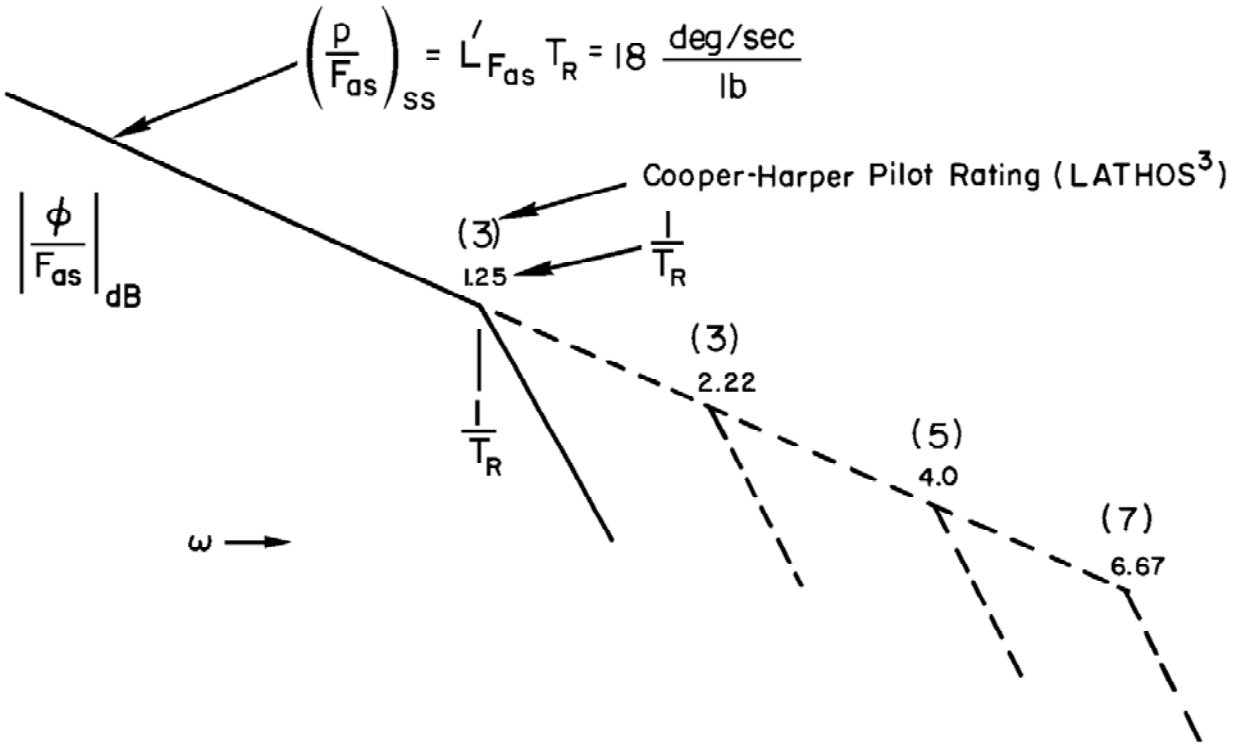


FIGURE 104. Effect of $1/T_R$ on high-frequency gain.

VERIFICATION

The equivalent system is to be fit to the higher-order system using algorithms similar to those specified in 5.2.2.1.1, over the frequency range from 0.1 rad/sec to 10.0 rad/sec. Use δ for deflection control systems (pilot controller deflection commands the control effector) and F for force control systems (pilot controller force commands the control effector).

Where the roll response to F_{as} contains a significant amount of Dutch roll mode, this transfer function shall be matched simultaneously with the equivalent-system transfer function of lateral-acceleration response to roll control force. (In general, this latter transfer function has a fourth-order numerator, but not all of its zeros may be within the frequency range of interest).

The roll rate response to a step roll control input for aircraft with conventional response is usually made up of three distinct modes: the roll mode, the spiral mode, and the Dutch roll mode. If the roll response is classical in nature (i.e., defined by the spiral, Dutch roll, and roll modes), conventional techniques may be utilized to determine T_R. If linearity is assumed, the principle of superposition applies. Then any point on the roll rate trace at any given time must be the sum of these three modes at that time (shown on figure 105 from AFFDL-TR-69-72, for a step command). Therefore, if the three modes can be identified on the roll rate trace, it is possible to extract the roll mode time constant, T_R. Methods of extracting values of T_R from flight test data are given in AFFDL-TR-69-72, Appendix VB. For a rate-limited input, take time = 0 at the midpoint of the ramp to determine T_R. Compliance is easily demonstrated through flight testing at the minimum and maximum specified operational altitudes, over the range of service speed, with the aircraft configured for maximum rolling moment of inertia.

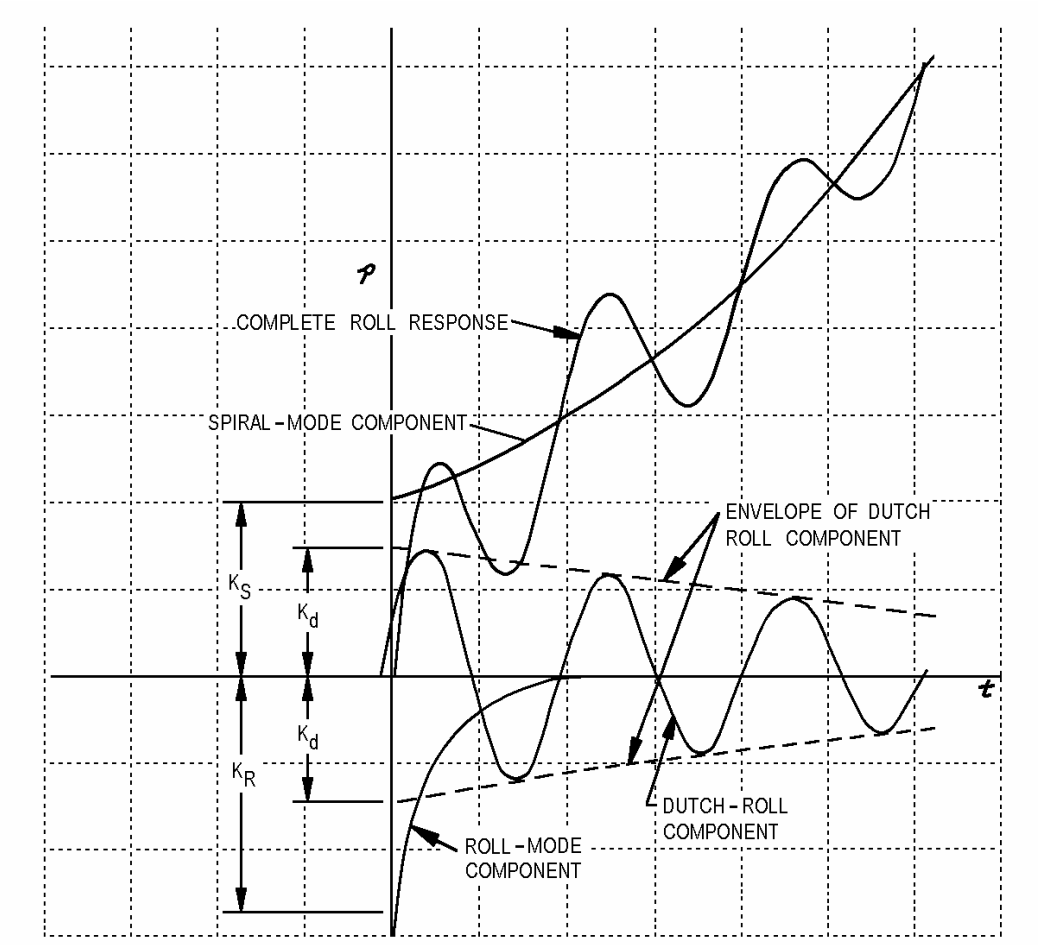


FIGURE 105. Dutch roll, roll, and spiral mode components of roll rate response to a step input.

5.2.3.1.2 Dutch roll frequency and damping.
The equivalent Dutch roll frequency, ω_d, and damping ratio, ζ_d, shall exceed the minimum values in table XLV. When $\omega_d^2|\phi/\beta|_d$ is greater than 20 $(rad/sec)^2$, the minimum $\zeta_d\omega_d$ shall be increased above the $\zeta_d\omega_d$ minimums listed in table XLV by:

Level 1: $\Delta\zeta_d\omega_d = 0.014\ (\omega_d^2|\phi/\beta|_d - 20)$

Level 2: $\Delta\zeta_d\omega_d = 0.009\ (\omega_d^2|\phi/\beta|_d - 20)$

Level 3: $\Delta\zeta_d\omega_d = 0.005\ (\omega_d^2|\phi/\beta|_d - 20)$

with ω_d in rad/sec.

TABLE XLV. Minimum Dutch roll frequency and damping.

Level	Flight Phase Category	Minimum ζ_d	Minimum $\zeta_d\omega_d$ (rad/sec)	Minimum ω_d (rad/sec)
1	A			
	B			
	C			
2	A			
	B			
	C			
3	A			
	B			
	C			

REQUIREMENT RATIONALE

This requirement assures that any lateral-directional oscillatory (Dutch roll) response to yaw control will be sufficiently stable and well damped. Requirements on Dutch roll damping are specified to limit the oscillations of the Dutch roll after it has been excited. Requirements on minimum Dutch roll frequency are specified to limit the amount of sideslip generated by a given yawing disturbance and to ensure that the aircraft will naturally tend to point the way it is going.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.1.1
MIL-STD-1797: 4.6.1.1.

Recommended values for table XLV:

Level	Flight Phase Category	Class	Minimum ζ_d *	Minimum $\zeta_d\omega_d$ * (rad/sec)	Minimum ω_d (rad/sec)
1	A (CO, GA, RR, TF, RC, FF, and AS)	I, II, III, and IV	0.4	0.4	1.0
	A	I and IV	0.19	0.35	1.0
		II and III	0.19	0.35	0.4
	B	All	0.08	0.15	0.4
	C	I, II-C, and IV	0.08	0.15	1.0
		II-L and III	0.08	0.10	0.4
2	All	All	0.02	0.05	0.4
3	All	All	0	--	0.4

* The governing damping requirement is that yielding the larger value of ζ_d, except that a ζ_d of 0.7 is the maximum required for Class III aircraft.

REQUIREMENT LESSONS LEARNED

Allowable Dutch roll oscillatory characteristics are specified in terms of minimum values of ζ_d, ω_d, and the product $\zeta_d\,\omega_d$; the last is also a function of $|\phi/\beta|_d$ when $|\phi/\beta|_d$ is very large. From examination of supporting data it is apparent that over a wide range of frequencies and $|\phi/\beta|_d$ response ratios, lines of constant damping ratio (ζ_d) fit the data quite well. In determining the minimum frequency (ω_d) boundaries, it was found that the more closely the low-frequency data were examined, the more difficult it became to assess the importance of low Dutch-roll frequency per se. Not surprisingly, there is support for raising the minimum acceptable value of ζ_d when ω_d is low, i.e., ζ_d and ω_d are not independent. This is reflected by specifying a minimum for the total damping ($\zeta_d\,\omega_d$).

The total damping has also been made a function of the product $\omega_d^2\,|\phi/\beta|_d$. While the data to support this are sparse, there is a clear need to account for possible turbulence effects on aircraft with high Dutch roll frequencies and high $|\phi/\beta|_d$.

Figure 106 illustrates the range of Dutch roll damping and frequency found on some airplanes (AGARD-CP-17). Airplanes include the B-52, B-58, XB-70, C-130, C-141A, C-5A, F-104, F-105, F-4D, A-7D, F-111, Boeing 707-300, 720B, 727, and an SST design. The symbols shown are for several different flight conditions.

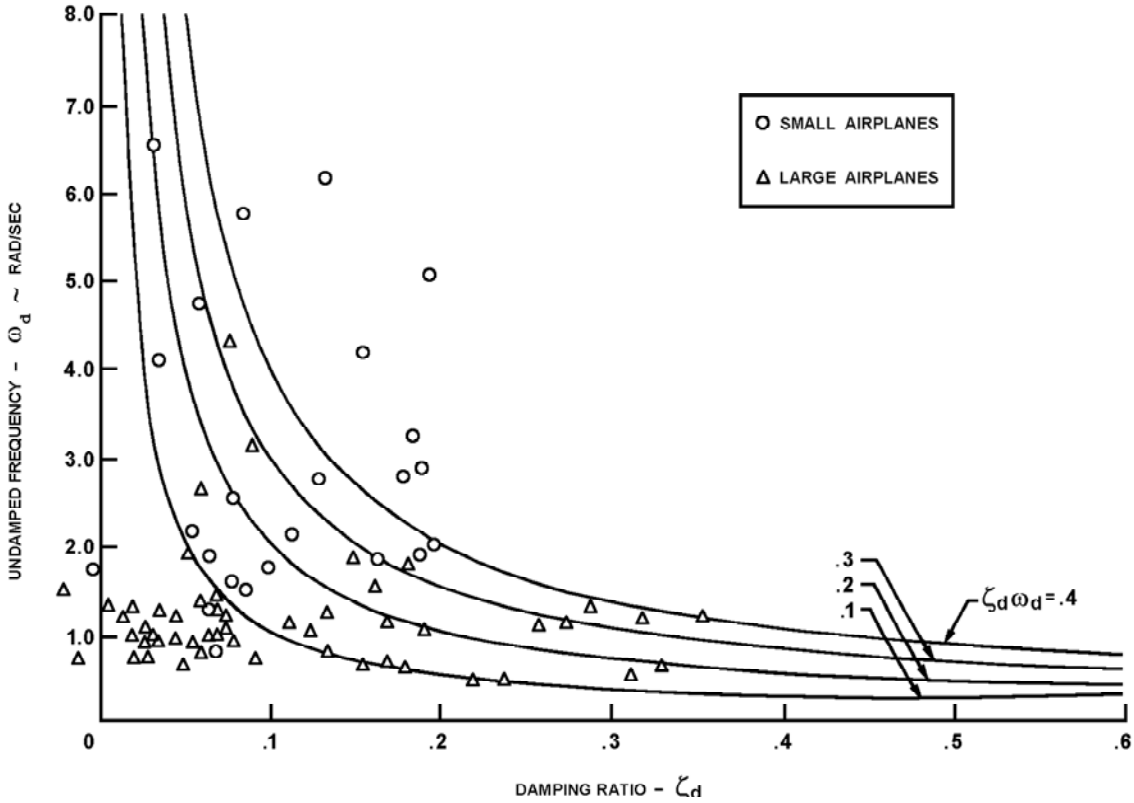


FIGURE 106. Dutch roll data on various airplanes (from AGARD-CP-17).

Characteristics for the Lockheed C-5A, C-141A, YC-141B, and L-1011 (AFFDL-TR-78-171) are plotted on figure 107 for the Category B Flight Phase with yaw damper inoperative. The L-1011 meets the minimum requirements for Level 2 operation, while the C-5A, C-141A, and YC-141B all fail the Level 2 damping requirements. AFFDL-TR-78-171 discusses the results for the C-141A and C-5A:

> "An evaluation of the C-141A Dutch roll recovery techniques with yaw damper inoperative was conducted by the Air Force Flight Test Center in Feb 1977. Results of the test ["C-141A Dutch Roll Recovery," AFFTC Technical Letter Report, by Picha and Klein] show Cooper-Harper rating values ranging from 2.0 to 5.0, using aileron only for recovery, which is the recommended Flight Handbook procedure. Over 100 Dutch roll maneuvers were accomplished during the evaluation, which consisted of regaining control of the aircraft and returning to a wings-level attitude from bank angles as high as 45º. It should also be noted that evaluating pilots do not rate operation of the C-5A with the stability augmentation system off below [worse than] the suggested Level 2 guidelines (6.5 Cooper-Harper rating scale). These data strongly indicate that the Level 2 minimum $\zeta_d\ \omega_d$ requirement of 0.05 rad/sec is too stringent."

Comments by SPOs on the application of Dutch roll requirements shows the opposite trend: minimum allowable values of ζ_d have been increased for some current aircraft. Concern also

was raised over applicability of the Dutch roll requirements to augmented aircraft. The following table summarizes these comments for specific airplanes:

F-16	Parameters in this section are not easily applied to highly augmented aircraft.
F-15, F-16, C-141	ζ_d was increased to 0.30 for Category A.
AMST, B-1	Values for ζ_d were increased on AMST for Level 1 Category B and C (0.08 to 0.20) and Levels 2 and 3 (0.02 to 0.08); the minimum values of ω_d were also increased: a requirement that addresses higher dynamic modes that are present with augmentation needs to be defined.
A-10	Increased ζ_d above that required by MIL-F-8785B was the key to achieving good air-to-ground accuracy, as described in 4.1.2 Requirement Lessons Learned.

Considerations other than flying qualities may determine the acceptability of Dutch roll characteristics: ride qualities in turbulence or in aileron rolls, or structural fatigue, for example.

VERIFICATION

It is intended that equivalent values of ζ_d and ω_d be used for augmented aircraft as well as those with simpler control systems. Inasmuch as there has been little work done to develop lower-order equivalent systems for the Dutch roll response, guidance on this area is limited.

The parameters should be found by matching the higher-order sideslip response to yaw control input, typically over the frequency range from 0.1 rad/sec to 10 rad/sec. For most aircraft an appropriate lower-order equivalent system for sideslip response to a rudder input is adequate:

$$\frac{\beta(s)}{\delta_{rp}(s) \text{ or } F_{rp}(s)} = \frac{K_{\beta_e}\, e^{-\tau_{\beta_e} s}}{s^2 + 2\,\zeta_{d_e}\,\omega_{d_e}\,s + \omega_{d_e}^2}$$

This is a simplification of the more complete three-degree-of-freedom classical form:

$$\frac{\beta(s)}{\delta_{rp}(s) \text{ or } F_{rp}(s)} = \frac{K_{\beta_e}\,(s + 1/T_{\beta_{1_e}})(s + 1/T_{\beta_{2_e}})\,(s + 1/T_{\beta_{3_e}})\, e^{-\tau_{\beta_e} s}}{(s + 1/T_{S_e})\,(s + 1/T_{R_e})\,(s^2 + 2\,\zeta_{d_e}\,\omega_{d_e}\,s + \omega_{d_e}^2)}$$

where typically (but not necessarily) one zero is very large, one is small (of the order of $1/T_S$), and the other is close to $1/T_R$ (see Aircraft Dynamics and Automatic Control).

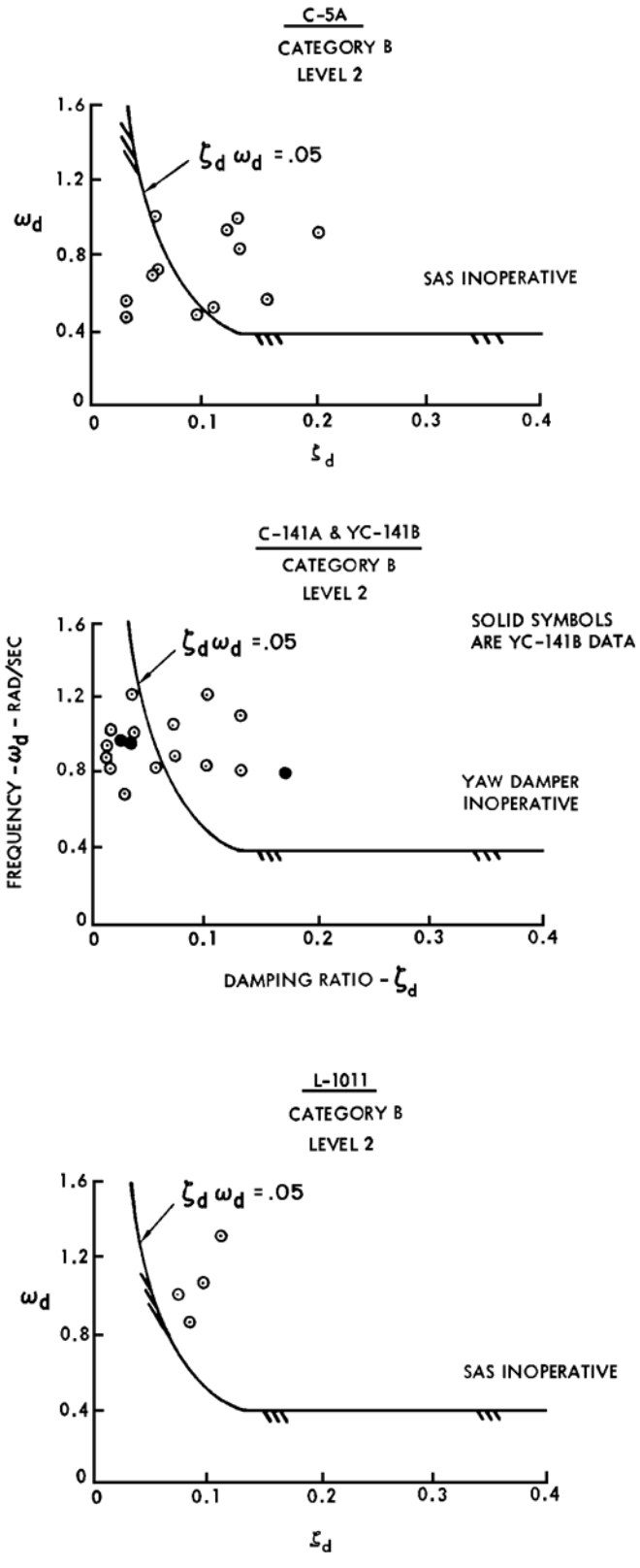


FIGURE 107. Lateral-directional damping for some Class III airplanes (from AFFDL-TR-78-171).

Alternatively, when $|\phi/\beta|_d$ is large, a significant portion of the Dutch roll response occurs in roll, so that it may be desirable to match equivalent sideslip and roll responses simultaneously in order to best match pilots' impressions. The ϕ/F_{as} equivalent-system transfer function of 5.2.3.1 shall be matched simultaneously with the equivalent-system β/F_{as} transfer function:

$$\frac{\phi(s)}{\delta_{as}(s) \text{ or } F_{as}(s)} = \frac{K_{\phi_e} (s^2 + 2\zeta_{\phi_e} \omega_{\phi_e} s + \omega_{\phi_e}^2) e^{-\tau_{\phi_e} s}}{(s + 1/T_{S_e})(s + 1/T_{R_e})(s^2 + 2\zeta_{d_e} \omega_{d_e} s + \omega_{d_e}^2)}$$

$$\frac{\beta(s)}{\delta_{as}(s) \text{ or } F_{as}(s)} = \frac{K_{\beta_e} (s + 1/T_{\beta_{1e}})(s + 1/T_{\beta_{2e}})(s + 1/T_{\beta_{3e}}) e^{-\tau_{\beta_e} s}}{(s + 1/T_{S_e})(s + 1/T_{R_e})(s^2 + 2\zeta_{d_e} \omega_{d_e} s + \omega_{d_e}^2)}$$

Use δ_{as} for deflection controllers (pilot controller deflection commands the control effectors) and F_{as} for force controllers (pilot controller force commands the control effectors). The algorithms described in 5.2.3.1 are useful for the fitting process, although any mutually-agreed matching technique may be used. No limits are set on ϕ or β numerator terms.

To determine the equivalent ζ_d and ω_d, for cases where τ_{β_e} is small, simple measurements from a time response of sideslip to rudder kick will frequently be sufficient. A pair of sustained lateral-directional control pulses or doublets has been found to yield good identification without a large departure from trim. Classical time response measures should be sufficient to extract ζ_d and ω_d from the sideslip time histories. The parameter $|\phi/\beta|_d$ (at any instant, the ratio of the amplitudes of the bank angle and sideslip angle envelopes in the Dutch roll mode) is a factor in determining the required total damping $\zeta_d \omega_d$. $|\phi/\beta|_d$ can be determined simply from flight or simulator time histories. Methods of extracting values of $|\phi/\beta|_d$ from flight test data are given in AFFDL-TR-69-72, Appendix VB. From the envelopes of the Dutch roll component oscillations in ϕ and β, measure $|\phi|$ and $|\beta|$ at any instant sufficiently far along in the response (see figure 108). The ratio of $|\phi|$ and $|\beta|$ is $|\phi/\beta|_d$. Since the ratio is invariant with time, any suitable instant may be chosen.

Compliance must be demonstrated through flight testing at the minimum and maximum specified operational altitudes, over the range of service airspeeds, with the aircraft configured for maximum yawing moment of inertia to get the lowest ω_d. Factors affecting the stability derivatives include: angle of attack, Mach number, and aerodynamic pressure. Aerodynamic directional stability may deteriorate noticeably as the c.g. moves aft.

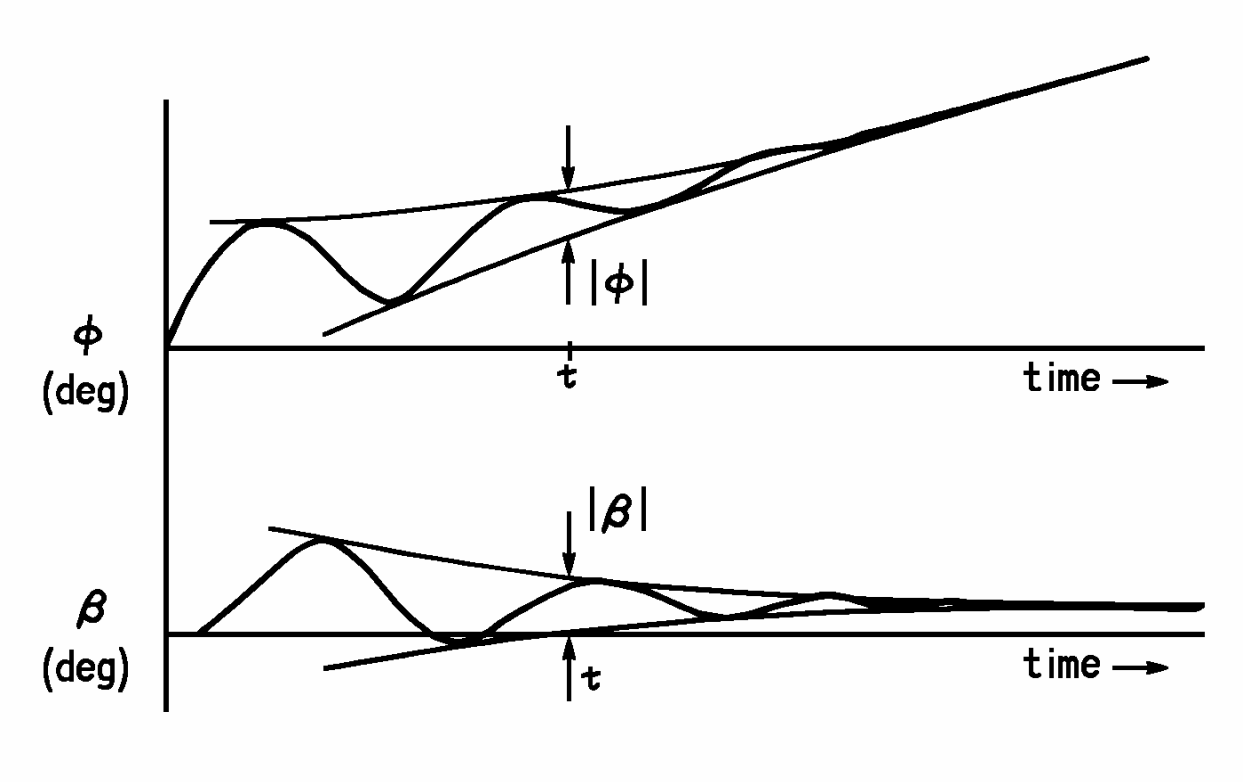


FIGURE 108. Definition of $|\phi/\beta|_d$.

5.2.3.1.3 Spiral stability.
The combined effects of spiral stability, flight control system characteristics, and rolling moment change with speed shall be such that following a disturbance in bank of up to 20º, the time for the bank angle to double shall be greater than the values in table XLVI. This requirement shall be met with the aircraft trimmed for wings-level, zero-yaw-rate flight and the cockpit controls free.

TABLE XLVI. Spiral mode minimum time to double amplitude.

Flight Phase Category	Level 1	Level 2	Level 3
A			
B			
C			

REQUIREMENT RATIONALE

The requirements on spiral stability are aimed primarily at restricting the amount of aileron that must be held during a turn, and at limiting the amount of attention required by the pilot to fly the aircraft by insuring that the aircraft will not diverge too rapidly in bank from a wings-level condition during periods of pilot inattention.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.1.3
MIL-STD-1797: 4.5.1.2.

Recommended values for table XLVI:

Flight Phase Category	Level 1	Level 2	Level 3
A and C	12 sec	8 sec	4 sec
B	20 sec	8 sec	4 sec

REQUIREMENT LESSONS LEARNED

This requirement is well established and the numbers come from operational experience, borne out by rough estimates of the lower limit of pilots' frequency range of active control. The spiral mode, characterized by slow rolling and yawing responses to a roll disturbance, is generally not a

problem for the pilot during fully-attended operations (air combat, landing, etc.). However, spiral divergence during low-gain tasks can be a nuisance and even a dangerous condition if the divergence is too rapid. A limit on time to double amplitude for the spiral mode is necessary for safety during such operations. There is as yet no clear need for, or definition of, a limit on positive spiral stability. Indeed, a coupled roll-spiral mode (5.2.3.1.4) might sometimes be desirable: for example, attitude command on landing approach, where fine tracking but no rapid, gross maneuvering is required. Roll and yaw control forces in steady turns are bounded in 5.2.3.8.5.

NADC-ED-6282 recommended retention of the existing $T_2 > 20$ sec requirement and also proposed a requirement $T_{1/2} > 10$ sec, on the degree of positive spiral stability permitted. The recommended allowable instability of table XLVI is similar, in that for Flight Phase Category B (analogous to the cruise configuration) the time to double amplitude is $T_2 > 20$ sec. But for Flight Phase Category A and C, where the pilot is generally closing a tight attitude loop, a less stringent value of $T_2 > 12$ sec was selected.

Grouping Category C Flight Phases with Category A Flight Phases is based on the consideration that during Category A and C Flight Phases the pilot is in more continuous control of the aircraft than in Category B Flight Phase and is therefore less concerned about long-term attitude characteristics. This point was demonstrated in the TIFS Phase I landing approach experiments reported in AFFDL-TR-71-164. Spiral roots with time to double of 9.6 sec were hardly noticed and a case with time to double of 6.4 sec, although noted, was not considered reason for downgrading the evaluation. Based on these data together with the extensive data in FAA-ADS-69-13 and NRC of Canada Rpt LTR-FR-12, it is recommended that the Level 2 limit of T_2 be 8 sec. Even this limit is a conservative interpretation of the data in FAA-ADS-69-13, which could be used to support a value of $T_2 = 6$ sec for Level 2. The gradient of pilot rating with time to double is steep, however, and a conservative interpretation is believed justified. The data of Cornell Aero Lab TB-574-F-6 (figure 109) tend to suggest, however, that a higher value of T_2 might be justified for Level 2.

For Level 3, a value of $T_2 > 4$ sec was selected as a compromise between what is flyable and what is controllable if the pilot cannot devote full attention to flying the aircraft. This subject was discussed as follows in Cornell Aero Lab TB-574-F-6:

> "The minimum tolerable time to double amplitude of the spiral divergence was very much longer than the minimum allowed by the existing handling qualities specifications [Navy BuAer SR-119B/USAF C-1815B]. It is believed that the concept of the spiral divergence being unimportant to the pilot, because it is slow enough to be controlled, had led to considerable confusion on the subject. It is true that the pilot can control an aircraft with a very rapid divergence (say, time to double amplitude of 2 or 4 seconds) when he has nothing to do but fly the airplane. Therefore, tests made with a rapid divergence where the pilot devoted full attention to flying, or made under conditions such as a landing approach where the pilot necessarily devotes nearly all of his time to flying the airplane, will show that the minimum tolerable time to double amplitude is very low. However, there are many circumstances where the pilot does not, and indeed

cannot, devote all of his attention to flying the airplane. He must read maps, work navigation problems, consult radio facilities handbooks or route maps, tune radios, and carry on various other activities. It is impossible for him to handle these tasks effectively if, every time he diverts his attention, the airplane starts spiralling off. It is perfectly reasonable, then, for pilots to find an airplane with a rapid spiral divergence perfectly flyable yet absolutely intolerable."

In NADC-ED-62-82, a limit of $T_{1/2} > 10$ sec on the degree of spiral stability was recommended primarily from consideration of Cornell Aero Lab TB-1094-F-1 and WADC-TR-59-135. TB-1094-F-1 stated that "the maximum desired spiral stability appears to be a time to half amplitude of 10 seconds." Based on closed-loop analysis, WADC-TR-59-135 suggested that $T_{1/2}$ less than approximately 7 to 14 sec would generally cause a degradation of pilot opinion. If the experimental in-flight data of TB-574-F-6 (figure 109) and TB-1094-F-1 (figure 110) are examined, however, it can be seen that good pilot ratings are obtained for $T_{1/2} = 10$ sec and that the flying qualities do not begin to degrade appreciably until $T_{1/2} = 5$ sec.

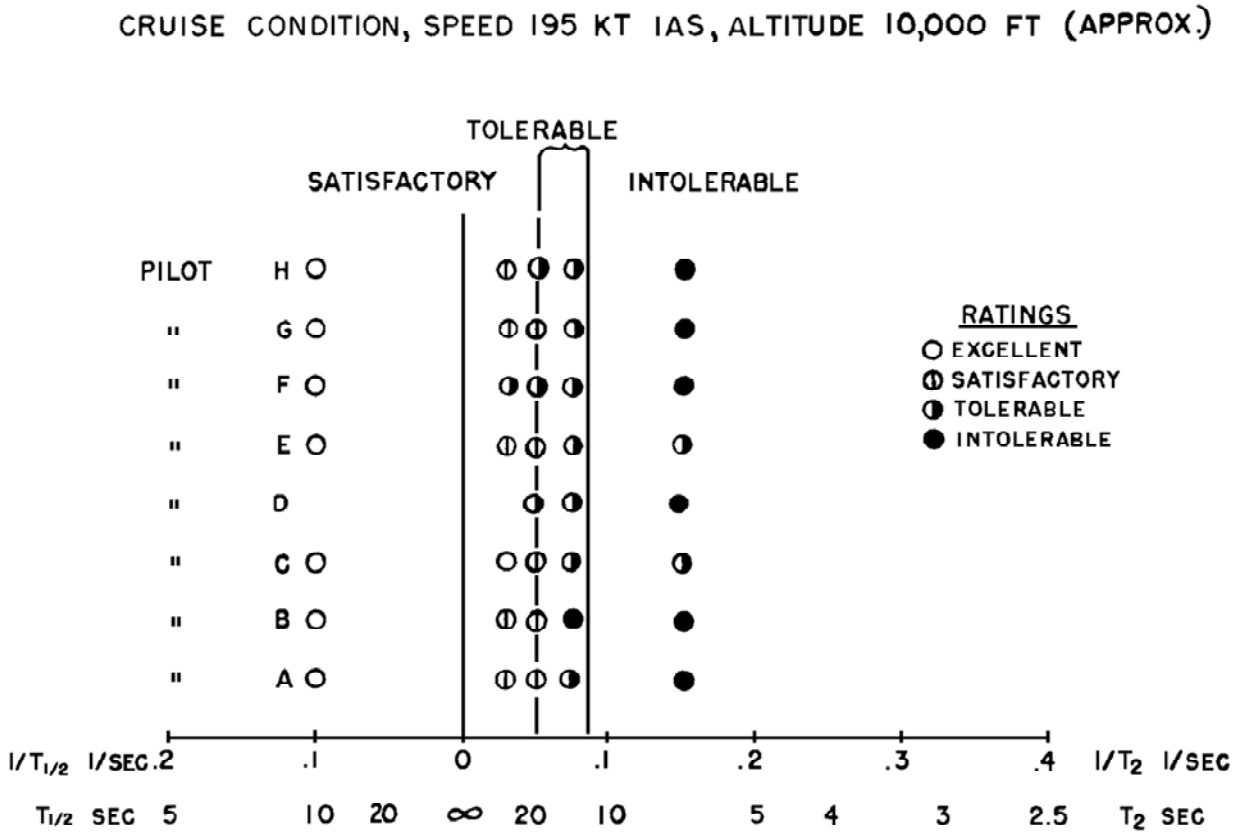


FIGURE 109. Limits of satisfactory and tolerable rates of spiral divergence (from Cornell Aero Lab TB-574-F-6).

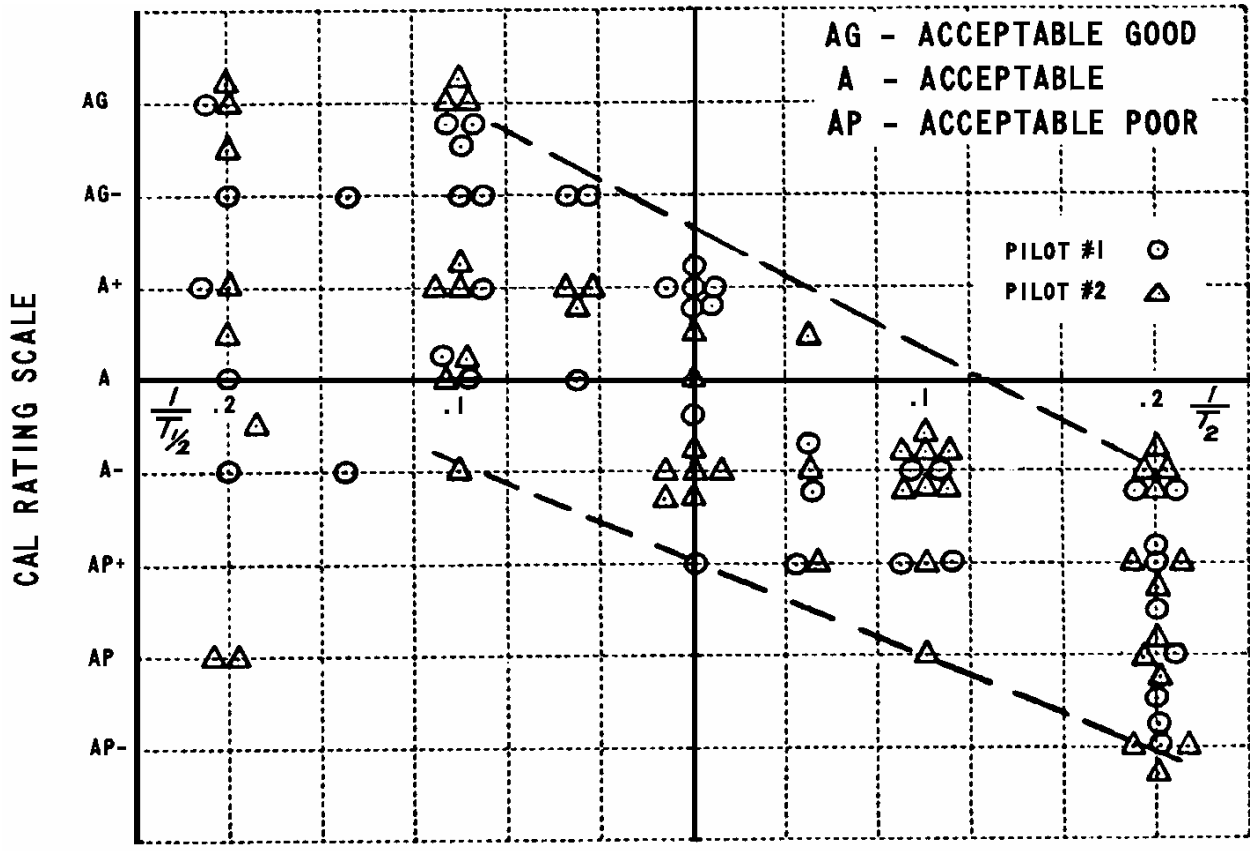


FIGURE 110. Data for all types of flying—pilot opinion versus spiral damping (from Cornell Aero Lab TB-1094-F-1).

Although there are some data that indicate there should be some limit on the degree of positive spiral stability, other data show that strong positive spiral stability can be beneficial. For example, in the program described in Cornell Aero Lab IH-2154-F-1, a wings-leveling device was installed in an aircraft that resulted in an effective highly convergent spiral. Although some pilots commented on the high forces required to hold the airplane in a turn, the flying qualities were considered to be quite acceptable and, in some respects, definitely preferable to neutral spiral stability.

For these reasons, it was decided not to recommend a requirement on positive T_S or $T_{1/2}$ at this time, but instead to recommend requirements on other factors associated with convergent spirals. That is, the limit on aileron forces in turns and the required roll maneuverability will effectively limit $T_{1/2}$.

It should be noted that the spiral requirements include the effect of lateral trim change with speed as well as the constant-speed spiral stability characteristics, since this is more representative of what the pilot sees than are constant-speed stability effects alone.

VERIFICATION

Because of the low frequency of the spiral mode its effects are easily masked by residual rolling moments, e.g., due to asymmetric loadings or control system friction. Additionally, if tests are conducted with the pitch control free, the resulting phugoid oscillation may alter the rate of spiral divergence. Values of equivalent spiral time constant can be obtained from the equivalent systems fit of the ϕ/F_{as} (or ϕ/δ_{as}) response described in 5.2.3.1. To include the effect of airspeed though, would require simultaneously considering the longitudinal degrees of freedom.

The very low frequency typical of the spiral mode makes the spiral time constant difficult to measure by dynamic parameter identification techniques. Bearing in mind that friction and asymmetric loading must be accounted for, no special flight test techniques are required.

5.2.3.1.4 Coupled roll-spiral oscillation.
The aircraft shall not exhibit a coupled roll-spiral mode in Category A Flight Phases. A coupled roll-spiral mode may be permitted for Category B and C Flight Phases provided that the product of ζ_{RS} and ω_{RS} exceeds the values in table XLVII. The aircraft shall not exhibit a coupled roll-spiral oscillation in Category C Flight Phases requiring rapid turning maneuvers such as short approaches.

TABLE XLVII. Minimum values for roll-spiral damping coefficient, $\zeta_{RS}\omega_{RS}$.

Level	$\zeta_{RS}\,\omega_{RS}$ (rad/sec)
1	
2	
3	

REQUIREMENT RATIONALE

Existence of a coupled roll-spiral oscillation (also called a lateral phugoid) results in poor bank angle control and excessive lateral stability for maneuvering flight. However, for rather benign flight conditions, as in conventional approach and landing, some roll attitude (or integral of p dt) stabilization has been found helpful. The requirement prohibits the degree of coupling which has been found deleterious.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.1.4
MIL-STD-1797: 4.5.1.3.

Recommended values for table XLVII:

Level	$\zeta_{RS}\,\omega_{RS}$ (rad/sec)
1	0.5
2	0.3
3	0.15

REQUIREMENT LESSONS LEARNED

The primary objections to a coupled roll-spiral mode are the lack of roll control effectiveness and high steady forces (more or less proportional to bank angle). In turning flight (see AFFDL-TR-72-41, pg 138) roll-spiral coupling can arise from unusual values of L'_β, L'_p, N'_p, and N'_r or from augmentation which employs bank angle feedback (see Aircraft Dynamics and Automatic Control). Experience with bank angle command systems has shown that in order to obtain reasonable control sensitivities, the control authority must be very low. One source of experience was a V/STOL control system blending study conducted on a moving-base simulator (NADC-77052-30). In that study it was clear that while a bank angle command system was desirable in hover, it was also desirable to switch to a rate command system at very low airspeeds. As with the AFFDL-TR-72-41 comments, the pilots found that the lack of roll control authority and high steady forces in the turns were unacceptable. For hands-off steady flight, some autopilots use a wings-leveler to good effect.

In cases where the coupled roll-spiral results from unusually large values of L'_β, N'_r, and N'_p / L'_p, a low L'_p may result in controllability problems. This was the case for the M2-F2 lifting body (NASA TN D-6496).

NASA CR-778, AFFDL-TR-65-39, and NASA TN D-5466 contain results of simulations involving coupled roll-spiral modes. In all cases the longitudinal characteristics of the baseline vehicle were rated Level 1 by the evaluation pilots. In addition, all lateral phugoid cases were characteristic of lifting bodies: large effective dihedral, low roll damping, and positive yaw acceleration due to roll rate.

The data of NASA CR-778 are for an in-flight simulation of a reentry vehicle using the Calspan variable-stability NT-33. Figure 111, figure 112, and figure 113 present results of the simulation utilizing spiral descent and landing approach (figure 111) and up-and-away flight (figure 112 and figure 113) maneuvers. Four pilots evaluated the configurations and inter-pilot variations in ratings were small for most cases. Figure 111 shows that in smooth air and with slight proverse yaw due to ailerons, $(\omega_\phi/\omega_d)^2 = 1.344$, a lateral phugoid is acceptable but unsatisfactory (Ratings were based on the 10-point CAL scale). Ratings degrade quickly in turbulence and as $(\omega_\phi/\omega_d)^2$ becomes much less than or greater than 1.0. Figure 113 shows poor ratings for all configurations; this may be due to the large value of $|\phi/\beta|_d$ (= 8.58). However, in general, in light

or less turbulence and with low yaw due to ailerons, a lateral phugoid mode is not shown to be objectionable (note that for figure 112 and figure 113, ζ_d is Level 2 in value) for low-demand (Category B) tasks.

The ground simulation of NASA TN D-5466 involved cruise and low-speed conditions, including several ILS approaches. These also, with the exception of the ILS approaches, are Category B maneuvers. However, the approach ratings were reported to correlate with the low-speed (general all-around flying) ratings, so all of these could be considered to be Category C data. Results are shown on figure 114. The boundaries of table XLVII are shown for comparison. In general, though the Dutch roll characteristics (ζ_d, ω_d, and $|\phi/\beta|_d$) and adverse aileron $[(\omega_\phi/\omega_d)^2]$ are varied, the data show definite trends with ζ_{RS} and ω_{RS}. They are also in agreement with the in-flight data of figure 111, figure 112, and figure 113, but they do not show strong support for table XLVII boundaries. Instead, they indicate that something like the following minimum values of the product $\zeta_{RS}\ \omega_{RS}$ would be more appropriate:

Level 1: $\zeta_{RS}\ \omega_{RS} > 0.5$

Level 2: $\zeta_{RS}\ \omega_{RS} > 0.07$

Level 3: $\zeta_{RS}\ \omega_{RS} > 0.0$

Quantitatively, correlation with the boundaries would jump from about 15% to almost 80%. However, the lack of any pilot commentary or of detailed descriptions of the piloting tasks, somewhat reduces the credibility of the data. In addition, AFFDL-TR-65-39 has data which disagrees entirely with both NASA TN D-5466 and NASA CR-778.

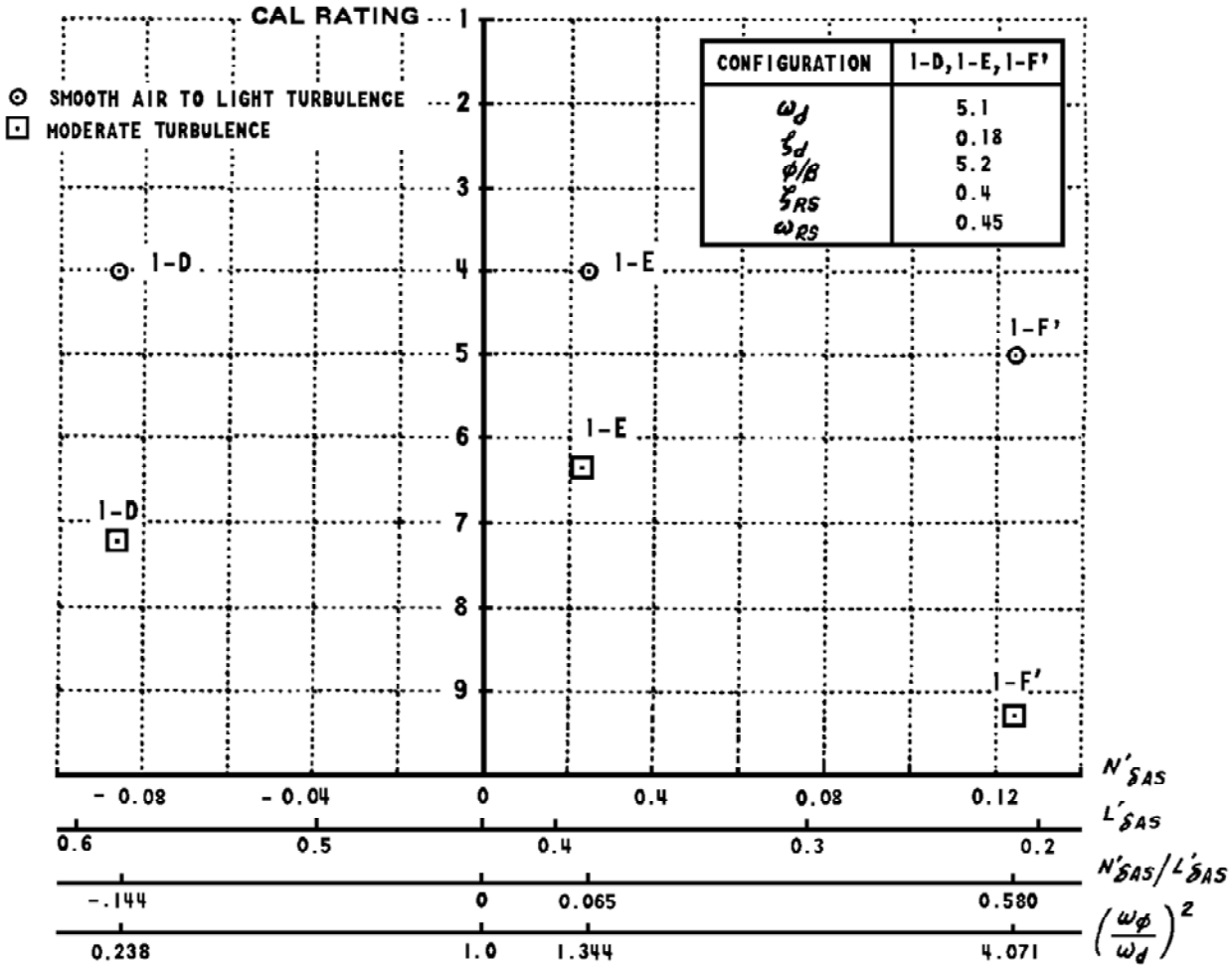


FIGURE 111. Composite pilot ratings for spiral descent of simulated reentry vehicle (from NASA CR-778).

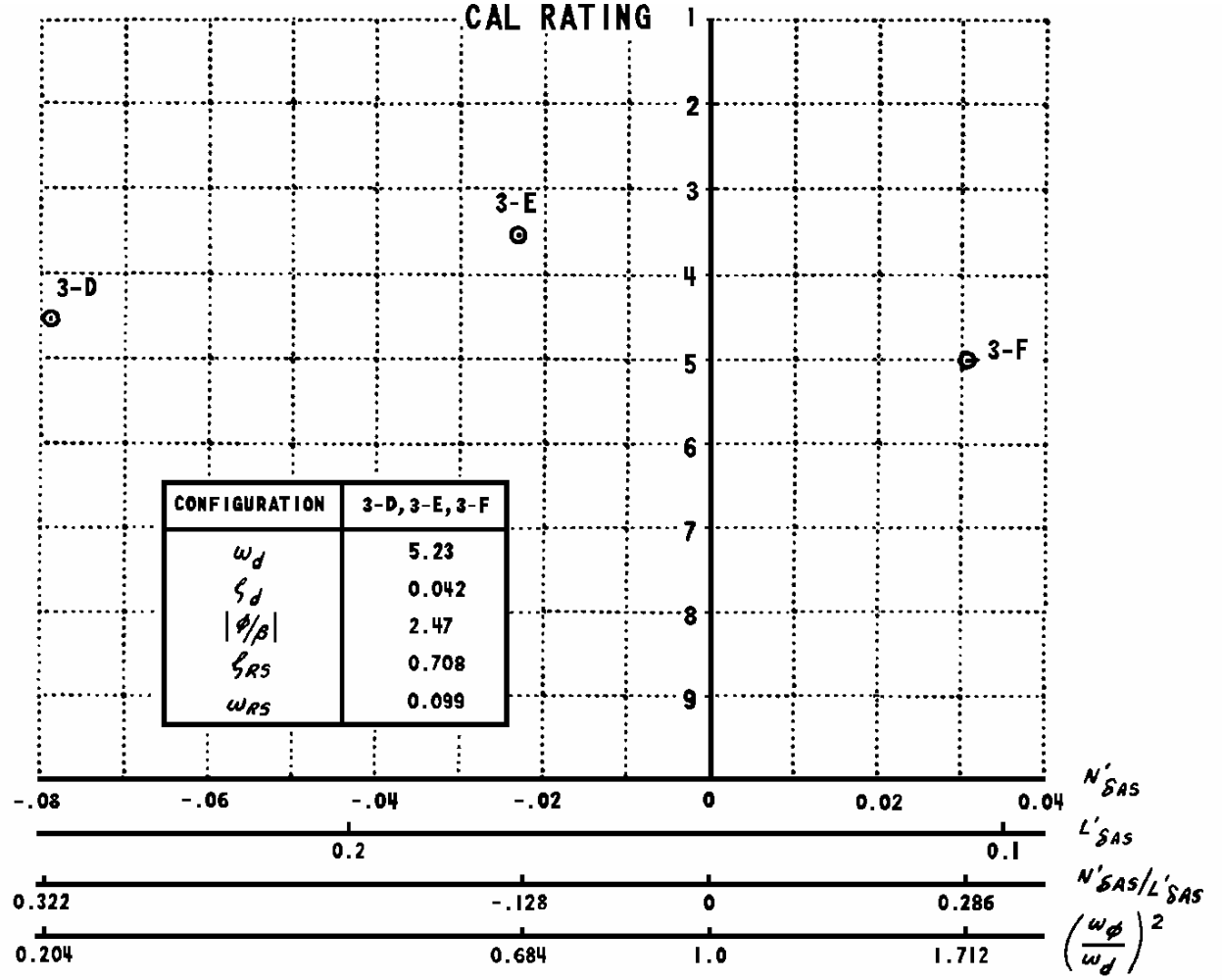


FIGURE 112. Composite pilot ratings for up-and-away flight; moderate $|\phi/\beta|_d$ (from NASA CR-778).

The ground simulation of AFFDL-TR-65-39 shows a much more pessimistic view of the lateral phugoid (figure 115). Even the best of the configurations was rated no better than 5 (CAL scale), and almost all were unflyable (PR of 10). In this simulation both open- and closed-loop pilot tasks were included. The closed-loop maneuvers considered climbing, diving, and level turns and both slow and rapid entries into 30º and 60º banks. Both smooth and simulated rough air were used. The open-loop tasks required the pilots copy a standard IFR clearance.

It is not clear why the data of AFFDL-TR-65-39 differ so dramatically from both the ground simulation of NASA TN D-5466 and the flight data of NASA CR-778. The data of AFFDL-TR-65-39 using real roll and spiral modes are also in disagreement with other such data, see figure 104. It is possible the high ratio of $|\phi/\beta|_d$ (ranging from 6.1 to 26.5) caused the degradation; from AFFDL-TR-65-39:

> "The most obvious conclusion is that the complex roll-spiral mode configurations that were investigated represent poor to very bad tactical aircraft, primarily because of the lack of roll damping and the resultant rolly characteristics."

However, NASA CR-778 also contains some high $|\phi/\beta|_d$ cases (figure 113), and for $(\omega_\phi/\omega_d)^2$ near 1.0 the average rating was 5 ($\zeta_{RS}\,\omega_{RS} = 0.057$). The details of the simulated turbulence used in AFFDL-TR-65-39 are not known; but it is possible that this had a major effect on the ratings (see figure 126 and AFFDL-TR-67-2).

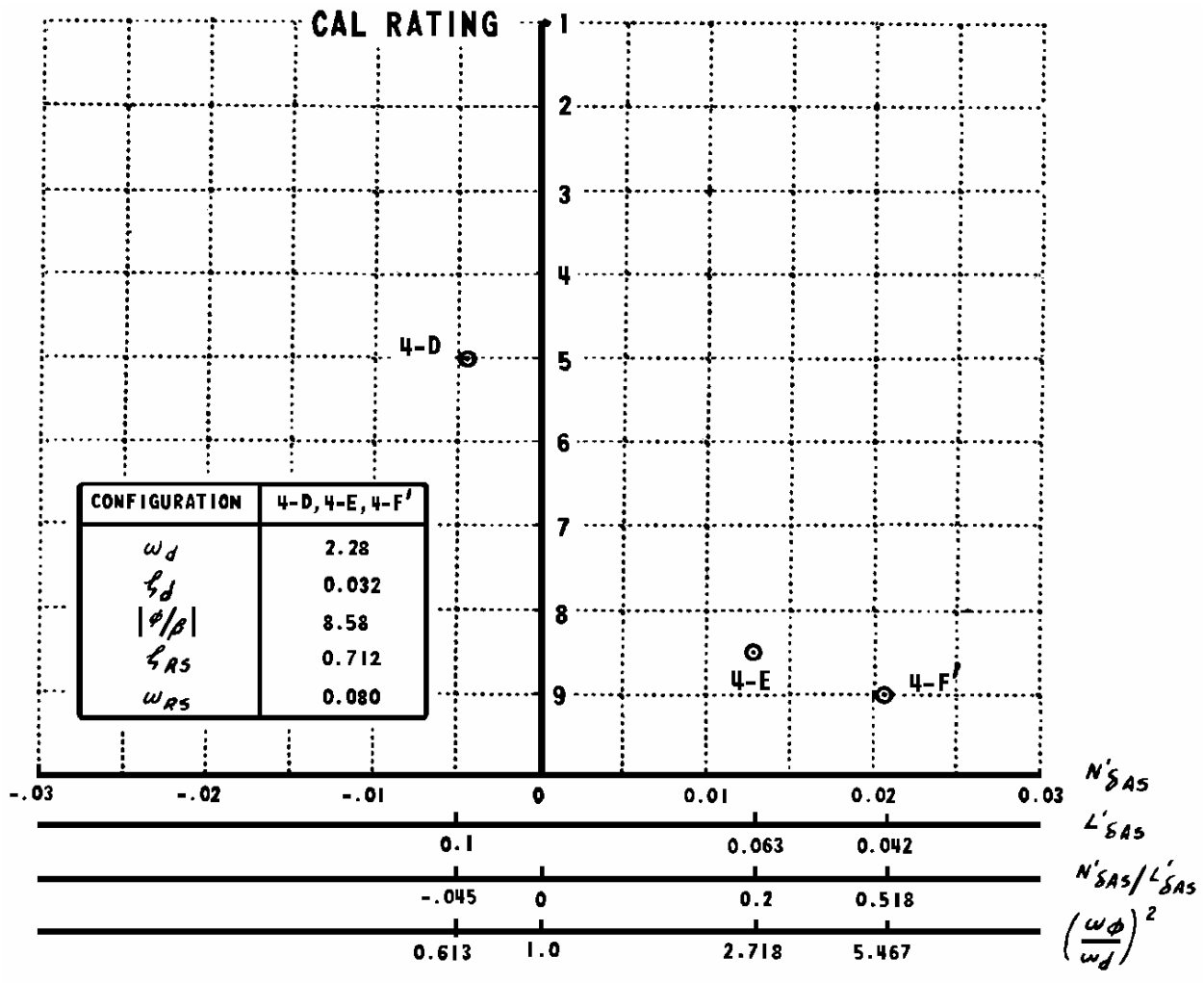


FIGURE 113. Composite pilot ratings for up-and-away flight; large $|\phi/\beta|_d$ (from NASA CR-778).

Experience with the M2-F2 lifting body (NASA TN D-6496) shows support for a strict requirement on the lateral phugoid and illustrates the insidious nature of the lateral phugoid mode. Figure 116 shows the variation in ζ_{RS} and ω_{RS} for the unaugmented and augmented M2-F2. In the actual vehicle, a second-order washout mode occurred, through p and r feedbacks, but it was near in frequency and damping to the lateral phugoid. Low-frequency washout zeroes

effectively cancel one of the modes so that the vehicle essentially behaves like a classical coupled roll-spiral configuration. In gliding landing tests of the M2-F2, energy management required flight at negative angles of attack. On numerous flights the M2-F2 entered divergent lateral-directional oscillations which were stopped only by pulling back to positive angles of attack. The analysis of NASA TN D-6496 showed the oscillations to be due to the coupled roll-spiral mode. Time histories in NASA TN D-6496 suggest that the M2-F2 became uncontrollable at an angle of attack of -2º; this coincides (figure 116) with $\zeta_{RS}\ \omega_{RS} = 0$. Addition of a center fin (the M2-F3) improved primarily the yawing characteristics of the vehicle. As figure 116 shows, even at large negative angles of attack the M2-F3 lateral phugoid mode is still stable. In fact, the roll and spiral modes are uncoupled ($\zeta_{RS} > 1$) until $\alpha = 2º$. Flight tests of the M2-F3 (SAS on) supported the prediction of good lateral flying qualities at negative angles of attack.

VERIFICATION

Analysis based on wind tunnel estimates can predict potential areas in which a lateral phugoid might exist; flight testing shall be concentrated in those regions. Starting from the complete lateral and directional transfer functions, it is straightforward to identify that the roll and spiral modes are coupled. That is, the two usual first-order terms in the denominator are combined into a second-order mode that identifies the roll-spiral damping. Flight and simulator evaluation tasks shall involve as rapid maneuvering as could be expected in the Flight Phase, in smooth and turbulent air. Existence of a coupled roll-spiral mode shall be allowed only during those flight conditions which do not require rapid bank angle maneuvering.

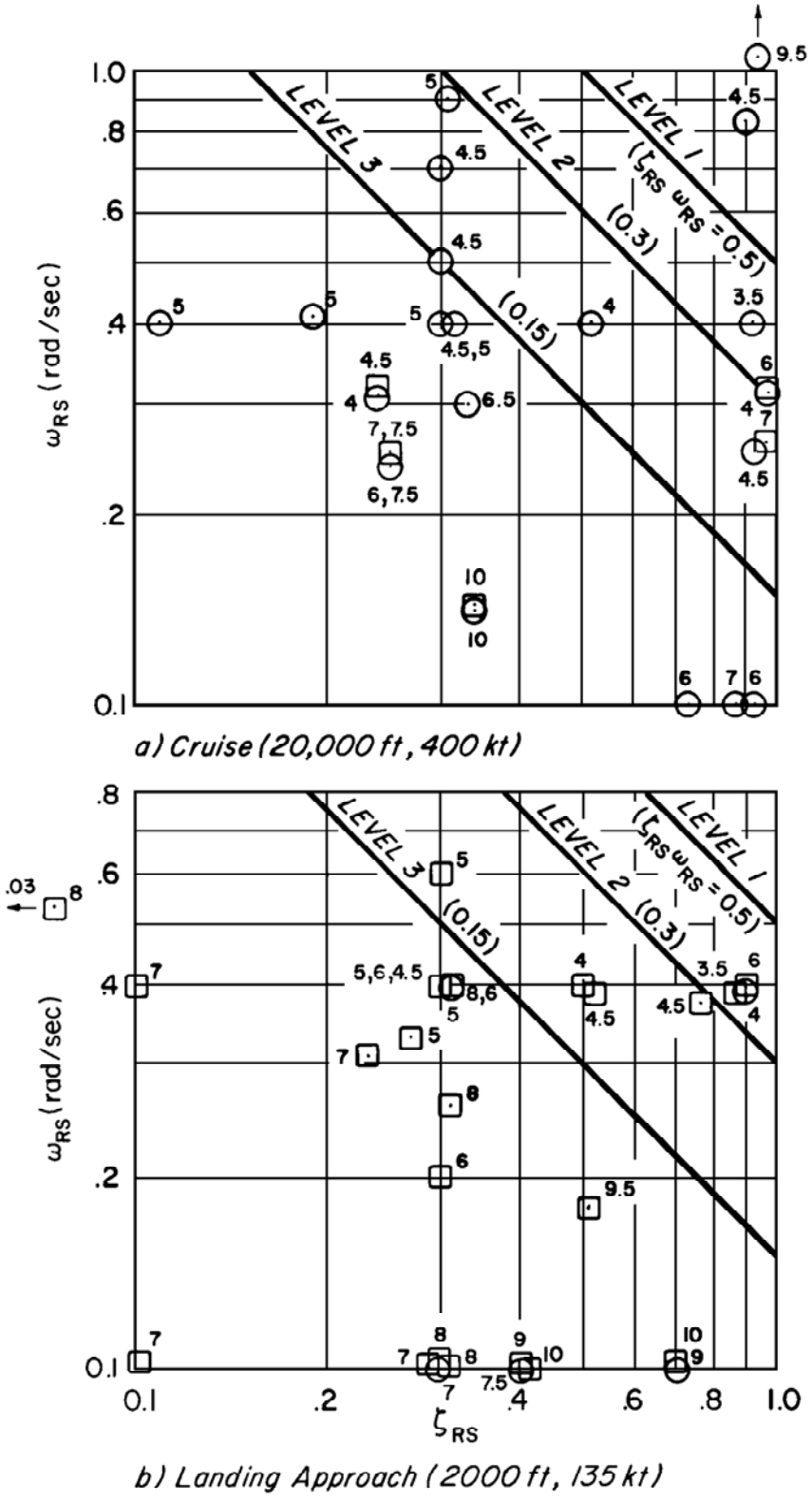


FIGURE 114. Pilot ratings for ground-simulation of NASA TN D-5466 (Dutch roll characteristics vary).

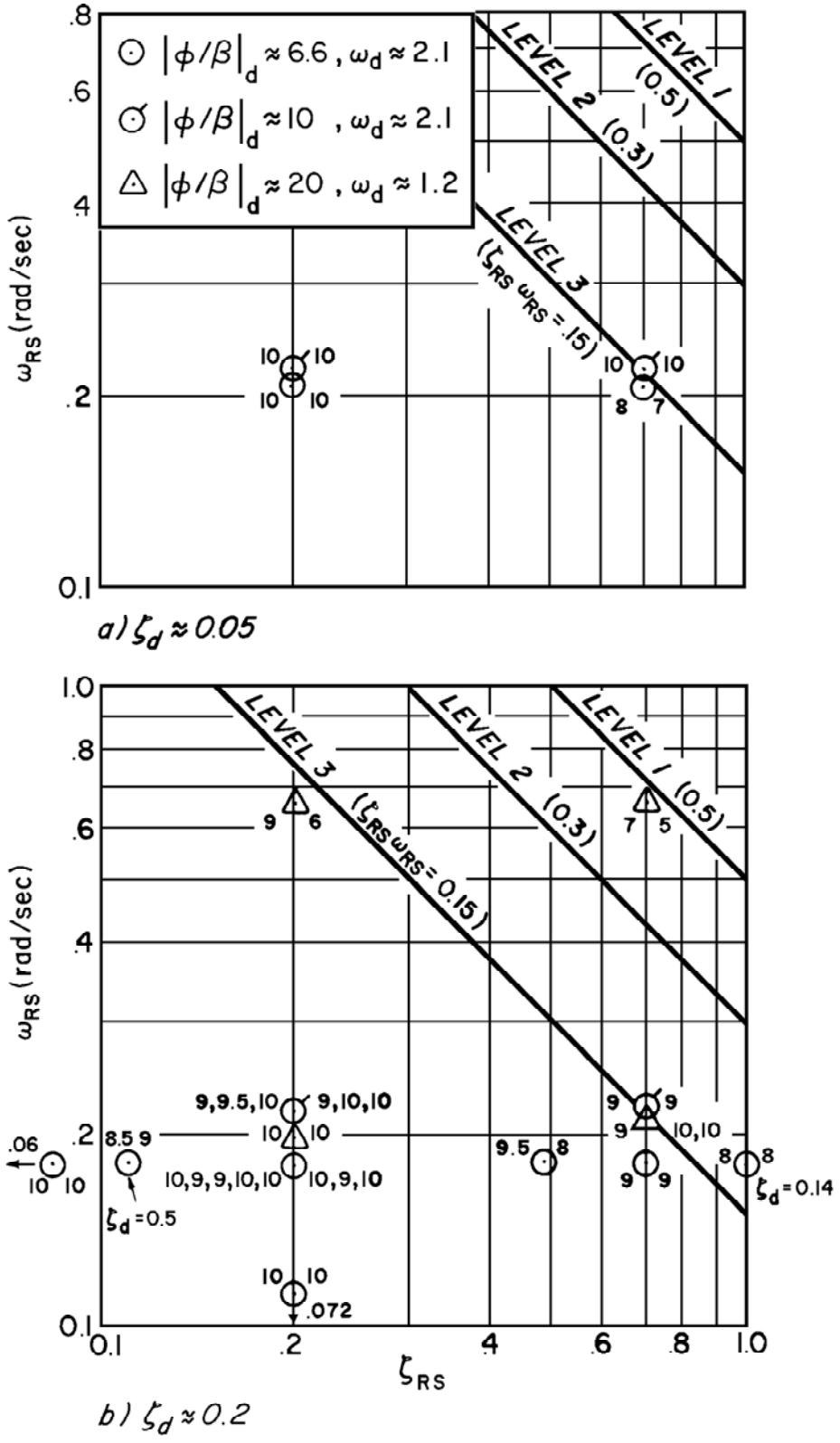


FIGURE 115. Pilot ratings for ground simulation of AFFDL-TR-65-39 [$(\omega_\phi/\omega_d)^2$ = 0.64-1.10].

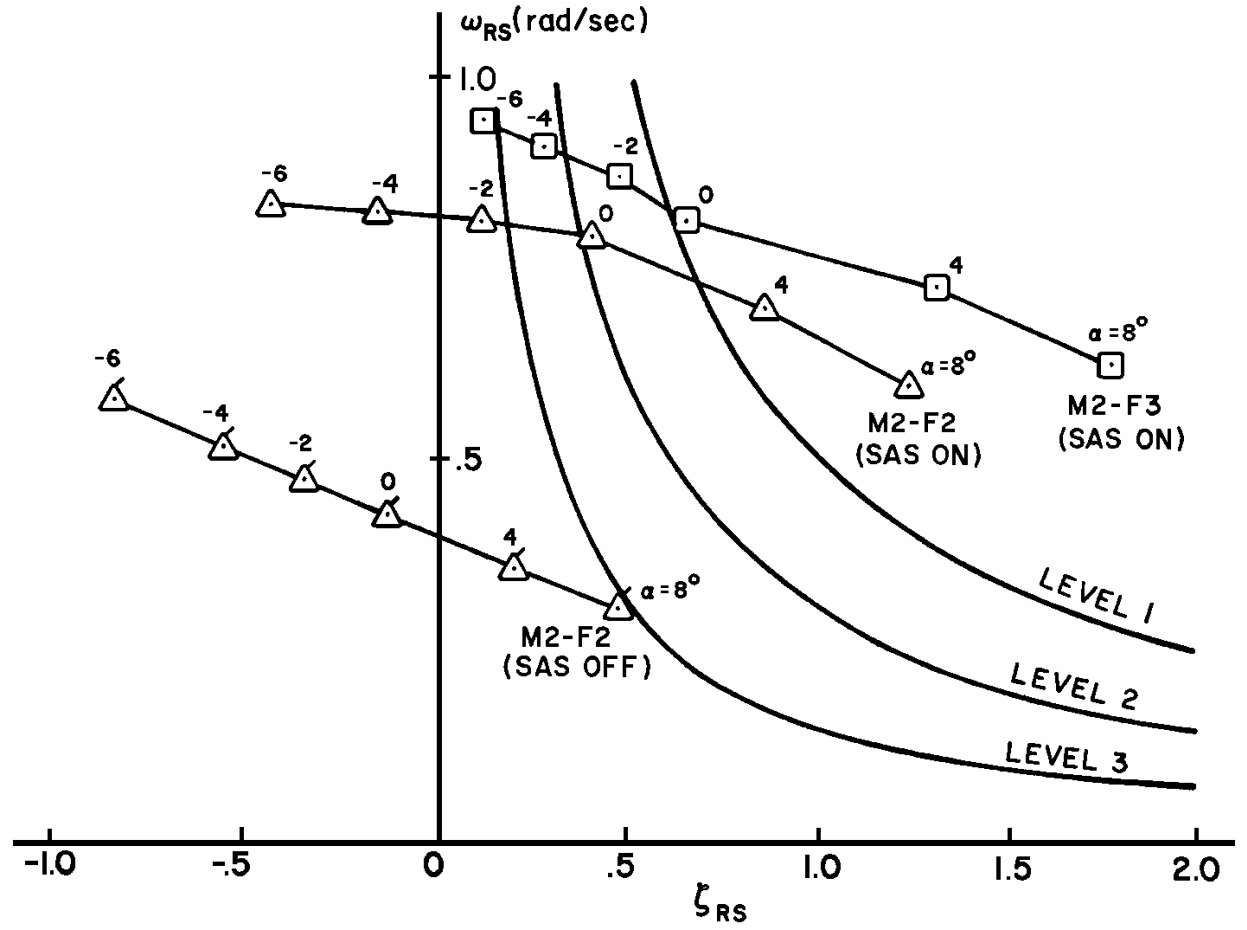


FIGURE 116. **Coupled roll-spiral mode characteristics for M2-F2 and M2-F3 lifting bodies (from NASA TN D-6496).**

5.2.3.1.5 Roll time delay.
The equivalent roll time delay, τ_{ϕ_e}, shall not exceed the values in table XLVIII.

TABLE XLVIII. Maximum allowable equivalent roll time delay.

Level	Equivalent Roll Time Delay
1	
2	
3	

REQUIREMENT RATIONALE

This requirement is intended to insure that combined delay contributions of prefilters, stability augmentation, servos, etc., does not degrade the pilot's tracking capability in roll control.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.5.3
MIL-STD-1797: 4.5.1.5.

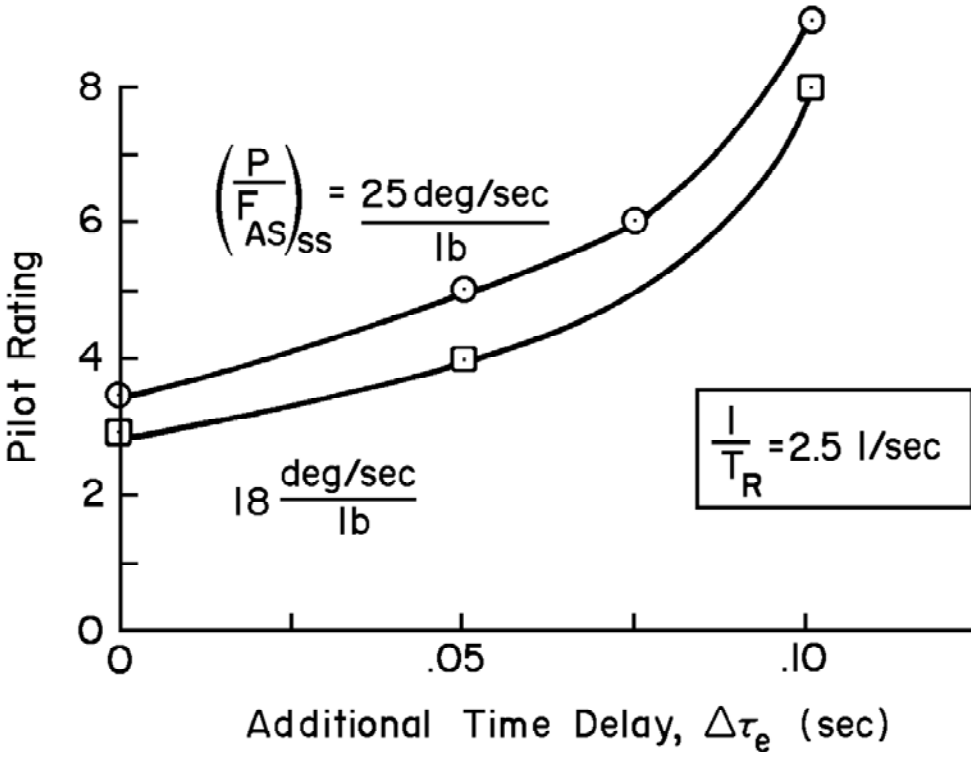
Recommended values for table XLVII:

Level	Equivalent Roll Time Delay
1	0.1 sec
2	0.2 sec
3	0.25 sec

REQUIREMENT LESSONS LEARNED

Based upon extensive research into the effect of time delays in the pitch axis, the important contributions of delays is well known. This requirement extends the time delay limits imposed on the pitch axis in 5.2.2.1.1.2 to the roll axis. For lack of enough substantive data, the limits remain unchanged from MIL-F-8785C. Precise roll tracking, like pitch tracking, is a high-bandwidth task. As more data become available, these limits shall be evaluated and adjusted if necessary.

Little in the way of hard data is available. However, the effect of equivalent time delay was found to be significant in the longitudinal axis (see 5.2.2.1.1.2). This result is seen to extend to the lateral axis based upon the LATHOS study of AFWAL-TR-81-3171 (figure 117). In fact, for the demanding task used in the LATHOS program (air-to-air refueling and bank angle tracking on the HUD) any time delay above the basic NT-33 value resulted in Level 2 ratings. AFWAL-TR-81-3118 reports similarly that for a simulated large transport in approach and landing any increase above the basic (0.12 sec) effective time delay could not support Level 1 ratings (figure 118). Hence, there is evidence to support a need for such a requirement—though not enough to support a change in the requirement.



Note: $\Delta\tau_e$ represents the additional time delay above the basic NT-33 control system value of approximately 0.05 sec.

FIGURE 117. Effect of time delay, LATHOS data (AFWAL-TR-81-3171).

Prefilters can add considerable equivalent time delay to the system. In the longitudinal axis, a first-order lag, $1/T_{LAG}$, of 3 rad/sec adds more than 0.1 sec to overall equivalent time delay, τ_{ϕ_e}. For the NT-33 LATHOS experiment, where basic τ_{ϕ_e} due to actuators was small (0.028 sec), this equivalent time delay was not significant. But on a highly-augmented aircraft, where structural filters, sensor filters, digital time delay, etc., may already contribute considerable lag, a prefilter could make the aircraft totally unacceptable due to excessive time delay. The effect of time delay on pilot rating was considerable in the LATHOS experiment as shown on figure 117.

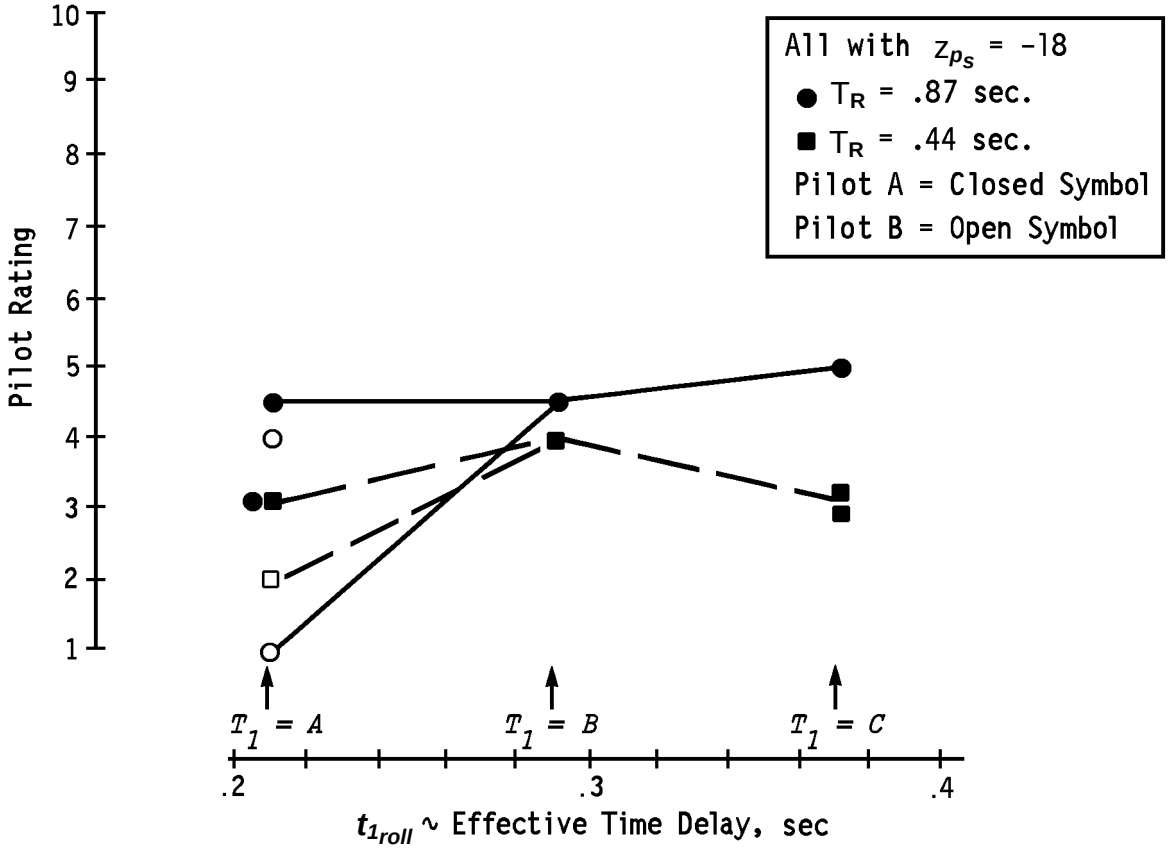


FIGURE 118. Pilot rating versus time delay—lateral-directional.

A control surface rate limit may increase the equivalent time delay or roll mode time constant as a function of the size of command.

VERIFICATION

Appropriate values of τ_{ϕ_e} will require equivalent system matching. See verification for 5.2.3.1, which discusses the use of equivalent system matching of the roll rate response to roll controller to extract T_{S_e}, T_{R_e}, and τ_{ϕ_e}. That section should be consulted for more information on the meaning of τ_{ϕ_e}. In the end, flight test data or a flight-verified analytical model shall be used to verify compliance.

5.2.3.1.6 Yaw time delay.
The equivalent yaw time delay, τ_{β_e}, shall not exceed the values in table XLIX.

TABLE XLIX. Maximum allowable equivalent yaw time delay.

Level	Equivalent Yaw Time Delay
1	
2	
3	

REQUIREMENT RATIONALE

This requirement is intended to insure that combined delay contributions of prefilters, stability augmentation, servos, etc., does not degrade the pilot's yaw control.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.5.3
MIL-STD-1797: No corresponding requirement.

Recommended values for table XLIX:

Level	Equivalent Yaw Time Delay
1	0.1 sec
2	0.2 sec
3	0.25 sec

REQUIREMENT LESSONS LEARNED

Based upon extensive research into the effect of time delays in the pitch axis, the important contributions of delays is well known. This requirement extends the time delay limits imposed on the pitch axis in 5.2.2.1.1.2 to the yaw axis. For lack of enough substantive data, the limits remain unchanged from MIL-F-8785C. Little in the way of hard data is available. As more data become available, these limits shall be evaluated and adjusted if necessary.

VERIFICATION

Appropriate values of τ_{β_e} will require equivalent system matching. See verification for 5.2.3.1, which discusses the use of equivalent system matching to extract τ_{β_e}. That section should be consulted for more information on the meaning of τ_{β_e}. In the end, flight test data or a flight-verified analytical model shall be used to verify compliance.

5.2.3.2 Lateral-directional dynamic response characteristics.
The lateral-directional time responses shall meet the requirements of 5.2.3.2.1 through 5.2.3.2.8.

REQUIREMENT RATIONALE

The time response design criteria limit characteristics of the roll and yaw responses to pilot commands. This format avoids explicit identification of dominant roots or equivalent system models by working directly with the roll and yaw responses. Nevertheless, by virtue of assuming a conventional-appearing response these criteria do rely on a form of equivalent system. The criteria are applicable to aircraft exhibiting conventional aircraft dynamic modes and to most lateral-directional augmentation systems.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 through 3.3.11.1.2 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.2.2, 3.3.2.2.1, 3.3.2.3, 3.3.2.4, 3.3.2.4.1, and 3.3.4.4
MIL-STD-1797: 4.5.1.4, 4.5.3, 4.5.4, and 4.6.2.

REQUIREMENT LESSONS LEARNED

VERIFICATION

Verification shall be via verification of requirements 5.2.3.2.1 through 5.2.3.2.8. Definitions of the parameters used in these requirements are shown in the time histories presented on figure 119 through figure 121. The figures present p, ϕ, and β time histories following an abrupt aileron stick input for four different zero locations of the $p/\delta_{as}|_{step}$ transfer function. These figures present examples of measurement of ψ_β, $\angle p/\beta$, p_{osc}/p_{av}, and $\Delta\beta_{max}$ for both right and left rolls. figure 119 shows the four zero locations, figure 120a through figure 120d show the time histories for right rolls, and figure 121a through figure 121d show the time histories for left rolls. Techniques for measuring the parameters used in these requirements are presented in Appendices VB and VC in AFFDL-TR-69-72.

Since several oscillations of the Dutch roll are required to measure these parameters, and since, for proper identification, large roll rates and bank angles changes must generally be avoided, step

roll control inputs should be small. It should be noted that since ψ_β is the phase angle of the Dutch roll component of sideslip, care must be taken to select a peak far enough downstream that the position of the peak is not influenced by the roll mode. In practice, peaks occuring one or two roll mode time constants after the aileron input will be relatively undistorted. Care must also be taken when there is ramping of the sideslip trace, since ramping will displace the position of a peak of the trace from the corresponding peak of the Dutch roll component. In practice, the peaks of the Dutch roll component of sideslip are located by first drawing a line through the ramping portion of the sideslip trace and then noting the times at which the vertical distance between the line and the sideslip trace is the greatest. (See figure 122 for an enlarged section of figure 120a).

Since the first local maximum of the Dutch roll component of the sideslip response occurs at t = 2.95 sec,

$$\psi_\beta = -\frac{360}{T_d} t_{n_\beta} + (n-1)\,360 = -\frac{360}{3.5}(2.95) = -303\ ^\circ$$

For example, if the roll performance requirement is $\phi_t = 30^\circ$ in 1 sec with rudder pedals free (as in the rolls of 5.2.3.5), then from the definitions, "k" for this condition is

$$k = \frac{(\phi_1)_{command}}{(\phi_1)_{requirement}} .$$

Therefore, from figure 120a through figure 121d:

Case (a), k = 9.1/30 = 0.30 Case (b), k = 8.1/30 = 0.27
Case (c), k = 6.8/30 = 0.23 Case (d), k = 6.0/30 = 0.20 .

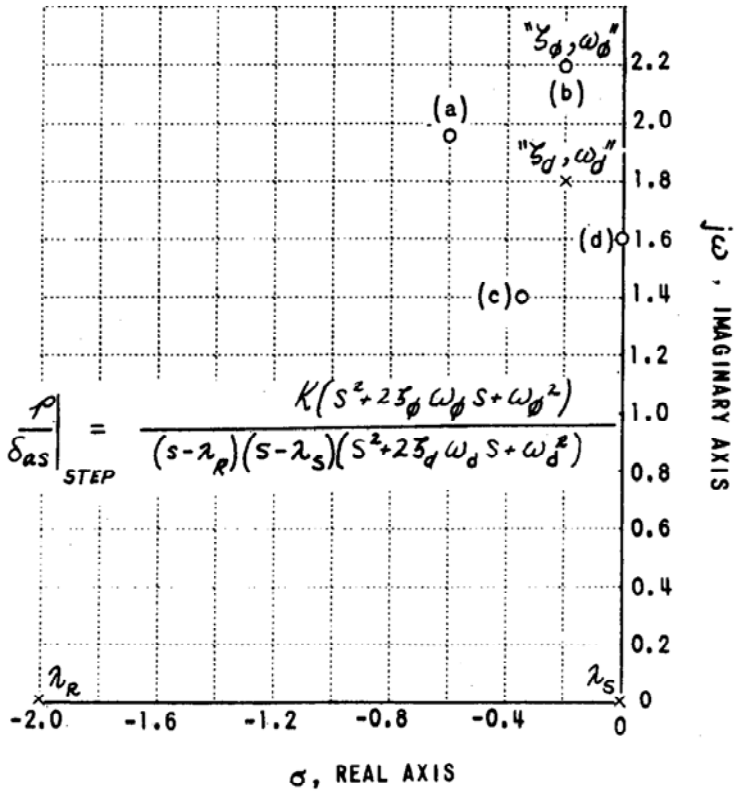


FIGURE 119. Examples of pole and zero locations of the $p/\delta_{as}|_{step}$ response function.

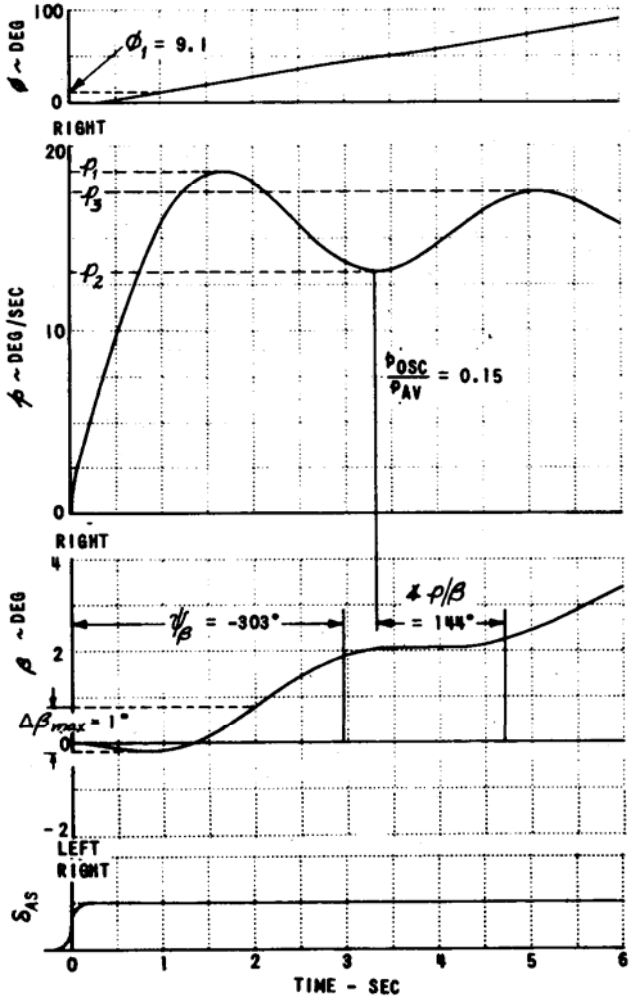


FIGURE 120a. Right roll—zero at position (a) of figure 119.

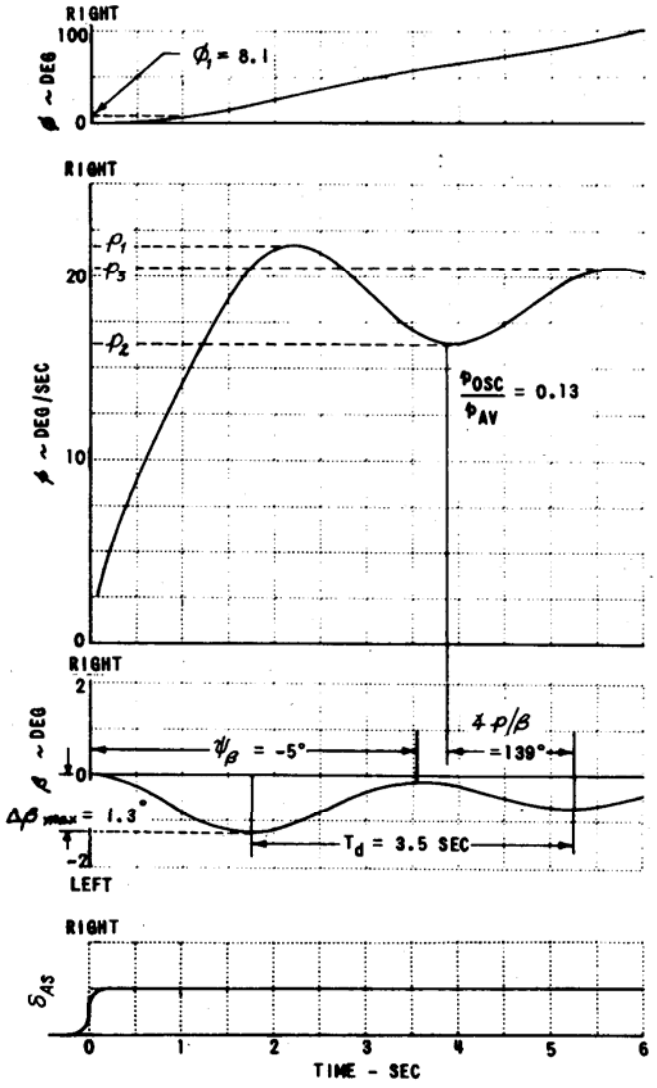


FIGURE 120b. Right roll—zero at position (b) of figure 119.

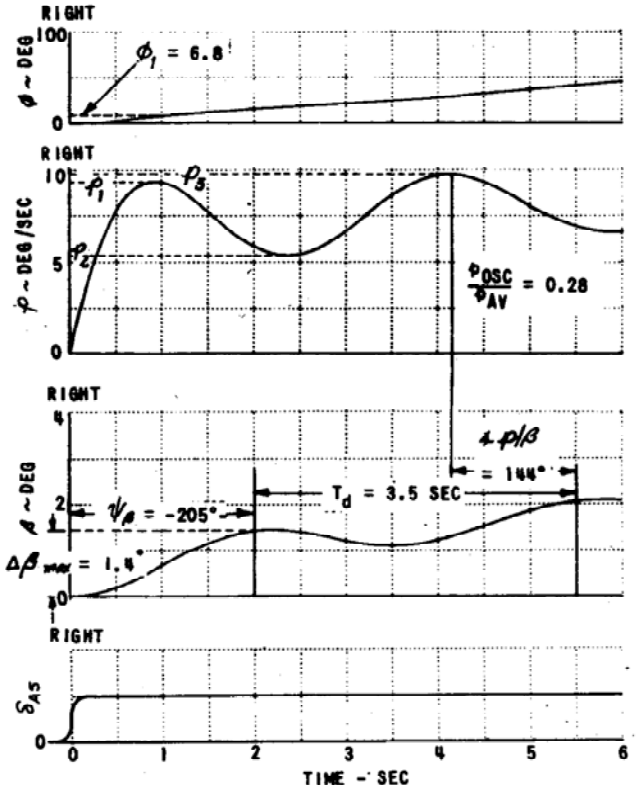


FIGURE 120c. Right roll—zero at position (c) of figure 119.

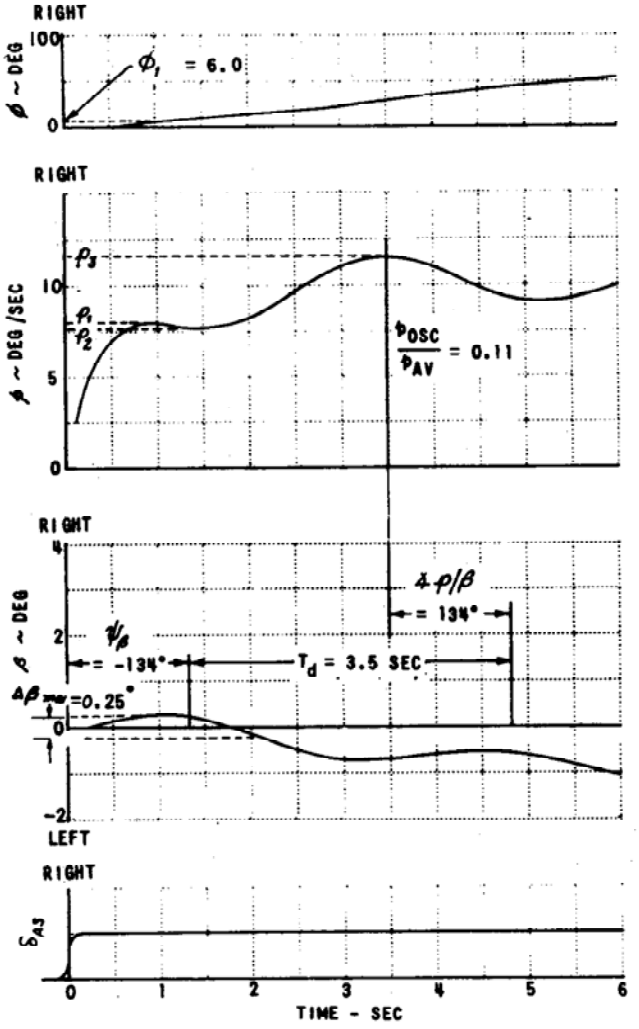


FIGURE 120d. Right roll—zero at position (d) of figure 119.

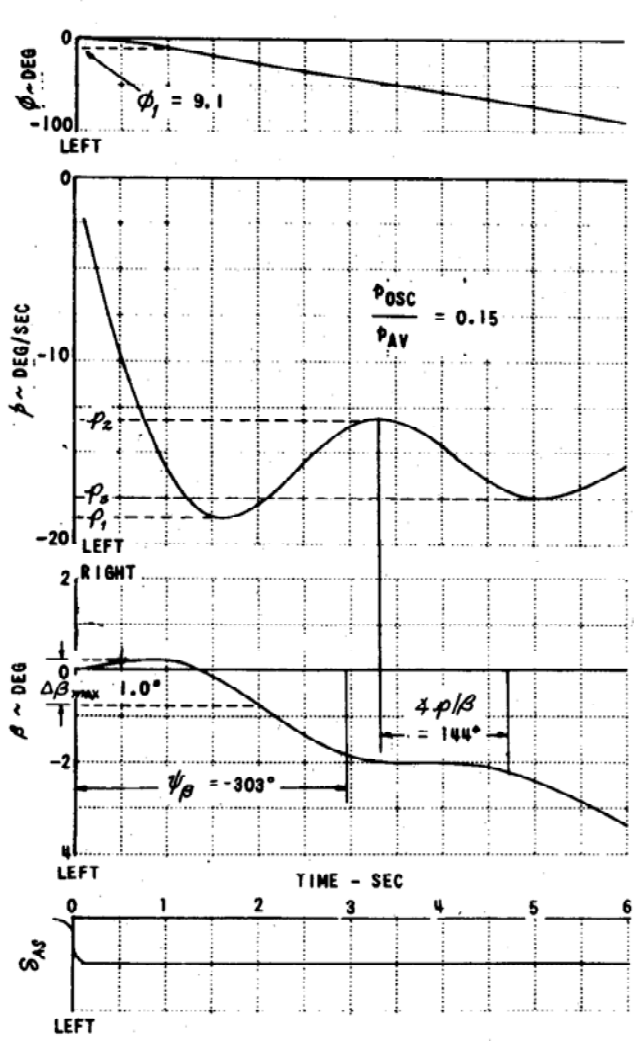


FIGURE 121a. **Left roll—zero at position (a) of figure 119.**

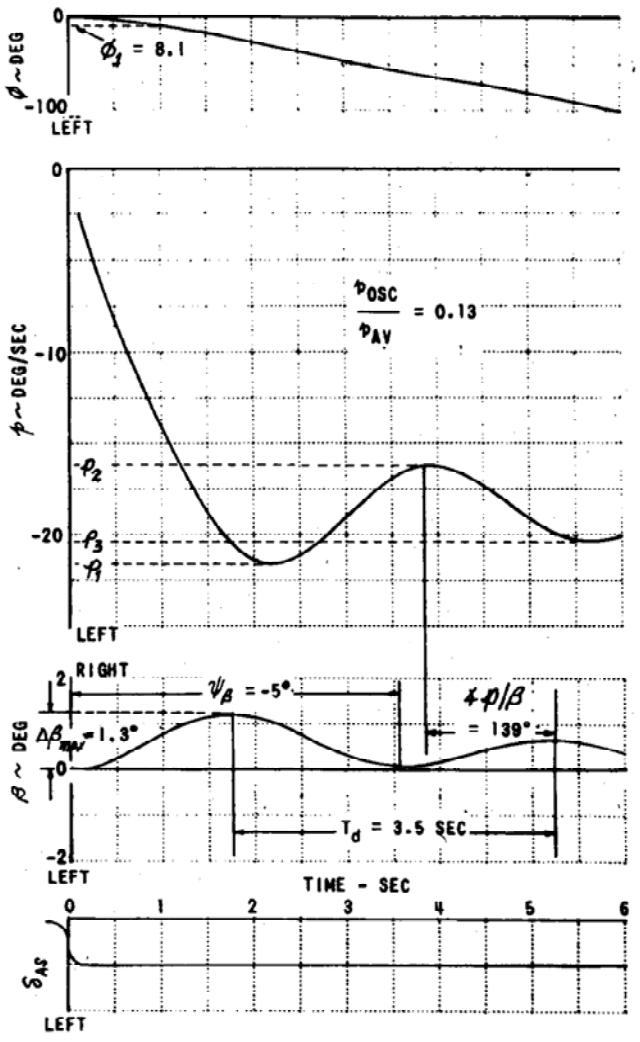


FIGURE 121b. **Left roll—zero at position (b) of figure 119.**

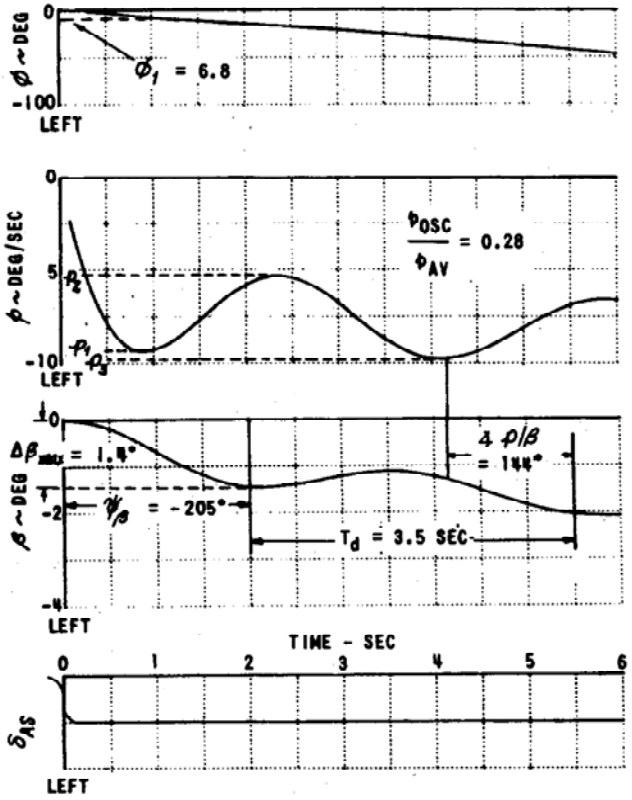


FIGURE 121c. Left roll—zero at position (c) of figure 119.

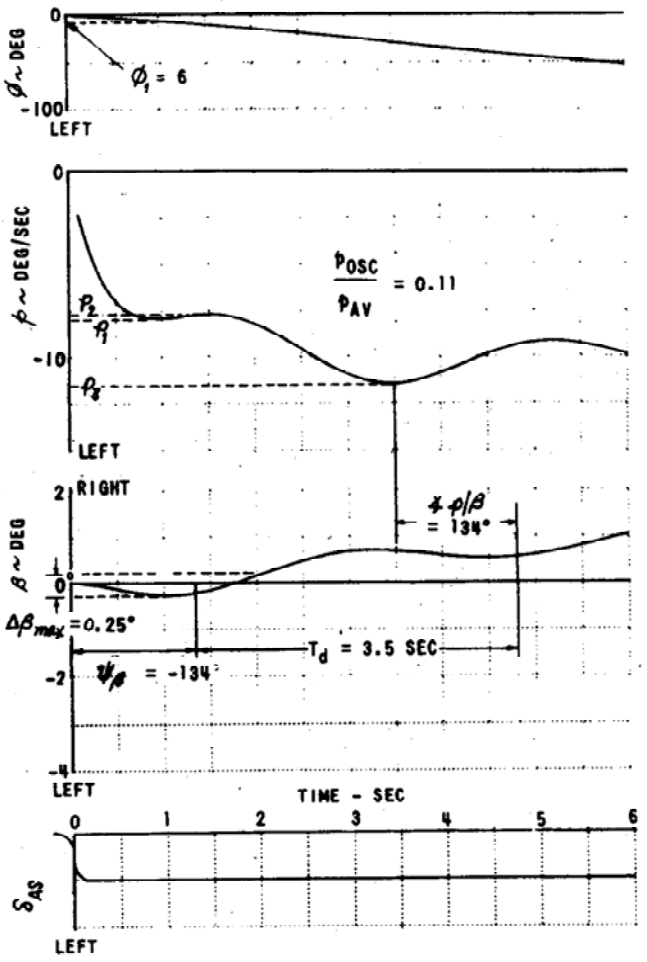


FIGURE 121d. Left roll—zero at position (d) of figure 119.

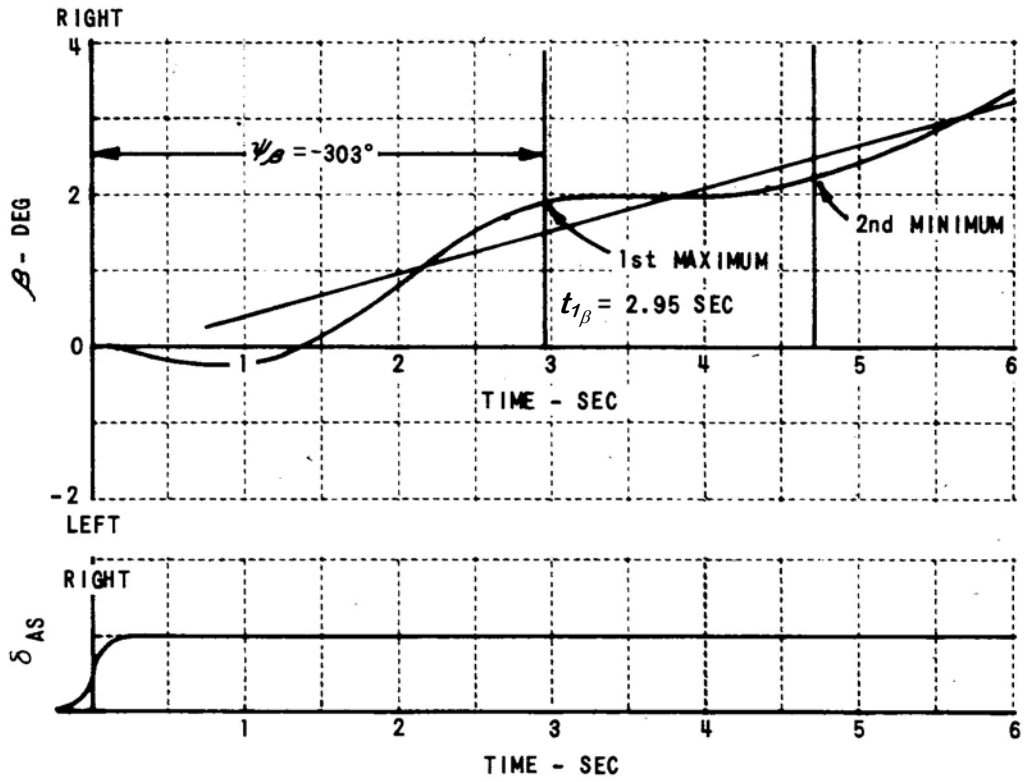


FIGURE 122. **Determining ψ_β from sideslip trace—zero at position (a) of figure 119.**

5.2.3.2.1 Roll rate response to small roll commands.
With yaw controls free, for step roll control commands up to the magnitude which causes a 60º bank angle change in $1.7T_d$ sec, the value of the parameter p_{osc}/p_{av} shall be within __(1)__.

REQUIREMENT RATIONALE

Requirements 5.2.3.2.1, 5.2.3.2.2, and 5.2.3.2.3 are directed at precise control of an aircraft with moderate to high $|\phi/\beta|_d$ response ratios combined with marginally-low Dutch roll damping. Such aircraft exhibit a tendency to develop oscillations in roll rate both open- and closed-loop. This characteristic clearly interferes with the pilot's precision of control and shall be kept to an absolute minimum for all but the most undemanding tasks. These three requirements are stated in different forms in order to catch various magnitudes of control input and response.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.2.2.1
MIL-STD-1797: 4.5.1.4.

Recommended value for Blank (1): "the boundaries of figure 123"

REQUIREMENT LESSONS LEARNED

The parameters p_{osc}/p_{av} and ψ_β are defined in Requirement Verification.

The existence of roll rate oscillations is directly traceable to the relative locations of the ω_{ϕ_e} and ω_{d_e} zeros in the p/F_{as} LOES transfer function:

$$\frac{p}{F_{as}} = \frac{L_{F_{as_e}}\, s\,(s^2 + 2\,\zeta_{\phi_e}\,\omega_{\phi_e}\,s + \omega_{\phi_e}^2)\, e^{-\tau_{\phi_e} s}}{(s + 1/T_{S_e})\,(s + 1/T_{R_e})\,(s^2 + 2\,\zeta_{d_e}\,\omega_{d_e}\,s + \omega_{d_e}^2)}$$

When the complex roots cancel ($\omega_{\phi_e} = \omega_{d_e}$ and $\zeta_{\phi_e} = \zeta_{d_e}$), the Dutch roll mode is not excited at all. When they do not cancel, the Dutch roll contamination occurs primarily in yaw and sideslip if $|\phi/\beta|_d$ is low (say less than 1.5) or primarily in the roll axis when $|\phi/\beta|_d$ is large. As mentioned above, the p_{osc}/p_{av} parameter is directed at cases where $|\phi/\beta|_d$ is large and ζ_{d_e} is low. Note that for all Category A operations the suggested Level 1 value of ζ_{d_e} in 5.2.3.1.2 effectively eliminates the need for this requirement.

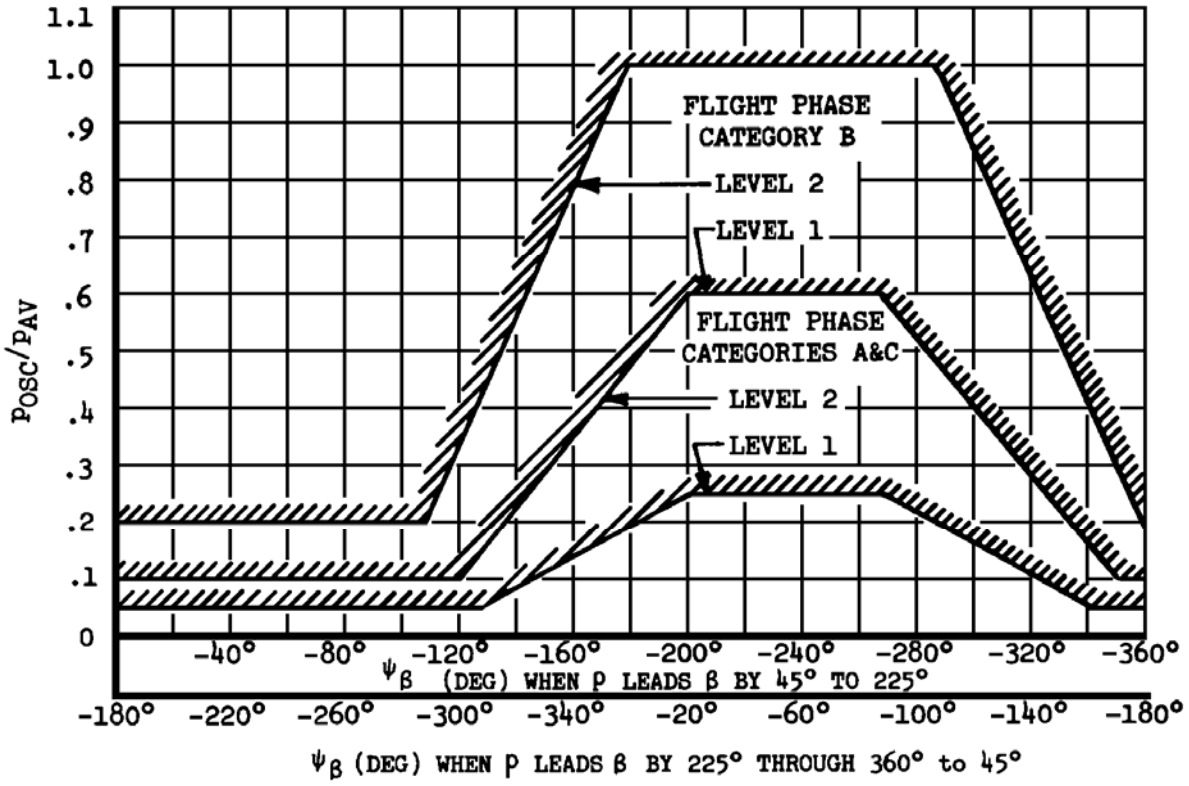


FIGURE 123. Roll rate oscillation limitations.

An extensive description of the derivation of p_{osc}/p_{av} and ψ_β is given in AFFDL-TR-69-72, which should be consulted for further detail. p_{osc}/p_{av} is the ratio of the oscillatory component of roll rate to the average component of roll rate following a rudder-pedals-free step lateral control input. Examples of measurements of p_{osc}/p_{av} are given in the following. The parameter ψ_β is shown in AFFDL-TR-69-72 to be a measure of the relative location of the Dutch roll pole and the ω_ϕ zero. It defines the phasing of the Dutch roll component of sideslip response following a step lateral control input; i.e.,

$$\frac{\beta_d}{F_{as}} = C_d\, e^{-\zeta_d \omega_d t} \cos(\omega_d \sqrt{1-\zeta_d^2}\; t + \psi_\beta)$$

This is illustrated graphically on figure 124 (from AFFDL-TR-69-72).

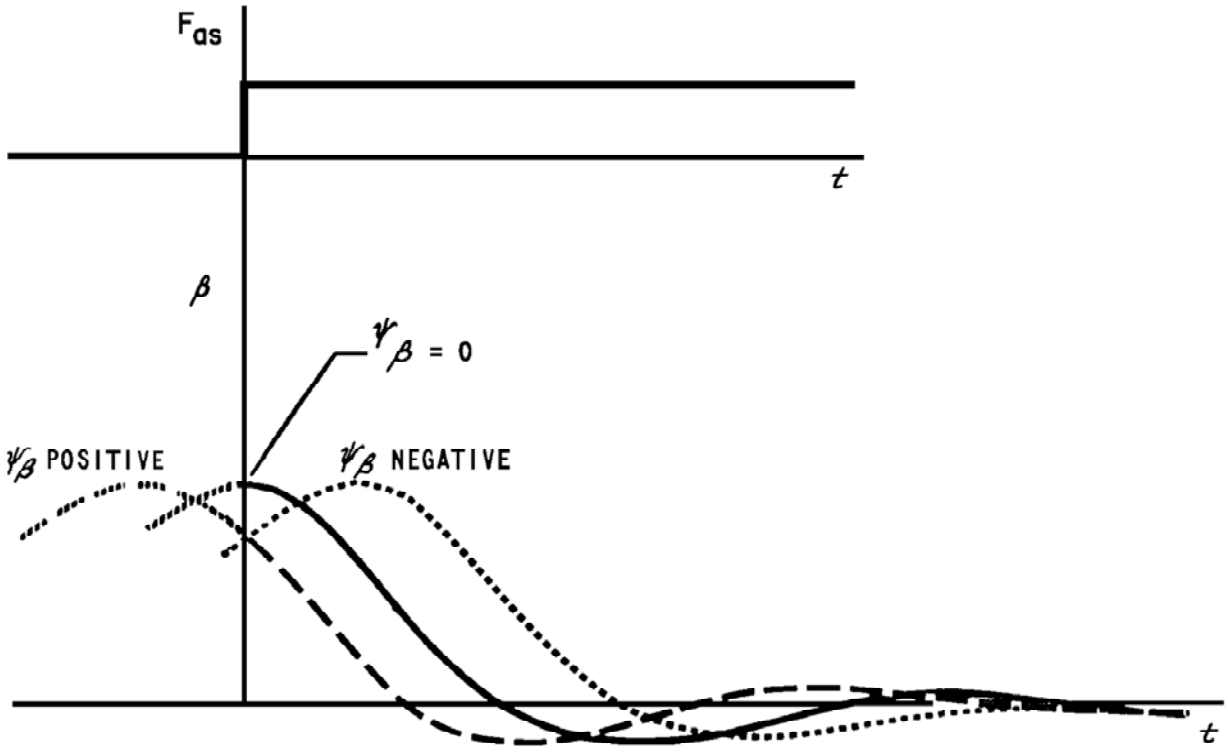


FIGURE 124. Phasing of the Dutch roll component of sideslip response following a step lateral control input.

The parameters p_{osc}/p_{av} and ψ_β have been used to specify the criteria as a function of Flight Phase Category and Level (figure 123). It should be noted that figure 123 has two ψ_β scales, one for positive dihedral (p leads β by 45º to 225º) and the other for negative dihedral (p leads β by 225º through 360º to 45º). Dihedral as used in flying qualities work seems to be an ambiguous and sometimes ill-defined parameter. Here the term refers to the phasing of roll and sideslip motions in Dutch roll mode, $\angle p/\beta$. With positive dihedral, in the Dutch roll oscillation left rolling accompanies right sideslipping and vice versa. In this context, positive dihedral normally means negative $L'_\beta + Y_v\, L'_r \doteq L'_\beta$, where

$$L'_\beta \equiv \frac{L_\beta + N_\beta\, I_{xz}/I_{xx}}{1 - I_{xz}^2/(I_{xx}\, I_{zz})} \ .$$

This is the expected result of positive geometric dihedral, hence the use of the term.

Since ψ_β (the phase angle in a cosine representation of the Dutch roll component of sideslip, negative for a lag) is a rather abstract parameter, it is well to consider it's physical implications and significance to the piloting of an aircraft. Very simply, ψ_β together with $\angle p/\beta$ indicates the oscillatory roll phasing in response to roll commands. It can be seen from figure 123 that the ratio of roll rate oscillation to steady-state roll rate is allowed to be much greater for some values

of ψ_β than for others. Specifically, the limits of p_{osc}/p_{av} for $0 \geq \psi_\beta \geq -90^o$ are far more stringent than for $-180^o \geq \psi_\beta > -270^o$. There are at least two reasons why this is so:

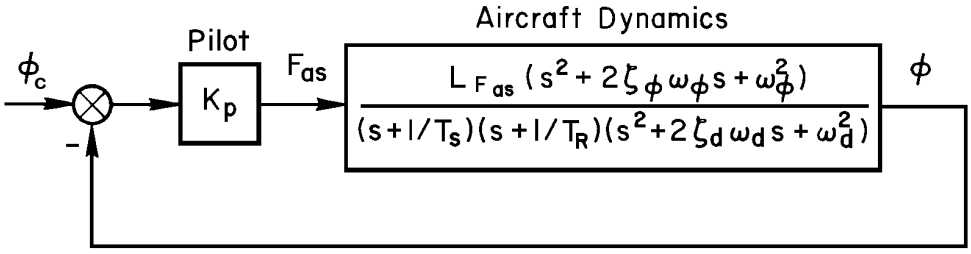
a) Differences in closed-loop stability in piloted-control

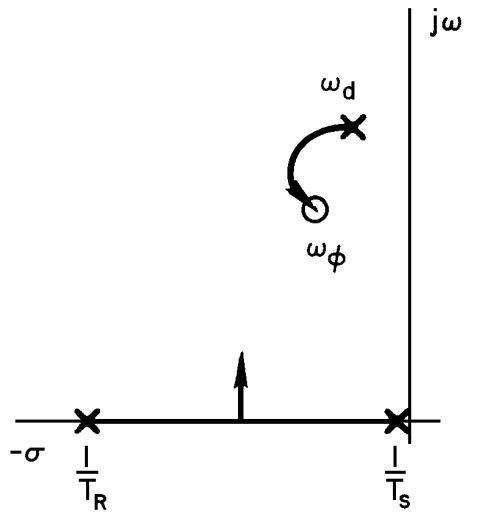
b) Differences in average roll rate.

From the root locus analysis on figure 125a, it can be shown that when the zero of the p/F_{as} transfer function lies in the lower quadrant with respect to the Dutch roll pole (which results in $-180^o \geq \psi_\beta \geq -270^o$), the closed-loop damping increases when the pilot closes a bank angle error to aileron loop. Conversely, it can be shown that when the zero lies in the upper quadrant with respect to the Dutch roll pole ($0 \geq \psi_\beta \geq -90^o$), the closed-loop damping decreases when the pilot applies aileron inputs proportional to bank angle error (figure 125b). In this case a pilot's tolerance of p_{osc}/p_{av} tends to decrease. Finally, when ζ_{d_e} becomes large the effect of the pole-zero location is diminished (figure 125c); i.e., the variation of damping due to the $\omega_{\phi_e}/\omega_{d_e}$ effect is small relative to ζ_{d_e}.

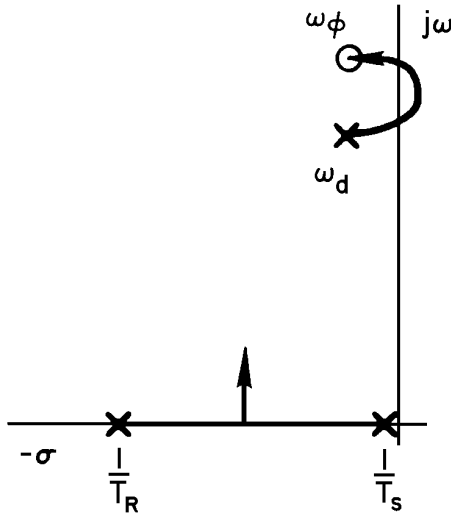
The connection between pole-zero location and the p_{osc}/p_{av} boundaries is shown on figure 126, where the Level 1 and Level 2 boundaries on figure 123 are mapped into ω_{ϕ_e} zero locations for several values of ω_{d_e} and ζ_{d_e}. Note that when ζ_{d_e} meets the Level 1 requirement ($\zeta_{d_e} > 0.19$) the acceptable region for the ω_{ϕ_e} is very large in the region to the left of and below ω_{d_e}. However, there is always a low tolerance for $\omega_{\phi_e} > \omega_{d_e}$ because the closed-loop damping decreases. There is still a relatively tight limit on $\omega_{\phi_e} > \omega_{d_e}$ for ζ_{d_e} of 0.25. This of course reflects the decrease in damping that occurs ($\zeta_{d_e} = 0.25$ is not much greater than the Level 1 limit of 0.19). An important aspect of the p_{osc}/p_{av} requirement is that it implicitly accounts for the allowable increase in the region of allowable ω_{ϕ_e} as ζ_{d_e} and ω_{d_e} increase.

The most complete set of Category A supporting data (figure 127 and figure 128), from flight tests of AFFDL-TR-67-98 using the variable-stability NT-33 shows conflicting results. The data of figure 127, for moderate $|\phi/\beta|_d$ ratios, agree quite well with the boundaries. Likewise, the figure 128 high $|\phi/\beta|_d$ points correlate, but all these data are rated Level 2 or 3, including the single point that falls in the Level 1 region. However, the cases for low $|\phi/\beta|_d$ (1.5) on figure 128 show extremely poor correlation. The reasons for this have not been resolved, but pilot comments indicate that the pilot was sensitive to the amounts of adverse aileron yaw included in many of the low $|\phi/\beta|_d$ cases. But even when there was no adverse aileron yaw the ratings were still generally poor.

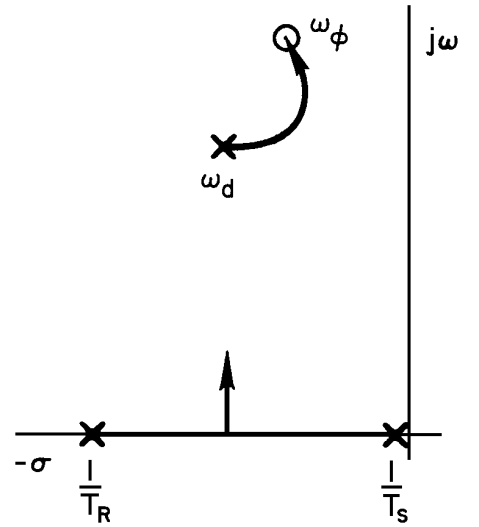




a) Low ζ_d; Zero Below Pole



b) Low ζ_d; Zero Above Pole



c) Moderate ζ_d; Zero Above Pole

FIGURE 125. Effect of relative pole-zero location on piloted control of bank angle.

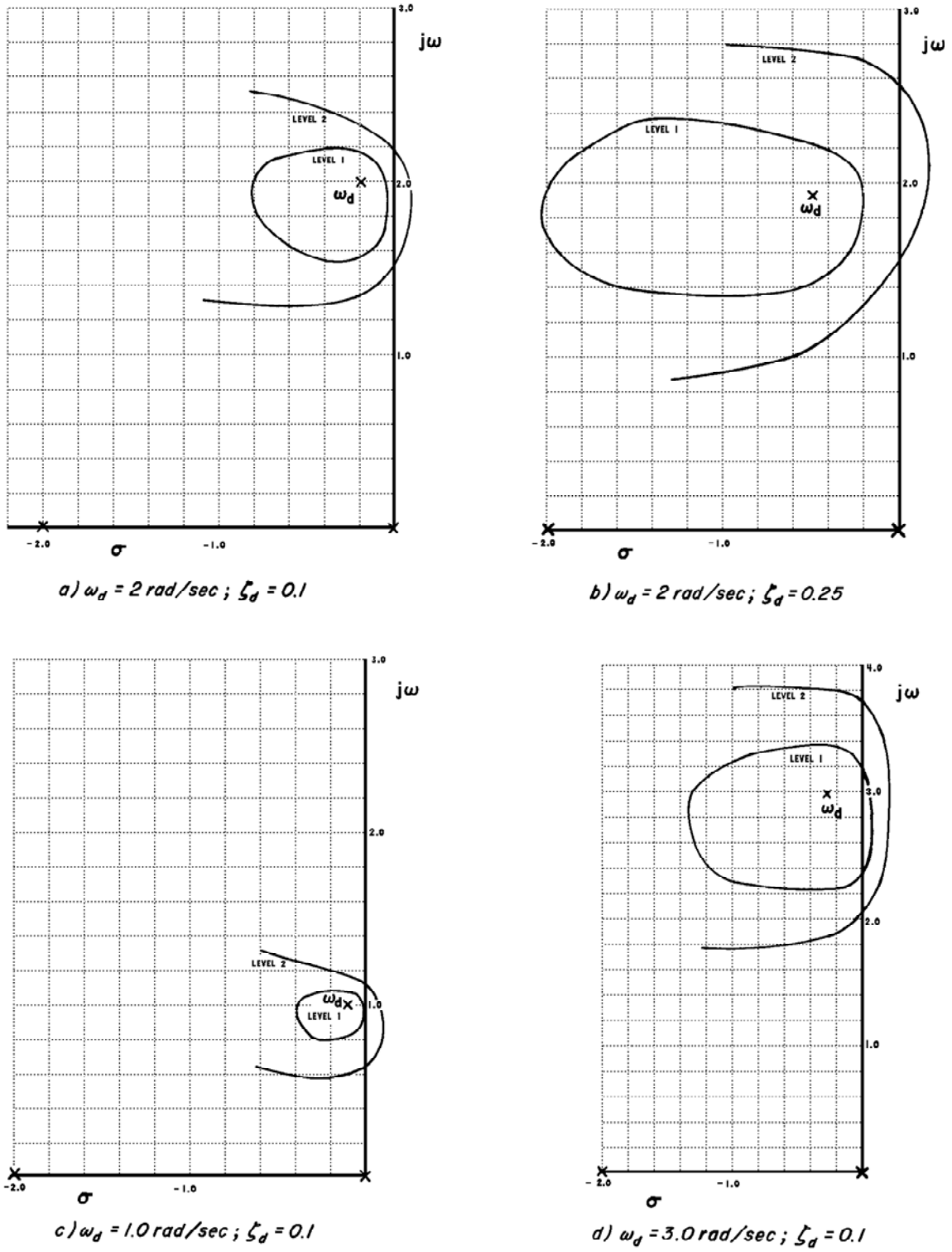


FIGURE 126. **Locations of the ω_ϕ zero corresponding to Category A and C and Level 1 and 2 boundaries on figure 123 (T_R = 0.5 sec, T_S = ∞) (from AFFDL-TR-69-72).**

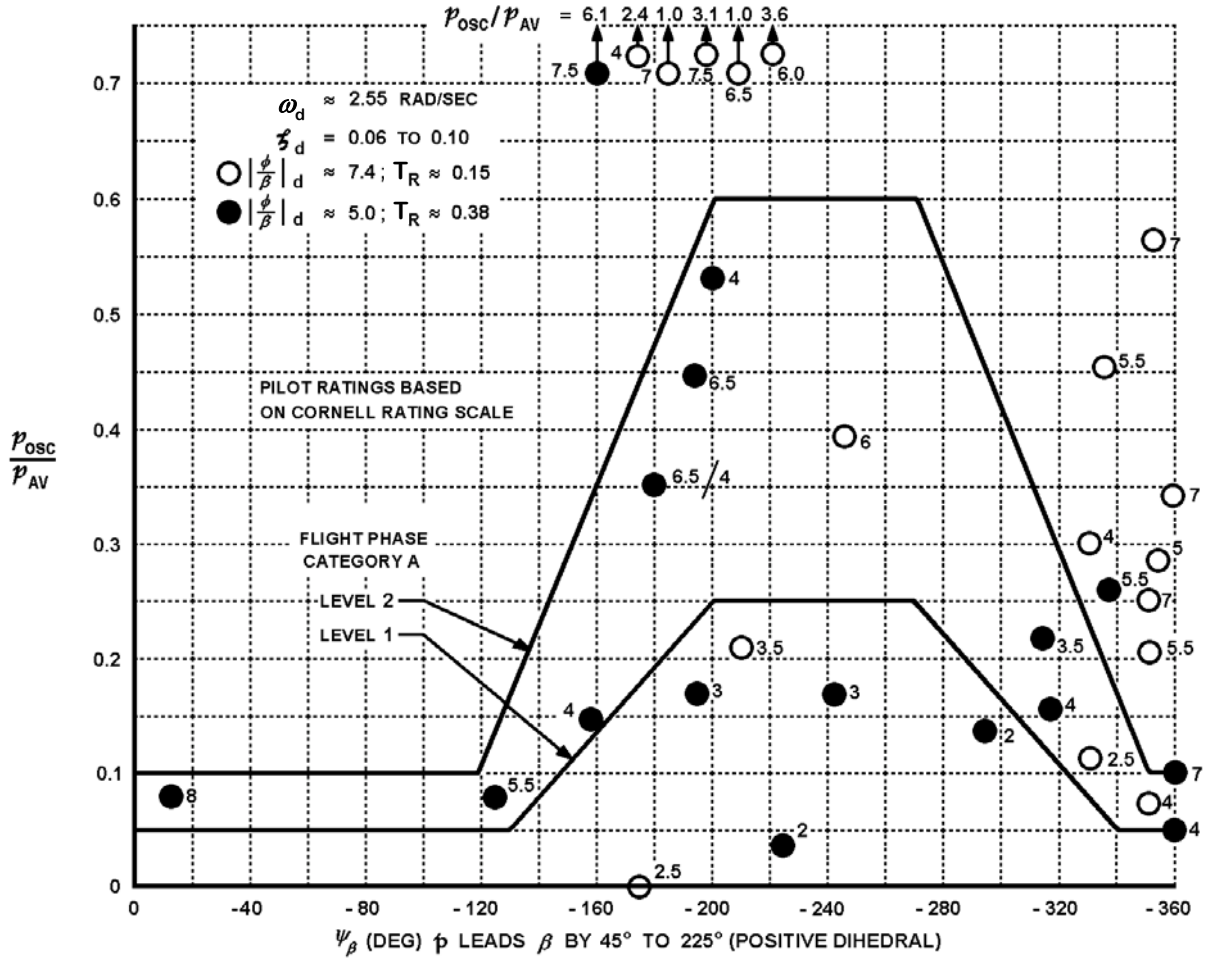


FIGURE 127. Flight Phase Category A data, moderate $|\phi/\beta|_d$ (from AFFDL-TR-67-98).

The Category B data (figure 129 and figure 130) show good correlation, but there are actually only about ten data points with which to evaluate the Level 2 and 3 regions (that is, data for which p_{osc}/p_{av} is large). Likewise, the Category C Level 2/3 boundary (figure 131) is not well defined by the data.

Similar data (AFFDL-TR-70-145) show support for the requirements (figure 132). Again, this is Category C data, though the test programs of Princeton Univ Rpt 727 and AFFDL-TR-70-145 were for approach and waveoff only and hence did not include landing. Pilot ratings might be slightly worse if landings had been required. In their favor, however, both tests did include artificial turbulence (and, for AFFDL-TR-70-145, simulated winds) which would be expected to increase pilot workload.

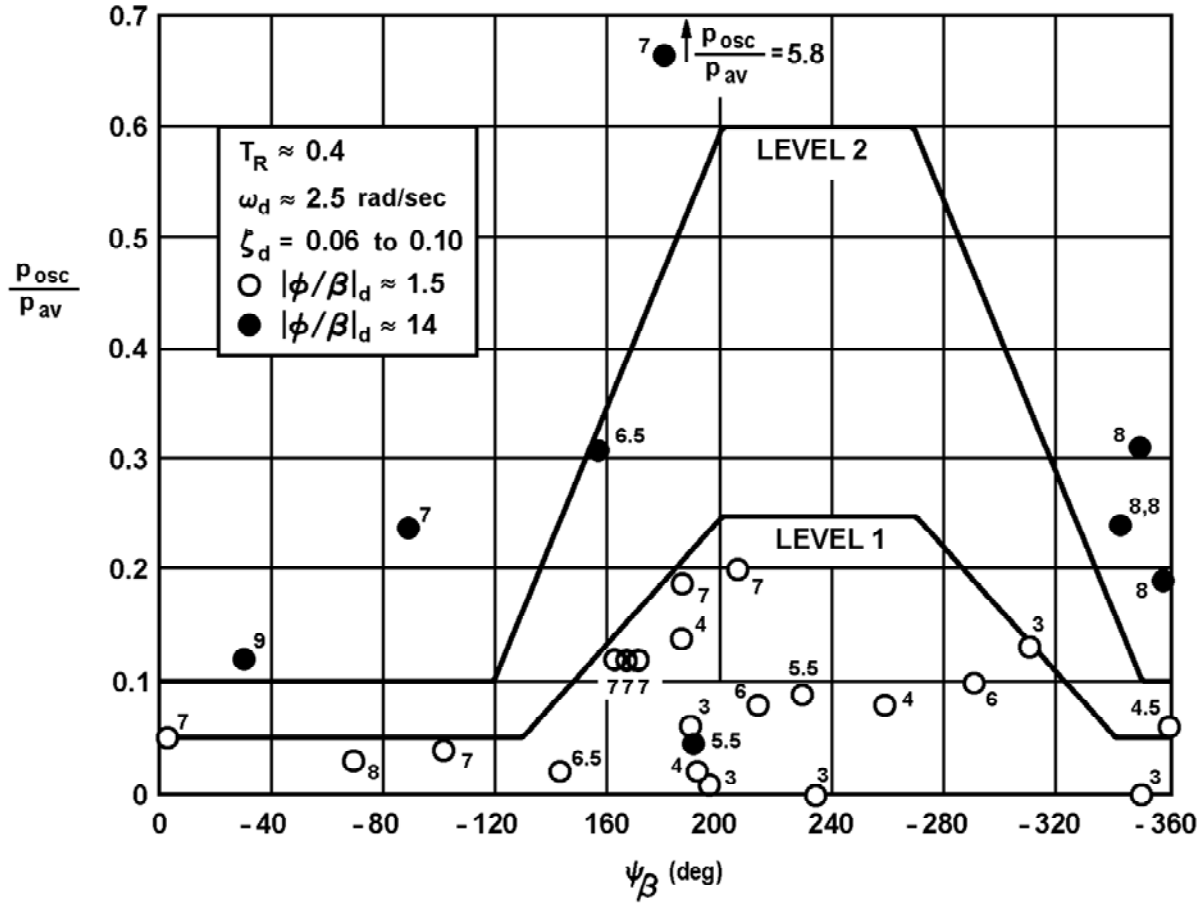


FIGURE 128. Flight Phase Category A data, large and small $|\phi/\beta|_d$ (from AFFDL-TR-67-98)

The thorough test matrix of AFFDL-TR-70-145 produced an abundance of data with which to draw some guidance for applying p_{osc}/p_{av}:

> "5.2.3.2.1 need not be applied if the $|\phi/\beta|_d$ is small (from figure 132, $|\phi/\beta|_d < 1.5$ generally produces good ratings and low p_{osc}/p_{av}; though the ratings of figure 128 for $|\phi/\beta|_d = 1.5$ are poor, p_{osc}/p_{av} is low).
>
> The criterion is most useful when ζ_d, ω_d, $\zeta_d\omega_d$, or T_R is near the Level 1/2 limit. For example, there are 21 cases on figure 132 with $\zeta_d = 0.3$ (where the Level 1 limit is 0.08), only one of which is predicted to be significantly worse than Level 1. Actually, five of the 21 are rated worse than Level 1, but only one is worse than PR = 4 ($p_{osc}/p_{av} = 4.2$, $\psi_\beta = -180°$, PR = 8)."

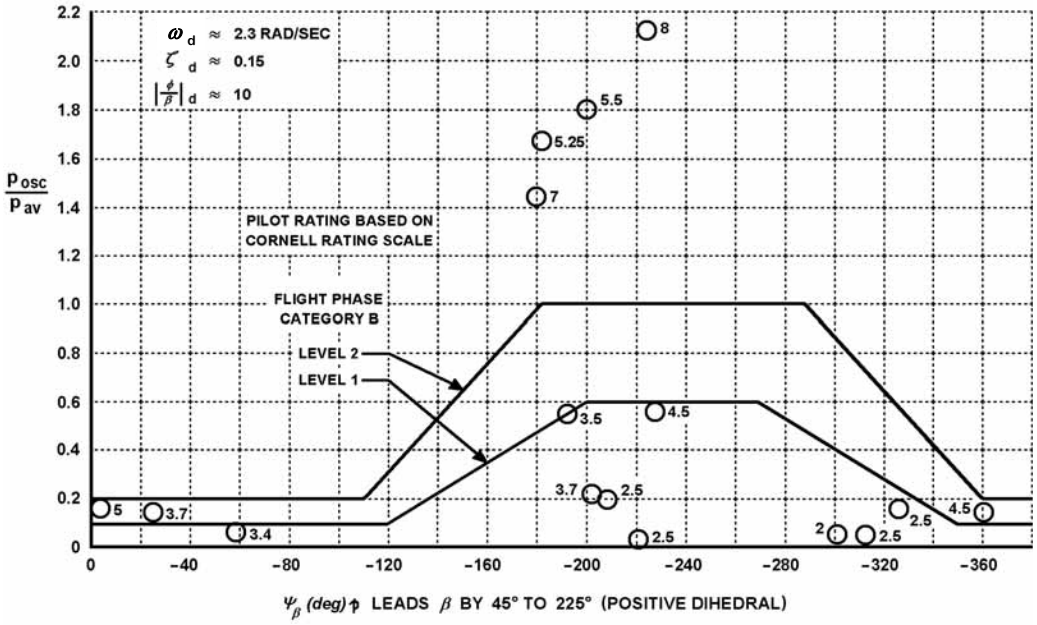


FIGURE 129. Flight Phase Category B data (from NASA CR-778).

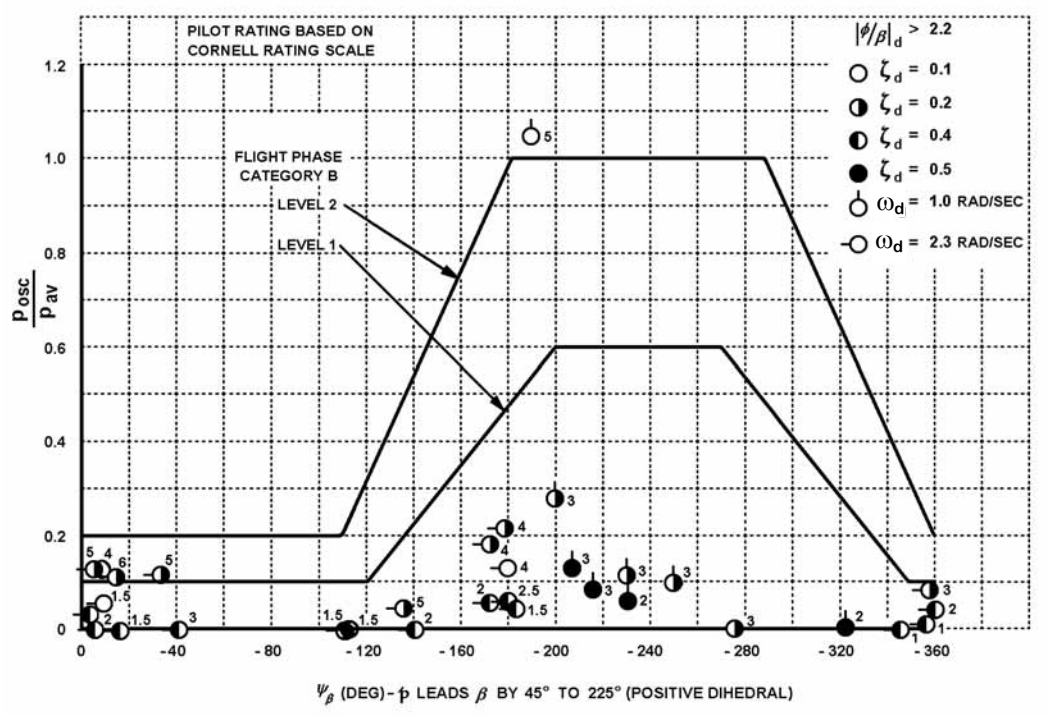


FIGURE 130. Flight Phase Category B data (WADD-TR-61-147).

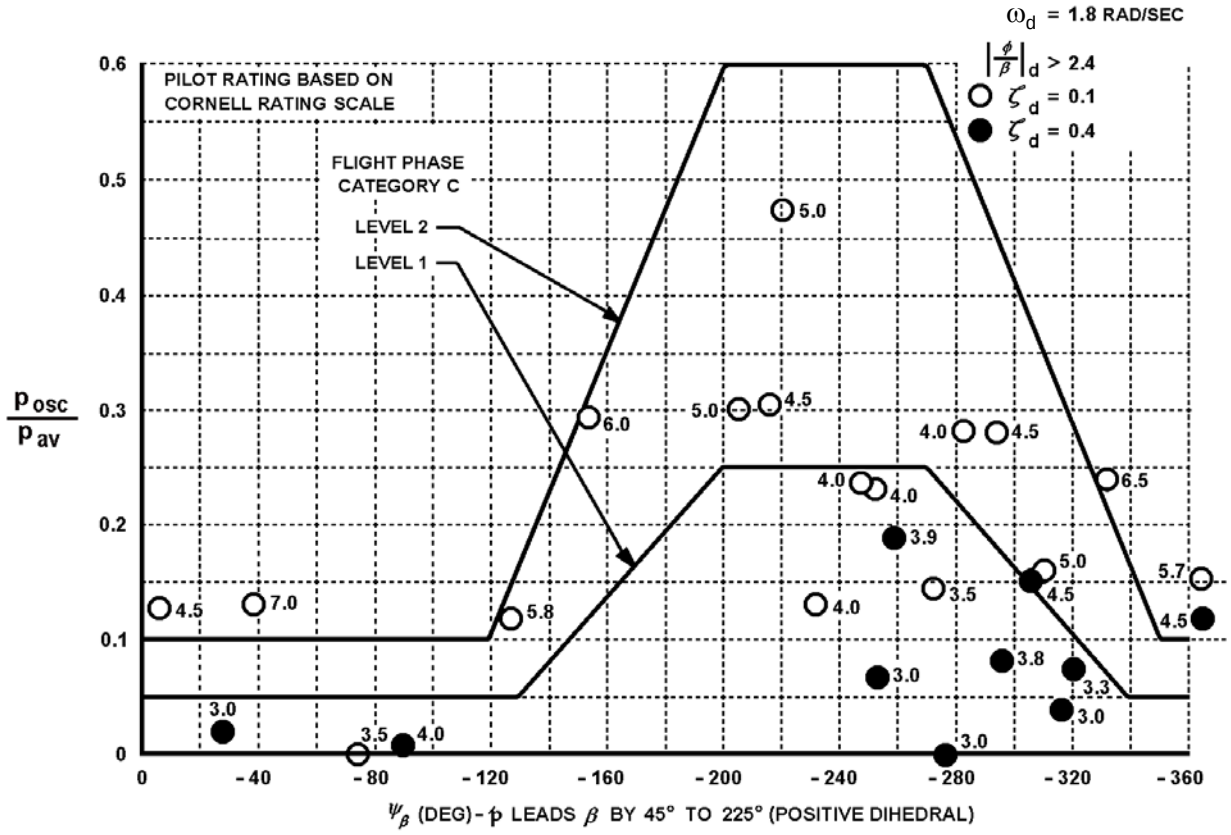


FIGURE 131. Flight Phase Category C data (from Princeton Univ Rpt 727).

Since almost no data exist on strong roll-sideslip coupling with negative dihedral, it was necessary to specify the negative-dihedral requirement through analogy with the positive-dihedral requirements previously described. Princeton Univ Rpt 604 did provide some data, however, which are presented on figure 133 for comparison with the roll rate oscillation requirement. The program of Princeton Univ Rpt 604 investigated lateral-directional instabilities relating to the X-15. In the course of this investigation, configurations were simulated either in flight or in a fixed-base simulator, or in both, that had:

a) Positive dihedral, proverse yaw due to aileron

b) Positive dihedral, adverse yaw due to aileron

c) Negative dihedral, proverse yaw due to aileron

d) Negative dihedral, adverse yaw due to aileron.

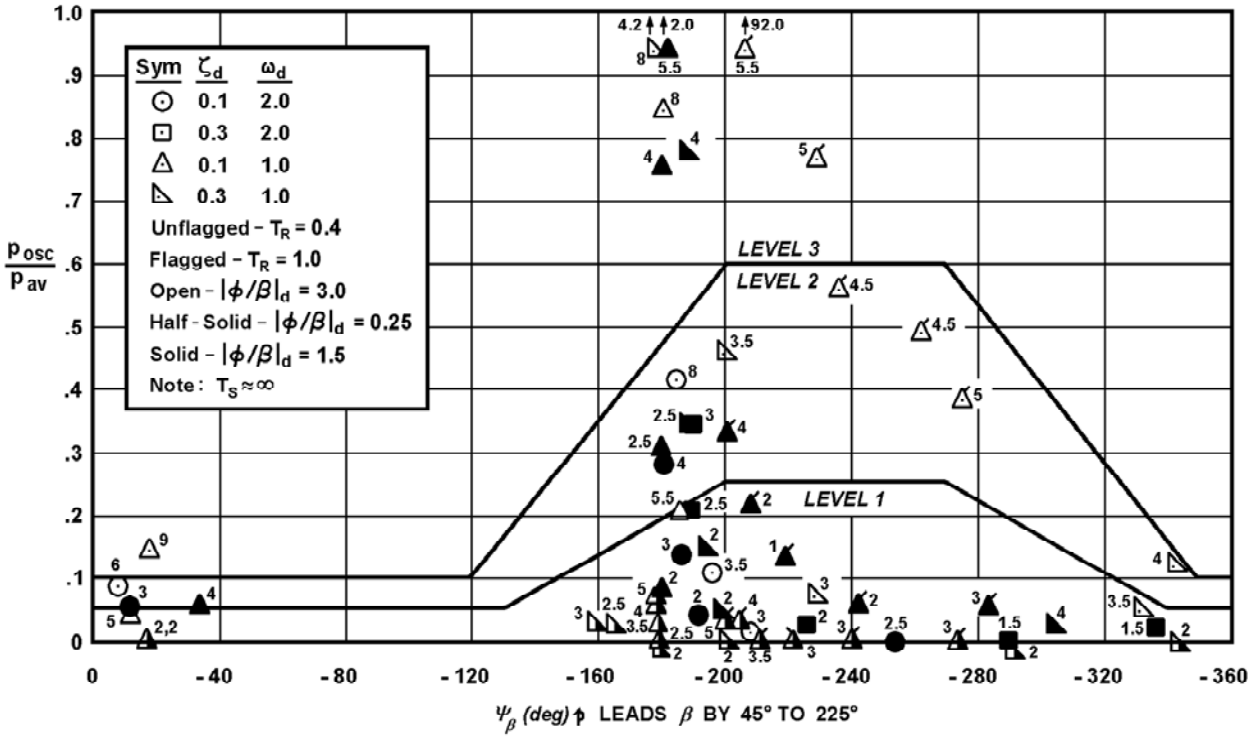


FIGURE 132. Category C data (approach and waveoff); Cooper-Harper pilot ratings (from AFFDL-TR-70-145).

These configurations, which all had very light Dutch roll damping and large $|\phi/\beta|_d$ response ratios, are plotted on figure 133. The parameters p_{osc}/p_{av} and $\angle p/\beta$ were obtained from time histories of the responses to aileron inputs. Configuration A, which falls well outside the Level 2 boundary of figure 133 was uncontrollable, attempts by the pilot to control the oscillation resulted in excursions of increasing magnitude for both sideslip angle and the roll rate.

Configuration b, which also falls well outside the Level 2 boundary of figure 133, was unacceptable because of the oscillatory response. However, it is significant that the pilot was able to control the aircraft, and, in fact, damp the oscillations when they occurred using only normal aileron control movements.

Configuration c, which marginally falls in the good area of figure 133, was controllable and it was found that attempts to control the roll angle in a normal manner also helped to reduce the excursions of the sideslip angle.

Configuration d, which falls in an area of marginal acceptably on figure 133, was uncontrollable because of pilot-in-the-loop oscillations.

Thus, with the possible exception of Configuration D, the pilot comments pertaining to the configurations were compatible with those expected from their roll-sideslip coupling characteristics as indicated by figure 133. Although Configuration d was rated worse than would be expected from the measured roll-sideslip coupling characteristics, the fact that the point fell in

the region of figure 133 where the amount of allowable roll rate oscillation changes rapidly with ψ_β would indicate the flying qualities of the configuration are sensitive to small changes in ψ_β. For example, if ψ_β were only 30º greater (or if the peaks on the time histories presented differed by only 0.2 sec from those of the configuration flown), the roll-sideslip coupling characteristics as indicated by figure 133 could be compatible with the pilot comments.

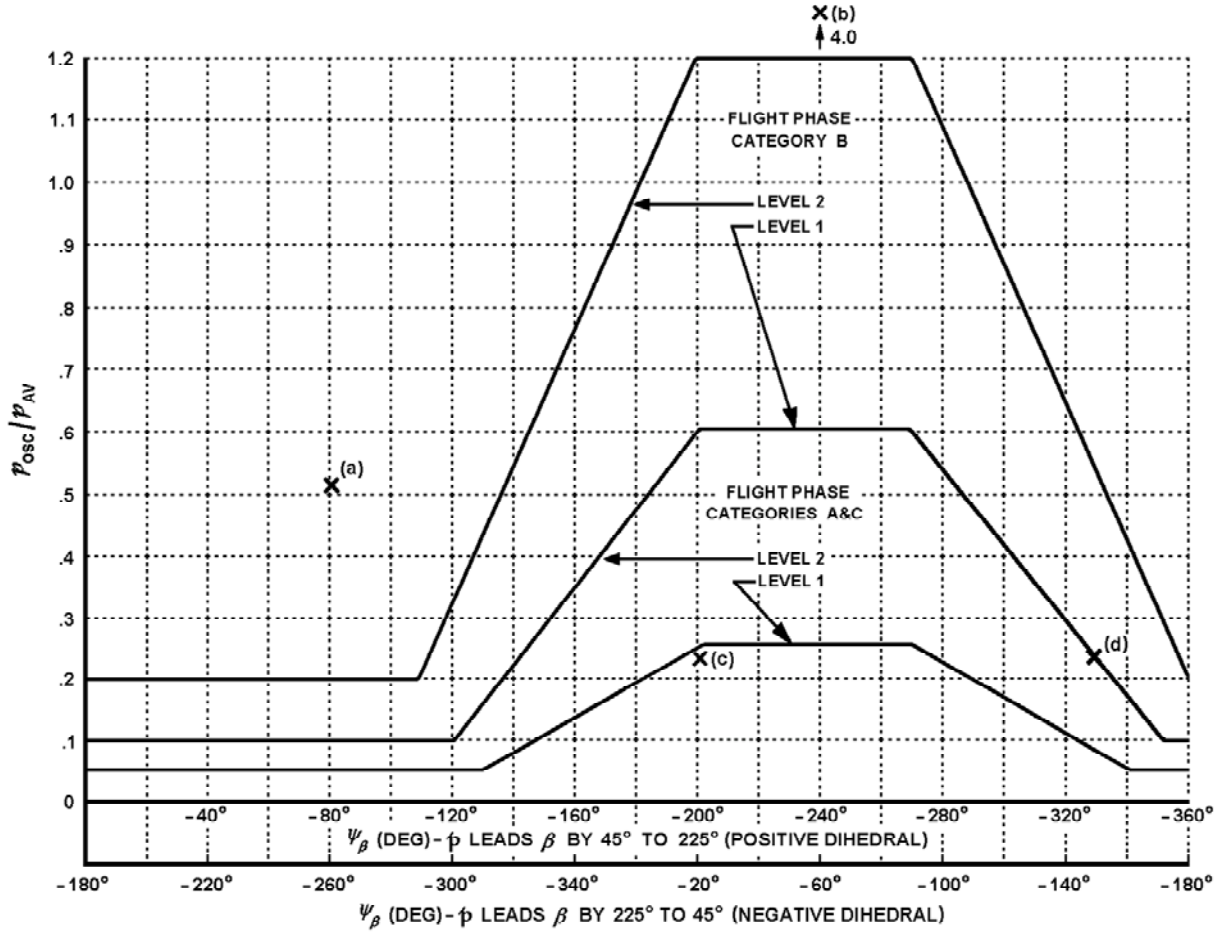


FIGURE 133. Positive and negative dihedral data of Princeton Univ Rpt 604.

As the following data reflects, correspondence with users of MIL-F-8785B and MIL-F-8785C shows the requirements for roll rate and bank angle oscillations (3.3.2.2, 3.3.2.2.1, and 3.3.2.3 of MIL-F-8785C) were generally ignored for the last generation of airplanes. However, these were highly-augmented aircraft, in which Dutch roll was cancelled. Unaugmented aircraft may still find these paragraphs useful. It is hoped the rationale presented in this document will aid the understanding of the intent. Roll control involves both fine tracking about neutral stick deflection and quick, accurate checking of high roll rate to stop accurately at a desired bank angle. Both control nonlinearities and gravitational/inertial nonlinearities can cause the characteristics to vary with control deflection or aircraft orientation in roll.

ASD Comments on MIL-F-8785B, 3.3.2, Lateral-Directional Dynamic Response Characteristics:

"F-16	Requirements 3.3.2.1, 3.3.2.2, 3.3.2.2.1, and 3.3.2.3 were deleted in the F-16 specification. These requirements were assessed to be based on a questionable data base and have historically been difficult to verify from flight test data.
F-15, F-16, C-141	Requirements 3.3.2.2.1 (p_{osc}/p_{av}) and 3.3.2.3 (ϕ_{osc}/ϕ_{av}) should not be a problem if 3.3.1.1 ($\zeta_{d_e}\omega_{d_e}$) is set at sufficient value; these paragraphs added little to the evaluation of the F-15.
AMST, B-1	Requirement 3.3.2.2 values seem too low based on DC-10 and B-1 data; 3.3.2.2.1 and 3.3.2.3 are redundant; and only 3.3.2.3 should be retained."

Military Specification MIL-F-8785C, 3.3.2.2, set limits on roll at the first minimum following the first peak in response to a step roll control input. That and p_{osc}/p_{av} are directly related, since (from AFFDL-TR-69-72):

> "The numerical values of the roll rate specified in 3.3.2.2 were transformed from the values of p_{osc}/p_{av} for adverse yaw in 3.3.2.2.1. Thus, the requirements of 3.3.2.2 and 3.3.2.2.1 are essentially identical for aircraft with adverse yaw. However, the requirement of 3.3.2.2 is far more lenient than the requirement of 3.3.2.2.1 for aircraft with proverse yaw."

VERIFICATION

The parameter p_{osc}/p_{av} is a measure of the ratio of the oscillatory component of roll rate to the average component of roll rate following a yaw-control-free step roll control command:

$$\zeta_{d_e} \le 0.2: \qquad \frac{p_{osc}}{p_{av}} = \frac{p_1 + p_3 - 2\,p_2}{p_1 + p_3 + 2\,p_2}$$

$$\zeta_{d_e} > 0.2: \qquad \frac{p_{osc}}{p_{av}} = \frac{p_1 - p_2}{p_1 + p_2}$$

where p_1, p_2, and p_3 are roll rates at the first, second, and third peaks, respectively (figure 119 through figure 121d).

The parameter ψ_β is the phase angle expressed as a lag for a cosine representation of the Dutch roll oscillation in sideslip, where

$$\psi_\beta = -\frac{360}{T_d}\,t_{n_\beta} + (n-1)\,360 \quad (^\circ)$$

with $t_{n\beta}$ defined as the time for the Dutch roll oscillation in the sideslip response to reach the n^{th} local maximum for a right step or pulse roll-control command, or the n^{th} local minimum for a left command. In the event a step control input cannot be accomplished, the control should be moved as abruptly as practical and, for purposes of this definition, time shall be measured from the instant the cockpit control deflection passes through half the amplitude of the commanded value. For pulse inputs, time shall be measured from a point halfway through the duration of the pulse.

For design, this requirement is intended to define areas of acceptable pole-zero locations for the ϕ/F_{as} transfer function. Review of the derivation and data base for the requirement has resulted in several guidelines and qualifications to be considered in interpreting the requirement.

> "The roll, Dutch roll, and spiral mode requirements (5.2.3.1.1, 5.2.3.1.2, and 5.2.3.1.3) shall first be met. If T_{S_e} is very small, the requirement may result in misleading values of p_{osc}/p_{av} (e.g., figure 134).
>
> For aircraft with very small L'_β and very large L'_r (AFFDL-TR-71-164), $\angle p/\beta|_d$ can be between 180 and 270º. As shown in AFFDL-TR-69-72, this condition is not adequately included in the approximations used to define ψ_β. This leaves some doubt as to the significance of p_{osc}/p_{av} for such data. The test results of AFFDL-TR-71-164 are not included in this report since the simulated airplane had Level 2 pitch characteristics which could have influenced pilot ratings. However, the peculiar problems encountered in measuring $\angle p/\beta$ are still of interest.
>
> If $|\phi/\beta|_d$ is small (generally less than about 1.5), p_{osc}/p_{av} will be small and the requirement may not add any new information. Then 5.2.3.2.5 and 5.2.3.2.6 can be very demanding requirements.
>
> The requirement is of most value when ζ_{d_e} is near the Level 1 boundary (0.1-0.2). For greater ζ_{d_e}, p_{osc}/p_{av} is inherently small; for less ζ_{d_e}, the handling deterioration will be indicated by falling outside the ζ_{d_e} boundary."

The type(s) of maneuver to be considered will depend upon flight control system mechanization and, to an extent, on the degree of maneuverability required. More detailed forms of the requirement are based on a linear model of the aircraft response, and so are valid only for motions which are well-described by transfer functions. To the extent that a particular approximation holds, it may be stipulated that showing compliance with one of these requirements (5.2.3.2.1, 5.2.3.2.2, and 5.2.3.2.3) adequately shows compliance with the rest.

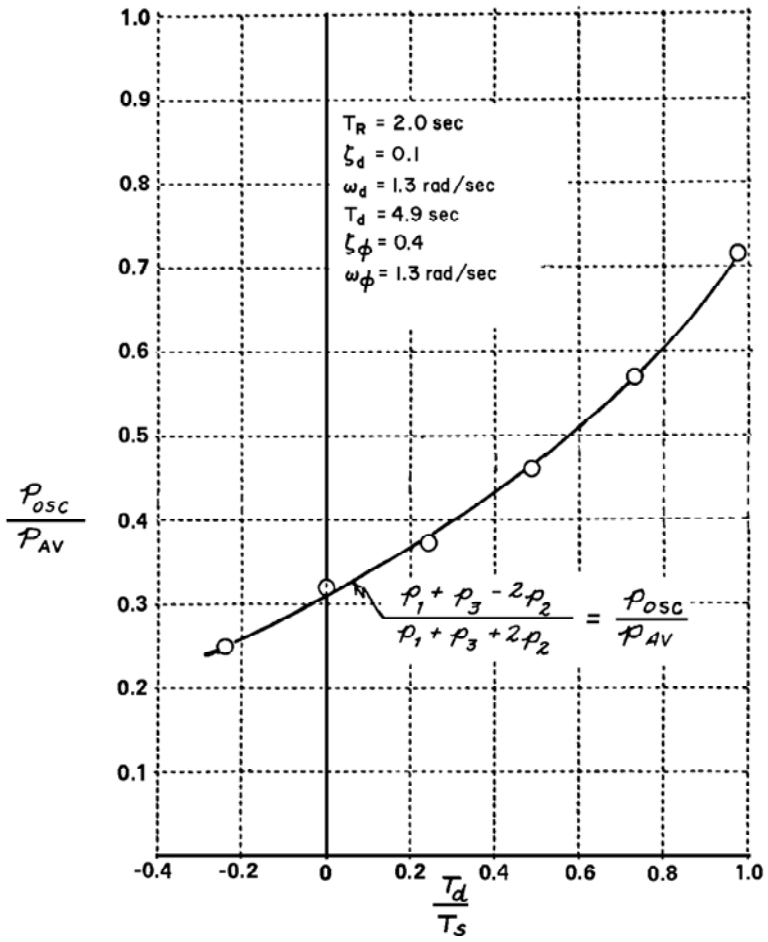


FIGURE 134. **p_{osc}/p_{av} as a function of the ratio of Dutch roll period and spiral root time constant (from AFFDL-TR-72-41).**

Closest scrutiny shall be given to those flight conditions where the roll or Dutch roll characteristics are marginally acceptable (i.e., on the boundary between Levels 1 and 2). In any case, consideration should be given to testing at the maximum operational altitude, over the range of service speeds. Data suitable for determining the critical conditions should be available from analysis and testing of roll performance (5.2.3.5). As has been indicated, for small $|\phi/\beta|_d$ this requirement will generally not be as important as the limitations on oscillatory sideslip (5.2.3.2.5 and 5.2.3.2.6).

5.2.3.2.2 Bank angle response to roll commands.
With yaw controls free, following a maximum roll control command impulse, the value of the parameter ϕ_{osc}/ϕ_{av} shall be within __(1)__.

REQUIREMENT RATIONALE

Requirements 5.2.3.2.1, 5.2.3.2.2, and 5.2.3.2.3 are directed at precise control of an aircraft with moderate to high $|\phi/\beta|_d$ response ratios combined with marginally low Dutch roll damping. Such aircraft exhibit a tendency to develop oscillations in roll rate both open- and closed-loop. This characteristic clearly interferes with the pilot's precision of control and shall be kept to an absolute minimum for all but the most undemanding tasks. The data for requirement 5.2.3.2.1 were obtained for small aileron steps, and practical problems arise when large inputs are used, so that requirement applies to small inputs only. However, a certain degree of precision is needed for large, and possibly open-loop as well as small, closed-loop inputs; so requirements 5.2.3.2.2 and 5.2.3.2.3 have been added to pertain to large control inputs.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.2.3
MIL-STD-1797: 4.5.1.4.

Recommended value for Blank (1): "the boundaries of figure 135"

REQUIREMENT LESSONS LEARNED

This requirement based on ϕ_{osc}/ϕ_{av} and ψ_β (3.3.2.3 of MIL-F-8785C) is expected to give results similar to p_{osc}/p_{av}. The three forms of this requirement (5.2.3.2.1, 5.2.3.2.2, and 5.2.3.2.3) have been retained:

For selection of the easiest form to measure

To account for both step and large pulse inputs

To handle flight control system nonlinearities, as in the F-16

To handle hands-on/hands-off differences in mechanization of stabilization, as in the F-4.

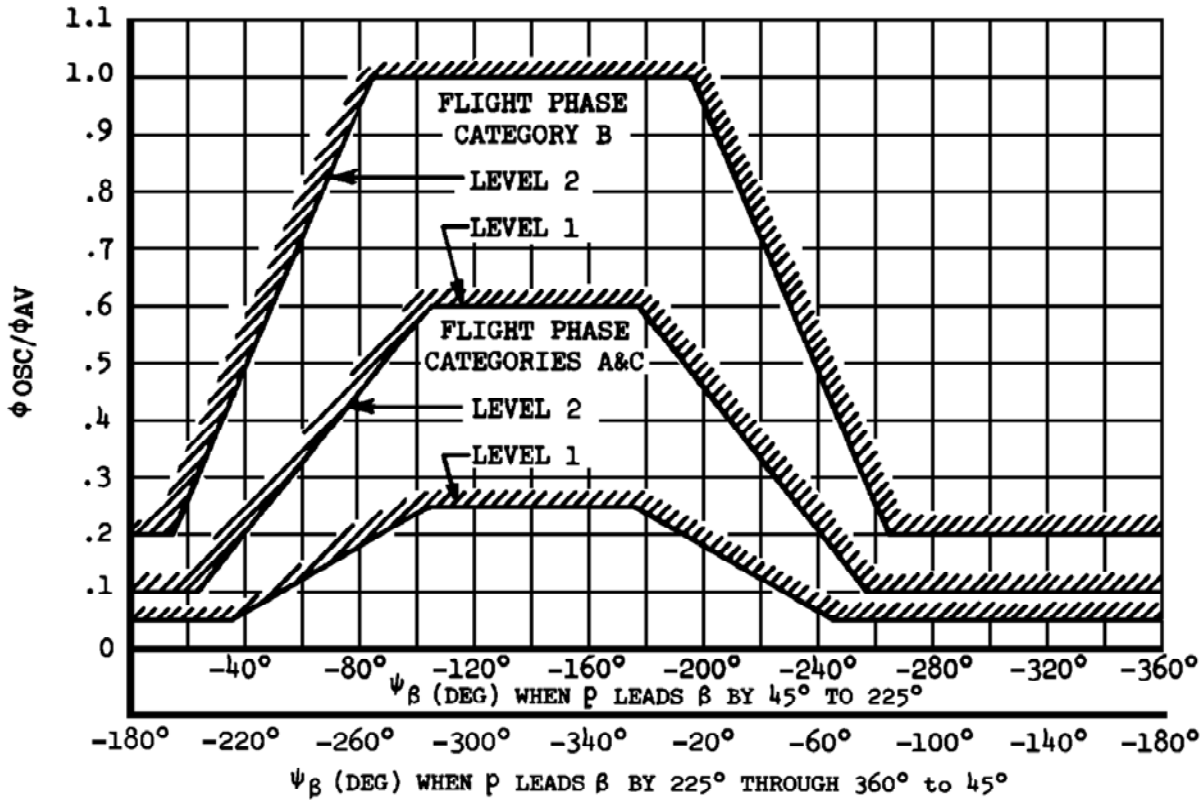


FIGURE 135. Bank angle oscillation limitations.

ϕ_{osc}/ϕ_{av} is a measure of the ratio of the oscillatory component of bank angle to the average component of bank angle following a pedal-free impulse roll control command:

$$\zeta_{d_e} \leq 0.2: \quad \frac{\phi_{osc}}{\phi_{av}} = \frac{\phi_1 + \phi_3 - 2\,\phi_2}{\phi_1 + \phi_3 + 2\,\phi_2}$$

$$\zeta_{d_e} > 0.2: \quad \frac{\phi_{osc}}{\phi_{av}} = \frac{\phi_1 - \phi_2}{\phi_1 + \phi_2}$$

where ϕ_1, ϕ_2, and ϕ_3 are bank angles in the first, second, and third peaks, respectively.

The parameter ψ_β is defined in 5.2.3.2.1.

Requirement 5.2.3.2.2 is essentially the same as 5.2.3.2.1 since, for linear motions, the bank angle time history following a pulse is the same as the roll rate time history following a step input. The advantage of using a pulse, rather than a step, is that much larger airborne inputs may be used since large bank angles do not result, and also ψ_β may be easier to measure since the β trace will not tend to ramp.

Since ψ_β for a pulse is $(90 + \sin^{-1}\zeta_{d_e})^o$ more positive than ψ_β for a step, the curves of figure 123 have been shifted to the left 90º plus an angle corresponding to $\zeta_{d_e} \approx 0.1$. The resulting curves of figure 135 shall be within 5º of the exact location for all damping ratios for which this requirement might be critical.

This paragraph, then, like 5.2.3.2.3 and 5.2.3.2.6, is an attempt to extend the roll-sideslip coupling requirements to larger control deflections. It may also help account for some flight control system nonlinearities such as stick position-dependent roll damping augmentation.

VERIFICATION

Closest scrutiny shall be given to those flight conditions where the roll or Dutch roll characteristics are marginally acceptable (i.e., on the boundary between Levels 1 and 2). In any case, consideration should be given to testing at the maximum operational altitude, over the range of service speeds. Data suitable for determining the critical conditions should be available from analysis and testing of roll performance (5.2.3.5). As indicated in 5.2.3.2.1, for small $|\phi/\beta|_d$ this requirement will generally not be as important as the limitations on oscillatory sideslip (5.2.3.2.5 and 5.2.3.2.6).

5.2.3.2.3 Roll rate response to large roll commands.
With yaw controls free, for a step roll control command held until bank angle has changed at least 90º, the roll rate at the first minimum (p_2) following the first peak shall be of the same sign as the roll rate at the first peak (p_1) and have a magnitude no less than the values in table L. For all Levels, the change in bank angle shall always be in the direction of the roll control command.

TABLE L. Minimum roll rate at the first minimum, p_2.

Level	Flight Phase Category	Minimum $\|p_2\|$
1	A	
	B	
	C	
2	A	
	B	
	C	

REQUIREMENT RATIONALE

Requirements 5.2.3.2.1, 5.2.3.2.2, and 5.2.3.2.3 are directed at precise control of an aircraft with moderate to high $|\phi/\beta|_d$ response ratios combined with marginally low Dutch roll damping. Such aircraft exhibit a tendency to develop oscillations in roll rate both open- and closed-loop. This characteristic clearly interferes with the pilot's precision of control and shall be kept to an absolute minimum for all but the most undemanding tasks. The data for requirement 5.2.3.2.1 were obtained for small aileron steps, and practical problems arise when large inputs are used, so that requirement applies to small inputs only. However, a certain degree of precision is needed for large, and possibly open-loop as well as small, closed-loop inputs; so requirements 5.2.3.2.2 and 5.2.3.2.3 have been added to pertain to large control inputs.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.2.2
MIL-STD-1797: 4.5.1.4.

Recommended values for TABLE L:

Level	Flight Phase Category	Minimum $\|p_2\|$
1	A and C	0.6 $\|p_1\|$
	B	0.25 $\|p_1\|$
2	A and C	0.25 $\|p_1\|$
	B	0

REQUIREMENT LESSONS LEARNED

The parameters p_1 and p_2 are defined in 5.2.3.2.

The numerical values of roll rate specified in 5.2.3.2.3 were transformed from the values of p_{osc}/p_{av} for adverse yaw in 5.2.3.2.1. Thus, the requirements of 5.2.3.2.1 and 5.2.3.2.3 are essentially identical for airplanes with adverse yaw. However, the requirement of 5.2.3.2.3 is far more lenient than the requirement of 5.2.3.2.1 for airplanes with proverse yaw.

The 90° roll amplitude limit of 5.2.3.2.3 arbitrarily matches the 45°-to-45°-bank maneuver of MIL-F-8785(ASG) for a similar requirement. Obviously fighters roll through larger angles. In that respect it would have been rational to make the amplitude limit a function of Class or Flight Phase. However, the design implications could be rather drastic. For that reason, application is restricted to maneuvers of roll amplitude less than 90°.

VERIFICATION

Closest scrutiny shall be given to those flight conditions where the roll or Dutch roll characteristics are marginally acceptable (i.e., on the boundary between Levels 1 and 2). In any case, consideration should be given to testing at the maximum operational altitude, over the range of service speeds. Data suitable for determining the critical conditions should be available from analysis and testing of roll performance (5.2.3.5). As has been indicated, for small $|\phi/\beta|_d$ this requirement will generally not be as important as the limitations on oscillatory sideslip (5.2.3.2.5 and 5.2.3.2.6).

5.2.3.2.4 Lateral acceleration at the pilot station.
With yaw control free and with it used to coordinate turn entry, the ratio of maximum lateral acceleration at the pilot station to maximum roll rate shall not exceed the values in table LI for the first 2.5 sec following a step roll control input.

TABLE LI. Lateral acceleration at the pilot station versus roll rate.

Level	Maximum $n_{y_{pilot_{max}}}/p_{max}$
1	
2	
3	

REQUIREMENT RATIONALE

Where the pilot is well forward of the center of gravity or well above the roll axis, turn entries with zero sideslip are accompanied by large lateral accelerations at the cockpit and entries with a centered ball may be impossible. This requirement is included to limit such accelerations to acceptable levels.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.5.4.

The recommended values for table LI:

Level	Maximum $n_{y_{pilot_{max}}}/p_{max}$
1	0.012 g/deg/sec
2	0.035 g/deg/sec
3	0.058 g/deg/sec

REQUIREMENT LESSONS LEARNED

Lateral acceleration at a distance (x, z) from the c.g. is given by

$$a_y = \dot{v} + r\,u - p\,w - g\cos\theta\sin\phi - z\,\dot{p} + x\,\dot{r}\,.$$

For a step aileron input the initial lateral acceleration is

$$\frac{n_{y_{p_0}}}{\dot{p}_0} = \frac{-z_p}{g} + \frac{x_p}{g}\frac{N'_{\delta_a}}{L'_{\delta_a}} + \frac{1}{g}\frac{Y_{\delta_a}}{L'_{\delta_a}} = \frac{-z_p}{g}\left(1 + \frac{x_p\, k_x^2\, C'_{n_{\delta_a}}}{z_p\, k_z^2\, C'_{\ell_{\delta_a}}} - \frac{b\, k_x^2\, C_{y_{\delta_a}}}{z_p\, C'_{\ell_{\delta_a}}}\right)$$

and in a steady turn, with θ and ϕ in stability axis ($w = 0$)

$$\dot{\psi} = \frac{g}{V\cos\beta}\left(\tan\phi + \frac{n_y}{\cos\theta\cos\phi}\right)$$

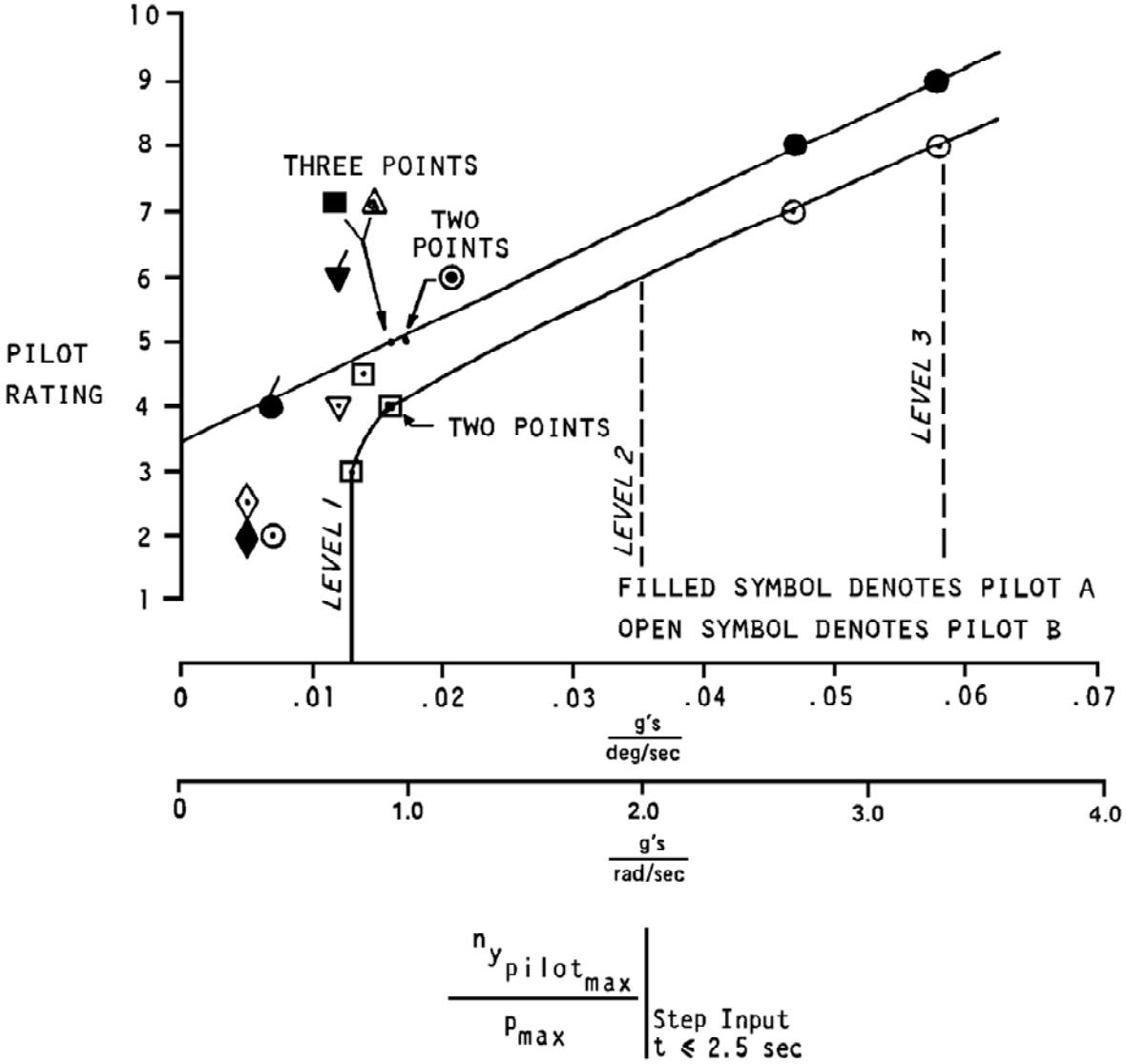
where $k_x^2 = I_{xx}/(mb^2)$ and $k_z^2 = I_{zz}/(mb^2)$ in terms of stability axes and primed derivatives which account for product of inertia effects; x_p and z_p are pilot station displacements from the center of gravity.

Concern over lateral acceleration is primarily for ride qualities, although in some cases aircraft control can be affected due to arm/bobweight effects, or just the jerkiness associated with a large offset from the roll axis.

A criterion based on the ratio of maximum pilot acceleration to maximum roll rate includes in it the recognition of pilot acceptance of high accelerations as a function of aircraft rolling performance. Such a criterion was proposed by Chalk in NASA CR-159236 for large aircraft, and the recommended values are based on flight results with the TIFS. Some refinement of the limits may be necessary for Class I, II, and IV aircraft, and some modification may be in order if the crew are given more effective lateral restraints.

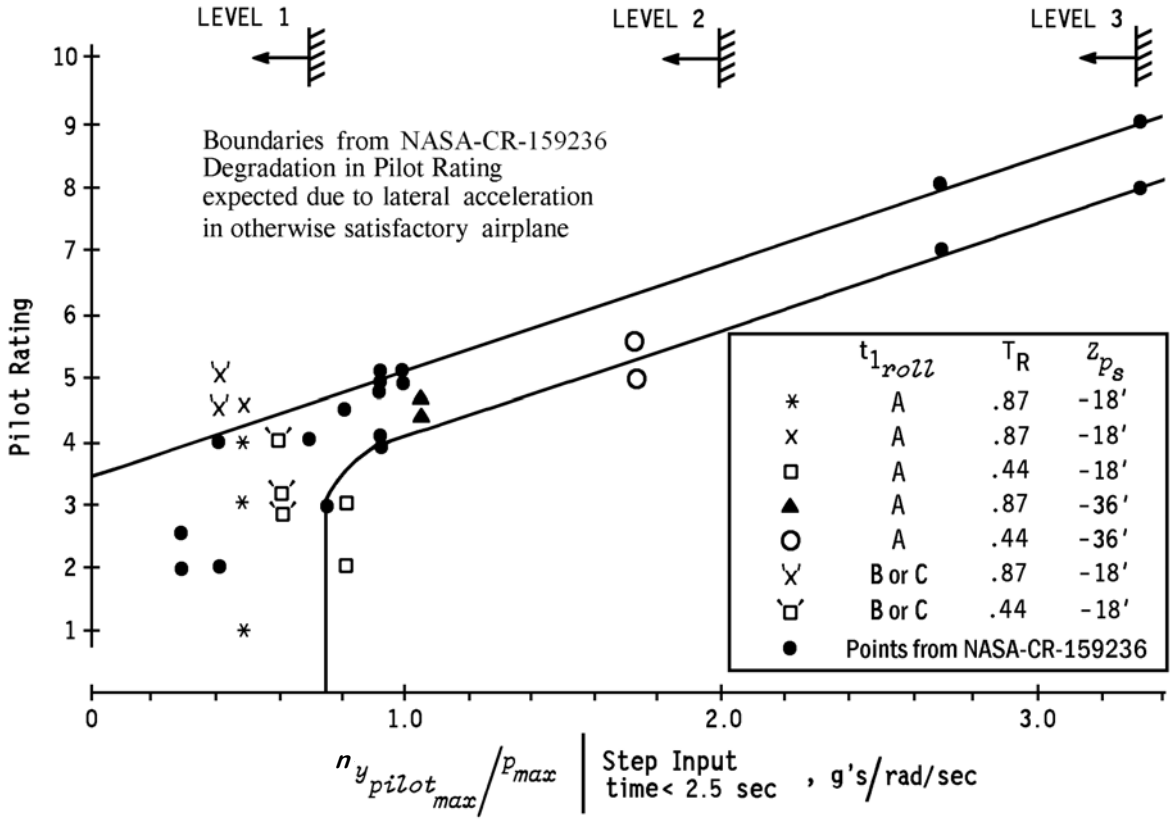
Objectionable lateral acceleration was noted in the C-5A (AFFDL-TR-75-3), and fighter pilots (e.g. F-15) have remarked about their helmets hitting the canopy during abrupt rolls. The criterion was derived in NASA CR-159236 as a proposed flying qualities requirement for Supersonic Cruise Research (SCR) aircraft. Figure 136 shows TIFS data compared with Cooper-Harper pilot ratings. Figure 137 gives more data for large aircraft from AFWAL-TR-81-3118. $t_{1_{roll}}$ indicates the level of delay in the configuration descriptions of AFWAL-TR-81-3118. $t_{1_{roll}}$ = A was the nominal effective time delay of the aircraft model, the feel system, the model surface servo, and TIFS model-following delay in roll (0.21 sec, total). $t_{1_{roll}}$ = B added a first-order prefilter lag to A such that the total effective delay was 0.29 sec. $t_{1_{roll}}$ = C was B with additional transport delay such that the total $t_{1_{roll}}$ was 0.37 sec. Correlation is quite good, though more data should be gathered, especially for other classes of aircraft.

Due to the tentative nature of this requirement, it should be applied primarily as a guideline until more data can be obtained. When dealing with aircraft in which the cockpit is either well forward of the center of gravity or well above the roll axis, designing the roll augmentor to minimize sideslip or c.g. lateral acceleration can produce unacceptable lateral accelerations at the pilot station. With direct side force control another possibility for lessoning the magnitude of lateral acceleration in rolling is to control the height of the roll axis: modify the effective Y_{δ_a}.



Notes: 1. Flagged points are configurations specifically downgraded by Pilot A due to poor Dutch roll damping—not lateral acceleration.
2. The lines indicate degradation in pilot rating to be expected because of ride qualities for an airplane with otherwise satisfactory flying qualities parameters.

FIGURE 136. Lateral acceleration criterion versus pilot rating from NASA CR-159236.



NOTE: The three * points are from the baseline configuration Long Aft, High $\bar{q}$, $t_{1_{roll}}$ = A; Pilot A—one point, Pilot B—two points. These were evaluated during the longitudinal variations and no lateral-directional comments were made.

FIGURE 137. Pilot rating versus lateral acceleration criteria.

VERIFICATION

Both large and small step roll control inputs shall be used and measurements made within the first 2.5 sec of application of the control. Conceivably Dutch roll phasing could make another time more critical, but 2.5 sec is a reasonably-long duration for a roll maneuver.

5.2.3.2.5 Yaw response to small roll control commands.
With yaw controls free and, in initial steady turns, fixed at the deflection for zero sideslip in the turn, the amount of sideslip following a step roll control command up to the magnitude that causes a 60º bank angle change within T_d or 2 sec (whichever is greater) shall be within __(1)__ for Levels 1 and 2.

REQUIREMENT RATIONALE

This requirement is intended to insure that any yaw control needed to coordinate turn entries and recoveries/reversals is not objectionable to the pilot. This requirement uses the value of sideslip angle that accompanies roll control inputs as the measure of acceptance.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.2.4.1
MIL-STD-1797: 4.6.2.

Recommended value for Blank (1): "the boundaries of figure 138"

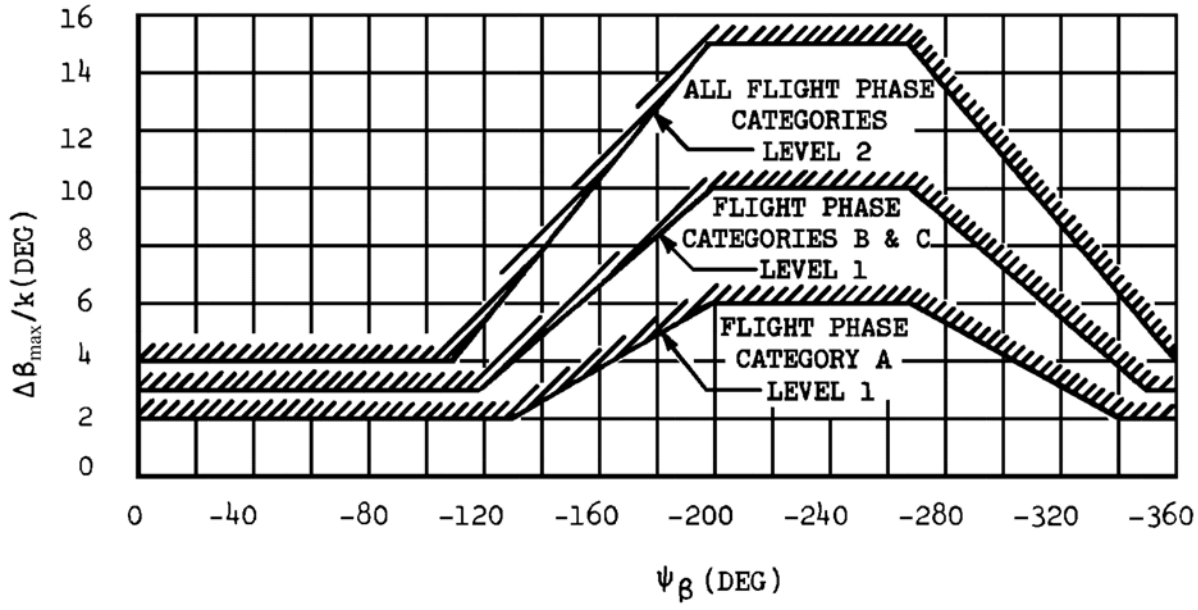


FIGURE 138. Sideslip excursion limitations for small roll control commands.

REQUIREMENT LESSONS LEARNED

The requirement presented applies to sideslip excursions. Other related motions of interest, but upon which no requirements can be laid, are lateral acceleration at the pilot station and the initial yawing response. This requirement was first introduced in MIL-F-8785B.

The primary source data from which the sideslip requirement evolved is the low $|\phi/\beta|_d$ (≈ 1.5) configurations of AFFDL-TR-67-98 (figure 139). Analysis of the data revealed that the amount of sideslip that a pilot will accept or tolerate is a strong function of the phase angle of the Dutch roll component of sideslip. When the phase angle is such that β is primarily adverse (out of the turn), the pilot can tolerate quite a bit of sideslip. On the other hand, when the phasing is such that β is primarily proverse (into the turn), the pilot can only tolerate a small amount of sideslip because of difficulty of coordination.

There is more to coordination, however, than whether the sideslip is adverse or proverse: the source and phasing of the disturbing yawing moment also significantly affect the coordination problem. If the yawing moment is caused by aileron and is in the adverse sense, then in order to coordinate, the pilot must phase either right rudder with right aileron or left rudder with left aileron. Since pilots find this technique natural they can generally coordinate well even if the yawing moment is large. If, on the other hand, the yawing moment is in the proverse sense or is caused by roll rate, coordination is far more difficult. For proverse yaw due to aileron, the pilot must cross control; and for either adverse or proverse yaw due to roll rate, required rudder inputs must be proportional to roll rate and also phased with respect to it. Pilots find these techniques unnatural and difficult to perform. Since yawing moments may also be introduced by yaw rate, it can be seen that depending on the magnitude and sense of the various yawing moments, coordination may be either easy or extremely difficult. If coordination is sufficiently difficult that pilots cannot be expected to coordinate routinely, the flying qualities requirements must restrict unwanted motions to a size acceptable to pilots.

Analysis further revealed that it was not so much the absolute magnitude of the sideslip that bothered the pilot, but rather the maximum change in sideslip. The latter was a better measure of the amount of coordination required. Thus, the data from this program were plotted on figure 140 as the maximum change in sideslip occurring during a rudder-pedal-fixed rolling maneuver, $\Delta\beta_{max}$, versus the phase angle of the Dutch roll component of the sideslip, ψ_β (see 5.2.3.2.1). Note that the Dutch roll damping ratio is Level 2 for the data of figure 140.

The phase angle, ψ_β, is a measure of the sense of the initial sideslip response, whether adverse or proverse, while $\Delta\beta_{max}$ is a measure of the amplitude of the sideslip generated. Both the sense and the magnitude affect the coordination problem.

It was observed from examination of the low $|\phi/\beta|_d$ data plotted on figure 140 that the break points and curves of iso-pilot ratings occurred at almost exactly the same value of ψ_β as for the moderate $|\phi/\beta|_d$ configurations (see the discussions in 5.2.3.2.1), even though the degradation of flying qualities was due to sideslip with the low $|\phi/\beta|_d$ configurations and to bank angle problems with the moderate $|\phi/\beta|_d$ configurations. Since the break points were so close and since the figures describe different manifestations of the same phenomena, the break points were made identical for both the low $|\phi/\beta|_d$ configurations ($\Delta\beta_{max}$ versus ψ_β) and moderate $|\phi/\beta|_d$ configurations (p_{osc}/p_{av} versus ψ_β). The sideslip excursions criteria were thus presented in the form as shown on figure 141.

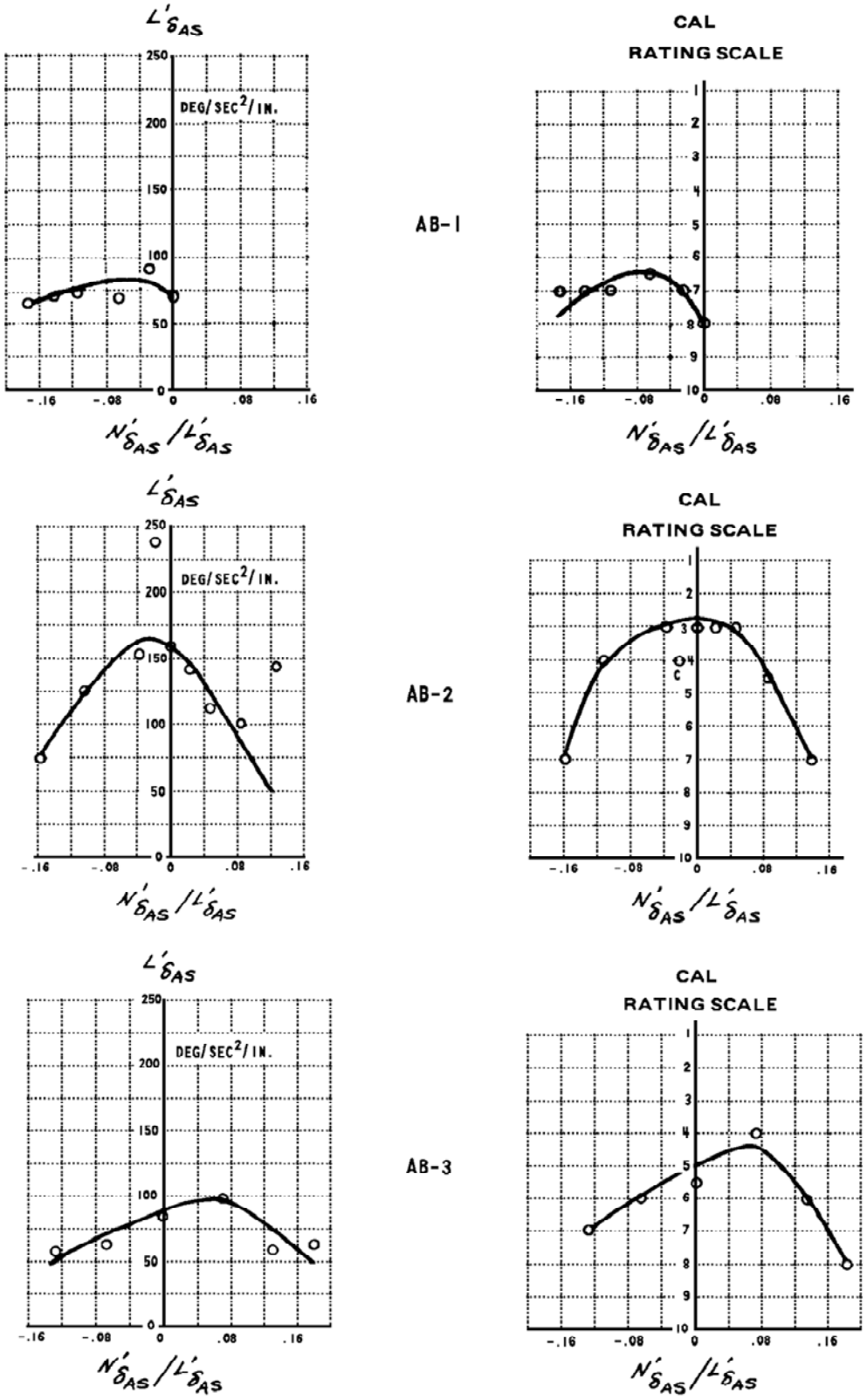


FIGURE 139. Pilot ratings and optimum aileron sensitivity (low $|\phi/\beta|_d$, medium T_R) (AFFDL-TR-67-98).

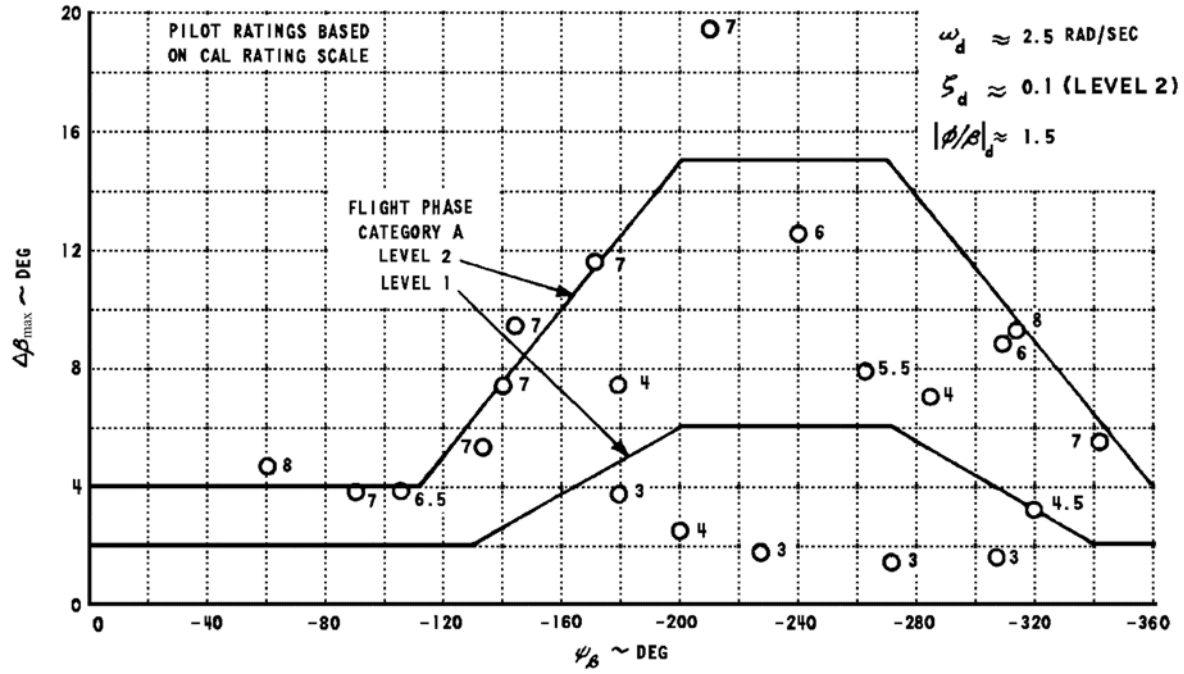


FIGURE 140. Flight Phase Category A data from AFFDL-TR-67-98.

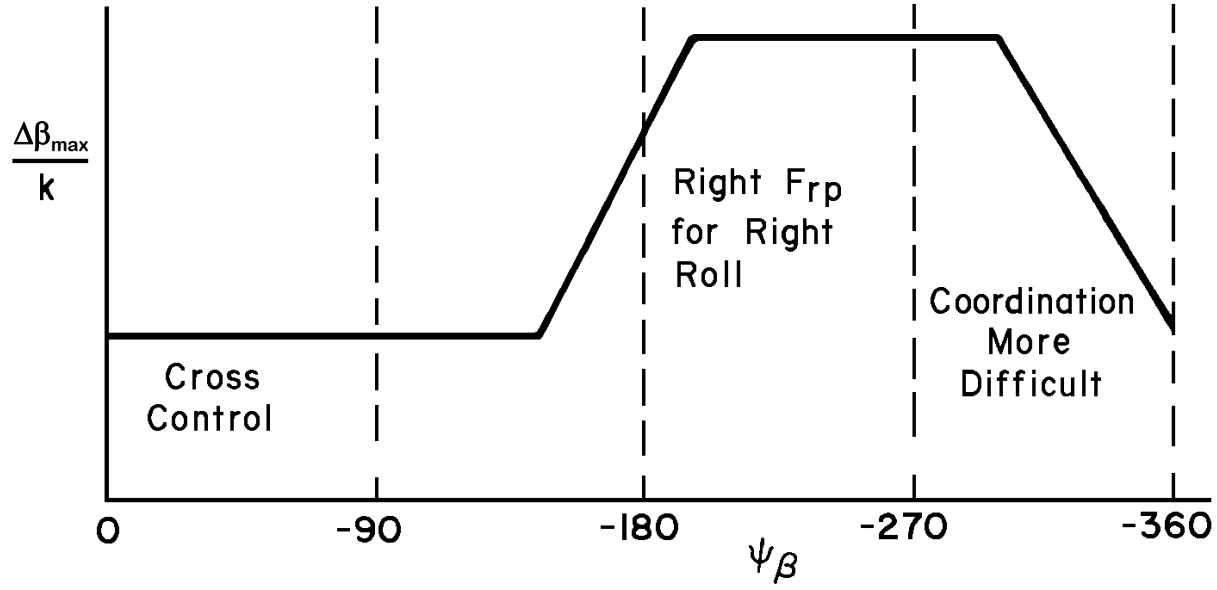


FIGURE 141. Form of the sideslip excursion criteria.

As with the p_{osc}/p_{av} requirement, it can be seen from figure 141 that the specified value of $\Delta\beta_{max}$ varies significantly with ψ_β. This difference is almost totally due to the differences of ability to coordinate during turn entries and exits. Since ψ_β is a direct indicator of the difficulty a pilot will experience in coordinating a turn entry, variation of allowable $\Delta\beta_{max}$ with ψ_β is to be expected. For $-180° \leq \psi_\beta \leq -260°$, the pilot may coordinate; that is, right rudder pedal for right rolls. Thus, even if large sideslip excursions occur, by coordinating in the normal manner sideslip oscillations can readily be minimized. As ψ_β varies from -270 to -360º, coordination becomes increasingly difficult, and in the range $-360° \leq \psi_\beta \leq -90°$ cross controlling is required to effect coordination. Since pilots do not normally cross control and, if they must, have great difficulty in doing so, for $-360° \leq \psi_\beta \leq -90°$, oscillations in sideslip either go unchecked or are amplified by the pilot's effort to coordinate with rudder pedals.

The parameter k relates the amount of allowable sideslip without rudder-pedal use to the roll performance requirements.

$$k = \frac{\phi_{t_{command}}}{\phi_{t_{requirement}}}$$

where

$\phi_{t_{command}}$ = the bank angle actually attained in the stated time for a given step roll command with yaw control pedals employed as specified in 5.2.3.5 and 5.2.3.5.1

$\phi_{t_{requirement}}$ = the bank angle required in the applicable time from 5.2.3.5 and 5.2.3.5.1 for the Class, Flight Phase Category, and Level under consideration

Through this tie to roll performance requirements, the effect of Class and some of the effects of Flight Phase and Level are taken into consideration. While $\Delta\beta_{max}$ is relatively straightforward, the parameter k is a function of Flight Phase, Aircraft Class, speed range, and actual roll performance. In addition, questions arise about the influence of the yaw controller, since the roll performance requirements of 5.2.3.5 allow use of yaw controls in some instances. While here $\Delta\beta_{max}$ is found with feet on the floor, except in steady turns requiring rudder-pedal coordination, k is a measure of roll response with combined roll and yaw controls.

Because the required ϕ_t values of 5.2.3.5 are different for Levels 1, 2, and 3, use of k has in the past involved separate values of $\phi_{t_{requirement}}$ for comparison with the Level 1 boundary and with the Level 2 boundary of $\Delta\beta_{max}/k$. The supporting data for this requirement do not show the need for such special treatment: the proper value of $\phi_{t_{requirement}}$ to use is that specified for Level 1 in 5.2.3.5 with rudder use if permitted there, and the proper value of $\phi_{t_{command}}$ will be the value obtained by performing the verification tests of 5.2.3.5. The idea is to limit roll-control-only sideslip to a value which is (a) proportional to the roll control used and (b) stated in terms of the amount of roll control needed to meet the Level 1 roll performance requirement.

As a result, the correct way to compute k is with the commanded value assuming rudder use as permitted by 5.2.3.5 and the required Level 1 ϕ_t value obtained from 5.2.3.5. The resulting $\Delta\beta_{max}/k$, whether Level 1 or not, is then compared to the requirements of 5.2.3.2.5. This is a perfectly adequate way to define the parameters.

The need for a requirement limiting yaw response to roll controller is a result of Dutch roll excitation for aircraft with low to moderate $|\phi/\beta|_d$. If $|\phi/\beta|_d$ is large, the Dutch roll will be most noticeable in roll rate, and p_{osc}/p_{av} is the important criterion (see 5.2.3.2.1). In general, the available data suggests that $\Delta\beta_{max}/k$ is not as useful as p_{osc}/p_{av} when $|\phi/\beta|_d$ is greater than 3.5 to 5.0.

Finally, some of the shortcomings discussed for p_{osc}/p_{av} in 5.2.3.2.1 are relevant here as well: for example, for very low Dutch roll frequencies, the step roll control inputs must be very small (see AFFDL-TR-72-41). However, although an unstable spiral mode causes divergence in all lateral-directional degrees of freedom, the small amount of sideslip in that mode makes the Dutch roll mode easier to sort out in the sideslip response than it is in the roll response (see "Aircraft Dynamics and Automatic Control").

VERIFICATION

To determine sideslip excursions small step roll control commands are called for in straight flight and in steady turns. To determine k, rolls are to be made with rudder use according to 5.2.3.5. The verification of 5.2.3.2.5, however, calls for rudder-pedals-fixed rolls from turns. With powered controls, generally, there will be no difference here between controls-fixed and controls-free.

Note from 5.2.3.2 that $\Delta\beta_{max}$ is the total excursion during the stated time period. Thus if β reverses during that time, the sum of right and left excursion magnitudes is called for.

The flight testing to obtain $\Delta\beta_{max}$ and $\phi_{t_{command}}$ shall cover the range of operational altitudes and service speeds. As with roll rate oscillations (5.2.3.2.1 and 5.2.3.2.3), the critical flight conditions for compliance with this requirement should in general become apparent during the roll performance testing of 5.2.3.5. The most important flight conditions for compliance demonstration are those with low $|\phi/\beta|_d$, less than 6.

5.2.3.2.6 Yaw response to large roll control commands.
With yaw controls free and, in initial steady turns, fixed at the deflection for zero sideslip in the turn, following a step roll control command held fixed until the bank angle has changed at least 90º, the ratio of the sideslip increment, $\Delta\beta_{max}$, to the parameter k (see 5.2.3.2.5) shall be less than the values specified in table LII.

TABLE LII. Maximum sideslip excursions for large roll control commands.

Level	Flight Phase Category	$\Delta\beta_{max}/k$	
		Adverse Sideslip (i.e., right roll command causes right sideslip)	Proverse Sideslip (i.e., left roll command causes right sideslip)
1	A		
	B		
	C		
2	A		
	B		
	C		

REQUIREMENT RATIONALE

This requirement is intended to insure that any yaw control needed to coordinate turn entries and recoveries/reversals is not objectionable to the pilot. This requirement uses the value of sideslip angle that accompanies roll control inputs as the measure of acceptance.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.2.4
MIL-STD-1797: 4.6.2.

Recommended values for TABLE LII:

Level	Flight Phase Category	$\Delta\beta_{max}/k$	
		Adverse Sideslip (i.e., right roll command causes right sideslip)	Proverse Sideslip (i.e., left roll command causes right sideslip)
1	A	6º	2º
	B and C	10º	3º
2	All	15º	4º

REQUIREMENT LESSONS LEARNED

VERIFICATION

5.2.3.2.7 Coordination in turn entry and exit.
The yaw control crossfeed necessary to maintain zero sideslip shall result in a ratio of initial yawing acceleration to initial rolling acceleration for a step roll control input, $(\dot{r}_0/\dot{p}_0)_{step}$, within __(1)__ for Levels 1 and 2. In addition, for values of $|\dot{r}_0/\dot{p}_0|_{step}$ less than 0.07, the crossfeed parameter $\delta'_{rp}(3)$ shall be within the limits of table LIII.

TABLE LIII. Limits on $\delta'_{rp}(3)$ for $|\dot{r}_0/\dot{p}_0|_{step} < 0.07$.

Level	Adverse Yaw	Proverse Yaw
1		
2		

REQUIREMENT RATIONALE

This requirement is intended to insure that any yaw control needed to coordinate turn entries and recoveries/reversals is not objectionable to the pilot. This requirement is offered as an alternative to bounds on sideslip excursions, $\Delta\beta_{max}/k$ versus ψ_β (5.2.3.2.5 and 5.2.3.2.6). It places limits directly on the magnitude and shaping of rudder pedal inputs required for turn coordination as the measure of acceptance. Hence it is intended as analytical guidance.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.6.2.

Recommended value for Blank (1): "the boundaries of figure 142"

Recommended values for table LIII:

Level	Adverse Yaw	Proverse Yaw
1	-0.39	0.11
2	-1.15	0.78

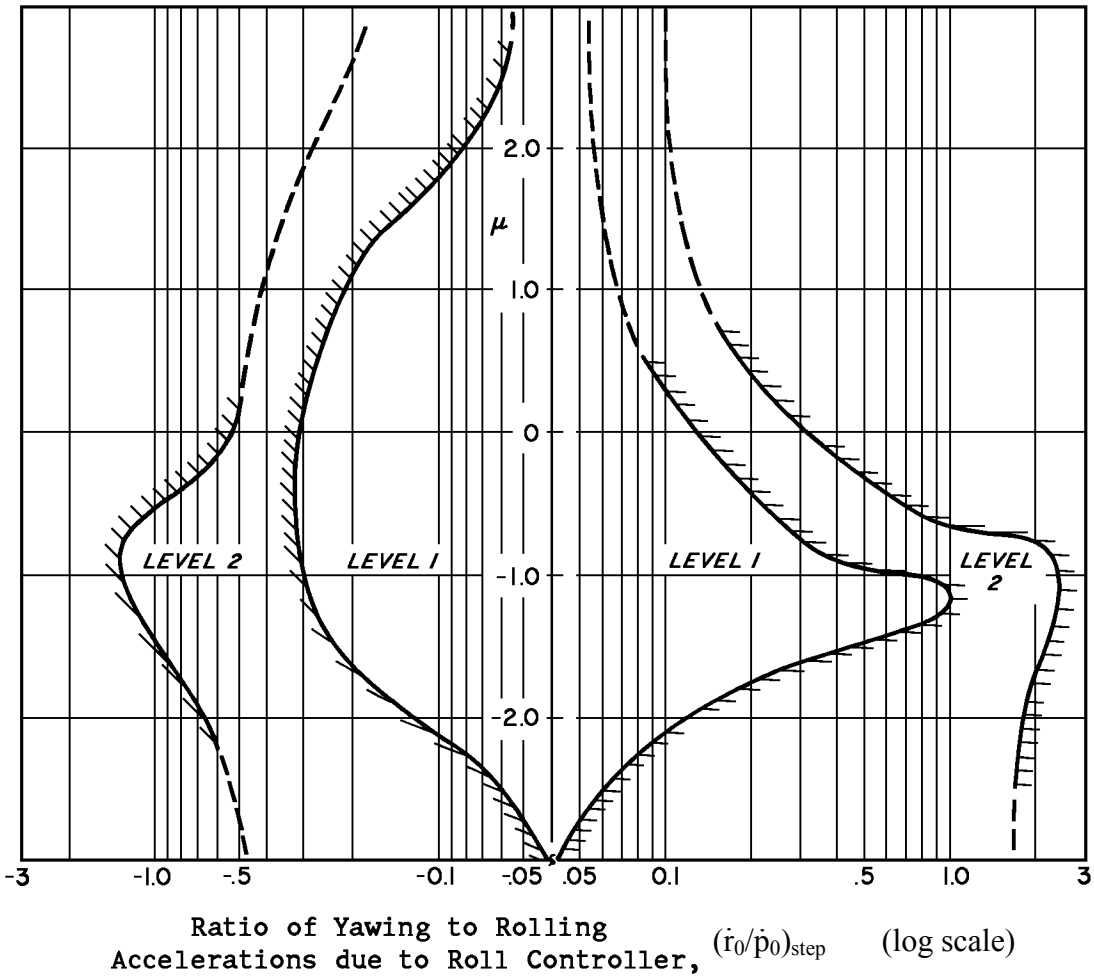


FIGURE 142. Crossfeed parameter boundaries.

REQUIREMENT LESSONS LEARNED

The ability to make precise changes in aircraft heading is a key factor in pilot evaluation of lateral-directional handling qualities. With the other qualities (e.g., adequate roll response, yaw

frequency/damping, etc.) assumed to be good, any deficiencies in heading control are directly traceable to excitation of the Dutch roll mode due to roll-yaw cross-coupling effects. It is a commonly accepted piloting technique to reduce these excursions by appropriate use of the aileron and rudder, usually referred to as "coordinating the turn." The problem is that existing criteria for heading control ($\Delta\beta_{max}/k$, or AFFDL-TR-72-41, FAA-RD-70-61, or WP-189-3) are based on aileron-only parameters, and the effects of rudder control are only indirectly apparent as they may have influenced individual pilot ratings. That these criteria are not satisfactory is shown in NASA CR-2017, where several configurations that violated boundaries based on aileron-only parameters were given good to excellent pilot ratings. The approach taken here is that for an otherwise acceptable aircraft the aileron-rudder shaping necessary to coordinate the turn is a dominant factor in pilot evaluation of heading control. In this regard it is important to recognize that heading control is basically an outer loop and cannot be satisfactory if the inner bank angle loop is unsatisfactory. Table LIV contains a set of requirements intended to serve as a checklist for good roll control. All derivatives that follow are in the stability axis system.

TABLE LIV. Ground rules for application of rating data to heading control criteria.

1) T_R meets the Level 1 requirements of 5.2.3.1.1
2) ω_d meets the Level 1 requirements of 5.2.3.1.2
3) ζ_d meets the Level 1 requirements of 5.2.3.1.2
4) $\lvert\phi/\beta\rvert_d < 1.5$ when turbulence is a factor and $\left\lvert N'_{\delta_{as}} / L'_{\delta_{as}} \right\rvert = \left\lvert \dot{r}_0 / \dot{p}_0 \right\rvert_{step} > 0.03$ or
5) Meets L'_β versus ω_d boundaries when $\left\lvert \dot{r}_0 / \dot{p}_0 \right\rvert_{step} \le 0.03$ (figure 143)
6) Meets Level 2 p_{osc}/p_{av} according to 5.2.3.2.1

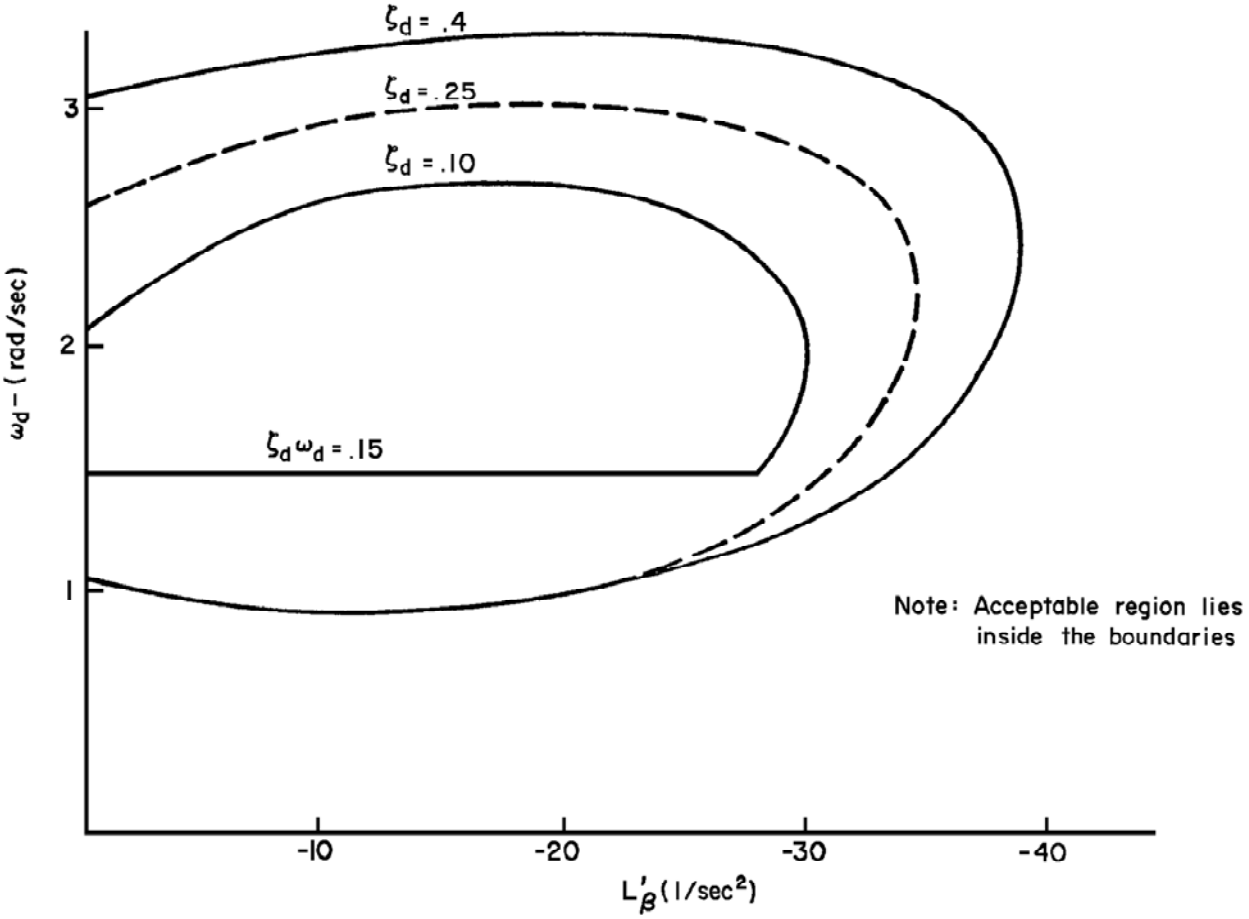


FIGURE 143. Pilot rating boundaries for acceptable roll control in turbulence with $N'_{\delta_{as}} / L'_{\delta_{as}} \leq 0.03$ (from Princeton Univ Rpt 797).

A. Analysis and Basic Concept

In general, coordinated flight implies minimum yaw coupling in roll entries and exits, which can be quantified in many ways, e.g.: a) zero sideslip angle ($\beta = 0$); b) zero lateral acceleration at the c.g.; c) turn rate consistent with bank angle and speed ($r = g \phi/U_0$); and d) zero lateral acceleration at the cockpit (ball in the middle).

Conditions a) through c) are equivalent when the side forces due to lateral stick, $Y_{\delta_{as}}$, and rudder pedal, $Y_{\delta_{rp}}$, are very small, which is usually the case. The fourth turn coordination criterion is complicated by pilot location effects which, however, appear to be more associated with ride qualities than with heading control itself (NASA CR-2017). Based on these considerations it appears that sideslip angle is an appropriate indicator of turn coordination. It has been suggested that pilots are taught to center the ball in turns and therefore a_y would be the more correct parameter. However, the real objective is to keep sideslip near zero so that the aircraft tracks bank angle. In fact, when turn coordination is critical, as on the AV-8 Harrier, a yaw string is

used to display β to the pilot. Also, glider pilots use a yaw string because turn coordination is a critical factor in these aircraft. Generally, sideslip bias at the nose due to steady yaw rate is small. Therefore, the proponents of β feel that β and not a_y is the appropriate parameter. For most flight conditions—even for most aircraft—the difference between a_y and β is very small. Accordingly, the following formulation undertakes to identify the parameters that govern the aileron-rudder shaping required to maintain coordinated flight as defined by zero sideslip angle (β = 0).

With an aileron stick or wheel crossfeed to the rudder pedals, Y_{CF}, the rudder pedal deflection produced is

$$\delta_{rp} = Y_{CF}\,\delta_{as} \qquad (1)$$

where δ_{as} is the lateral stick (or wheel) deflection at the pilot's grip. For the assumed ideal (zero sideslip) coordination, in terms of the β/δ_{as} and β/δ_{rp} transfer functions:

$$\beta = \left(\frac{N^{\beta}_{\delta_{as}}}{\Delta} + Y_{CF}\,\frac{N^{\beta}_{\delta_{rp}}}{\Delta} \right) \delta_{as} = 0 \qquad (2)$$

At large bank angles and high roll rates the equations of motion become nonlinear, with possibilities that trigonometric functions or higher-order terms will be of significant size. Nevertheless, the character of the motion generally will be shown by the linear analysis. $N^{\beta}_{\delta_{as}}$ and $N^{\beta}_{\delta_{rp}}$ are the numerators of the indicated transfer functions and Δ the denominator, as in "Aircraft Dynamics and Automatic Control." Therefore the ideal crossfeed is:

$$Y_{CF} = \frac{\delta_{rp}}{\delta_{as}} = -\frac{N^{\beta}_{\delta_{as}}}{N^{\beta}_{\delta_{rp}}} \qquad (3)$$

For augmented aircraft these numerators are of high order and cannot be generalized. However, aircraft with complex augmentation systems represented by higher-order systems (HOS) tend to respond to pilot inputs in a fashion similar to conventional unaugmented aircraft or low-order systems (LOS). In fact, experience with longitudinal pitch dynamics (see 5.2.2.1.1.2 and "Handling Qualities of Aircraft with Stability and Control Augmentation Systems – A Fundamental Approach") has shown that a HOS which cannot be fit to an LOS form generally is unsatisfactory to the human pilot.

Based on the approximate factors for conventional aircraft obtained from "Aircraft Dynamics and Automatic Control", the appropriate LOS form for Y_{CF} is, in its usual factored form, with time constants T_{β_i}:

$$Y_{CF} = \frac{N'_{\delta_{as}} \left[s + A_{as}\,(g/U_0) \right] \left[s + (1/T_{\beta_{as}}) \right]}{Y^{*}_{\delta_{rp}} \left[s + A_{rp}\,(g/U_0) \right] \left[s + (1/T_{\beta_{rp}}) \right] \left[s - \left(N'_{\delta_{rp}} / Y^{*}_{\delta_{rp}} \right) \right]} \qquad (4)$$

where

$$Y^{*}_{\delta_{rp}} = \left(\partial Y / \partial \delta_{rp}\right) / \left(m U_0\right) = \rho\, g\, U_0\, C_{y_{\delta_{rp}}} / \left(2\, W / S\right)$$

$$L'_j = \frac{L_j + N_j\, I_{xz} / I_{xx}}{1 - I^2_{xz} / \left(I_{xx}\, I_{zz}\right)} \qquad N'_j = \frac{N_j + L_j\, I_{xz} / I_{zz}}{1 - I^2_{xz} / \left(I_{xx}\, I_{zz}\right)}$$

$$A_i = \frac{L'_r - \left(L'_{\delta_i} / N'_{\delta_i}\right) N'_r}{L'_p - \left(L'_{\delta_i} / N'_{\delta_i}\right)\left[N'_p - \left(g / U_0\right)\right]}$$

$$1 / T_{\beta_i} = -L'_p + \left(L'_{\delta_i} / N'_{\delta_i}\right)\left[N'_p - \left(g / U_0\right)\right]$$

j = p, r, β, or δ_i, and i = as (aileron stick) or rp (rudder pedal). For the frequency range of interest, i.e., excluding both low and high frequencies [$A_i \left(g / U_0\right) << \omega << N'_{\delta_{rp}} / Y^{*}_{\delta_{rp}}$], Y_{CF} generally is of first order:

$$Y_{CF} = \frac{N'_{\delta_{as}} \left[s + 1 / T_{\beta_{as}}\right]}{N'_{\delta_{rp}} \left[s + 1 / T_{\beta_{rp}}\right]} \qquad (5)\ .$$

The approximation should hold in a great many cases. Whether or not this is a good approximation in any given case, it seems reasonable that pilots would not like a crossfeed more complicated than this. Now, the rudder sensitivity can be optimized separately and does not usually represent a basic airframe limitation, so it is appropriate to remove that from consideration. But even though the magnitude of δ_{rp} required for coordinating a roll command can be dispensed with thus, the size of the uncoordinated yaw response is of course a pilot concern. A suitable "gain" parameter, then, is $\left(\dot{r}_0 / \dot{p}_0\right)_{step} = N'_{\delta_{as}} / L'_{\delta_{as}}$. By this immediate cue a pilot can judge his need to add crossfeed for a step roll command. Accordingly, the LOS crossfeed parameter, Y'_{CF}, is given as:

$$Y'_{CF} = \frac{N'_{\delta_{as}} \left[s + 1 / T_{\beta_{as}}\right]}{L'_{\delta_{as}} \left[s + 1 / T_{\beta_{rp}}\right]} \qquad (6)$$

Equation 6 indicates that the aileron-to-rudder shaping required to maintain coordinated flight (β = 0) is directly related to the separation between the lateral stick (or wheel) and rudder pedal sideslip zeros.

As a basis for a direct correlation with pilot opinion, a "rudder shaping parameter," μ, is arbitrarily defined as the separation between $1 / T_{\beta_{rp}}$ and $1 / T_{\beta_{as}}$, normalized by $1 / T_{\beta_{rp}}$; i.e.,

$$\mu = \frac{\left(1/T_{\beta_{as}}\right) - \left(1/T_{\beta_{rp}}\right)}{1/T_{\beta_{rp}}}$$

which simplifies to

$$\mu = \left(T_{\beta_{rp}} / T_{\beta_{as}}\right) - 1 \qquad (7)\ .$$

Then, from either equation 5 or equation 6, $(1 + \mu)$ is the ratio of steady δ_{rp} to the initial δ_{rp}. The frequency response characteristics of Y'_{CF}, equation 6, as a function of the sign of $N'_{\delta_{as}} / L'_{\delta_{as}}$ and the value of μ, are shown on figure 144. The shaping of the rudder response is determined by μ. These parameters are summarized in terms of their analytical and pilot-centered functions in table LV.

The parameters $(\dot{r}_0 / \dot{p}_0)_{step}$ or $N'_{\delta_{as}} / L'_{\delta_{as}}$ and μ are a natural choice for correlation of heading-control pilot rating data since they, plus the value of $T_{\beta_{rp}}$, completely define the approximate aileron-to-rudder crossfeed necessary for turn coordination ($T_{\beta_{rp}}$ often is approximately the same value as T_R). Such an ideal crossfeed is difficult to isolate with simple flight test procedures, but is nevertheless considered a viable correlation concept because the rationale is straightforward and the parameters can be extracted from flight via parameter identification techniques. Correlation with available data is shown subsequently.

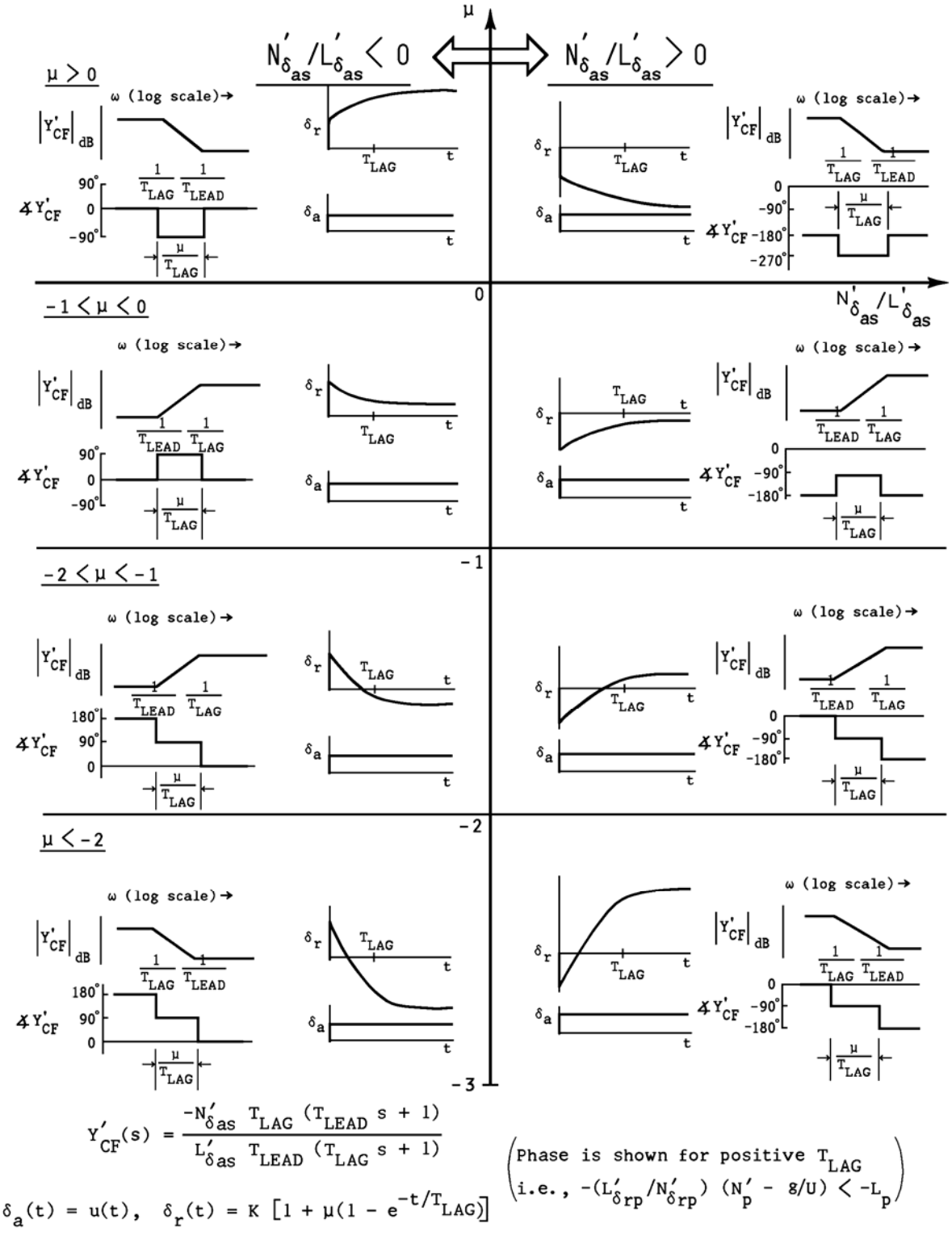


FIGURE 144. Bode asymptotes and time response of crossfeed.

TABLE LV. Parameters defining the LOS representation of the aileron-rudder crossfeed.

Parameter	Analytical Function	Pilot-Centered Function
μ	Defines shape of Y_{CF}	Determines complexity of rudder activity necessary for ideally coordinated turns. Also, defines phasing of heading response when rudder is not used.
$(\dot{r}_0/\dot{p}_0)_{step}$ or $N'_{\delta_{as}}/L'_{\delta_{as}}$	Defines magnitude of Y_{CF}	Determines high-frequency yawing induced by aileron inputs.

The rudder-pedal time history required to coordinate a unit step wheel or stick input is:

$$\delta_{rp}(t) = \left[1 + \mu\left(1 - e^{-t/T_{\beta_{rp}}}\right)\right] N'_{\delta_{as}} / N'_{\delta_{rp}} \qquad (8)$$

so

$$\delta_{rp}(t\rightarrow\infty)/\delta_{rp}(t\rightarrow 0+) = 1 + \mu \qquad (9)$$

Note that $\delta_{rp}(t)$ refers to the rudder pedal motion (other surface inputs are flight control system crossfeeds and SAS). The criterion value of t is properly set by the lower limit on the frequency range of interest for piloted heading control. The simulation experience of NASA CR-239 indicated that a minimum heading crossover frequency of 1/3 rad/sec was necessary for desirable handling qualities. Therefore, a time of 3 sec rather than infinity was selected as pertinent to a pilot-centered characterization of crossfeed properties. (While this makes little difference for the approximation to μ, it may be significant for the exact solution.) Recognizing further (Equation 4) that $T_{\beta_{rp}} \doteq -1/L_p$ approximately equals the roll mode time constant, T_R, and that the latter must generally be less than 1.0 to 1.4 sec for acceptable roll control (5.2.3.1.1) sets the following limits on the exponential in Equation 8:

AIRCRAFT	T_R	e^{-3/T_R}
Small, light or highly maneuverable	< 1.0 sec	< 0.049 sec
Medium to heavy weight, low to medium maneuverability	< 1.4 sec	< 0.117 sec

Accordingly, Equation 9 reduces within a maximum error of 5-10 percent, depending on Aircraft Class to

$$\mu \doteq \delta_{rp}(3) - 1 \qquad (10)$$

for a $\delta_{rp}(0)$ of 1. This simple relationship was used to compute μ for the pilot rating correlations shown under Supporting Data. It should also be noted that, since the high-frequency gain is set

to unity by normalizing with respect to $\delta_{rp}(0)$, the normalized rudder parameter, $\delta_{rp}(3)$, may be calculated from Y_{CF} (Equation 3, 4, or 5) or Y'_{CF} (Equation 6).

However, before this simple formula can be applied it is necessary to avoid the high-frequency responses that occur due to pairs of roots that frequently occur with complex SAS installations having associated higher-order β numerators. For example, a simple wash-out yaw rate feedback and a first-order lagged aileron/rudder crossfeed results in seventh-order β numerators of otherwise unaugmented aircraft. Most of the zeros of these polynomials occur at very high frequency, having negligible effect on the dynamics near the pilot's crossover frequency, and therefore shall not be accounted for in the shaping function μ. The standard procedure utilized to compute the values of μ was to eliminate all roots of the β parameters above values of 6 rad/sec in pairs, i.e., keeping their order relative to each other the same (e.g., a third over fourth order would be reduced to a second over third order, etc.). Roots above 6 rad/sec which do not occur in pairs are left unmodified. The point of deleting the high-frequency root pairs is just to find the rudder/aileron crossfeed ratio that the pilot should apply.

The following example illustrates a typical computation of μ and the effect of removing the high-frequency roots from Equation 2. The aileron/rudder crossfeed for one of the AFFDL-TR-69-41 configurations used in the pilot rating correlations is given as:

$$\frac{\delta_{rp}}{\delta_{as}} = \frac{0.19\,(s - 0.102)(s - 0.922)(s + 605.2)}{(s - 0.057)(s + 5.6)(s + 109.9)} \tag{11}$$

As discussed above, all roots above 6 rad/sec are removed in pairs and the high-frequency gain (0.19) is set to unity, resulting in the following equation:

$$\frac{\delta_{rp}}{\delta_{as}} = \frac{(s - 0.102)(s - 0.922)}{(s - 0.057)(s + 5.6)} \tag{12}$$

The rudder time responses to a unit wheel input for Equations 11 and 12 are plotted on figure 145. Removal of the high-frequency roots is seen to replace the initial rapid rudder reversal with a unity initial condition. These responses are essentially equivalent to the pilot, who sees the necessity to use immediate rudder with aileron inputs (which must be removed ½ sec later). The value of μ corresponding to this response is $\delta_{rp}(3) - 1 = -1.17$. In this case, the low-frequency pole and zero could also have apparently been removed with no ill effect.

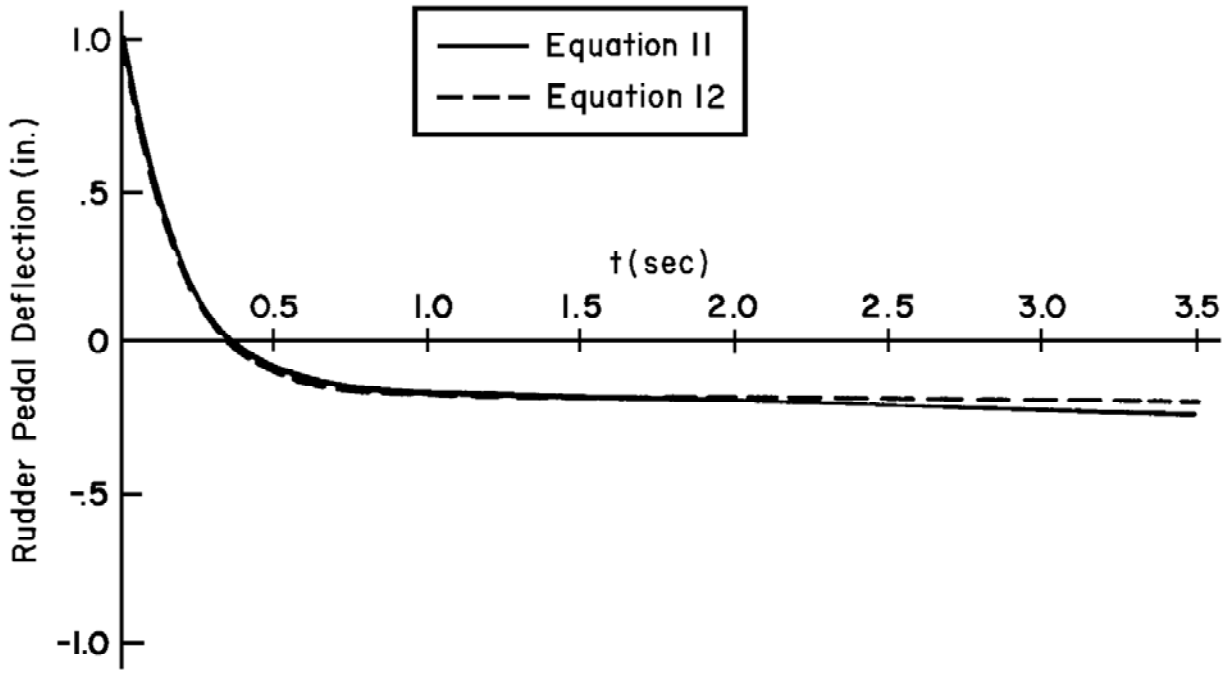


FIGURE 145. Effect of removing high-frequency roots from β numerators.

B. Physical Interpretation

A physical interpretation is that δ_{rp} is the sum of an initial amount to counter $N'_{\delta_{as}}\ \delta_{as}$ and a variable amount proportional to $N'_p\ p$. The possibilities are enumerated in table LVI.

TABLE LVI. Physical interpretation of μ.

Value of Rudder Shaping Parameter	Roll-Yaw Cross-Coupling Characteristics
$\mu > 0$	$N'_{\delta_{as}}$ and N'_p are additive, indicating that the cross-coupling effects increase with time after an aileron input.
$\mu = 0$	$N'_p = g/U_0$, indicating that all roll-yaw cross-coupling is due to $N'_{\delta_{as}}$. The aileron-rudder crossfeed is therefore a pure gain.
$-1 < \mu < 0$	$N'_{\delta_{as}}$ and N'_p are opposing. Initial cross-coupling induced by $N'_{\delta_{as}}$ is reduced by N'_p as the roll rate builds up. Exact cancellation takes place when $\mu = -1$, resulting from a zero rudder requirement for steady rolling.
$\mu < -1$	Low-frequency and high-frequency cross-coupling effects are of opposite sign, indicating a need for rudder reversals for coordination. If rudder is not used, the nose will appear to oscillate during turn entry and exit.

The iso-opinion lines on figure 142 indicate that some values of the rudder shaping parameter, μ, are more desirable than others in that they are less sensitive to an increase in aileron yaw. The following observations help to explain this trend in terms of pilot-centered considerations:

1. Moderately high proverse (positive) $N'_{\delta_{as}}$ is acceptable in the region where $\mu \doteq -1$. Physically, this corresponds to a sudden initial heading response in the direction of turn followed by decreasing rudder requirements. (Required steady-state rudder is zero when μ = -1, see figure 144.) It is felt that the pilots are accepting the initial proverse yaw as a heading lead and are not attempting to use cross-control rudder.

2. The allowable values of proverse $N'_{\delta_{as}}$ decrease rapidly as μ becomes more negative than -1. Physically this corresponds to an increase in the need for low-frequency rudder activity (see figure 144). A requirement for cross-control is particularly objectionable.

3. The pilot ratings are less sensitive to the required rudder shaping when $N'_{\delta_{as}}$ is negative [adverse yaw, negative $(\dot{r}_0 / \dot{p}_0)_{step}$]. Recall that adverse yaw is consistent with conventional piloting technique (rudder with the turn to augment roll into the turn).

C. $(\dot{r}_0 / \dot{p}_0)_{step}$ Near Zero

Control cross-coupling effects are obviously not a factor when $\left|N'_{\delta_{as}} / L'_{\delta_{as}}\right| \left(\left|\dot{r}_0 / \dot{p}_0\right|_{step}\right)$ is small. This may occur when the basic control cross-coupling is negligible or with augmentation systems that result in ideal crossfeed, Y_{CF}, having denominators of higher-order dynamics than numerators (e.g., the augmented $N'_{\delta_{as}}$ is zero). When $\left|N'_{\delta_{as}} / L'_{\delta_{as}}\right|$ is identically zero, for unaugmented conventional aircraft the required aileron-rudder crossfeed takes the form

$$Y'_{CF} = \frac{N'_{\delta_{rp}}}{L'_{\delta_{as}}} Y_{CF} = \frac{-\left(N'_p - g/U_0\right)\left[s + \left(g/U_0\right) N'_r / \left(N'_p - g/U_0\right)\right]}{s\left(s + 1/T_{\beta_{rp}}\right)} \qquad (13)$$

The Bode asymptotes are shown on figure 146. The rudder magnitude required to coordinate mid-frequency and high-frequency aileron (wheel) inputs is seen to be dependent on the roll cross-coupling, $g/U_0 - N'_p$, whereas low-frequency rudder requirements are dependent on N'_r. The required rudder shaping has the characteristic of a rate system (ramp δ_{rp} to step δ_{as}) at low and high frequency. Accordingly, aileron-rudder shaping per se is not the essence of the problem, which reduces instead to concern with the general magnitude of the rudder crossfeed.

From figure 146 it is seen that $g/U_0 - N'_p$ provides a good measure of such magnitude; and, in fact, correlation of pilot rating data (for $\left|N'_{\delta_{as}} / L'_{\delta_{as}}\right| < 0.03$) with $g/U_0 - N'_p$ is quite good.

However, difficulties associated with estimating an effective g/U_0 - N'_p for augmented airframes present practical problems which make this parameter somewhat unattractive. Also, for configurations with $1/T_{\beta_{as}}$ close to $1/T_{\beta_{rp}}$ the effects due to N'_r (see figure 146) can be important. A more general approach is to compute a time history based on a unit step aileron stick input into Y_{CF}. Physically, this represents the required rudder magnitude for coordination of a unit step aileron control input; that is (from Equation 5 and 6):

$$\delta'_{rp} = Y'_{CF}\,\delta_{as} = \frac{N'_{\delta_{rp}}}{L'_{\delta_{as}}}\,\delta_{rp} \qquad (14)$$

Utilizing the same response time considerations as in the computation of μ, $\delta'_{rp}(3)$ is suggested as the correlating parameter when $\left|N'_{\delta_{as}}/L'_{\delta_{as}}\right|$ is small or when the denominator of Y_{CF} is of higher order than the numerator. What specifically constitutes a small value of $\left|N'_{\delta_{as}}/L'_{\delta_{as}}\right|$ has proven to be somewhat difficult to quantify. Reasonably good correlations were found by plotting the $\left|N'_{\delta_{as}}/L'_{\delta_{as}}\right| < 0.03$ pilot ratings versus $\delta'_{rp}(3)$ as shown on figure 147. More recent experience in utilizing the parameter has revealed that $\left|N'_{\delta_{as}}/L'_{\delta_{as}}\right| < 0.07$ results in better correlations. When $\left|N'_{\delta_{as}}/L'_{\delta_{as}}\right|$ is between 0.03 and 0.07, both figure 142 and table LVI shall be checked and the most conservative result utilized.

In summary, $\delta'_{rp}(3)$ is calculated by obtaining the response of a unit step input into the transfer function Y_{CF} (Equation 3) at t = 3 sec and multiplying the result by $N'_{\delta_{as}}/L'_{\delta_{as}}$.

D. Complex Rudder Shaping

As discussed earlier, reasonable fits of the HOS to the LOS form implicit in Equation 9 presume that the required aileron-to-rudder shaping is at least monotonic (or with at most one reversal) in the region of piloted control. This assumes that if the required rudder coordination to a step aileron input is of significant size and non-monotonic in the region of control, pilot opinion will be poor. Since the authors of this standard have been unable to find any configurations having such a non-monotonic shape which have been tested for pilot opinion of heading control, the mismatch effects cannot yet be quantified.

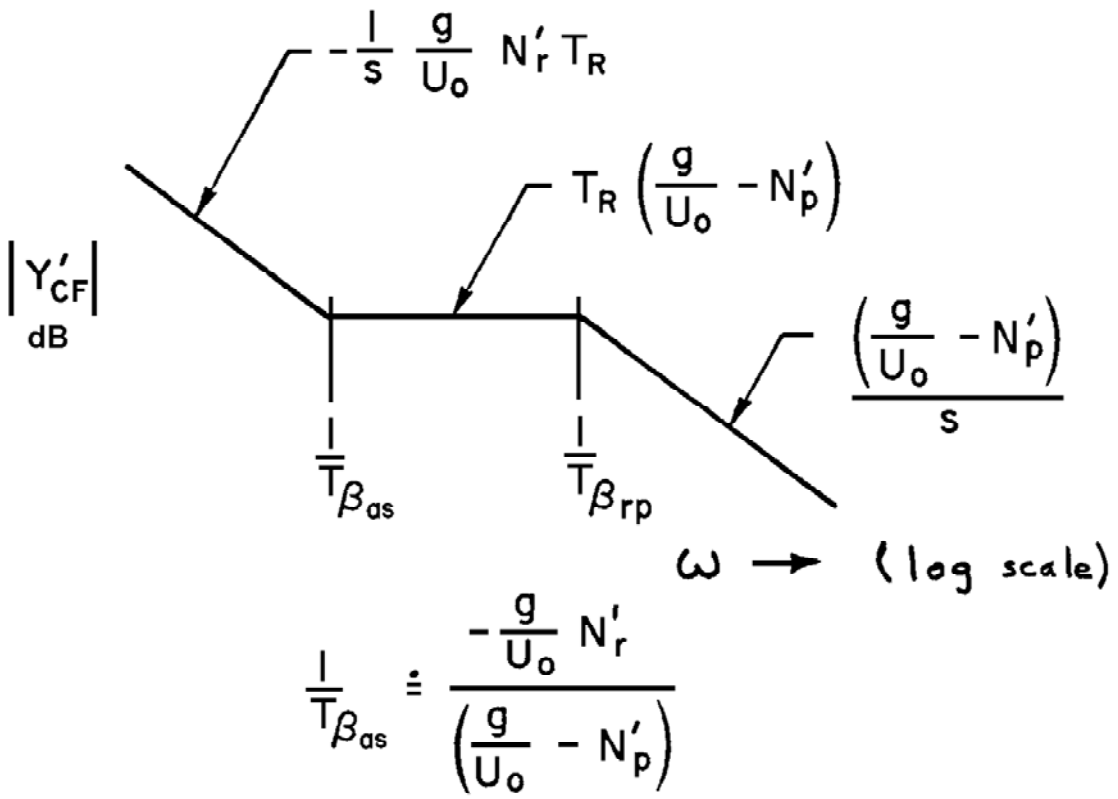


FIGURE 146. Required crossfeed for $N'_{\delta_{as}} = 0$.

E. Supporting Data

A summary of the data sources considered is given in table LVII. Each of the data points found to be applicable to heading control (i.e., met the ground rules) is plotted and faired on a logarithmic grid of $(\dot{r}_0 / \dot{p}_0)_{step}$ versus μ on figure 148. Only in-flight and moving-base simulator data were considered. With the exception of one or two points, the data from all the sources in table LVII correlate quite nicely with the criterion (figure 148). The few points that do not fit are rated better than the other data in the same region.

It is significant that the pilot rating correlations are not dependent on the type of aircraft and in fact are shown to be valid for vehicles ranging from light aircraft to fighter, STOL, and Space Shuttle configurations. This result indicates that good heading control characteristics are dependent on a fundamental aspect of pilot technique (aileron-rudder coordination) and that such factors as aircraft size, weight, approach speed, etc. can be neglected for all practical purposes. It is felt that the invariance of ratings with aircraft configuration is related to the pilot's ability to adapt to different situations and to rate accordingly. Finally, the excellent correlations of pilot ratings with the aileron-rudder crossfeed characteristics indicate that the required rudder coordination is indeed a dominant factor in pilot evaluation of heading control.

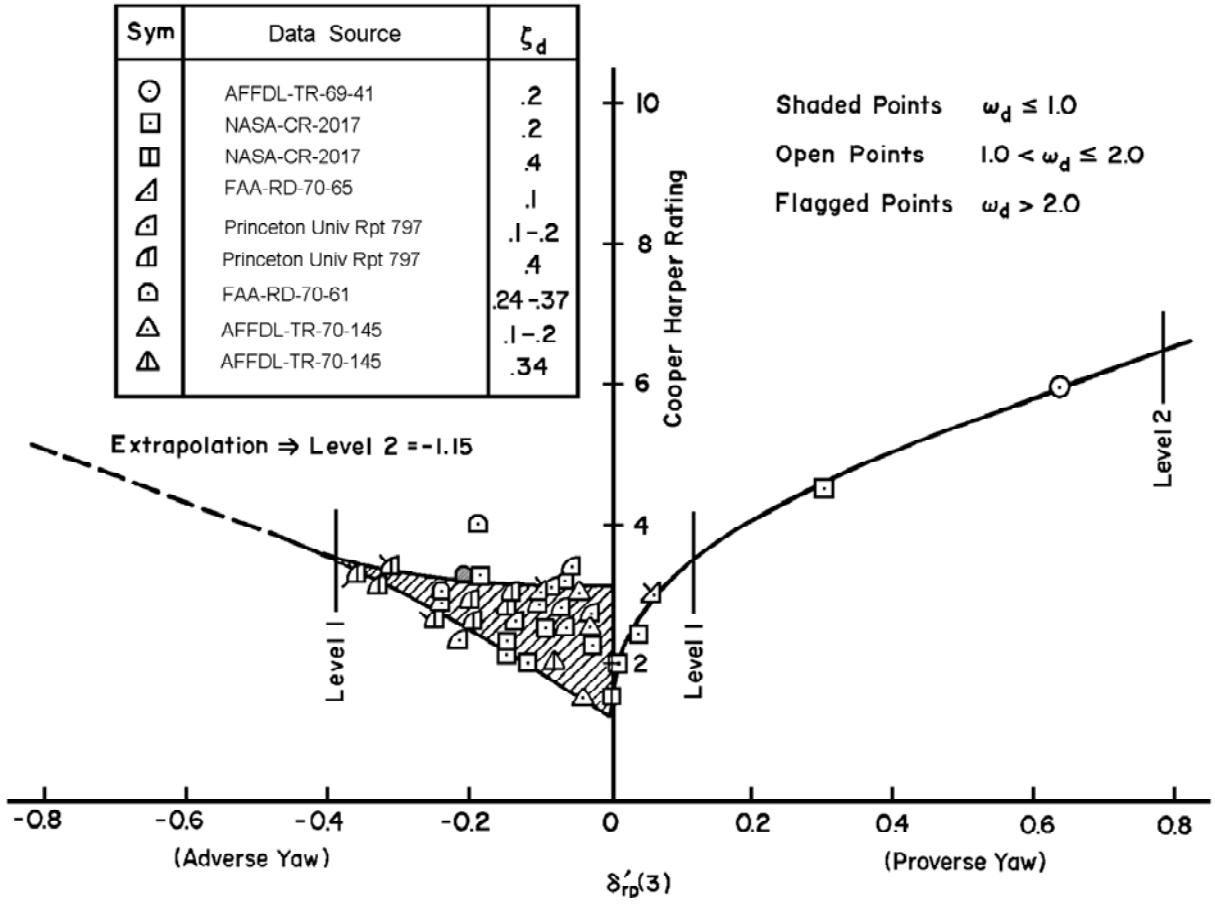


FIGURE 147. Pilot rating correlations when $\left|N'_{\delta_{as}} / L'_{\delta_{as}}\right|$ is small.

The rudder shaping parameter is attractive as a heading control criterion because the handling quality boundaries are easily interpreted in terms of pilot-centered considerations. Its shortcoming is centered about determining parameter values.

A review of data sources for figure 148 indicates that the requirement as devised is applicable primarily to low-speed flight, and especially to Category C (approach and landing) Flight Phases. Large $N'_{\delta_{as}}$ or N'_p, however, are unlikely at high speed. Further work is necessary in this area to determine any refinements for the other Categories. However, since low-speed flight with a high-gain lateral task defines the most extreme condition for control coordination, the requirement of figure 142 covers the worst cases.

Figure 149 (from Systems Technology, Inc., TR-1090-1) compares the rudder shaping parameter for several aircraft with the figure 142 requirements. Available pilot rating data for the F-111 with and without adverse yaw compensator (AYC) support the boundaries. The pilot ratings are from Systems Technology, Inc., TR-199-1, where it is stated that "the F-111B without the adverse yaw compensator lies in an unacceptable region….[The] rudder sequencing criteria…for the F-111B with and without AYC…are in agreement with the actual ratings it received."

TABLE LVII. Summary of current data.

Type of Aircraft Simulated	Description of Simulator	Reference	Total Number of Data Points	Number of Points Meeting Ground Rules
Executive jet and military Class II	Variable stability NT-33	AFFDL-TR-70-145	84	16
STOL	Variable stability helicopter	AFFDL-TR-69-41	109	30
General aviation light aircraft	Variable stability Navion	FAA-RD-70-65 Part 2	26	6
Space shuttle vehicle	6DOF moving-base	NASA CR-2017	52	52
STOL	3DOF moving-base	FAA-RD-70-61	8	7

VERIFICATION

Clearly, application of the rudder shaping criterion can only be through analysis, necessitating a good analytical model of the numerators of the augmented aircraft. With these, application of the criterion is straightforward. Values derived from wind tunnel tests and estimates shall be updated and verified by flight test parameter identification wherever roll-induced yawing is a problem.

The rules for application of this shaping criterion are summarized as follow, recalling that $(\dot{r}_0 / \dot{p}_0)_{step} = N'_{\delta_{as}} / L'_{\delta_{as}}$:

1. If $\left| N'_{\delta_{as}} / L'_{\delta_{as}} \right| < 0.03$, skip to Step 6.

2. Formulate Y_{CF} by taking the ratio of the β/δ_{as} and the β/δ_{rp} transfer functions. For augmented aircraft the transfer function must include the effects of augmentation: $Y_{CF} = \left(N^{\beta}_{\delta_{as}} / N^{\beta}_{\delta_{rp}} \right)_{aug}$.

3. Remove all roots greater than 6 rad/sec in pairs, keeping the relative numerator and denominator order of Y_{CF} constant. Roots above 6 rad/sec that do not occur in pairs are left unmodified. Set the high-frequency gain of Y_{CF} equal to unity (as in Equation 12).

4. Calculate $\delta_{rp}(3)$ from the time response of Y_{CF} (as modified by Step 3) to a unit step input, i.e., $\delta_{rp}(3) = £^{-1}\{Y_{CF}(s)/s\}$ evaluated at t = 3 sec.

5. Calculate μ as: $\mu = \delta_{rp}(3) - 1$ and plot on figure 142.

6. If $\left|N'_{\delta_{as}} / L'_{\delta_{as}}\right| \leq 0.07$, calculate the normalized rudder required, as follows:

   a. Calculate the magnitude at t = 3 sec of the time response of Y_{CF} (from Step 2) to a unit step input.

   b. Multiply the result by $N'_{\delta_{rp}} / L'_{\delta_{as}}$, i.e., $\delta'_{rp}(3) = Y_{CF}(3)\ N'_{\delta_{rp}} / L'_{\delta_{as}}$.

   c. Compare $\delta'_{rp}(3)$ with table LIII.

7. If $0.03 \leq \left|N'_{\delta_{as}} / L'_{\delta_{as}}\right| \leq 0.07$, utilize the more conservative result from Steps 5 and 6.

8. If the configuration does not meet the requirements, see figure 144 and table LVI to determine the type of expected piloting problems.

In the end, the transfer functions shall be identified from flight data.

The most important flight conditions for compliance demonstration are those with low $|\phi/\beta|_d$, less than 6.

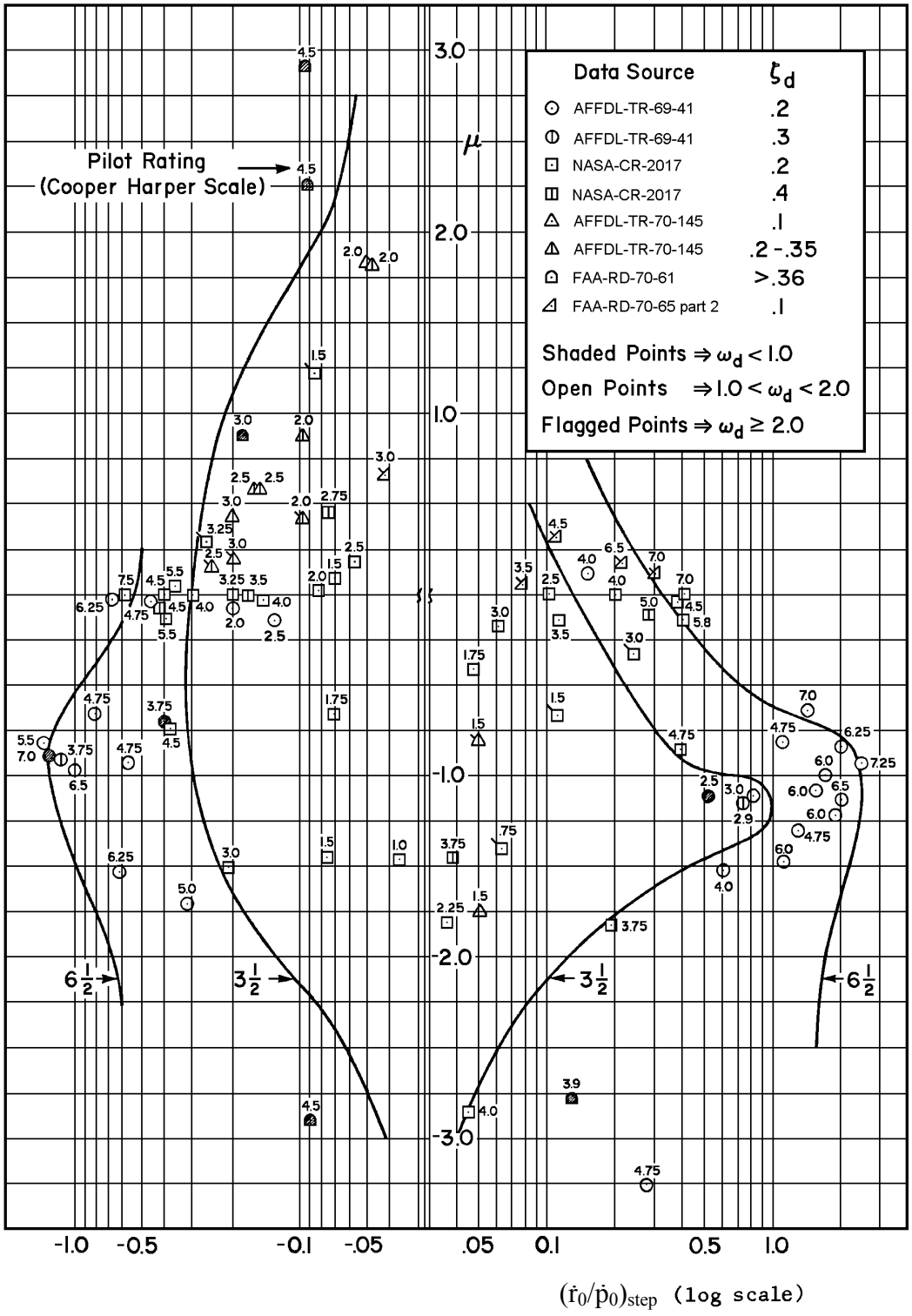


FIGURE 148. Pilot rating correlation with crossfeed parameters.

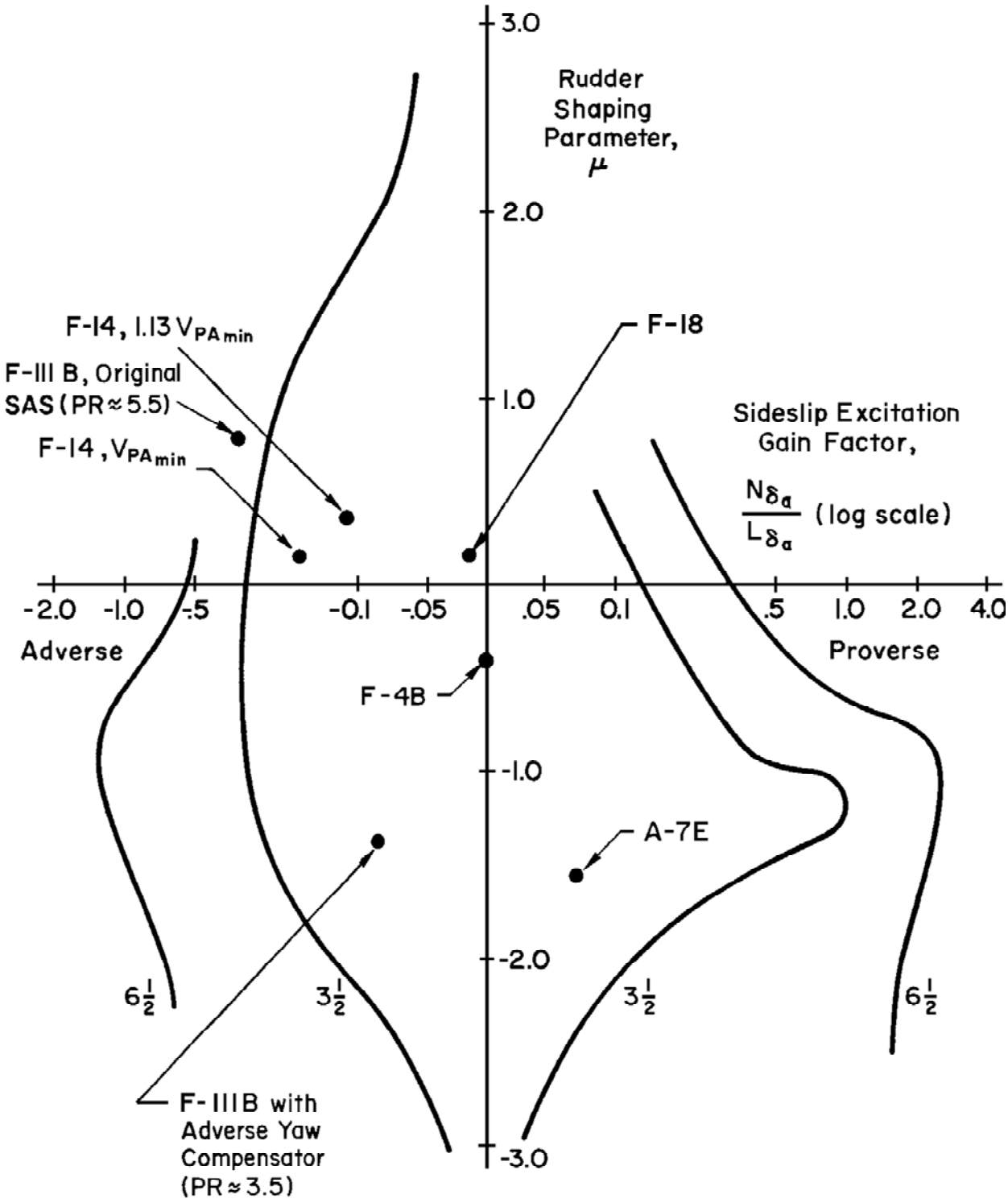


FIGURE 149. Required aileron-rudder sequencing for several operational aircraft, SAS/CAS-on (from Systems Technology, Inc., TR-1090-1)

5.2.3.2.8 Linearity of roll response to roll controller.
There shall be no objectionable nonlinearities in the variation of roll response with roll control deflection or force. Sensitivity or sluggishness in response to small control deflections shall be avoided.

REQUIREMENT RATIONALE

Nonlinear responses to control inputs can result in poor handling qualities. This requirement is intended to disallow forms and magnitudes of nonlinear response that interfere with precision control of bank angle.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.4.4
MIL-STD-1797: 4.5.3.

REQUIREMENT LESSONS LEARNED

The requirement is directed at precision of control. Objectionable nonlinearities can be those due to excessive friction, detents, nonlinear force gradients (especially abrupt gradient changes short of maximum deflection), nonlinear $C_{\ell_{\delta_{as}}}$ or $C_{n_{\delta_{as}}}$, spoiler lag, etc.

It is objectionable nonlinearities, not nonlinearities per se, at which this requirement is directed. Experience with roll command augmentation systems, in which roll rate response is made directly proportional to stick force, shows that some degree of nonlinearity is quite often desirable, if not necessary. This is discussed in 5.2.3.8.2. Figure 173 should be consulted for additional guidance.

VERIFICATION

It has not been possible to specify values for tolerable levels of nonlinearities, so reliance must be placed on qualitative pilot evaluations. Testing using ¼, ½, ¾, and full aileron are commonly used to demonstrate compliance. Such tests can also be used to help determine k for use in 5.2.3.2.5 and 5.2.3.2.6. Spoilers located in a region of separated flow will have a type-type nonlinearity.

Any problems with control nonlinearities should be evident in a series of large- and small-amplitude rapid target acquisition and precise tracking maneuvers. The target may be another maneuvering aircraft, a runway threshold, the horizon, lateral acceleration, etc., as appropriate.

5.2.3.3 Roll PIO.
There shall be no tendency for sustained or uncontrollable roll oscillations resulting from efforts of the pilot to control the aircraft. The phase angle of the bank angle frequency response to roll stick force at the criterion frequency, ω_c, shall be greater than or equal to __(1)__. Furthermore, the requirements of 5.2.3.1.1, 5.2.3.1.4, 5.2.3.1.5, 5.2.3.2.1 through 5.2.3.2.3, 5.2.3.5 through 5.2.3.5.3, 5.2.3.8.2, 5.2.3.8.3, 5.2.8.2.1, 5.2.8.3.1, 5.2.8.3.2, and 5.2.8.3.5 shall be met.

REQUIREMENT RATIONALE

The roll-axis is stated in addition to the general requirement of 5.2.1.6 to emphasize its importance for the roll axis and to allow incorporation of a more quantitative requirement.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.3
MIL-STD-1797: 4.5.2.

Recommended value for Blank (1): -180º

REQUIREMENT LESSONS LEARNED

The statement that requirements 5.2.3.1.1, 5.2.3.1.4, 5.2.3.1.5, 5.2.3.2.1 through 5.2.3.2.3, 5.2.3.5 through 5.2.3.5.3, 5.2.3.8.2, 5.2.3.8.3, 5.2.8.2.1, 5.2.8.3.1, 5.2.8.3.2, and 5.2.8.3.5 shall also be met would seem to be redundant, since these are already requirements. However, recent history would seem to indicate that, because the term "PIO" does not appear in the titles of these requirements, the importance of these requirements in precluding PIO is not appreciated. Many recent PIO incidents can be traced directly to problems addressed by these requirements. Therefore, these requirements are repeated here in the PIO requirement to insure that their significance in precluding PIO tendencies is understood.

The extension of the Smith-Geddes criteria to the lateral-directional axis was developed in AFWAL-TR-81-3090. In AFWAL-TR-81-3090, the lateral-directional criteria was used to analyze the YF-16, the X-15, and the M2-F2 and M2-F3 lifting bodies and also compared with results from handling qualities research projects with variable-stability aircraft: an approach and landing evaluation with the NT-33 (AFWAL-TR-81-3116), an investigation of reentry vehicle lateral-directional dynamics on the NT-33 (WADD-TR-61-147), and one configuration from lateral-directional studies on the Princeton Navion (Princeton University Report No. 727). Most of the data support the PIO criteria, and, in those cases where PIO was predicted but not encountered, handling qualities were usually poor.

See 5.2.2.1.5 for discussion of applicable considerations and data, in that case directed at longitudinal PIOs in general. The M2-F2 lifting body (NASA TN D-6496) encountered several divergent PIOs during flight testing. The primary cause was found to be the coupled roll subsidence/spiral mode (see Lessons Learned for 5.2.3.1.4)

Another cause of observed lateral PIO tendencies is the ω_ϕ/ω_d effect noted and explained on figure 125 and also in Norair Rpt No. NOR-64-143. Another prevalent cause is associated with control surface rate saturations. In this case the pilot tries to apply lateral control at a rate greater than the maximum surface rate, thereby getting out of phase if tight tracking is attempted. The quantitative aspects of such rate limiting are given in the appendix of Norair Rpt No. NOR-64-143 and involve gain and phase decrements that are functions of the ratio of commanded rate to saturation rate.

Pilot-in-the-loop oscillations on recent aircraft have been related to roll responses which are both too low (F/A-18) and too high (YF-16). These cases are discussed under 5.2.3.5.1 and 5.2.3.8.2, respectively. In a number of cases optimization of p/F_{as} in a fixed-base simulator has resulted in gross oversensitivity in actual flight. Control sensitivity, control surface rate limiting, control surface saturation, and equivalent time delay are critical factors in roll PIO.

VERIFICATION

This requirement shall apply to all flight conditions and tasks, and to all Levels, since zero or negative control loop damping is to be avoided under all flight conditions and Failure States. The existence of a PIO tendency is difficult to assess. A high-stress task such as approach and landing with a lateral-directional offset, air-to-air tracking, or terrain following, may reveal PIO proneness. Demanding tracking tasks, aggressive control, sensitive response, proverse yaw, low Dutch roll damping, and equivalent time delay are factors varying with flight condition which may tend to incite roll PIOs. Lateral acceleration induced on the pilot in rolling may contribute.

The value of ω_c is determined as follows:

$$\omega_c = (0.24 \text{ rad-oct/dB-sec})\ \overline{S} + 6.0 \text{ rad/sec}$$

where $\overline{S}$ is the average slope of $\left|\frac{\phi(s)}{F_{as}(s)}\right|$ in dB/oct over the interval from 1 to 6 rad/sec.

This requirement is the Smith-Geddes PIO criteria extended to the lateral-directional axis. The origins of the Smith-Geddes criteria are traced in 5.2.2.1.5, and discussions of the calculation of $\overline{S}$ and ω_c can be found there. Application in the lateral-directional axis is similar to that in the longitudinal axis, except that it is applied to ϕ/F_{as} instead of to θ/F_{es}.

Pilot-in-the-loop oscillations were a consideration in setting the boundaries of 5.2.3.1.1, 5.2.3.1.4, 5.2.3.1.5, 5.2.3.2.1 through 5.2.3.2.3, 5.2.3.5 through 5.2.3.5.3, 5.2.3.8.2, and 5.2.3.8.3. Pilot-vehicle analysis in the manner described in the discussion of these paragraphs should help in the design stage.

5.2.3.4 Yaw PIO.
There shall be no tendency for sustained or uncontrollable yaw oscillations resulting from the efforts of the pilot to control the aircraft in the air or on the ground. Furthermore, the requirements of 5.2.3.1.6, 5.2.3.2.5, 5.2.3.2.6, 5.2.3.2.7, 5.2.8.2.1, 5.2.8.3.1, 5.2.8.3.2, and 5.2.8.3.5 shall be met.

REQUIREMENT RATIONALE

This requirement, in addition to the general requirement of 5.2.1.6, is inserted to provide more specific criteria for any task that might involve high-bandwidth control in yaw or lateral acceleration. An example might be yaw pointing for fine tracking.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.3
MIL-STD-1797: 4.6.3.

REQUIREMENT LESSONS LEARNED

The pitch-axis PIO requirement, 5.2.2.1.5, discusses some causes of PIOs. Factors known to contribute to lateral-directional PIOs are large effective or equivalent time delays, excessive friction or hysteresis in the flight control system, and the "ω_ϕ/ω_d" effect described at length in AFFDL-TR-69-72 (see 5.2.3.2.1 discussion). Depending upon the cause, ground-based simulation may or may not prove a useful investigation technique—often it does not.

The statement that requirements 5.2.3.1.6, 5.2.3.2.5, 5.2.3.2.6, 5.2.3.2.7, 5.2.8.2.1, 5.2.8.3.1, 5.2.8.3.2, and 5.2.8.3.5 shall also be met would seem to be redundant, since these are already requirements. However, as with the corresponding requirements in the pitch and roll axes, because the term "PIO" does not appear in the title of these requirements, the importance of these requirement in precluding PIO may not be appreciated. Therefore, these requirements are repeated here in the PIO requirement to insure their significance in the prevention of PIO tendencies is understood.

VERIFICATION

Due to the lack of a reliable quantitative measure, the requirement is written in terms of subjective evaluations. It is, of course, hoped that meeting the other quantitative requirements of this standard will prevent a directional PIO. This requirement is identical to the qualitative roll-axis requirement of 5.2.3.3.

The existence of a yaw PIO tendency is difficult to assess. Therefore, no specific flight conditions or tasks are recommended, though a high-stress task such as approach and landing with a lateral offset, terrain following, air-to-ground tracking, or in-flight refueling (receiver) may reveal PIO proneness.

This requirement shall apply to all flight conditions and tasks, and to all Levels, since zero or negative closed-loop damping is to be avoided under any flight condition or Failure State. High-bandwidth yaw-control tasks are uncommon. The dynamic yaw response requirements (5.2.3.2.5, 5.2.3.2.6, and 5.2.3.2.7) are designed to account for the need of rudder pedal in rolling, but may not cover all contingencies. A precision closed-loop task, performed aggressively, is needed. Some direct side force modes may involve high-bandwidth yaw control; see AFWAL-TR-81-3027 and Sammonds, et al., for example.

Pilot-In-the-loop Oscillations were a consideration in setting the boundaries of 5.2.3.1.6, 5.2.3.2.5, 5.2.3.2.6, and 5.2.3.2.7. Pilot-vehicle analysis in the manner described in the discussion of these paragraphs should help in the design stage.

5.2.3.5 Roll control effectiveness.
For full roll commands to the right and to the left, initiated abruptly both from steady, coordinated (zero lateral acceleration) bank angles and from wings-level flight, the time to bank shall be no more than the limits in table LVIII, with time measured from the initiation of control force application. Requirements 5.2.3.5 through 5.2.3.5.2 apply throughout the applicable speed-altitude-load factor envelopes, except that the structural limits on combined rolling and normal acceleration need not be exceeded. Pitch control shall be held fixed throughout the maneuver and __(1)__. During the roll the air vehicle shall exhibit no tendency to depart from controlled flight at any angle of attack. For Flight Phase TO, the time required to bank may be increased proportional to the ratio of the rolling moment of inertia at takeoff to the largest rolling moment of inertia at landing for weights up to the maximum authorized landing weight.

TABLE LVIII. Roll performance.

Level	Speed Range	Time to Achieve the Stated Bank Angle Change (seconds)		
		Category A	Category B	Category C
1				
2				
3				

REQUIREMENT RATIONALE

Roll power is specified in terms of bank angle change in a given time, a form related to operational use of the air vehicle, to allow necessary maneuvering and attitude regulation.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:

MIL-F-8785C: 3.3.4 and 3.3.4.2
MIL-STD-1797: 4.5.8.1.

Recommended values for table LVIII for Class I air vehicles:

Level	Speed Range	Time to Achieve the Stated Bank Angle Change (seconds)		
		Category A	Category B	Category C
		$\phi_t = 60°$	$\phi_t = 60°$	$\phi_t = 30°$
1	$V_{o_{min}} \leq V < V_{o_{max}}$	1.3	1.7	1.3
2	$V_{min} \leq V \leq V_{max}$	1.7	2.5	1.8
3	$V_{min} \leq V \leq V_{max}$	2.6	3.4	2.6

Recommended values for table LVIII for Class II-L air vehicles:

Level	Speed Range	Time to Achieve the Stated Bank Angle Change (seconds)		
		Category A	Category B	Category C
		$\phi_t = 45°$	$\phi_t = 45°$	$\phi_t = 30°$
1	$V_{o_{min}} \leq V < V_{o_{max}}$	1.4	1.9	1.8
2	$V_{min} \leq V \leq V_{max}$	1.9	2.8	2.5
3	$V_{min} \leq V \leq V_{max}$	2.8	3.8	3.6

Recommended values for table LVIII for Class II-C air vehicles:

Level	Speed Range	Time to Achieve the Stated Bank Angle Change (seconds)		
		Category A	Category B	Category C
		$\phi_t = 45°$	$\phi_t = 45°$	$\phi_t = 25°$
1	$V_{o_{min}} \leq V < V_{o_{max}}$	1.4	1.9	1.0
2	$V_{min} \leq V \leq V_{max}$	1.9	2.8	1.5
3	$V_{min} \leq V \leq V_{max}$	2.8	3.8	2.0

Recommended values for table LVIII for Class III air vehicles:

Level	Speed Range	Time to Achieve Stated Bank Angle Change (seconds)		
		Category A	Category B	Category C
		$\phi_t = 30°$	$\phi_t = 30°$	$\phi_t = 30°$
1	$V_{o_{min}} \leq V < 1.8V_{o_{min}}$	1.8	2.3	2.5
	$1.8V_{o_{min}} \leq V < 0.7V_{o_{max}}$	1.5	2.0	2.5
	$0.7V_{o_{max}} \leq V \leq V_{o_{max}}$	2.0	2.3	2.5
2	$V_{min} \leq V < 1.8V_{min}$	2.4	3.9	4.0
	$1.8V_{min} \leq V < 0.7V_{max}$	2.0	3.3	4.0
	$0.7V_{max} \leq V \leq V_{max}$	2.5	3.9	4.0
3	$V_{min} \leq V \leq V_{max}$	3.0	5.0	6.0

Recommended values for table LVIII for Class IV air vehicles:

Level	Speed Range*	Time to Achieve the Stated Bank Angle Change (seconds)				
		Category A			Category B	Category C
		$\phi_t = 30°$	$\phi_t = 50°$	$\phi_t = 90°$	$\phi_t = 90°$	$\phi_t = 30°$
1	$V_{o_{min}} \le V < V_{o_{min}} + 20$ kts	1.1			2.0	1.1
	$V_{o_{min}} + 20$ kts $\le V < 1.4V_{o_{min}}$	1.1			1.7	1.1
	$1.4V_{o_{min}} \le V < 0.7V_{o_{max}}$			1.3	1.7	1.1
	$0.7V_{o_{max}} \le V \le V_{o_{max}}$		1.1		1.7	1.1
2	$V_{min} \le V < V_{min} + 20$ kts	1.6			2.8	1.3
	$V_{min} + 20$ kts $\le V < 1.4V_{min}$	1.5			2.5	1.3
	$1.4V_{min} \le V < 0.7V_{max}$			1.7	2.5	1.3
	$0.7V_{max} \le V \le V_{max}$		1.3		2.5	1.3
3	$V_{min} \le V < V_{min} + 20$ kts	2.6			3.7	2.0
	$V_{min} + 20$ kts $\le V < 1.4V_{min}$	2.0			3.4	2.0
	$1.4V_{min} \le V < 0.7V_{max}$			2.6	3.4	2.0
	$0.7V_{max} \le V \le V_{max}$		2.6		3.4	2.0

*NOTE: In some particular cases, some of these speed ranges may not exist. The requirements apply, of course, only within speed ranges that do exist.

Recommended values for blank (1):

For Class IV-L air vehicles: "yaw control pedals shall remain free for Level 1; but otherwise, yaw control pedals may be used to reduce sideslip that retards roll rate (not to produce sideslip that augments roll rate) if such control inputs are simple, easily coordinated with roll control inputs and consistent with piloting techniques for the air vehicle mission."

For Class IV-C air vehicles: "yaw control pedals shall remain free for Level 1 in all Flight Phase Categories and for Level 2 in Flight Phase Category C; but otherwise, yaw control pedals may be used to reduce sideslip that retards roll rate (not to produce sideslip that augments roll rate) if such control inputs are simple, easily coordinated with roll control inputs and consistent with piloting techniques for the air vehicle mission."

For all other carrier-based air vehicles: "yaw control pedals shall remain free for Level 1 and Level 2 in Category C Flight Phases; but otherwise, yaw control pedals may be used

to reduce sideslip that retards roll rate (not to produce sideslip that augments roll rate) if such control inputs are simple, easily coordinated with roll control inputs and consistent with piloting techniques for the air vehicle mission."

For all other Classes of air vehicles: "yaw control pedals may be used to reduce sideslip that retards roll rate (not to produce sideslip that augments roll rate) if such control inputs are simple, easily coordinated with roll control inputs and consistent with piloting techniques for the air vehicle mission."

REQUIREMENT LESSONS LEARNED

The recommended tables, definitions, and wording of this requirement are collations of the various roll control effectiveness sections of MIL-F-8785C. Considerable flight test data and pilot commentary are available on a larger number of modern aircraft. These will be shown to be extremely valuable in supporting the roll power requirements. However, for most of the data to be presented it is difficult to assure that all other lateral-directional characteristics (e.g., ζ_d, ω_d, T_R, T_S, $\Delta\beta_{max}/k$) were Level 1, so qualitative evaluations may be somewhat biased by these other effects. Also, the operational speed ranges for Class III and IV aircraft are not always known, so it is necessary to compare the data only generally with the specific performance requirements. In some instances, data are given as specified in MIL-F-8785B—times for a certain bank angle change; in others it may be given as required by an earlier version of the specification, MIL-F-8785: bank angle change in 1 sec.

Complicating factors in roll performance include adverse or favorable yaw, yaw due to rolling (5.2.3.2), and cross-axis coupling (5.2.4.3), control rate limiting may significantly decrease the roll performance. Except for configurations with low $|\phi/\beta|$ and low N_{δ_a}, the roll response may be significantly contaminated by the Dutch roll mode. Although crossfeed to the rudder can help, the required crossfeed varies with angle of attack.

See the discussions immediately below for recent experience with the roll requirements. Because these requirements are both difficult and costly to meet, they have generated much argument. Aircraft with large lateral stability (C_{ℓ_β}) at high angle of attack may need more roll control than specified separately here and in the sideslip requirements of 5.2.3.9.4 and 5.2.3.10 because of sensitivity to lateral gusts. In addition, asymmetric loadings (of fuel or stores, whether intentional or the result of failure or malperformance) shall be taken into account.

Discussion of Class III requirements

Class III roll requirements have been defined in terms of three speed ranges. The basic requirements for Levels 2 and 3 were relaxed somewhat in MIL-F-8785C from MIL-F-8785B:

Category B, Level 2: 30º in 3.3 sec, instead of 3.0

Category B, Level 3: 30º in 5.0 sec, instead of 4.0

Category C, Level 2: 30º in 4.0 sec, instead of 3.2

Category C, Level 3: 30º in 6.0 sec, instead of 4.0.

AFFDL-TR-72-41 concluded from a "review of roll control used in various experiments…[that] the roll control authority requirements…for Category C Flight Phases are excessive for airplanes that do not have high sensitivity to crosswind and turbulence. Data clearly indicate that there is an interaction between the roll control authority and the amount of roll damping and roll sensitivity to side velocity." The data (primarily for Class II and III airplanes) were:

Level 1 boundary: 16º in 1.8 sec (more control power than that was needed to counteract crosswinds and turbulence)
20º in 1.8 sec was adequate for Class II
16º in 1.8 sec was not exceeded for Class III

Level 2 boundary: 10º in 1.8 sec (more control power than that was needed to counteract crosswinds and turbulence)
6º in 2 sec.

Figure 150 shows, on the same basis as in AFFDL-TR-69-72, that by the more demanding of these data the present Level 1 boundary is substantiated but the Level 2 boundary may be too lenient.

Roll performance of the C-5A is shown on figure 151. The airplane does not meet the specification; however, "the roll acceleration available was considered satisfactory by the Joint Test Team on the basis of the offset landing maneuver, which was considered a practical test of lateral-directional maneuverability." In cruise the airplane was considered acceptable. ASD-TR-78-13 retained the MIL-F-8785B requirements for application to a production AMST, where the critical design case was to balance the rolling moment at stall with one engine failed. Thus, although there is some justification for relaxing the Class III roll requirements, that must be done considering the aircraft mission and potential operation. In AFFDL-TR-66-148 both Drake and Ashkenas find a need for a steady roll rate capability of 15º/sec or greater for the smooth offset-correction maneuvers typical of transport operation.

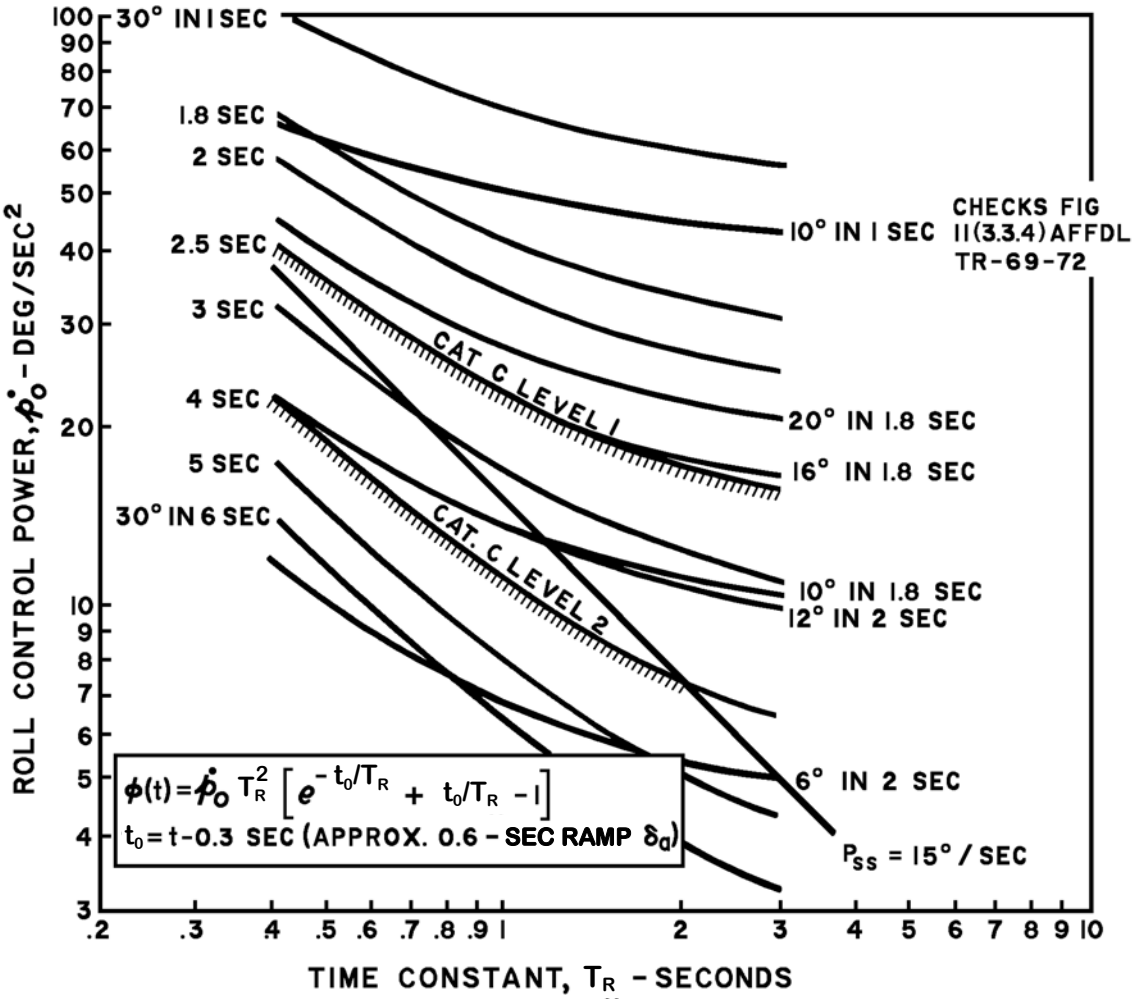
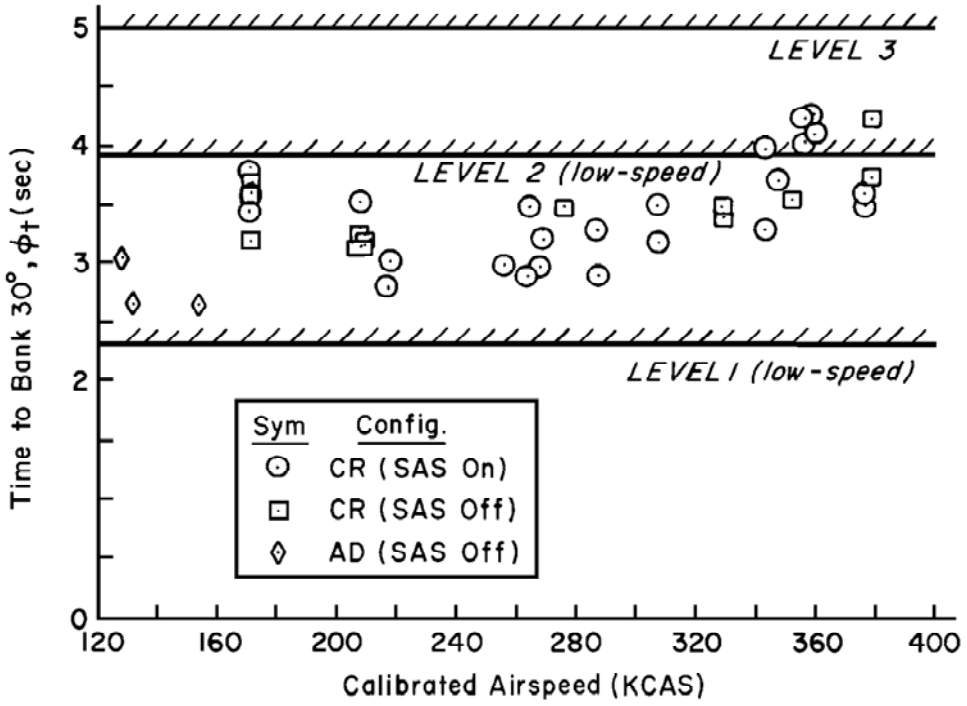
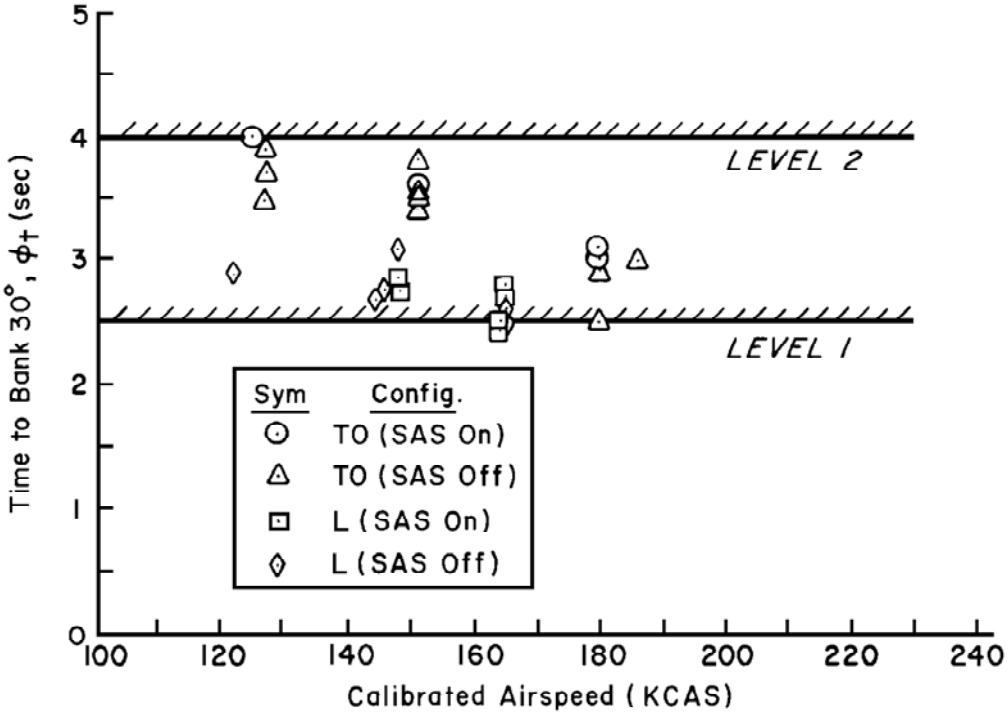


FIGURE 150. Roll control effectiveness parameters for Class III aircraft, Category C.



a) Category B Flight Phases



b) Category C Flight Phases

FIGURE 151. C-5 flight test data (from AFFDL-TR-75-3).

AFFDL-TR-72-141 contains limited substantiation of the Class III, Category A and C requirements. For the P-3B airplane,

> "…at speeds above about 240 KEAS, it was possible to achieve an attitude change of 30 degrees in about 1.5 seconds. Discussions with qualified P-3B pilots revealed that at speeds near and above 240 KEAS the aircraft had excellent roll maneuvering capability (aileron displacement is reduced, but forces are still a bother). These comments are concerned adequate to act as substantiation of the Level 1 requirement for Category A Flight Phases…In Flight Phase PA at 125 KEAS full wheel deflection (112 degrees) produces a 30 degree bank angle change in 2.6 seconds. This roll response was evaluated as acceptable but unpleasant. Part of the unpleasant aspect of this response is coupled with high lateral control forces and large lateral control displacements, assigned a Cooper Rating 3. The 30 degree bank angle change in 2.6 seconds is considered to be the minimum acceptable roll performance for Level 1 flying qualities and thus substantiates the Level 1 boundary of the specification (30 degree change in 2.5 seconds)."

However, for a larger Class III airplane, the C-5A (AFFDL-TR-75-3), comparison with the performance requirements is poor, as figure 151 shows. Similar data for the C-141A, YC-141B, and the L-1011 are given on figure 152, from AFFDL-TR-78-171. "Although neither the C-5A nor the C-141A/YC-141B comply with the rolling performance requirements, qualitative pilot comments indicate that both airplanes have acceptable rolling performance in the cruise configuration." In the landing configuration, for the C-5A (AFFDL-TR-75-3),

> "…the roll acceleration available was considered satisfactory by the Joint Test Team on the basis of the offset landing maneuver, which was considered a practical test of lateral directional maneuverability. The offset landing maneuver consists of approaching the runway with a 200-ft lateral misalignment on a 3 degree glideslope. At an altitude of 200 feet, the airplane is aligned with the runway centerline prior to touchdown."

But for the C-141, according to Woodcock and Mably (AFFDL-TR-66-148):

> "Currently the aircraft is restricted to 120 kts [crosswind] and may remain so because of low lateral control power…the [Military Airlift] command pilots interviewed generally consider it adequate for present operating conditions. AFFTC believes that as the minimum ceiling is reduced from 500 feet, C-141 operation will become limited by lateral control."

Acceptability of low roll performance may be a function of the amount of lateral acceleration felt by the pilot; i.e., as acceleration due to rapid roll control inputs increases, pilot acceptance of the roll response decreases. For the C-5A, AFFDL-TR-75-3:

> "In order to meet the Level 1 requirements, the lateral control system would have to be improved to obtain a higher bank angle change in the first second of roll.

> On an aircraft with a very large rolling moment of inertia this would be difficult to accomplish. An increased initial roll response on the C-5A would further aggravate the very noticeable side kick, or lateral acceleration component, in the cockpit and troop compartment that is experienced during full abrupt control input. The side kick occurs since the cockpit and troop compartment are located considerably above the principal roll axis of the airplane."

A requirement to operate into forward-area airstrips, or to land on short segments of bomb-damaged runways, demands more maneuvering than does airline-type operation. This should be borne in mind when evaluating comments based on operational experience.

The flight program of NASA TN D-5957 investigated roll requirements in Cruise (Category B) for transport aircraft. A NASA Lockheed JetStar was equipped with a model-following simulation to produce pure rolling response to ailerons; i.e.,

$$\frac{p}{\delta_a} = \frac{L_{\delta_a} T_R}{T_R\, s + 1} \ .$$

The evaluation consisted of various rolling and turning maneuvers, including rapid rolls to 30º bank angle. Cooper-Harper ratings for three pilots are compared on figure 153 with times to bank 30º. Only those cases for which T_R is Level 1 are shown. These data support the Level 1 and 2 roll requirements extremely well, and suggest that the Level 3 requirement could be relaxed from 5 sec to at least 8 sec or greater. They also show that a lower limit exists at somewhat less than 1 sec due to high roll sensitivity.

It must be pointed out that since the JetStar is considerable smaller, and of different design than the C-5A, the test pilots of NASA TN D-5957 would not have been subjected to the large lateral accelerations discussed above. It seems clear, however, that there is some need for addressing the incompatibility between the required Level 1 roll performance and the lateral acceleration on the occupants when objectionable accelerations result. Pilot ratings for a CV-990 (NASA TN D-6811) in Category B and C flight support the Level 1 boundaries recommended for Class III, as shown on figure 154.

Discussion of Class IV requirements

For Class IV aircraft the ROSH and ROTH have been divided into speed ranges with different requirements for the different speeds. This reflects lessons learned previous roll requirements to be met at the speed extremes of the envelopes. In general, this requirement retains the MIL-F-8785B roll requirements in the "Medium" speed range, with some relaxation in the other speed ranges.

The initial proposals for the speed ranges were defined using ASD experience with the F-15 and F-16. At the 1978 Flying Qualities Symposium the authors of MIL-F-8785C presented a modified definition of the speed range (AFFDL-TR-78-171). The suggested modification—to have the four speed ranges as a function of load factor—was incorrect. The intent is to require a certain roll performance at all load factors in an operationally useful speed range, as shown on figure 155. The problem then becomes one of defining the required speeds in a general way.

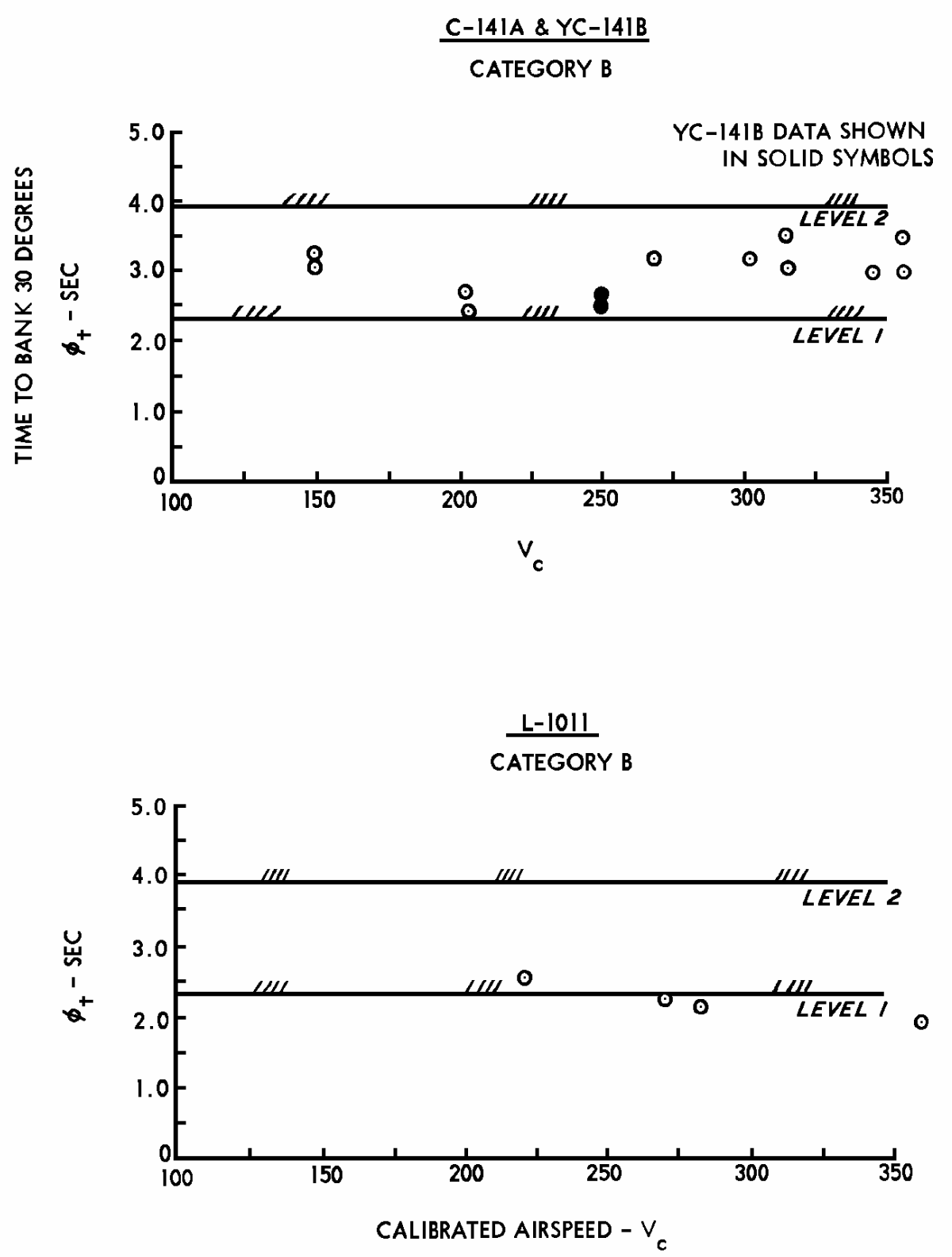


FIGURE 152. Roll performance for Class III aircraft (from AFFDL-TR-78-171).

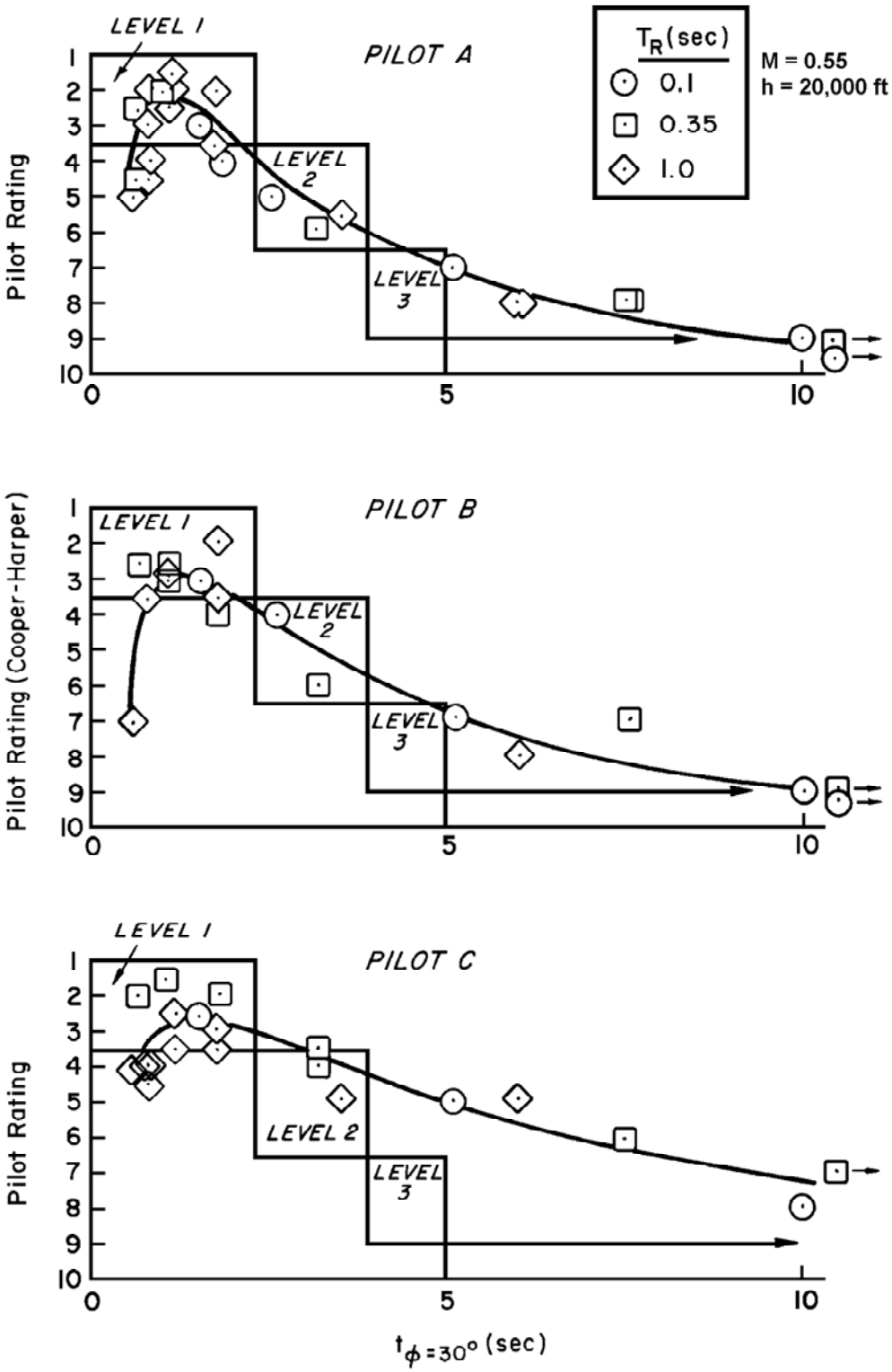


FIGURE 153. Comparison of pilot ratings for Class III aircraft in Category B Flight Phases with recommended roll performance requirements (NASA TN D-5957).

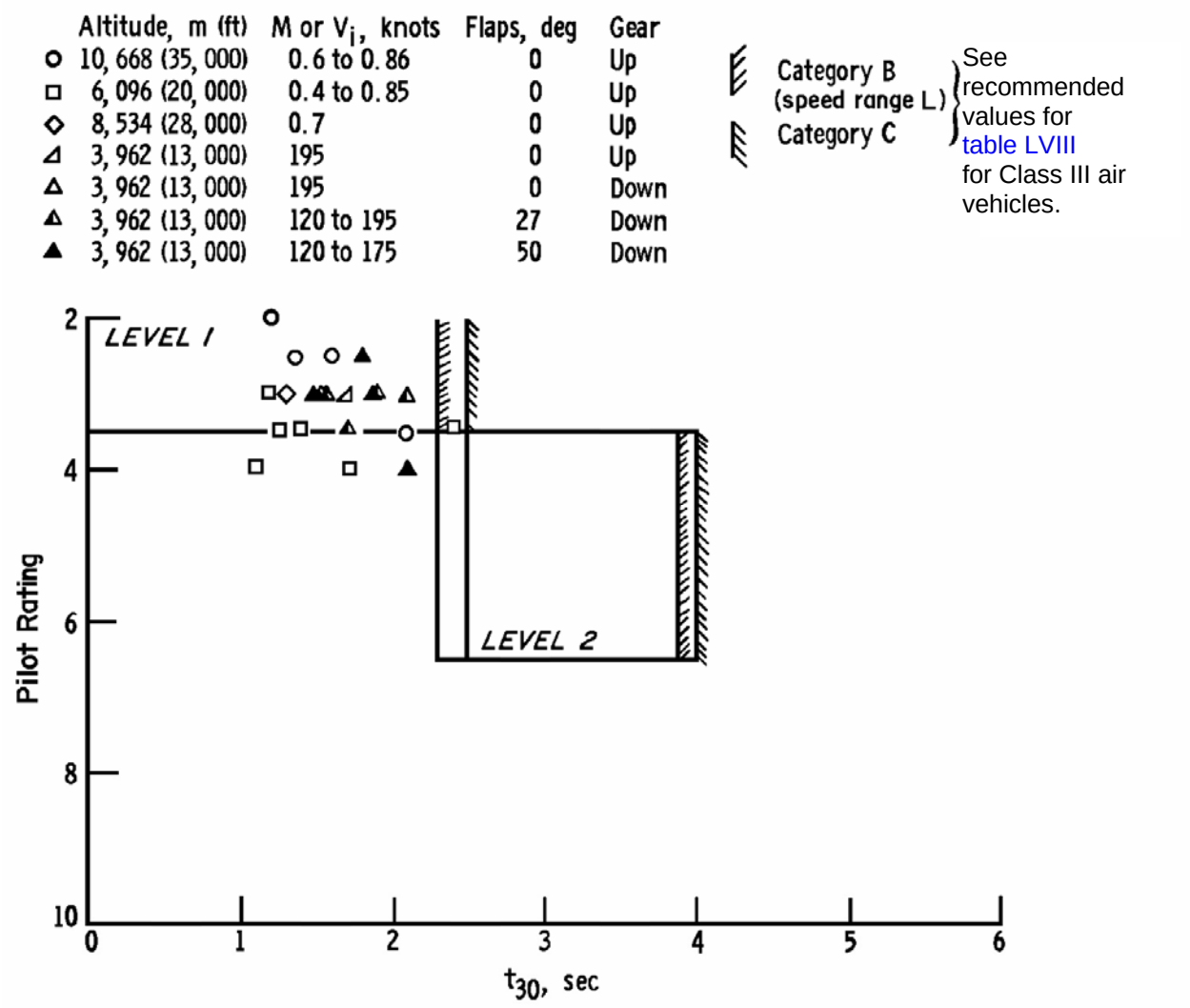


FIGURE 154. Time to bank 30° for CV-990 (NASA TN D-6811).

The definitions proposed in "Proposals for Revising MIL-F-8785B, 'Flying Qualities of Piloted Airplanes'" have been retained. The authors of this standard believe these represent the requirement for superior roll performance at combat flight conditions. The contractor and procuring activity should retain that philosophy in developing the system requirements. It may be that these definitions still do not cover all cases. It is emphasized that the proposed speed ranges should be tailored to the specific application. The intent is to provide sufficient roll maneuverability to do the task at the normal speed for that task, with a relaxation permitted for speeds at which the design penalty is to severe or less maneuverability is normally acceptable. In line with these speed ranges, the bank angle changes are compatible with the speed at which the roll performance will be demonstrated.

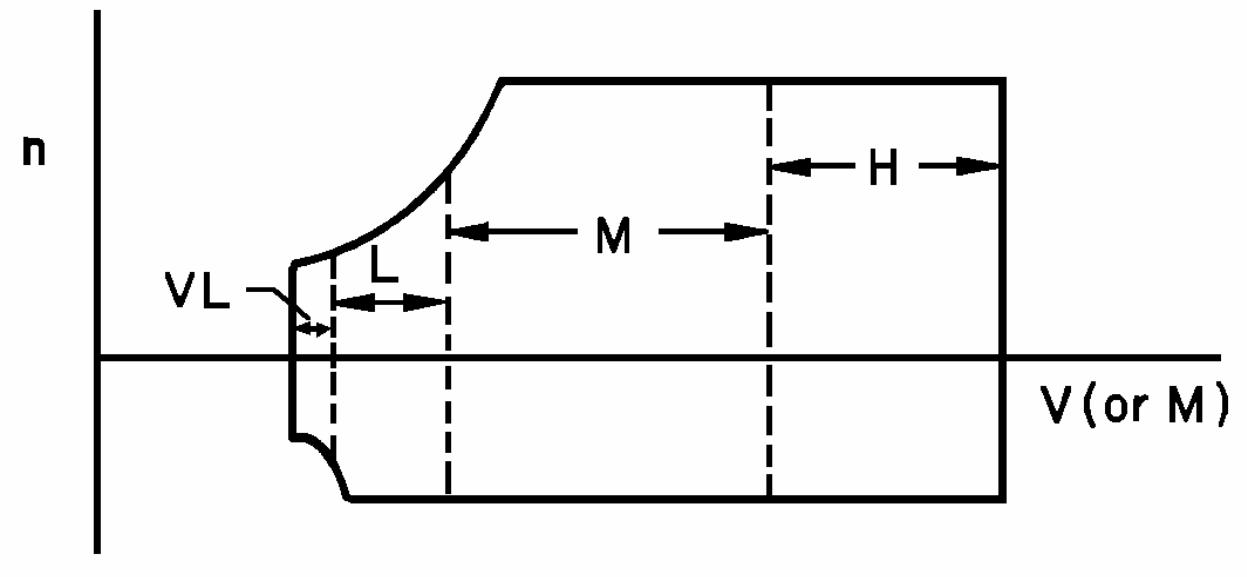


FIGURE 155. Division of flight envelope into speed ranges.

Relaxations in roll performance at low speeds are concessions to the difficulty of doing better without adding excessive structural weight, actuator size, etc. The authors of this standard are reluctant to relax the performance requirement, and harbor misgivings. The single, outstanding factor that influenced convergence and kill in a recent air combat simulation (AFWAL-TR-80-3060) was high roll performance at low airspeed. This was a fixed-base simulation, however, and the results must be balanced against feedback that pilots may not be able to use such roll rates at extreme flight conditions. Although past studies and analyses have indicated no need for roll performance that great, pilots have remained adamant on the requirement for 90º bank in the first second (e.g., AFFDL-TR-69-72).

Data for the F-4 (AFFDL-TR-70-155) are given on figure 156 and figure 157. Times to bank for these figures were not actual test values, but were calculated from known roll characteristics for various F-4 aircraft. Therefore, there is some inherent inaccuracy in the values of ϕ_t plotted. Figure 156 compares pilot ratings and ϕ_t with the recommended Category B requirements for Class IV aircraft (for all speed ranges except VL). While the data are sparse, it at least appears that the Level 1 requirement of 90º in 1.7 sec is not too stringent. AFFDL-TR-70-155 recommends that "For Class IV-L and IV-C aircraft, the Level 1 minimum time to bank 30° shall be relaxed to 1.3 seconds and the lower Level 3 boundary shall be relaxed to 30 degrees in 2.8 seconds." Of course, the impact of figure 156 data is mitigated by recognizing the possible inaccuracies involved in computing ϕ_t without actual test data.

Flight test data for the F-14A in power approach are shown on figure 158, from Navy Rpt No. SA-C7R-75. Roll performance in terms of bank angle change in 1 sec suggests the F-14A would meet the recommended Class IV, Category C, requirement of 30º in 1.1 sec at low angles of attack, and fail to meet it at high angles of attack. Navy Rpt No. SA-C7R-75 states that "Lateral control effectiveness in the PA configurations was adequate to perform the bank angle changes required during approach."

The F-15C (AFFTC-TR-80-23) meets the Category A Level 1 requirement of 90º in 1.3 sec ($1.4V_{min} \leq V < 0.7V_{max}$) as figure 159 shows. Unfortunately, the authors of this standard possess no pilot ratings or comments which relate specifically to F-15 roll performance.

F/A-18A roll performance in PA (landing and takeoff) configurations is seen to be marginally Level 1 (figure 160) below 180 kts; i.e., the responses are very close to the required 30º in 1.1 sec.

The roll performance of the YF-16, shown in table LIX and figure 161, is comparable to the capabilities of present-generation USAF fighter aircraft and is judged to be fairly good in comparison to the recommended roll control power requirements for Class IV aircraft (both 5.2.3.5 and 5.2.3.5.1), but less than some of the requirements. A roll damper which allows it to stop precisely, more quickly than it can start rolling, appears partly responsible for the satisfactory pilot ratings.

F-16 roll performance in 360º rolls for the "CR (cruise) configuration" (figure 162) compares quite well with the recommended requirements for Flight Phase CO (table LX). The F-16 was Level 2 below 180 kts, Level 3 below 155 kts in Power Approach (figure 163). According to AFFTC-TR-79-10, "the pilots were pleased with the F-16A/B CR configuration roll performance. PA configuration roll performance was acceptable."

TABLE LIX. YF-16 roll performance (from AFFTC-TR-75-15).

$\bar{q}$ (psf)	V (KCAS)	M	h_c (ft)	α* (deg)	p_{max} (deg/sec)	t_{90} (sec)	t_{360} (sec)	$\delta_{a_{avg}}$ (deg)	$\delta_{r_{max}}$ (deg)
70	135	0.45	36K	14	-155	1.60	3.55	17	12
270	300	0.79	30K	5	187	1.15	2.65	-10	-2
1140	650	1.58	30K	2	-168	1.25	2.90	10	-4

* Select angle of attack [mid-value among noseboom (pneumatic) angle of attack, left cone angle of attack, and right cone angle of attack; automatically selected by flight control system].

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall proceed or accompany careful build-up to suspected critical flight conditions.

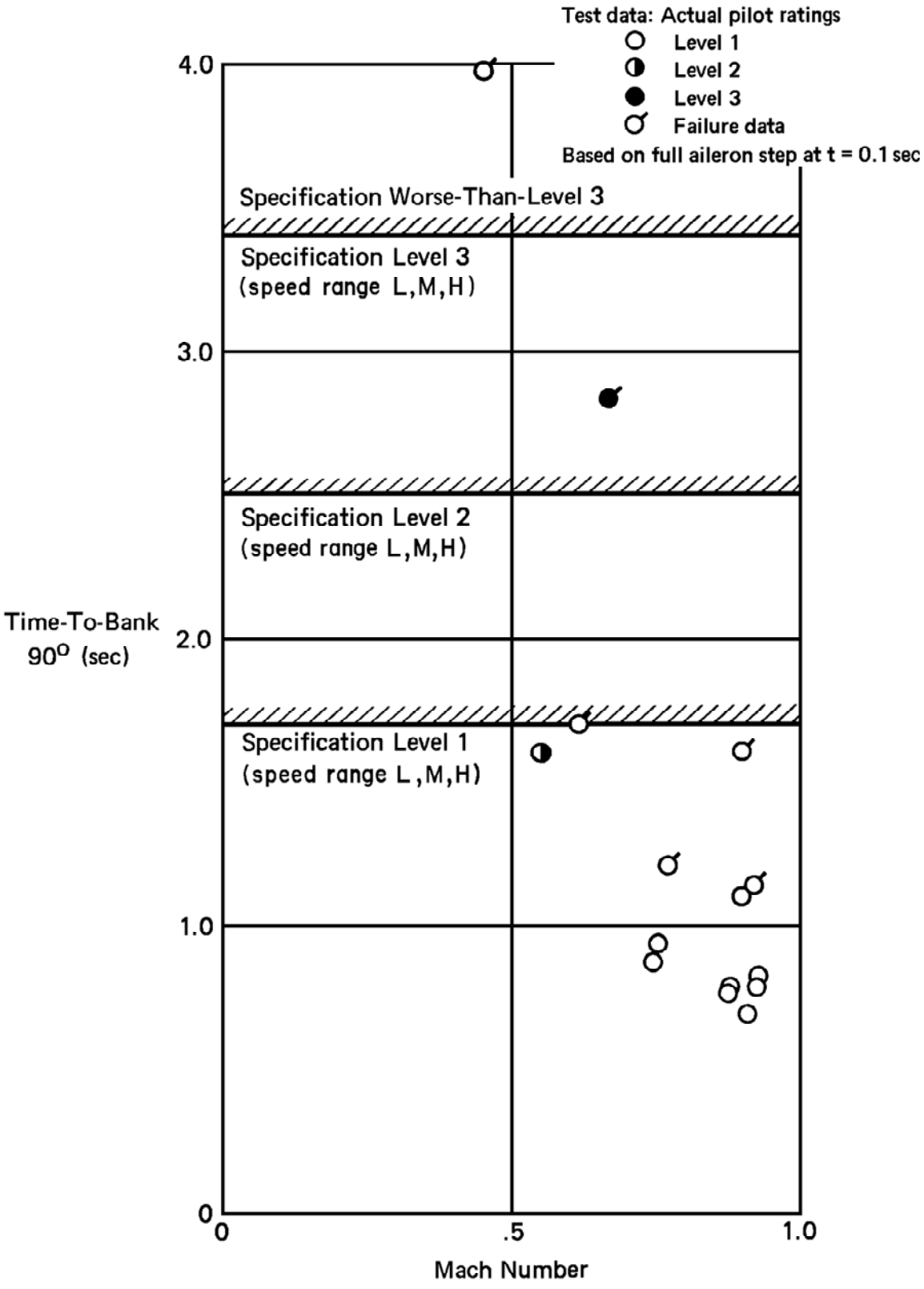


FIGURE 156. **F-4 roll control effectiveness, time-to-bank 90°, CR configuration (from AFFDL-TR-70-155).**

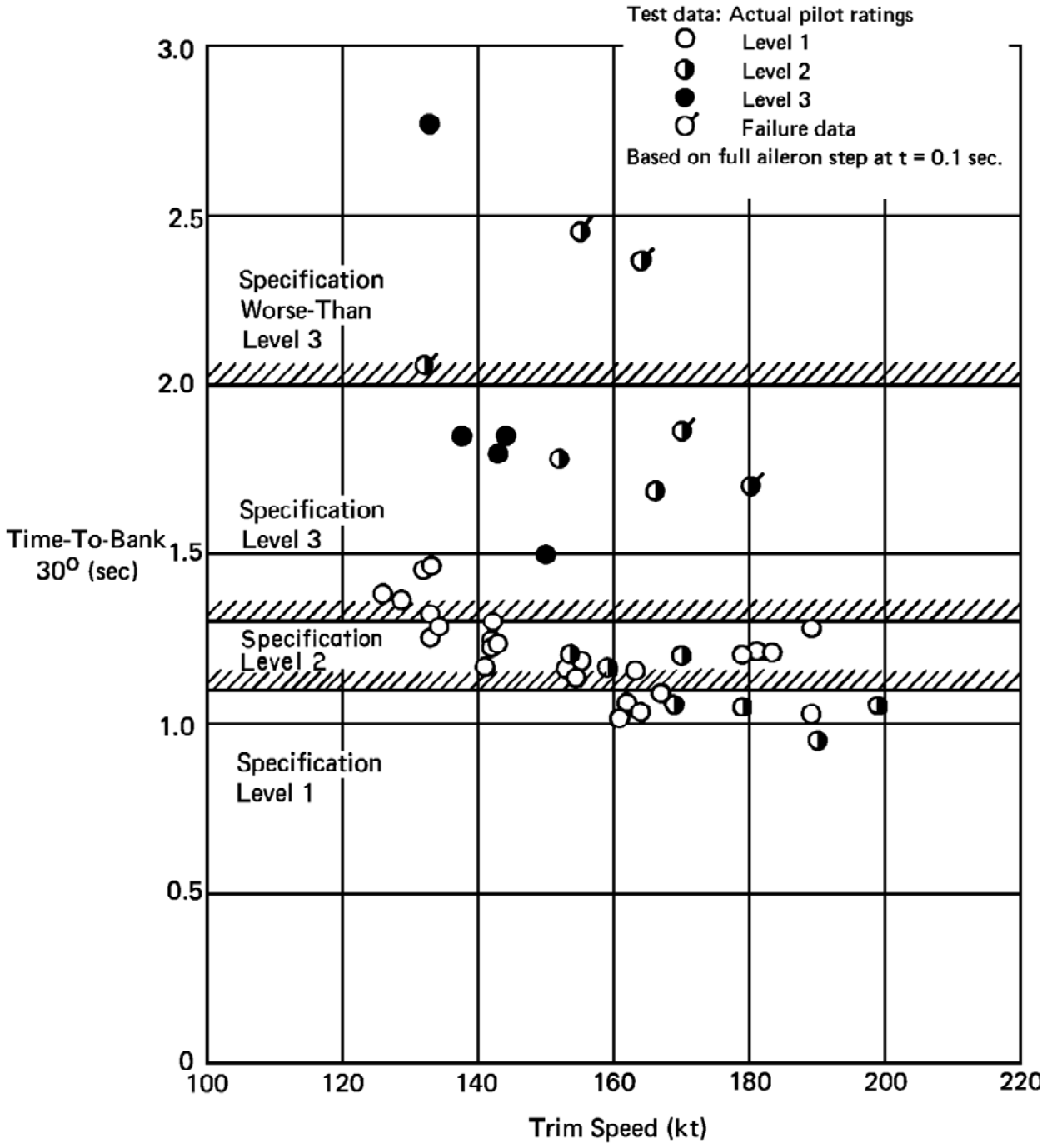
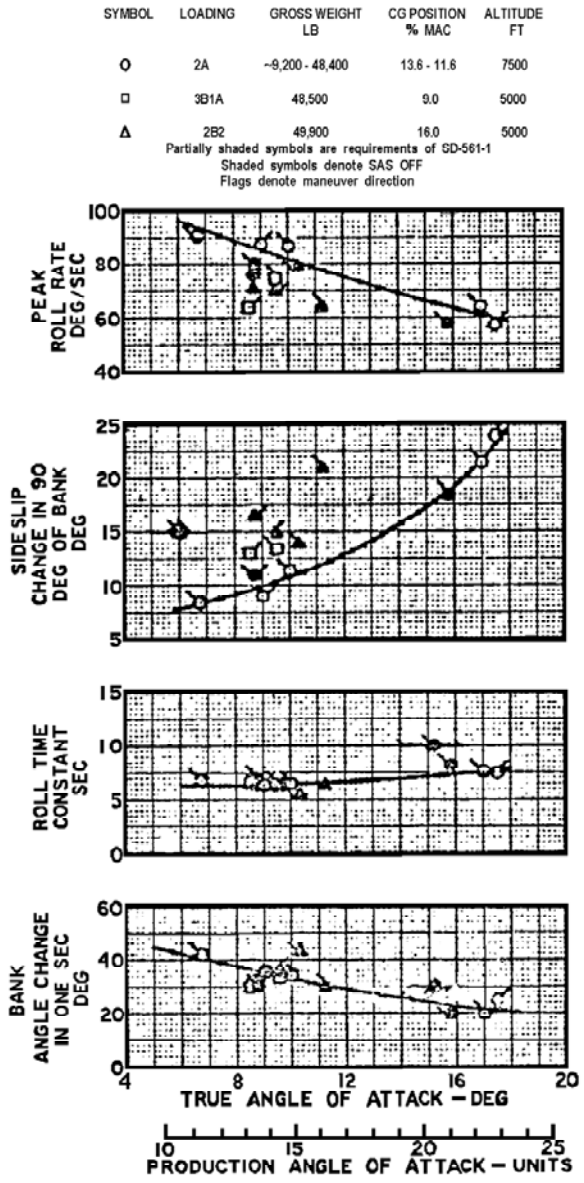


FIGURE 157. **F-4 roll control effectiveness, time-to-bank 30°, CR configuration (from AFFDL-TR-70-155).**



Loading 2A
Four AIM-7 Missiles
Loading 3B1A
Two AIM-54 Missiles
Two AIM-7 Missiles
Two AIM-9 Missiles
Loading 2B2
Four AIM-7 Missiles
Two AIM-9 Missiles

FIGURE 158. F-14A rolling performance in configuration PA; DLC on (from Navy Rpt No. SA-C7R-75).

CAS-ON
PRCA-AUTO
0.8 to 1.0 Mach Number
Dynamic Pressure ≤ 400 psf

Notes:

- *Fairings were taken from AFFTC-TR-76-48 and represent TF-15A flight test results*
- *Shaded symbols denote power approach configuration obtained from bank-to-bank rolls of 90 deg to -90 deg*

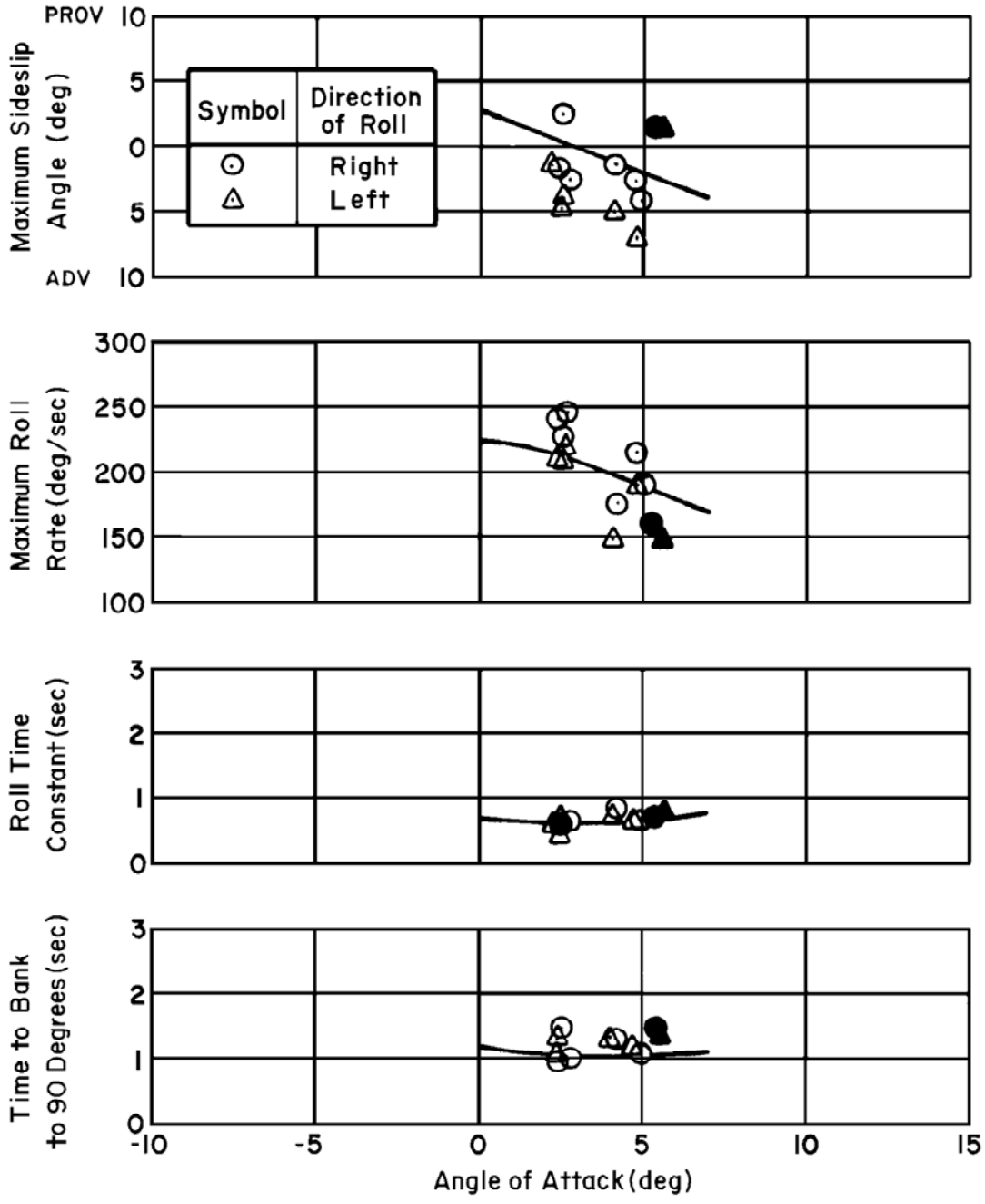


FIGURE 159. F-15C aileron roll characteristics (from AFFTC-TR-80-23).

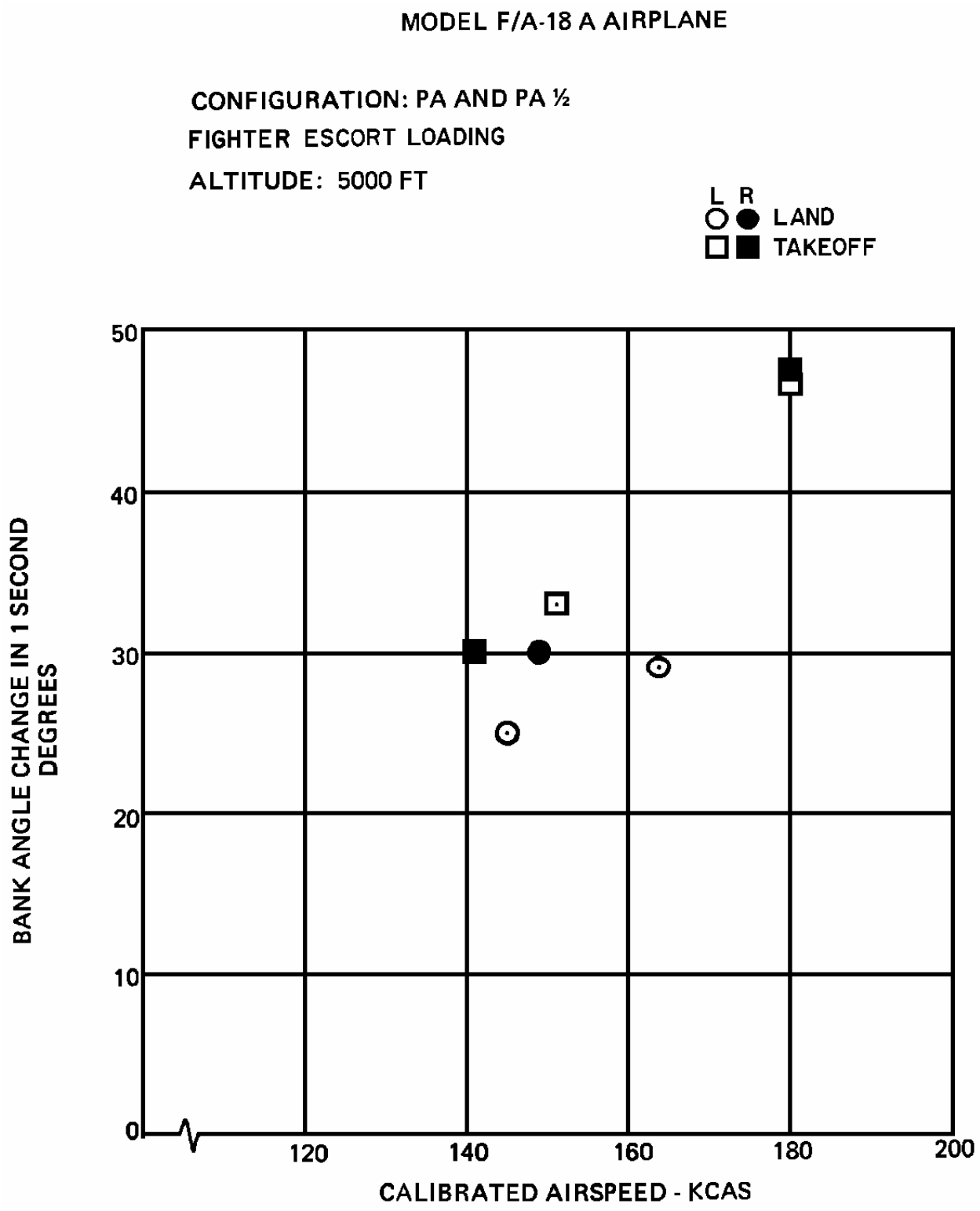


FIGURE 160. **F/A-18A roll performance characteristics in configuration PA (from Navy Rpt No. SA-14R-81).**

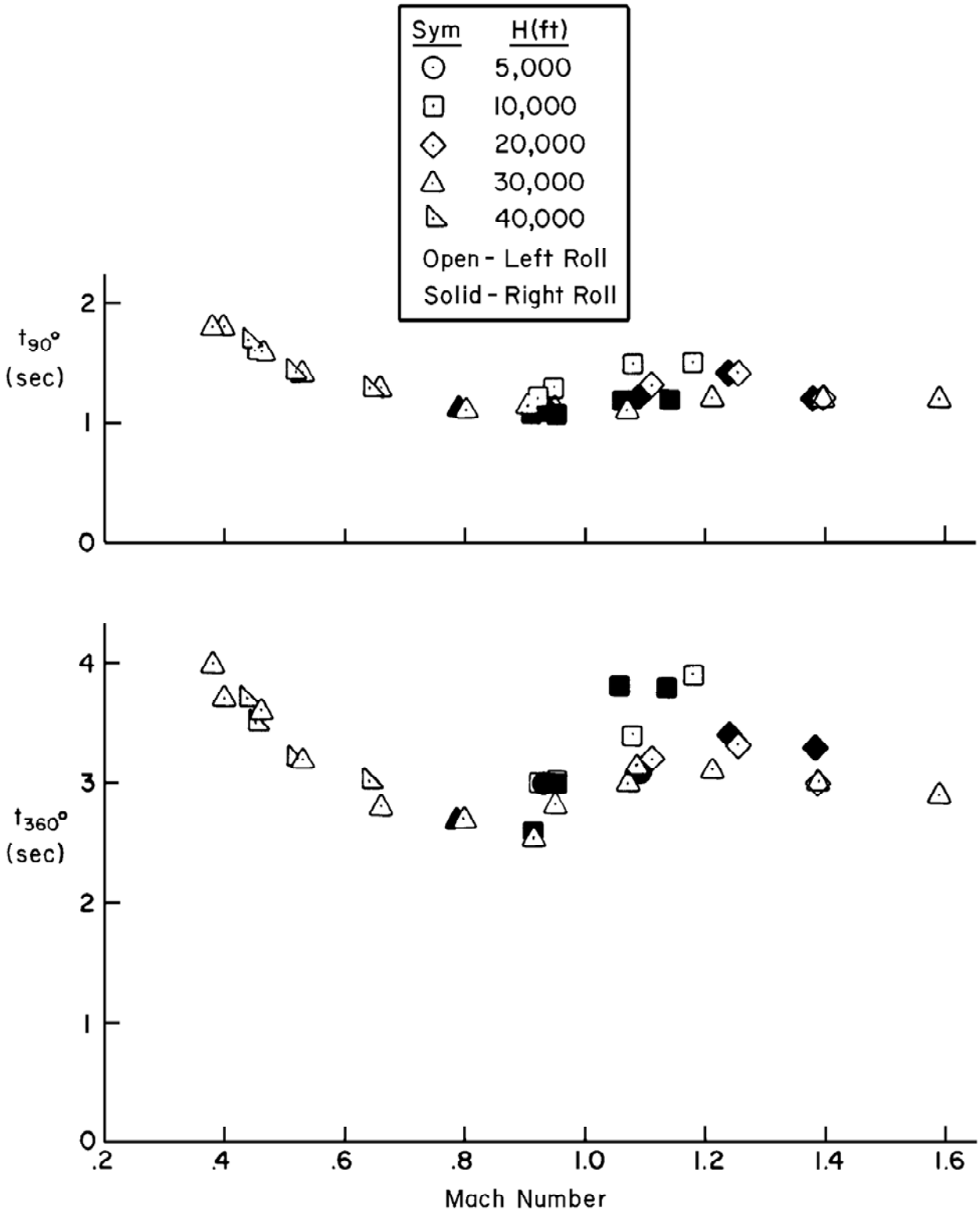


FIGURE 161. YF-16 rolling performance in cruise configuration (AFFTC-TR-75-15).

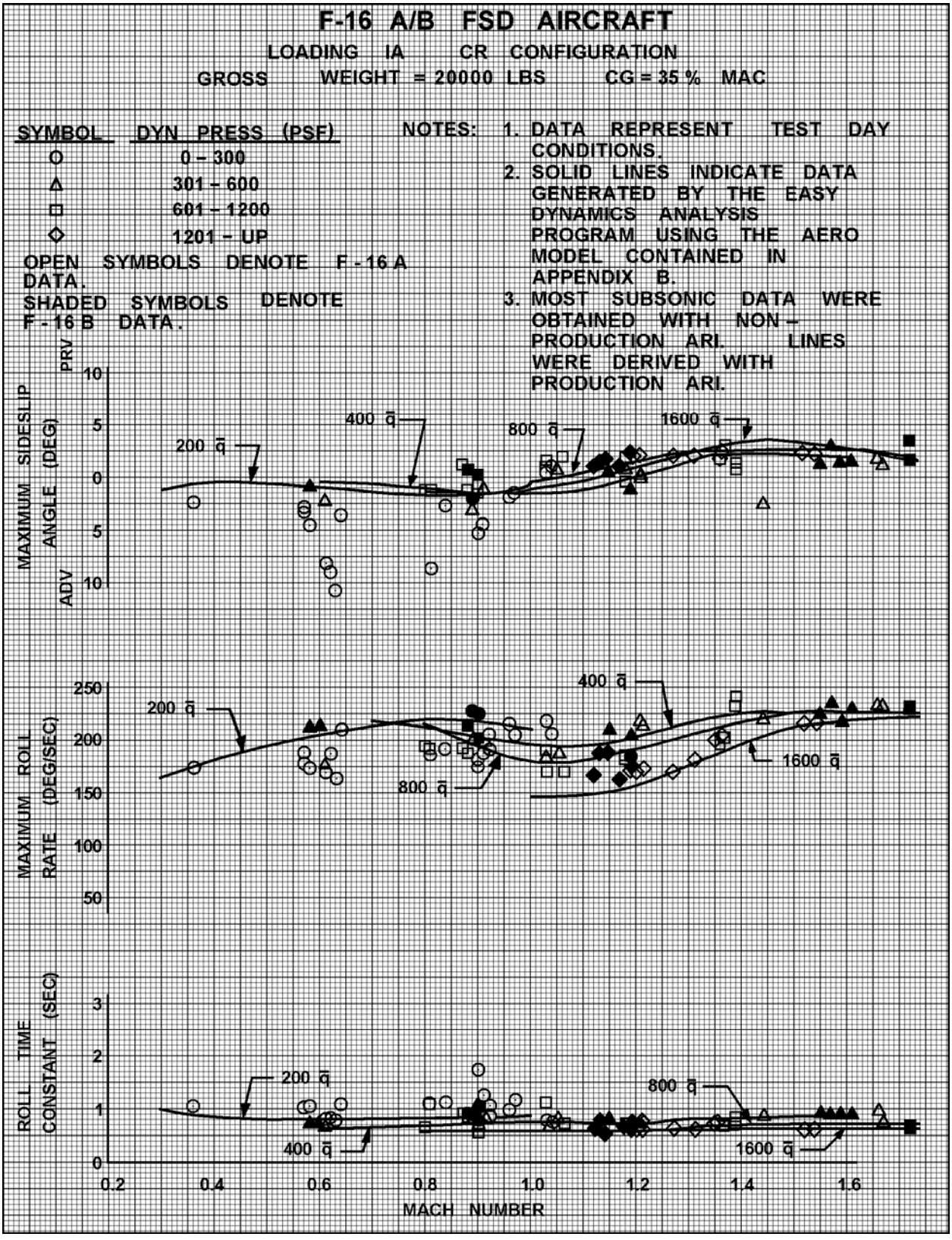


FIGURE 162. F-16A/B roll performance, CR configuration (from AFFTC-TR-79-10).

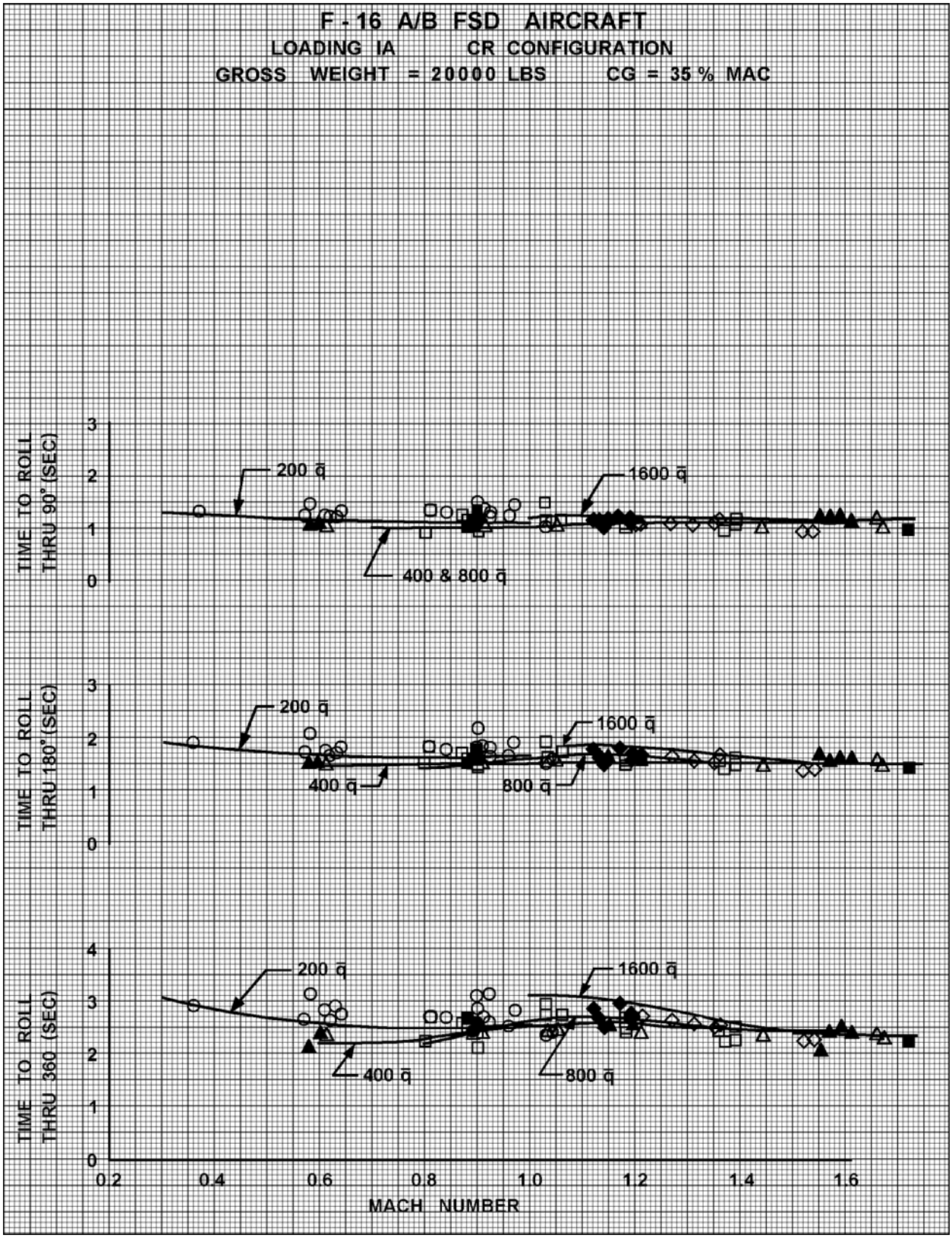


FIGURE 162. F-16A/B roll performance, CR configuration (from AFFTC-TR-79-10) - Continued.

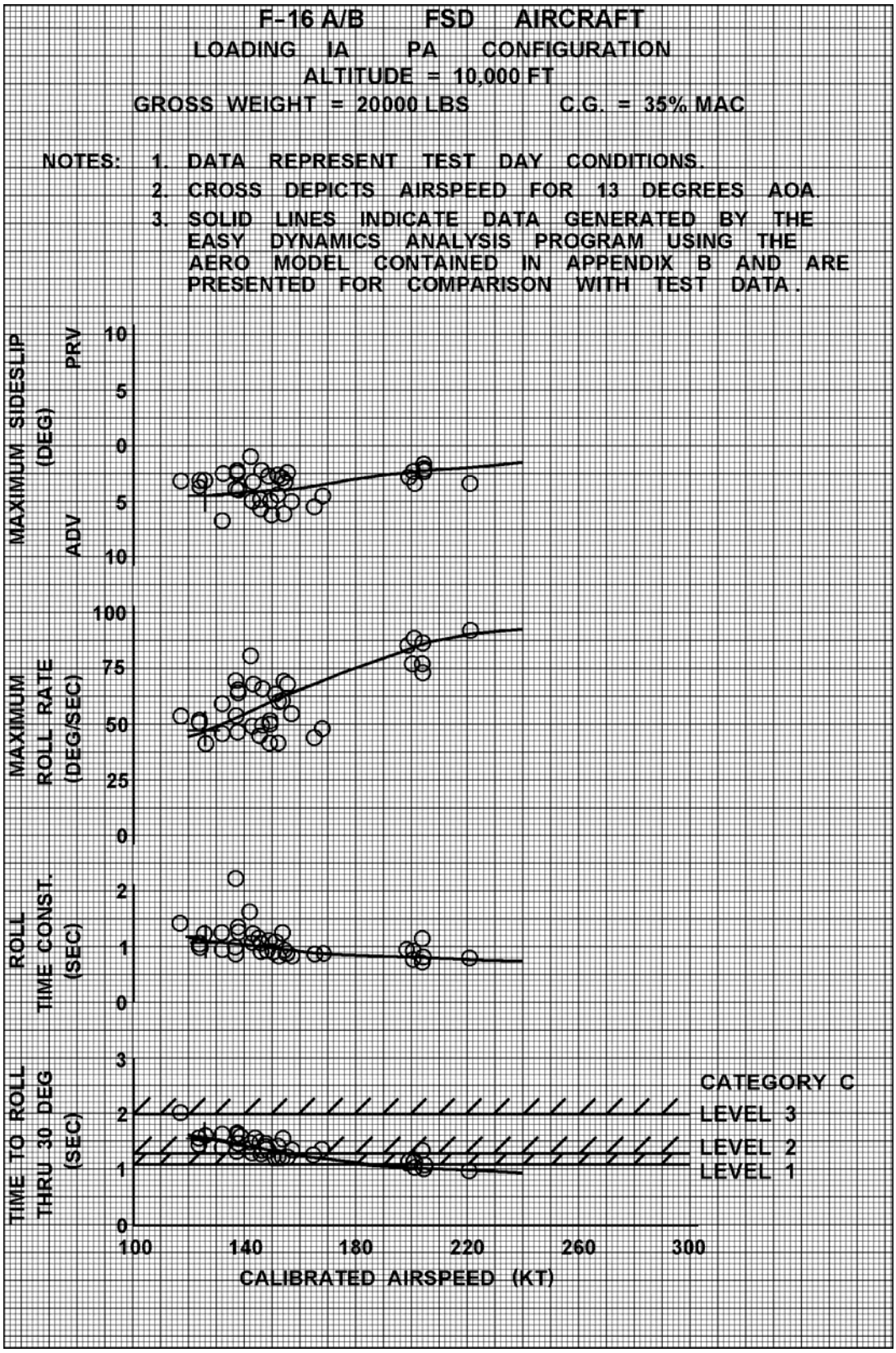


FIGURE 163. F-16A/B roll performance, PA configuration (from AFFTC-TR-79-10).

The requirements apply throughout the normal load factor, as well as the airspeed and altitude range, of each Region of Handling Qualities. Sharp step controller force inputs get the highest performance, and the technique is to roll through the specified bank angle change. In build-ups, a chain is sometimes used to allow quick, accurate partial deflection of the controller. While a bank-and-stop or sidestep maneuver might be somewhat more representative of operational use, rolling through a given bank angle gives more consistent, repeatable flight test results. Nevertheless, these other maneuvers and also gust response shall also be evaluated qualitatively in the course of simulation and flight testing.

Since a true step is virtually impossible to achieve in practice, time may be measured from the midpoint of the control input transient, as shown on figure 164, for the most abrupt input feasible.

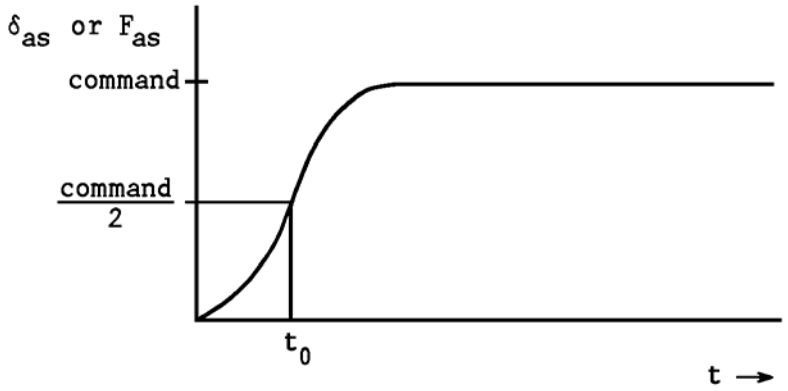


FIGURE 164. Measurement of control input time.

Some of these requirements are divided into airspeed regions or relaxed for heavy loadings along the wing. These provisions indicate some important critical factors. Slow surface actuators will lower the roll performance. Note how this effect was approximated on figure 150; it is better, of course, to perform the analysis more exactly (use more degrees of freedom and an actual ramp that corresponds to the actual rate limit). Surfaces used for both roll and pitch may have to give priority to pitch control, thus at some conditions limiting the amount available for roll. Low speed, high angle of attack (high lateral stability and large aileron yaw) and high dynamic pressure (aeroelastic deformation) are common critical flight conditions.

Of major importance in designing and producing an acceptable aircraft is the definition of the roll axis used in compliance. The roll axis is not specified exactly in these requirements. Its desired orientation varies with the pilot's intent: turns (or straightening out) to modify the flight path, barrel rolls to slow down, aileron rolls to start split-S', etc. The most frequent, usually most important, use is the first-named, for turn entry or exit. With respect to the direction of flight, a roll axis tilted up corresponds to adverse yaw (nose lagging the turn entry) in stability axes; while a nose-down tilt indicates proverse yaw. At high angle of attack, rotation about any axis other than the flight path will generate sideslip, thus exciting Dutch roll motion or even

departure from controlled flight. Other studies have shown that a major contributor to departure is the $p\alpha$ term in the side force equation:

$$\sum Y = m\ V\ (\dot{\beta} + r - p\ \alpha)$$

and $p\alpha_0$ is, of course, zero in stability axes.

However, compared to conventional body axes the cockpit is higher above a flight-path-aligned roll axis at high angles of attack. The result is spurious responses to roll control inputs: lateral acceleration as in the C-5A, F-15, etc.; visual slewing, e.g., of a runway threshold, found troublesome for the YF-16. These effects involve the kinematic relationships:

$$A_{y_p} = V_0\ \dot{\beta} + x_p\ \dot{r} - z_p\ \dot{p}$$

$$v_p = V_0\ \beta + x_p\ r - z_p\ p\ .$$

Direct side force control can be combined with roll control to raise the roll axis if desired. Also, at high angle of attack, rolling about other than the principal x axis creates a flywheel effect producing an incremental nose-up pitching moment, $p^2 I_{xz}$. See 5.1.3 Verification and 5.2.4.3 Lessons Learned for further discussion.

In tracking a target, roll about the sight axis can avoid a troublesome pendulum effect: the initial motion of a depressed sight is opposite to the desired heading change.

All things considered, generally it appears best to generate and measure the roll motion in stability axes, examining the results carefully at high angle of attack, where the difference between stability axes and body axes is greatest. In order to achieve the needed roll performance it may be necessary to accept some uncomfortable lateral accelerations. However, if possible these accelerations shall not exceed the limits established in 5.2.3.2.4.

5.2.3.5.1 Additional roll requirements for Class IV air vehicles.
In Flight Phase CO, the time to bank during 360° rolls initiated at 1-g shall be no more than the limits in table LX. In Flight Phase CO, the time to bank for rolls initiated from coordinated turns, keeping approximately constant normal load factor, at load factors between $0.8n_{o-}$ and $0.8n_{o+}$, shall be no more than the limits in table LXI. At load factors beyond this range, the change in bank angle shall always be in the direction of the roll control command, and the change in bank angle after 1 sec shall be greater than 0.0º. The requirements in Flight Phase GA with large complements of external stores may be relaxed from those specified in table LVIII; however, for any expected external loading the time to bank shall be no more than the limits in table LXII for rolls initiated from coordinated turns, keeping approximately constant normal load factor, at load factors between $0.8n_{o-}$ and $0.8n_{o+}$.

TABLE LX. Flight Phase CO roll performance in 360º rolls.

Level	Speed Range*	Time to Achieve the Stated Bank Angle Change (seconds)			
		$\phi_t = 30°$	$\phi_t = 90°$	$\phi_t = 180°$	$\phi_t = 360°$
1	$V_{o_{min}} \le V < V_{o_{min}} + 20$ kts				
	$V_{o_{min}} + 20$ kts $\le V < 1.4V_{o_{min}}$				
	$1.4V_{o_{min}} \le V < 0.7V_{o_{max}}$				
	$0.7V_{o_{max}} \le V \le V_{o_{max}}$				
2	$V_{min} \le V < V_{min} + 20$ kts				
	$V_{min} + 20$ kts $\le V < 1.4V_{min}$				
	$1.4V_{min} \le V < 0.7V_{max}$				
	$0.7V_{max} \le V \le V_{max}$				
3	$V_{min} \le V < V_{min} + 20$ kts				
	$V_{min} + 20$ kts $\le V < 1.4V_{min}$				
	$1.4V_{min} \le V < 0.7V_{max}$				
	$0.7V_{max} \le V \le V_{max}$				

*NOTE: some of these speed ranges may not exist in particular cases. The requirements apply, of course, only within speed ranges which do.

TABLE LXI. Flight Phase CO loaded roll performance.

Level	Speed Range*	Time to Achieve the Stated Bank Angle Change (seconds)			
		$\phi_t = 30°$	$\phi_t = 50°$	$\phi_t = 90°$	$\phi_t = 180°$
1	$V_{o_{min}} \le V < V_{o_{min}} + 20$ kts				
	$V_{o_{min}} + 20$ kts $\le V < 1.4V_{o_{min}}$				
	$1.4V_{o_{min}} \le V < 0.7V_{o_{max}}$				
	$0.7V_{o_{max}} \le V \le V_{o_{max}}$				
2	$V_{min} \le V < V_{min} + 20$ kts				
	$V_{min} + 20$ kts $\le V < 1.4V_{min}$				
	$1.4V_{min} \le V < 0.7V_{max}$				
	$0.7V_{max} \le V \le V_{max}$				
3	$V_{min} \le V < V_{min} + 20$ kts				
	$V_{min} + 20$ kts $\le V < 1.4V_{min}$				
	$1.4V_{min} \le V < 0.7V_{max}$				
	$0.7V_{max} \le V \le V_{max}$				

*NOTE: some of these speed ranges may not exist in particular cases. The requirements apply, of course, only within speed ranges which do.

TABLE LXII. Flight Phase GA loaded roll performance.

Level	Speed Range*	Time to Achieve the Stated Bank Angle Change (seconds)			
		$\phi_t = 30°$	$\phi_t = 50°$	$\phi_t = 90°$	$\phi_t = 180°$
1	$V_{o_{min}} \le V < V_{o_{min}} + 20$ kts				
	$V_{o_{min}} + 20$ kts $\le V < 1.4V_{o_{min}}$				
	$1.4V_{o_{min}} \le V < 0.7V_{o_{max}}$				
	$0.7V_{o_{max}} \le V \le V_{o_{max}}$				
2	$V_{min} \le V < V_{min} + 20$ kts				
	$V_{min} + 20$ kts $\le V < 1.4V_{min}$				
	$1.4V_{min} \le V < 0.7V_{max}$				
	$0.7V_{max} \le V \le V_{max}$				
3	$V_{min} \le V < V_{min} + 20$ kts				
	$V_{min} + 20$ kts $\le V < 1.4V_{min}$				
	$1.4V_{min} \le V < 0.7V_{max}$				
	$0.7V_{max} \le V \le V_{max}$				

*NOTE: some of these speed ranges may not exist in particular cases. The requirements apply, of course, only within speed ranges which do.

REQUIREMENT RATIONALE

Roll power is specified in terms of bank angle change in a given time, a form related to operational use of the air vehicle, to allow necessary maneuvering and attitude regulation.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.4.1 through 3.3.4.1.2
MIL-STD-1797: 4.5.8.1.

Note that 5.2.3.5.1 applies only for Class IV air vehicles and is not applicable if the vehicle under consideration is not a Class IV air vehicle.

Recommended values for 1-g roll performance table LX:

Level	Speed Range	Time to Achieve the Stated Bank Angle Change (seconds)			
		$\phi_t = 30°$	$\phi_t = 90°$	$\phi_t = 180°$	$\phi_t = 360°$
1	$V_{o_{min}} \le V < V_{o_{min}} + 20$ kts	1.0			
	$V_{o_{min}} + 20$ kts $\le V < 1.4V_{o_{min}}$		1.4	2.3	4.1
	$1.4V_{o_{min}} \le V < 0.7V_{o_{max}}$		1.0	1.6	2.8
	$0.7V_{o_{max}} \le V \le V_{o_{max}}$		1.4	2.3	4.1
2	$V_{min} \le V < V_{min} + 20$ kts	1.6			
	$V_{min} + 20$ kts $\le V < 1.4V_{min}$	1.3			
	$1.4V_{min} \le V < 0.7V_{max}$		1.3	2.0	3.4
	$0.7V_{max} \le V \le V_{max}$		1.7	2.6	4.4
3	$V_{min} \le V < V_{min} + 20$ kts	2.5			
	$V_{min} + 20$ kts $\le V < 1.4V_{min}$	2.0			
	$1.4V_{min} \le V < 0.7V_{max}$		1.7	3.0	
	$0.7V_{max} \le V \le V_{max}$		2.1		

Recommended values for Flight Phase CO loaded roll performance table LXI:

Level	Speed Range	Time to Achieve the Stated Bank Angle Change (seconds)			
		$\phi_t = 30°$	$\phi_t = 50°$	$\phi_t = 90°$	$\phi_t = 180°$
1	$V_{o_{min}} \leq V < V_{o_{min}} + 20$ kts	1.0			
	$V_{o_{min}} + 20$ kts $\leq V < 1.4V_{o_{min}}$		1.1		
	$1.4V_{o_{min}} \leq V < 0.7V_{o_{max}}$			1.1	2.2
	$0.7V_{o_{max}} \leq V \leq V_{o_{max}}$		1.0		
2	$V_{min} \leq V < V_{min} + 20$ kts	1.6			
	$V_{min} + 20$ kts $\leq V < 1.4V_{min}$	1.3			
	$1.4V_{min} \leq V < 0.7V_{max}$			1.4	2.8
	$0.7V_{max} \leq V \leq V_{max}$		1.4		
3	$V_{min} \leq V < V_{min} + 20$ kts	2.5			
	$V_{min} + 20$ kts $\leq V < 1.4V_{min}$	2.0			
	$1.4V_{min} \leq V < 0.7V_{max}$			1.7	3.4
	$0.7V_{max} \leq V \leq V_{max}$		1.7		

Recommended values for Flight Phase GA loaded roll performance table LXII:

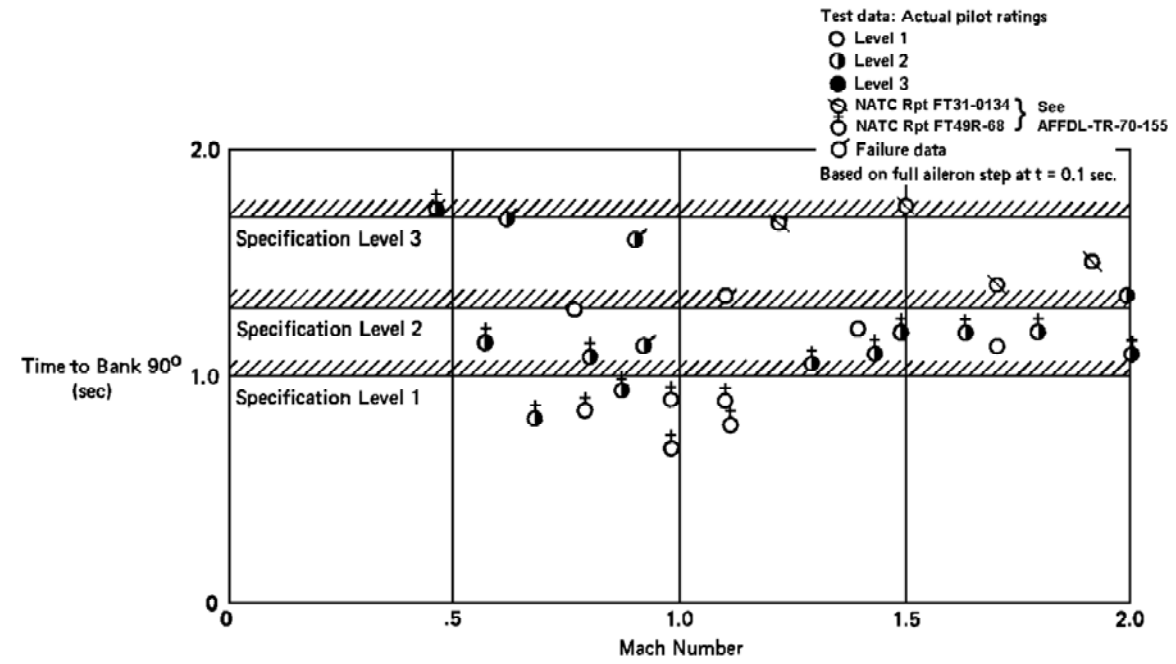
Level	Speed Range	Time to Achieve the Stated Bank Angle Change (seconds)			
		$\phi_t = 30°$	$\phi_t = 50°$	$\phi_t = 90°$	$\phi_t = 180°$
1	$V_{o_{min}} \leq V < V_{o_{min}} + 20$ kts	1.5			
	$V_{o_{min}} + 20$ kts $\leq V < 1.4V_{o_{min}}$		1.7		
	$1.4V_{o_{min}} \leq V < 0.7V_{o_{max}}$			1.7	3.0
	$0.7V_{o_{max}} \leq V \leq V_{o_{max}}$		1.5		
2	$V_{min} \leq V < V_{min} + 20$ kts	2.8			
	$V_{min} + 20$ kts $\leq V < 1.4V_{min}$	2.2			
	$1.4V_{min} \leq V < 0.7V_{max}$			2.4	4.2
	$0.7V_{max} \leq V \leq V_{max}$		2.4		
3	$V_{min} \leq V < V_{min} + 20$ kts	4.4			
	$V_{min} + 20$ kts $\leq V < 1.4V_{min}$	3.8			
	$1.4V_{min} \leq V < 0.7V_{max}$			3.4	6.0
	$0.7V_{max} \leq V \leq V_{max}$		3.4		

REQUIREMENT LESSONS LEARNED

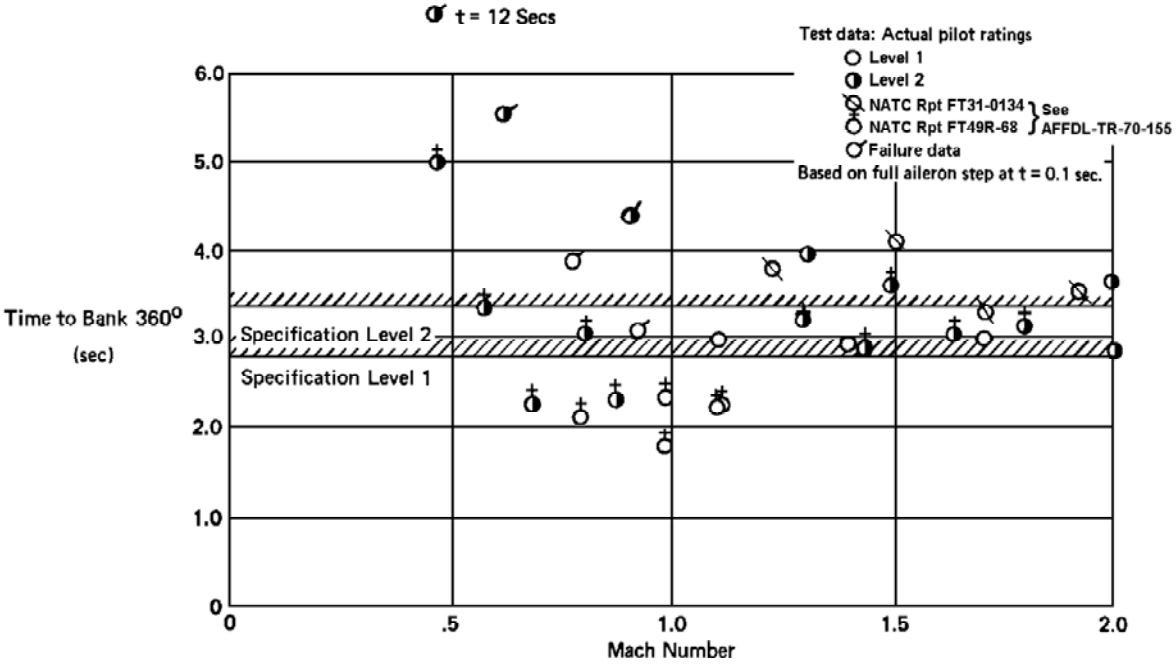
The recommended tables, definitions, and wording of this requirement are collations of the various roll control effectiveness sections of MIL-F-8785C.

Data for the F-4 (AFFDL-TR-70-155) are given on figure 165. Times to bank for these figures were not actual test values, but were calculated from known roll characteristics for various F-4 aircraft. Therefore, there is some inherent inaccuracy in the values of ϕ_t plotted. For Category A flight (figure 165) it is suggested that the limits may be too severe.

The boundaries drawn on figure 165 for times to bank 90º and 360º are for the most stringent recommended requirements of table LX; i.e., for the medium speed range ($1.4V_{min} \leq V < 0.7V_{max}$). The data tend to support the Level 1 limits, but suggest that the Level 2 limits could be relaxed considerably.



a) Time to Bank 90 deg



b) Time to Bank 360 deg

FIGURE 165. F-4 roll control effectiveness; CO configuration. Limits shown for medium speed range, loaded rolls (from AFFDL-TR-70-155).

F-5E data (figure 166, from AFFDL-TR-78-171) for Flight Phase CO at elevated load factors do not agree well with the recommended Level 1 limits of table LXI. The F-5E meets the requirements only in the high-speed range ($0.7V_{max} \leq V \leq V_{max}$). AFFDL-TR-78-171 describes the roll performance as "very satisfactory in operational use," and according to AFFDL-TR-71-134, "the F-5 has exhibited favorable roll performance in air combat situations where both the rudder and ailerons were used at low speed and at high angles of attack."

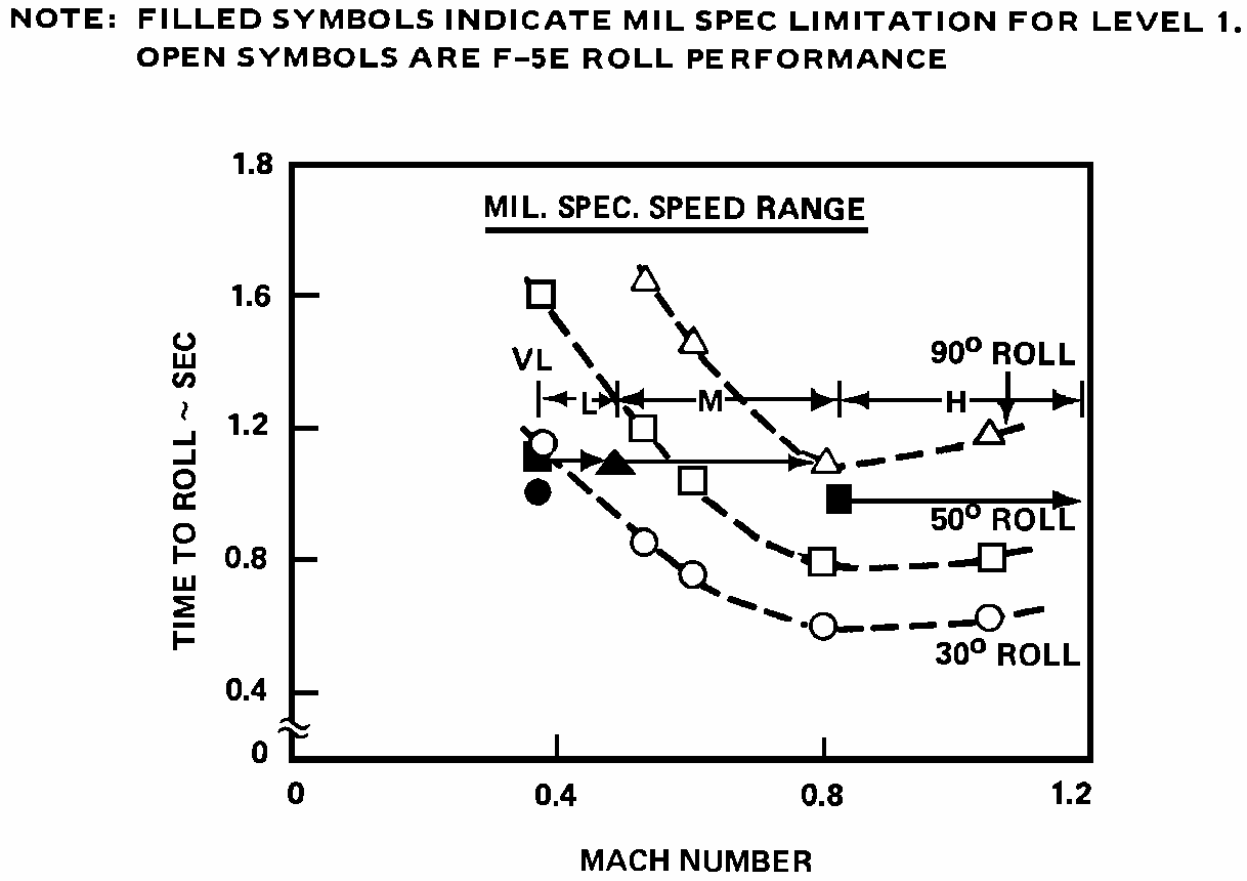


FIGURE 166. F-5E roll performance at 0.8 n_L, configuration CO (from AFFDL-TR-78-171).

Navy Rpt No. SA-14R-81 documents performance and handling qualities testing of a Navy F/A-18A airplane. This test airplane included various control system modifications over the prototype F/A-18A to correct deficiencies in the airplane's roll response. Results of 1-g, 360-deg rolls in cruise configuration are shown on figure 167 and figure 168. At moderate altitudes and at low speeds, the F/A-18A roll performance is quite good. However, for combinations of low altitude and high speed, the airplane is seen to be extremely sluggish. Steady-state roll rates as low as 50º/sec were encountered at 500 ft altitude. From Navy Rpt No. SA-14R-81:

> "In that portion of the flight envelope where the time to 90° is less than or equal to 1 sec, the fleet pilot will be able to rapidly and efficiently maneuver the airplane to track an aggressive target as well as perform rapid evasive maneuvers

> required during air-to-air and air-to-ground tactical maneuvers…. The excessive time to roll to 90° at low to medium altitude, high q conditions will preclude the pilot's ability to effectively perform the air-to-air and air-to-ground missions."

The difference in time to bank for left versus right rolls (see figure 167 and figure 168) was due to a lateral trim offset in the F/A-18A tested: "The large positive [control stick] deflection required for 1-g level flight significantly reduced the amount of control deflection change available to command a right roll as opposed to a left roll." While figure 167 shows Level 1 roll performance at low airspeeds, fine, precise control was found to be sluggish: "the pilot was unable to perform the fine tracking task at 200 KCAS/2g [15,000 ft altitude] (HQR 7)…. The sluggish lateral response characteristics…rapidly led to an out-of-phase condition and resultant nondivergent lateral PIO." It should be noted, however, that there may have been other problems resulting from T_R (5.2.3.1.1) or τ_{ϕ_e} (5.2.3.1.5).

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall precede or accompany careful build-up to suspected critical flight conditions. In 360º rolls, even test pilots tend to relax the control input before reaching 360º bank.

It must be noted that external tanks (both full and empty centerline and empty wing tanks) shall be considered in choosing the air-to-air configurations.

At high angle of attack, pilots of some aircraft (F-4 and F/A-18 for example) use the rudder pedals to roll, because of the large adverse yaw due to the roll control effectors. Use of an aileron-rudder interconnect or rudder-aileron interconnect, may be necessary to provide satisfactory roll performance at high angles of attack.

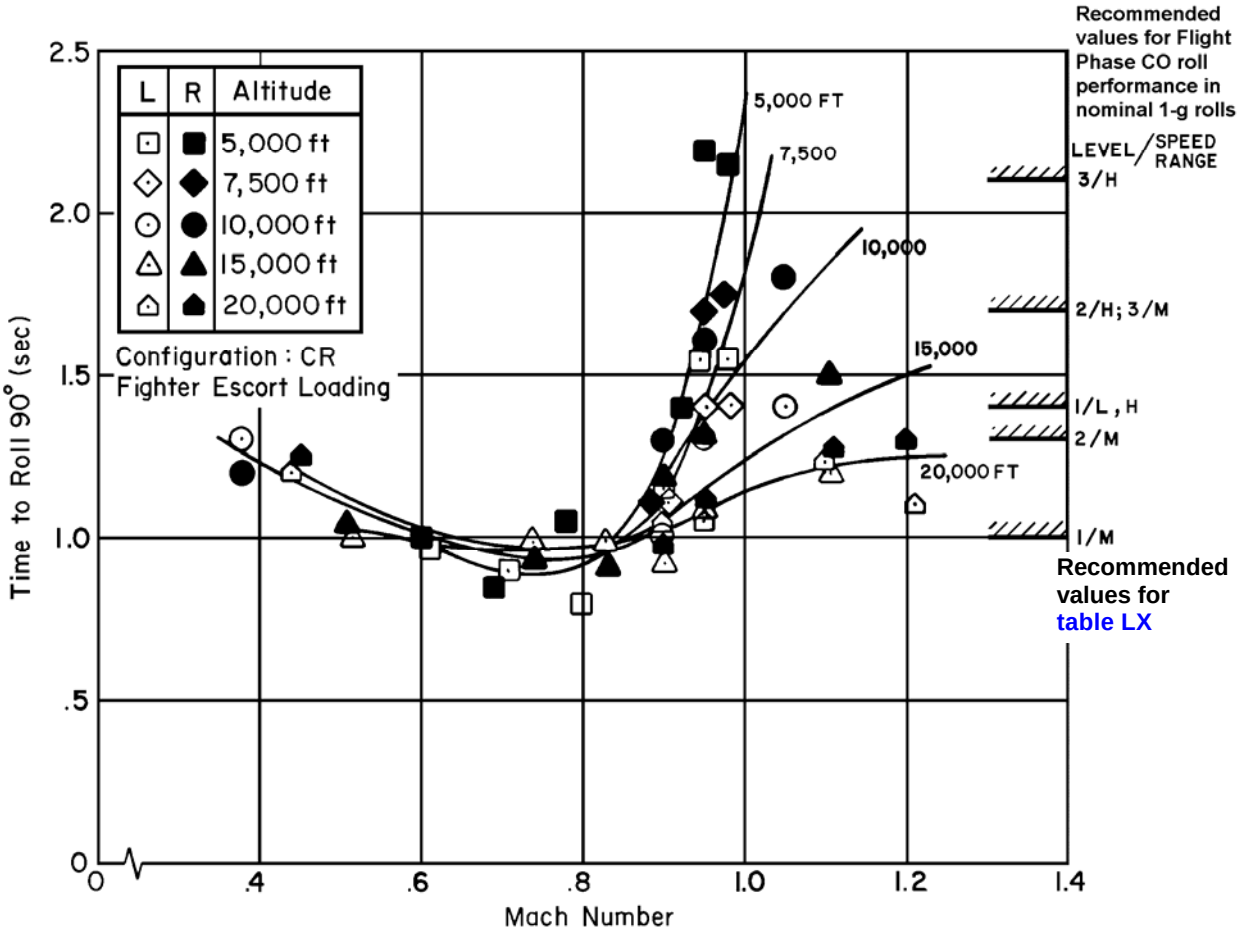


FIGURE 167. Time to roll 90° versus match for F/A-18A (Navy Rpt No. SA-14R-81).

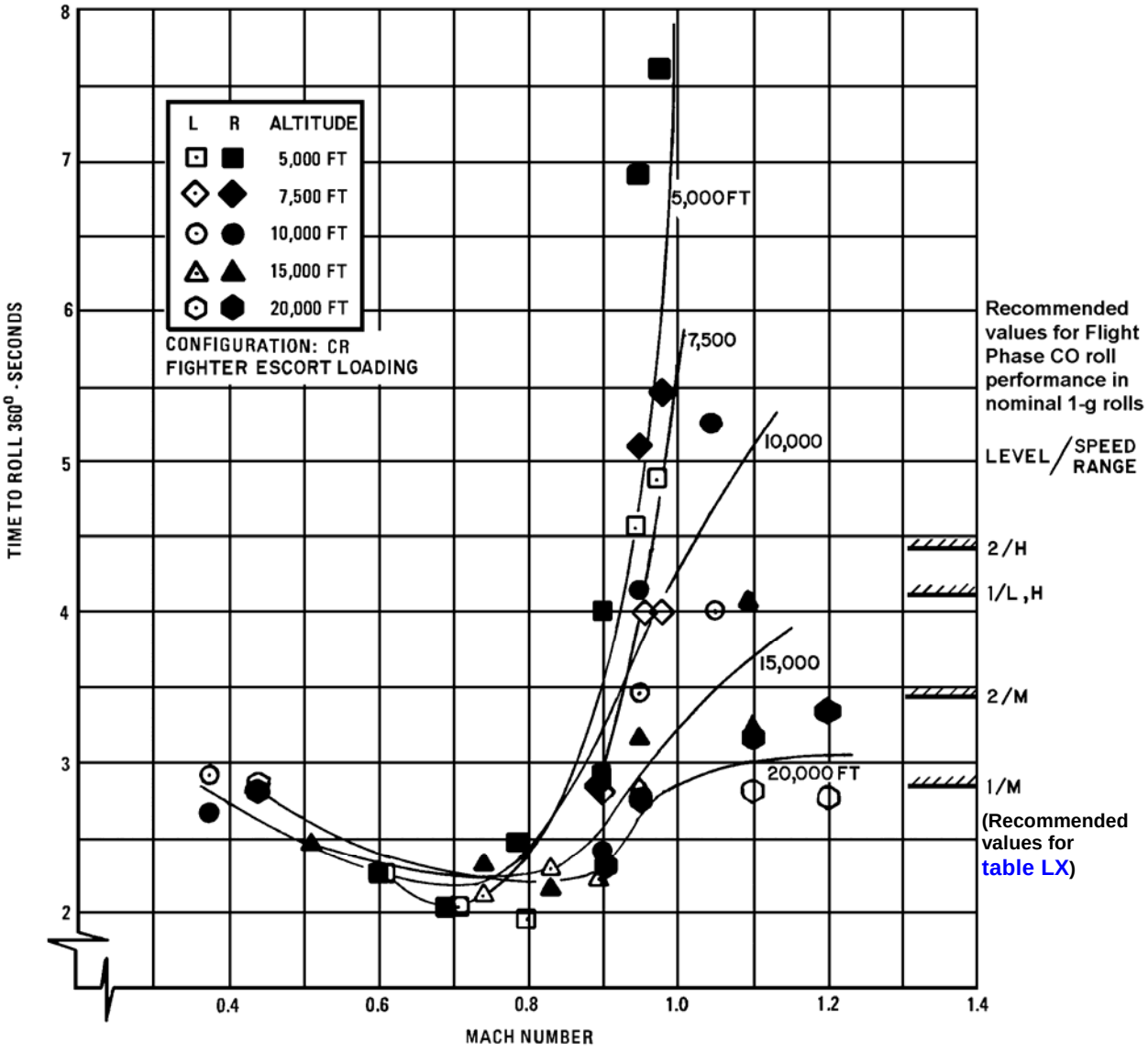


FIGURE 168. Time to roll 360° versus match for F/A-18A (Navy Rpt No. SA-14R-81).

5.2.3.5.2 Roll termination.
Clean and with symmetric and asymmetric air-to-air and air-to-ground loadings, abrupt roll control inputs shall be used to terminate the roll maneuvers after achieving the bank angle changes specified in 5.2.3.5. These abrupt roll control inputs shall not cause air vehicle motions which result in loss of control, stall, or exceedance of structural limits.

REQUIREMENT RATIONALE

Abrupt termination of rolling maneuvers can cause large overshoots in sideslip, angle of attack and load factor. These characteristics shall not detract from the mission capability of the aircraft.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.5.8.1.

REQUIREMENT LESSONS LEARNED

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall precede or accompany careful build-up to suspected critical flight conditions.

5.2.3.5.3 Roll control power with asymmetric loads.
For Category A Flight Phases, with any asymmetric loading necessary to meet operational mission requirements, roll control power shall be sufficient to hold the wings level at the maximum load factors specified in 5.2.2.1.7.3 with adequate control margin (5.1.3).

REQUIREMENT RATIONALE

This requirement is necessary for service employment. It has flight safety implications. Normal operation commonly involves the selective firing of weapons or release of stores, and hung stores.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.4.1.2
MIL-STD-1797: 4.5.8.6.

REQUIREMENT LESSONS LEARNED

The scope of this requirement has been expanded from MIL-F-8785C to cover all applicable Flight Phases instead of just GA.

A critical case for the L-19 was the addition of a wire-laying mission involving carriage of a large reel under one wing. Some airplanes—the F-15 is a recent example—have been prone to develop significant fuel asymmetries due to small differences in head pressure between left and right wing fuel pumps and other fuel system tolerances. Dive pullouts involving elevated load factors will accentuate the effects of loading asymmetries. Some F-100s were lost from asymmetric operation of leading-edge slats (powered, aerodynamically operated on their own, without pilot action) in dive-bombing pullouts.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Verification is a matter of flying at the envelope boundaries with those asymmetric loadings needed to meet operational requirements or likely to be encountered. Analysis and simulation shall precede or accompany careful build-up to suspected critical flight conditions. Low airspeed and high normal load factor will generally be critical. The level of static stability may affect the critical conditions. See the equations for asymmetric thrust in 5.2.3.11 and 5.2.3.11.2.

5.2.3.6 Lateral-directional control with speed changes.
When initially trimmed directionally with symmetric power, the trim change with speed shall be such that wings-level straight flight can be maintained over a speed range of ±30% of the trim speed or ±100 kts equivalent airspeed, whichever is less, (except where limited by the boundaries of the ROTH) with yaw control pedal forces not greater than the values of table LXIII without retrimming.

TABLE LXIII. Maximum yaw control force during speed changes.

Level	Maximum Yaw Control Force (lbs)
1	
2	
3	

REQUIREMENT RATIONALE

This requirement is to insure that speed effects on yawing moment are not unduly distracting.

REQUIREMENT LESSONS LEARNED

The scope of this requirement has been expanded from MIL-F-8785C to cover all applicable Flight Phases instead of just GA.

A critical case for the L-19 was the addition of a wire-laying mission involving carriage of a large reel under one wing. Some airplanes—the F-15 is a recent example—have been prone to develop significant fuel asymmetries due to small differences in head pressure between left and right wing fuel pumps and other fuel system tolerances. Dive pullouts involving elevated load factors will accentuate the effects of loading asymmetries. Some F-100s were lost from asymmetric operation of leading-edge slats (powered, aerodynamically operated on their own, without pilot action) in dive-bombing pullouts.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Verification is a matter of flying at the envelope boundaries with those asymmetric loadings needed to meet operational requirements or likely to be encountered. Analysis and simulation shall precede or accompany careful build-up to suspected critical flight conditions. Low airspeed and high normal load factor will generally be critical. The level of static stability may affect the critical conditions. See the equations for asymmetric thrust in 5.2.3.11 and 5.2.3.11.2.

5.2.3.6 Lateral-directional control with speed changes.
When initially trimmed directionally with symmetric power, the trim change with speed shall be such that wings-level straight flight can be maintained over a speed range of ±30% of the trim speed or ±100 kts equivalent airspeed, whichever is less, (except where limited by the boundaries of the ROTH) with yaw control pedal forces not greater than the values of table LXIII without retrimming.

TABLE LXIII. Maximum yaw control force during speed changes.

Level	Maximum Yaw Control Force (lbs)
1	
2	
3	

REQUIREMENT RATIONALE

This requirement is to insure that speed effects on yawing moment are not unduly distracting.

5.2.3.6.1 Lateral-directional control with speed changes - asymmetric loads.
When initially trimmed directionally with each expected asymmetric loading (including hung stores and all asymmetric loadings necessary to meet operational mission requirements) at any speed within the ROSH, it shall be possible to maintain a straight flight path at all other speeds within the limits of the ROSH with yaw-control-pedal forces not greater than the limits of table LXIV, without retrimming.

TABLE LXIV. Maximum yaw control forces for asymmetric loads.

Level	Maximum Yaw Control Force (lbs)
1	
2	
3	

REQUIREMENT RATIONALE

This requirement is included to assure adequate lateral-directional control power to compensate for any specified condition of asymmetric loading and to insure asymmetric loadings do not result in excessive yaw pedal force demands on the pilot. This requirement is necessary for service employment. It has flight safety implications.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1.3 and 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.5.1.1
MIL-STD-1797: 4.5.8.6, 4.6.6.3, and 4.6.7.5.

Recommended values for table LXIV:

Level	Maximum Yaw Control Force (lbs)
1	100
2	100
3	180

REQUIREMENT LESSONS LEARNED

These limits are quite high for prolonged application; but then retrimming would be expected. If asymmetric loadings are expected to be encountered regularly, the designer may want to reduce the yaw control force limits. The recommended limits have been taken from MIL-F-8785C and 180 lbs seems unreasonably high (see 5.2.3.10).

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall precede or accompany careful build-up to suspected critical flight conditions. Flight testing with the asymmetric loadings selected in section 5 shall be conducted throughout the ROSH. Low airspeed is generally critical.

5.2.3.7 Yaw control forces in waveoff (go-around).
The response to thrust, configuration, and airspeed change shall be such that the pilot can maintain straight flight during waveoff (go-around) initiated at speeds down to V_S(PA) with yaw control pedal forces not exceeding the values in table LXV when trimmed at $V_{o_{min}}$(PA). Bank angles up to 5º are permitted.

TABLE LXV. Maximum yaw control force for waveoff (go-around).

Level	Maximum Pedal Force (lbs)
1	
2	
3	

REQUIREMENT RATIONALE

The possibility of large, transient yaw pedal forces being needed on initiation of go-arounds necessitates a limit.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.5.2
MIL-STD-1797: 4.6.7.7.

Recommended values for table LXV:

Air Vehicle Class	Level	Maximum Pedal Force (lbs)
Propeller-driven Class IV and all propeller-driven carrier-based air vehicles	1 and 2	100
	3	180
All others	1 and 2	40
	3	180

REQUIREMENT LESSONS LEARNED

Pedal forces tend to be very high during the go-around due to the large change in configuration that occurs between the approach and go-around. The requirement values were taken directly from MIL-F-8785C. Lower limits are warranted to accommodate female and smaller male pilots, if appropriate data can be made available. This very important requirement shall be applied for all aircraft, at various weights and loadings, to assure that go-arounds will not overtax the pilot at a very stressful, busy time.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall precede or accompany careful build-up to suspected critical flight conditions. This is one of several flying qualities to be checked during go-arounds. Go-arounds may be simulated at a safe (but low) altitude, covering the speed range specified. Aft center of gravity, low airspeed-low weight, and high airspeed-high weight shall be checked.

5.2.3.8 Lateral-directional control forces and displacements.
The lateral-directional control forces and displacements shall meet the requirements of 5.2.3.8.1 through 5.2.3.8.6. When a cockpit control is adjustable for pilot physical dimensions or comfort, the control force limits in these requirements refer to the mean adjustment. A force referred to any other adjustment shall not differ by more than 10% from the force referred to the mean adjustment.

REQUIREMENT RATIONALE

The 10% force allowance for control adjustments is intended to assure keeping within the threshold of pilot perception.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

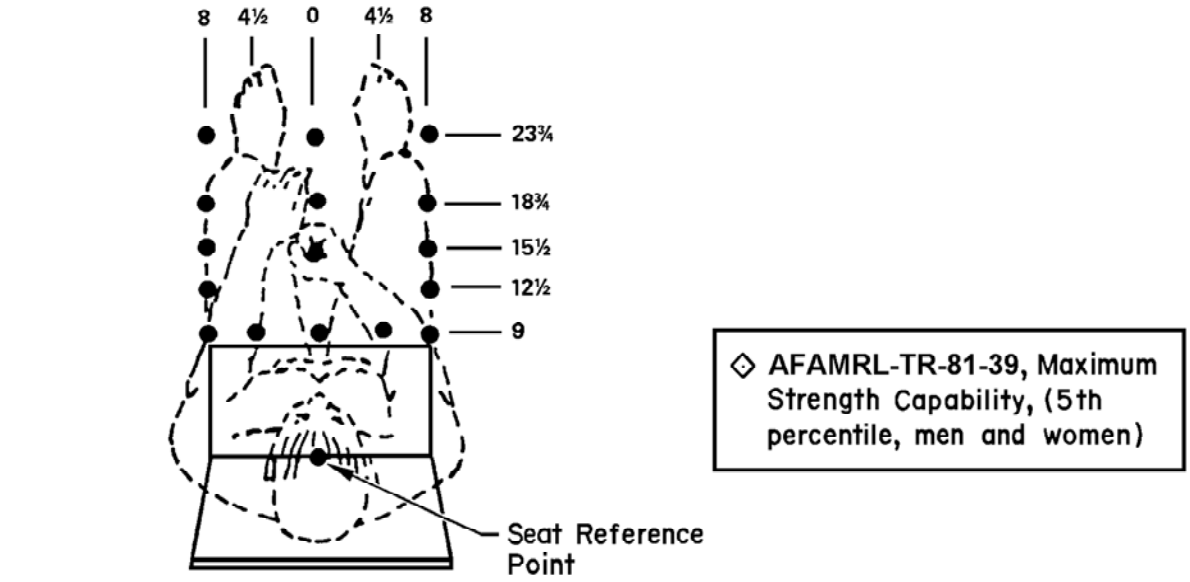
The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.5.2.4

MIL-STD-1797: 4.1.12.3, 4.5.9, and 4.6.7.

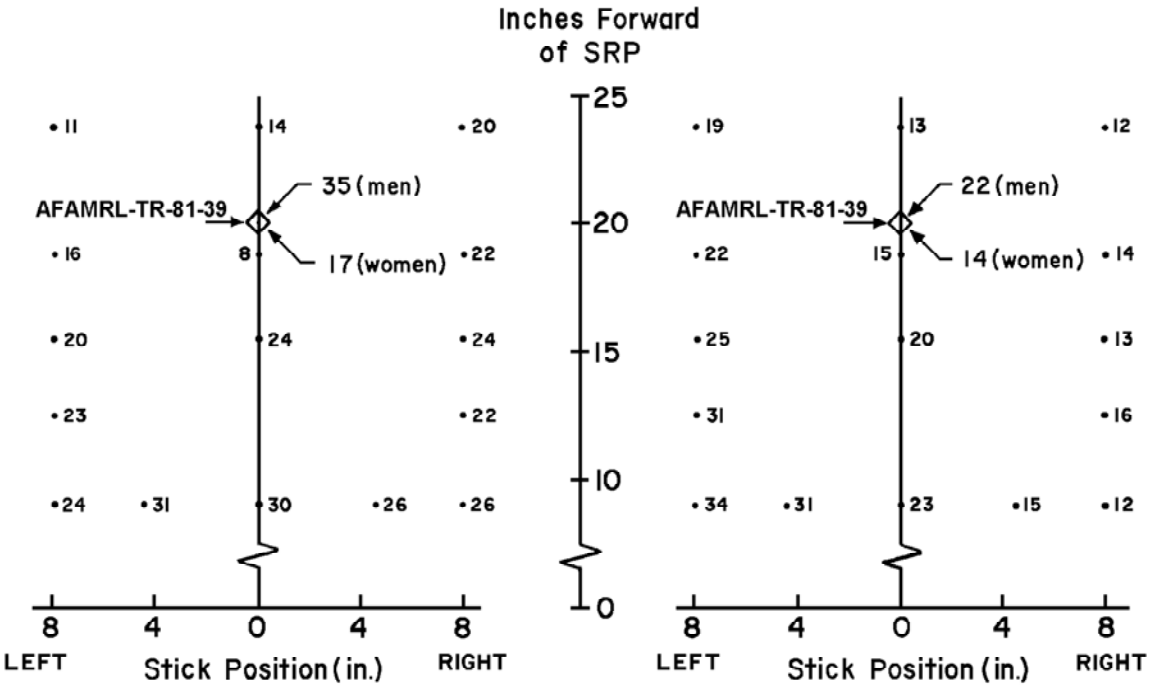
REQUIREMENT LESSONS LEARNED

The requirements of this section cover broad areas for forces and displacements of roll and yaw controllers. Many of the requirements (most specifically, the roll axis control force limits) set absolute upper limits on allowable control forces in various maneuvers. As with the pitch axis forces 5.2.2.1.8, there is some concern as to whether the limits are reasonable for continuous maneuvering, and whether they are attainable by female pilots. While any substantiating data are scarce, there are some available dealing with maximum forces for single-application tasks (Human Engineering Guide to Equipment Design and AFAMRL-TR-81-39). The purpose of this discussion is to briefly review these data. Since there are many variables involved in developing adequate controller characteristics, no attempt has been made to set any new requirements based upon this information; it is intended only as information.

Figure 169 shows the effect of arm/stick geometry on maximum applied force to the left and to the right for the 5th percentile male, and figure 170 shows the effect of upper arm angle on maximum applied force to the left and to the right for the 5th and 95th percentile male. The data in these illustrations are from Human Engineering Guide to Equipment Design. Single test points from a more recent study by McDaniel (AFAMRL-TR-81-39) are show on figure 169 as a comparison between male and female strength characteristics for operating an aircraft control stick. Figure 169b and figure 169c show that the maximum applied force to the left and to the right (depending on arm/stick geometry) varies and is not symmetric. The difference in strength characteristics between men and women (single points from AFAMRL-TR-81-39) shown on figure 169b and figure 169c is almost a factor of two. There is also a large difference in the forces attained by the men in the two tests (e.g., the data on figure 169b show about 8 lbs at the same location where the AFAMRL-TR-81-39 tests show 35 lbs). The reason for this difference is not known.



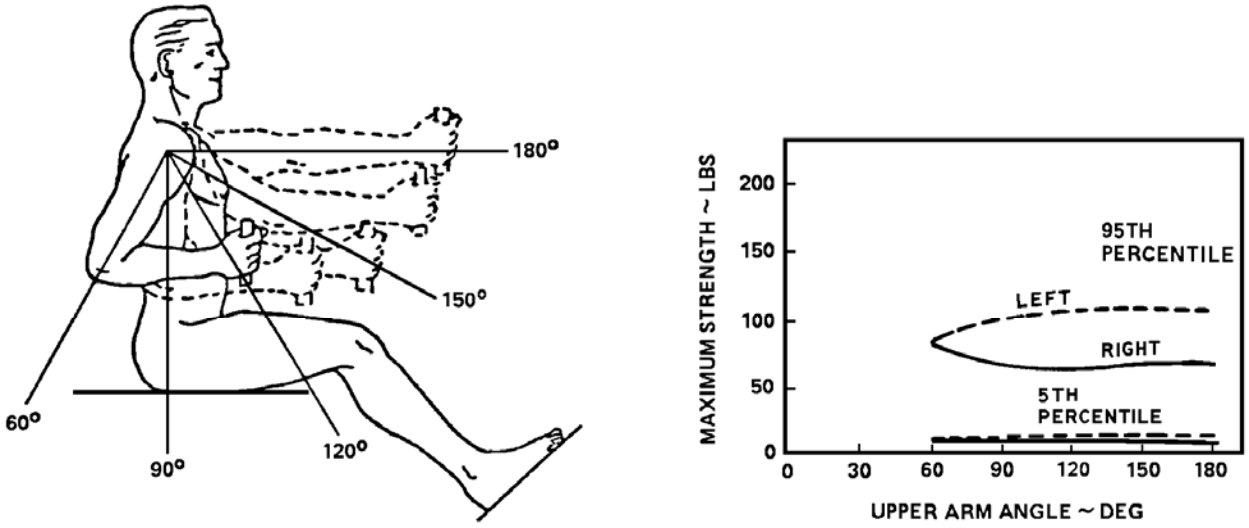
a) Physical Layout for Stick Force Tests (stick 13½ inches above seat Reference point)
See Human Engineering Guide to Equipment Design



b) Maximum Force to the Left (lb)

c) Maximum Force to the Right (lb)

FIGURE 169. Effect of arm/stick geometry on maximum applied force to the left and to the right by the right arm for the 5th percentile male (Human Engineering Guide to Equipment Design).



a) Physical Layout For Test

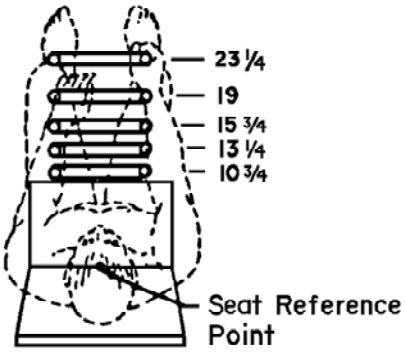
b) Maximum Strength vs. Upper Arm Angle

FIGURE 170. Effect of upper arm angle on maximum applied force to the left and to the right for the 5th and 95th percentile male (from Human Engineering Guide to Equipment Design).

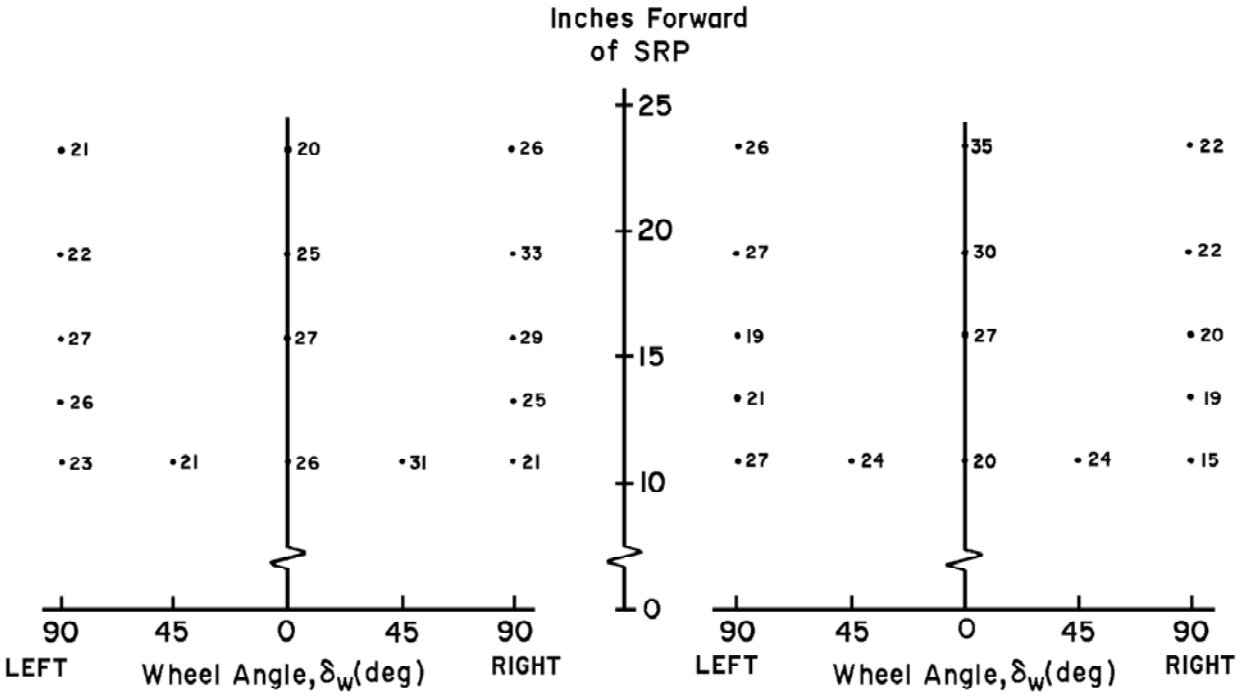
The 5th, 50th, and 95th percentiles of maximum forces exerted on an aircraft control stick by 61 men and 61 women (from AFAMRL-TR-81-39) are summarized in table LXVI. Figure 171 illustrates the effect of arm position and wheel angle on maximum applied force to the left and to the right for the 5th percentile male.

TABLE LXVI. Maximum forces exerted on aircraft control stick (lbs) by 61 men and 61 women (AFAMRL-TR-81-39).

Control Stick Direction	Men			Women		
	Percentile			Percentile		
	5th	50th	95th	5th	50th	95th
Stick left	35	52	74	17	26	35
Stick right	22	35	43	14	19	18



a) Physical Layout for Wheel Force Tests (wheel 18 inches above seat reference point)
See Human Engineering Guide to Equipment Design



b) Maximum Force to the Left (lb)

c) Maximum Force to the Right (lb)

FIGURE 171. Effect of arm position and wheel angle on maximum applied force to the left and to the right for the 5th percentile male (Human Engineering Guide to Equipment Design).

Human Engineering Guide to Equipment Design discusses one- versus two-handed operation of controls and some general principles of control design; for example:

> "…Controls requiring large forces shall be operated with two hands (which, for most controls, about doubles the amount of force that can be applied), depending on control type and location and on the kind and direction of movement, as follows:
>
> a. When two hands are used on wheel controls, rotational forces are effectively doubled in most cases
> b. With two hands on stick or lever controls located along the body midline…push right or left is increased about 50%
> c. When two hands are used on stick or lever controls located on either side of the body midline, at or beyond the shoulder…, pull right on controls located to the left is slightly better with two hands than with only the right hand, and push right on controls located to the right is slightly better with two hands than with only the left hand…. For controls requiring single applications of force or short periods of continuous force, a reasonable maximum resistance is half of the operator's greatest strength. For controls operated continuously, or for long periods, resistances shall be much lower.
>
> The quantitative rudder pedal force requirements of this design guide are only on maximum forces, not on gradients. There are little data to define a quantitative requirement on yaw control force/position gradients; and this issue has seldom been critical."

VERIFICATION

Verification shall be via verification of requirements 5.2.3.8.1 through 5.2.3.8.6.

For a stick controller, the roll control force is that component of applied force exerted by the pilot in the direction to command roll, acting at the center of the stick grip in a direction perpendicular to a line between the center of the stick grip and the stick pivot. For a wheel controller, the roll control force is the total moment applied by the pilot about the wheel axis in the plane of the wheel, divided by the average radius from the wheel pivot to the pilot's grip.

The yaw control pedal force is the difference of push-force components of forces exerted by the pilot on the yaw control pedals, lying in planes parallel to the plane of symmetry, measured perpindicular to the pedals at the normal point of application of the pilot's instep on the respective yaw control pedals.

5.2.3.8.1 Lateral-directional breakout forces.
Breakout forces (the cockpit control force required to affect movement of the control surface in flight), including friction, preload, etc., shall be within the limits of table LXVII.

TABLE LXVII. Allowable lateral-directional breakout forces.

Type of controller	Level	Minimum (lb)	Maximum (lb)
Roll controller	1		
	2		
	3		
Yaw controller	1		
	2		
	3		

REQUIREMENT RATIONALE

The cockpit control should not respond to very small inadvertent pilot commands, yet proportional control movements must transmit pilot commands precisely. Quantitative limits assure that centering and breakout characteristics of the roll and yaw controller will allow effective control without adding unduly to the pilot's workload.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.5.2.1
MIL-STD-1797: 4.5.9.4 and 4.6.7.11.

Recommended values for table LXVII:

Type of Controller	Level	Classes I, II-C, and IV		Classes II-L and III	
		Minimum (lb)	Maximum (lbs)	Minimum (lb)	Maximum (lbs)
Centerstick	1 and 2	0.5	2	0.5	4
	3	0.5	4	0.5	8
Wheel	1 and 2	0.5	3	0.5	6
	3	0.5	6	0.5	12
Sidestick	1 and 2	0.5	1		
	3	0.5	4		
Pedal	1 and 2	1	7	1	14
	3	1	14	1	28

REQUIREMENT LESSONS LEARNED

The recommended values for table LXVII come from a combination of a long history (but unfortunately not well documented) of flight test and operational experience and variable-stability aircraft evaluations. The recommended sidestick breakout forces for table LXVII are based upon recommendations of AFFDL-TR-79-3126. Pilots have objected to higher yaw control breakout forces on some aircraft. Requirement 5.2.8.3.2 has additional, qualitative requirements on control centering and breakout forces.

VERIFICATION

Verification shall be performed on the actual flight article. Changes to the flight control system may make it necessary to check the production version as well as the flight test aircraft. Measurement of breakout forces on the ground will ordinarily suffice in lieu of actual flight measurement, provided that qualitative agreement between ground measurement and flight observation can be established. It is possible that vibration, temperature, pressurization, etc., effects may influence the results. If there is any question, flight measurement in the critical environment will be definitive.

5.2.3.8.2 Roll control sensitivity.
The roll response to roll control force sensitivity shall not be greater than the limits of table LXVIII. In case of conflict between the requirements of 5.2.3.8.2 and 5.2.3.8.3, the requirements of 5.2.3.8.2 shall govern. The term, "sensitivity" does not include breakout force.

TABLE LXVIII. Maximum roll control sensitivity for stick-controlled aircraft.

Level	Flight Phase Category	Maximum sensitivity (deg/sec/lb)
1	A	
	C	
2	A	
	C	

REQUIREMENT RATIONALE

For all aircraft, control sensitivity is an important flying qualities factor. Oversensitivity can caused pilot-in-the-loop oscillations. The gradient as well as the maximum force (5.2.3.8.3) is important. This gradient requirement will tend to be more critical for Class IV aircraft, in Category A Flight Phases.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.4.1.3
MIL-STD-1797: 4.5.9.3.

Recommended values for table LXVIII:

Level	Flight Phase Category	Maximum Sensitivity (deg/sec/lb)
1	A	Centerstick: 15.0 Sidestick or wheel: Not objectionable to the pilot
	C	Centerstick: 7.5 Sidestick or wheel: Not objectionable to the pilot
2	A	Centerstick: 25.0 Sidestick or wheel: Not objectionable to the pilot
	C	Centerstick: 12.5 Sidestick or wheel: Not objectionable to the pilot

REQUIREMENT LESSONS LEARNED

The roll power requirements of 5.2.3.5, in combination with the roll forces of 5.2.3.8.3, effectively specify roll control sensitivity except for aircraft with stick shaping networks such as discussed below. This paragraph is intended to place an absolute, firm upper limit on gradients to prevent excessive sensitivity, which can easily result in a roll PIO. Over-sensitivity has also been encountered on wheel-control aircraft.

Effects of nonlinear force/deflection characteristics shall be considered for this requirement, e.g.: it is possible to design a control system which easily meets the limits specified for large inputs, but which still produces unacceptably high local sensitivity for small inputs. Nonlinear stick shaping, stick filters, and minimizing the lateral augmentor gains can circumvent this problem. The following discussion relates to Class IV aircraft.

Roll Command Augmentation System (CAS)

The elements of a roll CAS are shown on figure 172. In a roll rate CAS, filtered pilot control inputs are compared directly to actual roll response. CAS authority can be limited, with parallel direct links (e.g., the F-14, F-15, F/A-18, and B-1); or high command gains can be incorporated with full authority (e.g., the F-16). Examples of both types will be reviewed here.

From FTC-TR-73-32, AFFTC-TR-75-15, AFFTC-TR-79-10, FTC-TD-72-1, AFFTC-SD-69-5, and AFFTC-TR-76-15, with a high-authority roll CAS, responses to command inputs are sharp and rapid; control precision can be excellent; in turbulence, hands-off operation is improved. However, some distinct problems have arisen as well: over-sensitivity to small control inputs, over-control with large inputs; pilot-in-the-loop oscillations; and the phenomenon known as roll ratcheting. Causes of and cures for these shortcomings will be discussed.

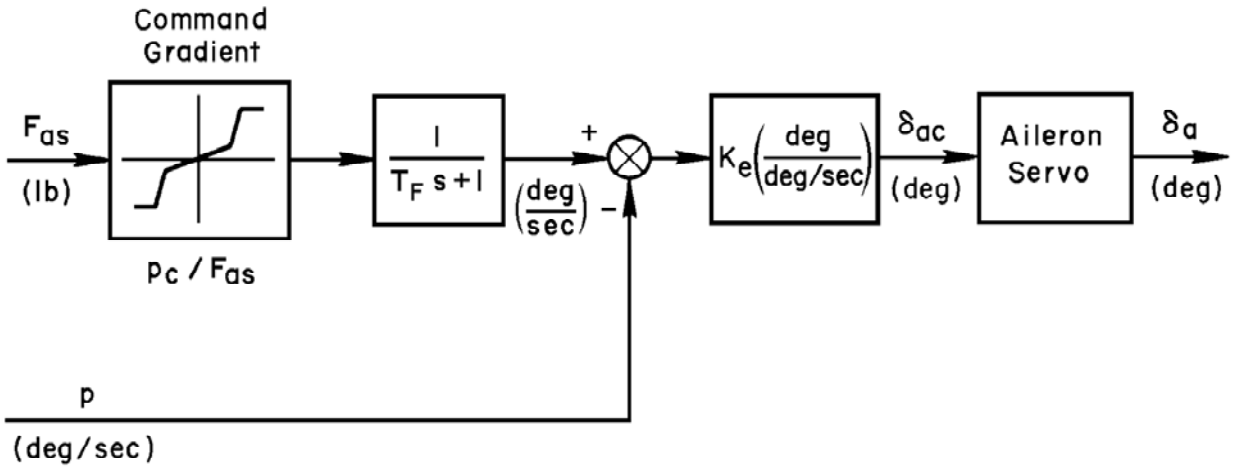


FIGURE 172. Block diagram representation of full-authority roll rate command augmentation systems.

a. Gradient Shaping

A key element is the gradient between commanded roll rate and stick forces, p_c/F_{as} (figure 172). High-gain, high-authority systems have had problems with extreme sensitivity for small inputs and inadequate roll performance with large inputs (AGARD-CP-319); saturation and small T_R contribute. A cure has been to decrease p_c/F_{as} for small inputs via a nonlinear stick-shaping network, while allowing a high gradient for larger inputs. The resulting parabolic p_c/F_{as} shaping appears as shown on figure 173. Experience with a limited number of CASs has shown that command networks which fall within the range shown on figure 173 will have acceptable response properties, as long as the roll time constant and roll performance requirements and recommendations are satisfied.

Figure 173 also reflects the range of actual maximum roll rates (achieved) for these command networks. In general, for small inputs (on the order of ½ stick travel or less) the commanded roll rates were obtained, but larger force inputs generally did not produce the larger rates commanded; that is, the command was not held in long enough. Possible explanations are actuator rate limiting (which would increase the effective T_R), a relatively small desired bank-angle change, and the pilot's inability to stop precisely at the desired bank angle.

In almost all high-performance or high-altitude augmented aircraft a yaw damper minimizes undesirable yawing motions while rolling. These effects on handling qualities and performance will be considered separately.

b. Roll Responses for Conventional Aircraft

Fighter aircraft with conventional, fully-powered hydraulic servos but without CAS (for instance, 1950s-generation fighters) generally have linear stick-to-surface linkages, i.e., δ_a response to F_{as} is linear (above breakout). However, wind tunnel and flight tests of these aircraft show that aileron effectiveness is nonlinear with deflection; large deflections produce an incrementally larger rolling moment than do small deflections. The result is a parabolic force/response curve. As an example, figure 174 shows the p_{max}/F_{as} curves for three aircraft, taken from flight or wind tunnel/flight test results. Parabolic p_c/F_{as} networks, therefore, artificially supply to the pilot what some aircraft without CAS have naturally.

c. Roll CAS Gradients

Evolution of the F-16 CAS shaping network is a valuable lesson (figure 175). The unique characteristics of the near-isometric sidestick controller undoubtedly played a significant role. Simulation of the YF-16 (AFFTC-TR-75-15) prior to first flight produced a very steep p_c/F_{as} gradient with maximum stick forces of 4 lbs. In-flight simulation of YF-16 takeoffs and landings in the USAF/Calspan variable-stability NT-33 resulted in a decrease in the initial gradient by a factor of two. During a high-speed taxi test, a divergent lateral PIO was encountered on the prototype YF-16 after the aircraft inadvertently became airborne (figure 176). The pilot, committed to fly, was then able to backoff on his control gains; the PIO stopped and a normal landing was made.

This PIO was directly traceable to excessive stick sensitivity around zero. After the flight (dubbed "Flight 0"), the stick sensitivity was reduced further and the PIO tendency disappeared. The final F-16A/B (fixed-stick) (AFFTC-TR-79-10) gradient was reduced even more. With the latest F-16 variable roll prefilter (AFFTC-TR-80-29), it has been possible to increase the CAS gradient somewhat.

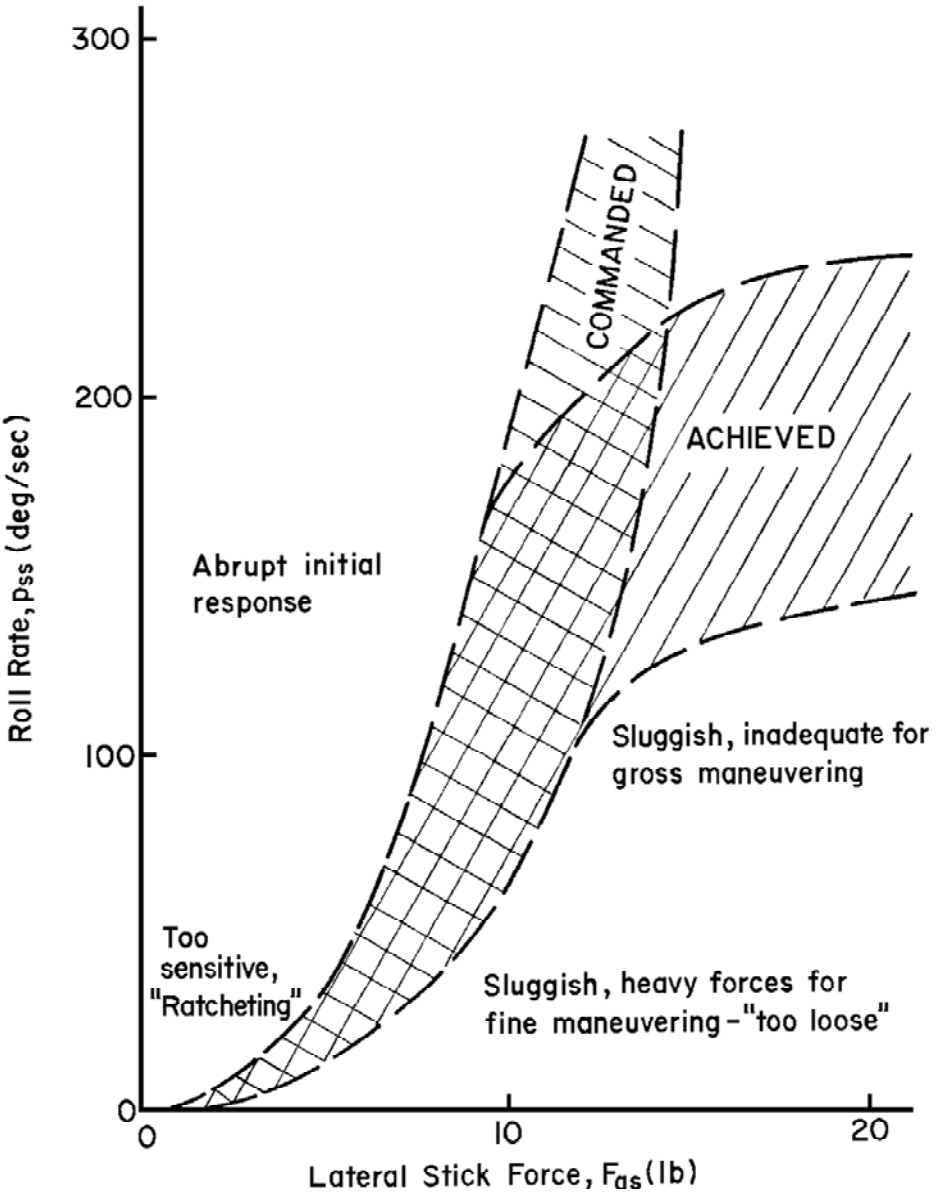


FIGURE 173. **Range of acceptable nonlinear roll command shaping networks based on flight test (Class IV aircraft, Flight Phase Category A, right roll).**

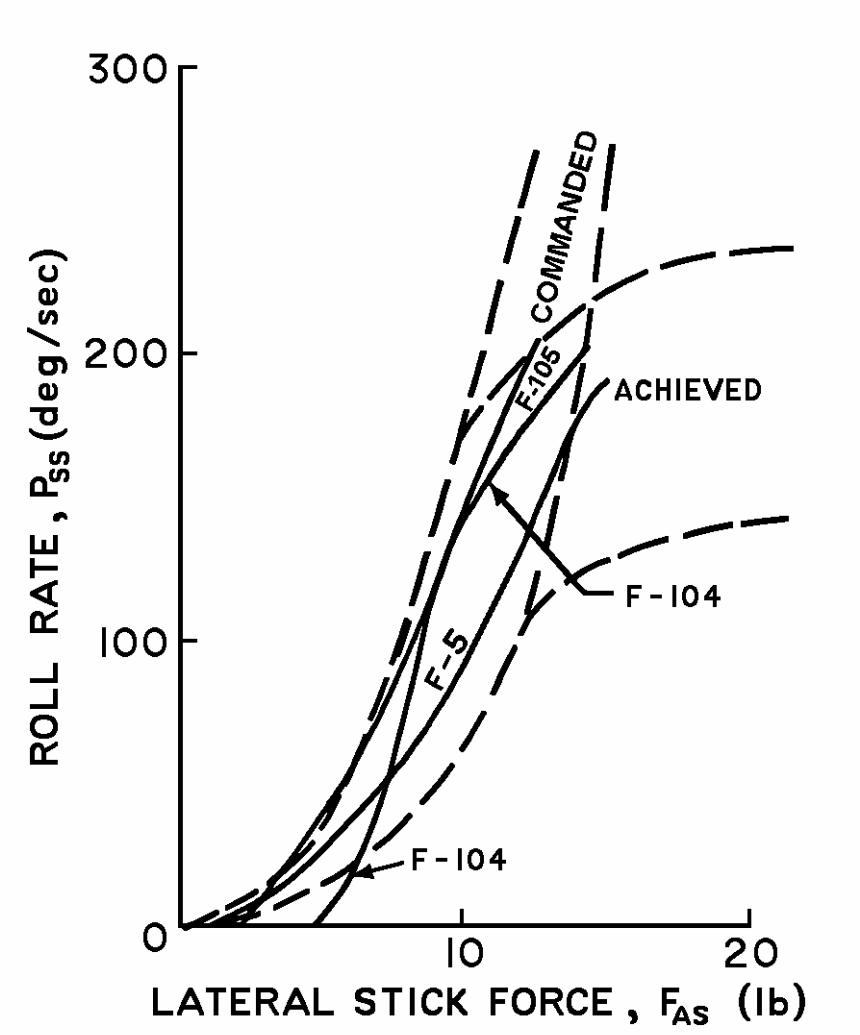


FIGURE 174. **Comparison of p_{max}/F_{as} for several conventional Class IV aircraft with CAS curves of figure 173.**

F-16 roll performance is discussed under 5.2.3.8.6. The final YF-16 command gradient of AFFTC-TR-75-15 (figure 175) produced acceptable response for small, precision stick inputs, though pilot comments indicate that excessive sensitivity "when encountered, was usually related to the small-amplitude, high-frequency inputs associated with the closed-loop, high-gain tasks of formation, refueling, tracking, and landing." Hence the nonlinear stick shaping and roll damping were reasonably successful in achieving acceptable maximum rolling performance without excessively compromising the small-amplitude precision tracking characteristics.

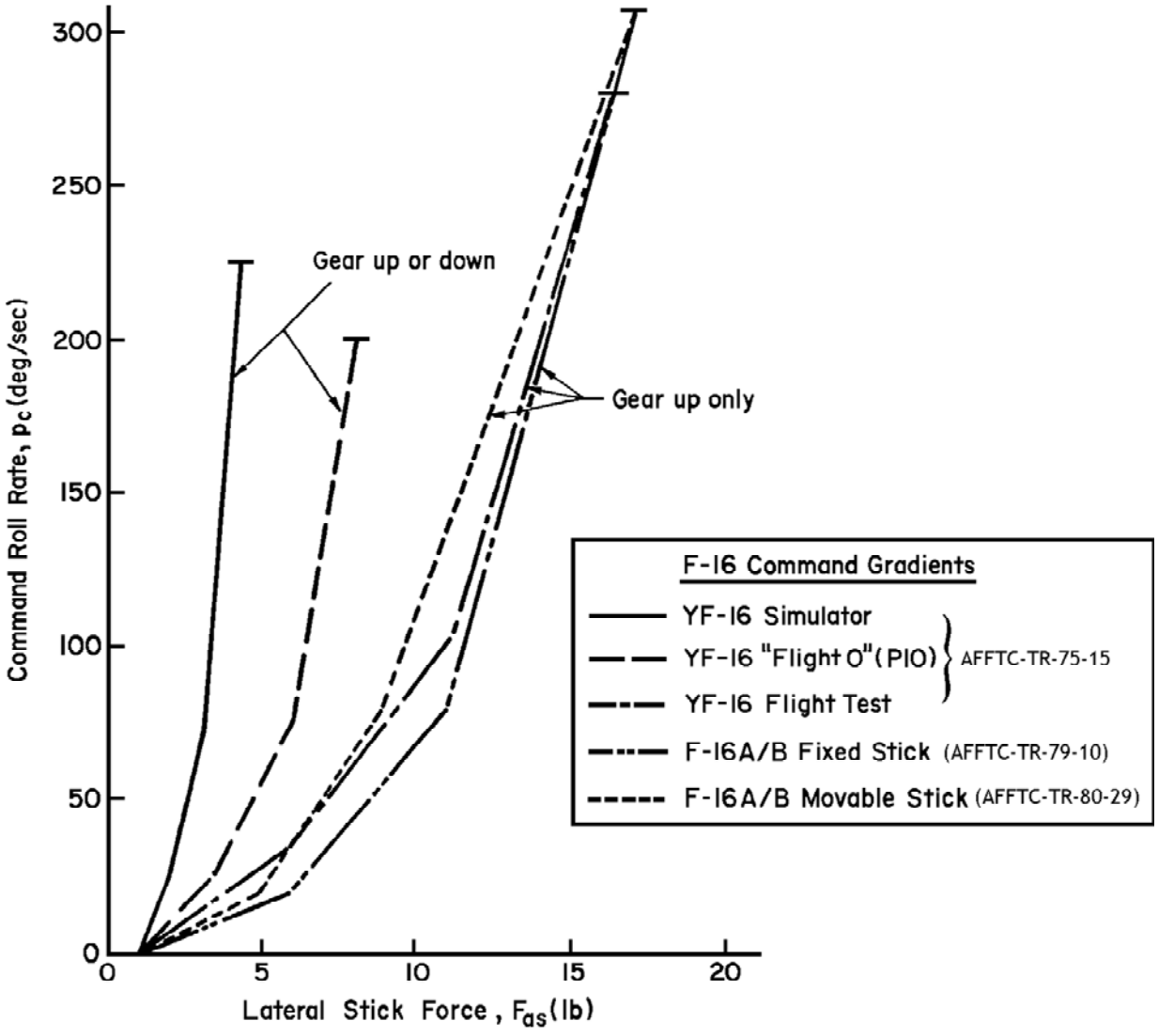


FIGURE 175. Evolution of the F-16 CAS shaping network.

d. Roll ratcheting

Roll ratcheting has been reported on most CAS-equipped aircraft, including the F-4 SFCS (FTC-TR-73-32), YF-16 (AFFTC-TR-75-15), F-16 (AFFTC-TR-79-10), and YA-7D DIGITAC (AFFTC-TR-76-15). It was also experienced during the Calspan Lateral High-Order System (LATHOS) program of AFWAL-TR-81-3171. All of these cases will be discussed in detail in the following paragraphs.

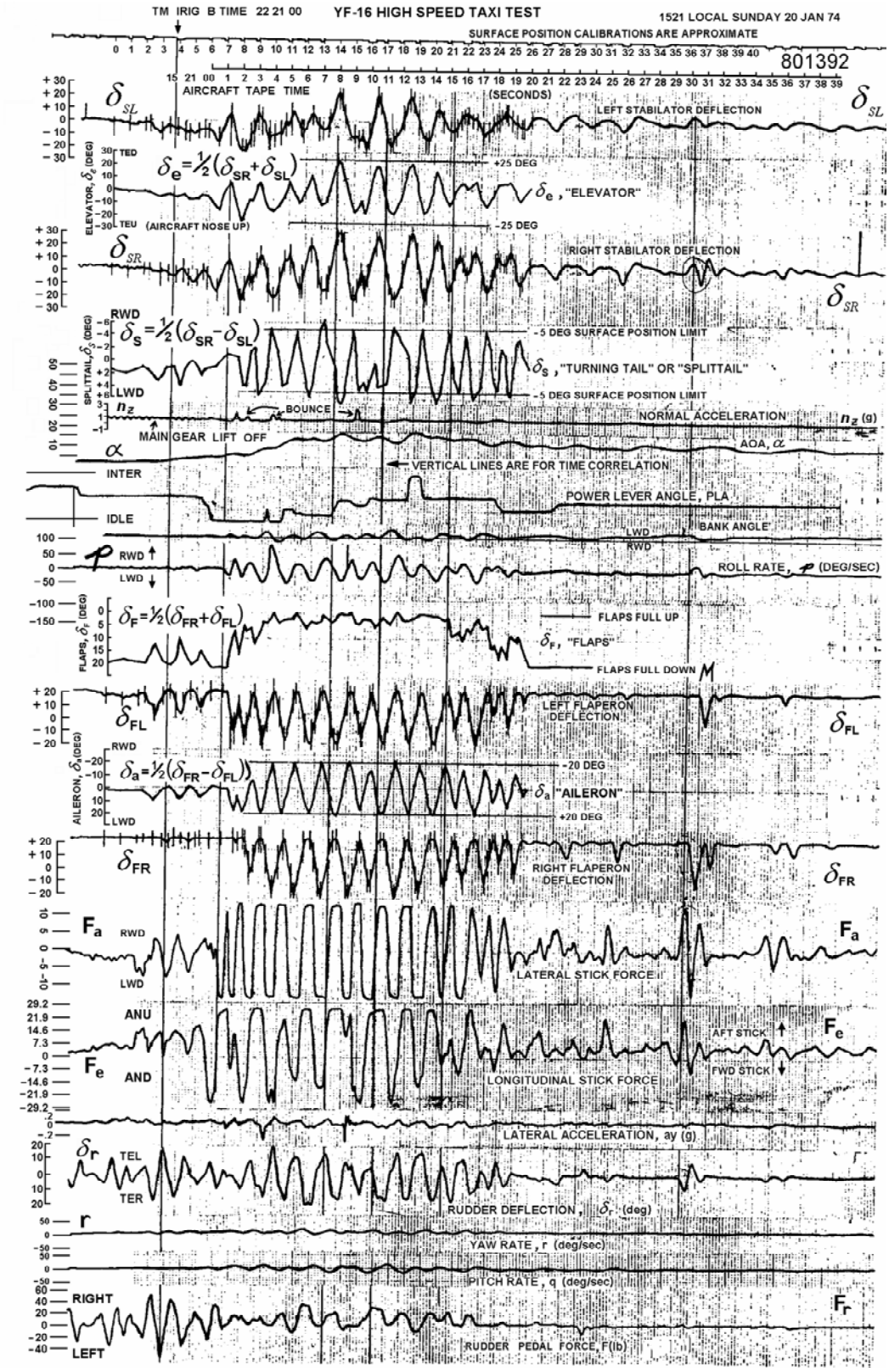


FIGURE 176. YF-16 PIO due to excessive lateral stick sensitivity (from AFFTC-TR-75-15).

An example of DIGITAC roll ratcheting (AFFTC-TR-76-15) shown on figure 177 was encountered during a series of bank-to-bank maneuvers: a limit cycle at a frequency of about 18 rad/sec. The roll CAS is presented on figure 178; p_c/F_{as} was Curve 1 on figure 178, indicating that stick shaping is not a cure for this problem: the stick sensitivity is reduced only around zero, allowing ratcheting to occur when the lateral stick force is non-zero, as on figure 177. Figure 178 and table LXIX document several of the CAS networks flown on the DIGITAC in developing an optimum CAS. This is an excellent review, since several gradients, prefilter lags, and error gains were evaluated.

TABLE LXIX. Descriptions of YA-7D digital CAS networks.

CAS*	Error Gain K_e* (deg/deg/sec)	Prefilter Lag $1/T_F$ (rad/sec)	Typical Pilot Comments
1	1.0	10	Much too sensitive to sharp inputs (PR = 7); ratcheting (see figure 177.)
2	0.5	10	Eliminated high sensitivity; steady-state response too low
3	1.0	3	Filter reduced sharp inputs, although not enough (PR = 7)
4	0.375	3	Lateral PIO tendency in fine tracking (includes 0.75-lb breakout)
5	0.375	3	Best all-around response

* See figure 178.

As table LXIX reflects, CAS 2 with only a reduction by one-half in K_e eliminated the roll sensitivity. However, with the sensitivity reduced, the pilots then noted that the steady-state roll response was much too low (Halving K_e only reduced the steady-state response about 14%). With $K_e = 1.0$ and the prefilter lag time constant increased from 1/10 to 1/3 sec (CAS 3), the roll sensitivity was reduced, although not enough. It was clear CASs 1-3 that: a) the roll response for large inputs was too low; b) an increase in the prefilter lag helped reduce sharp inputs; c) a reduction in the error gain eliminated ratcheting. Therefore, CAS 4 was evaluated. This involved a new p_c/F_{as} gradient (figure 178), including a 0.75-lb breakout and lower T_F and K_e (table LXIX). It also produced a mild PIO tendency during air-to-air tracking, probably due to the breakout. Finally, a slightly more sensitive gradient with no breakout (Curve 5, figure 178) was found to be best for all-around response.

Another example of roll ratcheting, experienced on the YF-16 (AFFTC-TR-75-15), is given on figure 179. The pilot was attempting a steady roll with less than full control input. The ratcheting is seen to be a lightly damped oscillation at a frequency of about 12 rad/sec. But in the second (later) roll on figure 179, the pilot was able to perform a roll without encountering ratcheting: “full authority rolls did not involve the oscillation.”

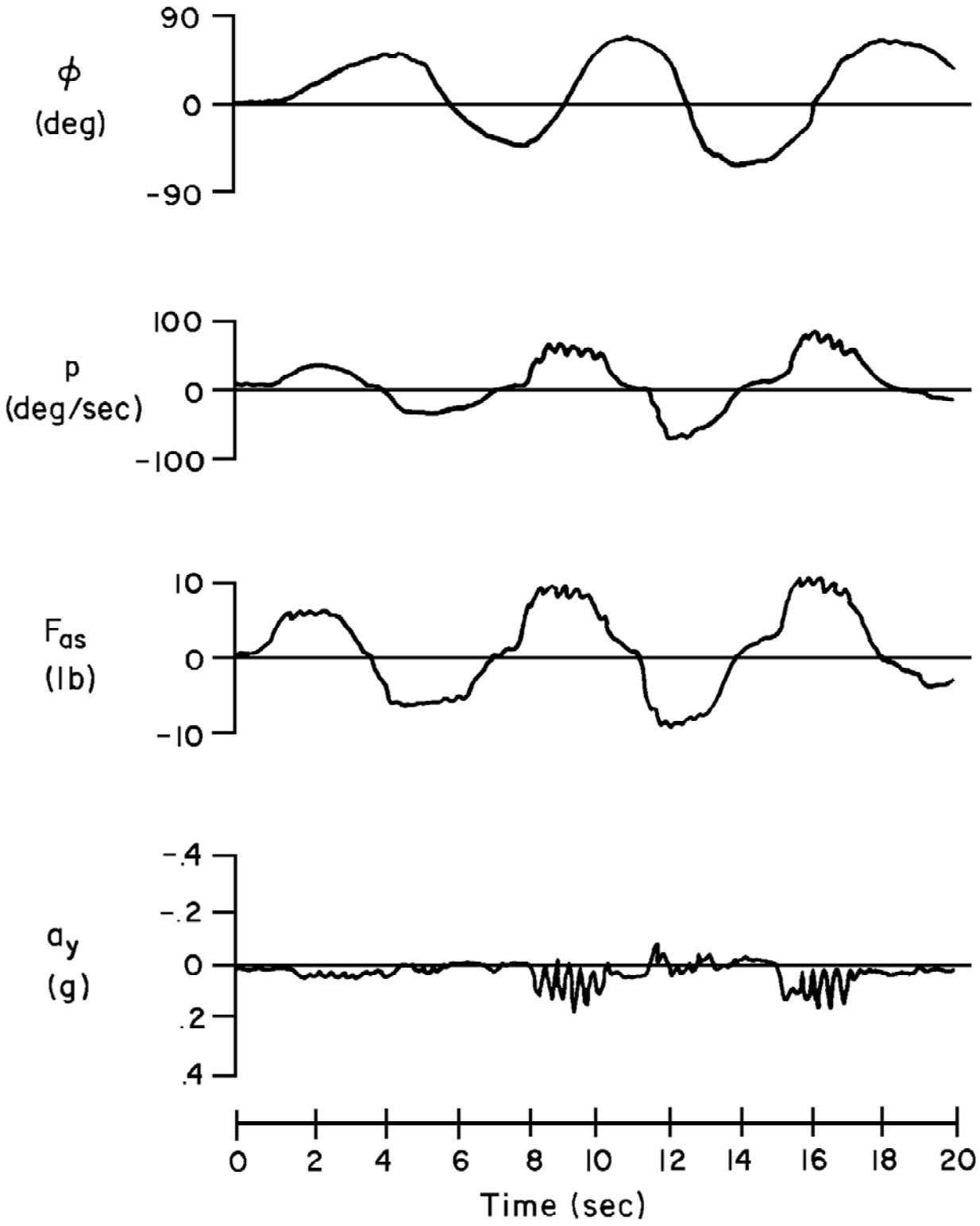


FIGURE 177. **Roll ratchet during banking maneuvers (DIGITAC, AFFTC-TR-76-15), h = 20,000 ft, M = 0.75.**

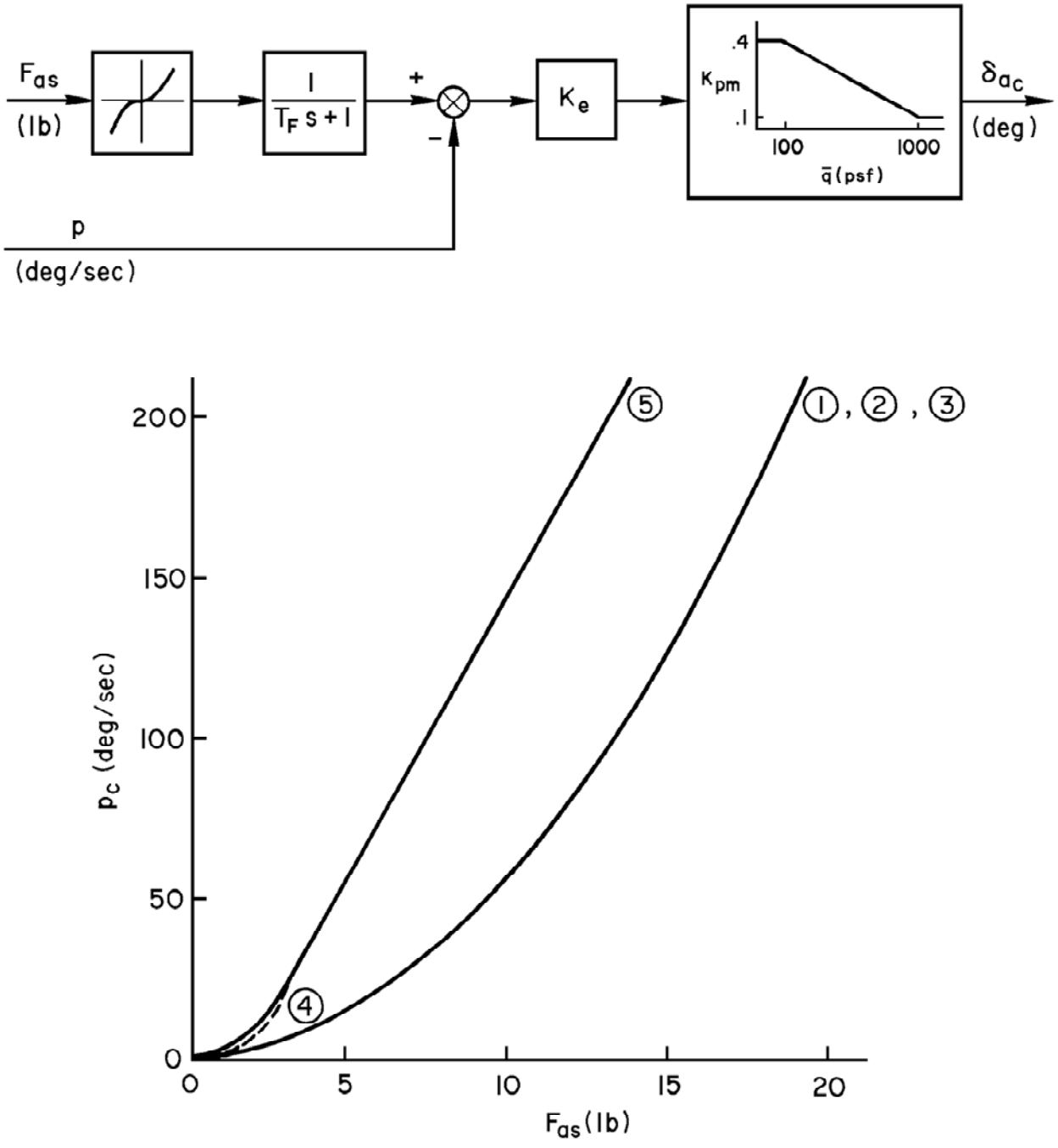
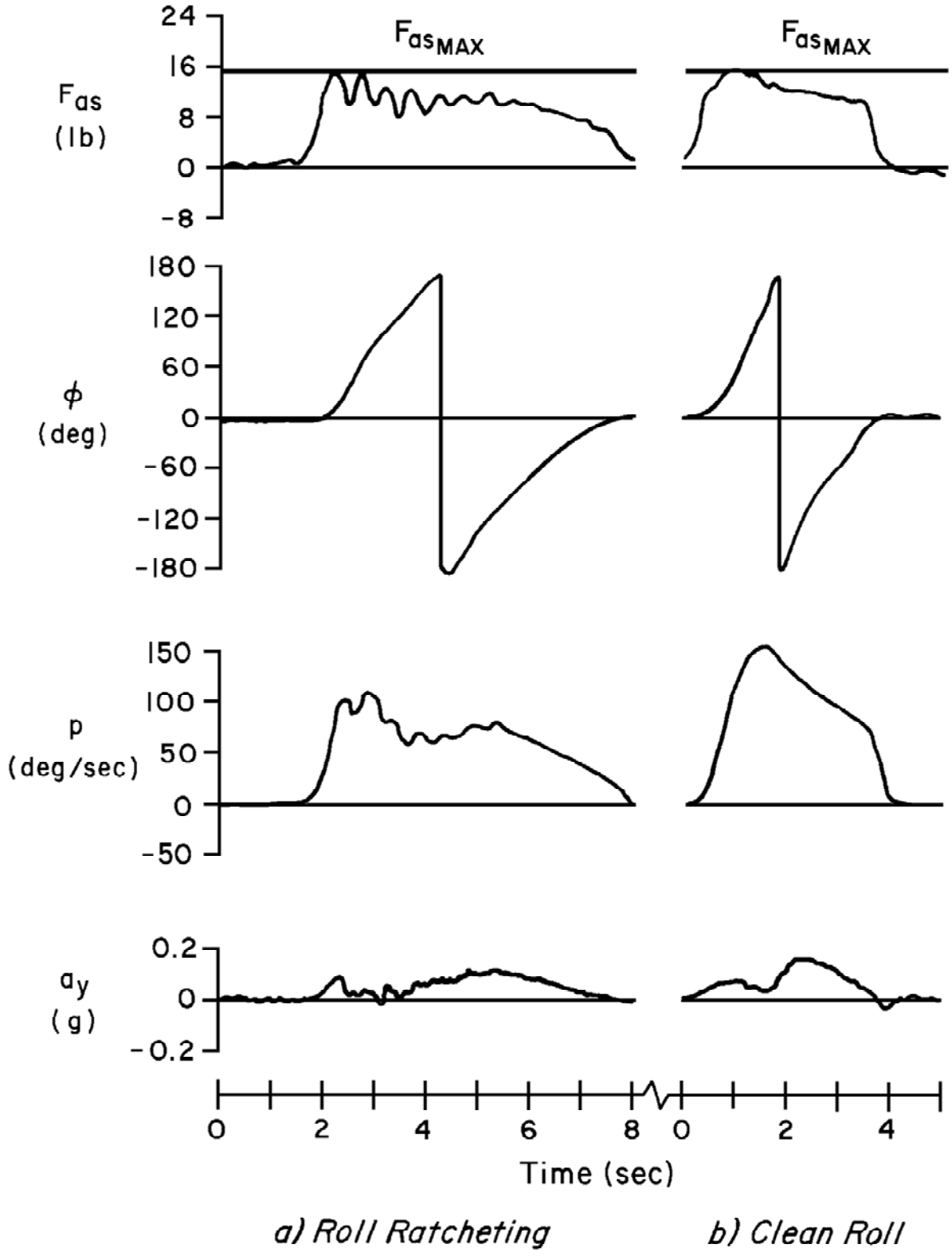


FIGURE 178. Evolution of roll CAS network for YA-7D DIGITAC (AFFTC-TR-76-15).



*The roll in (b) was performed 32 sec after (a) and was satisfactory.

FIGURE 179. Steady rolls on YF-16 (AFFTC-TR-75-15), h = 10,000 ft, M = 0.80.

The roll ratcheting experienced on the YF-4E SFCS (FTC-TR-73-32) was of a somewhat different character, as it occurred primarily during fine maneuvering rather than during large-input rolls. A representative time history is not available, but FTC-TR-73-32 describes "an oversensitive roll response which was universally objectionable to the pilots. It tended to manifest itself in uncomfortably high roll accelerations during rolling maneuvering and roll 'ratcheting' or jerkiness around neutral, particularly during tasks involving precise control." One pilot commented that the ratcheting "becomes less noticeable during up-and-away flight. However, this problem is definitely noticeable while performing a close task such as formation or air-to-air tracking."

The final example occurred during flight evaluations on the USAF/Calspan variable-stability NT-33. An investigation of lateral flying qualities of highly-augmented fighter aircraft (dubbed LATHOS for Lateral High-Order System, AFWAL-TR-81-3171) represents an excellent database for detailed discussion on many of the handling quality concerns for modern aircraft.

Mechanization of the lateral control was such that it may be considered a CAS. That is, the NT-33 variable-stability system was set up to command a certain ratio of steady-state roll rate to stick force (p_{ss}/F_{as}) with a certain time constant (T_R). Additionally, the spiral and Dutch roll modes were suppressed for roll commands ($T_S \doteq \infty$ and $\omega_\phi / \omega_d \doteq 1$). Thus, with a variable prefilter and a time delay to account for actuator lags,

$$\frac{p}{F_{as}} \doteq \frac{T_F\, L'_{F_{as}}\, e^{-0.28s}}{(s + 1/T_F)\,(s + 1/T_R)}$$

Ratcheting was noted some times but not others, for T_R values of 0.45 sec or less, more often at the higher sensitivities. Figure 180 illustrates the ratcheting from a HUD tracking task. (The very high frequency oscillations at 50-60 rad/sec are aileron buzz, resulting from an instability in the NT-33 variable-stability system with certain high command or feedback gains. In some instances the pilots complained about the buzz. This buzz was noted, when present, in the pilot comment summaries; where it may have influenced pilot ratings, that evaluation was discounted.) A ratcheting is best seen in the p and F_{as} traces, at a frequency of about 16 rad/sec.

Figure 181 compares the p_{ss}/F_{as} gradients flown on LATHOS in Category A tasks (air-to-air tracking, HUD tracking, and aerial refueling) with the acceptable range from figure 173. Zero breakout or friction forces were mechanized. Several values of prefilter lag, T_F, were used in Configurations 5-2 and 5-3. Figure 182 shows the influence of T_F on pilot ratings.

For Configuration 5-2 (p_{ss}/F_{as} = 10), the roll response for small inputs lies well above the acceptable range on figure 181, while the response for large inputs falls below the range of acceptable gradients. The pilot comments for Configuration 5-2 are consistent with this observation. Typical comments were: "Took off pretty smartly initially, but felt heavy for final response…Not predictable for fine tasks…Quick, sharp, ratcheting." Pilot ratings for this case were Level 3 (PR = 7, 6, 7). These ratings and comments were for T_F = 0.025 sec. However, increasing T_F did little to improve the ratings (see figure 182) because of the inadequate response. Pilot comments reflect this: "Gross acquisition sluggish…Sensitivity low…Took a lot of force."

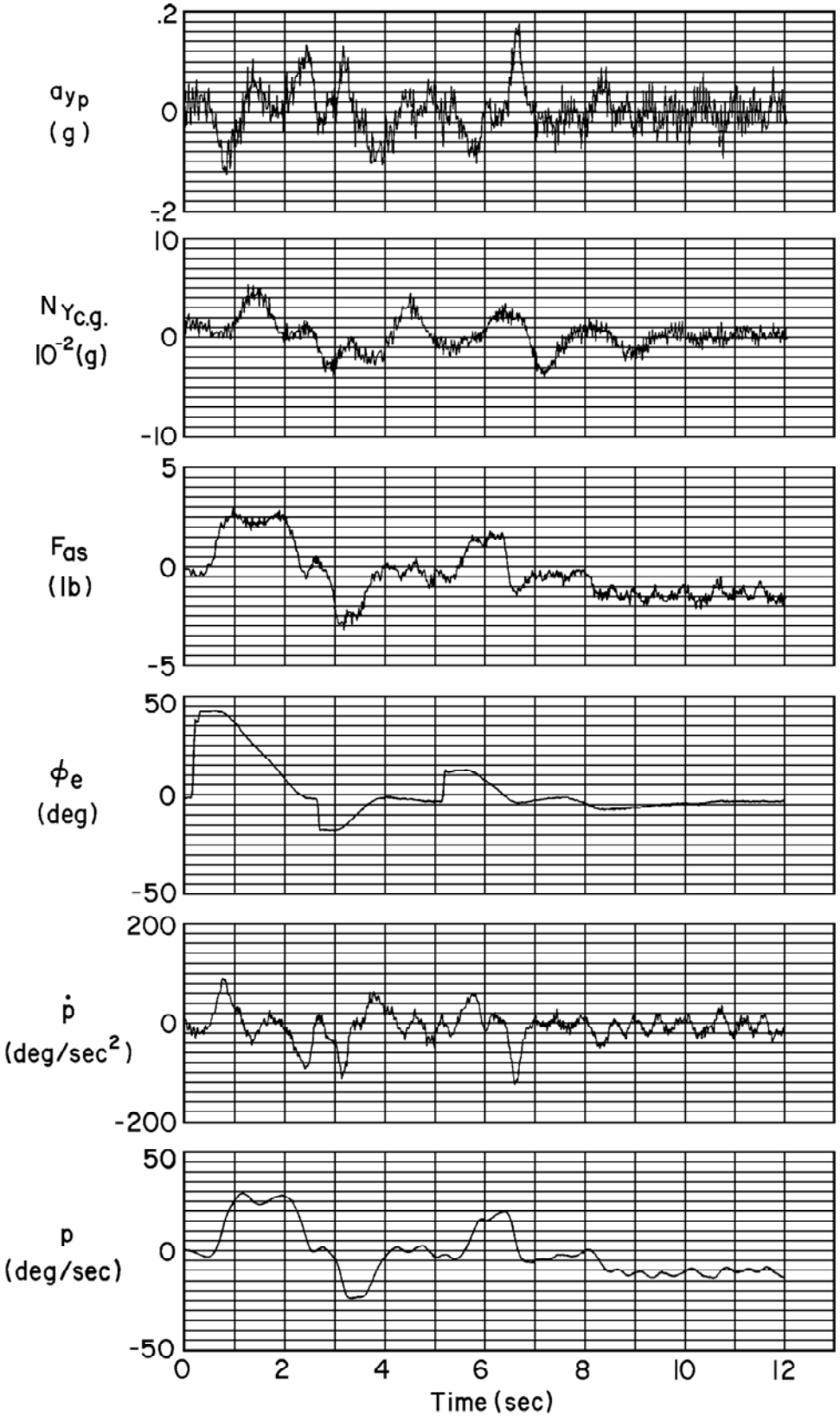


FIGURE 180. Roll ratcheting experienced on LATHOS (AFWAL-TR-81-3171) Configuration 5-2.

For Configuration 5-3 (p_{ss}/F_{as} = 18, figure 181) the final response is improved, but the small-control-input response is much too sensitive. Pilot comments for the 40 rad/sec filter case (T_F = 0.025, figure 182) reflect this: "Gross acquisition – no problem. Fine tracking was characterized by jerkiness…Had the perception that the stick was moving in my hand." Prefilters of 3.3-10 rad/sec (T_F = 0.3 and 0.10, figure 182) produced Level 1 pilot ratings, a trend like that found on DIGITAC. With T_F = 1.0, however, a PR of 7 was given; this was "Smooth but sluggish…Wouldn't respond to aggressive inputs."

It is noteworthy that the LATHOS configurations prone to roll ratcheting were: all of those with $L_{F_{as}} > 50$ (deg/sec^2)/lb, plus one observation of "slight ratcheting" for $L_{F_{as}} = 40$ and one "beginning" at $L_{F_{as}} = 30$, $T_R = 0.2$. Ratcheting occurred with T_R as long as 0.45 sec. (As can be seen on figure 102, the minimum $L_{F_{as}}$ evaluated increased as T_R decreased.)

Finally, two nonlinear gradients (5-3N2 and 5-3N3, figure 181) had the effect of reducing the sensitivity for small inputs while still providing good power for large inputs. For 5-3N2, a pilot rating of 4-1/2 was given due to "beginning of ratcheting – not strong…Jerky even with small inputs." For 5-3N3, a PR of 4 was similarly given because "Initial response [was] too abrupt….Adequate final roll rate for large inputs."

The LATHOS results are very similar to those for DIGITAC; i.e., ratcheting was reduced by addition of a roll prefilter around 3 rad/sec. However, as discussed in 5.2.2.1.1 and 5.2.2.1.2.2, stick prefilters are a major contributor to overall equivalent time delay, τ_e, and effective time delay, t_1. For example, figure 25a shows that a 3 rad/sec prefilter contributes about 0.1 sec to the overall time delay in the longitudinal axis. For sophisticated aircraft control systems, with structural filters, sensor filters, etc. included, a prefilter as low as 3 rad/sec could cause an unacceptably large delay. The prefilter thus should not be looked on as a final solution.

Yet another source of over-sensitivity is noted in AGARD-CP-319, as illustrated on figure 183. A current design trend is to use high-gain forward-loop stabilization for robust, fast, stable response, with a prefilter to shape the response to pilot commands as desired. The high gain tends to saturate the actuators, etc., far short of maximum command. When the sensitivity is adjusted for large commands, then, in Navy experience, the system can be grossly over-sensitive for small inputs.

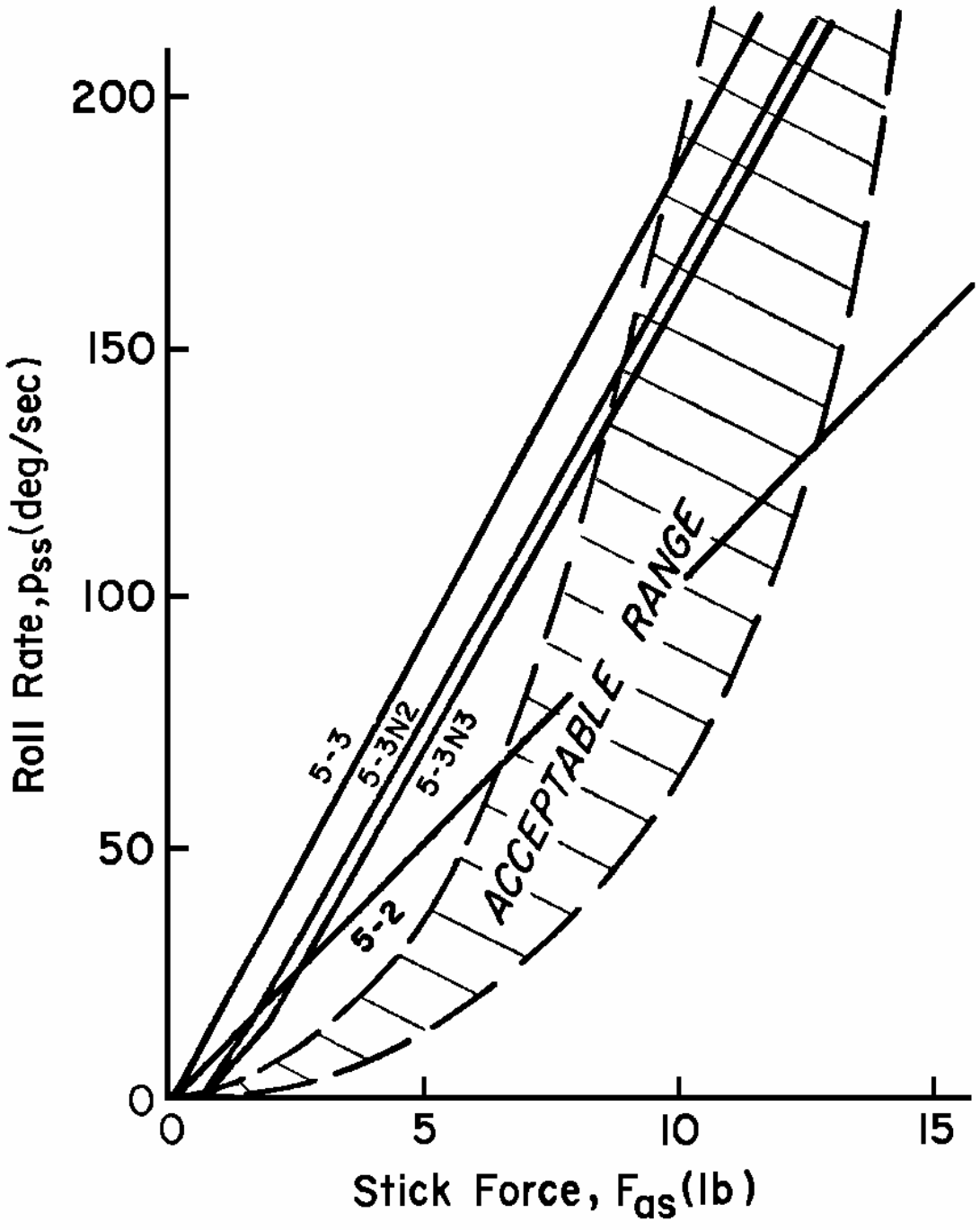


FIGURE 181. **Roll gradients for LATHOS Configurations 5-2 and 5-3 (T_R = 0.15 sec) compared with acceptable range from figure 173.**

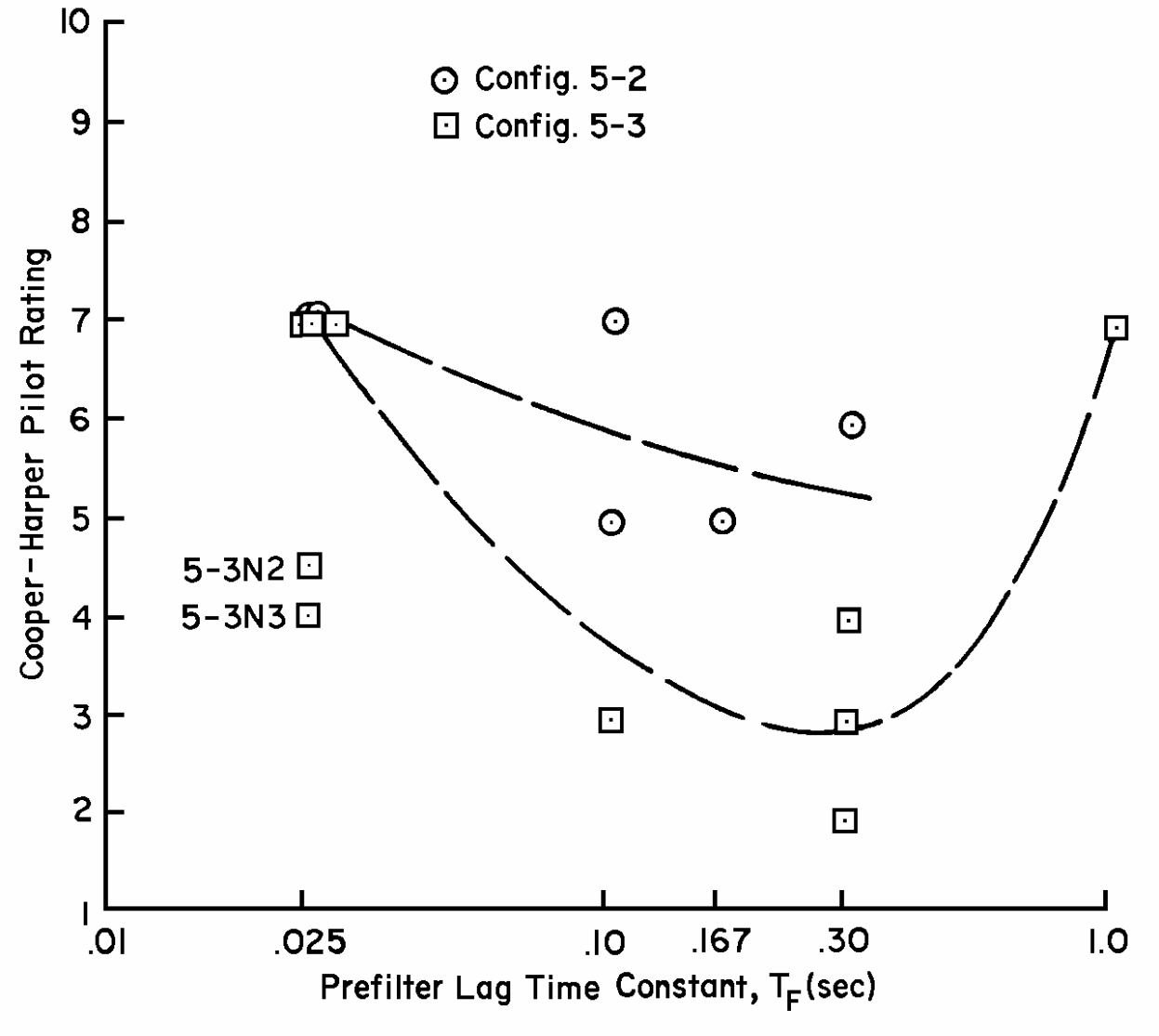


FIGURE 182. **Influence of prefilter lag on pilot ratings, T_R = 0.15 sec (AFWAL-TR-81-3171).**

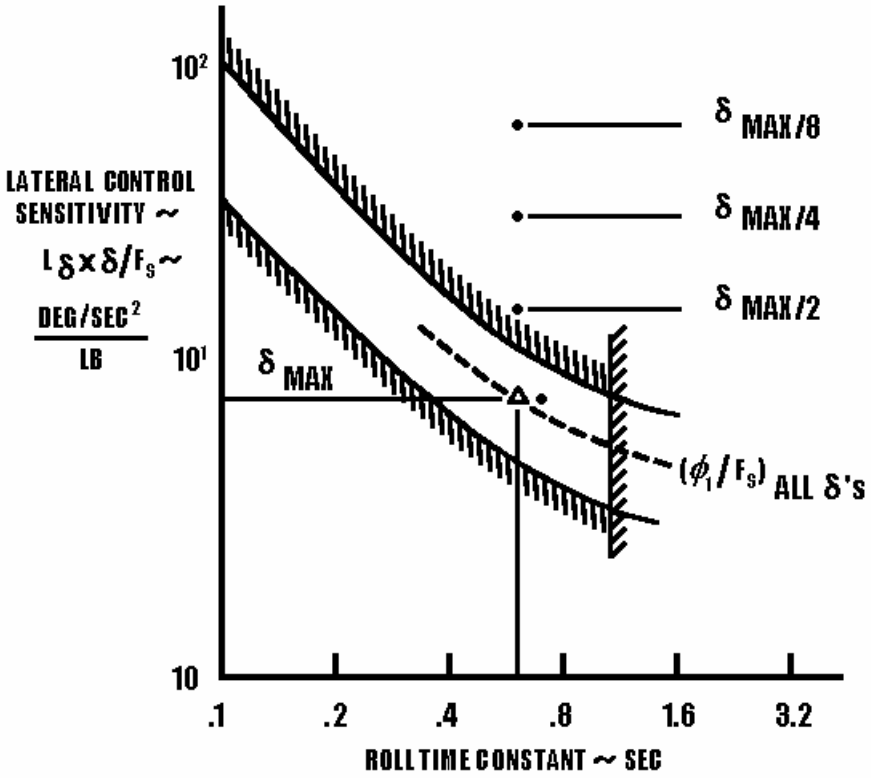


FIGURE 183. Typical roll sensitivity—high gain.

e. Guidance for acceptable sensitivity

The following guidelines are offered to obtain adequate roll control power for large control inputs:

Avoid excessive time delay, which is a source of lateral PIO tendencies. Stick filters will eliminate roll ratcheting. However, the break frequency should be carefully evaluated in terms of time delay (see figure 117).

Utilize nonlinear stick shaping in the region specified on figure 173.

Avoid control saturation and excessively large values of $1/T_R$ by minimizing the gain on the roll rate feedback. Figure 102 suggests $1/T_R < 4$.

As a rule of thumb, use no more than half the optimum sensitivity determined in a ground-based simulation (AGARD-CP-333).

Program of AFFDL-TR-67-98 (Flight Phase Category A)

In this in-flight lateral-directional flying qualities program for a typical fighter mission the pilots were allowed to select the sensitivity of the aileron control. (The spring rate was F_{as}/δ_{as} = 3.81 lbs/in.) Only the resulting optimum sensitivities are shown on figure 184 because most configurations were Level 2 or worse in Dutch roll or roll mode characteristics. Many of the points are associated with poor pilot ratings. Hence there is a tacit assumption that the optimum roll sensitivity is the same for Level 1 and 2 values of ω_d and ζ_d.

Program of NASA Memo 1-29-59A (Flight Phase Category A)

In this program, which utilized a rolling simulator and several fighter aircraft, a parametric variation of L_{δ_a} $\delta_{a_{max}}$ and T_R was made to determine lateral control requirements for fighter aircraft performing fighter missions. The results, then, should be directly comparable to those of AFFDL-TR-67-98. The NASA Memo 1-29-59A data are plotted along with the AFFDL-TR-67-98 data on figure 184. Optimum values and the values corresponding to Level 1 and Level 2 flying qualities are also shown. The spring rate, F_{as}/δ_{as}, was 2 lbs/in.

It can be seen from figure 184 that the data points of constant pilot rating lie approximately along lines of constant ϕ_1 (bank angle change in 1 sec). This suggests that, at least for Class IV aircraft which perform fighter missions, roll response sensitivity can be best expressed in terms of ϕ_1/F_{as} (bank angle change in 1 sec/lb). It can further be seen that the data points from NASA Memo 1-29-59A for Level 1 roll response sensitivity lie along a curve of ϕ_1/F_{as} = 15º/lb; for Level 2 flying qualities ϕ_1/F_{as} = 25º/lb. Both sets of data indicate that for optimum roll response sensitivity ϕ_1/F_{as} shall be between 10 and 20º/lb. A possible exception is indicated by the low-T_R data of AFFDL-TR-67-98, where the pilots selected somewhat lower optimum roll response sensitivities.

Figure 185 shows actual data from the tests of NASA Memo 1-29-59A. The pilot ratings from the moving-base simulation (figure 185b) clearly support the gradient limits of table LXVIII. Differences between the fixed and rolling simulator results are presumable due to the additional accelerations the pilots sensed in the rolling simulator. These results indicate that evaluations of roll sensitivity should involve a moving-base simulator as a minimum.

Program of Princeton Univ Rpt 727 (Flight Phase Category C)

In order to compare the fighter aircraft data for up-and-away flight with data for the landing approach, consider the in-flight data of Princeton Univ Rpt 727 shown on figure 186. From comparisons of figure 184 and figure 186 it can be seen that the optimum roll response sensitivity, maximum satisfactory roll response sensitivity (Level 1), and maximum tolerable roll response sensitivity (Level 2), in terms of rolling acceleration per force for the landing approach, are about half those for the respective Levels of flying qualities for Flight Phase Category A. This proportion is reflected in table LXVIII.

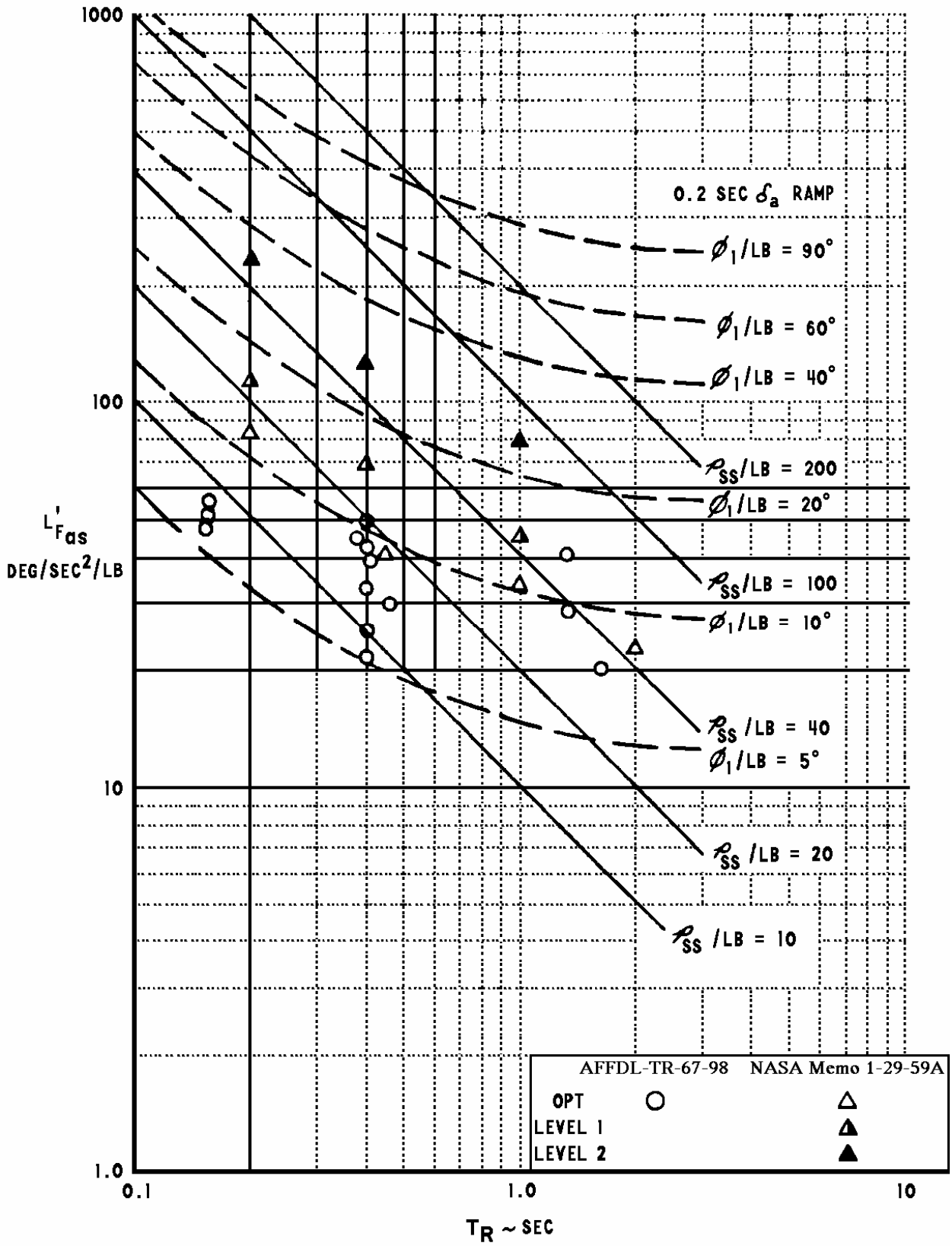


FIGURE 184. Flight Phase Category A—force sensitivity.

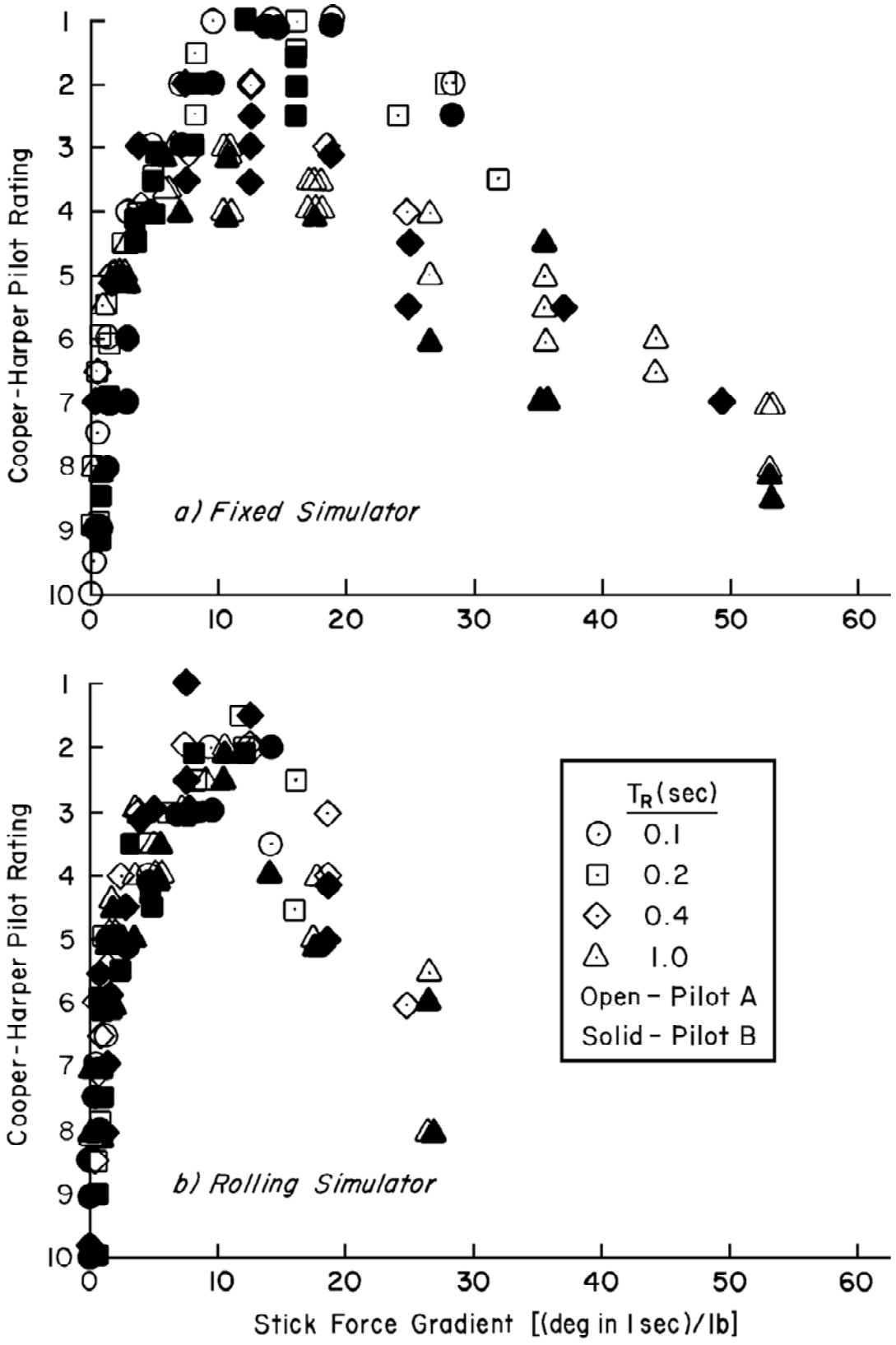


FIGURE 185. Pilot ratings from NASA Memo 1-29-59A (Category A Flight Phase).

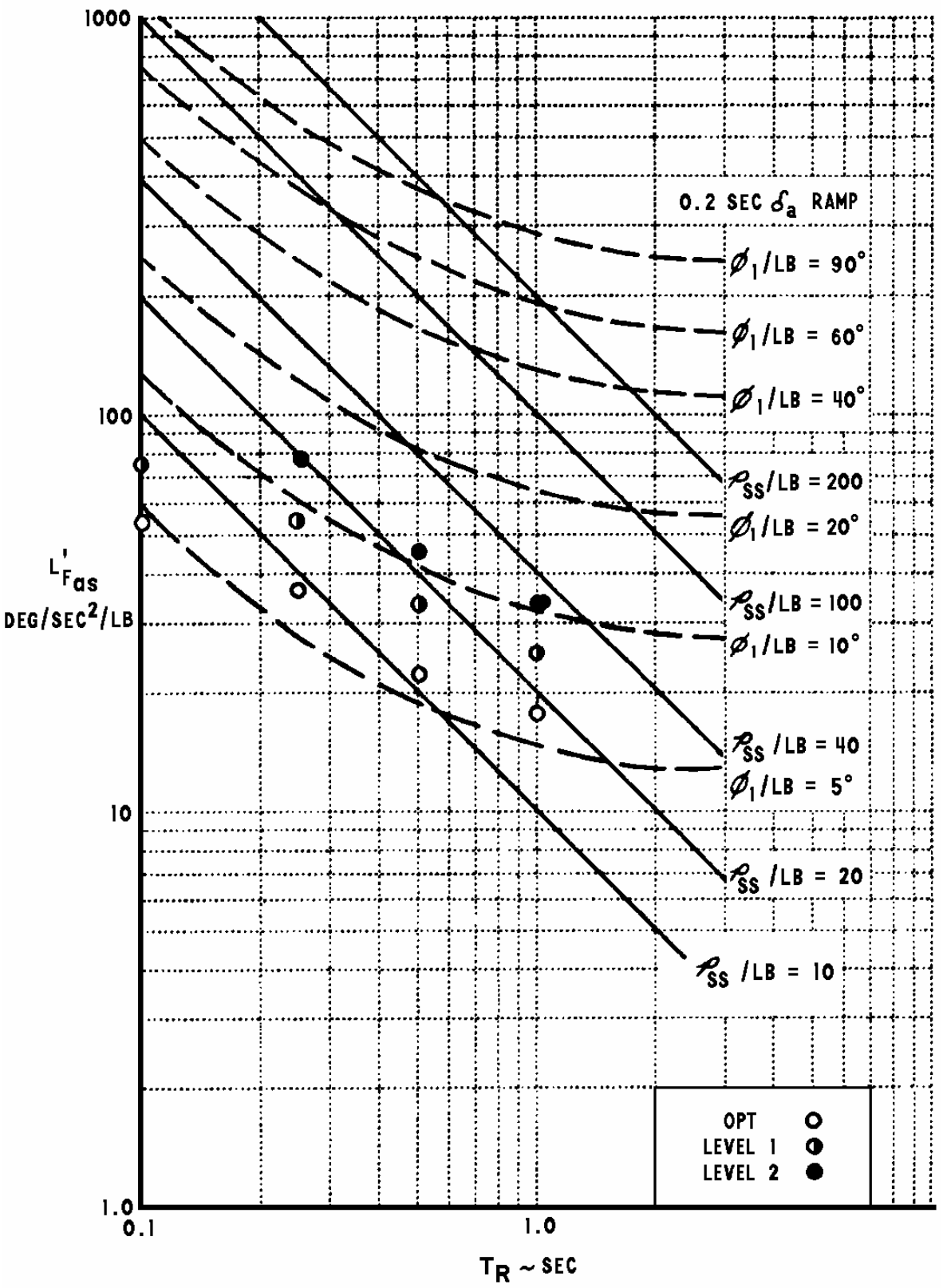


FIGURE 186. Flight Phase Category C—force sensitivity (from Princeton Univ Rpt 727).

The F-5 roll gradients in both Category A and C flight fall well within the Level 1 limits (AFFDL-TR-71-134). Similarly, flight test data from Navy Rpt No. SA-14R-81 show that the gradients for the original F/A-18A in Category A and C Flight Phases are within the Level 1 limits.

As discussed above, aircraft with high-gain, high-authority roll augmentation systems require a parabolic stick shaping network. Such a network makes it possible to maintain the required sensitivity for small stick deflections without giving up rolling performance for large stick deflections. The F/A-18 has a parabolic stick shaping network and the data quoted represent sensitivities for small stick deflections such as used for precision tracking.

Although over-sensitivity has also been encountered in sidestick and wheel-control aircraft, data are insufficient to recommend quantitative requirements.

VERIFICATION

Verification will be in connection with build-ups and demonstrations of roll performance (5.2.3.8.6), and evaluations in operational-type maneuvers involving precise tracking. Roll commands of all sizes need to be investigated for precise control tasks. Both minimum and maximum sensitivities are specified. The critical factors are enumerated and discussed in 5.2.3.8.2 Requirement Lessons Learned.

5.2.3.8.3 Roll control force limits in rolls.
The roll control force required to obtain the roll performance specified in 5.2.3.5 shall be neither greater than the maximum in table LXX nor less than the minimum in table LXXI. For two-handed operation, the force limits apply to the sum of right- and left-hand forces.

TABLE LXX. Maximum roll control force.

Level	Flight Phase	Maximum force (lbs)
1	A	
	B	
	C	
2	A	
	B	
	C	
3	A	
	B	
	C	

TABLE LXXI. Minimum roll control force.

Level	Minimum force (lbs)
1	
2	
3	

REQUIREMENT RATIONALE

In combination with the roll control power requirements of 5.2.3.5, this paragraph specifies control force gradients for good flying qualities. Pilot force capabilities must be observed and over-sensitivity avoided.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.4.3
MIL-STD-1797: 4.5.9.2.

Recommended values for table LXX:

Level	Class	Flight Phase Category	Maximum Force (lbs)		
			Centerstick	Wheel	Sidestick
1	I, II-C, and IV	A and B	20	40	Forces shall not be so large or so small as to be objectionable to the pilot.
		C	20	20	
	II-L and III	A and B	25	50	
		C	25	25	
2	I, II-C, and IV	A and B	30	60	
		C	20	20	
	II-L and III	A and B	30	60	
		C	30	30	
3	All	All	35	70	

Recommended values for table LXXI:

Level	Minimum Force (lbs)
1	Breakout force + 25% of the Level 1 values in table LXX
2	Breakout force + 12.5% of the Level 2 values in table LXX
3	Breakout force

REQUIREMENT LESSONS LEARNED

The maximum and minimum forces are unchanged from MIL-F-8785C.

VERIFICATION

Control forces shall be determined while verifying compliance with 5.2.3.5. Hinge moments, which vary with dynamic pressure, are an additional consideration in unpowered control systems. For aircraft with centerstick controllers, 5.2.3.8.2 shall be applied along with this requirement.

5.2.3.8.4 Yaw control forces in rolls.
In the rolling maneuvers described in 5.2.3.5, but with coordination allowed, directional control effectiveness shall be adequate to maintain zero sideslip with pedal forces not greater than the limits in table LXXII.

TABLE LXXII. Maximum yaw control force in rolls.

Level	Flight Phase Category	Maximum Pedal Force (lbs)
1	A	
	B	
	C	
2	A	
	B	
	C	
3	A	
	B	
	C	

REQUIREMENT RATIONALE

This requirement is aimed at insuring adequate yaw control effectiveness for coordination during rapid turn entries and exits as well as during steady rolls, without extreme rudder pedal forces. The coordinated, banked turn is a basic flight maneuver. This requirement is a companion to the requirement on coordination in turn entry and exit (5.2.3.2.7). It serves as a control effectiveness requirement as well—though usually not a critical one. For parametric expressions of the required rudder pedal deflections, see 5.2.3.2.7 Requirement Lessons Learned.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.2.5
MIL-STD-1797: 4.6.7.1.

Recommended values for table LXXII:

Aircraft Class	Level	Flight Phase Category	Maximum Pedal Force (lbs)
I, II, and III	All	All	100
IV	1	A	50
		B and C	100
	2 and 3	All	100

REQUIREMENT LESSONS LEARNED

VERIFICATION

Yaw pedal forces shall be recorded during testing for compliance with 5.2.3.5 in which rudder pedals are used to reduce adverse yaw. Analytically, sideslip can be suppressed and the required pedal force and deflection determined. During tests of roll performance and roll yaw coupling, any critical flight conditions for this yaw control requirement will become evident.

5.2.3.8.5 Control forces in steady turns.
For Levels 1 and 2, with the aircraft trimmed for wings-level straight flight, it shall be possible to maintain steady coordinated turns in either direction at __(1)__ with a pedal force not exceeding __(2)__. It shall be possible to maintain steady turns in either direction at the same bank angles with yaw controls free with a roll control force not exceeding __(3)__.

REQUIREMENT RATIONALE

These requirements limit the allowable control forces in steady turns. The object of the first requirement is to insure that high yaw pedal forces are not required to maintain steady coordinated turns. The maximum allowable pedal forces shall be relatively small for banked turns, where the forces may be sustained for some time. The second requirement limits the pilot effort required to perform steady uncoordinated turns. The object is to insure that only modest steady roll control forces are needed when rudder pedals are not used, and consequently that excessive rudder pedal use is not needed for coordination. Retrimming is undesirable. Not only does that add to pilot workload, but the reference for straight flight is then lost.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.2.6
MIL-STD-1797: 4.5.9.5.1 and 4.6.7.2.

Recommended value for Blank (1): Class I and II: 45º
Class III: 30º
Class IV: 60º

Recommended value for Blank (2): 40 lbs

Recommended value for Blank (3): Centerstick controller: 5 lbs
Wheel controller: 10 lbs
Sidestick controller:

REQUIREMENT LESSONS LEARNED

These requirements apply to Levels 1 and 2 only, since it is expected that Level 3 operations would not involve prolonged turns at large bank angles. For Blank (1), the recommended steepness of the turn is a function of Aircraft Class, to correspond with normal operational use.

Good design practice would dictate considerably lower rudder pedal forces than the upper limit recommended for Blank (2). Zero lbs would be ideal.

VERIFICATION

Roll and yaw control forces shall be recorded at any flight conditions where high forces are noted in the course of the flight test program. Flight testing at the specified bank angle (and

corresponding normal load factor), at maximum operational altitude and minimum operational velocity, generally presents the greatest potential for large yawing moments and large effective dihedral. Measurement is straightforward in either flight or simulator. Slipstream, engine gyroscopic, or other asymmetries may make forces different to the right and to the left.

5.2.3.8.6 Roll control displacement limits.
The control throw necessary to meet the roll performance requirements specified in 5.2.3.5 shall not exceed __(1)__ in either direction.

REQUIREMENT RATIONALE

The allowable control displacement must be limited to allow proper control in both pitch and roll and to bound the time and effort needed to reach sufficient travel. An upper limit on allowable control throw helps assure that undue contortions will not be required to fly the aircraft. If the throw is too small, of course, the aircraft can be overly sensitive to small inputs. However, from a comfort (and safety) standpoint, maximum throw is more crucial.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.4.5
MIL-STD-1797: 4.5.9.1.

Recommended value for Blank (1): Wheel controller: 60º *
Centerstick controller: Not objectionable to the pilot
Sidestick controller: Not objectionable to the pilot

*This requirement may be relaxed to 110º for wheel controllers with completely mechanical systems.

REQUIREMENT LESSONS LEARNED

Pilots have been critical of the larger wheel throws which were prevalent in the past, before the advent of powered controls. The small wheel control throw of 60º is attainable in normal, one-handed operation without undue physical effort. A wheel throw of 110º for completely mechanical systems has been specified in deference to the design problem posed by such systems. Stops are normally provided at the column to limit wheel travel.

As data from NASA CR-635 indicate (figure 187), to maintain a desirable roll response sensitivity in terms of roll performance per degree of wheel deflection, the smaller the wheel throw, the lower the required roll performance. (This is valid providing roll effectiveness is equal to or greater than the specified roll effectiveness requirements.)

NATO Rpt 508A makes recommendations concerning the amount of wheel throw for one-handed operation. Although the comments pertain to VTOL vehicles, the recommendation may

well be of general applicability. The NATO Rpt 508A recommendation is that for one-handed operation the wheel throw shall not exceed 60º in each direction.

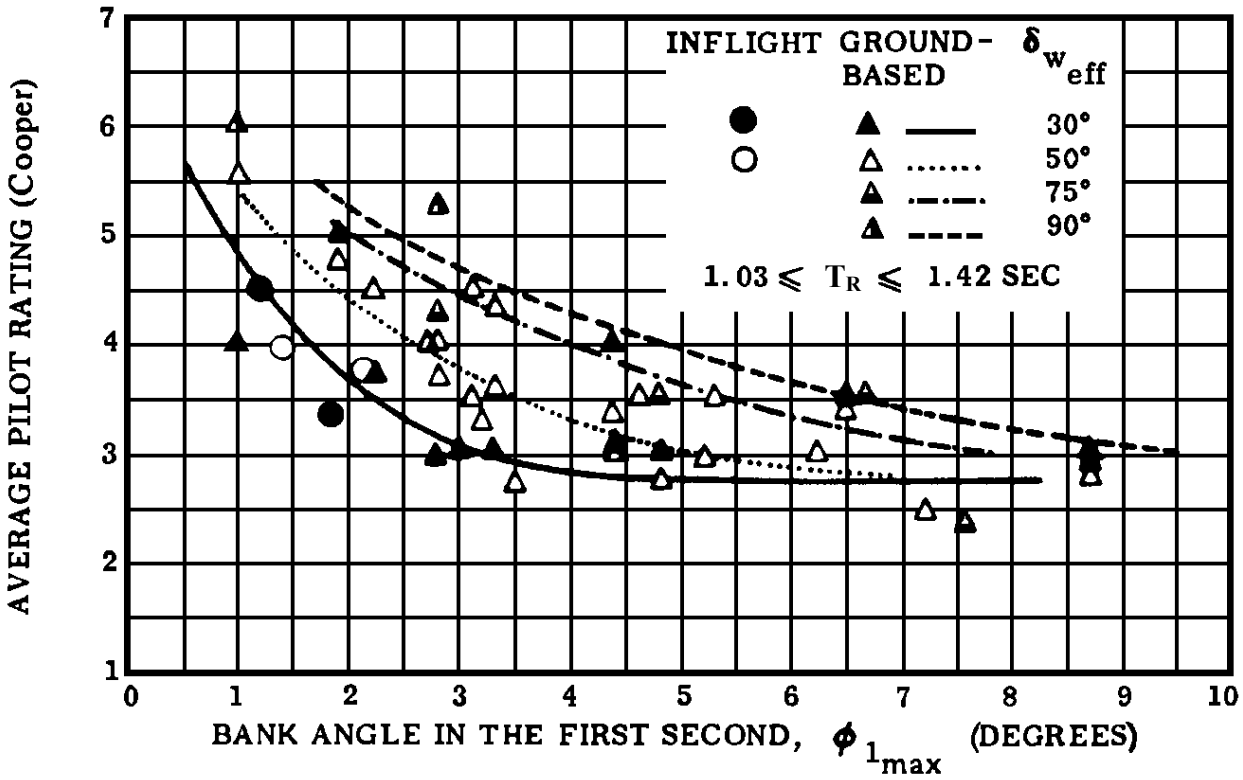


FIGURE 187. Variation of pilot rating with bank angle in the first second for four values of effective angle (from NASA CR-635).

VERIFICATION

Normally full wheel or stick throw will meet this requirement. If the roll performance requirements can be met with less than full wheel or stick throw, flight tests will show the throw actually required.

5.2.3.9 Steady sideslips.
Requirements 5.2.3.9.1 through 5.2.3.9.4 apply to yaw-and-roll control induced steady, upright, zero-yaw-rate sideslips with the aircraft trimmed for wings-level straight flight, at sideslip angles up to those produced or limited by:

a. Full yaw-control pedal deflection, or

b. 250 lbs of yaw control pedal force, or

c. Maximum roll control or surface deflection, or

d. __(1)__.

REQUIREMENT RATIONALE

These requirements are intended to provide static directional stability for reasonable sideslip angles.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.6
MIL-STD-1797: 4.6.1.2.

Recommended value for Blank (1):
For single-propeller-driven aircraft: "During waveoff (go-around), yaw-control-pedal deflection in the direction opposite to that required for wings-level straight flight need not be considered beyond the deflection for a 10º change in sideslip from the wings-level straight flight condition."
For all other aircraft: Blank (1) is not applicable.

REQUIREMENT LESSONS LEARNED

VERIFICATION

5.2.3.9.1 Yaw control force and deflection in steady sideslips.
Right yaw-control-pedal force and deflection shall produce left sideslips and left yaw-control-pedal force and deflection shall produce right sideslips. For Levels 1 and 2, the variation of sideslip angle with yaw-control-pedal deflection shall be essentially linear for sideslip angles between __(1)__, and for larger sideslip angles, an increase in yaw-control-pedal deflection shall always be required for an increase in sideslip. For Levels 1 and 2, the variation of sideslip angle with yaw-control-pedal force shall be essentially linear for sideslip angles between __(2)__, and for larger sideslip angles, an increase in yaw-control-pedal force shall always be required for an increase in sideslip.

REQUIREMENT RATIONALE

These requirements are intended to provide static directional stability for reasonable sideslip angles, and to prevent rudder lock at any possible rudder deflection. These requirements regulate against the classical rudder lock problem with unpowered controls, wherein pedal force reverses at large sideslip angles (see Perkins and Hage, "Airplane Performance Stability and Control"). Controls-free static directional stability for small sideslip angles is implied by the 5.2.3.1.2 requirement that ω_d be greater than zero—except that a favorable product of inertia effect can make up dynamically for a certain amount of static instability. However, generally, verification of the dynamic requirement is limited to small or moderate sideslip angles. Although meeting the static stability requirement does not necessarily assure dynamic stability, it does tend to promote it while at least insuring against an unnatural steady-state response.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.6.1
MIL-STD-1797: 4.5.5. and 4.6.1.2.

Recommended value for Blank (1): ±15º

Recommended value for Blank (2): ±10º

REQUIREMENT LESSONS LEARNED

The recommended requirements are very straightforward in interpretation and application. "Essentially linear", it should be noted, is qualitative rather than absolute. It is sensible to insure static stability over a given range, which may go beyond the range of sideslips in normal operation (but that still might reasonably be encountered occasionally), and that the response to yaw controls not be unnatural. The requirement for positive rudder deflection gradients with β provides a clue to the pilot of increasing sideslip in normal operation. Rudder lock in large sideslips, which creates difficulty in recovering, is prohibited. Rudder lock results from a reversal of rudder floating tendency (with unpowered controls) at large sideslip angles; a dorsal fin is a useful fix. See, e.g., "Airplane Performance Stability and Control."

Complimentary roll-axis requirements will be found in 5.2.3.9.2 and 5.2.3.9.3.

VERIFICATION

The most critical trimmed flight condition will be at the most aft center of gravity with any destabilizing external stores. Test conditions shall cover the ROTH speed range at the minimum, intermediate and maximum operational altitudes.

In order to totally remove the effects of aerodynamic damping from this static stability requirement, the requirement is stated for straight flight, with no yawing. Therefore, generally in the sideslips the wings will not be level. Thus there may be some small but probably insignificant difference from the wings-level sideslip specified by the FARs. Measurements shall cover rudder deflections from zero to maximum (which may be a function of flap deflection or landing gear position, dynamic pressure, etc.), and any differences between right and left sideslips need to be determined.

5.2.3.9.2 Bank angle in steady sideslips.
For the sideslips of 5.2.3.9, an increase or no change in right bank angle shall accompany an increase in right sideslip, and an increase or no change in left bank angle shall accompany an increase in left sideslip.

REQUIREMENT RATIONALE

This requirement insures that the bank angle in straight, steady sideslips is conventional, in accordance with pilot training and experience.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.6.2
MIL-STD-1797: 4.5.5.

REQUIREMENT LESSONS LEARNED

This requirement is intended to apply at positive normal load factor. The bank angle is the angle necessary to straighten the flight path while sideslipping. While there is some evidence that pilots do not object to zero bank in straight sideslips, weak or zero side-force characteristics lessen the usefulness of the ball-bank indicator for trimming and for coordinating turns, and seat-of-the-pants feel. Opposite bank angle seems disconcerting. Therefore, the requirement is merely that bank angle does not change adversely with sideslip. Normally this requirement on side-force is easily met.

VERIFICATION

A series of steady sideslips, as specified in 5.2.3.9, shall be performed at flight conditions over the operating envelope of the aircraft, measuring sideslip and bank angles, yaw and roll rates (to assure that they are zero), and yaw and roll controller and control surface deflections, and control forces.

5.2.3.9.3 Roll control force and deflection in steady sideslips.
For the sideslips of 5.2.3.9, right or zero roll control deflection and force shall accompany right sideslips, and left or zero roll control deflection and force shall accompany left sideslips. For Levels 1 and 2, the variation of roll control deflection and force with sideslip angle shall be essentially linear. This requirement may, if necessary, be excepted for waveoff (go-around) if task performance is not impaired and no more than __(1)__ of roll control power available to the pilot and no more than __(2)__ of roll control force are required in a direction opposite to that specified herein.

REQUIREMENT RATIONALE

This requirement insures that the roll control in straight, steady sideslips is conventional, in accordance with pilot training and experience.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.6.3 and 3.3.6.3.1
MIL-STD-1797: 4.5.5.

Recommended value for Blank (1): 50%

Recommended value for Blank (2): 10 lbs

REQUIREMENT LESSONS LEARNED

This requirement is intended to apply at positive normal load factor. The requirement involved principally the roll static stability and control hinge moments or feel. The relaxation for waveoff (go-around) has been found both necessary on occasion and tolerable. In this exception, allowable roll-control force is not made a function of the type of controller, since one-handed operation must be assumed for the waveoff or go-around maneuver. The phrase “available to the pilot” is used because control surface position can be commanded by both the pilot and the stability augmentation system.

This requirement normally (but not necessarily) will preclude negative effective dihedral. It is not desirable to use rudder-to-aileron crossfeeds to meet this requirement inasmuch as such

mechanizations augment $L_{\delta_{rp}}$ and not L_β as desired. Other requirements: 5.2.3.5.3, 5.2.3.6.1, 5.2.3.9.4, and 5.2.3.10.1 limit the amount of positive effective dihedral.

In reviewing this requirement (for MIL-F-8785B), consideration was given to putting some lower limit on dihedral effect since data such as those presented in Princeton Univ Rpt 727 (see figure 92) indicate that zero or low L_β is undesirable. Princeton Univ Rpt 727 indicates that the zero- L_β configurations were down-rated because the pilots were forced to use rudder pedals to damp the Dutch roll oscillations. Fighter pilots, in particular, desired some dihedral to enable them to damp the Dutch roll using ailerons alone. On the other hand, pilots evaluating a prototype assault transport (YC-15) with zero effective dihedral liked it, commenting on the uncoupled yaw response to rudder.

VERIFICATION

A series of steady sideslips, as specified in 5.2.3.9, shall be performed at flight conditions over the operating envelope of the aircraft, measuring sideslip and bank angles, yaw and roll rates (to assure that they are zero), and yaw and roll controller and control surface deflections, and control forces. While linearity with β is not always required, excursions to large sideslip shall serve to bring out any disconcerting behavior that might not be encountered otherwise in flight testing.

5.2.3.9.4 Roll control power in steady sideslips.
For Levels 1 and 2, positive effective dihedral (right roll control for right sideslip and left roll control for left sideslip) shall never be so great that more than __(1)__ of roll control power available to the pilot, and no more than __(2)__ of roll control force, are required for sideslip angles which might be experienced in service employment.

REQUIREMENT RATIONALE

These requirements assure adequate roll power for operation in sideslips, with some additional control margin to correct for turbulence, etc. They also insure that the amount of roll control force necessary to achieve a steady sideslip condition is never tiring for the pilot.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.6.3.2
MIL-STD-1797: 4.5.8.2 and 4.5.9.5.4.

Recommended value for Blank (1): 75%

Recommended value for Blank(2):	Centerstick controller:	10 lbs
	Wheel controller:	20 lbs
	Sidestick:	

REQUIREMENT LESSONS LEARNED

This requirement places a limit on dihedral effect in order to assure controllability. The 25 percent margin has generally proved sufficient, although it does not explicitly address gust sensitivity.

The pilot must be able to maneuver and to cope with disturbances, so a control margin is required. The type of controller relates directly to aircraft usage; that is, the size of sideslip that might be experienced in service employment, which varies with aircraft type as well as details of design missions. This requirement alone, however, is not enough to assure that excessive lateral stability will not make the vehicle overly sensitive to lateral gusts.

VERIFICATION

Analysis should reveal any large dihedral effect. If compliance with any of the sideslip requirements is doubtful or marginal, flight testing will need to be more extensive. Otherwise, analytical demonstration verified through qualitative flight testing shall be sufficient.

5.2.3.10 Lateral-directional control in crosswinds.
It shall be possible to takeoff and land with normal pilot skill and technique in 90º crosswinds, from either side, of velocities up to those specified in table LXXIII. Roll control forces shall not exceed the limits of table LXXIV, and yaw control pedal forces shall not exceed __(1)__ for Level 1 or __(2)__ for Levels 2 and 3.

TABLE LXXIII. Crosswind velocity.

Level	Crosswind
1	
2	
3	

TABLE LXXIV. Maximum roll control forces in crosswinds.

Level	Maximum roll control force (lbs)
1	
2	
3	

REQUIREMENT RATIONALE

These requirements assure good roll- and yaw-axis flying qualities in crosswind takeoffs and landings. They also assure that roll and yaw control forces in crosswind takeoffs and landings are acceptable, within pilot capabilities, and not overly burdensome. This is simply a method of insuring that crosswind operations do not require more roll control force than do normal rolling maneuvers.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.7
MIL-STD-1797: 4.5.6, 4.5.9.5.3, 4.6.4, and 4.6.7.4.

Recommended values for table LXXIII:

Level	Class	Crosswind
1 and 2	I	20 kts
	II, III, and IV	30 kts
	Water-based air vehicles	20 kts
3	All	50% of the values for Levels 1 and 2

Recommended values for table LXXIV:

Level	Class	Maximum Force (lbs)		
		Centerstick	Wheel	Sidestick
1	I, II-C, and IV	20	20	Forces shall not be so large or so small as to be objectionable to the pilot.
	II-L and III	25	25	
2	I, II-C, and IV	20	20	
	II-L and III	30	30	
3	All	35	70	

Recommended values for blanks (1) and (2) are:
Blank (1): 100 lbs
Blank (2): 180 lbs

REQUIREMENT LESSONS LEARNED

The crosswinds recommended above will affect not only these requirements, but also the asymmetric thrust requirements (5.2.3.11), which should be reviewed at the same time.

Appendix 2 of AFFDL-TR-69-72 presents a survey of wind data from which the values were derived. While the "requirement could be relaxed to 25 kts and still achieve at least 99.5 percent operational effectiveness," specific airfields such as the Azores are more troubled by crosswinds. Also the values given are averages over 1-minute intervals (at about 15 ft height), taken hourly over a 5 to 10 year period. "There are large time and space variations in wind speed, and terrain features exert a large influence."

Either roll or yaw control may be critical, depending upon the aircraft design. Roll control difficulty in crosswind operation may result from high lateral stability and is influenced by the crosswind technique adopted. Gusts can be expected to accompany strong crosswinds.

Both crabbing and sideslipping techniques have been used effectively for landing approach. In a steady crosswind of magnitude C at 90º to the runway, for takeoff and landing a pilot must align his flight path with respect to the air mass so that it has a component normal to the runway that exactly cancels his drift from the crosswind. Denoting the required angle to the runway by χ and the resultant airspeed by V_a, in the nearly level flight and attitude he must make

$$V_a \sin \chi + C = 0$$

or

$$\chi = \sin^{-1} (- C/V_a)$$

so that the ground speed, V (the inertial velocity) will be along the runway (figure 189). The pilot can do this entirely by crabbing, with zero sideslip; in that case $\psi = \chi$, the yaw deviation angle. When $\psi = 0$ the alignment must be entirely by sideslip: $\chi = \beta$ for small ϕ, with the aircraft pointed down the runway centerline but having a side (y-direction) velocity with respect to the air mass equal and opposite to C (assuming that $\cos \phi = 1$). The general case, with both crab and sideslip, is illustrated on figure 189. There, it is evident that

$$\chi = \psi + \beta$$

so that

$$\beta = \sin^{-1} (- C/V_a) - \psi \quad .$$

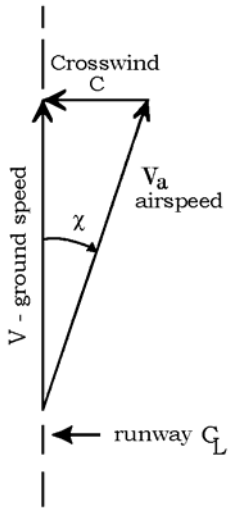


FIGURE 188. Approach geometry in crosswinds (runway heading = 0º).

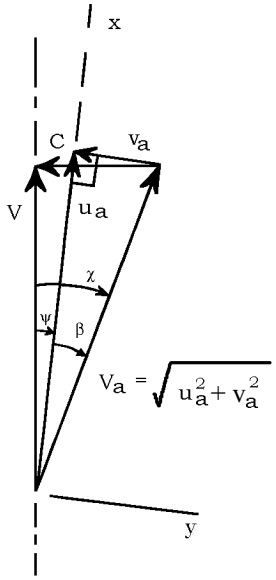


FIGURE 189. General case—both crab and sideslip (runway heading = 0º).

Figure 190 shows arbitrary orientations, when the inertial flight path (V_G) is aligned with the runway.

$$\frac{W}{\sin\varepsilon} = \frac{V_a}{\sin\sigma} = \frac{V_G}{\sin(\psi_W - \psi_G)}$$

$$\chi_a = \psi_G - \varepsilon = \psi_G - \sin^{-1}\left[\frac{W}{V_a}\sin(\psi_W - \psi_G)\right]$$

Using radian measure,

$$\chi_a \approx \psi_G - \frac{W}{V_a}\sin(\psi_W - \psi_G)$$

it can be shown that in stability axes ($\alpha = 0$),

$$\sin\beta = \sin(\chi_a - \psi)\cos\gamma / \cos\phi$$

$$\approx \sin(\chi_a - \psi)$$

for the usually small γ and ϕ. Then

$$\beta \approx (\psi_G - \psi) - \frac{W}{V_a}\sin(\psi_W - \psi_G)$$

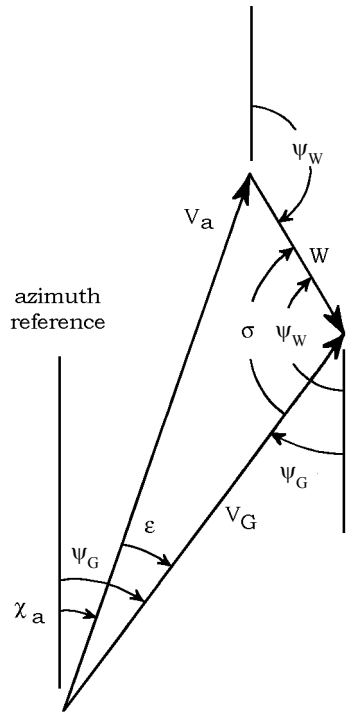


FIGURE 190. Arbitrary orientation, V_G aligned with runway.

Two extremes of crosswind compensation are wings-level crabbing and sideslipping with zero crab. In the crab, $\beta = 0$ and $\psi = \chi_a$. In the sideslip, to the left (negative) for the case shown—nose to the right of V_a:

$$\beta = -\varepsilon$$

That is,

$$\psi = \chi_a + \varepsilon$$

Heading is along the ground track and sideslip is

$$\beta = -(\psi_W - \psi_G)\ W/V_a$$

With sideslip, the aircraft heading with respect to the runway is small; with crabbing, a timely rudder kick (supplemented if necessary by aileron) must reduce the crab angle to a small value at touchdown. Landing in a level, crabbed attitude, the main-gear friction force gives a stabilizing, restoring moment for a nose-gear configuration.

The side-load capacity of the landing gear is normally intended to allow for the expected dispersion in crab angle at touchdown, rather than to allow keeping much crab intentionally. Crosswind landing gear of one sort or another is a design option, of course, but taxiing must not be made too difficult.

The SAE ARP 842B crosswind is 30 percent of $V_S(L)$ or 40 kts, half of that when there are two failures. One Air Force meteorologist has commented that 35 kts is about as much crosswind as is ever seen. On the ground, other factors in addition to aircraft aerodynamics enter: acceleration/deceleration, symmetric and differential braking, nose-wheel steering, thrust vectoring, and any drag parachute. The capabilities and limitations of all these factors play a part. Drag chutes can be particularly troublesome in a crosswind after touchdown. Keeping the nose high for aerodynamic braking keeps down the load on the main gear, thus limiting the effectiveness of differential braking.

From “Reengined KC-135 Shows Performance Gains in Test”:

> Changes in the nose-wheel steering system on the KC-135R also have been evaluated. Pilots said that combined rudder pedals and “tiller” steering are superior to the small hand wheel used on the KC-135A, particularly during crosswind landings and takeoffs. Takeoff procedures used in the older KC-135A required the pilot to steer the aircraft with his left hand via a small hand wheel, keeping his right hand on the control column yoke. The copilot operates the throttles until the aircraft reaches rudder effectiveness speed and the nosewheel steering control is no longer needed for directional control. At that point, the pilot switches his left hand to the control yoke and his right hand on the throttle quadrant.

> By connecting the nosewheel steering to the rudder pedals on the KC-135R, the pilot can control the aircraft's direction with his feet, simplifying normal and crosswind takeoffs. A tiller has replaced the old nosewheel steering control wheel and is used for maneuvering into and out of ramp parking spaces.

On the ground, directional stability is influenced by dihedral effect. The rolling moment generated by sideslip loads the downwind landing gear, tending to roll the aircraft about it. A wide main-gear track helps a lot.

The allowable maximum pedal forces for crosswind operations are somewhat high, but this is a result of the fact that such operations are generally of very short duration. The recommended limits are taken directly from MIL-F-8785C. However, they are felt to be quite high for continuous maneuvering and may be excessive for female pilots. It is recommended that lower limits be considered for new aircraft. AFAMRL-TR-81-39 shows that of 61 men and 61 women tested in a short-duration (4 sec) force test, almost all could exert 180 lbs to rudder pedals. However, continuous maneuvering would likely be difficult for male or female pilots.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall precede or accompany careful build-up to suspected critical flight conditions. For application as a control-authority design requirement, crosswind values shall be assumed to be invariant with altitude. However, some caution should be exercised in applying the recommended crosswinds, since actual crosswinds normally include unsteady gusts, and designing to just meet this requirement might not leave a margin for safety in real crosswinds. The takeoff and landing techniques, not specified, should be those normally employed in crosswind operation of that aircraft.

Flight testing in 90º crosswinds of the specified velocity may be unacceptably hazardous for Level 2 and 3 operation, especially with the gustiness that often accompanies winds of that magnitude. However, actual takeoffs and landings need to be made in crosswinds up to the specified values in order to demonstrate compliance in Level 1 operation. The use of simulation is not recommended because of the difficulty in developing an accurate model of the landing gear dynamics. Thus, buildups to large crosswinds are advisable, with some use of flight-validated simulation. Some indication of aircraft crosswind capabilities can be obtained from tests at altitude or low approaches, and crosswind evaluations can be carried as alternate flight cards for target of opportunity testing.

The design crosswinds are just about impossible to find when needed for flight test, so flight testing must take advantage of what winds occur during the program, with due regard for safe buildups and avoiding extreme gustiness. At the Air Force Flight Test Center there should be no difficulty in finding appropriate crosswinds for a flight-test buildup—although the winds must be taken as they come.

Since flight testing close to the ground in 90º crosswinds of the specified velocity may be unacceptably hazardous for Level 2 and 3 operations, those takeoffs and landings may be

performed in some crosswinds less than (but close to) the required velocity. Additional slow flight may then be conducted at a safe (but low) altitude.

For bank angle and control deflections in steady sideslips see 5.2.3.9.

If the force at full roll control deflection is greater than the limit, measurements of sideslip and roll control force in crosswind operation shall furnish the verification.

5.2.3.10.1 Final approach in crosswinds.
Yaw-and-roll control power shall be adequate to
aintain wings level with at least __(1)__ of sideslip in the power approach with yaw control pedal forces not exceeding the values specified in table LXXIV. For Level 1, roll control shall not exceed either __(2)__ of force or __(3)__ of control power available to the pilot. For Levels 2 and 3, roll control force shall not exceed __(4)__.

REQUIREMENT RATIONALE

These requirements define roll and yaw control power for operations near the ground, accounting for expected crosswinds. Roll and yaw control power must be available for maneuvering and countering atmospheric disturbances while sideslipping. Additionally, the ability to roll moderately in operational sideslipping, including compensating for a gust disturbance, is necessary to minimize flight-safety operational restrictions. Roll control power available to the pilot is called out in recognition that flight control system features, for example stability augmentation, may give the pilot less than full authority at times. This requirement also limits the roll and yaw control forces required for crosswind takeoffs and landings to tolerable values.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.7.1
MIL-STD-1797: 4.5.8.3, 4.5.9.5.4, and 4.6.6.1.

Requirement 5.2.3.10.1 does not apply for land-based air vehicles equipped with crosswind landing gear or otherwise constructed to land in a large crabbed attitude. For all other air vehicles the recommended values for blanks (1) through (4) are:
Blank (1): 10º
Blank (2): 10 lbs
Blank (3): 75%
Blank (4): 20 lbs.

REQUIREMENT LESSONS LEARNED

See 5.2.3.10.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall proceed or accompany careful build-up to suspected critical flight conditions. As with all requirements on control power, it is desirable to show compliance through actual testing. However, as with all crosswind requirements, suitable conditions can be difficult to find. Operation in some level of crosswind that falls within the maximum specified in 5.2.3.10 should provide some indication of trends. Careful analysis to extrapolate these results to the limits of this requirement may then be accepted in lieu of further testing.

5.2.3.10.2 Takeoff run and landing rollout in crosswinds.
Yaw and roll control power, in conjunction with other normal means of control, shall be adequate to maintain a straight path on the ground or other landing surface during takeoff run and landing rollout in Calm air and in crosswinds up to the values in table LXXIII, with cockpit control forces not exceeding the values specified in 5.2.3.10. This requirement applies on __(1)__ runways. Aerodynamic control power alone shall be sufficient to maintain control at all airspeeds above __(2)__. For very slippery runways, these requirements need not apply for crosswind components at which the force tending to blow the air vehicle off the runway exceeds the opposing tire-runway frictional forces with the tires supporting all of the air vehicle's weight.

REQUIREMENT RATIONALE

These requirements define roll and yaw control power for operations on the ground, accounting for expected crosswinds. Control power must be available for maneuvering and countering atmospheric disturbances on the ground. This requirement also limits the roll and yaw control forces required for crosswind takeoffs and landings to tolerable values.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.7.2 and 3.3.7.2.1
MIL-STD-1797: 4.5.8.3, 4.5.9.5.3, 4.6.6.1, and 4.6.7.4.

Recommended values for blanks (1) and (2) are:
Blank (1): All air vehicles: dry and wet
Air vehicles intended to operate in snow and ice: dry, wet, snow-packed, and icy
Blank (2): For Class IV air vehicles: 50 kts
Class I, Class II, and Class III air vehicles: 30 kts

REQUIREMENT LESSONS LEARNED

An aft center of gravity will put less weight on the nose gear, a consideration for nose-wheel steering. At touchdown, since lifting forces are high, the landing-gear ground reactions may be low; thus it may be necessary to depress the upwind wing as in a steady sideslip to achieve balanced lateral forces.

When lateral control is lost at lower speeds and it is no longer possible to hold the wings level, the resulting lateral component of lift may cause a large unbalance of lateral forces. Although it may be possible to maintain heading, the aircraft skids downwind. Application of upwind brake to counteract the skidding may or may not be successful. A full analysis gets rather involved, but AFFDL-TR-65-218 gives some simplified equations. Also see 5.2.3.10 Requirement Lessons Learned.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall precede or accompany careful build-up to suspected critical flight conditions. As with all requirements on control power, it is desirable to show compliance through actual testing. However, as with all crosswind requirements, suitable conditions can be difficult to find. Operation in some level of crosswind that falls within the maximum specified in 5.2.3.10 should provide some indication of trends. Careful analysis to extrapolate these results to the limits of this requirement may then be accepted in lieu of further testing.

5.2.3.10.2.1 Additional requirements for carrier-based air vehicles.
All carrier-based air vehicles shall be capable of maintaining a straight path on the ground without the use of wheel brakes, at airspeeds of __(1)__and above, during takeoffs and landings in a 90º crosswind of at least __(2)__, with cockpit control forces not exceeding the values specified in 5.2.3.10.

REQUIREMENT RATIONALE

These requirements define roll and yaw control power for operations on the ground, accounting for expected crosswinds. Control power must be available for maneuvering and countering atmospheric disturbances on the ground. This requirement also limits the roll and yaw control forces required for crosswind takeoffs and landings to tolerable values.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.7.2.2
MIL-STD-1797: 4.5.9.5.3, 4.6.6.1, and 4.6.7.4.

Note that 5.2.3.10.2.1 only applies for carrier-based air vehicles:
Blank (1): 30 kts
Blank (2): $0.10V_S(L)$

REQUIREMENT LESSONS LEARNED

This requirement is taken relatively unchanged from MIL-F-8785, 3.4.13.1.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall precede or accompany careful build-up to suspected critical flight conditions. As with all requirements on control power, it is desirable to show compliance through actual testing. However, as with all crosswind requirements, suitable conditions can be difficult to find. Operation in some level of crosswind that falls within the maximum specified in 5.2.3.10 shall provide some indication of trends. Careful analysis to extrapolate these results to the limits of this requirement may then be accepted in lieu of further testing.

5.2.3.10.3 Taxiing wind speed limits.
It shall be possible to taxi on a dry surface at any angle to a __(1)__ wind.

REQUIREMENT RATIONALE

Control power must be available for maneuvering and countering atmospheric disturbances on the ground while getting to and from the runway. The conditions under which it must be possible to taxi have been specified since there is generally no point in being able to takeoff or land in a given crosswind if the aircraft cannot be taxied.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.7.3
MIL-STD-1797: 4.5.8.3 and 4.6.6.1.

Recommended values for Blank (1) are: For Class I air vehicles: 35-kt
For Class II, III, and IV air vehicles: 45-kt

REQUIREMENT LESSONS LEARNED

The winds specified for taxi are somewhat higher than the specified crosswinds, since taxiing must be possible in any direction. The wind speeds specified are a compromise between what is desired and what is reasonable to require. At some point while taxiing, the aircraft will likely be broadside to the wind, even though the wind tends to be somewhat aligned with the active runway. Dihedral effect will tend to repress the downwind wing.

Landing gear characteristics, for example nose-wheel centering and castering, affect ground handling. Some crosswind gears have made taxiing difficult.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall precede or accompany careful build-up to suspected critical flight conditions. Ground taxi shall be performed in winds that are at least close in average magnitude to those specified. Choice of wind conditions shall account for variability and gustiness.

5.2.3.11 Lateral-directional control with asymmetric thrust.
Following sudden asymmetric loss of thrust from any factor, the air vehicle shall be safely controllable in the crosswinds of table LXXIII from the unfavorable direction. The air vehicle transient motions following sudden asymmetric loss of thrust shall be such that dangerous conditions can be avoided by pilot corrective action. Requirements 5.2.3.11 through 5.2.3.11.5 shall be met for the appropriate Flight Phases when any single failure or malperformance of the propulsive system including inlet, exhaust, engines, propellers, or drives causes loss of thrust on one or more engines or propellers. Also, the effect of the failure or malperformance on all subsystems powered or driven by the failed propulsive system shall be included when meeting these requirements. A realistic time delay of at least __(1)__ between the failure and initiation of pilot corrective action shall be incorporated when determining compliance with these requirements (5.2.1.5).

REQUIREMENT RATIONALE

The transient and steady-state effects of asymmetric thrust must be limited to amounts which can be compensated by pilot control action. These requirements assure directional control by the pilot under adverse conditions (i.e., crosswinds) and insure a match between upsetting yawing moments due to asymmetric thrust and restoring moments from static directional stability.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1.3 and 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.9 and 3.3.9.3
MIL-STD-1797: 4.5.8.3 and 4.6.5.1.

Recommended value for Blank (1): 1 sec

REQUIREMENT LESSONS LEARNED

Asymmetric loss of thrust may be caused by many factors including engine failure, inlet unstart, propeller failure or propeller-drive failure. Generally takeoffs are planned to account for possible loss of an engine; thus reaching a control limit may restrict performance even in normal crosswind takeoff or landing.

Considering variations in pilot alertness to possible failure, 2 sec may be a more representative time delay. One second shall be regarded as an absolute minimum. Also consult 5.2.1.5 Requirement Lessons Learned for realistic time delay.

Failure of any automatic compensation device is a consideration. A rational ground-rule would be to include the probability of such failure in the calculations for 5.1.1.3.1 by assuming a probability of 1 that the critical engine failure occurs and adding any failures which result in nuisance actuation of the device.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall precede or accompany careful build-up to suspected critical flight conditions. For these requirements, crosswind values shall be assumed to be invariant with altitude.

Simulated engine failures must be performed in flight, covering at least the critical conditions and covering the range of service speed and altitude for the pertinent Flight Phases. Fuel cutoff is a representative way to simulate many critical propulsion failures. Any limitations imposed by these requirements must be observed during the entire flight test program, including demonstration of compliance with performance requirements.

Generally, all the possible consequences of propulsion system failures must be considered. For example, inlet unstart may cause a disturbance in all axes. Another kind of failure is represented by damage to other parts of the aircraft caused by thrown turbine blades: for example, hydraulic lines shall be routed (or enough armor used) so that the thrown engine, fan, or propeller parts cannot sever all hydraulic systems needed for flight control. Analysis and simulation, rather than flight test, are of course in order for such considerations. At high speeds, a structural limit may be the critical factor.

Of primary importance in applying these requirements is choice of the most critical flight conditions. From a safety of flight standpoint, the most sensitive airborne condition in most cases shall be at V_{MCA}, the airborne minimum control speed. The following excerpt from FAR Part 25 paragraph 25.149 serves as a reasonable guideline for designing for V_{MCA} (the FAA uses V_{MC} for minimum airborne control speed):

> (b) V_{MC} is the calibrated airspeed at which, when the critical engine is suddenly made inoperative, it is possible to recover control of the airplane with that engine still inoperative, and maintain straight flight either with zero yaw or, at the option of the applicant, with an angle of bank of not more than 5º.
>
> (c) V_{MC} may not exceed 1.2 V_S with:
>
> (1) Maximum available takeoff power or thrust on the engines;
>
> (2) The most unfavorable center of gravity;
>
> (3) The airplane trimmed for takeoff;
>
> (4) The maximum sea level takeoff weight (or any lesser weight necessary to show V_{MC});
>
> (5) The airplane in the most critical takeoff configuration existing along the flight path after the airplane becomes airborne, except with the landing gear retracted; and

(6) The airplane airborne and the ground effect negligible; and

(7) If applicable, the propeller of the inoperative engine:

(i) Windmilling;

(ii) In the most probable position for the specific design of the propeller control; or

(iii) Feathered, if the airplane has an automatic feathering device…

For equilibrium at any given bank angle ϕ, after a thrust change ΔT

$$\beta = \frac{-\left[C_{L_l}\phi + C_y(\Delta T)\right]\left(1 - \frac{C_{n_{\delta_a}} C_{\ell_{\delta_r}}}{C_{\ell_{\delta_a}} C_{n_{\delta_r}}}\right) + \frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}}\left[C_n(\Delta T)\left(1 - \frac{C_{n_{\delta_a}} C_{\ell_{\delta_r}}}{C_{\ell_{\delta_a}} C_{n_{\delta_r}}}\right) - C_\ell(\Delta T)\left(\frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}} - \frac{C_{y_{\delta_a}} C_{n_{\delta_r}}}{C_{\ell_{\delta_a}} C_{y_{\delta_r}}}\right)\right]}{C_{y_\beta}\left(1 - \frac{C_{n_{\delta_a}} C_{\ell_{\delta_r}}}{C_{\ell_{\delta_a}} C_{n_{\delta_r}}}\right) - C_{n_\beta}\frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}}\left(1 - \frac{C_{n_{\delta_a}} C_{\ell_\beta}}{C_{\ell_{\delta_a}} C_{n_\beta}}\right) - \frac{C_{y_{\delta_a}}}{C_{\ell_{\delta_a}}}\left(C_{\ell_\beta} - \frac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}} C_{n_\beta}\right)}$$

$$\approx \frac{-C_{L_l}\phi + \frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}}\left[C_n(\Delta T)\left(1 - \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}} C_\ell(\Delta T)\right)\right]}{C_{y_\beta} - \frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}} C_{n_\beta}\left(1 - \frac{C_{n_{\delta_a}} C_{\ell_\beta}}{C_{\ell_{\delta_a}} C_{n_\beta}}\right)}$$

$$\delta_r = \frac{-C_{n_\beta}\left(1 - \frac{C_{n_{\delta_a}} C_{\ell_\beta}}{C_{\ell_{\delta_a}} C_{n_\beta}}\right)\beta - C_n(\Delta T) + \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}} C_\ell(\Delta T)}{C_{n_{\delta_r}}\left(1 - \frac{C_{n_{\delta_a}} C_{\ell_{\delta_r}}}{C_{\ell_{\delta_a}} C_{n_{\delta_r}}}\right)}$$

For δ_a, see 5.2.3.11.2 Verification.

More approximately yet, when $C_n(\Delta T) = -y_T \Delta T$ and $C_\ell(\Delta T) = 0$, derivatives are invariant with angle of attack, and rudder control for static balance is critical, the airborne minimum control speed is given by

$$V_{MC} = 17.2 \cdot \sqrt{\frac{-.0873(W/S) + \left| \frac{y_T}{b} \cdot \frac{\Delta T}{S} \cdot \frac{C_{y_\beta}}{C_{n_\beta}\left(1 - \frac{C_{n_{\delta_a}} C_{\ell_\beta}}{C_{\ell_{\delta_a}} C_{n_\beta}}\right)} \right|}{\left| \left(\frac{C_{y_\beta} C_{n_{\delta_r}}}{C_{n_\beta}\left(1 - \frac{C_{n_{\delta_a}} C_{\ell_\beta}}{C_{\ell_{\delta_a}} C_{n_\beta}}\right)} - C_{y_{\delta_r}} \right) \delta_{r_{max}} \right|}} \quad \begin{matrix} \text{kt} \\ \text{EAS} \end{matrix}$$

with $\phi = 5^\circ$ in the failed-engine-up sense (For $\phi = 0$, delete the W/S term). Either force or deflection limits may determine $\delta_{r_{max}}$.

Yaw controls free: with linear rudder hinge moments,

$$\beta = \frac{-C_n(\Delta T)}{C^*_{n_\beta}} \left(\frac{1 - \frac{C_{n_{\delta_a}} C_\ell(\Delta T)}{C_{\ell_{\delta_a}} C_n(\Delta T)}}{1 - \frac{C_{n_{\delta_a}} C^*_{\ell_\beta}}{C_{\ell_{\delta_a}} C^*_{n_\beta}}} \right)$$

$$\delta_a = \frac{\frac{C^*_{\ell_\beta}}{C^*_{n_\beta}} C_n(\Delta T) - C_\ell(\Delta T)}{C_{\ell_{\delta_a}} \left(1 - \frac{C_{n_{\delta_a}} C^*_{\ell_\beta}}{C_{\ell_{\delta_a}} C^*_{n_\beta}}\right)}$$

$$\sin\phi = \left(\frac{1}{C_{L_1}}\right) \left[-C_y(\Delta T) + \left(\frac{1 - \frac{C_{y_{\delta_a}} C^*_{\ell_\beta}}{C_{\ell_{\delta_a}} C^*_{n_\beta}}}{1 - \frac{C_{n_{\delta_a}} C^*_{\ell_\beta}}{C_{\ell_{\delta_a}} C^*_{n_\beta}}} \right) \frac{C^*_{y_\beta}}{C^*_{n_\beta}} C_n(\Delta T) + \left(\frac{C_{y_{\delta_a}} - \frac{C^*_{y_\beta}}{C^*_{n_\beta}} C_{n_{\delta_a}}}{1 - \frac{C_{n_{\delta_a}} C^*_{\ell_\beta}}{C_{\ell_{\delta_a}} C^*_{n_\beta}}} \right) \frac{C_\ell(\Delta T)}{C_{\ell_{\delta_a}}} \right]$$

where

$$C^{*}_{y_\beta} = C_{y_\beta} - C_{y_{\delta_r}} \, C_{hr_\beta} \big/ C_{hr_{\delta_r}}$$

$$C^{*}_{n_\beta} = C_{n_\beta} - C_{n_{\delta_r}} \, C_{hr_\beta} \big/ C_{hr_{\delta_r}}$$

$$C^{*}_{\ell_\beta} = C_{\ell_\beta} - C_{\ell_{\delta_r}} \, C_{hr_\beta} \big/ C_{hr_{\delta_r}}$$

Except for some extreme wartime situations, aircraft operations will be limited to conditions deemed safe in the event of an engine failure. To minimize the danger (in the event of another failure) of flight testing near the ground, tests might be conducted down to the minimum safe altitude, extrapolated to sea level—accounting for any derivative changes with angle of attack or thrust coefficient—and spot checked at the lower altitude. This technique has worked well with some airplanes, but not so well for the B-1.

Crosswind landings performed to demonstrate compliance with 5.2.3.10 must include simulated asymmetric thrust with a crosswind blowing in the unfavorable direction [i.e., from the direction of the bad engine(s)]. Operation with an engine failure will in most aircraft be Level 2, which implies a pilot rating of only 6-1/2 or better in a 10-kt crosswind. It is clear that the subjective nature of this requirement (safely controllable) allows considerable leeway in its application.

5.2.3.11.1 Thrust loss during takeoff run.
It shall be possible for the pilot to maintain control of the air vehicle on the takeoff surface following sudden asymmetric loss of thrust from the most critical propulsive source, allowing a realistic time delay of at least __(1)__ between the failure and initiation of pilot corrective action. Thereafter it shall be possible to achieve and maintain a straight path on the takeoff surface without a deviation of more than __(2)__ from the path originally intended, with yaw control pedal forces not exceeding __(3)__. For the continued takeoff, the requirement shall be met when thrust is lost at speeds from the refusal speed (based on the shortest runway from which the air vehicle is designed to operate) to the maximum takeoff speed with takeoff thrust maintained on the operative engine(s), using only controls not dependent upon friction against the takeoff surface and not dependent upon release of the pitch, roll, yaw, or throttle controls. For the aborted takeoff, the requirement shall be met at all speeds below the maximum takeoff speed; however, additional controls such as nosewheel steering and differential braking may be used. Automatic devices which normally operate in the event of a thrust failure may be used in either case.

REQUIREMENT RATIONALE

The transient and steady-state effects of asymmetric thrust must be limited to amounts which can be compensated by pilot control action. The requirement for adequate control of the ground path insures that, following loss of thrust during the takeoff run, the pilot can either safely abort or

safely continue the takeoff. This requirement provides an additional assurance of flight safety. Safe operation in the event of a propulsive failure is a critical consideration in multi-engine aircraft. Yaw control force is an important aspect.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1.3 and 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.9.1
MIL-STD-1797: 4.5.8.3, 4.6.5.1, 4.6.6.2, and 4.6.7.8.

Recommended values for blanks (1) through (3) are:
Blank (1): 1 sec
Blank (2): 30 ft
Blank (3): 180 lbs.

REQUIREMENT LESSONS LEARNED

The Federal Aviation Regulations, among other sets of requirements, have similar requirements. V_{MCG}, the minimum controllable airspeed on the ground, is defined as the minimum airspeed at which the runway deviation requirements of this paragraph can be met. For operation at forward bases, narrower runways may have to be a design consideration in combination with the main-gear track.

Considering the capabilities of the entire population, 120 lbs may be a more satisfactory, safer upper limit for yaw control pedal forces. Pending further data, that limit (as in FAR Part 23 for prolonged operation) might be used here.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall proceed or accompany careful build-up to suspected critical flight conditions. Verification shall be accomplished during demonstration of compliance with 5.2.3.11.

With the aircraft configured at its lightest weight, simulated engine failures during takeoff shall be performed in the conditions specified in 5.2.3.11. Ground simulation is not recommended for ultimate verification of compliance with the ground roll portion of this requirement because of the known problems with developing an accurate landing gear model.

5.2.3.11.2 Thrust loss after takeoff.
After liftoff, it shall be possible for the pilot to achieve straight flight without a change in selected configuration following sudden asymmetric loss of thrust from the most critical factor at speeds from $V_{min}(TO)$ to $V_{max}(TO)$, allowing a realistic time delay of at least __(1)__ between the failure and initiation of pilot corrective action. Thereafter, the pilot shall be able to maintain straight flight throughout the climbout and to perform __(2)__ banked turns with and against the inoperative propulsive unit. The yaw-control-pedal force required to maintain straight flight throughout the climbout with asymmetric thrust shall not exceed __(3)__. Roll control shall not exceed either the force limits specified in table LXXV or __(4)__ of available roll control power, with takeoff thrust maintained on the operative engine(s) and trim at normal setting for takeoff with symmetric thrust. Automatic devices which normally operate in the event of a thrust failure may be used, and during straight flight the air vehicle may be banked up to __(5)__ away from the inoperative engine.

TABLE LXXV. Maximum roll forces for thrust loss after takeoff.

Level	Maximum Roll Control Force (lbs)
1	
2	
3	

REQUIREMENT RATIONALE

The transient and steady-state effects of asymmetric thrust must be limited to amounts which can be compensated by pilot control action. This requirement insures that following thrust loss after takeoff the pilot can safely go around or continue climbout. The intent is that $V_{min}(TO)$ normally should be set by other considerations and adequate control provided down to that speed. The requirement for turn capability, similar to FAR 25.147(c), addresses the need to insure maneuvering capability in airport environments to avoid obstacles that become a threat due to the heading change likely incurred with the loss of an engine. For flight safety, a roll control margin must be provided for gentle maneuvering and for countering gusts while coping with a propulsive failure. This requirement also assures that roll control forces required to counter the effects of a failed engine are within a pilot's capabilities. This requirement provides an additional assurance of flight safety. Safe operation in the event of a propulsive failure is a critical consideration in multi-engine aircraft. Yaw control force is an important aspect.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1.3 and 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.9.2

MIL-STD-1797: 4.5.8.3, 4.5.8.4, 4.5.9.5.5, 4.6.5.1, 4.6.6.2, and 4.6.7.8.

The recommended values for table LXXV are:

Level	Class	Maximum Force (lbs)		
		Centerstick	Wheel	Sidestick
1	I, II-C, and IV	20	20	Forces shall not be so large or so small as to be objectionable to the pilot.
	II-L and III	25	25	
2	I, II-C, and IV	20	20	
	II-L and III	30	30	
3	All	35	70	

The recommended values for blanks (1) through (5) are:
Blank (1): 1 sec
Blank (2): 20º
Blank (3): 180 lbs
Blank (4): 75%
Blank (5): 5º.

REQUIREMENT LESSONS LEARNED

The Federal Aviation Regulations, among other sets of requirements, have similar requirements.

The requirement not to exceed 75 percent of available roll control is based on judgement and experience, since no hard data are available. The 25 percent margin, though arbitrary, has generally proved sufficient to assure flight safety when an engine fails. A rational alternative would be to apply a gust at least commensurate with the expected mean value of turbulence (σ = 2.9 ft/sec, AFFDL-TR-69-72) at the critical time, but that added complication seems unwarranted.

This requirement may well determine the minimum approach speed for a powered-lift aircraft. While the YC-15 relied on the pilot to retain control, the YC-14 incorporated automatic reconfiguration and control inputs to handle thrust asymmetry.

Five degrees is about the greatest bank angle possible without significantly reducing the vertical component of lift. In B-1B flight testing it was found that other alternatives to the 5º bank angle limitation, such as reducing the asymmetry or accelerating, were acceptable.

The maximum force limits recommended for table LXXVI are the same as those recommended for table LXXI.

Again, 120 lbs may be a more satisfactory, safer upper limit for yaw control pedal forces considering the capabilities of the entire pilot population. Pending further data, that limit (as in FAR Part 23 for prolonged operation) might be used here.

Requirement 5.2.3.11.5 has additional requirements on a second propulsive failure.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall precede or accompany careful build-up to suspected critical flight conditions. Verification shall be accomplished during demonstration of compliance with 5.2.3.11.

While airborne, the necessary steady roll control deflection is

$$\delta_a = \frac{-\left(C_{\ell_\beta} - C_{n_\beta} C_{\ell_{\delta r}} / C_{n_{\delta r}}\right)\beta - C_\ell(\Delta T) + C_n(\Delta T) C_{\ell_{\delta_r}} / C_{n_{\delta_r}}}{C_{\ell_{\delta a}}\left(1 - \frac{C_{n_{\delta a}} C_{\ell_{\delta r}}}{C_{\ell_{\delta a}} C_{n_{\delta r}}}\right)}$$

where

$$\beta \doteq \frac{-C_{L_1}\phi + \left[C_n(\Delta T) - C_\ell(\Delta T) C_{n_{\delta a}} / C_{\ell_{\delta a}}\right] C_{y_{\delta r}} / C_{n_{\delta r}}}{C_{y_\beta} - \left(1 - \frac{C_{n_{\delta a}} C_{\ell_\beta}}{C_{\ell_{\delta a}} C_{n_\beta}}\right) \frac{C_{y_{\delta r}}}{C_{n_{\delta r}}} C_{n_\beta}}$$

Roll control forces shall be determined in the course of verifying compliance with 5.2.3.11.

5.2.3.11.3 Thrust loss during waveoff /go-around.
At any airspeed down to $V_{min}(L)$ it shall be possible to achieve and maintain steady, straight flight with waveoff (go-around) thrust on the remaining engine(s) following sudden asymmetric loss of thrust from the most critical factor, allowing a realistic time delay of at least __(1)__ between the failure and initiation of pilot corrective action. Configuration changes within the capability of the crew while retaining control of the air vehicle, and automatic devices that normally operate in the event of a propulsion failure, may be used.

REQUIREMENT RATIONALE

The transient and steady-state effects of asymmetric thrust must be limited to amounts which can be compensated by pilot control action.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1.3 and 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.5.8.3 and 4.6.5.1.

The recommended value for Blank (1) is: 1 sec.

REQUIREMENT LESSONS LEARNED

See 5.2.3.11 Requirement Lessons Learned.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall precede or accompany careful build-up to suspected critical flight conditions. Verification shall be accomplished during demonstration of compliance with 5.2.3.11.

5.2.3.11.4 Yaw controls free.
The static directional stability shall be such that at all speeds above $1.4V_{min}$, with asymmetric loss of thrust from the most critical propulsive source while the other engine(s) develop normal rated thrust, the air vehicle with yaw-control-pedals free can be balanced directionally in steady straight flight. The trim settings shall be those required for wings-level straight flight prior to the failure. Roll control forces shall not exceed the limits specified in table LXXVI.

TABLE LXXVI. Maximum roll forces for asymmetric thrust with yaw controls free.

Level	Flight Phase Category	Maximum Roll Control Force (lbs)
1	A	
	B	
	C	
2	A	
	B	
	C	
3	A	
	B	
	C	

REQUIREMENT RATIONALE

The transient and steady-state effects of asymmetric thrust must be limited to amounts which can be compensated by pilot control action. This requirement is intended to preclude the consequences of stalling the vertical tail in case of an engine failure and to assure that roll control forces required to counter the effects of a failed engine are within the pilot's capabilities.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1.3 and 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.9.4
MIL-STD-1797: 4.5.9.5.5 and 4.6.5.1.

Recommended values for table LXXVI:

Level	Class	Flight Phase Category	Maximum force (lbs)		
			Centerstick	Wheel	Sidestick
1	I, II-C, and IV	A and B	30	60	Forces shall not be so large or so small as to be objectionable to the pilot.
		C	20	20	
	II-L and III	A and B	30	60	
		C	30	30	
2	I, II-C, and IV	A and B	30	60	
		C	20	20	
	II-L and III	A and B	30	60	
		C	30	30	
3	All	All	35	70	

REQUIREMENT LESSONS LEARNED

The maximum force limits recommended for table LXXVI are the same as those recommended for table LXXI except that with yaw controls free the Level 2 values from table LXXI are used for both Levels 1 and 2.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall precede or accompany careful build-up to suspected critical flight conditions. With rudder pedals free, larger bank and sideslip angles generally will be needed.

Roll control forces shall be determined in the course of verifying compliance with 5.2.3.11.

5.2.3.11.5 Two-engine failures in multi-engine air vehicles (more than two engines). At the one-engine-out speed for maximum range with any engine initially failed, it shall be possible upon sudden failure of the most critical remaining engine to stop the transient motion and thereafter to maintain straight flight from that speed to the speed for maximum range with both engines failed, allowing a realistic time delay of at least __(1)__ between the failure and initiation of pilot corrective action. In addition, it shall be possible to effect a safe recovery at any speed from $V_{o_{min}}(C)$ to $V_{min}(C)$ following sudden simultaneous failure of the two most critical failing engines, allowing a realistic time delay of at least __(2)__ between the failure and initiation of pilot corrective action.

REQUIREMENT RATIONALE

The transient and steady-state effects of asymmetric thrust must be limited to amounts which can be compensated by pilot control action. This requirement provides an additional assurance of flight safety, including the more remote but possible event of two sequential or simultaneous propulsive failures. The unlikely event of two critical engines failing has occurred in service often enough to warrant guarding against the resultant loss of an aircraft.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1.3 and 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.9.5
MIL-STD-1797: 4.6.6.2.

These requirements only apply to multi-engine air vehicles (more than two engines).

The recommended value for both blanks (1) and (2) is: 1 sec.

REQUIREMENT LESSONS LEARNED

Federal Aviation Regulations, among other sets of requirements, have similar requirements.

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall precede or accompany careful build-up to suspected critical flight conditions. The specialized nature of this requirement, as well as the general qualitative terms in which it is worded, makes it straightforward to apply. Buildups may be advisable in flight testing.

Flight testing at altitudes covering the ROSH will be at conditions stated by the requirements, e.g., for two engines out the airspeeds will be either speed for maximum range with one engine out or representative speeds at and above $V_{o_{min}}(C)$. At high speeds, structural limits may govern the allowable excursions.

5.2.3.12 Wings-level turn.
For the wings-level-turn mode of control, the air vehicle response shall meet the requirements of 5.2.3.12.1 through 5.2.3.12.4.

REQUIREMENT RATIONALE

These requirements are included to specify the response characteristics of aircraft utilizing an extra DFC mode for wings-level turns.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.6.1.3.

Note that 5.2.3.12 through 5.2.3.12.4 apply only to air vehicles with a wings-level-turn mode of control.

REQUIREMENT LESSONS LEARNED

The wings-level-turn mode, allowing changes in heading to occur at zero bank angle, is sometimes referred to as the "a_y" mode (see, for example, AIAA Paper 77-1119); a sketch of the aircraft motions and a summary of the useful features is given on figure 191.

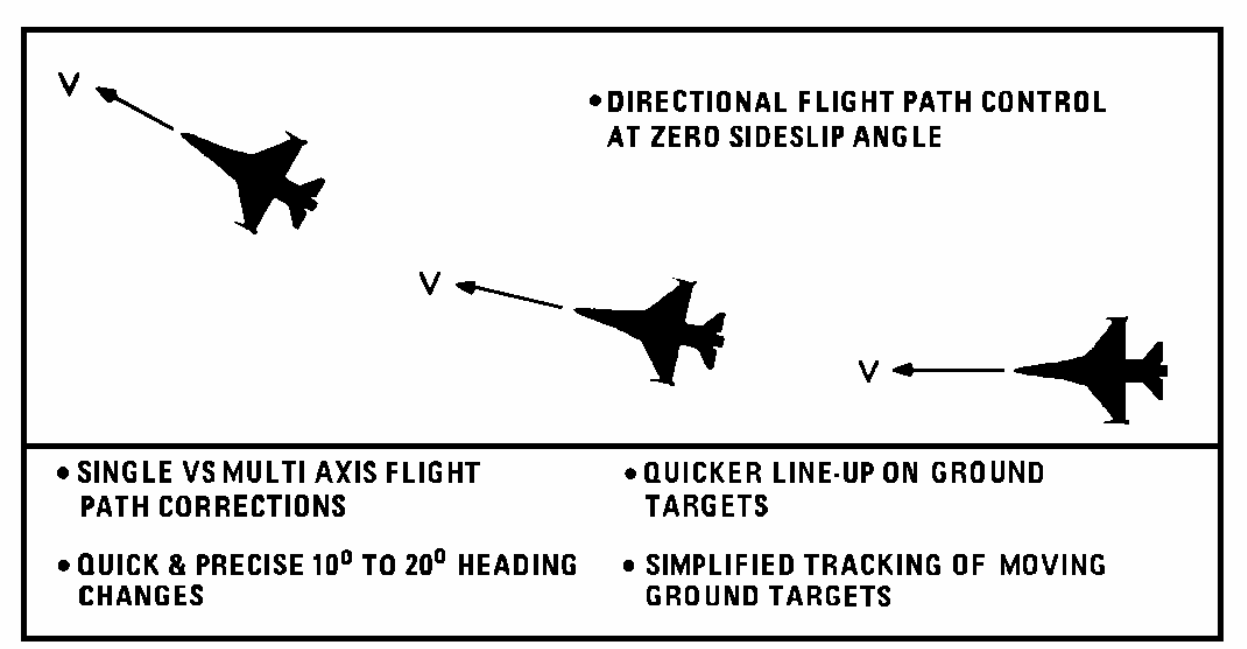


FIGURE 191. Wings-level-turn mode.

The F-16 CCV utilized a wings-level-turn mode (AIAA Paper 77-1119) with considerable success. Pilots found the mode particularly useful for air-to-ground missions. The aircraft was capable of approximately 0.8 g, and at least one pilot reported that this would not be excessive if adequate pilot lateral restraints could be provided. Two DFC controllers were tried: the conventional rudder pedals and a CCV thumb button. The rudder pedals were the favored controller (see AFFDL-TR-78-9).

When mechanizing a wings-level-turn mode, consideration should be given to the impact of the particular mechanization on aircraft flying qualities and performance degradation. A large drag rise can occur when aerodynamic surfaces are deflected to generate the wings-level turn, causing a rapid airspeed bleedoff. The loss of aircraft velocity may offset any tactical advantage gained from using the mode.

VERIFICATION

Verification shall be via verification of requirements 5.2.3.12.1 through 5.2.3.12.4.

5.2.3.12.1 Dynamic response to direct side force controller.
Turns shall occur at approximately zero sideslip angle and zero or constant small bank angle when using the direct side force controller. The bandwidth of the open-loop response of heading, lateral flight path angle, or lateral velocity to the direct side force input shall be greater than the values in table LXXVII.

TABLE LXXVII. Bandwidth limits for wings-level-turn mode.

Task	Control Variable	Required Bandwidth (rad/sec)	
		Level 1	Level 2
Air-to-air tracking			
Air-to-ground tracking/pointing/strafing			
Dive bombing			
Formation flying			
Air-to-air refueling			
Approach			

REQUIREMENT RATIONALE

These requirements are included to specify the response characteristics of aircraft utilizing an extra DFC mode for wings-level turns.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.6.1.3.

Recommended values for table LXXVII:

Task	Control Variable	Required Bandwidth (rad/sec)	
		Level 1	Level 2
Air-to-air tracking	Heading angle if sideslip is not an important factor for weapon release. Lateral flight path angle if sideslip must be small for weapon release.	1.25	0.60
Air-to-ground tracking/pointing/strafing	Heading angle	1.25	0.60
Dive bombing	Lateral flight path angle or lateral velocity	1.25	0.60
Formation flying	Lateral flight path angle or lateral velocity	0.30	0.12
Air-to-air refueling	Lateral flight path angle or lateral velocity	0.30	0.12
Approach	Lateral flight path angle or lateral velocity	0.30	0.12

REQUIREMENT LESSONS LEARNED

The bandwidth criterion used in this requirement makes the fundamental assumption that the primary factor in the pilot's evaluation of a DFC mode is his ability to exert tight control to minimize errors and thereby improve closed-loop tracking performance. The criterion originates from an old, well-accepted idea: that a measure of the handling qualities of an aircraft is its response characteristics when operated in a closed-loop compensatory tracking task. The bandwidth criterion was also used to set limits on pitch attitude dynamics in 5.2.2.1.3.1.

The bandwidth (ω_{BW}) is a measure of the maximum frequency at which such closed-loop tracking can take place without requiring the pilot to supply favorable dynamic compensation and without threatening stability. It follows that aircraft capable of operating at a large value of bandwidth will have superior performance, readily achieved. An implicit characteristic of the requirement is that inter-axis coupling or contamination, regardless of type or source, affects the pilots opinion only so far as it affects the bandwidth. This is a highly desirable feature of the requirement because the very large varieties of coupling that can occur would make it virtually impossible to classify and set limits on each type of coupling. For additional background on the use of bandwidth as a criterion for DFC modes the reader is referred to AFWAL-TR-81-3027, AFFDL-TR-76-78, and AFFDL-TR-78-9.

The variable-stability flight test experiment of AFWAL-TR-81-3027 provides supporting data for both limiting values of bandwidth and validity of bandwidth as a criterion for DFC modes. The Cooper-Harper pilot ratings from the AFWAL-TR-81-3027 experiment are plotted versus heading bandwidth on figure 192 for the air-to-air tracking task using the wings-level-turn mode. The open symbols on figure 192 indicate that variations in heading bandwidth were achieved via yaw coupling. That is, the crossfeed gain from DFC control (pedal) to the rudder was increased above its nominal value to achieve favorable yaw coupling and reduced below its nominal value to achieve unfavorable yaw coupling. The heading bandwidth was varied via changes in roll coupling, i.e., the DFC to aileron gain, for solid symbols on figure 192.

To the pilot, favorable yaw coupling appears as a tendency for the nose to move abruptly in the direction of the commanded turn, whereas unfavorable yaw coupling is a tendency for the nose initially to swing away from the commanded turn. When flying a configuration with favorable roll coupling, the pilot will observe a tendency for the aircraft to roll in the direction of the commanded wings-level turn, thereby augmenting the yaw response (provided roll is not too large). Finally, adverse roll coupling is a tendency for the aircraft to bank away from the commanded wings-level turn. The validity of bandwidth as a criterion for DFC is supported by the following observations from figure 192:

> The pilot ratings for Configurations WLT4 and WLT15 (adverse yaw coupling) are approximately the same as the pilot rating for Configuration WLT13 (adverse roll coupling). As can be seen from figure 192, all of these configurations have approximately the same heading bandwidth of between 0.7 and 0.8 rad/sec.
>
> Configuration WLT3 (slight adverse yaw coupling) has approximately the same pilot rating as Configuration WLT14 (slight adverse roll coupling). The bandwidths of these configurations are both approximately 1.1 rad/sec.
>
> Configurations WLT10 and WLT12 have significant favorable roll coupling and correspondingly high values of heading bandwidth. Configuration WLT5 also has a large value of heading bandwidth (4.1 rad/sec) by virtue of its highly proverse yaw coupling. Figure 192 indicates these configurations are all rated approximately the same.

The above examples provide strong evidence to indicate that satisfactory wings-level-turn flying qualities depend primarily on the ability of the pilot to increase his tracking bandwidth to some established level by tightening up on the controls.

The rating data on figure 192 indicate that even the best wings-level-turn configurations barely meet the classical definition of Level 1 flying qualities (e.g., Cooper-Harper pilot rating equal to or better than 3-1/2). However, when one considers that the task involves tracking a target undergoing large and rapid bank angle reversals, it is difficult to conceive of any configuration that would correspond to the adjectival descriptions of a pilot rating of 3 (i.e., “minimal pilot compensation required for desired performance”). The pilot commentary in AFWAL-TR-81-3027 indicates the WLT1 configuration had very acceptable flying qualities and the desired performance in tracking was easily obtained (but apparently involved more than minimal

compensation). Hence, the inability to achieve average pilot ratings better than 3 is believed attributable not to the configuration but rather to the difficulty of the task involved. Pilot ratings of 2 for wings-level-turn mode were obtained in AIAA-80-1628-CP. The tracking task in that case was a ground target that performed a discrete step change in position, a significantly less demanding task than the air-to-air tracking utilized in AFWAL-TR-81-3027.

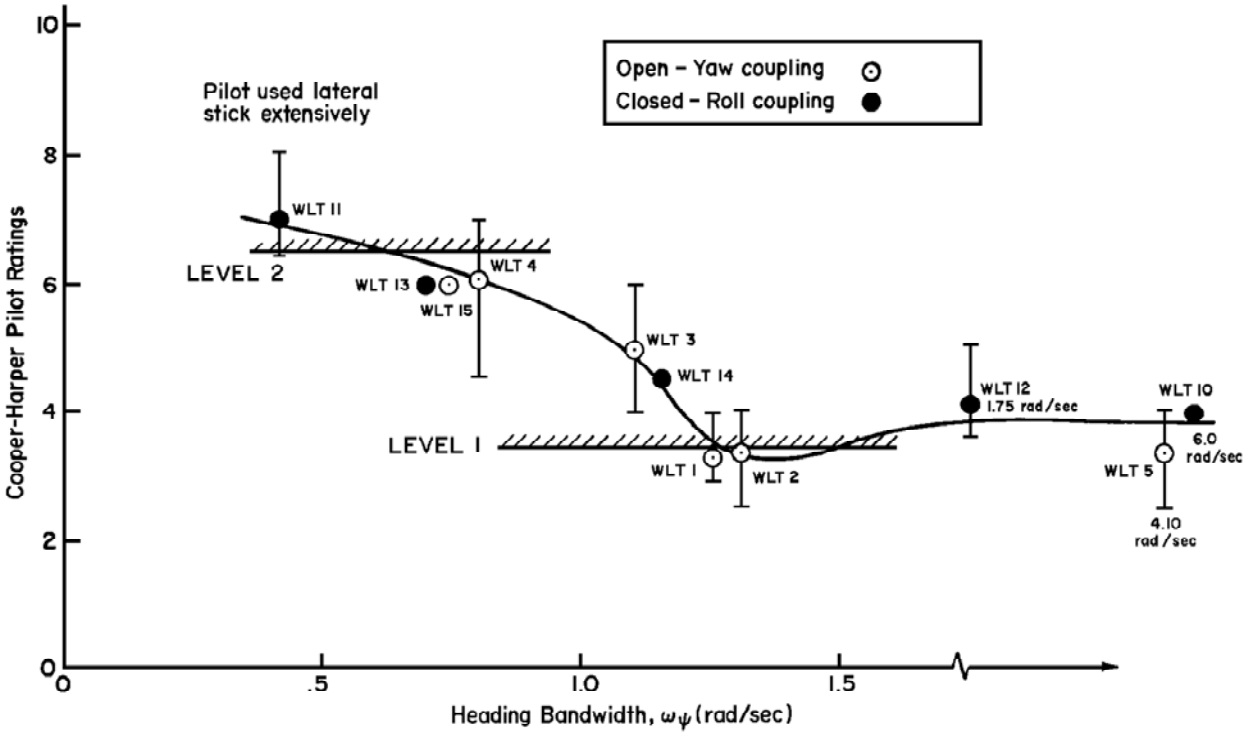


FIGURE 192. Correlation of pilot ratings with heading bandwidth; wings-level-turn mode; air-to-air tracking task.

The path deviation bandwidth requirements recommended for the approach task are based on the lateral translation mode results given in AFWAL-TR-81-3027, as well as heading control results obtained for conventional aircraft in previous programs. An example of such results is shown on figure 193, taken from NASA CR-2017. Figure 193 indicates that most points below a heading bandwidth of 0.3 rad/sec are Level 2 or worse. For lack of any better data, the Level 2 boundary was defined (from figure 193) as 0.12 rad/sec.

"Criteria for Side-Force Control in Air-to-Ground Target Acquisition and Tracking", elaborating on the NASA Ames FSAA results of NASA TM 81266 and AIAA Paper 80-1628-CP for a dive-bombing task, recommends a bandwidth of at least 2.3 rad/sec for a Level 1 wings-level-turn mode, almost double the AFWAL-TR-81-3027 recommendation adopted here. Different tasks, simulators, and pilots may have contributed to this difference in results.

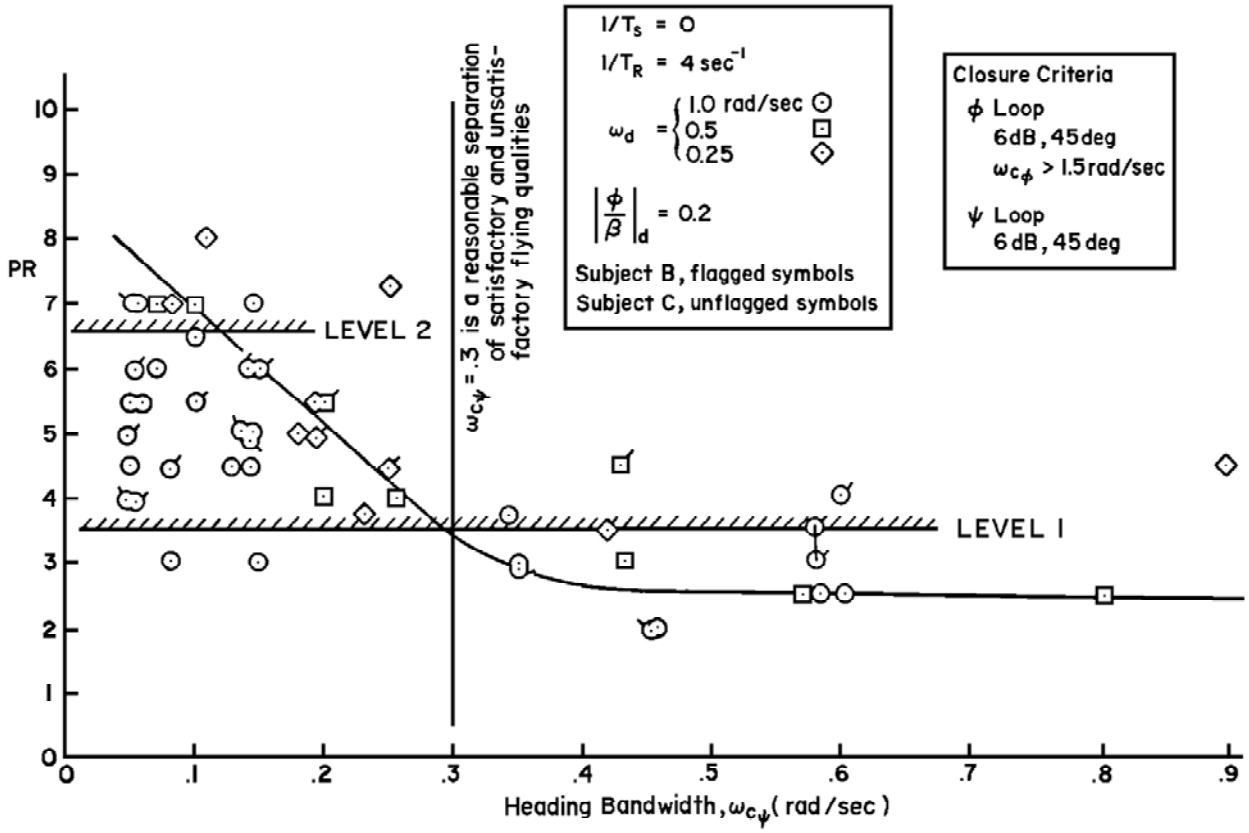


FIGURE 193. Correlation of pilot ratings with heading bandwidth for conventional aircraft; ILS approach task.

VERIFICATION

The bandwidth frequency, ω_{BW}, to be used in table LXXVII is defined from the frequency response plot of heading or lateral flight-path angle to cockpit direct-force control input (i.e., χ/F_{DFC} or ψ/F_{DFC}. Specifically, it is the lower of two frequencies: the frequency at which the phase margin is 45º, or the frequency for a gain margin of 6 dB (figure 194). Before this definition is applied, first determine the frequency for neutral closed-loop stability (180º phase angle) from the phase portion of the Bode plot (ω_{180}). The next step is to note the frequency at which the phase margin is 45º. This is the bandwidth frequency defined by phase, $\omega_{BW_{phase}}$. Then, note the amplitude corresponding to ω_{180} and add 6 dB to it. The frequency at which this value occurs on the amplitude curve is $\omega_{BW_{gain}}$. Finally, the bandwidth, ω_{BW}, is the lesser of $\omega_{BW_{phase}}$ and $\omega_{BW_{gain}}$.

If $\omega_{BW} = \omega_{BW_{phase}}$, the system is said to be phase-margin limited. On the other hand, if $\omega_{BW} = \omega_{BW_{gain}}$ the system is gain-margin limited; that is, the aircraft is driven to neutral stability when the pilot increases his gain by 6 dB (a factor of 2). Gain-margin-limited aircraft may have a great deal of phase margin, Φ_M, but then increasing the gain slightly causes Φ_M to decrease

rapidly. Such systems are characterized by frequency-response amplitude plots that are flat, combined with phase plots that roll off rapidly, such as shown on figure 194.

A discussion of Fast Fourier Transform procedures that can be used to generate frequency response (Bode) plots from flight test or simulator data is given in 5.2.2.1.3.1 Verification. Once the Bode plots of heading or lateral flight path angle to DFC input are obtained it is a simple matter to determine the bandwidth as shown on figure 194.

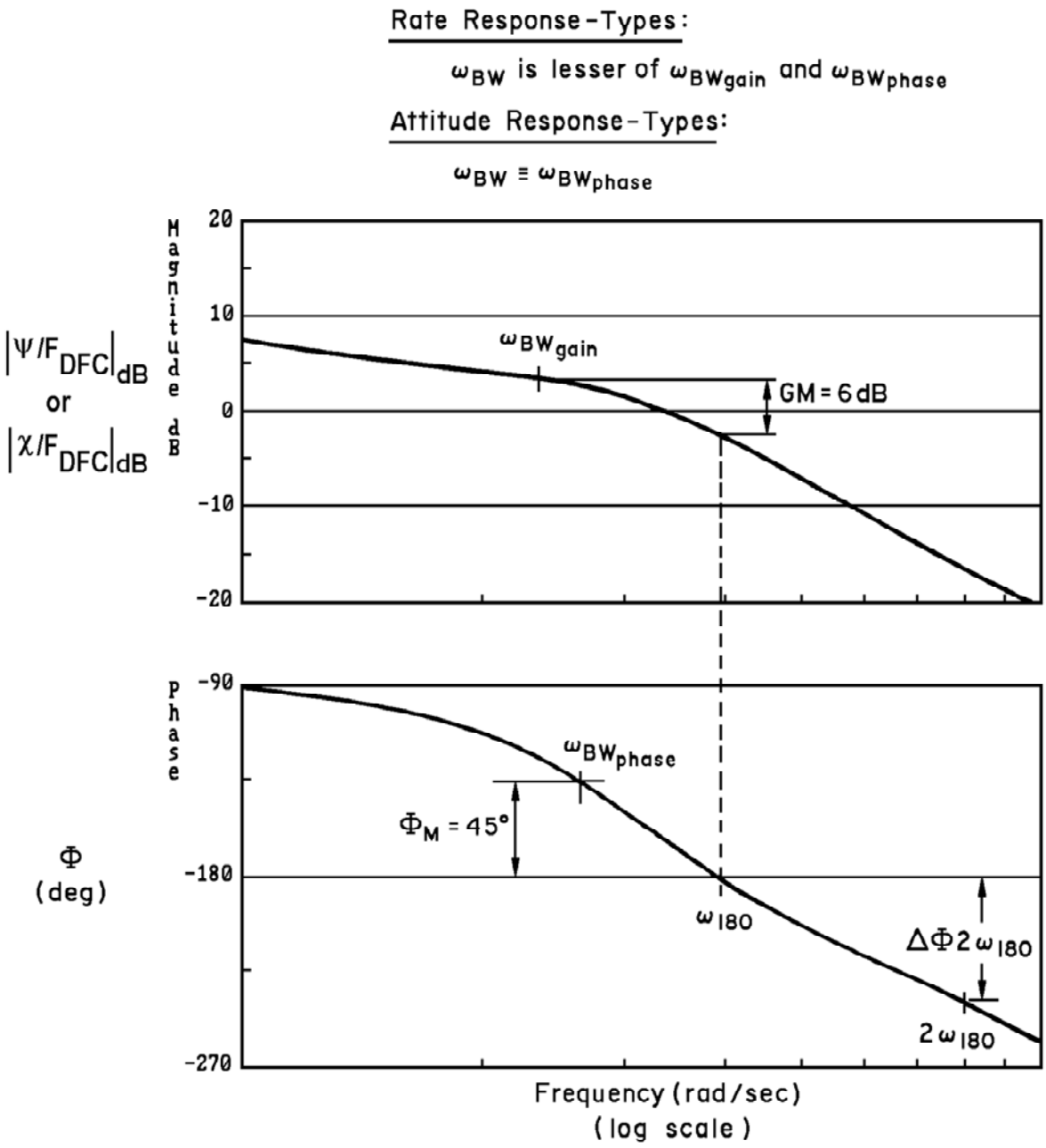


FIGURE 194. Definition of bandwidth frequency.

5.2.3.12.2 Steady-state response to direct side force controller.
Maximum direct side force inputs shall produce a steady-state lateral acceleration of at least __(1)__.

REQUIREMENT RATIONALE

These requirements are included to specify the response characteristics of aircraft utilizing an extra DFC mode for wings-level turns.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.6.1.3.

Recommended value for Blank (1):
If wings-level-turn mode is required for air-to-ground tasks: ±1 lateral g
If wings-level-turn mode is required for air combat maneuvers: ±2.5 lateral g's

REQUIREMENT LESSONS LEARNED

This requirement shall relate to the turn rate or lateral acceleration necessary to accomplish the desired task. They shall be determined by analysis, ground- and in-flight simulation based upon task requirements. AFFDL-TR-76-78 indicated that 1 lateral g was sufficient for air-to-ground and AFWAL-TR-81-3027 indicated that 2.5 lateral g would be required for air combat maneuvering. Obviously, some form of lateral pilot restraint would be required. As mentioned in 5.2.3.12.4, recent studies from the AFAMRL centrifuge (Investigation of the Effects of g_y and g_z on the AFTI/F-16 Control Inputs, Restraints, and Tracking Performance) have indicated that a restrained pilot can track up to about 2.5 lateral g. At this time, however, the operational usefulness of such large lateral accelerations has not been established.

VERIFICATION

The lateral acceleration produced by a given DFC deflection will vary with dynamic pressure, $\bar{q}$.

5.2.3.12.3 Direct side force controller forces and displacements.
Direct side force controller sensitivity shall be sufficiently high that satisfactory coordination can be achieved without unduly high control forces, yet sufficiently low that occasional, improperly-coordinated control inputs will not cause a degradation in flying qualities Level. Use of the primary direct side force controller shall not require the use of another control manipulator to meet 5.2.3.12.1.

REQUIREMENT RATIONALE

These requirements are included to specify the response characteristics of aircraft utilizing an extra DFC mode for wings-level turns.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.6.1.3.

REQUIREMENT LESSONS LEARNED

When the rudder pedals are to be used as the direct side force controller, the requirements of 5.2.3.8 may be used as a guide. If a special purpose controller such as a thumb switch or lever is to be used, acceptable characteristics shall be determined in flight test.

Some guidance regarding DFC sensitivity may be found in AFWAL-TR-81-3027 flight tests of the wings-level-turn mode. The in-flight simulator was set up so that DFC sensitivity could be varied. The pilots were asked to vary the control sensitivity of each new configuration to determine the optimum value, thereby eliminating it as a variable in the problem. It was discovered the pilot ratings were not dependent on small variations in control sensitivity for either uncoupled or adversely-coupled configurations.

The acceptability of configurations with large values of favorable yaw or roll coupling tended to be significantly more dependent on control sensitivity. This is shown by comparing figure 195 for high favorable yaw coupling and figure 196 for very high favorable roll coupling with figure 197 for low coupling. It is interesting to note that the nominal value of control sensitivity used for low coupling (0.008 g/lb) was found to be unacceptably high for the favorable coupling cases. The scatter in the data shown on figure 196 is primarily due to pilot MP. In order to help explain why MP's ratings are higher, his comments have been annotated near the appropriate data points on figure 196. It is clear that his poor ratings are based on his fundamental objection to utilizing roll coupling to improve tracking bandwidth, although his comments for the lowest sensitivity case indicate that adequate performance could be obtained in this mode.

One interpretation is that pilot MP's rating of 5 was given to discourage intentional design of proverse roll coupling to improve tracking bandwidth. Hence, even though large values of favorable roll coupling may be inferred as acceptable to produce Level 1 flying qualities, the

designer is cautioned against using such coupling to overcome an inherently low bandwidth. This is especially pertinent for configurations having a cockpit farther from the roll axis (than in the Navion) and therefore subjecting the pilot to more roll induced lateral acceleration.

The use of secondary controls was allowed in AFWAL-TR-81-3027. That is, the pilots were specifically instructed to utilize the centerstick to improve tracking if such control techniques seemed warranted. This was done for consistency with the real-world situation in which pilots might well use the wings-level-turn mode for fine tuning and the basic aircraft controls for gross maneuvering. Such control usage was found to conform to the normal pilot-centered requirements for separation of controls, i.e., only one control can be utilized at the primary closed-loop frequency, with all other controls limited to performing trimming-like functions. In the AFWAL-TR-81-3027 experiments the pilots utilized the centerstick any time the target bank angle appeared excessively large so that the DFC side force generators were approaching their limit. Such low-frequency secondary control usage was found to be entirely acceptable. However, attempts to utilize the secondary control to improve the tracking bandwidth of the primary DFC were unsuccessful.

VERIFICATION

Compliance can only be determined in flight test, since the necessary combination of visual and lateral acceleration cues cannot be obtained in a ground-based simulator.

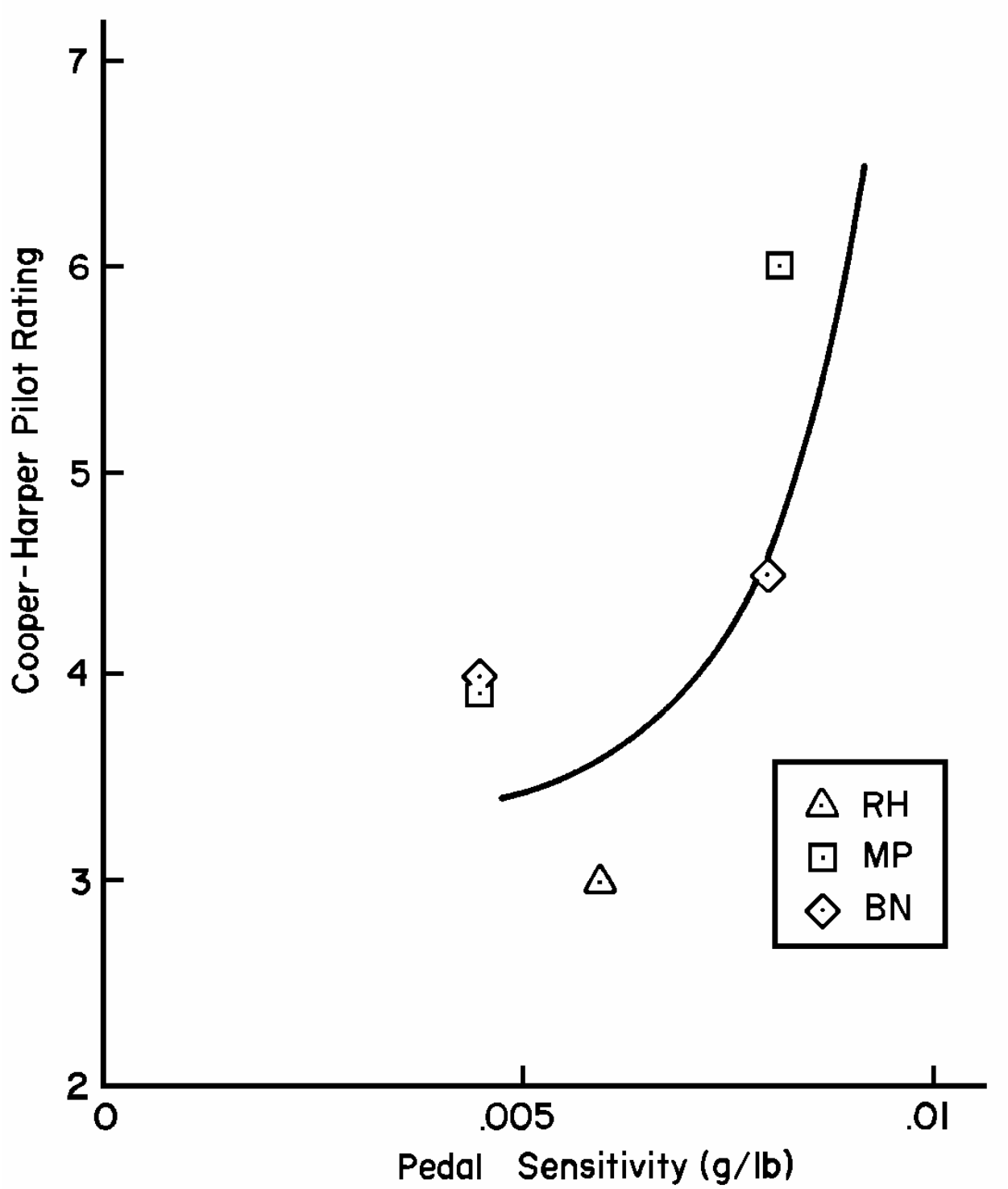


FIGURE 195. Effect of DFC manipulator sensitivity configuration WLT 5 (high favorable yaw coupling) (from AFWAL-TR-81-3027).

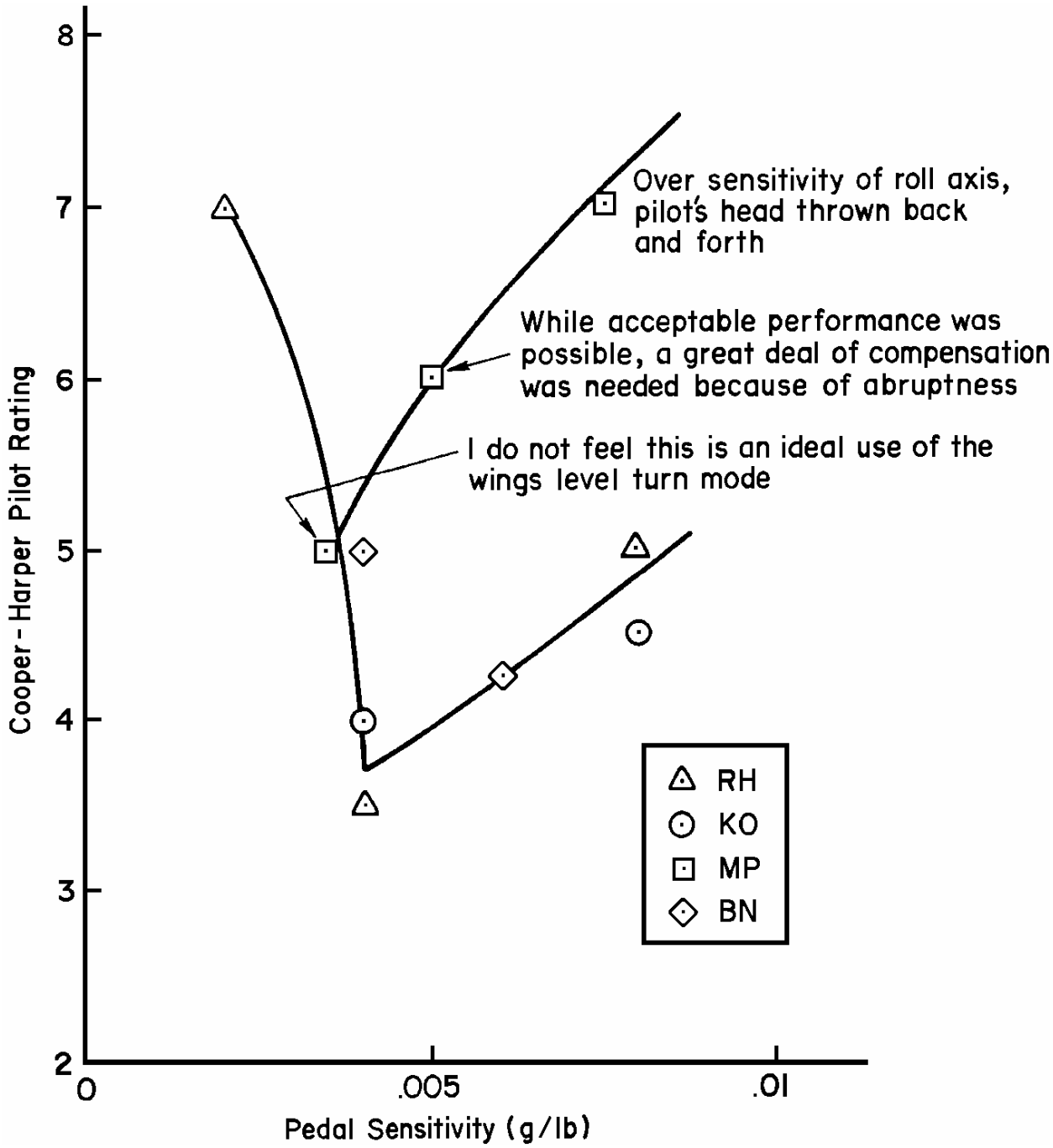


FIGURE 196. Effect of DFC manipulator sensitivity configuration WLT 12 (very high favorable roll coupling) (from AFWAL-TR-81-3027).

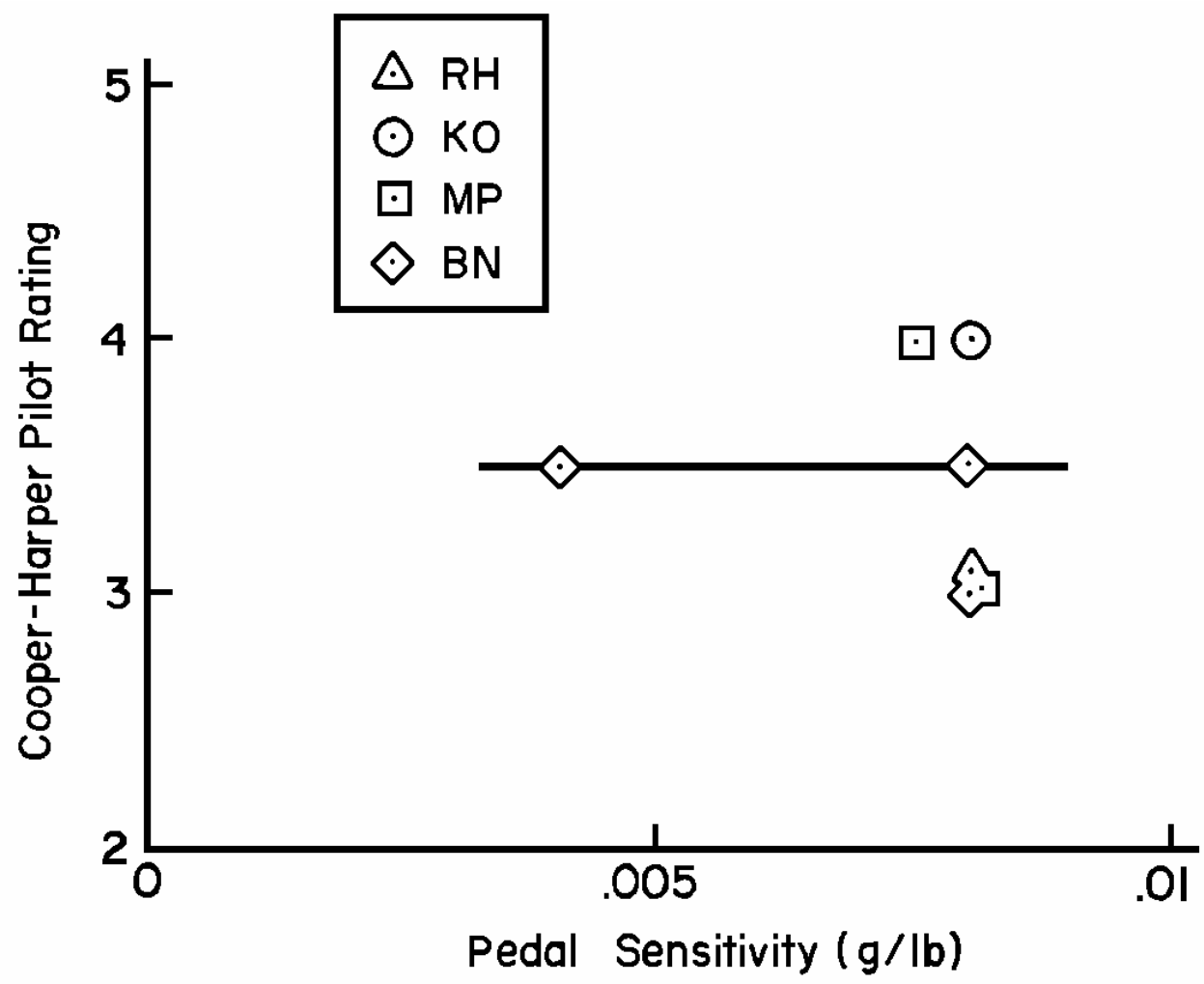


FIGURE 197. Effect of DFC manipulator sensitivity configuration WLT 1 (very low coupling) (from AFWAL-TR-81-3027).

5.2.3.12.4 Crew restraints.
Abrupt, large direct side force inputs shall not produce pilot head or arm motions which interfere with task performance. Crew restraints shall __(1)__.

REQUIREMENT RATIONALE

These requirements are included to specify the response characteristics of aircraft utilizing an extra DFC mode for wings-level turns.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.6.1.3.

Recommended value for Blank (1):

If DFC control power is sufficient to produce 0.5 lateral g or more: "be provided. The crew restraint shall not obstruct the crew's normal field of view, nor interfere with manipulation of any cockpit control required for task performance."

If DFC control power produces less than 0.5 lateral g: There is no requirement for crew restraint and the sentence shall be deleted.

REQUIREMENT LESSONS LEARNED

The variable-stability aircraft in the flight test experiment of AFWAL-TR-81-3027 was a Princeton University Navion, which has an operational speed of 105 kts. This resulted in lateral accelerations that were a factor of 5 lower than would occur at typical air combat speeds. However, recent AFAMRL centrifuge data (Investigation of the Effects of g_y and g_z on the AFTI/F-16 Control Inputs, Restraints, and Tracking Performance) indicate that pilots can track in a lateral acceleration environment of 2.5 g when properly constrained. Inasmuch as the Navion was limited to 0.5 g, it appears that this value represents the upper limit of pilot tolerance to lateral accelerations without side restraints.

VERIFICATION

Compliance can only be determined in flight test, since the necessary combination of visual and lateral acceleration cues cannot be obtained in a ground-based simulator.

5.2.3.13 Lateral translation.
For the lateral translation mode of control, the air vehicle response shall meet the requirements of 5.2.3.13.1 through 5.2.3.13.4.

REQUIREMENT RATIONALE

These requirements specify the useful range of response characteristics of aircraft that utilize direct force control (DFC) in the lateral translation mode, which changes aircraft lateral velocity at zero bank angle and constant heading.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.7.1.

Note that 5.2.3.13 through 5.2.3.13.4 apply only to air vehicles with a lateral translation mode of control.

REQUIREMENT LESSONS LEARNED

Figure 198 (taken from AIAA Paper 77-1119) gives a sketch of the airplane motions with a summary of the useful features of this mode.

VERIFICATION

Verification shall be via verification of requirements 5.2.3.13.1 through 5.2.3.13.4.

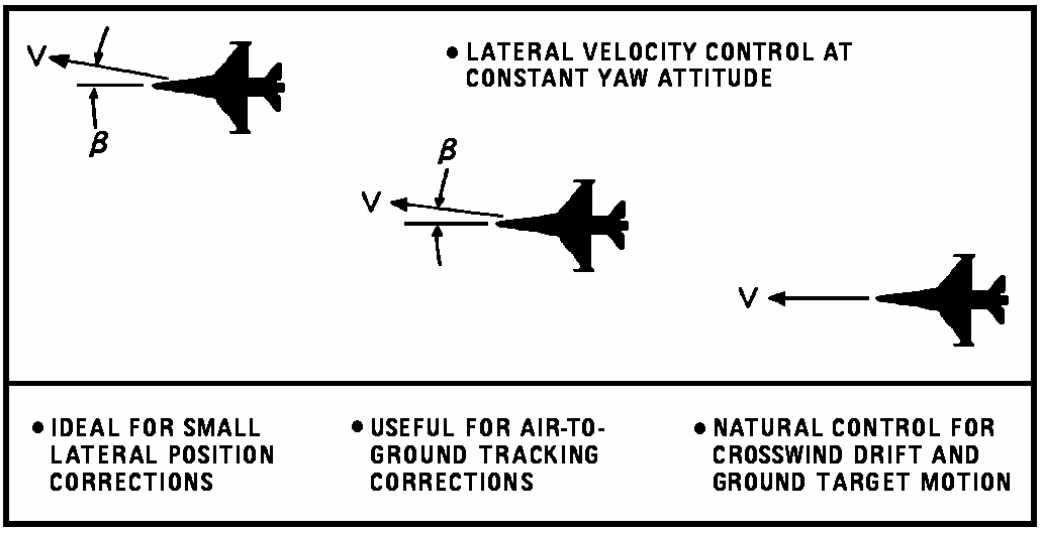


FIGURE 198. Lateral translation mode.

5.2.3.13.1 Dynamic response to lateral translation controller input.
Lateral translations shall occur at essentially zero bank angle and zero change in heading angle. Response to direct side force control inputs shall be acceptable to the crew in performing the mission tasks.

REQUIREMENT RATIONALE

These requirements specify the useful range of response characteristics of aircraft that utilize direct force control (DFC) in the lateral translation mode, which changes aircraft lateral velocity at zero bank angle and constant heading.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.7.1.

REQUIREMENT LESSONS LEARNED

There are no data of sufficient quality upon which to set a quantitative lower limit on lateral translation bandwidth. Until the required data become available, the qualitative requirement will have to suffice.

The only supporting data available are the F-16 CCV (AIAA Paper 77-1119) and two configurations flight tested in AFWAL-TR-81-3027. The AFWAL-TR-81-3027 are shown in table LXXVIII. Unfortunately, the control sensitivities were not optimized, which tended to obscure the results. Hence, it was not possible to define specific limits on bandwidth for the lateral translation mode. However, the results are presented here to provide some insight.

TABLE LXXVIII. Summary of Cooper-Harper pilot ratings for lateral translation mode.

Configuration	Bandwidth (rad/sec)	Formation				Air-to-Air			
		MP	WN	RH	KO	MP	WN	RH	KO
LT1	1.5	2.5	2.5		3	6	4	5	4
LT1Y	4.0	5			3.5		5	2.5	2.5

Configuration LT1Y was developed to test the bandwidth hypothesis by increasing the inherent bandwidth of Configuration LT1 via favorable yaw coupling. Unfortunately, the control sensitivities were not systematically varied for the LT1Y configuration. A review of the pilot comments indicated that the primary deficiency of the LT1Y mode was the jerky or abrupt nature of heading changes to CCV control inputs. Such comments are typical for aircraft with excessive control sensitivity, and the evaluation of Configuration LT1Y cannot be confidently

ascribed to its dynamics or compared directly with Configuration LT1. The scatter in pilot ratings for LT1Y in table LXXVIII is probably a measure of the degree to which each pilot objected to the excessive control sensitivity.

The F-16 CCV lateral translation mode was simply a decoupling of axes so that pure translation resulted from DFC inputs. From AFWAL-TR-81-3027 (page 13) the lateral velocity response for a perfectly decoupled aircraft is

$$\frac{v}{\delta_{DFC}} = \frac{Y_{\delta_{SF}}}{s - Y_v}$$

It follows that the basic response of the lateral translation mode is limited by the inverse time constant Y_v, which tends to be a small number, on the order of 0.2 to 0.3 for contemporary aircraft (0.25 for the F-16 at the test condition). Physically this means that even with perfect decoupling the lateral translation mode could require a special piloting technique due to a tendency for the aircraft to continue drifting laterally for a time upon release of the CCV control. An example of this is quoted below from AIAA Paper 77-1119 (CCV Flight No. 38-F16):

> A technique not previously evaluated using lateral translation involves reversing the command before the original side velocity had coasted to a stop, thereby providing increased deceleration to expedite the stop. This method of operation substantially improved the usefulness of the … mode. In previous evaluations of this mode the side velocity was allowed to coast to a stop after the applied command was removed.

The above pilot commentary indicates that the basic DFC response was unacceptably slow (low Y_v), but that a special piloting technique could be utilized to make the mode acceptable, that is, effectively generating lead to augment Y_v. It follows logically that a better lateral translation mode could be developed by augmenting Y_v via feedback of sideslip to the direct side force controller. This has implications on the frequency response characteristics of the servo drive as well as the authority required for the direct side force control. This F-16 CCV experience indicates that, in addition to decoupling, the lateral translation mode requires a β or $\dot{y}$ feedback to augment Y_v and thereby obtain the crisp response required to make this mode useful.

A Y_v of 0.25 results in a lateral position bandwidth of about 0.3. The above commentary indicates that this value of bandwidth corresponds more to Level 2 than to Level 1 flying qualities. Hence the minimum acceptable bandwidth (for the formation flying task) lies somewhere between 0.3 rad/sec and 1.5 rad/sec (see table LXXVIII). The rationale for using a bandwidth criterion for DFC modes is given in the discussion of 5.2.3.12. Procedures to be followed in application of the bandwidth criterion are given there.

VERIFICATION

Because of the scarcity of data upon which to base a limiting value of bandwidth for this mode, there are no firm values recommended. Until better data are obtained, the final demonstration of compliance with this requirement shall be by evaluation of operational tasks during flight test. In

fact, the data generated during such tests shall be utilized to upgrade the requirements. The critical conditions will depend upon the operational use which is to be made of the control mode. For a given sideslip angle, lateral acceleration increases with airspeed.

5.2.3.13.2 Steady-state response to lateral translation controller input.
Maximum control force input shall produce at least __(1)__ of sideslip at the flight conditions where lateral translation is to be utilized.

REQUIREMENT RATIONALE

These requirements specify the useful range of response characteristics of aircraft that utilize direct force control (DFC) in the lateral translation mode, which changes aircraft lateral velocity at zero bank angle and constant heading.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.7.1.

Recommended value for Blank (1): ±4º

REQUIREMENT LESSONS LEARNED

Analysis and simulation of task performance may yield a better-suited value than that recommended above.

VERIFICATION

The lateral acceleration produced by a given DFC deflection will vary with dynamic pressure, $\bar{q}$.

5.2.3.13.3 Lateral translation controller forces and displacements.
Lateral translation controller sensitivity shall be sufficiently high that satisfactory coordination can be achieved without unduly high control forces, yet sufficiently low that occasional, improperly-coordinated control inputs will not cause a degradation in flying qualities Level. Use of the primary lateral translation controller shall not require the use of another control manipulator to meet 5.2.3.13.1.

REQUIREMENT RATIONALE

These requirements specify the useful range of response characteristics of aircraft that utilize direct force control (DFC) in the lateral translation mode, which changes aircraft lateral velocity at zero bank angle and constant heading.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.7.1.

REQUIREMENT LESSONS LEARNED

If conventional cockpit controls are to be used as the DFC controller, the requirements for these controls provide some guidance; i.e., 5.2.3.8.

VERIFICATION

Compliance can only be determined in flight test, since the necessary combination of visual and lateral acceleration cues cannot be obtained in a ground-based simulator.

5.2.3.13.4 Crew restraints.
Abrupt, large lateral translation inputs shall not produce pilot head or arm motions which interfere with task performance. Crew restraints shall __(1)__.

REQUIREMENT RATIONALE

These requirements specify the useful range of response characteristics of aircraft that utilize direct force control (DFC) in the lateral translation mode, which changes aircraft lateral velocity at zero bank angle and constant heading.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.7.1.

Recommended value for Blank (1):

If lateral translation control power is sufficient to produce 0.5 lateral g or more: "be provided. The crew restraint shall not obstruct the crew's normal field of view, nor interfere with manipulation of any cockpit control required for task performance."

If lateral translation control power produces less than 0.5 lateral g: There is no requirement for crew restraint and the sentence shall be deleted.

REQUIREMENT LESSONS LEARNED

From the references cited in 5.2.3.13.1, 0.5 g is about the maximum tolerable lateral acceleration without special restraints.

VERIFICATION

Compliance can only be determined in flight test, since the necessary combination of visual and lateral acceleration cues cannot be obtained in a ground-based simulator.

5.2.4 Cross-axis responses.
The cross-axis responses of the air vehicle shall meet the requirements of 5.2.4.1 through 5.2.4.6.

REQUIREMENT RATIONALE

To keep from needlessly increasing pilot workload, controllers shall perform their functions without deleterious side effects serious enough to cause objections for Level 1 or 2, or a flight safety problem for Level 3.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C:
MIL-STD-1797: 4.1.11.8, 4.8.

REQUIREMENT LESSONS LEARNED

See Requirement Lessons Learned in 5.2.4.1 through 5.2.4.6.

VERIFICATION

Verification shall be via verification of requirements 5.2.4.1 through 5.2.4.6.

5.2.4.1 Longitudinal control forces in sideslips.
For Levels 1 and 2, with the air vehicle trimmed for straight, level flight with zero sideslip, the pitch control force required to maintain constant speed in steady sideslips with up to __(1)__ of pedal force in either direction shall not exceed the pitch control force that would result in a 1-g change in normal acceleration. In no case, however, shall the pitch control force exceed __(2)__ in pull or __(3)__ in push. If a variation of pitch control force with sideslip does exist, then increasing pull force shall accompany increasing sideslip, and the magnitude and direction of the force change shall be similar for right and left sideslips. For Level 3, there shall be no uncontrollable pitching motions associated with the sideslips described above.

REQUIREMENT RATIONALE

There are two primary reasons for having requirements for maximum longitudinal forces in sideslips: to insure that small amounts of sideslip inadvertently developed during normal operations do not result in large or possibly dangerous angle of attack changes; and to limit the longitudinal corrections required when the pilot intentionally changes the sideslip angle, as in a crosswind landing. This requirement limits the unwanted coupling of pitch to sideslip to an acceptable level. This keeps pilot workload in crosswinds (or in failures, such as one-engine-out operation on multi-engine aircraft) to a minimum.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.3.7
MIL-STD-1797: 4.2.8.6.4.

Recommended value for Blank (1): 50 lbs

Recommended value for Blank (2):	Centerstick controller:	10 lbs
	Wheel controller:	15 lbs
	Sidestick controller:	The forces must be acceptable to the pilot.

Recommended value for Blank (3):	Centerstick controller:	3 lbs
	Wheel controller:	10 lbs
	Sidestick controller:	The forces must be acceptable to the pilot.

REQUIREMENT LESSONS LEARNED

AFFDL-TR-72-141, using a Class III aircraft (P-3B), shows support for the pull limit of 15 lbs for wheel controllers.

The Level 3 requirement stipulates that pitching motions due to sideslip shall not further aggravate a Level 3 aircraft into an uncontrollable state. Uncontrollable is used here to indicate a

divergent pitch response and should not be interpreted to mean a mild uncommanded buffet or oscillation.

VERIFICATION

Flight test is needed to verify that all of the significant contributions have been accounted for: production asymmetries, engine gyroscopic moments, propeller slipstream, etc. Analysis and flight test shall encompass the extremes of the operational altitude range and the service flight speed range. The Level 3 requirement may be difficult to verify in practice, except in those cases where external failure conditions (e.g., one engine out) would create a Level 3 state.

5.2.4.2 Lateral-directional control in dives and pullouts.
Yaw and roll control power shall be adequate to maintain wings level and zero sideslip, without retrimming, throughout the dives and pullouts of 5.2.2.1.8.7 and 5.2.2.1.8.8. Within the ROTH, roll control forces shall not exceed __(1)__ and yaw control pedal forces shall not exceed __(2)__.

REQUIREMENT RATIONALE

For safety, roll and yaw control power must be adequate to perform any dive maneuvers, not only within the ROTH but also throughout the RORH. Excessive roll control forces in intended symmetric dives and pullouts increase the difficulty of maintaining symmetric flight. Excessive yaw in dives presents both control and structural problems, so any required rudder pedal forces must be within the pilot's capability.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.3.8
MIL-STD-1797: 4.5.8.5, 4.5.9.5.2, and 4.6.7.6.

Recommended values for blanks (1) and (2) are:
Blank (1): For propeller-driven air vehicles: 20 lbs
For other air vehicles: 10 lbs

Blank (2): For propeller-driven air vehicles: 180 lbs
For other air vehicles: 50 lbs.

REQUIREMENT LESSONS LEARNED

Slight geometric asymmetries may produce rolling, yawing, or hinge moments. F-100 aircraft were lost when the aerodynamically-operated automatic leading-edge slats stuck closed asymmetrically during pullouts from dive-bombing runs. Friction, binding, and sideslip were possible causes.

Transonic flow phenomena have been responsible for some asymmetric moments. Aeroelasticity tends to reduce roll control effectiveness at high dynamic pressure. Some other possible contributors to reduce roll effectiveness are compressibility effects, high hinge moments, and with a combined pitch/roll control surface, the pitch control input. Asymmetric loadings, either internal or external, will affect the roll control needed.

The control force requirements distinguish between propeller-driven and all other aircraft because of the normal torque and crossflow slipstream effects due to turning propellers. Ten lbs of roll control force can easily be held with one hand, and 20 lbs is manageable.

Although 5.2.3.6 limits the rudder pedal forces needed to maintain straight flight over the same speed range given here, pullouts may be a more severe test. This more relaxed requirement is intended to assure controllability in extreme cases. The limits recommended here seem unreasonably high for female pilots (see 5.2.3.10).

VERIFICATION

Verification shall be by analysis, simulation, and flight test. This is a flight safety item. Analysis and simulation shall proceed or accompany careful build-up to suspected critical flight conditions.

Verification will be in connection with the verification of 5.2.2.1.8.7 and 5.2.2.1.8.8. The qualitative nature of the first part of this requirement makes it straightforward to apply. Roll and yaw control forces shall be measured in flight during the required dives.

5.2.4.3 Cross-axis coupling in roll maneuvers.
In yaw-control-free, pitch-control-fixed maximum performance rolls through __(1)__ and rolls which are checked at a given bank angle, entered from straight flight or from turns, pushovers, or pullups ranging from 0g to $0.8n_L$, the resulting yaw or pitch motions and sideslip or angle of attack changes shall neither exceed structural limits nor cause other dangerous flight conditions such as uncontrollable motions or roll autorotation. During maneuvers involving rolls through angles up to __(2)__ and rolls which are checked at a given bank angle, the yawing and pitching shall not be so severe as to impair the tactical effectiveness of the maneuver. Rudder pedal inputs used to roll the air vehicle with roll control fixed, or when used in a coordinated manner with roll control inputs, shall not result in departures in pitch, roll, or yaw. These requirements define Levels 1 and 2.

REQUIREMENT RATIONALE

Both aerodynamic and inertial cross-coupling of pitch and yaw motions with rolling are common for modern air vehicles. The ensuing motions can be violent in nature, leading to prolonged loss of control.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.3
MIL-STD-1797: 4.8.1.

Blank (1) and blank (2):	Class I and Class IV:	360º (but not to exceed the structural roll angle limit applicable to the normal load factor)
	Class II and Class III:	120º (but not to exceed the structural roll angle limit applicable to the normal load factor)

REQUIREMENT LESSONS LEARNED

The roll angle restrictions mentioned in the recommended values for Blanks (1) and (2) are the structural design limits, as for example in JSSG-2006. This requirement should be coordinated with the structural requirements. In one instance, flying qualities predictions indicated good roll performance in 360º rolls at high load factor. However, the structural group did not know of this requirement and only designed the structure for 1-g, 360º rolls. It was pointed out that the 360º rolls at the high-g were required for various combat maneuvers.

The dynamics of the pitch and yaw coupling associated with rapid rolls are complex and nonlinear. In general, the dynamics involve interactions among the aircraft inertia properties, its aerodynamic properties and the kinematics of the rolling motion; coupling is more severe the more the mass is concentrated along the fuselage. The cross-axis coupling phenomena related to flight near or beyond stall are treated in 5.2.5.

It should be noted that inertial pitch/roll coupling may set the pitch control power requirements on the aircraft. This is especially true for Class IV aircraft, on which very high roll rates are common, or where reduced static stability is employed in the longitudinal axis, as with the F-16.

The so-called stability-axis yaw damper (in which body-centerline-axis r - p α is fed back to the rudder) reduces the adverse sideslip in high-angle-of-attack maneuvering, thus minimizing aerodynamic coupling; but thereby it increases roll rate and with it the p r and p^2 inertial coupling terms in the pitch axis, viz.

$$\dot{q} = \frac{I_{zz} - I_{xx}}{I_{yy}} p\, r + \frac{I_{xz}}{I_{yy}} (r^2 - p^2) + \frac{1}{I_{yy}} M_{aero}$$

Use of such augmentation requires a compromise between allowable sideslip and inertial coupling. Moreover, I_{xz} is generally significant in stability (flight-path-oriented in the steady state) axes but zero (by definition) in principal axes. Thus rolling about the flight path rather than the principal x axis influences another term to aggravate pitch coupling. Nevertheless, stability-axis rolling has been found helpful. So have increased directional stability and yaw damping. Analysis and simulation shall indicate the best combination of stability augmentation.

From figure 199, the inertial pitching moment of the mass element is

$$\Delta M = -(p^2\, z\, \Delta m)\, x = -p^2\, x\, z\, \Delta m$$

Integrating over the principal x axis, $M = -I_{xz}\, p^2$.

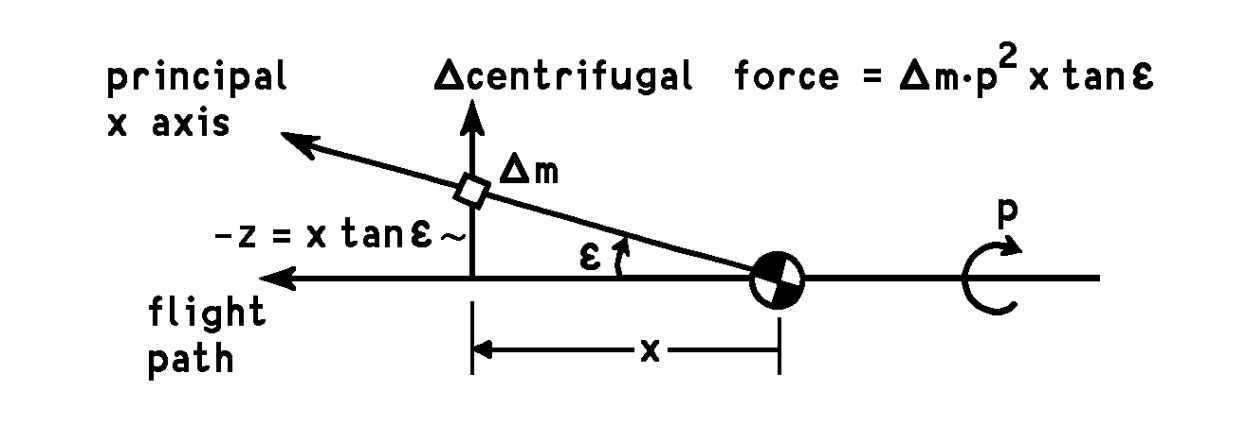


FIGURE 199. Orientation of the principal x axis relative to the flight path.

It has sometimes been difficult to meet both this requirement and the roll performance requirement, to the extent that a fix for inertial coupling adopted during the flight-test phase was to severely limit the roll control power. This, of course, has a detrimental effect on the fighter's air combat capability. Thus, early design attention is in order.

During prototype evaluation of the YF-16 lightweight fighter, coupled loss of control was encountered on two separate occasions. According to AFFTC-TR-75-15, "Lateral performance at low dynamic pressure was sufficiently high that roll and yaw rates could be generated which produced a nose-up pitching moment that could not be controlled by a full trailing-edge-down elevator."

AFFTC-TR-75-15 concludes:

> "The most significant conclusion and recommendation concerning the handling qualities of the YF-16 deal with coupling: A potential for loss of control due to inertial pitch/roll coupling was predicted after the completion of stabilizer saturation tests conduced during the High Angle of Attack test phase. The potential was later inadvertently demonstrated during the air combat maneuvering evaluation. The single spin of the flight test program was also coupling-related. Considering the production potential of the design, it is significant that: (1) two coupled departures were experienced during the prototype program, and (2) both occurred during controlled evaluations flown by highly qualified and experienced pilots. The deficiency represents a serious hazard to the safe operational use of the aircraft. The YF-16's potential for inertial pitch/roll-coupled departure should be eliminated even though its occurrence is associated with the outer portions of the useful flight envelope. The flexibility afforded by the electronic flight control system should be fully explored as an alternative to more complicated and costly means of correcting the deficiency. A reduction in the roll rate available to the pilot at high angles of attack should be considered. External aerodynamic configuration changes should be made to eliminate the potential for inertial pitch/roll coupling only if the deficiency cannot be corrected by modification of the flight control computer."

This last AFFTC recommendation must be taken as a concession, after the fact, to expediency. While the control laws can be changed more easily than the aerodynamic configuration there effectiveness here is severely limited by available control authority and rate (see 5.1.3 Requirement Lessons Learned). Also, increasing control law complexity generally creates additional phase lag which may cause piloting problems.

VERIFICATION

The nonlinear, violent manner of cross-axis coupling makes flight testing almost mandatory. However, ground simulation can give some insight into potential problem areas in the flight regime. It may be necessary to perform maneuvers with control combinations other than those specified to determine coupling problems. If the roll controller alone does not meet the control power requirements of 5.2.3.5 at high angle of attack, pilots may use supplemental rudder pedal deflection during air combat with Class IV aircraft; if pertinent, the pedal usage shall be evaluated. Also, as already noted, nose-down pitching may increase susceptibility to roll inertial coupling.

At least 5-degree-of-freedom nonlinear analysis (constant speed is generally a good assumption) is needed to portray the motions or the characteristics. A range of control deflections shall be

examined: aircraft-nose-down pitch control is commonly critical (see, for example, Dynamics of Atmospheric Flight). However, with the additional assumptions of a constant roll rate about the principal axis and negligible gravity, an approximation to the critical roll rate can be found more easily as a first cut. Near and beyond the critical roll rate, large pitch or yaw excursions (or acceleration to an extremely high roll rate) can be expected. A quadratic expression approximating the critical roll rate squared, p_{cr}^2, is

$$\left(\omega_{sp}^2 - \frac{b\left(k_z^2 - k_x^2\right)}{\bar{c}^2 k_y^2} p_{cr}^2\right)\left(\omega_d^2 - \frac{k_y^2 \bar{c}^2/b^2 - k_x^2}{k_z^2} p_{cr}^2\right)$$

$$+ \frac{g\rho V}{4(W/S)k_y^2 k_z^2}\left[C_{m_q} + 2\frac{b^2}{\bar{c}^2}\left(k_z^2 - k_x^2\right)C_{y_\beta}\right]\left[C_{n_r} - 2\left(k_y^2 \bar{c}^2/b^2 - k_x^2\right)\left(C_{L_\alpha} + C_D\right)\right]p_{cr}^2 = 0$$

in terms of the uncoupled approximations for ω_{sp}^2 and ω_d^2:

$$\omega_{sp}^2 \approx \frac{-g}{C_{L_1} k_y^2 \bar{c}}\left[\frac{C_{m_\alpha}}{C_{L_\alpha} + C_D} + \frac{g\rho\bar{c}}{4(W/S)}C_{m_q}\right]$$

$$\omega_d^2 \approx \frac{g}{C_{L_1} k_z^2 b}\left[C_{n_\beta} + \frac{g\rho b}{4(W/S)}C_{y_\beta}C_{n_r}\right]$$

For flight testing, a careful buildup with prior analysis and simulation is indicated, for the sake of safety.

5.2.4.4 Crosstalk between pitch and roll controllers.
The pitch- and roll-control force and displacement sensitivities and breakout forces shall be such that intentional inputs to one control axis will not cause objectionable inputs to the other.

REQUIREMENT RATIONALE

Force and displacements requirements for pitch and roll controllers are separately specified elsewhere, but their operation in combination through the same stick or column can cause problems if these characteristics are incompatible.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.4
MIL-STD-1797: 4.8.2.

REQUIREMENT LESSONS LEARNED

Control harmony has several aspects. One problem is that pitch and roll control forces must be in the proper ratio for gross unsymmetrical maneuvers, to enhance proper coordination of the maneuver. Another problem is that unless the pitch and roll control sensitivities and breakout forces are properly matched, intentional inputs to one control can result in inadvertent inputs to the other. For example, many heavy airplanes with unboosted controls have had aileron forces that were much too high with respect to the elevator forces. As a result, it was difficult to control pitch attitude accurately when rolling rapidly into a turn. In addition, it is often difficult to pull the centerstick straight back in Class IV highly-maneuverable aircraft due to arm and manipulator geometry and the lack of appropriate arm support. If lateral forces are low compared to longitudinal forces, some inadvertent lateral input is inevitable. The intent of this requirement is to prevent troublesome situations of that sort.

For design guidance, 5 percent cross-coupling has been suggested by one source as "normal and not objectionable." In the absence of data, though, the authors of this standard are reluctant to propose any specific hard limit.

This requirement may be of increasing significance with continued development of sidestick controllers. The force stick of the YF-16 was especially susceptible to longitudinal/lateral interactions (crosstalk), as described in AFFTC-TR-75-15:

> "…lateral versus longitudinal stick force crossplots and other quantitative data indicated that the YF-16's prototype force controller was susceptible to crosstalk during both classical evaluation and mission-oriented tasks. The pilots did not identify crosstalk as operationally significant, possibly because they subconsciously reacted to aircraft motion and modified their force inputs accordingly. Further development of the force controller should reflect that the stick's rotational orientation may not be optimum when aligned parallel with the

longitudinal and lateral axes of the aircraft. Preliminary quantitative data analysis suggests that the stick should be rotated clockwise (as seen from above) up to approximately 15º."

Later operational experience with the F-16 indicated that crosstalk was indeed a noticeable problem. For example, left banks were commonly observed during the pitch rotation for takeoff and landing. The F-16's later, moveable sidestick axis was rotated 12º clockwise, essentially eliminating crosstalk in takeoff and in landing.

VERIFICATION

Compliance with this requirement shall be shown through the course of normal simulation or flight testing; pilot comments and crossplots as mentioned above shall reveal any potential deficiencies.

5.2.4.5 Control harmony.
Operational maneuvers which require combined control inputs shall not require the pilot to apply simultaneous control forces greater than the magnitudes in table LXXIX.

TABLE LXXIX. Maximum simultaneous control forces.

Control Axis	Max Simultaneous Control Forces (lbs)
Pitch	
Roll	
Yaw	

REQUIREMENT RATIONALE

Normal maneuvering involving two or three controllers can be taxing, and precise maneuvering difficult, if any one controller requires large force inputs, even if each of the forces meets its single requirement. The cockpit control forces required to perform maneuvers which are normal for the aircraft shall have magnitudes which are related to the pilot's ability to produce such forces in combination. The pilot cannot apply forces simultaneously to all three controls that are as large as those forces that can be applied to one control at a time.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.4.1
MIL-STD-1797: 4.8.3.

Recommended values for table LXXIX:

Control Type	Pitch (lbs)	Roll (lbs)	Yaw (lbs)
Sidestick	20	15	
Centerstick	50	25	
Wheel (two-handed tasks)	75	40	
Wheel (one-handed tasks)	50	25	
Pedal			175

REQUIREMENT LESSONS LEARNED

The control forces recommended for table LXXIX are considered to be limiting values compatible with the pilot's ability to apply simultaneous forces. Larger simultaneous control forces shall not be required to perform any customary and expected maneuvers.

The 40 lbs allowed for wheel forces is a carryover from MIL-F-8785C. It is based on the use of two hands, a rare occurrence in most flying tasks since one hand is on the throttle(s) during maneuvering. The sidestick forces are based upon both the maximum forces on the F-16 moveable stick (AFFTC-TR-79-10) and results of the USAF Test Pilot School evaluations (AFFDL-TR-79-3126). The forces chosen are 75-90 percent of the maximum forces used.

Another very important aspect of control harmony, the relative timing of pilot forces and deflections of the several controls used to perform an operational maneuver, is hard to quantify. There are also the force and deflection relationships, such as in a turn entry, steady turns, and reversals. These aspects of control harmony are subjective in nature.

VERIFICATION

Maximum forces shall not exceed the combined forces specified herein when maneuvers such as dives, steady turns, and stalls are performed; and the test pilots shall evaluate the control harmony. The key to applying this requirement is in definition of normal maneuvers. Such maneuvers shall be all maneuvers required by the missions defined for the aircraft, including training.

5.2.4.6 Control cross-coupling.
No controller (continuous or discrete) which affects the motion of the aircraft shall create a secondary response which is objectionable (for Levels 1 and 2) or dangerous (for Level 3).

REQUIREMENT RATIONALE

To keep from needlessly increasing pilot workload, controllers shall perform their functions without deleterious side effects serious enough to cause objections for Levels 1 or 2, or a flight safety problem for Level 3.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.3, 3.4.11, and 3.6.3
MIL-STD-1797: 4.1.11.8.

REQUIREMENT LESSONS LEARNED

Operation of controllers intended for flight path or speed control shall not cause objectionable pitch response characteristics, and vice versa. (It is recognized, too, that some coupling in the right direction might actually be favorable. For example, a slight nose-up response to an increase in throttle would improve the short-term flight path response for STOL aircraft operating on the back side of the power-required curve; but while using the throttle for airspeed control, pitching up is the wrong direction of response to a thrust increase.)

Crossfeed and feedbacks in the stability and control augmentation system should generally reduce the severity of trim changes, if not eliminate them; but aerodynamic cross-coupling may vary greatly with angle of attack.

"Unique Flight Characteristics of the AD-1 Oblique-Wing Research Airplane" discusses control cross-coupling problems encountered on NASA's first oblique-wing airplane. In particular a pitch-roll coupling was noted. For even a symmetric airplane, spoiler deflection for roll control may produce disconcerting pitching; spoiler deflection as a direct lift control affects drag.

AIAA Paper 78-1500 recounts a B-1 design problem with its rolling/pitching horizontal stabilizer. While deflection limits can cause similar trouble, in this case limited hinge moment capability at high speed caused a roll command effectively to command pitch motion too. While for this case high dynamic pressure was critical, other aerodynamic and inertial coupling is often critical at high angle of attack, low dynamic pressure.

This general requirement is in addition to specific requirements, among them the ones on pilot-in-the-loop oscillations (5.2.1.6, 5.2.2.1.5, 5.2.3.3, and 5.2.3.4), response to failures (5.2.1.5 and 5.2.3.11), transients and trim changes, configuration or mode changes (5.2.8.1 and 5.2.9.2), pitch/flight path/airspeed interactions (5.2.2.1.2.3, 5.2.2.1.2.4, 5.2.2.1.3.3, and 5.2.2.1.6), lateral acceleration response to a roll command (5.2.3.2.4), roll control in steady sideslips (5.2.3.9), yaw

response to roll controller (5.2.3.2.5, 5.2.3.2.6, 5.2.3.2.7, and 5.2.3.8.4), yaw axis control in go-around (5.2.3.7), direct force control (5.2.3.12 and 5.2.3.13), and other general requirements of 5.1.3, 5.2.1, 5.2.8, and 5.2.9.

VERIFICATION

Any difficulties should become apparent during normal maneuvering. Also check Failure States and any control reconfigurations.

5.2.5 High angle of attack requirements.
Air Vehicle flying qualities at high angles of attack shall meet the requirements of 5.2.5.1 through 5.2.5.6.4.

REQUIREMENT RATIONALE

Requirements 5.2.5.1 through 5.2.5.6.4 concern stall warning, stalls, departures from controlled flight, post-stall gyrations, spins, recoveries, and related characteristics. They apply at speeds and AOAs which in general are outside the ROTH. They are intended to assure safety and the absence of mission limitations due to high-AOA flight characteristics.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.2
MIL-STD-1797: 4.8.4.

Note that the requirements apply as well to asymmetric loadings called out in the contract or experienced in normal operation.

REQUIREMENT LESSONS LEARNED

A large number of air vehicle incidents have been attributed to loss of control at high AOA. It is conjectured that many losses in Vietnam combat with no evidence to determine a cause might well have been due to loss of control at high AOA. Whereas previous requirements have concentrated on demonstration of acceptable stall and spin characteristics, these requirements emphasize prevention of loss of control (departure) as well. These requirements define flight demonstration maneuvers and control abuse appropriate to each Air Vehicle Class and mission. The requirements in this regime of nonlinearities remain largely qualitative.

The stall and spin requirements that follow are related by their application at high AOA outside the ROTH. Therefore, this requirement is retained to serve as an overview of characteristics and problems with high-AOA flight. The discussions presented in Verification Lessons Learned summarize recent insights and information on high-AOA flight, applicable in general to any of the stall/spin requirements. Based upon the requirements of this paragraph, high AOA is considered to be at and above the AOA for stall warning (5.2.5.2), generally outside the ROTH

in which the bulk of the other flying qualities requirements apply. Thus the AOA value which is considered high will vary with the situation: air vehicle, configuration, and Mach number.

Future advanced fighter air vehicles may have the capability to fly/maneuver in the post-stall region. This capability will exist through the use of improved high-AOA aerodynamics, digital flight control systems, and thrust-vectored control. Manned simulation studies indicate tactical utility and increased combat effectiveness available via high-AOA maneuvering. Consequently, continued flying qualities research is needed to establish stability and control requirements for flight operations in this region. Suggested areas to address are:

Definition of the post-stall region

Control power requirements to provide deep-stall recovery capability

Control power requirements to prevent departures from controlled flight

Engine operating requirements and means to fulfill them

Post-stall warning and pilot cues

Multi-axis, nonlinear dynamics at high-AOA, with good representation of the aerodynamics

Roll, pitch, and yaw rate capability (where rolling about the flightpath is mostly body-axis yawing)

Agility

Deceleration/acceleration capability (nobody wants to stay long in a state of very low energy)

Maximum allowable/usable sideslip and yaw rate at high AOA

Design criteria for departure resistance

Falling leaf prevention

Aerodynamic means to improve departure/post-stall characteristics, compatible with high-performance, low-observable, etc.

Thrust-vectoring control power requirements for high-AOA stabilization and control

Cockpit display and field of view requirements at high AOA.

These needs are listed to indicate the considerations necessary if an air vehicle is to be designed for effective post-stall flight. Data are lacking for more quantitative recommendations.

VERIFICATION

A high-AOA flight test program (a) is necessary to bring out under controlled conditions any idiosyncrasies of the air vehicle that might be encountered later in service use, and (b) needs careful preparation, including prior analysis and model testing, provision of emergency recovery means, propulsion system modification such as continuous ignition, backup hydraulic power, etc.

Verification shall be by demonstration in analysis, simulation, and flight test. The air vehicle shall demonstrate compliance with the requirements of 5.2.5.1 through 5.2.5.6.4 by flight test according to table LXXX. Reasonable delayed recovery attempts after a stall or departure, and exaggerated misapplication of controls following a stall or departure, to simulate possible incorrect pilot responses shall be investigated under the least conservative circumstances to ascertain the degree of spin susceptibility/resistance for operational users. When spins result as a natural consequence of testing through departures from controlled flight or as a result of deliberate spin attempts, a satisfactory spin recovery technique shall be demonstrated. The use of prolonged pro-spin control to sustain a developed spinning condition is __(1)__.

TABLE LXXX. Flight demonstration maneuvers for air vehicles.

Test Phase	Flight Phase Category	Control Application	Maneuver Requirements

For Class II and III air vehicles the following would be applicable to blank (1):

> The use of prolonged pro-spin control to sustain a developed spinning condition is not required.

For most Class I and Class IV air vehicles the following would be applicable to blank (1):

> The use of prolonged pro-spin control to sustain a developed spinning condition is required for no more than 3 turns.

For trainer-type air vehicles to be cleared for intentional spins, and for Class I and Class IV air vehicles in which sufficient departures and developed spins do not result during Test Phases A, B, and C, the following would be applicable to blank (1):

> The use of prolonged pro-spin control to sustain a developed spinning condition is required for 15 sec or until 3 turns of the fully-developed spin has been encountered, whichever occurs first.

Recommended values for table LXXX follow.

For Class I air vehicles which do not employ AOA limiting devices:

Test Phase	Control Application	Flight Phase Category	Maneuver Requirements
A Stalls	Pitch control applied to achieve the specified AOA rate. Roll, yaw, and decoupled[1] control inputs as normally required for the maneuver task. Recovery initiated after the pilot has a clear indication of: a) a definite g break, or b) a rapid, uncommanded angular motion, or c) the aft stick stop has been reached and AOA is not increasing, or d) sustained intolerable buffet.	A and B	Entry conditions[2]: 1) One-g stall with smooth AOA rate[3] 2) Accelerated[4] stall with smooth AOA rate 3) One-g stall with abrupt AOA rate[5] of at least 4°/sec 4) Accelerated stall with abrupt AOA rate of at least 4°/sec 5) Tactical[6]
	For those air vehicles where clear indications of stall are not evident and where the minimum permissible speed is other than the stall speed, recovery may be initiated somewhat beyond the arbitrary limit(s).	C	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate
B Stalls with aggravated control inputs	Pitch control applied to achieve the specified AOA rate. Roll, yaw, and decoupled control inputs as normally required for the maneuver task. When conditions a), b), c), or d) from above have been attained, controls briefly misapplied[7] intentionally or in response to unscheduled air vehicle motions, before recovery is initiated.	A and B	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate 3) One-g stall with abrupt AOA rate of at least 4°/sec 4) Accelerated stall with abrupt AOA rate of at least 4°/sec 5) Tactical
		C	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate
C Stalls with aggravated and sustained control inputs[8]	Pitch control applied to achieve the specified AOA rate. Roll, yaw, and decoupled control inputs as normally required for the maneuver task. When conditions a), b), c), or d) from above have been attained, controls are misapplied[7,9] intentionally or in response to unscheduled air vehicle motions, and held for 3 sec[9,10] before recovery attempt is initiated	A and B	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate 3) One-g stall with abrupt AOA rate of at least 4°/sec 4) Accelerated stall with abrupt AOA rate of at least 4°/sec 5) Tactical
D Post-Stall Gyration, Spin and Deep Stall Attempts[8] (This phase required only for training air vehicles which may be intentionally spun and for air vehicles in which sufficient departures and developed spins did not result during Test Phases A, B, and C to define characteristics of each possible out-of-control mode)	Pitch control applied to achieve the specified AOA rate. Roll, yaw, and decoupled control inputs as normally required for the maneuver task. When conditions a), b), c), or d) from above have been attained, controls applied in the most critical[11] manner to attain each possible mode of post-stall motion, and held for various lengths of time up to 15 sec or 3 fully-developed spin turns, whichever occurs first, before the recovery attempt is initiated.[9,12]	A and B	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate 3) One-g stall with abrupt AOA rate of at least 4°/sec 4) Accelerated stall with abrupt AOA rate of at least 4°/sec 5) Tactical

For Class II air vehicles which do not employ AOA limiting devices:

Test Phase	Control Application	Flight Phase Category	Maneuver Requirements
A Stalls	Pitch control applied to achieve the specified AOA rate. Roll, yaw, and decoupled[1] control inputs as normally required for the maneuver task. Recovery initiated after the pilot has a clear indication of: a) a definite g break, or b) a rapid, uncommanded angular motion, or c) the aft stick stop has been reached and AOA is not increasing, or d) sustained intolerable buffet. For those air vehicles where clear indications of stall are not evident and where the minimum permissible speed is other than the stall speed, recovery may be initiated somewhat beyond the arbitrary limit(s).	A and B	Entry conditions[2]: 1) One-g stall with smooth AOA rate[3] 2) Accelerated[4] stall with smooth AOA rate 3) One-g stall with abrupt AOA rate[5] of at least 2^{o}/sec 4) Accelerated stall with abrupt AOA rate of at least 2^{o}/sec 5) Tactical[6]
		C	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate
B Stalls with aggravated control inputs	Pitch control applied to achieve the specified AOA rate. Roll, yaw, and decoupled control inputs as normally required for the maneuver task. When conditions a), b), c), or d) from above have been attained, controls briefly misapplied[7], intentionally or in response to unscheduled air vehicle motions, before recovery is initiated.	A and B	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate 3) One-g stall with abrupt AOA rate of at least 2^{o}/sec
		C	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate
C Stalls with aggravated and sustained control inputs[8]	Pitch control applied to achieve the specified AOA rate. Roll, yaw, and decoupled control inputs as normally required for the maneuver task. When conditions a), b), c), or d) from above have been attained, controls are misapplied[7,9], intentionally or in response to unscheduled air vehicle motions, and held for 3 sec[9,10] before recovery attempt is initiated.	A and B	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate 3) One-g stall with abrupt AOA rate of at least 2^{o}/sec

For Class III air vehicles which do not employ AOA limiting devices:

Test Phase	Control Application	Flight Phase Category	Maneuver Requirements
A Stalls	Pitch control applied to achieve the specified AOA rate. Roll, yaw, and decoupled[1] control inputs as normally required for the maneuver task. Recovery initiated after the pilot has a clear indication of: a) a definite g break, or b) a rapid, uncommanded angular motion, or c) the aft stick stop has been reached and AOA is not increasing, or d) sustained intolerable buffet. For those air vehicles where clear indications of stall are not evident and where the minimum permissible speed is other than the stall speed, recovery may be initiated somewhat beyond the arbitrary limit(s).	A and B	Entry conditions[2]: 1) One-g stall with smooth AOA rate[3] 2) Accelerated[4] stall with smooth AOA rate 3) One-g stall with abrupt AOA rate[5] of at least 1^{o}/sec 4) Accelerated stall with abrupt AOA rate of at least 1^{o}/sec 5) Tactical[6]
		C	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate
B Stalls with aggravated control inputs	Pitch control applied to achieve the specified AOA rate. Roll, yaw, and decoupled control inputs as normally required for the maneuver task. When conditions a), b), c), or d) from above have been attained, controls briefly misapplied[7], intentionally or in response to unscheduled air vehicle motions, before recovery is initiated.	A and B	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate 3) One-g stall with abrupt AOA rate of at least 1^{o}/sec
		C	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate

For Class IV air vehicles which do not employ AOA limiting devices:

Test Phase	Control Application	Flight Phase Category	Maneuver Requirements
A Stalls	Pitch control applied to achieve the specified AOA rate. Roll, yaw, and decoupled[1] control inputs as normally required for the maneuver task. Recovery initiated after the pilot has a clear indication of: a) a definite g break, or b) a rapid, uncommanded angular motion, or c) the aft stick stop has been reached and AOA is not increasing, or d) sustained intolerable buffet. For those air vehicles where clear indications of stall are not evident and where the minimum permissible speed is other than the stall speed, recovery may be initiated somewhat beyond the arbitrary limit(s).	A and B	Entry conditions[2]: 1) One-g stall with smooth AOA rate[3] 2) Accelerated[4] stall with smooth AOA rate 3) One-g stall with abrupt AOA rate[5] of at least 8°/sec 4) Accelerated stall with abrupt AOA rate of at least 8°/sec 5) Tactical[6]
		C	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate
B Stalls with aggravated control inputs	Pitch control applied to achieve the specified AOA rate. Roll, yaw, and decoupled control inputs as normally required for the maneuver task. When conditions a), b), c), or d) from above have been attained, controls briefly misapplied[7], intentionally or in response to unscheduled air vehicle motions, before recovery is initiated.	A and B	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate 3) One-g stall with abrupt AOA rate of at least 8°/sec 4) Accelerated stall with abrupt AOA rate of at least 8°/sec 5) Tactical
		C	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate
C Stalls with aggravated and sustained control inputs[8]	Pitch control applied to achieve the specified AOA rate. Roll, yaw, and decoupled control inputs as normally required for the maneuver task. When conditions a), b), c), or d) from above have been attained, controls are misapplied[7,9], intentionally or in response to unscheduled air vehicle motions, and held for 3 sec[9,10] before recovery attempt is initiated.	A and B	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate 3) One-g stall with abrupt AOA rate of at least 8°/sec 4) Accelerated stall with abrupt AOA rate of at least 8°/sec 5) Tactical
D Post-Stall Gyration, Spin and Deep Stall Attempts[8] (This phase required only for training air vehicles which may be intentionally spun and for air vehicles in which sufficient departures and developed spins did not result during Test Phases A, B, and C to define characteristics of each possible out-of-control mode)	Pitch control applied to achieve the specified AOA rate. Roll, yaw, and decoupled control inputs as normally required for the maneuver task. When conditions a), b), c), or d) from above have been attained, controls applied in the most critical[11] manner to attain each possible mode of post-stall motion, and held for various lengths of time up to 15 sec or 3 fully-developed spin turns, whichever occurs first, before the recovery attempt is initiated.[9,12]	A and B	Entry conditions: 1) One-g stall with smooth AOA rate 2) Accelerated stall with smooth AOA rate 3) One-g stall with abrupt AOA rate of at least 8°/sec 4) Accelerated stall with abrupt AOA rate of at least 8°/sec 5) Tactical

For Class I and Class IV air vehicles which employ AOA limiting devices:

Test Phase	Control Application	Flight Phase Category	Maneuver Requirements[2]
A	Longitudinal maneuvers to the limiter AOA ranging from 1-g decelerations to maximum-g decelerations and wind-up turns and pushover-pullups to the limiter AOA over the airspeed range between the minimum sustainable speed and maximum level flight speed. Recovery attempts shall be initiated immediately after departure occurs.	A, B, and C	1-g decelerations to the limiter AOA will be performed using a slow control rate which will produce a speed deceleration of approximately 1 kt/sec. AOA rate during accelerated maneuvers and pushover-pullups to the limiter AOA will be increased in a build-up fashion until the maximum attainable AOA rate or the maximum suitable AOA rate (if considered less than maximum attainable) is demonstrated.
B	Combinations of pitch, roll, yaw, and decoupled[1] controls applied while the air vehicle is at or near limiter AOA outside the airspeed range between minimum sustainable speed and maximum level flight speed. These maneuvers include roll and sideslips at limiter AOA. Recovery attempts shall be initiated immediately after departure occurs.	A, B, and C	The control inputs shall consist of applying controls in the most critical directions, combinations, and rates of application. AOA rate during accelerated maneuvers and pushover-pullups to the limiter AOA will be increased in a build-up fashion until the maximum attainable AOA rate or the maximum suitable AOA rate (if considered less than maximum attainable) is demonstrated. Controls shall be applied below limiter AOA if predictions indicate the possibility of inducing a departure. This includes combinations of roll, yaw, and decoupled control inputs below limiter AOA while AOA is increasing both at smooth and abrupt rates. The air vehicle will not be required to roll in excess of 360^{o}.
C	Combinations of pitch, roll, yaw, and decoupled controls applied while the air vehicle is at or near limiter AOA outside the airspeed range between minimum sustainable speed and maximum level flight speed. These maneuvers include high pitch attitude, low airspeed recoveries and high-speed dive pullouts. Recovery attempts shall be initiated immediately after departure occurs.[8]	A and B	The control inputs shall consist of applying controls in the most critical directions, combinations, and rates of application. AOA rate during accelerated maneuvers and pushover-pullups to the limiter AOA will be increased in a build-up fashion until the maximum attainable AOA rate or the maximum suitable AOA rate (if considered less than maximum attainable) is demonstrated. Controls shall be applied below limiter AOA if predictions indicate the possibility of inducing a departure. This includes combinations of roll, yaw, and decoupled control inputs below limiter AOA while AOA is increasing both at smooth and abrupt rates. The air vehicle will not be required to roll in excess of 360^{o}.
D	Combinations of pitch, roll, yaw, and decoupled controls of a gross and abnormal nature not likely to occur during operational use of the air vehicle.[8,11,12] This includes deliberate out-of-control events held for various lengths of time up to 15 sec or 3 spin turns, whichever is longer, to demonstrate out-of-control recovery.	A and B	The control inputs shall consist of applying controls in the most critical directions, combinations, and rates of application.

For Class II air vehicles which employ AOA limiting devices:

Test Phase	Control Application	Flight Phase Category	Maneuver Requirements[2]
A	Longitudinal maneuvers to the limiter AOA ranging from 1-g decelerations to maximum-g decelerations and wind-up turns and pushover-pullups to the limiter AOA over the airspeed range between the minimum sustainable speed and maximum level flight speed. Recovery attempts shall be initiated immediately after departure occurs.	A, B, and C	1-g decelerations to the limiter AOA will be performed using a slow control rate which will produce a speed deceleration of approximately 1 kt/sec AOA rate during accelerated maneuvers and pushover-pullups to the limiter AOA will be increased in a build-up fashion until the maximum attainable AOA rate or the maximum suitable AOA rate (if considered less than maximum attainable) is demonstrated.
B	Combinations of pitch, roll, yaw, and decoupled[1] controls applied while the air vehicle is at or near limiter AOA outside the airspeed range between minimum sustainable speed and maximum level flight speed. These maneuvers include roll and sideslips at limiter AOA. Recovery attempts shall be initiated immediately after departure occurs.	A, B, and C	The control inputs shall consist of applying controls in the most critical directions, combinations, and rates of application. AOA rate during accelerated maneuvers and pushover-pullups to the limiter AOA will be increased in a build-up fashion until the maximum attainable AOA rate or the maximum suitable AOA rate (if considered less than maximum attainable) is demonstrated. Controls shall be applied below limiter AOA if predictions indicate the possibility of inducing a departure. This includes combinations of roll, yaw, and decoupled control inputs below limiter AOA while AOA is increasing both at smooth and abrupt rates. The air vehicle will not be required to roll in excess of 120^{o}.
C	Combinations of pitch, roll, yaw, and decoupled controls applied while the air vehicle is at or near limiter AOA outside the airspeed range between minimum sustainable speed and maximum level flight speed. These maneuvers include high pitch attitude, low airspeed recoveries and high-speed dive pullouts. Recovery attempts shall be initiated immediately after departure occurs.[8]	A and B	The control inputs shall consist of applying controls in the most critical directions, combinations, and rates of application. AOA rate during accelerated maneuvers and pushover-pullups to the limiter AOA will be increased in a build-up fashion until the maximum attainable AOA rate or the maximum suitable AOA rate (if considered less than maximum attainable) is demonstrated. Controls shall be applied below limiter AOA if predictions indicate the possibility of inducing a departure. This includes combinations of roll, yaw, and decoupled control inputs below limiter AOA while AOA is increasing both at smooth and abrupt rates. The air vehicle will not be required to roll in excess of 120^{o}.

For Class III air vehicles which employ AOA limiting devices:

Test Phase	Control Application	Flight Phase Category	Maneuver Requirements[2]
A	Longitudinal maneuvers to the limiter AOA ranging from 1-g decelerations to maximum-g decelerations and wind-up turns and pushover-pullups to the limiter AOA over the airspeed range between the minimum sustainable speed and maximum level flight speed. Recovery attempts shall be initiated immediately after departure occurs.	A, B, and C	1-g decelerations to the limiter AOA will be performed using a slow control rate which will produce a speed deceleration of approximately 1 kt/sec. AOA rate during accelerated maneuvers and pushover-pullups to the limiter AOA will be increased in a build-up fashion until the maximum attainable AOA rate or the maximum suitable AOA rate (if considered less than maximum attainable) is demonstrated.
B	Combinations of pitch, roll, yaw, and decoupled[1] controls applied while the air vehicle is at or near limiter AOA outside the airspeed range between minimum sustainable speed and maximum level flight speed. These maneuvers include roll and sideslips at limiter AOA. Recovery attempts shall be initiated immediately after departure occurs.	A, B, and C	The control inputs shall consist of applying controls in the most critical directions, combinations, and rates of application. AOA rate during accelerated maneuvers and pushover-pullups to the limiter AOA will be increased in a build-up fashion until the maximum attainable AOA rate or the maximum suitable AOA rate (if considered less than maximum attainable) is demonstrated. Controls shall be applied below limiter AOA if predictions indicate the possibility of inducing a departure. This includes combinations of roll, yaw, and decoupled control inputs below limiter AOA while AOA is increasing both at smooth and abrupt rates. The air vehicle will not be required to roll in excess of 120^{o}.

[1] Decoupled controls are defined as unconventional controls such as direct normal force, direct side force, pitch pointing, yaw pointing, vertical translation, lateral translation, and flightpath control using thrust vectoring (excluding V/STOL flight).

[2] Establish the ranges and increments of the following variables to be tested for their influence on high-AOA flight characteristics:

a. Configuration

b. Gross weight

c. Center of gravity

d. Flight control system status

e. Loadings, both with internal and with external stores; critical combinations of aerodynamic and inertial loadings to include:

1) Symmetric, fuselage heavy

2) Symmetric, wing heavy

3) Asymmetric (maximum allowable asymmetry)

4) Any other loadings found critical in preliminary test and analysis

f. Entry speed, altitude, and attitude

g. Thrust and engine gyroscopic effects.

Power settings shall include:

a. Takeoff (TO) configuration:

1) All engines at TO thrust

2) Critical engine inoperative, others at TO thrust (stall approach, Test Phase A only)

b. Power approach (PA) configuration:

1) All engines at normal approach thrust

2) Critical engine inoperative, others at required approach thrust

c. Climb (C) configuration:

1) All engines at normal climb thrust

2) Critical engine inoperative, others at normal climb thrust

d. Cruise (CR) configuration:

1) All engines at thrust for level flight (TLF)

2) All engines at idle thrust

e. Combat (CO) configuration:

1) All engines at military rated thrust (MRT)

2) All engines at maximum augmented thrust (MAT).

Throttle settings for those cases where flameouts or compressor stalls occur shall include:

a. Throttle retarded to idle from the maneuver entry setting position for a malfunctioning engine (for MAT, MRT, TLF)

b. Throttle left at the entry setting position until the entry has been accomplished (for MAT, MRT, TLF) unless compliance would result in exceeding engine operating limitations.

Flight control system configurations shall include:

a. All modes (normal or degraded) of flight control systems that have a reasonable probability of being engaged during or previous to flight at high AOA

b. Degraded, reversion, reconfiguration, and backup modes that can be engaged during flight at high AOA.

The test air vehicles shall be configured so that these modes can be safely engaged during flight test.

The air vehicles shall be trimmed [controls and throttle(s)] at settings consistent with the maneuver tasks. The effects of each designated flight test variable shall be determined individually in each required Test Phase or until such effects are definitely established and predictable for succeeding Test Phases. Variables need to be tested in combination only when that variable could possibly yield less conservative results from those obtained by individual testing.

[3] Smooth, 1-g entries shall be approached using a slow control rate which would produce a speed deceleration of approximately 1 kt/sec for normal stalls (1g). Smooth, accelerated entries shall be approached using a control rate to achieve an AOA rate of approximately .5º/sec.

[4] Accelerated entries, encompassing a representative range of Mach number, dynamic pressure, and allowable load factor, shall include wind-up turns, constant altitude turns, and wings-level pullouts from dives appropriate to the Air Vehicle Class and mission.

[5] In the required abrupt entries, the entry AOA rate may be limited by maximum control deflections and rate. The magnitude of the abrupt entry rates may be graduated in Test Phases A through C, commensurate with the increasing severity of control requirements, but the stated minimum AOA rates shall be achieved in Test Phase C.

[6] These entries shall be initiated from offensive/defensive, ground attack, or other tactical maneuvers associated with the capability and Class of the air vehicle. The maneuvers, conducted with a suitable AOA rate, may include:

a. Inverted stalls and aborted vertical reversements, loops, or Immelmans to investigate inverted out-of-control events

b. High-AOA turn reversals with roll control only, with coordination attempted, and with yaw control only

c. High pitch attitudes (greater than 45º)

d. Head-out-of-cockpit air combat maneuvering or ground attack maneuvering

e. High-g, supersonic turns/transonic decelerations

f. Sudden idle power/speed brake decelerations

g. Sudden asymmetric thrust transients prior to stall.

[7] Misapplied controls shall consist of moving controls in the most critical directions an amount significantly greater than that expected during operational use. This should generally require somewhat less than full deflection depending upon the mission and expected pilot reactions.

[8] In addition to the demonstration of satisfactory spin, out-of-control, and deep stall recovery procedures, the effects of both premature and delayed application of these recovery procedures shall be investigated during the final phase of testing.

[9] The test pilot shall insure that routine familiarity with stalls, post-stall gyrations, and spins does not negate the intent of the delay/misapplication simulation and does not result in premature application of spin recovery controls before a developed spin has been attained (as subsequently confirmed by the flight records, when necessary).

[10] This time requirement may be increased for air vehicles that do not exhibit a clear indication to the pilot of impending loss of control.

[11] With respect to spin attempts, "critical" control positions shall include, but not be necessarily restricted to, pro-spin settings. For some combinations of air vehicle state and entry test variables, the spinning motion may be sustained with controls in positions (neutral, out-of-control recovery settings, or stick forward, for example) other than full pro-spin positions. A recovery attempt with controls displaced from the former positions may result in recovery capability, duration, or reversal tendency materially different from that which would occur if recovery were initiated from the full pro-spin condition. The possibility of reversal or secondary stall shall be investigated by holding full recovery control for a brief period after recovery is attained. If it appears possible to encounter these circumstances in service use, the "critical" controls may be any setting necessary to define out-of-control modes and determine recovery characteristics specifically applicable to operational users.

[12] For trainer air vehicles, recovery shall also be demonstrated from a fully-developed spin if such a spin is attainable within a limited number of turns after spin entry.

The following approach shall be used to achieve the verification objectives:

a. Before flight test, analysis and simulation shall be used to investigate predicted high-AOA flight characteristics for a range of configurations, maneuvers, control inputs, and flight conditions. An initial flight test plan shall be developed from the results of this study.

b. The flight test program shall integrate flight demonstration, analysis, and simulation in a build-up approach to critical test points. At each step, analysis and simulation shall be used to investigate predicted air vehicle response. Data from flight demonstration shall be used to validate or update the analytical and simulation models.

Requirements 5.2.5.1 through 5.2.5.6.4 apply at speeds and AOAs which in general are outside the ROTH. These requirements apply for all Air Vehicle Normal States and Extreme States in straight unaccelerated flight and in turns and pullups with attainable normal accelerations up to n_L. These requirements also apply to Air Vehicle Failure States that affect stall characteristics. Requirements 5.2.5.2 through 5.2.5.4.3 and 5.2.5.6.4 apply for all stalls, including stalls entered abruptly.

A full range of internal and external loadings shall be tested. If modifications such as a spin chute or a flight test nose boom might change the aerodynamic characteristics, some testing might have to be repeated in the service configuration. AFFDL-TR-65-218, among others, indicates ranges of some of the critical inertial parameters that generally give certain spin and recovery characteristics. Cancelled specification MIL-F-83691 (for the Air Force) and MIL-D-8708 (for the Navy) give further guidance.

A critical design review of the departure and spin characteristics shall be performed using pilots who represent the contractor and the procuring activity in a manned simulation program. Prior to CDR, the contractor shall develop an evaluation plan that:

a. Indicates the range of air vehicle gross weight and c.g. positions, and Air Vehicle Normal, Extreme, and Failure States associated with each flight phase.

b. Specifies maneuvers and control inputs to be evaluated for each Flight Phase. The control inputs evaluated shall be broadly classed into four techniques:

Technique (1): Ordinary control inputs

Technique (2): Misapplied control inputs

Technique (3): Consecutive misapplied control inputs

Technique (4): Pro-spin control inputs (optional)

The plan shall be tailored to the Class and structural design criteria of the air vehicle. The cancelled MIL-F-83691 gives guidance. Aerodynamic data shall be of sufficient quality and quantity to recognize at least the initial characteristics of divergence and spin. The piloted simulation shall address the maneuvers associated with each Flight Phase, with some extra simulation time allotted to the pilots to allow them to evaluate entry maneuvers not covered by the plan.

Prior to flight test evaluation of departure and spin characteristics, installation of a recovery system on the flight test air vehicle is recommended to allow for recovery from out-of-control flight and to permit the air vehicle to be landed safely with the engines inoperable.

Typical emergency recovery systems used during high angle of attack and out-of-control flight testing, unless demonstrated to be unwarranted due to inherent system design and redundancy, include a spin recovery system, emergency battery electrical power system, emergency hydraulic power system, an emergency fuel boost pump, continuous engine ignition capability, APU scoop, liquid oxygen breathing system, and an enhanced crew restraint system. The spin recovery system shall be capable of recovering the air vehicle from a spin or other out-of-control flight condition if normal techniques are not effective in recovering the air vehicle to a controlled condition. The recovery system shall be installed such that it does not snag on the control surfaces, regardless of control surface deflection during or after deployment jettison. The essential cockpit instruments, displays, and controls, including radio communications, landing gear control, and emergency stores jettison functions shall remain functional through use of emergency backup systems. Fuel pressure and APU pressure recovery must be adequate to achieve engine airstart at altitude. A reliable aircrew breathing system that is not dependent on engine bleed air or air vehicle electrical power shall be available. Additional pilot restraint that does not adversely affect ejection seat operation or procedures may be required for high yaw rate or violent high angle of attack maneuvering. Additionally, cockpit indications of the condition of the emergency recovery systems shall be available to the pilot.

A flight test plan similar to the one used for simulation shall be developed before initiation of flight test. Through flight test, frequent procuring activity/contractor coordination meetings should be held to review results to date and determine the safest course to achieve program

objectives. The initial flight test evaluation shall include a careful buildup to the maneuvers of Technique (1) (ordinary control inputs) in addition to those maneuvers necessary to determine stability derivatives and calibration of the air data sensors. After evaluation of Technique (1), a careful buildup test to positive and negative AOAs and sideslip in excess of planned production limit settings is needed to verify and define stability derivatives. The simulator aerodynamic data, sensor effects, and subsystem effects (e.g., hydraulic/electrical power) shall be updated. Continued piloted simulations shall be performed to evaluate departure characteristics for Technique (2) control inputs (misapplied control inputs) and Technique (3) control inputs (the effects of consecutively misapplied control inputs). The effects of Technique (2) control inputs shall then be test flown. The results of flight test and updated simulation results shall be utilized for system refinement and pilot handbook information. After completion of the departure phase of the F-15 program, a spin recovery program was initiated using Technique (4) maneuvers.

A survey of 33 air vehicle manufacturers, research and test agencies, and operational commands and squadrons (AFWAL-TR-81-3108) provides considerable information on "mission phases or tasks involving high-AOA flight, past or present flying qualities problems, stall/departure/spin encounter, future desires, etc." While information was sought on all classes of air vehicle, most of the concern dealt with departure/spin resistance for Class IV, highly-maneuverable air vehicles. It is realized, however, that this is very important for all classes. Mission requirements such as initial pilot training, forward air control, increased gross weight, or high-altitude start of long-range cruise (to name just a few) expose Class I, II, and III air vehicles to any departure tendency. Concern of operational pilots covered "inadequate cues, flight control system limiters which obviously remove the pilot from control, and adequate control power."

A major consensus derived from the AFWAL-TR-81-3108 survey is that, for Class IV air vehicles:

> "… high-AOA maneuvering in combat, although spectacular or glorious, is not a primary tactic. It is definitely a subordinate area, but one which shall not limit use of the aircraft. High-AOA is equated with high energy loss, slowing velocity, and becoming an easy target for the opponent's gun or missile. It is much more desirable to maintain high specific energy by avoiding hard maneuvering. High-AOA combat generally results from pitting aircraft of similar performance and maneuvering capabilities against one another. If the opponents have dissimilar performance capabilities, the fight generally will not last long enough to degenerate to high-AOA. Thus most high-AOA flight results from air combat maneuver (ACM) training against the same type of aircraft. It generally involves gun fighting, and new weapon systems coming into the inventory are counted on to reduce gun fighting.
>
> Thus considering that high-AOA maneuvering is subordinate to the primary mission but shall not limit the air vehicle's usefulness, the major expressed concern involved departure/spin resistance, flight cues, and the role of the flight control system."

More recently still, renewed interest in extreme AOAs for air combat maneuvering has been seen. This need is one matter that must be settled at the outset of a design, since it can have a great impact on the configuration.

Typically Class I air vehicles have much lighter wing loadings than the rest. Most are designed to meet FAA regulations (FAR Part 23), then adapted for military operations; so high-AOA flight is looked at differently for their usage. Similarly, due to the very large inertias and limited maneuvering of Class III air vehicles in all axes, high-AOA departures or large uncommanded motions are not structural design considerations, so vehicle design shall assure that such maneuvers are not likely to be encountered. The major concern for departures and spins (5.2.5.5 through 5.2.5.6.4) is therefore Class IV air vehicles.

In terms of design philosophy, AFWAL-TR-81-3108 concludes that there are three separate schools of thought: aerodynamic dominance (e.g., the F-5), balanced aerodynamics and flight control system (F-15), and flight control system dominance (F-16). The military using agencies "expressed views advocating specification- and design-restraint…High-AOA flying qualities specification requirements shall not dictate aircraft configuration, flight control system complexity, or even overly compromise primary mission performance." Despite this desire, recent designs (F-15, F-16, F-18, F-20 for example) owe some of their dominant external configuration features, considerable control system logic, or both, to high α considerations. These features were deemed necessary just to avoid excessive occurrences of loss of control.

The reader is referred to AFWAL-TR-81-3108's excellent summary for additional references and more detail on high-AOA requirements, characteristics, and criteria.

On the F-15 and some other air vehicles for which vortices off the nose are prime contributors to the high-AOA characteristics, the flight test nose boom had a marked effect on departure. External stores and store or internal fuel asymmetries have also been found to influence some air vehicles' high-AOA characteristics significantly.

FTC-TD-73-2 provides additional guidance and lessons learned on flight test demonstration of departure resistance and post-departure characteristics.

The availability of valid air vehicle simulation models and other prediction tools is critical to the safe conduct of flight test operations and the reduction of the actual flight tests necessary to document air vehicle characteristics. During preparation for high angle of attack flight testing of the F/A-18E air vehicle, the test team sought to optimize flight testing, minimize resource expenditures, and expedite decisions and schedules by ensuring the fidelity of the high angle of attack simulation model. High confidence in the flight simulation allowed much of the build-up to be eliminated and the interval between test points to be increased. A key to the test team's approach was that the F/A-18E high angle of attack testing was conducted in four phases. During the first phase air vehicle flying qualities at high angle of attack were qualitatively evaluated during benign maneuvering and flight data were gathered to verify and correct the aerodynamic database used in the simulation. The second phase explored build-up to spin entry to confirm the recovery characteristics of the air vehicle from out-of-control conditions. The third phase was designed to determine the control inputs and flight conditions that would cause

departures from controlled flight, document the departure boundaries, and demonstrate each type of departure. The test team made extensive use of off-line simulation in the third phase by performing tens of thousands of simulation runs evaluating all conceivable combinations of pilot input at hundreds of different flight conditions. Various loadings, asymmetric thrust conditions, aft center of gravity locations, and pitch attitudes were investigated. The flight test conditions were selected from the conditions identified as prone to departure in the simulation. Flight test results were then used to refine the aerodynamic database used in the simulator to allow for development and modification of control laws such that departures were minimized or eliminated. This approach to high angle of attack testing was also instrumental in revealing a major controllability problem with the air vehicle. Simulation demonstrated that simultaneous application of lateral and longitudinal stick at low angle of attack produced an inertial coupling yawing moment which saturated the rudder causing uncontrollable yaw. The test team was able to identify and correct this problem prior to actually performing the maneuver in flight, which served to eliminate the hazard and greatly reduced risk. The fourth and final phase consisted of dedicated spin testing to analyze spin recovery techniques. The extensive use of simulation and phased approach to flight test buildup served to minimize risk and instilled confidence in the effectiveness of the recovery controls and predictability of the departure conditions and modes.

5.2.5.1 Warning cues.
Warning or indication of approach to stall or loss of air vehicle control shall be clear and unambiguous.

REQUIREMENT RATIONALE

The seriousness of the consequences of stalling, departure, or spinning demands clear, unambiguous cues to warn the pilot.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.1.1
MIL-STD-1797: 4.8.4.1.

REQUIREMENT LESSONS LEARNED

This requirement is essentially identical to 5.2.1.1.1. Its addition here is based upon three observations: 1) the requirement of 5.2.1.1.1 is intended for any dangerous flight condition, not specifically for high-AOA flight; 2) there may be some instances (e.g., air combat maneuvering at high AOA) which should not be considered dangerous; and 3) warning cues provided on many recent air vehicles for high-AOA flight are considered inadequate.

Provision of a consistent, useful warning cue to the pilot continues to be a problem, as shown by a survey of pilots of Class IV air vehicles (AFWAL-TR-81-3108). AFWAL-TR-81-3108 summarizes:

> "Lack of adequate high-AOA maneuvering/stall non-visual (e.g., tactile) cues rank very high on the pilots' problem list. Such cues are a primary source of information when attention is directed away from the instruments—as is generally the situation surrounding stall encounter. Cues are equally important in air combat to establish maximum and/or optimum maneuver conditions. It appears that very few aircraft have adequate non-visual cues. In particular, single-crew aircraft require a separation of information channels which might be compared with the need for frequency separation in highly augmented aircraft with uncoupled modes of control. That is, artificial devices such as pitch or rudder pedal shakers can be (and are) masked by buffet; aural tones can be (and are) masked by radio communications or missile arming and lock-on tones. The preferred cues are buffet itself and possibly the most consistent and desirable tactile cues—stick force and position. These were stressed over and over by the operational pilots.
>
> The key cues which provide positive indication of changing aircraft AOA or energy state are:
>
> Stick force (per kt or g)
>
> Stick position
>
> Buffet level
>
> Uncommanded aircraft motion
>
> Artificial warning devices."

It must be emphasized here that the intent of this requirement, like that of all the high-AOA requirements, is not to force an artificial limit on the air vehicle. The AFWAL-TR-81-3108 survey of using agencies concludes that:

> "Prevention of dangerous flight conditions via maneuver limiters drew strong objections from a large segment of the military community. Such devices are viewed as double-edged swords; they inflexibly protect the aircraft (and crew) from inexperienced or inept piloting at the cost of an (arbitrary) imposed safety margin. In so doing they become a pilot equalizer, and make aircraft maneuvering performance predictable to the enemy. Finally, protective limit requirements generally vary with aircraft loading (external or internal) and therefore to be effective entail considerable complexity."

Additional information on the F/A-18A at high AOAs ("F/A-18A High Angle of Attack/Spin Testing") shows that it has inadequate buffet and natural stick force cues, though an α-feedback in the control augmentation system provides a good artificial stick force cue. A warning tone is also employed.

Artificial warnings have proven to be inadequate on many air vehicles. AFWAL-TR-81-3108 discusses the F-111/FB-111 in particular:

> "It was designed to have (and does have) the very best flying and ride qualities throughout its operational flight envelope. It is described as the Cadillac® of military aircraft. This is accomplished largely through the incorporation of:
>
> High-gain authority command augmentation systems
>
> Maneuver enhancement devices (automatic configuration changes)
>
> Automatic series trim
>
> As a result, the flying qualities pertaining to stick force, stick position, and aircraft motion remain essentially invariant until stall or departure occurs. There is little buffet, and even this does not change appreciably with AOA. Thus, the aircraft suddenly falls off a "cliff." Three artificial cues—a stick shaker, a horn, and panel lights—are provided, which activate at 14º AOA, well below the departure AOA of 20-21º. However, these have met with little success in preventing stalls and loss of control. A control system modification is now being retrofitted which will restore the needed stick force/position cues."

The FB-111 control system modifications include a stall inhibitor, a sideslip reducer, and an increase in stick force cues as the AOA limit is approached. This system can be defeated at low airspeed by various combinations of control inputs.

VERIFICATION

Verification shall be by demonstration in the flight tests of the tasks of 5.2.5. Proof of compliance in these demonstration tasks will consist of pilot comments. The comments should indicate that the warning cues are clear and unambiguous.

Due to the complex nature of high-AOA flight, final demonstration of compliance with this requirement will necessitate flight testing. Wind tunnel testing will give a preliminary indication of natural buffeting. If artificial warning cues are utilized, verification may include ground simulation. Pilots shall evaluate the adequacy of the cues in operational-type maneuvering, as well as in test stall approaches.

The landing configuration at low weight may be critical because of a lower buffet intensity at the lower airspeed; or there may be less buffeting in the clean configuration.

Evaluation of pilot cues is inherently subjective. The warning shall be evaluated by a number of pilots. If there is no consensus, acceptability shall be based on adequacy for most service pilots in the intended missions.

5.2.5.2 Stall approach.
The onset of stall warning (5.2.5.1) shall occur within the speed limits of table LXXXI for 1-g stalls and within the lift limits of table LXXXII for accelerated stalls, but not within the ROSH.

TABLE LXXXI. Speed range for onset of stall warning for 1-g stalls.

Flight Phase	Minimum Speed for Onset of Warning	Maximum Speed for Onset of Warning
Approach		
All others		

TABLE LXXXII. Lift range for onset of stall warning for accelerated stalls.

Flight Phase	Minimum Lift for Onset of Warning	Maximum Lift for Onset of Warning
Approach		
All others		

5.2.5.2.1 Stall warning.
Stall warning shall be easily perceptible and shall consist of buffeting or shaking of the air vehicle, shaking of the cockpit controls, or both. An increase in intensity of the warning with further increase in AOA shall be sufficiently marked to be noted by the pilot. The warning shall continue until the AOA is reduced to a value less than that for warning onset.

5.2.5.2.2 Uncommanded oscillations prior to stall.
Prior to the stall, uncommanded oscillations shall not be so severe as to require the pilot's full attention to retain control in the maneuver.

5.2.5.2.3 Cockpit controls prior to stall.
At all AOAs up to the stall, the cockpit controls shall remain effective in their normal sense, and small control inputs shall not result in departure from controlled flight.

REQUIREMENT RATIONALE

Approach to stall must always be clearly indicated to the pilot with a margin (airspeed or AOA) sufficient to recover from the incipient stall, yet small enough to be meaningful.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.2.1.1 through 3.4.2.1.1.2
MIL-STD-1797: 4.8.4.2.1.
Recommended values for table LXXXI:

Flight Phase	Minimum Speed for Onset of Warning	Maximum Speed for Onset of Warning
Approach	Higher of $1.05V_S$ or $V_S + 5$ kts	Higher of $1.10V_S$ or $V_S + 10$ kts
All others	Higher of $1.05V_S$ or $V_S + 5$ kts	Higher of $1.15V_S$ or $V_S + 15$ kts

Recommended values for table LXXXII:

Flight Phase	Minimum Lift for Onset of Warning	Maximum Lift for Onset of Warning
Approach	82% of $C_{L_{stall}}$	90% of $C_{L_{stall}}$
All others	75% of $C_{L_{stall}}$	90% of $C_{L_{stall}}$

REQUIREMENT LESSONS LEARNED

Even where the ROSH and ROTH coincide, as they may at the low-speed boundaries, there shall be sufficient margin from stall that warning shall still be required to occur outside the ROSH.

A requirement limiting uncommanded oscillations (as in 5.2.5.2.2) is quite subjective: one pilot may want no uncommanded motion associated with approach to stall, while another may consider some such motion a necessary evil or even a cue of occasional value, and so find oscillations acceptable. The results of the piloted simulations of AFWAL-TR-80-3141 suggest that a noticeable "g-break" indicated stall while any aperiodic uncommanded motion (in any axis) of greater than 20º/sec signified departure.

The accelerated stall margins are in terms of $C_{L_{stall}}$, as in MIL-F-8785C. They correspond to the airspeed stall margins for unaccelerated flight. That was a change from Interim Amendment-1 United States Air Force (USAF), which used AOA margins in recognition of the very shallow lift curve slope characteristic of low aspect ratio and swept wings in the stall approach region. Our C_L margin corresponds then to a rather wide α margin, thus tending to restrict the usable AOA range more than may be necessary. Nevertheless, upon reflection, the authors of this standard were convinced that: (a) accelerated stall warning requirements must be consistent with those for unaccelerated stalls (for which an error speed margin is both rational and well accepted); and (b) the large $\Delta\alpha$ does not provide enough extra lift to warrant more special consideration. The required progressive warning should assist fighter pilots to extract that last little bit of g from their air vehicle in a dogfight or dive pullout. Perhaps it could be supplemented by a tone or some other indication of nearness to stall AOA.

With limited aerodynamic design capability and inadequate data on pilot desires, more detailed specification of stall warning margins seems unwarranted. However, gaining pilot acceptance of

dynamic stall warning may require some additional tailoring of the air vehicle. Possibly the warning range desired for accelerated stall would be mission-dependent (for example, air-to-ground versus air-to-air), considering the average altitude available for recovery, the rapidity of speed bleed-off for the vehicle and weapon configuration, and departure susceptibility and severity. Data are insufficient to establish such mission-dependent criteria, and implementation on air vehicles would be difficult, so the requirements of MIL-F-8785C have been retained.

Adequate, timely warning is of paramount importance if the stall and post-stall characteristics are less than satisfactory. However, even if there is an effective stall limiter, a pilot needs a readily perceived indication of approaching an air vehicle limit.

See AFWAL-TR-80-3141 for more accounts of experience.

VERIFICATION

Verification shall be by demonstration in the flight tests of 5.2.5. Proof of compliance in these demonstration tasks will consist of pilot comments and time histories of the stall approaches. While wind tunnel tests and analyses can provide estimates, final verification must be by flight test. Verification of 5.2.5.2 through 5.2.5.2.3 apply for all stalls, including stalls entered abruptly.

The stall angle of attack at constant speed for the configuration, weight, center-of-gravity position and external store combination associated with a given Air Vehicle Normal State is defined as the lowest of:

a. Angle of attack for the highest steady load factor, normal to the flight path, that can be attained at a given speed or Mach number

b. Angle of attack, for a given speed or Mach number, at which uncommanded pitching, rolling, or yawing occurs (5.2.5.3)

c. Angle of attack, for a given speed or Mach number, at which intolerable buffeting is encountered

Stall speed as defined in 4.1.4.3 is a steady state. To determine V_S and α_S the stall approach shall be made slowly to eliminate any dynamic effects. At the 1 kt/sec break rate called for by the FAR Part 23 and 25, an airspeed somewhat lower than the present V_S may be reached before the stall break. Trial will determine a rate that is slow enough for the particular air vehicle. (FAR Part 25 landing approaches, however are at 1.4 times the stall speed; while MIL-C-5011 called for landing approach at 1.3 times a somewhat higher V_S, which may be comparable to the FAR approach speed.) Nevertheless, rapid stall entries are also to be evaluated.

Conditions for determining V_S are shown in table LXXXIII. The aircraft shall be trimmed initially at approximately $1.2V_S$ with the settings shown; thereafter the trim and throttle settings shall be held constant. In flight test, it is necessary to reduce speed very slowly (typically ½ kt/sec or less) to minimize dynamic lift effects. The load factor will generally not be exactly 1 g when stall occurs; when this is the case, V_S is defined as follows:

$$V_S = \frac{V}{\sqrt{n_f}}$$

where V and n_f are the measured values at stall, n_f being the load factor normal to the flight path.

TABLE LXXXIII. Conditions for determining V_S.

Flight Phase	Thrust Settings*	Trim Setting
Climb (C)	Normal climb	For straight flight
Descent (D)	Normal descent	For straight flight
Emergency Descent (ED)	Idle	For straight flight
Emergency Deceleration (DE)	Idle	For straight flight
Takeoff (TO)	Takeoff	Recommended takeoff setting
Approach (PA)	Normal approach	For normal approach
Waveoff/Go-Around (WO)	Takeoff	For normal approach
Landing (L)	Idle	For normal approach
All other	TLF at $1.2V_S$	For straight flight

* Either on all engines or on remaining engines with critical engine inoperative, whichever yields the higher value of V_S

Suitability of the warning shall be evaluated in operational conditions. Beyond low subsonic speeds, V_S and α_S are functions of Mach number.

Stall buffet will be less intense at lower speeds, corresponding to lightweight and high-lift configurations (unless the high-lift configuration enhances the stall buffeting).

5.2.5.3 Stall characteristics.
During the unaccelerated stalls of 5.2.5.2, the pilot shall be able to keep roll, yaw, and downward pitch excursions within __(1)__ degrees of the stall attitude. No pitch-up tendencies shall occur in stalls, unaccelerated or accelerated, except a) in the unaccelerated stalls of 5.2.5.2 mild nose-up pitch is acceptable if no pitch control force reversal occurs and no dangerous, unrecoverable, or intolerable flight conditions result, and b) in the accelerated stalls of 5.2.5.2, a mild nose-up tendency is acceptable if the operational effectiveness of the air vehicle is not compromised and the air vehicle has adequate stall warning, pitch control effectiveness is such that it is possible to stop the pitch-up promptly and reduce the AOA, and at no point during the stall approach, stall, or recovery, does any portion of the air vehicle exceed structural limit loads.

REQUIREMENT RATIONALE

In order for an air vehicle to be controllable in a developed stall, uncommanded angular excursions must be of a manageable magnitude and, in the case of pitch excursions, shall be in a direction that will enhance controllability.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.2.1.2
MIL-STD-1797: 4.8.4.2.2.

Recommended value for blank (1):

Air Vehicle Class	Excursion limit
Classes I, II, and III	20º
Class IV	30º

These limits are the amount of attitude change at stall.

REQUIREMENT LESSONS LEARNED

The stated tolerance of mild pitch-up is a concession to allow configurations giving significant performance benefits not to be penalized unduly for their stall characteristics. The designer shall make every reasonable attempt to avoid pitch-up or, failing that, to minimize it.

There is no mention of angular rates, which may be more important to the pilot at stall. The transients due to failure of the primary flight control system (5.2.1.5.1) are recommended to be less than ±0.5 g laterally or longitudinally and ±10º/sec roll rate within 2 sec. A similar constraint could be defined for unaccelerated stalls. For particular applications—such as air combat fighters—the intended mission may dictate tighter limits, or even prohibit such excursions, whether open- or closed-loop.

AFWAL-TR-80-3141 and AFFDL-TR-74-61 point out that cases exist in which a pilot's attempts at stabilization do not help, but actually induce instability. For example, with the A-7,

aerodynamic coupling between the longitudinal and lateral-directional motions while sideslipping is shown to be the cause of departure from controlled flight. While sideslip is not specifically mentioned in these requirements, some sideslip is common, even unavoidable at high AOAs. However, air vehicles rarely have a decent zero sideslip reference. The contractor should consider whether a sideslip requirement is appropriate.

This requirement legislates against severe pitch-up tendencies, but some non-violent pitch-down which can be arrested often is a desirable characteristic. For large (Class III) air vehicles, excessive nose-down pitching is undesirable, according to AFWAL-TR-81-3108, because of:

> "…the very large inertias involved and the excessive altitude loss which is incurred before recovery. The regions where stall is most usually encountered may also be important, e.g., pitch-down due to stall at cruise ceiling could lead to Mach overspeed, while pitch-down in the landing pattern could easily lead to a non-recoverable dive. The preferred recovery sequence is to set the aircraft nose on the horizon, add full power, and wait for the aircraft to regain flying speed. The preferred metric is the dwell time between recovery initiation and regaining of flight speed. Altitude loss due to settling is less than that due to a diving recovery."

This preference obviously depends on the drag characteristics at stall AOA. The technique described is consistent with training procedures used for civil transport air vehicles. For example, "Out for a Spin" describes the stall series used in Boeing 747 training and recurrent checks:

> "…a V_{ref} (final approach) speed is computed for the landing weight and a bug position next to this number on the airspeed gauge. The first stall is made clean with wings level, the next in a 20º bank with 10º of flaps, and the third straight ahead with the gear down and landing flaps (30º). In each exercise the engines remain spun up but at low thrust settings. These configurations approximate those seen in near airport maneuvering.
>
> The recovery from each is the same: at buffet or stick shaker, apply go-around thrust, lower the nose to 5º above the horizon, and level the wings. When properly executed, the 747 will resume normal flight with little or no loss of altitude. Rough handling insures a secondary buffet or shaker, or both, and substantial altitude loss."

VERIFICATION

Verification shall be by demonstration in the flight tests of 5.2.5. Proof of compliance in these demonstration tasks will consist of pilot comments and time histories of the stalls.

While wind tunnel tests and analyses can provide estimates, final verification must be by flight test.

The flight test program should be conducted cautiously, with a build-up from less severe configurations, loadings, entries, and failure states, to more severe ones.

Stall characteristics generally tend to deteriorate as the c.g. moves aft. External stores may affect the aerodynamic or inertial characteristics significantly, and any asymmetries which can reasonably expected shall be test flown.

The extent of required stall penetration during flight testing has been argued extensively. Civil and military definitions of V_S differ (see 5.2.5.2 Verification). Also, the desired degree of penetration depends to an extent on intended use, which may be more severe for military than for civil use. In general, even military cargo and transport air vehicles shall be taken at least to a stall break that is discernible on the flight records.

5.2.5.4 Stall prevention and recovery.
It shall be possible to prevent the stall by moderate use of the pitch control alone at the onset of the stall warning.

5.2.5.4.1 Stall recovery.
It shall be possible to recover from a stall by simple use of the pitch, roll, and yaw controls with cockpit control forces not to exceed the limits of table LXXXIV, and to regain level flight without excessive loss of altitude or build-up of speed. Throttles shall remain fixed until an angle of attack below the stall has been regained unless compliance would result in exceeding engine operating limitations.

TABLE LXXXIV. Control force limits for stall recovery.

Axis of Response	Maximum Control Force for Recovery
Pitch	
Roll	
Yaw	

5.2.5.4.2 Control power for stall recovery.
In the straight-flight stalls of 5.2.5.2, with the air vehicle trimmed at an airspeed not greater than $1.4V_S$, pitch control power shall be sufficient to recover from any attainable AOA.

5.2.5.4.3 One-engine-out stalls for multi-engine air vehicles (more than one engine). It shall be possible to recover safely from stalls with the critical engine inoperative and thrust on the remaining engines at the settings of table LXXXV:

TABLE LXXXV. Thrust settings to recover from one-engine-out stalls.

Flight Phase	Thrust
TO	
C	
PA	
WO	

REQUIREMENT RATIONALE

Except for practice or test, stalling is generally an unexpected and potentially dangerous event. Therefore, recovery must be easy and instinctive. Some multi-engine air vehicles exhibit violent, unacceptable rolling or yawing tendencies in engine-out stalls, while the need to maximize air vehicle performance for recovery from an engine failure increases the possibility of stalling.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.2.1.3 and 3.4.2.1.3.1
MIL-STD-1797: 4.8.4.2.3 and 4.8.4.2.4.

Recommended values for table LXXXIV:

Control Type	Pitch (lbs)	Roll (lbs)	Yaw (lbs)
Sidestick	20	15	
Centerstick	50	25	
Wheel (two-handed tasks) (one-handed tasks)	 75 50	 40 25	
Pedal			175

Recommended values for table LXXXV:

Flight Phase	Thrust
TO	Takeoff
C	Normal climb
PA	Normal approach
WO	Waveoff

Requirement 5.2.5.4.3 applies to multi-engine air vehicles (more than one engine) only.

REQUIREMENT LESSONS LEARNED

Prevention of and recovery from the stall must always be simple for the pilot. Military Specification MIL-F-8785C included the requirement that throttles remained fixed until "speed has begun to increase." This has been removed in recognition of the method of stall recovery used for both light trainer (Class I) and heavy (Class III) air vehicles: release back pressure on the wheel, lower the nose to the horizon, and add power—whether airspeed has begun to increase or not.

As long as the wing is unstalled, the addition of power will aid in flying out of the stall with minimal altitude loss. If stalling may produce engine flameout, however, recoveries with appropriate thrust (or lack of it) shall be investigated. Also note that control to balance propeller torque may limit the application of power for recovery at very low airspeed.

A potential quantitative criterion for specifying stall recovery for Class III air vehicles is dwell time (AFWAL-TR-81-3108). As mentioned in 5.2.5.3 Requirement Lessons Learned, this is the time between occurrence of the stall and recovery of flying speed. The criterion is in accordance with standard practice for stall recovery: keeping the nose at the horizon and adding thrust, rather than letting the nose fall through the horizon before thrust is applied.

AGARD-CP-260 shows that for three Class III air vehicles (S-3A, L-1011, and C-5A) maximum nose-down pitch acceleration at the stall was less than or equal to 0.08 rad/sec^2 for 90% of the stalls. It therefore suggests that a pitch recovery criterion could be that the pitch control produce $\ddot{\theta}$ greater than 0.08 rad/sec^2 at stall.

During stall testing of the F-16A/B with aft c.g., pitch-up to an upright deep-stall was encountered, requiring a spin chute for recovery. Figure 200 shows a time history of a deep stall. Analysis of the F-16 flight control system suggests that the deep stall condition may have been aggravated by anti-spin stability augmentation (SAS), (figure 201) which is activated at $\alpha \geq 29^{\circ}$, combined with a longitudinal stick gain to remove the pilot from the loop. Figure 200 shows the point at which the anti-spin SAS became active (t = 10 sec, δ_F is differential flaperon deflection). A lateral limit cycle oscillation developed, possibly caused by the anti-spin SAS, and cross-axis coupling caused the air vehicle to pitch to still higher α and subsequent deep stall, with full nose-down stabilator deflection.

Recovery from this deep stall (which might arguably be termed a post-stall gyration but certainly is prohibited by 5.1.3) without a spin chute requires a manual pitch override (MPO) in the longitudinal SAS (AFFTC-TR-79-18):

> "A manual pitch override system was installed in the test aircraft to allow pilot control of the stabilator in a deep stall condition (upright or inverted), and thus allow the aircraft to be "rocked out" of the deep stall…. This pitch override system required the pilot to hold a toggle switch, located on the left console, in the OVRD position during usage. The switch was spring loaded to the NORM position. When selected, the pitch override (a) eliminated the negative g limiter to allow TED stabilator control and (b) for AOA greater than or equal to 29º,

> eliminated the AOA limiting and pitch integrator functions to allow trailing edge up (TEU) stabilator control."

An MPO switch was included in production air vehicles, but according to AFFTC-TR-79-18, its operational utility is questionable:

> "The MPO was an effective upright deep stall recovery device when utilized properly…. However, the ability of the operational pilot to properly and readily adapt to the usage of the MPO remains a concern. During flight tests with pilots who were extremely familiar with the deep stall environment, as many as four total cycles of stick were required before an effective cycle was achieved. The primary difficulty encountered involved improper phasing with existing pitch oscillations. Proper phasing became much more difficult when severe roll oscillations existed. The rolling tendency (to as much as 90º bank angle) masked the pitching motion of the aircraft."

Such phasing between stick and air vehicle motion could be considered a violation of this requirement: that is, this is not a "simple" use of the pitch control. What is expedient as a "fix" is not necessarily acceptable for standard use.

Loss of an engine in low-speed flight will often lead to a stall, especially in a critical flight phase such as takeoff. The large yawing and rolling moments produced by an engine-out situation can then induce a spin if recovery from the stall is not immediate.

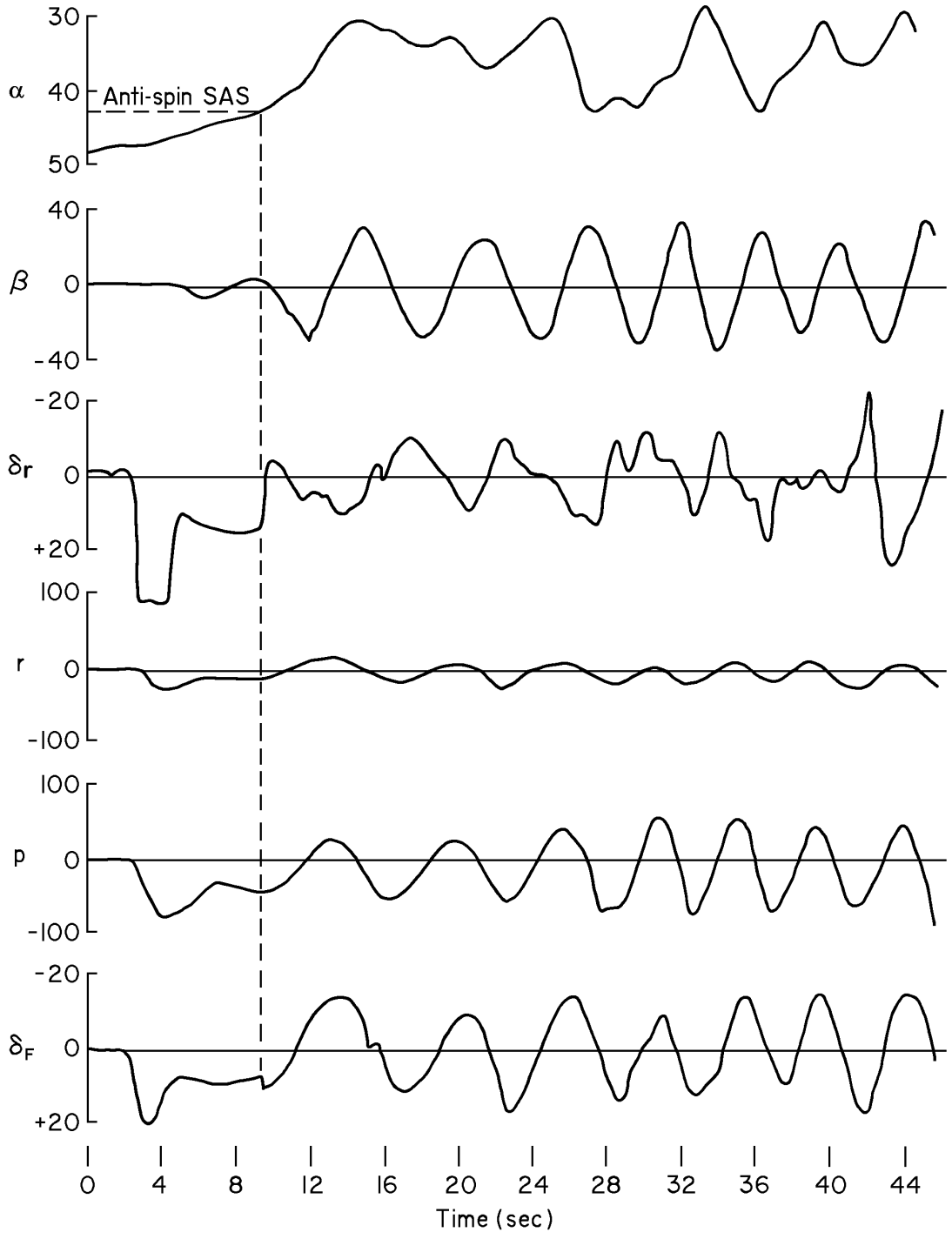


FIGURE 200. Time history of aft-c.g. deep stall encountered by F-16 (AFFTC-TR-79-18).

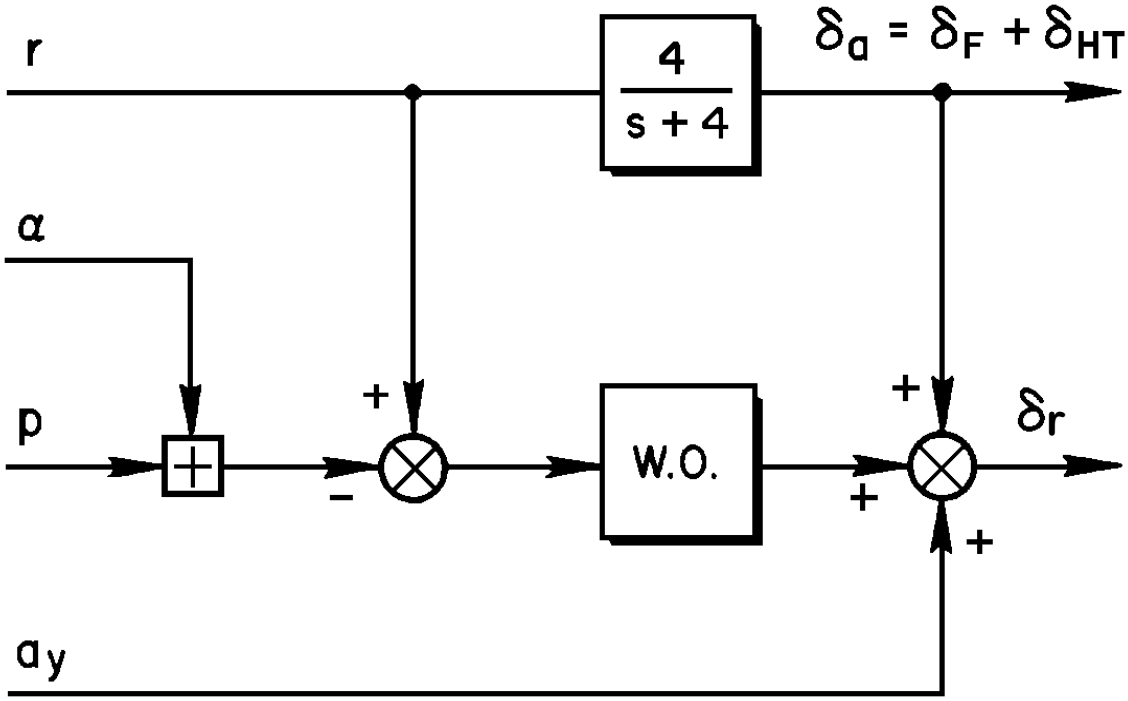


FIGURE 201. Anti-spin SAS for F-16B ($\alpha \geq 29^{o}$).

For civil air vehicles, FAR Part 25 requires that recovery be possible "with the remaining engines at up to 75% of maximum continuous power, or up to the power at which the wings can be held level with the use of maximum control travel, whichever is less." FAR Part 23 is more severe in that it has the additional requirement that the air vehicle not display any undue spinning tendency during the single-engine stall demonstration.

Throttling back on the operative engine(s) during recovery is allowable.

There is some evidence that stalls with one-engine inoperative and the other(s) at high power have led to departures and, in some cases, an out-of-control flat spin. This has occurred on contemporary fighter air vehicles as well as on light twin-engine air vehicles, usually as a result of delayed recovery controls. It is conjectured that several C-133 air vehicles lost at sea had suffered an engine failure at the start of long-range cruise, at or above the service ceiling, where stall margin is minimal. Artificial stall warning, having been found undependable, was sometimes turned off; and poor roll control and a severe roll-off accompanied stall, more so with only three engines.

VERIFICATION

Verification shall be by demonstration in the flight tests of 5.2.5. Proof of compliance in these demonstration tasks will consist of pilot comments and time histories of the stall preventions and stall recoveries.

While wind tunnel tests and analyses can provide estimates, final verification must be by flight test.

Both stall approaches broken off at stall warning and complete stalls to an AOA great enough to identify V_S are to be performed—with caution and careful build-up at a safe altitude, as with all high-AOA testing. Again, the degree of stall penetration depends to an extent on the intended use of the air vehicle. On air vehicles for which high-AOA testing does not proceed beyond the stall AOA, verification of control to recover from any attainable AOA may be by analysis of wind tunnel data.

These and other high-AOA tests shall be preceded by thorough study of available model tests and simulation results for the particular air vehicle.

For one-engine-out stalls, the same precautions shall be observed as in testing other stalls, only more so. These tests will normally follow the symmetric thrust stalls, incrementally increasing good engine thrust on successive stalls. Where propellers or fans direct airflow over the wing, the side with reduced thrust will generally stall first. Roll control effectiveness may also be reduced by lessened dynamic pressure or even local stalling at the ailerons or spoilers.

5.2.5.5 Departure from controlled flight.
The air vehicle shall be __(1)__ .

5.2.5.5.1 Departure warning.
Adequate warning of approach to departure (5.2.5.1) shall be provided.

5.2.5.5.2 Uncommanded motions.
The air vehicle shall exhibit no uncommanded motion which cannot be arrested promptly by simple application of pilot control.

5.2.5.5.3 Departure avoidance following sudden asymmetric loss of thrust.
At all AOAs within the ROSH, following sudden asymmetric loss of thrust from the most critical factor, it shall be possible to avoid departure without exercise of exceptional pilot skill.

REQUIREMENT RATIONALE

Departure resistance is a prime concern for high-AOA flight. So far it has been difficult to arrive at an agreed upon method of predicting departure susceptibility.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.2.2.1
MIL-STD-1797: 4.8.4.3.1.

Recommended words for blank (1) in 5.2.5.5:

> Most air vehicles: "resistant to departure from controlled flight, post-stall gyrations, and spins"
>
> Air vehicles specifically intended for operation at high AOA: "extremely resistant to departure from controlled flight, post-stall gyrations, and spins"
>
> Trainers specifically intended to provide spin recovery training: "resistant to departure from controlled flight, post-stall gyrations, and spins, but shall be capable of a developed spin and consistent recovery"

REQUIREMENT LESSONS LEARNED

The definitions of departure susceptibility and resistance from the cancelled specification MIL-F-83691 are pertinent here:

> "Extremely susceptible to departure: departure from controlled flight will generally occur with the normal application of pitch control alone, or with small roll and yaw control inputs.
>
> Susceptible to departure: departure from controlled flight will generally occur with the application or brief misapplication of pitch and roll or yaw controls that may be anticipated in operational use.
>
> Resistant to departure: departure from controlled flight will only occur with a large and reasonably sustained misapplication of pitch and roll and yaw controls.
>
> Extremely resistant to departure: departure from controlled flight can only occur after an abrupt and inordinately sustained application of gross, abnormal or pro-departure controls."

On a pragmatic basis, the normal requirement would be for an air vehicle to be resistant to departure. If the high-AOA region is particularly important, the air vehicle may be required to be extremely resistant. Also, the procuring activity may further designate that certain (training) air vehicles shall be capable of a developed spin and consistent recovery—so that pilots will not experience such phenomena cold on some later air vehicle.

The requirement is intended to apply to all air vehicles. The terms "large", "reasonably sustained", "abrupt", and "inordinately sustained", however, are to be interpreted according to the air vehicle Class and mission. The specification MIL-F-8785C required the air vehicle be "extremely resistant"; it is recommend the requirement be reduced to "resistant" in most cases. In the words of AFWAL-TR-81-3108, "The requirement of 'extremely resistant to departure' can be expected to dictate aircraft configuration or flight control system complexity, or both—precisely what the using commands warn against. Their preference is that the aircraft be departure/spin resistant." See also 5.2.5 Verification. Easing this requirement also allows for those (admittedly rare) occasions when pilots of Class IV air vehicles want to use departure as a

last ditch evasive maneuver during air combat. The major difference, reflected in the definitions above, is in requiring "reasonably sustained application of … controls" and "inordinately sustained application of gross, abnormal, pro-departure controls" for producing a departure. This difference should not be important except during air-to-air combat.

A requirement for a departure warning (see 5.2.5.1) reflects pilots' concerns. According to AFWAL-TR-81-3108:

> "Warning is needed which is separate and distinct from stall warning. Margins (maximum and minimum) between warning onset and actual departure shall be dependent upon pitch control power (how rapidly the aircraft can transit the warning region), departure severity, spin susceptibility, and aircraft mission."

VERIFICATION

Verification shall be by demonstration in the flight tests of 5.2.5. Proof of compliance in these demonstration tasks will consist of pilot comments and time histories of the departures.

When the post-stall region is not banded by structural design considerations, flight testing is a necessity since it is difficult to define an accurate aerodynamic model for post-stall flight.

Engineering preparation for departure and spin tests should be done with care. Tasks include:

a. Determination of recovery devices necessary for departure/spin program.

b. Effect of these devices on the aerodynamic characteristics and inertial characteristics of the vehicle.

c. The limits of operation of these devices.

d. A sensitivity analysis of the aerodynamic characteristics and their influence on departure and spin susceptibility. This shall be done on the manned simulator as it may influence the flight test technique used to explore the AOAs where unfavorable aerodynamic nonlinearities are expected.

e. Go/no-go criteria shall be established; i.e., if a control surface position, sideslip angle, AOA, or rate differs by a defined amount from a predicted value, then testing shall be discontinued until further analysis is performed.

See FTC-TD-73-2 for a comprehensive discussion of considerations for a stall, post-stall, and spin flight test program.

In any simulation, the designer may prefer fixed-base over moving-base to avoid problems with confusing or unrealistic motions that might influence pilots' perceptions. Even for Class III air vehicles, which will have no spin flight tests, stall/post-stall wind tunnel tests and analysis are in order.

Stall AOA (or $C_{L_{max}}$) is dictated by performance requirements. However, experience with the F-5 series air vehicles and the F-15 leads to the conclusion that a sharp increase in longitudinal stability, starting slightly below stall AOA, allows the pilot full use of the transient pitch performance for air combat maneuvers. This aerodynamic characteristic limits AOA overshoots during abrupt pullup and rolling maneuvers. It also provides rapid recovery at low dynamic pressure with neutral pitch control. Though yaw departures occur in a limited portion of the flight regime, the F-15 does not continue into a spin, but pitches down due to its inherent longitudinal stability at high AOA.

A configuration that is longitudinally unstable at or above stall is undesirable for Class IV air vehicles. AOA limiters are usually implemented in this case; limiters can be defeated, however, during low speed maneuvers such as zoom climbs and high-AOA rolls. To preclude AOA overshoots as the limit is approached, a rate anticipation system is usually incorporated into the flight control system. This feature reduces the transient maneuvering performance of the vehicle. Pitching moment curves with a strong unstable break are poor for Class IV applications. When departure occurs, it is violent and can preclude safe ejection of the crew. Even if there is a large amount of nose-down control power, the low dynamic pressure encountered as the air vehicle pitches up results in slow nose-down recovery.

Wind tunnel data that present rolling moment (C_ℓ) and yawing moment (C_n) as functions of AOA for zero sideslip shall be evaluated to ensure that no excessively large moment values occur (e.g., from asymmetric vortex shedding from the nose) that could cause departures. The aerodynamic effect of the flight test boom, if located on the nose, shall be determined.

Several quantitative requirements were suggested in AFWAL-TR-80-3141. While there is insufficient support for incorporating them as such, they may be very useful for guidance in early analysis. A fixed-base piloted simulation of an F-4J found that:

> "…pilot perception of lateral-directional departure susceptibility is related to one zero of the numerator $N^{\phi}_{\delta_{stk}}$ [δ_{stk} is lateral stick, commanding ailerons and spoilers] becoming negative. Root magnitudes more negative than -0.5 rad/sec were consistently rated as departure susceptible, while those less negative (or positive) are rated as departure resistant. This criterion reflects a closed-loop divergence rate limit related to the pilots' threshold for uncommanded motion or ability to cope. As such it is a pilot-centered criterion which should be applied for any flight situation, although it has been identified in a low Mach number, fixed-base simulation. It is consistent with the empirically established airframe-alone departure/spin criterion boundaries of Weissman [ASD-TR-72-48] and extends applicability of that criterion [Lateral Control Divergence Parameter, LCDP] to highly augmented airframe cases. It is also consistent with previous in-flight simulation of maximum controllable aperiodic divergence rates. Finally, it serves as both a design guide and a flying qualities specification item."

The Lateral Control Divergence Parameter is defined with stability-axis derivatives as

$$LCDP = C'_{n_\beta} - \frac{C'_{n_{\delta a}}}{C'_{\ell_{\delta a}}} C'_{\ell_\beta}$$

Generally, LCDP shall be greater than about -0.001. For the unaugmented airframe, according to AFWAL-TR-80-3141:

> "A value of $1/T_{\phi_1}$ = -0.5 corresponds for the airframe tested to an effective LCDP of -0.001 and thus is consistent with and supports the empirically derived LCDP departure boundary developed by Weissman."

And finally, a recommendation that no aperiodic uncommanded motions exceed 20º/sec was made, "based on a rough average of the simulation pilots' commentary as to their definitions of departure." This qualitative requirement is subject to the usual interpretation problems. There is a need for some limit between what is labeled a "stall" or a "departure."

An aggressive analytical approach to evaluate and design for departure resistance was taken for the F-5E, F-15, F-16, and YF-17 programs. This approach included obtaining good quality wind tunnel data to 90º AOA at M = 0.2 and above the stall AOA at M = 0.6, 0.9, and 1.2. These data included longitudinal and lateral-directional stability and control data with store configurations. Due to aerodynamic nonlinearities at large positive and negative AOAs, data points shall be closely spaced (approximately 3º apart in AOA and sideslip). These data were used to optimize flap schedules and evaluate buffet onset, and as input data for five-degree- and six-degree-of-freedom analyses of large amplitude maneuvers. The control laws were then determined and optimized for flying qualities and departure resistance. Maneuvers included bank-to-bank rolls at maximum AOA, rolls at negative AOAs, pushovers, pullups, and other maneuvers chosen for analysis of departure and spin characteristics. The F-15 program also used free-flight model tests to evaluate departure and spin resistance. Low Reynolds number data were used to correlate with model drop tests. This provided confidence in the analyses of the departure resistance that used high Reynolds number data. New wind tunnel techniques to enhance our analysis capability are emerging. Rotary balance tests can be used to obtain the dynamic derivatives for a more accurate analysis of stall, departure, and spin resistance. After these analyses are performed, manned simulation shall be used to verify control laws, flying qualities, and departure resistance prior to flight, and as an adjunct to flight testing.

AGARD-CP-199 presents a number of aspects and views on stall/spin problems. Research contracts sponsored by the Air Force have generated information into the causes of departures and spins. Some of these are discussed in AFFDL-TR-78-171; examples of resulting reports include AFFDL-TR-74-61, AFWAL-TR-80-3141, ASD-TR-72-48, and AFWAL-TR-81-3108.

In the fixed-base piloted simulation of AFWAL-TR-80-3141, various maneuvers (bank-to-bank and wind-up turns, and pullups) were performed with and without a target air vehicle. The simulated air vehicle was based upon an F-4J, and aerodynamic parameters were varied to assess

the effects of these parameters on handling qualities. Evaluations of departure susceptibility or resistance (based upon the MIL-F-83691 definitions) were different for the two pilots.

Although generally flight evaluations have been made with limiters operating, it has generally been found that these limiters could be defeated, tricked into allowing penetration to higher angles. Asymmetric loadings, either intentional or naturally occurring can affect post-stall behavior. An increase in control power or a decrease in stability (say, in pitch) can allow attainment of conditions theretofore unreachable.

5.2.5.6 Recovery from post-stall gyrations and spins.
The proper recovery technique(s) shall be readily apparent to the pilot, be simple and easy to apply under the motions encountered, and provide prompt recovery from all post-stall gyrations, incipient spins, and developed spins. It is desirable that the recovery techniques for post-stall gyrations, incipient spins, and developed spins be the same or at least compatible.

5.2.5.6.1 Turns and altitude loss for recovery.
For all modes of spin that can occur, recoveries shall be attainable within the limits of table LXXXVI, measured from the initiation of recovery action.

TABLE LXXXVI. Maximum turns and altitude loss for recovery.

Flight Phases	Maximum Number of Turns for Recovery	Allowable Altitude Loss for Recovery*
Categories A & B		
PA		

*not including dive pullout

5.2.5.6.2 Avoidance of Spin Reversal.
Avoidance of a spin reversal or an adverse mode change shall not depend upon precise pilot control, timing, or deflection.

5.2.5.6.3 Control Forces for Recovery.
Safe and consistent recovery and pullouts shall be accomplished without exceeding the forces in table LXXXVII, and without exceeding structural limitations.

TABLE LXXXVII. Maximum control forces for recovery.

Control Axis	Maximum Control Forces (lbs)
Pitch	
Roll	
Yaw	

5.2.5.6.4 Operation of automatic stall, departure, spin prevention or recovery devices. Operation of automatic stall, departure, and spin prevention or recovery devices and flight control modes shall not interfere with the pilot's ability to prevent or recover from stalls and departures.

REQUIREMENT RATIONALE

Recovery from post-stall gyrations and spins must be possible and prompt, with simple control application. Even for air vehicles in which flight demonstration is not feasible, the post-stall characteristics need to be known in order to give guidance for inadvertent encounters.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.2.2.2
MIL-STD-1797: 4.8.4.3.2.

Recommended values for table LXXXVI:

Air Vehicle Class	Flight Phases	Maximum Number of Turns for Recovery	Allowable Altitude Loss for Recovery*
I	Categories A & B	1-1/2	1000 ft
	PA	1	800 ft
Other classes	Categories A & B	2	5000 ft
	PA	1	1000 ft

*Not including dive pullout

Recommended values for table LXXXVII:

Type of Controller	Maximum Control Forces (lbs)		
	Pitch	Roll	Yaw
Sidestick	20	15	
Centerstick	50	25	
Wheel			
(two-handed tasks)	75	40	
(one-handed tasks)	50	25	
Pedal			175

Where requirements exist for spin recovery parachute systems, standard design practices shall be followed as found in the Deployable Aerodynamic Decelerator subsection (JSSG-2010-12) of JSSG-2010 Crew Systems Specification Guide and Handbook.

REQUIREMENT LESSONS LEARNED

AFWAL-TR-81-3109 describes the evolution of these requirements:

> "Prior to Amendment 1 to MIL-F-8785B there had been no general requirements on post-stall gyrations, as distinguished from spins. MIL-F-8785B had only a reference to the then-current spin demonstration requirements of the Air Force (MIL-S-25015) and the Navy (MIL-D-8708). For aircraft to be spun, MIL-S-25015 required ready recovery from incipient and fully-developed spins (5 turns, 1 turn spins for landing, 2 turns inverted). MIL-F-8785B Amendment 1 kept the MIL-S-25015 numbers of turns for spin recovery and added more bounds on altitude loss during recovery. The Class I requirements are similar to those of FAR Part 23 for the Aerobatic Category. Amendment 2 deleted all altitude bounds, on the premise that wing loading and drag are set by other considerations, leaving only turns for recovery to determine altitude loss, and that these bounds and turns for recovery could not reasonably be reduced further. Amendment 2 also deleted a number of Amendment 1's specifics on departure techniques, as well as an Amendment 1 requirement that the start of recovery be apparent within 3 seconds or 1 spin turn. Those specification features indicated desirable tests and characteristics but added considerable detail in areas where design capability is lacking. That material is felt to be more pertinent to a flight demonstration specification such as MIL-S-83691."

Changes from MIL-F-8785C reflect pilots' views on spin recovery. The specification of recovery in terms of altitude loss, as in Amendment 1 of MIL-F-8785B, based upon what the pilot really is concerned about, was considered. For example, the piloted simulation of AFWAL-TR-80-3141 included an air vehicle model that would not spin, but showed:

> "…low-frequency wallowing that masked departure. At the same time, the wallowing does not generate sufficiently rapid motion to excite inertia coupling and PSG. All pilots tend to continue fighting to maintain control well past full stall, incurring excessive altitude loss. However, if controls were released at any time, the aircraft would immediately go into a nose-low spiral and recovery by itself."

High-AOA characteristics were otherwise considered quite good, but the excessive loss of altitude was unacceptable: "Pilot commentary indicated the overall departure ratings were heavily influenced by altitude loss and mission phase." Specification of altitude loss, however, was deemed impractical.

AFWAL-TR-81-3108 also shows preference for an altitude-based metric:

> "Altitude loss per turn can vary drastically with different spin modes (e.g., steep versus flat), and a given vehicle may exhibit more than one spin mode. The allowable altitude loss, which is highly mission related (e.g., air-to-ground versus air-to-air), appears to be a more appropriate recovery metric than turns for recovery."

However, with a rate of descent in a spin roughly proportional to wing loading, W/S, it would seem extremely difficult for a high-W/S fighter to recovery in much less altitude than presently required. Ideally the altitude loss requirement would also be a function of altitude above the ground, since a PSG at 80,000 ft would not be as critical as one at 2,000 ft above the ground. Although air density variations exert some influence on the motions, such a requirement is not felt to be practical.

It is best if the same technique, or very similar techniques, can be used to recover from all post-stall gyrations, incipient spins, and developed spins. A recovery technique independent of the direction of motion—releasing or centering the controls for example,—is very desirable because pilots easily become disoriented in violent post-stall motions. Such recovery characteristics, however, may not be achievable without some automation. A "panic button" has been suggested.

The F-4 series of air vehicles serve as excellent examples of what is good and bad with this requirement. AFFDL-TR-70-155 summarizes a wide body of experience in spin testing of the F-4. The air vehicle was predicted by model tests to have steep erect and inverted oscillatory modes, as well as a flat spin mode. AFFDL-TR-70-155 quotes flight test reports concerning spin testing. For the F-4B:

> "A typical spin was initiated by applying pro-spin controls at the stall which resulted in the airplane yawing in the direction away from the applied aileron. After the initial yaw the airplane would pitch nose down to about 60º to 80º at the ¼-turn position followed by an increase in yaw rate. After ½ turn in yaw the airplane would pitch up to near level and in some cases 10º to 20º ANU, depending upon the energy conditions at entry. The yaw rate was usually at a minimum when the pitch attitude (and AOA) was at a maximum. The airplane was concurrently oscillating 60º in roll with no apparent relationship to pitch or yaw. The motions were extremely oscillatory for the first 2 to 3 turns. After 3 to 4 turns steady state conditions were approached and although the oscillations remained, the amplitude and period became constant.… Pro-spin controls were held for up to 4½ turns. The characteristics of the spin were similar for both left and right spins; however, each spin was different in some aspect from the others even under apparently identical entry conditions."

Standard recovery from incipient undeveloped spins was consistent and effective in all but flat spins. Also the recovery requirement was not met, since the pilot had to determine the direction of the motion and timing was critical:

> "The recovery technique used after one turn in the incipient stage and in the fully-developed spin was full aft stick, full rudder against the spin, and full aileron with the spin. This technique would generally affect recovery in ½ to 1½ turns…. The primary visual cue that recovery had been effected was the cessation of yaw. As the yaw rate stopped, the controls had to be neutralized rapidly to prevent a reversal. The time at which controls were neutralized was critical. If controls were neutralized before the yaw rate ceased, the airplane would accelerate back

> into the spin…, and if they were not neutralized within the one second after the yaw rate stopped, the spin direction would reverse… in most cases, the recovery was indistinct because of residual oscillations, particularly in roll. Even though the yawing had been arrested and the AOA was below stall the aircraft would roll up to 540º in the same direction as the terminated spin. The residual oscillations were easily mistaken for a continuation of the spin."

A flat spin led to loss of the air vehicle (figure 202). The air vehicle was stalled with throttles idle and pro-spin controls. It entered a post-stall gyration, but did not progress to an incipient spin. "After 15 seconds the pilot attempted to terminate the post-stall gyration by neutralizing the rudder and aileron and by placing the stick forward of neutral"; control motions in keeping with the requirement that the recovery not be dependent on determination of the direction of motion. However, according to AFFDL-TR-70-155:

> "A left yaw rate developed, and the airplane entered a left incipient spin. After 1-2 turns the oscillations diminished and the flat spin mode became apparent. Anti-spin controls were applied but had no significant effect on the spin characteristics. The drag chute was deployed at 33,000 ft, but again it streamed, did not blossom, and had no effect on the spin. At 27,000 ft the emergency spin recovery chute was deployed, but it also streamed. As a last resort the flight controls were cycled in an attempt to induce oscillations in the spin motions and/or to change the weight characteristics between the airplane and the spin chute. The only apparent effect of the control cycling was an increase in yaw rate to above 100º/sec."

These results serve to emphasize the importance of approaching spin testing with great care.

Recovery from the F-16 deep-stall (5.2.5.4 Requirement Lessons Learned) required both a manual pitch SAS override switch and proper application of longitudinal stick to "rock" the air vehicle out of the stall—an action which required the pilot to determine the direction of motion, albeit in pitch and not yaw.

High-AOA testing of the F-18 ("F/A-18A High Angle of Attack/Spin Testing") has uncovered spin modes not unlike those of the F-4:

> A low yaw rate spin was identified using asymmetric thrust to force the entry. It was characterized by yaw rates between 20º and 40º/sec, and angle of attack between 50º and 60º, a steep nose-low attitude, and fairly smooth pitch and roll rates.
>
> An oscillatory intermediate mode with yaw rates between 50º and 80º/sec and an angle of attack between 60º and 80ºg
>
> A smooth flat mode at 90º to 140º/sec yaw rate with an angle of attack between 80º and 85º

The latter two modes were entered by defeating the Control Augmentation System (CAS) and removing all feedback control limiting.

During these tests, 150 entries were attempted with over 100 resultant spins. Since the low-α mode could be entered with CAS on, a manual CAS defeat switch was installed to allow pilot access to maximum control authority for recovery. Using this switch and lateral stick into the spin, a single recovery technique was identified for all three spin modes.

The low-α mode spin has an aspect like the F-16's deep stall, and recovery with a CAS defeat switch is similar. Again, recovery from all three spin modes required determination of the direction of motion to apply lateral stick into the spin.

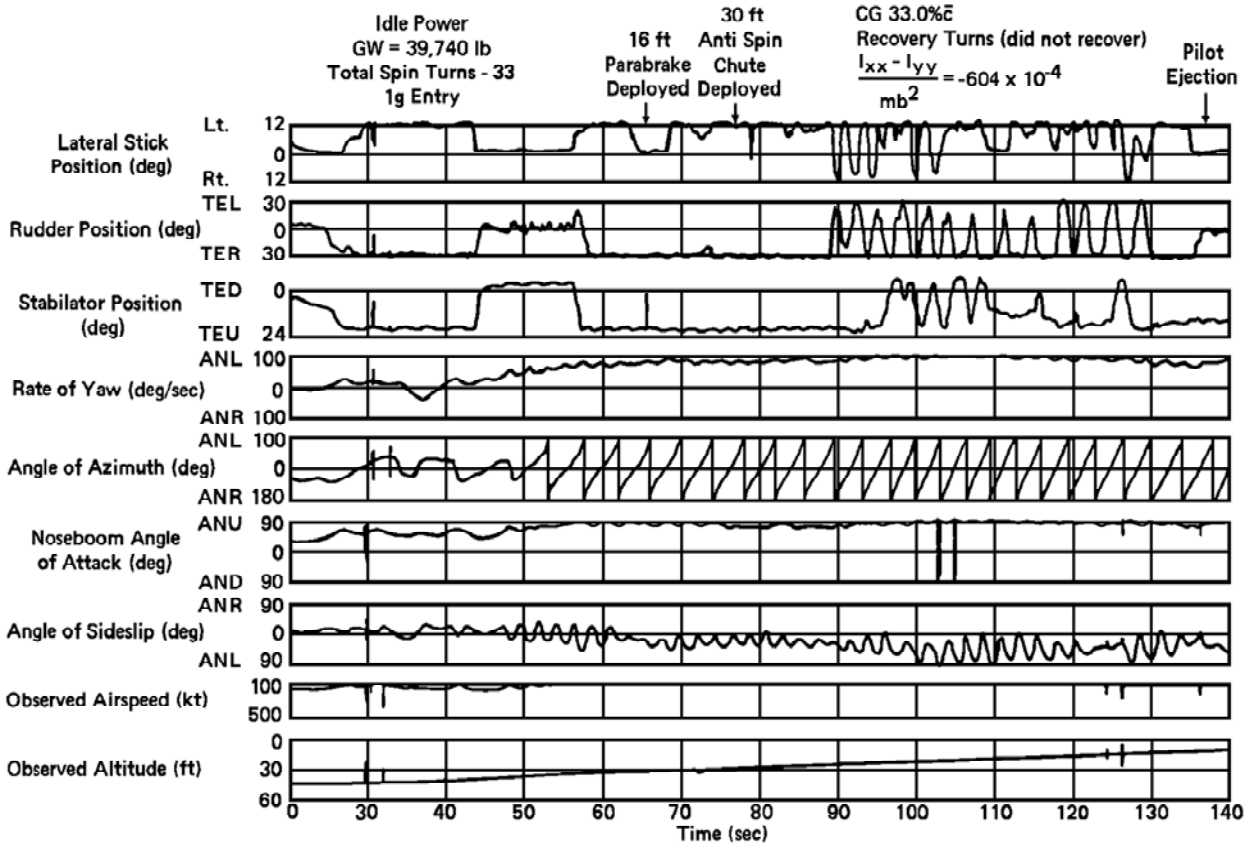


FIGURE 202. Left flat spin, F-4B (from AFFDL-TR-70-155).

The 40 lbs allowed for wheel forces is a carryover from MIL-F-8785C. It is based on the use of two hands, a rare occurrence in most flying tasks since one hand is on the throttle(s) during maneuvering. AFFDL-TR-72-141, a validation of MIL-F-8785B using a Class III air vehicle (P-3B) indicates pilot support for the limits for Class III air vehicles. The sidestick forces are based upon both the maximum forces on the F-16 movable stick (AFFTC-TR-79-10) and results of the USAF Test Pilot School evaluations (AFFDL-TR-79-3126). The forces chosen are 70-90 percent of the forces used.

The F-16 and F/A-18 have fly-by-wire control systems incorporating (respectively) manual pitch override or spin recovery mode switches which change the SCAS feedbacks. However, flight test experience has shown that employment of this type of technology can lead to difficulties associated with the anti-spin flight control system mode interfering with or delaying post-stall gyration spin recovery. As noted earlier, automatic engagement of the F-16 anti-spin stability augmentation system may have aggravated the deep stall condition. For the F/A-18, the flight control system is designed to automatically revert to the ASRM if in a spin. The pilot is given anti-spin control authority should he desire to use it—anti-spin control inputs are not automatically applied. During initial F/A-18 operational evaluation testing, an F/A-18 crashed in a low yaw rate spin because the ASRM did not engage and provide the pilot with full anti-spin control authority. Also, there were occurrences during which the special cockpit displays for spin recovery provided incorrect information. There are no specific requirements presently with regard to the safe operation of the automatic/manual post-stall recovery modes of a flight control system or of associated display operation. Some points to consider are:

Engagement/disengagement thresholds of automatic spin recovery flight control modes shall be designed such that they do not inhibit or prevent recovery.

Displayed recovery information shall always present the correct flight control system status (e.g., mode) and recovery control information.

A successful spin test was completed on the F-15 clean and with external stores. Recovery was defined as an absence of yaw rate and a steadily decreasing AOA at an AOA of 20º. Data was cut off. It was later found out that recovery to a safe airspeed in level controlled flight varied with store loading. When the air vehicle was configured with stores, it wallowed more during recovery and took significantly more altitude to regain flying speed than the clean configuration. Furthermore, the pilots recommend slower control inputs during the dive pullout than with the clean configuration. Such information shall be determined as a part of analysis and test, and incorporated into the pilot's manual.

VERIFICATION

Verification shall be by demonstration in the flight tests of 5.2.5. Proof of compliance in these demonstration tasks shall consist of pilot comments and time histories of the departure recoveries.

When the post-stall region is not banded by structural design considerations, flight testing is a necessity since it is difficult to define an accurate aerodynamic model for post-stall flight.

This requirement will be verified in flight test only for air vehicles that must be designed to withstand the forces of post-stall gyrations and spins. For other air vehicles the requirement is only to determine that post-stall and spin characteristics are satisfactory by appropriate wind tunnel, spin tunnel, or free-flight model testing and analysis. This should provide some confidence in the pilots' handbook material and thus help to save the air vehicles when they inadvertently get beyond prescribed flight limits. The requirement then has implications for design of the structure and other subsystems. The designer shall weight the benefits of assured recovery against any design penalties so as not to unduly compromise the air vehicle.

In addition to analysis, common verification techniques include high-AOA wind tunnel tests (e.g., in the NASA Ames 12-ft wind tunnel which has a rotary mount in addition to a high-AOA sting), free-flight model tests in NASA Langley full-scale tunnel, drop model tests with a more or less elaborate flight control system, and (where structural design permits) flight testing. Tests at NASA facilities, such as the Langley spin tunnel, of course need NASA approval. The procuring activity shall see that, contractually, results of all such testing are to be made available to their project office.

Detailed analysis and model testing shall precede flight verification to determine the characteristics of the particular design. Conditions investigated will be those of 5.2.5.5.

A wide variety of test philosophies has been applied to recent air vehicles. Of course the severity of control abuse is a function of air vehicle type and missions. There has been controversy in applying the qualitative guidance of the cancelled MIL-F-83691. For fighters, testing generally has progressed to the stage of finding ways to defeat any limiters, without turning them off. The F-18 test program went further, exploring limiters off behavior. The first high-AOA tests of the Mirage 2000 were made with the limiter off because the pilots did not trust limiters. However, the F-15 spun nicely after just manipulating the stick to trick the limiter into allowing large pro-spin control surface deflections. For the F-15, loading asymmetries (internal fuel, external stores) caused significant variations in spin characteristics.

Small changes in aerodynamic or inertial configuration have in some cases profoundly affected entry or recovery characteristics. While there are those who for safety rely completely on analysis and model tests to predict recoverability prior to flight tests (for example, the Mirage 2000 Mathe in AGARD-CP-333), in this country we have insisted on an additional recovery device for flight verification: a spin chute or spin recovery rockets. The latter have been used on the T-28 and supplemental spin testing on the F-100. Care must be taken that the recovery device does not change the air vehicle motion characteristics.

5.2.6 Carrier operations.

In addition to the preceding requirements, carrier-based air vehicles shall meet the requirements of 5.2.6.1 through 5.2.6.6.

REQUIREMENT RATIONALE

Carrier-based air vehicles are required to be launched from catapults, landed in arresting gear, and operated from aircraft carriers of a variety of sizes. The carrier-based air vehicle must have deck handling qualities and flying qualities compatible with the shipboard environment. Requirements 5.2.6.1 through 5.2.6.6 address carrier-specific operational requirements and flying qualities, and are not meant to be stand-alone requirements. Rather, they are meant to provide requirements above and beyond the requirements in this standard that apply to all fixed-wing military aircraft, whether they are CTOL, carrier-based, or V/STOL aircraft. They are meant to be treated as an extension of the Primary requirements (see 5.1) specifically for carrier operations.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There were no corresponding requirements in MIL-F-8785C or MIL-STD-1797A.

Requirements 5.2.6.1 through 5.2.6.6 apply only to carrier-based air vehicles.

REQUIREMENT LESSONS LEARNED

See Requirement Lessons Learned in 5.2.6.1 through 5.2.6.6.

VERIFICATION

Verification shall be via verification of requirements 5.1.1.1 and 5.1.2 using the tasks recommended in table LXXXVIII and by meeting the requirements of 5.2.6.1 through 5.2.6.6.

The air vehicle must be evaluated by pilots in high-gain, closed-loop tasks to insure the air vehicle has achieved the required Levels of flying qualities. For the most part, these tasks must be performed in actual flight. However, for conditions which are considered too dangerous to attempt in actual flight (e.g., certain flight conditions outside the ROTH, flight in Extraordinary atmospheric disturbances, etc.), the closed-loop evaluation task can be performed in a simulator.

Selected Air Vehicle Normal States shall be analyzed and evaluated in simulation and flight test for each applicable Flight Phase. The selected Air Vehicle Normal States shall include the loadings and configurations recommended below for each task plus a limited number of values of those parameters such as weight, moments of inertia, c.g. position, wing sweep, and throttle setting which vary continuously over a range of values during the Flight Phase. The values selected shall include the most critical values and the extremes encountered during the Flight Phase in question.

Verification of 5.1.1.1 (a), (b), and (d) and 5.1.2 for carrier operations shall be by flight test in the performance of appropriate carrier tasks from table LXXXVIII. These tasks shall be flown by test pilots at the flight conditions recommended below under various levels of atmospheric disturbances. Proof of compliance in these demonstration tasks will consist of pilot comments and Cooper-Harper (C-H) ratings. The comments and ratings should indicate that the flying qualities are no worse than the required Level of flying qualities for each combination of Air Vehicle State, Flight Phase, and environmental condition.

Verification of 5.1.1.1 (c) for carrier operations shall be by flight test to show that the air vehicle can be flown to particular flight conditions outside of the ROTH and return readily and safely to the ROTH or ROSH. The specific flight conditions to be evaluated shall be those flight conditions appropriate to carrier operations which are outside the ROTH but inside the ROHR and which are predicted by analysis and simulation to have the worst handling qualities or to be unrecoverable. Obviously, an emergency recovery system shall be installed for these tests, regardless of whether the condition is predicted to be recoverable or not—the purpose of flight testing is to verify analytical predictions. Further discussion of emergency recovery systems can be found in 5.2.5 Verification. Proof of compliance in these demonstrations will consist of pilot

comments. The pilot comments should indicate that the air vehicle can be readily and safely returned to the ROTH or ROSH and that no exceptional pilot inputs are necessary to recover the air vehicle.

For a discussion of how to use piloted evaluations as verification see 5.1.1.1 Verification. Selection of appropriate tasks from table LXXXVIII shall include consideration of the following:

During the requirements definition process, the procuring activity together with the contractor(s) and the responsible test organization, shall select several closed-loop tasks with which to evaluate the air vehicle in flight test. During system development, ground-based and in-flight simulations shall be used to get an initial appraisal of how well the air vehicle will perform the tasks in flight. The simulations can also be used to train the test pilots and refine the tasks, performance objectives, and test procedures.

Pilot comments shall be considered the most important data. A C-H rating is only a summary of observed flying qualities characteristics expressed as a single number. Pilot comments identify the specific deficiencies, if any, that must be corrected. Furthermore, the "long-look" technique filters the effects of deficiencies on the C-H rating because, over several runs, the pilot learns to compensate for some deficiencies. Since pilot comments are given for every run, the comments will identify all observed deficiencies, even those which can be compensated for. Comments on succeeding runs provide guidance on the pilot's ability to compensate for the deficiencies and the final C-H rating indicates the relative significance of these deficiencies. Therefore, pilot comments must be recorded and analyzed for every test run.

Time and cost constraints prohibit piloted evaluation of every task in every possible air vehicle configuration at every possible point in the Flight Envelope. The conditions that must be evaluated are the most common operating conditions, operating conditions critical to the mission of the air vehicle, and the worst-case conditions, especially those where the quantitative, open-loop flying qualities requirements are violated by wide margins. For air vehicles with multiple flight control modes, all mode transitions shall be evaluated at common, mission critical, and worst-case conditions, especially mode switches which are done automatically, as opposed to those deliberately switched by the pilot. Furthermore, the degradation due to atmospheric disturbances shall be demonstrated by evaluation at different levels of disturbances. Evaluation of the effect of Extraordinary atmospheric disturbances may be performed in ground simulation. When using simulation to predict the degradation of flying qualities due to Extraordinary atmospheric disturbances, it will be necessary to correlate C-H ratings gathered from simulation sessions in lower levels of turbulence with C-H ratings obtained from flight test for the same task in similar levels of turbulence.

The following paragraphs discuss some recommended tasks and some suggested performance objectives for each task.

TABLE LXXXVIII. Suggested handling qualities evaluation tasks for carrier operations.

Suggested Tasks	Suggested Performance Objectives
Catapult Launch	Desired Performance After launch, maintain flyaway AOA or pitch attitude within ±1°
	Adequate Performance After launch, maintain flyaway AOA or pitch attitude within ±1°
Case II Recovery	Desired Performance During penetration descent, maintain rate of descent within 4000 ±500 ft/min
	Adequate Performance During penetration descent, maintain rate of descent within 4000 ±500 ft/min
Carrier Approach (V_{MCA}, 180° Start and 90° Start)	Desired Performance During approach turn, capture and maintain bank angle within ±2° Capture and maintain approach AOA within ±1/2° Capture and maintain glideslope within ±1/2 ball Capture and maintain final approach heading within ±1° Touchdown on centerline within ±5 ft
	Adequate Performance During approach turn, capture and maintain bank angle within ±5° Capture and maintain approach AOA within ±1° Capture and maintain glideslope within ±1 ball Capture and maintain final approach heading within ±2° Touchdown on centerline within ±10 ft
Carrier Approach (V_{MCA}, Straight-in Approach, and IMC)	Desired Performance Capture and maintain approach AOA within ±1/2° Capture and maintain glideslope within ±1/2 ball Capture and maintain final approach heading within ±1° Touchdown on centerline within ±5 ft
	Adequate Performance Capture and maintain approach AOA within ±1° Capture and maintain glideslope within ±1 ball Capture and maintain final approach heading within ±2° Touchdown on centerline within ±10 ft
Bolter	Desired Performance Capture flyaway attitude (either AOA or pitch) within ±1°, with no more than one overshoot
	Adequate Performance Capture flyaway attitude (either AOA or pitch) within ±2°, with no more than two overshoots
Long Bolter	Desired Performance
	Adequate Performance
Waveoff (Constant AOA)	Desired Performance Maintain flyaway angle of attack within ±1/2° Maximum stick force during waveoff no more than 10 lbs
	Adequate Performance Maintain flyaway angle of attack within ±1° Maximum stick force during waveoff no more than 10 lbs
Waveoff (Constant Pitch)	Desired Performance Maintain flyaway pitch attitude within ±1/2° Maximum stick force during waveoff no more than 10 lbs
	Adequate Performance Maintain flyaway pitch attitude within ±1° Maximum stick force during waveoff no more than 10 lbs

Catapult Launch

The following task may be used to demonstrate that the aircraft has the flying qualities necessary to execute a catapult launch and flyaway. A hands-off catapult launch is highly desirable. No unusual or excessive control inputs shall be required of the pilot (no stick programming). When hands-off, pitch rates that are too high in either direction can be disconcerting to the pilot.

Set recommended longitudinal trim. Execute a standard, hands-off catapult launch. The airplane shall promptly capture the flyaway attitude, with little or no input from the pilot, and maintain either AOA or pitch. Perform representative clearing turn. Test conditions shall include each takeoff configuration; full-aft and full-forward c.g. locations; minimum, intermediate, and maximum gross weight; MIL and MAX A/B (as applicable); variable catapult endspeed in accordance with performance of the catapult capability (such as C-13 Mod 1).

Case II Recovery

The purpose of this task is to determine the airplane's compatibility with current recovery profiles and procedures and its interoperability with other ship-based platforms. In order to maintain quick and efficient recovery cycles, the CV and LHA/LHD NATOPS requires ship-based fixed-wing aircraft to perform a rapid penetration descent out of the marshal pattern. The aircraft must possess sufficient deceleration capability to attain the desired descent rate for the desired airspeed. Any significant deviation from this profile will alter the intervals between aircraft, resulting in an inefficient recovery cycle with several foul-deck waveoffs. In addition, the airplane must be able to quickly slow to landing speed following a carrier break turn.

The maneuver begins in the marshal pattern at 15,000 ft, 30 nm from the ship (angels + 15). Begin the Case II recovery with a 250-kt, 4000-fpm penetration descent to a platform altitude of 5,000 ft. Reduce the rate of descent to 2,000 fpm to enter the carrier pattern at 800 ft. Initiate a break over the bow at an airspeed of 250 to 450 KCAS. After the carrier break, decelerate to onspeed AOA prior to the abeam position on downwind. Test conditions shall include clean loading and minimum recovery weight.

Carrier Approach (V_{MCA}, 180º start)

The purpose of this task is to determine the flying qualities during power approach to a carrier landing. This maneuver recreates the standard Case I recovery starting from the downwind leg to evaluate flying qualities during the turn to final and the final approach leg.

The nominal approach start conditions are with the airplane configured for landing, 1 nautical mile abeam the target touchdown point, 600 ft AGL, onspeed AOA, and on a course 180º off final approach heading. Follow the Case I VFR pattern to perform a carrier recovery. Test conditions shall include maximum and minimum carrier landing weights, full-aft and full-forward c.g. locations, clean and minimum lateral-directional stability loadings, a 3.5º glideslope with minimum recovery WOD, a 4º glideslope with 35-kt WOD, and manual and autothrottles, if applicable.

Carrier Approach (V_{MCA}, 90º start)

The purpose of this task is to determine the flying qualities during power approach to a carrier landing. This maneuver recreates a standard Case I approach with an overshooting start. A prompt approach turn followed by several lineup corrections are required to enable recovery.

Nominal approach start conditions are with the airplane offset 0.6 nm from, and on a heading perpendicular to, the extended centerline of the angle deck at a distance 1.1 nm aft of the target touchdown point. Immediately initiate a final approach turn. Continue approach to execute a shipboard recovery. Test conditions shall include maximum and minimum carrier landing weights, full-aft and full-forward c.g. locations, clean and minimum lateral-directional stability loadings, a 3.5º glideslope with minimum recovery WOD, a 4º glideslope with 35-kt WOD, and manual and auto-throttles, if applicable. Initial altitude shall be variable between a centered, high, or low ball. Initial airspeed shall be variable within ±10 kts of the recommended approach speed.

Carrier Approach (V_{MCA}, Straight-in Approach)

The purpose of this task is to determine the flying qualities during power approach to a carrier landing. This maneuver highlights the subtasks of AOA, glideslope, and lineup control required to satisfactorily perform a shipboard recovery.

Nominal approach start conditions are with the airplane trimmed in desired rate of descent, on heading, at a distance of 0.75 nm aft of the target touchdown point. Immediately correct for deviations in airspeed, altitude, and lineup. Continue approach to execute a shipboard recovery. Test conditions shall include maximum and minimum carrier landing weights, full-aft and full-forward c.g. locations, clean and minimum lateral-directional stability loadings, a 3.5º glideslope with minimum recovery WOD, a 4º glideslope with 35-kt WOD, and manual and auto-throttles, if applicable. Initial altitude shall be variable between a centered, high, or low ball. Initial airspeed shall be variable within ±10 kts of the recommended approach speed. Initial lineup deviation shall be variable ±140 ft measured perpendicular to the extended centerline of the angle deck.

Carrier Approach (IMC)

The purpose of this task is to determine the flying qualities during power approach to a carrier landing. This maneuver highlights the subtasks of AOA, glideslope, and lineup control required to satisfactorily perform a shipboard recovery.

The nominal approach start conditions are in IMC with the airplane trimmed on glideslope at the desired rate of descent, on heading and lineup, at a distance of 1.25 nautical miles aft of the target touchdown point. Follow landing aids (ILS needles or equivalent) to one of three breakout conditions:

1) 0.5 nautical miles from touchdown, altitude corresponding to 1-ball high
2) 0.75 nautical miles from touchdown, altitude corresponding to 1-ball low
3) 0.75 nautical miles from touchdown, 140-ft offset from, and on a heading parallel to, the extended centerline of the angle deck

Upon breakout, transition to a visual approach, correcting immediately for deviations in altitude and lineup. Continue the approach to execute a shipboard recovery. Test conditions shall include maximum and minimum carrier landing weights, full-aft and full-forward c.g. locations, clean and minimum lateral-directional stability loadings, a 3.5º glideslope with minimum recovery WOD, a 4º glideslope with 35-kt WOD, and manual and auto-throttles, if applicable.

Bolter

The following task may be used to demonstrate that the aircraft has the performance and flying qualities necessary to execute bolters. Following a bolter, it is necessary to fly off the end of the angle deck with no apparent settle off the bow. No unusual or excessive control inputs shall be required of the pilot.

Perform hook-up touch-and-go landing. Attain flyaway attitude before reaching the end of the angle deck. Test conditions shall include each approach configuration, maximum and minimum carrier-landing weights, full-aft and full-forward c.g. locations, MIL and MAX A/B thrust, a 3.5º glideslope with minimum recovery WOD, and a 4º glideslope with 35-kt WOD.

Long Bolter

The following task may be used to assess longitudinal control power available to prevent excessive settle-off-bow following a long bolter. Sufficient control power must be available to arrest the nose down pitching moment induced by the airplane landing on its main gear, such that the airplane can safely fly away. Some settle off the bow can be expected.

Perform hook-up touch-and-go landing. Target touchdown shall be well beyond the landing area, such that the main gear contact the flight deck, but the airplane exits the angle deck before the nose gear touches down. Test conditions shall include each approach configuration, maximum and minimum carrier landing weights, full-aft and full-forward c.g. locations, MIL and MAX A/B thrust, a 3.5º glideslope with minimum recovery WOD, and a 4º glideslope with 35-kt WOD.

Waveoff (Constant AOA)

The following task may be used to demonstrate that the aircraft has the performance and flying qualities necessary to execute a waveoff using constant AOA technique. For safe carrier operations, adequate waveoff flying qualities and performance are required. The waveoff shall not require an excessive change in longitudinal control deflection to maintain trim AOA. However, in high thrust to weight (T/W) airplanes a constant AOA waveoff maneuver may result in a large increase in pitch attitude, which consequently would increase the hook-to-eye distance and decrease the hook clearance over the ramp. A late waveoff of this type may significantly increase the likelihood of in-flight arrestments and/or hook slaps.

The following two tasks describe the procedures for waveoff (constant AOA):

Task A: Nominal approach
Establish approach on nominal glideslope at onspeed AOA. At waveoff command, select desired waveoff thrust setting. Maintain onspeed AOA until positive rate of climb is achieved. Do not retrim during the task.

Task B: High-come-down:
Establish approach on nominal +1 ball high glideslope at onspeed AOA. Select Idle for 2 sec. Select desired waveoff thrust setting. Maintain onspeed AOA until positive rate of climb is achieved. Do not retrim during the task.

Test conditions shall include maximum and minimum carrier landing weights, full-aft and full-forward c.g. locations, MIL and MAX A/B thrust, a 3.5º glideslope with minimum recovery WOD, and a 4º glideslope with 35-kt WOD.

Waveoff (Constant Pitch)

The following task may be used to demonstrate that the aircraft has the performance and flying qualities necessary to execute a waveoff using constant pitch technique. For safe carrier operations, adequate waveoff flying qualities and performance are required. A constant pitch attitude waveoff will generally provide less performance than if the constant AOA technique were used. However, the constant pitch technique may be warranted to preclude a large increase in pitch attitude which may create a high likelihood for in-flight arrestments and/or hook slaps.

The following two tasks describe the procedures for waveoff (constant pitch)

Task A: Nominal approach
Establish approach on nominal glideslope at onspeed AOA. At waveoff command, select desired waveoff thrust setting. Maintain constant pitch attitude until positive rate of climb is achieved. Do not retrim during the task.

The following two tasks describe the procedures for waveoff (constant pitch)

Task A: Nominal approach
Establish approach on nominal glideslope at onspeed AOA. At waveoff command, select desired waveoff thrust setting. Maintain constant pitch attitude until positive rate of climb is achieved. Do not retrim during the task.

Task B: High-come-down:
Establish approach on nominal + 1-ball high glideslope at onspeed AOA. Select Idle for 2 sec. Select desired waveoff thrust setting. Maintain constant pitch attitude until positive rate of climb is achieved. Do not retrim during the task.

Test conditions shall include maximum and minimum carrier landing weights, full-aft and full-forward c.g. locations, MIL and MAX A/B thrust, a 3.5º glideslope with minimum-recovery WOD, and a 4º glideslope with 35-kt WOD.

Verification of 5.1.2 shall be by simulation or flight test in the performance of the tasks of table LXXXVIII under various levels of atmospheric disturbances, including the carrier landing disturbance model described below. This model shall be used in analysis and piloted simulations to determine aircraft control response and path control accuracy during carrier landing operations. This model supplements but does not replace the low-altitude disturbance model of 5.1.2.

For ship-based operations, the definitions of Common, Uncommon, and Extraordinary steady winds shall be as shown on figure 203 in lieu of the definitions given in 5.1.2 Verification. The steady wind intensities shown on figure 203 are a combination of ambient steady wind and ship speed. Consequently, these wind intensities are applicable only to requirements for ship-based operations.

The terminal approach carrier disturbance model shall be used during simulation of the last ½ mile of the carrier approach. The u velocity component is aligned with the wind over deck. Total disturbance velocities are computed by adding segments caused by:

1) random free-air turbulence (u_1, v_1, and w_1)
2) steady ship-wake disturbance (u_2 and w_2)
3) periodic ship-motion-induced turbulence (u_3 and w_3)
4) random ship-wake disturbance (u_4, v_4, and w_4).

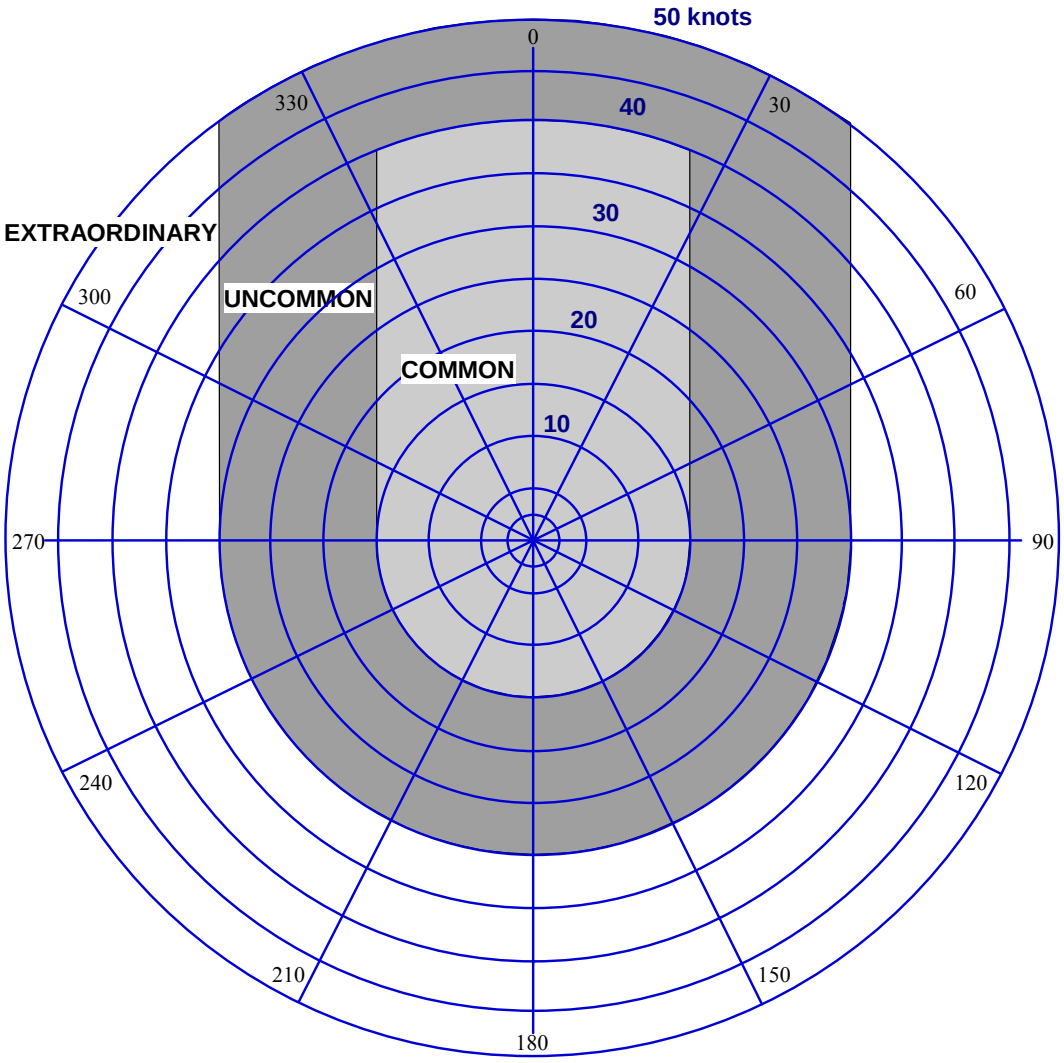


FIGURE 203. Steady-state wind intensities for ship-based requirements.

The total air disturbance components (u_g, v_g, and w_g) are then computed as:

$$u_g = u_1 + u_2 + u_3 + u_4$$

$$v_g = v_1 + v_4$$

$$w_g = w_1 + w_2 + w_3 + w_4 .$$

The input to all of the random disturbance filters shall be generated by filtering the wide-band Gaussian output of zero-mean, unit-variance random-number generators.

The free-air turbulence components which are independent of aircraft relative position are represented by filtering the output of white noise generators described above to produce the following spectra:

$$\Phi_{u_1}(\Omega) = \frac{200}{1 + (100\,\Omega)^2} \qquad (\text{ft/sec})^2 \text{ per rad/ft}$$

$$\Phi_{v_1}(\Omega) = \frac{939\left[1 + (400\,\Omega)^2\right]}{\left[1 + (1000\,\Omega)^2\right]\left[1 + \left(\frac{400}{3}\,\Omega\right)^2\right]} \qquad (\text{ft/sec})^2 \text{ per rad/ft}$$

$$\Phi_{w_1}(\Omega) = \frac{71.6}{1 + (100\,\Omega)^2} \qquad (\text{ft/sec})^2 \text{ per rad/ft}$$

The steady components of the carrier airwake consist of a reduction in the steady wind and a predominant upwash aft of the ship which are functions of range. Figure 204 illustrates the steady wind functions $u_2/V_{w/d}$ and $w_2/V_{w/d}$ as functions of position aft of the ship center of pitch.

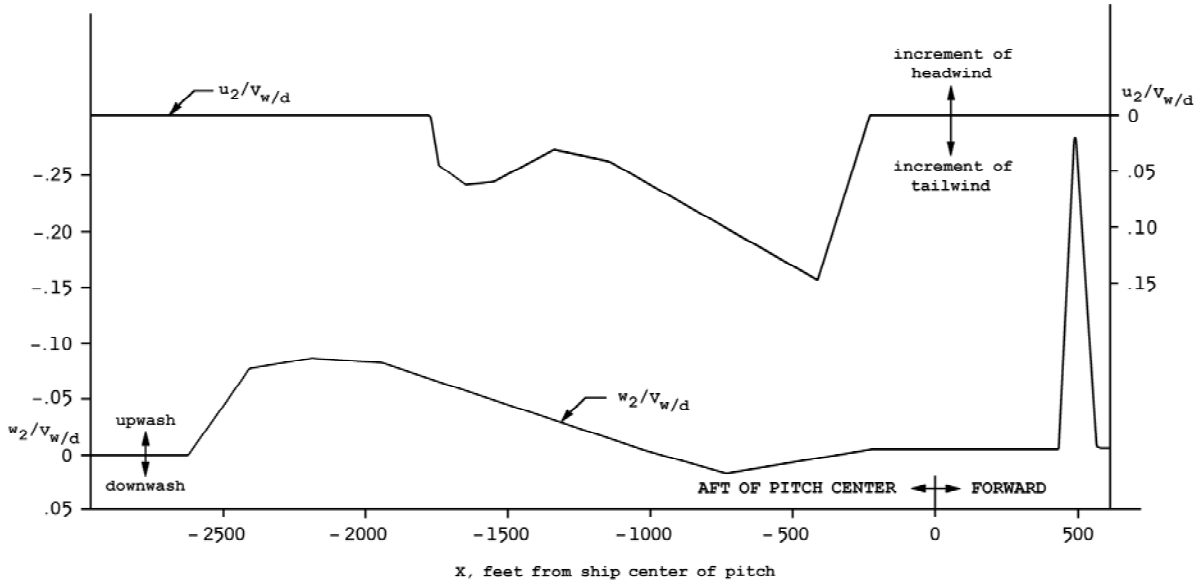


FIGURE 204. CV ship burble steady wind ratios.

The periodic component of the wake varies with ship pitching frequency, pitch magnitude, wind over deck, and aircraft range. These components are computed as follow:

$$u_3 = \theta_s \, V_{w/d} \, (2.2 + 0.0009 \, X) \, C$$

$$w_3 = \theta_s \, V_{w/d} \, (4.98 + 0.0018 \, X) \, C$$

$$C = \cos \left\{ \omega_p \left[t \left(1 + \frac{V - V_{w/d}}{0.85 \, V_{w/d}} \right) + \frac{X}{0.85 \, V_{w/d}} \right] + \Phi \right\}$$

where: ω_p = ship pitch frequency, rad/sec
θ_s = ship pitch amplitude, rad
Φ = random phase, rad.

The u component is set to zero for X < -2236 ft, and the w component is set to zero for X < -2536 ft.

The ship-related random velocity components are computed by filtering the white noise as follows:

$$u_4 = \frac{\sigma_{u_4}(X) \sqrt{2 \, T_{u_4}(X)} \, (\text{Input})}{T_{u_4}(X) \, j\omega + 1}$$

$$w_4 = v_4 = \frac{0.035 \, V_{w/d} \sqrt{6.66} \, (\text{Input})}{3.33 \, j\omega + 1}$$

where: $\sigma_{u_4}(X)$ = rms amplitude – ft/sec (figure 205)

$T_{u_4}(X)$ = time constant – sec (figure 205)

$$\text{Input} = \left[\begin{matrix} \text{Random number} \\ \text{output} \end{matrix} \right] \left[\frac{j\omega}{j\omega + 0.1} \right] \sin(10 \pi t) \ .$$

In addition to the atmospheric disturbances associated with degraded environmental conditions, simulation of Flight Phase Category C operations from a ship shall include the ship motions resulting from these conditions. Common environmental conditions shall include the ship motion resulting from sea states up to sea state 3. Uncommon environmental conditions shall include the ship motion resulting from sea states up to sea state 5. Extraordinary environmental conditions shall include the ship motion resulting from sea states up to sea state 7. Environmental conditions worse than sea state 7 need not be considered for Category C operations from a ship.

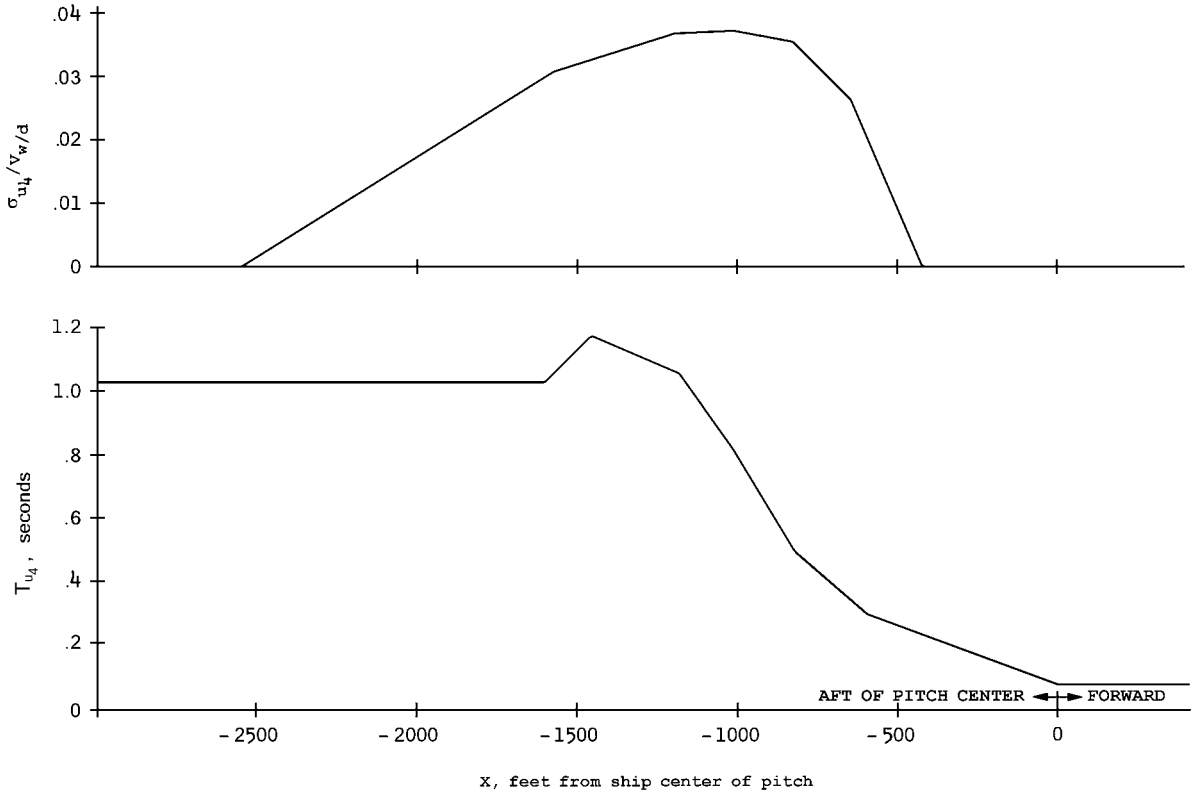


FIGURE 205. u-component burble time constant and variance.

Guidance for the magnitude of ship motions that should be used in a simulation of Category C operations are presented in table LXXXIX. The ship motions may be represented by a sum of three sinusoidal terms in the following proportions:

pitch and heave - 70% at 0.3 rad/sec, 20% at 0.7 rad/sec, 10% at 0.9 rad/sec
roll and yaw - 70% at 0.6 rad/sec, 20% at 1.2 rad/sec, 10% at 1.8 rad/sec .

The corresponding Common, Uncommon, and Extraordinary atmospheric disturbances shall be used simultaneously with the ship motions.

TABLE LXXXIX. Ship motions (degrees or feet).

Condition	Pitch	Roll	Heave	Yaw
Common	± 2	± 5	± 5	
Uncommon	± 3	± 7	± 10	
Extraordinary	± 4	± 10	± 15	

5.2.6.1 Deck handling.
The air vehicle shall have ground handling characteristics that allow operation from the constricted spaces aboard ships in degraded environmental conditions.

REQUIREMENT RATIONALE

Carrier-based air vehicles are required to taxi expeditiously, in confined spaces with various obstacles, and execute precise lineup on the catapult. The deck handling characteristics of the air vehicle can have a significant effect on the launch cycle time.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

This requirement applies only to carrier-based air vehicles.

REQUIREMENT LESSONS LEARNED

Factors that affect deck handling include power required to initiate air vehicle movement, engine acceleration characteristics, braking characteristics, pivoting of wheels, overshoots, steering effectiveness, and turn radius. Precise directional control and speed control are essential for shipboard operations.

VERIFICATION

Verification shall include shore-based evaluations followed by testing aboard an aircraft carrier. Deck handling characteristics shall be demonstrated when operating in the tight confines of the carrier hangar and flight deck, during flight deck taxiing, during shipboard long taxi, while performing final alignment on the catapult and tight turns, during low speed taxi on a pitching deck, and when clearing the landing area following shipboard arrested landings.

Shorebased evaluations will allow for a preliminary evaluation of ground handling but shipboard verification is required to confirm that the deck handling characteristics of the air vehicle are actually compatible with the carrier environment, which includes deck surfaces, confined spaces, obstacles, degraded weather conditions, and ship movement.

All shipboard testing shall strive to simulate rapid, expeditious operations aboard the aircraft carrier to evaluate handling qualities under the most representative conditions.

During initial shipboard trials of the F/A-18A, the air vehicle was discovered to be prone to roll back when taxiing at very low speed on a pitching deck. The pilot had to use continuous throttle inputs to prevent the air vehicle from rolling backwards and subsequent brake input to maintain controlled low speed. Precise speed control utilizing this technique during catapult hook-up and maneuvering close to the edge of the flight deck was complicated by imprecise brake pedal pressure feedback cues to the pilot. Objectionable time lag between pilot braking action and brake system

response during carrier deck taxi operations combined with excessive sensitivity of nose wheel steering around neutral to create a deficiency in deck handling characteristics.

5.2.6.2 Catapult launch.
With __(1)__ wind-over-deck (WOD), Level 1 flying qualities shall be achieved during catapult launches from all in-service catapult types, for all operating weights and center of gravity location combinations, subject to constraints on catapult minimum end speed. Sink off bow shall be no more than __(2)__ and pitch rate shall be no more than __(3)__ following catapult launch.

REQUIREMENT RATIONALE

The carrier-based air vehicle must be suitable for catapult launching from all catapult positions on various classes of aircraft carriers under operationally-representative environmental conditions. Flying qualities shall be satisfactory throughout the catapult launch sequence and subsequent maneuvering.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Fill in blank (1) with the minimum acceptable WOD for the class ship(s) from which the air vehicle will operate.

Fill in blank (2) with the maximum acceptable distance that the air vehicle shall sink (in feet) following catapult launch.

Fill in blank (3) with the maximum acceptable pitch rate following catapult launch.

This requirement applies only to carrier-based air vehicles.

REQUIREMENT LESSONS LEARNED

Experience has demonstrated that pilots prefer stick-free catapult launches where control inputs are not required during the catapult launch and pilot input is only required after launch to correct any excessive rates, the angle of attack, or sink off bow.

VERIFICATION

Simulation and shore-based testing shall be used to predict the minimum airspeed for catapult launches and to demonstrate the flying qualities characteristics approaching the minimum airspeed predicted prior to flying in the carrier environment. The minimum speed to maintain acceptable flying qualities and low speed handling characteristics shall be verified in the carrier environment. Flying qualities shall be evaluated during shipboard catapult launches from each unique catapult type and location (waist and bow). The effects of lateral and longitudinal center

of gravity, excess airspeed, and crosswinds on longitudinal trim requirements, rotation and flyaway characteristics during catapult launching shall be evaluated.

Catapult launches must be safely demonstrated on shorebased equipment prior to deploying in the more hazardous environment of the carrier. The compatibility of the air vehicle must be demonstrated for all of the carriers and catapults from which the air vehicle will be expected to operate.

The rotation to flyaway attitude shall be such that settle off the bow is kept to a minimum while avoiding excessively high pitch rates which can be disconcerting and disorienting to the pilot. Dynamic overshoots of the flyaway attitude shall be minimized, and any resultant automatic nose-down correction shall be small and not uncomfortable for the pilot. The effects of stored energy in the nose and main landing gear shall not cause undesirable pitch characteristics. Lateral-directional flying qualities shall allow an aggressive clearing turn shortly after launch. It shall not be necessary to retrim the air vehicle rapidly or to use large longitudinal control movements immediately after launch.

5.2.6.3 Carrier approach and landing.
Level 1 flying qualities shall be achieved during approach and landing on carrier decks for all recovery weights, with airspeed slow enough to require a WOD of no more than __(1)__.

REQUIREMENT RATIONALE

Flying qualities during approach and landing are important in preventing engagement at speeds beyond arresting gear capability, ensuring touchdown within the narrow landing area is consistently achievable, and responding to turbulence around the carrier and pitching and rolling of the carrier deck. The air vehicle flying qualities shall allow the air vehicle to integrate satisfactorily with existing carrier recovery approach and landing procedures and be compatible with normal recovery patterns. Flying qualities during the carrier approach and landing phase shall be satisfactory to allow optimum shipboard recovery with reasonable pilot workload.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Fill in blank (1) with the maximum acceptable WOD for the class ship(s) from which the air vehicle will operate.

This requirement applies only to carrier-based air vehicles.

REQUIREMENT LESSONS LEARNED

Experience has shown that the best approach characteristics in the carrier environment are dependent on the stability of angle of attack and attitude and the characteristics of engine response. The visual cueing provided to the pilot is also important in achieving satisfactory

handling qualities. The head-up-display has proven to be an extremely helpful tool in the carrier approach and landing environment. Important factors that affect the characteristics of the approach are the field of view, lineup control, gust sensitivity, and flight control input required with power application.

VERIFICATION

Flying qualities during the carrier approach and landing phase shall be analyzed using simulation, demonstrated using shorebased facilities, and finally verified through shipboard testing. General approach and landing characteristics shall be evaluated during approach and arrested landings, initially in the power approach configuration in level flight and on the glide path during field carrier landing practice. Qualitative evaluation tasks that shall be used to evaluate approach and landing characteristics in the carrier environment include airspeed control, heading control, altitude control, angle of attack control, roll attitude control, heading capture, lineup control, and glideslope control. All applicable types of arrested landings shall be demonstrated including on center, off center, and free flight engagement. The flying qualities shall be evaluated qualitatively over the speed band while making glide path corrections by changing rate of descent at constant thrust and by varying thrust while maintaining constant angle of attack.

The shipboard environment is inherently more dangerous than shore-based operations. Air vehicle shipboard suitability testing shall be preceded by a) simulation analysis to determine minimum speeds and associated flying qualities at the minimum speeds, and b) shore-based testing to evaluate the air vehicle in a nearly-representative environment. The final determination of optimum approach and landing technique and verification of flying qualities can only be determined aboard carriers.

Testing shall confirm suitability of recommended approach angle of attack and associated airspeeds for the approved landing gross weights. The glideslope, angle of attack, and lineup must be easily controllable within precise limits. Airspeed and altitude control on the downwind leg shall be adequate to ensure the proper interval between air vehicles and to increase the likelihood of initiating the final approach at the correct speed and glideslope. Bank control during the approach turn shall be adequate to minimize lineup errors at rollout. Deviations from glideslope shall be easy to correct to maintain sufficient clearance above the ramp and to touch down repeatably within the arresting wires. Control of angle of attack is critical to prevent engagement at speeds beyond arresting gear capability, and to prevent excursions into regions of potentially poor flying qualities. Lineup control is critical to ensure touchdown within the narrow landing area. The flying qualities, especially glide path control, shall be demonstrated to remain satisfactory when subjected to the considerable turbulence aft of the ship, unpredictable roll gusts, and the pitching and rolling carrier deck. All normal and emergency approach configurations shall be considered. Verification must include the effects of minimum recovery headwind, wind-over-deck as high as possible to evaluate burble effects, crosswind recovery, different glide slope angles, and different high-lift device positions.

Glide path corrections are typically made with initial longitudinal control inputs, therefore, it is desirable that the air vehicle have maneuvering capability at a constant thrust setting for small changes in angle of attack. For making large corrections to glide path, it is desirable that the

change in thrust required decrease for an increase in angle of attack. The typical correction technique is to correct to the glide slope with longitudinal control, readjusting to the approach angle of attack and then adjusting thrust to correct for the original erroneous thrust setting so that rapid glide path corrections are possible. Experience shows that airspeed instability is the most undesirable single characteristic in the carrier approach.

5.2.6.4 Bolter.
From a 4º glideslope at approach speed, it shall be possible to achieve nosewheel liftoff and attain flyaway attitude by the end of the angle deck, assuming the arresting hook just misses the last wire on the classes of carriers from which it is required to operate.

REQUIREMENT RATIONALE

Adequate flying qualities must exist to allow the air vehicle to continue flight when the arresting hook does not engage the arresting cable, because of hook bounce or touchdown outside of the landing area, when attempting an arrested landing.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

This requirement applies only to carrier-based air vehicles.

REQUIREMENT LESSONS LEARNED

Longitudinal control power shall be sufficient to offset the nose-down pitching moment caused by the main landing gear touchdown. A preliminary guideline developed by a joint NASA/Navy test team suggests that the air vehicle should be able to generate a pitch acceleration of at least ±0.20 rad/sec/sec within the first second of pilot maximum longitudinal stick command, at a dynamic pressure for aerodynamic stall, the most critical center of gravity, and with control effectors deflected to maintain stability. The pilot shall be able to capture the flyaway attitude easily, with no unusual or excessive control inputs and negligible pilot-in-the-loop oscillation tendency.

VERIFICATION

Flying qualities during a bolter shall be evaluated during shore-based and shipboard touch-and-go landings in all landing flap configurations and with simulated single engine failure. Bolters shall be performed with the air vehicle in an operationally-representative loading which produces the most critical nose-heavy moment and with thrust at, or below, an intermediate setting. Bolter technique shall employ aggressive use of aft stick to capture pitch attitude for flyaway.

Flying qualities during a bolter must be demonstrated in the shipboard environment in order to assess the impact of degraded environmental conditions, ship motion, and the variation of pilot gain and technique due to increased risk of hazard to the air vehicle and pilot.

Flyaway attitude shall be reached by the end of the angle deck with a combination of fuel and stores loading which produces the most critical nose-heavy moment with thrust not exceeding an intermediate setting. A normal bolter is defined as one where the entire angle deck landing area beyond the last crossdeck pendant is available for the maneuver. There shall be no tendency for pitch oscillations during the bolter. Rotation rates shall be uniform and predictable. There shall not be any tendency for air vehicle overrotation or pilot-in-the-loop oscillations. Adequate bolter performance must be demonstrated for carriers with the shortest distance between the last available arresting cable and the angle deck round down at air vehicle forward center of gravity positions with minimal wind-over-deck.

5.2.6.5 Waveoff.
The air vehicle shall possess __(1)__ flying qualities during waveoff in order to enable timely and safe termination of a shipboard approach. Loss of altitude following waveoff initiation shall be no more than __(2)__.

REQUIREMENT RATIONALE

The air vehicle is required to fly out of the carrier approach pattern due to foul deck or incorrect glide path to the ship and, therefore, must possess the flying qualities necessary for premature termination of a shipboard landing approach.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommend value for blank (1): Level 1

Recommend value for blank (2): 30 ft

This requirement applies to only carrier-based air vehicles.

REQUIREMENT LESSONS LEARNED

The altitude lost during a waveoff and the time required to regain a positive rate of climb shall be minimized to permit waveoffs in close proximity to landing. The pilot shall be able to capture the flyaway attitude easily, with no unusual or excessive control inputs and no tendency for pilot-in-the-loop oscillations. Automatic or manual configuration or control law changes shall not cause any undesirable response.

VERIFICATION

Flying qualities during a waveoff shall be evaluated during shore-based and shipboard approaches to landing in all landing flap configurations and with simulated single engine failure. Waveoff flying qualities shall be demonstrated through attitude captures and angle of attack control tasks. Safe termination of a shipboard approach shall be demonstrated by maintaining

the approach angle of attack throughout the maneuver until a positive rate of climb is established or maintaining constant pitch attitude to eliminate the increase in pitch attitude and the likelihood of an in-flight engagement. Waveoff shall be demonstrated for angles of attack above the recommended approach angle of attack.

Flying qualities during a waveoff must be demonstrated in the shipboard environment in order to assess the impact of degraded environmental conditions, ship motion, and the variation of pilot gain and technique due to increased risk of hazard to the air vehicle and pilot.

5.2.6.6 Single engine failure (multi-engine air vehicles).
At the minimum catapult end airspeed, in configuration for power approach, and 5 kts below the airspeed for power approach for the landing conditions, the dynamic response following critical engine failure shall not prevent the pilot from controlling the air vehicle with reasonable inputs to reach steady state single engine flight conditions.

REQUIREMENT RATIONALE

A multi-engine carrier-based air vehicle must have adequate flying qualities to allow for safe completion or termination of a catapult launch or shipboard approach and landing following a single engine failure.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

This requirement applies only to carrier-based air vehicles with more than one engine.

REQUIREMENT LESSONS LEARNED

The air vehicle shall be safely controllable throughout the motions ensuing from failure of the most critical engine. Following the transients, the rudders and lateral control surfaces shall be capable of holding zero yawing and rolling velocities, while achieving and maintaining straight flight parallel to the centerline of the angle deck. During the waveoff, angle of attack shall be maintained constant at the original angle of attack until the target pitch attitude is attained. Control forces required shall not be excessive.

VERIFICATION

One-engine-out flying qualities during waveoff and bolters shall be demonstrated during shore-based and shipboard touch-and-go landings in all appropriate configurations, thrust settings on the operating engine(s), and gross weights from a stabilized approach on glide slope. Flying qualities during a dynamic simulated engine failure shall be approximated during simulated one-engine-out operations at a stabilized catapult takeoff flyaway attitude.

Since a dynamic single engine failure will likely end in a bolter or waveoff, it must be demonstrated that these maneuvers can be safely executed. One-engine-out operations can be adequately simulated with the critical engine retarded to idle.

5.2.7 V/STOL specific requirements.
In addition to other requirements in this standard, V/STOL air vehicles shall meet the requirements of 5.2.7.1 through 5.2.7.2.2.

REQUIREMENT RATIONALE

Requirements 5.2.7.1 through 5.2.7.2.2 address V/STOL specific operational requirements and flying qualities, and are not meant to be stand-alone requirements. Rather, they are meant to provide requirements above and beyond the requirements elsewhere in this standard that apply to all fixed-wing military aircraft, whether they are CTOL, carrier-based, or V/STOL aircraft. More specifically, for all airspeeds above the speed at which the V/STOL aircraft has fully converted to wing-borne flight (the conversion airspeed), all applicable portions of this entire document apply. Although 5.2.7.1 through 5.2.7.2.2 are meant to address requirements and verification techniques for flight at all speeds below the conversion airspeed (i.e., the powered-lift flight regime), there are still many other requirements in this standard, outside of 5.2.7.1 through 5.2.7.2.2, which are applicable and pertinent to flight at speeds below the conversion speed.

REQUIREMENT GUIDANCE

These requirements flow down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There were no corresponding requirements in MIL-F-8785C or MIL-STD-1797A.

These requirements apply only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

See MIL-F-83300 and AFFDL-TR-70-88.

VERIFICATION

Verification shall be via verification of requirements 5.1.1.1 and 5.1.2 using the tasks recommended in table XC and by meeting the requirements of 5.2.7.1 through 5.2.7.2.2. The rationale for verification of V/STOL operational requirements stems from Mission Area Assessments, Mission Need Statements, Concept of Operations and other documentation from USMC, US Army, and other armed services requiring V/STOL capabilities to perform their missions.

Verification of V/STOL flying qualities requirements is based on NASA Technical Paper 3356, NASA TM 104021, and NADC-82146-60.

The air vehicle must be evaluated by pilots in high-gain, closed-loop tasks to insure the air vehicle has achieved the required Levels of flying qualities. For the most part, these tasks must be performed in actual flight. However, for conditions which are considered too dangerous to attempt in actual flight (e.g., certain flight conditions outside the ROTH, flight in Extraordinary atmospheric disturbances, etc.), the closed-loop evaluation task can be performed in a simulator.

Selected Air Vehicle Normal States shall be analyzed and evaluated in simulation and flight test for each applicable Flight Phase. The selected Air Vehicle Normal States shall include the loadings and configurations recommended below for each task plus a limited number of values of those parameters such as weight, moments of inertia, c.g. position, wing sweep, and throttle setting which vary continuously over a range of values during the Flight Phase. The values selected shall include the most critical values and the extremes encountered during the Flight Phase in question.

Verification of 5.1.1.1 (a), (b), and (d) and 5.1.2 for V/STOL operations shall be by flight test in the performance of appropriate V/STOL tasks from table XC. These tasks shall be flown by test pilots at the flight conditions recommended below under various levels of atmospheric disturbances. Proof of compliance in these demonstration tasks will consist of pilot comments and Cooper-Harper (C-H) ratings. The comments and ratings should indicate that the flying qualities are no worse than the required Level of flying qualities for each combination of Air Vehicle State, Flight Phase, and environmental condition.

Verification of 5.1.1.1 (c) for V/STOL operations shall be by flight test to show that the air vehicle can be flown to particular flight conditions outside of the ROTH and return readily and safely to the ROTH or ROSH. The specific flight conditions to be evaluated shall be those flight conditions appropriate to V/STOL operations which are outside the ROTH but inside the ROHR and which are predicted by analysis and simulation to have the worst handling qualities or to be unrecoverable. Obviously, an emergency recovery system shall be installed for these tests, regardless of whether the condition is predicted to be recoverable or not—the purpose of flight testing is to verify analytical predictions. Further discussion of emergency recovery systems can be found in 5.2.5 Verification. Proof of compliance in these demonstrations will consist of pilot comments. The pilot comments should indicate that the air vehicle can be readily and safely returned to the ROTH or ROSH and that no exceptional pilot inputs are necessary to recover the air vehicle.

For a discussion of how to use piloted evaluations as verification see 5.1.1.1 Verification. Selection of appropriate tasks from table XC shall include consideration of the following:

During the requirements definition process, the procuring activity together with the contractor(s) and the responsible test organization, shall select several closed-loop tasks with which to evaluate the air vehicle in flight test. During system development, ground-based and in-flight simulations shall be used to get an initial appraisal of how well the air vehicle will perform the tasks in flight. The simulations can also be used to train the test pilots and refine the tasks, performance objectives, and test procedures.

TABLE XC. Suggested handling qualities evaluation tasks for V/STOL operations.

Suggested Tasks	Suggested Performance Objectives
Short Takeoff (STO) Ground-based	Desired Performance Maintain centerline tracking within ±1.5 ft Maintain takeoff attitude within ±1° Maintain transition attitude and angle of attack within TBD No loss of altitude or aircraft settle experienced during the transition
	Adequate Performance Maintain centerline tracking within ±5 ft Maintain takeoff attitude within ±2° Maintain transition attitude and angle of attack within TBD No loss of altitude or aircraft settle experienced during the transition
Short Takeoff (STO) Shipboard	Desired Performance Maintain takeoff flyaway attitude (AOA or pitch) within ±1.0° No loss of altitude or aircraft settle experienced during the transition. Maintain nose gear within STO tram lines during deck run (±1.5 ft)
	Adequate Performance Maintain takeoff flyaway attitude (AOA or pitch) within ±2.0° No loss of altitude or aircraft settle experienced during the transition. Maintain nose gear within ±5 ft of the center of the STO tram line
Short Takeoff (STO) With ski jump	Desired Performance Maintain takeoff flyaway attitude (AOA or pitch) within ±1.0° No loss of altitude or aircraft settle experienced during the transition. Maintain nose gear within STO tram lines during deck run (±1.5 ft)
	Adequate Performance Maintain takeoff flyaway attitude (AOA or pitch) within ±2.0° No loss of altitude or aircraft settle experienced during the transition. Maintain nose gear within ±5 ft of the center of the STO tram line
Vertical Takeoff (VTO)	Desired Performance Maintain ascent attitude within ±0.5° Maintain aircraft c.g. above the takeoff point within a conical section by circle with a diameter of 5 ft at the surface and a concentric circle with a diameter of 20 ft at a height of 50 ft.
	Adequate Performance Maintain ascent attitude within ±1.0° Maintain aircraft c.g. above the takeoff point within TBD
Shipboard Recovery Pattern	Desired Performance Excursions no greater than ±2 ft from the formation position No PIO
	Adequate Performance Excursions no greater than ±4 ft from the formation position
Powered-lift Landing	Desired Performance Maintain flight path angle within ±1° Touch down within ±200 ft longitudinally and ±10 ft laterally of aimpoint After touchdown, track centerline within ±10 ft
	Adequate Performance Maintain flight path angle within ±2° Touch down within ±400 ft longitudinally and ±20 ft laterally of aimpoint After touchdown, track centerline within ±25 ft

TABLE XC. Suggested handling qualities evaluation tasks for V/STOL operations - Continued.

Suggested Tasks	Suggested Performance Objectives
Hover and Station Keeping	Desired Performance For Nominal Ascent/Descent, capture target height within ±3 ft with no more than one overshoot of up to 5 ft. For Aggressive Ascent/Descent, capture target height within ±3 ft with no more than two overshoots of up to ±10 ft Maintain aircraft c.g. above the reference point within a conical section by circle with a diameter of 5 ft at the surface and a concentric circle with a diameter of 20 ft at a height of 50 ft.
	Adequate Performance For Nominal Ascent/Descent, capture target height within ±5 ft with no more than one overshoot of up to 10 ft. For Aggressive Ascent/Descent, capture target height within ±5 ft with no more than two overshoots of up to ±15 ft Maintain aircraft c.g. above the reference point within TBD
Hovering Translation	Desired Performance Maintain altitude within ±5 ft Maintain ground track within ±5 ft
	Adequate Performance Maintain altitude within ±10 ft Maintain ground track within ±10 ft
Ground-based Vertical Landing (VL)	Desired Performance Aircraft c.g. at touchdown shall be within a 3-ft-diameter circle centered at the desired touchdown point. Aircraft landing attitude at touchdown within ±0.5^{o} of reference
	Adequate Performance Aircraft c.g. at touchdown shall be within a 5-ft-diameter circle centered at the desired touchdown point. Aircraft landing attitude at touchdown within ±1.0^{o} of reference
Shipboard Vertical Landing (VL)	Desired Performance 45^{o} bearing line maintained within ±5 ft of desired track. Aircraft position above the desired touchdown point stabilized within ±10 ft. Aircraft c.g. at touchdown within a 3-ft-diameter circle centered at the desired touchdown point. Aircraft landing attitude at touchdown within ±0.5^{o} of reference
	Adequate Performance 45^{o} bearing line maintained within ±10 ft of desired track. Aircraft position above the desired touchdown point stabilized within ±20 ft. Aircraft c.g. at touchdown within a 5-ft-diameter circle centered at the desired touchdown point. Aircraft landing attitude at touchdown within ±1.0^{o} of reference
Rolling Vertical Landing (Shore Based)	Desired Performance Maintain flight path angle within ±1^{o} Touch down within ±50 ft longitudinally and ±2 ft laterally of aimpoint After touchdown, track centerline within ±2 ft
	Adequate Performance Maintain flight path angle within ±2^{o} Touch down within ±100 ft longitudinally and ±5 ft laterally of aimpoint After touchdown, track centerline within ±5 ft
Waveoff from a Hover	Desired Performance Maintain target waveoff attitude within ±0.5^{o}
	Adequate Performance Maintain target waveoff attitude within ±1.0^{o}

TABLE XC. Suggested handling qualities evaluation tasks for V/STOL operations - Continued.

Suggested Tasks	Suggested Performance Objectives
Accelerating Transition from Hover or Vertical Takeoff (VTO)	Desired Performance Maintain optimum transition attitude within ±0.5° No loss of altitude during the maneuver.
	Adequate Performance Maintain optimum transition attitude within ±1.0° No loss of altitude during the maneuver.
Decelerating Transition (no crosswind)	Desired Performance No objectionable transients during the conversion Maintain optimum transition attitude within ±0.5° Deviations in the flight path angle less than ±0.5°
	Adequate Performance Maintain optimum transition attitude within ±1.0° Deviations in the flight path angle less than ±1.0°
Decelerating Transition (with crosswind)	Desired Performance Maintain optimum transition attitude within ±0.5° Deviations in the flight path angle less than ±0.5° Maintain centerline within ±10 ft
	Adequate Performance Maintain optimum transition attitude within ±1.0° Deviations in the flight path angle less than ±1.0° Maintain centerline within ±20 ft
Waveoff from a Decelerating Approach	Desired Performance Maintain target waveoff attitude within ±1°
	Adequate Performance Maintain target waveoff attitude within ±2°
Waveoff from a Performance-Deficit Condition	Desired Performance Maintain target waveoff attitude within ±1°
	Adequate Performance Maintain target waveoff attitude within ±2°

Pilot comments shall be considered the most important data. A C-H rating is only a summary of observed flying qualities characteristics expressed as a single number. Pilot comments identify the specific deficiencies, if any, that must be corrected. Furthermore, the "long-look" technique filters the effects of deficiencies on the C-H rating because, over several runs, the pilot learns to compensate for some deficiencies. Since pilot comments are given for every run, the comments will identify all observed deficiencies, even those for which compensation can be made. Comments on succeeding runs provide guidance on the pilot's ability to compensate for the deficiencies and the final C-H rating indicates the relative significance of these deficiencies. Therefore, pilot comments must be recorded and analyzed for every test run.

Time and cost constraints prohibit piloted evaluation of every task in every possible air vehicle configuration at every possible point in the Flight Envelope. The conditions that must be evaluated are the most common operating conditions, operating conditions critical to the mission of the air vehicle, and the worst-case conditions, especially those where the quantitative, open-loop flying qualities requirements are violated by wide margins. For air vehicles with multiple flight control modes, all mode transitions shall be evaluated at common, mission critical, and worst-case conditions, especially mode switches which are done automatically, as opposed to

those deliberately switched by the pilot. Furthermore, the degradation due to atmospheric disturbances shall be demonstrated by evaluation at different levels of disturbances.

Land-based V/STOLs shall use the atmospheric disturbance model defined in 5.1.2 Verification. For those requirements that apply to ship-based V/STOL operations, the atmospheric disturbance model of 5.2.6 shall be used. Additionally, shipboard V/STOLs shall use the atmospheric disturbance model of 5.1.2 for those requirements that apply to land operations. Evaluation of the effect of Extraordinary atmospheric disturbances may be performed in ground simulation. When using simulation to predict the degradation of flying qualities due to Extraordinary atmospheric disturbances, it will be necessary to correlate C-H ratings gathered from simulation sessions in lower levels of turbulence with C-H ratings obtained from flight test for the same task in similar levels of turbulence.

The following paragraphs discuss some recommended tasks and some suggested performance objectives for each task.

Short Takeoff

If the aircraft is expected to operate from austere sites, the following task may be used to demonstrate that the aircraft has the performance and flying qualities necessary to execute a short-field takeoff. Centerline tracking is very important if the aircraft will be operating from roads and confined areas.

Begin with the aircraft configured for STO, with the power lever at IDLE and toe brakes held. Command maximum thrust, and as tire skid begins, release the brakes. At the predetermined STO airspeed, initiate an inceptor input that will result in rotating the thrust vector angle to the recommended STO position. Maintain runway centerline during the takeoff roll. Rotate to and maintain optimum attitude for the takeoff. Upon passing 50 ft, capture and maintain the required attitude and angle of attack for the accelerating transition to wingborne flight. The maneuver is complete when the aircraft reaches 1.2 $V_{min_{power-off}}$. Test conditions shall include the minimum and maximum STO weights, full-forward and full-aft c.g. locations, and the store loading that is most destabilizing in pitch. Some suggested performance objectives are given in table XC.

If the aircraft is expected to operate from the confined flight deck areas aboard LHA and LHD class ships, the following task may be used to demonstrate that the aircraft has the performance and flying qualities necessary to execute a short takeoff from a flat-deck ship. No unusual or excessive control inputs shall be required to capture and maintain the flyaway attitude. Sensitivity to mistrims shall be manageable. Ground handling and over-the-nose field-of-view shall be sufficient to maintain the aircraft within the STO tram lines throughout its entire deck run.

Command maximum thrust and, as tire skid begins, release the brakes. Maintain the nose gear within the STO tram lines during the takeoff roll. As the flight deck is exited (or earlier if appropriate), initiate an inceptor input that will result in rotating the thrust vector angle to the recommended STO position. Rotate to and maintain optimum attitude for the takeoff and accelerating transition. Continue the takeoff through transition and configuration change. Test

conditions shall include representative ship motion and WOD conditions, minimum to maximum takeoff gross weights, full-forward to full-aft c.g. locations, the loading that is most destabilizing in pitch, deck runs from 250 to 750 ft, and both proper and improper trim settings. Some suggested performance objectives are given in table XC.

If the aircraft is expected to operate from the confined flight deck areas aboard ships equipped with ski jumps, the following task may be used to demonstrate that the aircraft has the performance and flying qualities necessary to execute a short takeoff from ski jump ramps. No unusual or excessive control inputs shall be required to capture and maintain the flyaway attitude. The sensitivity to mistrim shall be manageable. Ground handling and over-the-nose field-of-view shall be sufficient to maintain the aircraft within the STO tram lines throughout its entire deck run.

Command maximum thrust and, as tire skid begins, release the brakes. Maintain the nose gear within the STO tram lines during the takeoff roll. As the flight deck is exited (or earlier if appropriate), initiate an inceptor input that will result in rotating the thrust vector angle to the recommended STO position. Rotate to (if necessary) and maintain optimum attitude for the takeoff and accelerating transition. Initiate transition after reaching the inflection point (point of minimum rate of climb). Continue the takeoff through the configuration change. Test conditions shall include representative ship motion and WOD conditions, minimum to maximum takeoff gross weights, full-forward to full-aft c.g. locations, the loading that is most destabilizing in pitch, deck runs from 250 to 450 ft (or as limited by landing gear structural limits), and both proper and improper trim settings. Some suggested performance objectives are given in table XC.

Vertical Takeoff

This task is intended to demonstrate that the aircraft has sufficient the performance and flying qualities necessary to execute a vertical takeoff. There shall be no adverse propulsion-induced effects which would force the pilot to compensate in order to maintain attitude as well as maintain position over the takeoff point.

With the aircraft converted for powered-lift flight and setup on a hover pad or shipboard landing spot at an idle engine condition, apply maximum thrust for vertical takeoff. Maintain takeoff attitude and stabilize in a hover at 100 ft. During ascent, maintain the aircraft position over the takeoff point. Test conditions shall include the maximum vertical takeoff weight. Some suggested performance objectives are given in table XC.

Powered-lift Landing

If the aircraft is intended to operate from austere sites with short runway lengths, this task is intended to demonstrate that the aircraft has the performance and flying qualities necessary to execute a landing at airspeeds below the wingborne 1-g stall speed.

Perform the approach as required to compensate for crosswinds. Adjust the power, thrust vector, and AOA as appropriate for the control system. Throughout the approach phase, the pilot

maintains precise flightpath angle control. At touchdown, the pilot attempts to put the main wheels down inside a designated touchdown zone. After touchdown, the pilot steers from the touchdown point to the runway centerline and thereafter stays on the runway centerline while bringing the air vehicle to a stop. Some suggested performance objectives are given in table XC.

Hover and Station Keeping

This task is intended to verify that the flight vehicle has the flying qualities in the powered-lift flight phase necessary to maintain a stabilized position with respect to a fixed ground reference point or a moving reference point on a ship, as well as to change the vertical position. The vertical-height control response characteristics shall be smooth and predictable. The pilot shall have the capability to change vertical position aggressively without any increase in compensation required to maintain horizontal position with respect to the fixed or moving reference point. Engine thrust response characteristics shall be linear and predictable and there shall be no vertical PIO tendencies during the height capture task.

Height capture tasks and station-keeping tasks shall be accomplished using a fixed visual reference. Vertical height references can range from 25 to 100 ft. The aircraft c.g. shall remain above the reference point. There are two tasks:

> Task A: Height Captures from a Descent
> Begin and stabilize on a 5-ft/sec descent rate (Nominal). Capture and stabilize at a target vertical height position. Maintain horizontal position with respect to the fixed or moving reference point. Repeat above task using 10-ft/sec descent rate (Aggressive).
>
> Task B: Height Captures from an Ascent
> Begin and stabilize on a 5-ft/sec ascent rate (Nominal). Capture and stabilize at a target vertical height position. Maintain horizontal position with respect to the fixed or moving reference point. Repeat above task using 10-ft/sec ascent rate (Aggressive).

Test conditions shall include a stabilized hover at 100 ft above the reference point for descent captures and 25 ft for ascent captures, and both minimum and maximum vertical landing weights.

Hovering Translation

This task is intended to verify that the flight vehicle has the flying qualities and performance in the powered-lift terminal flight phase necessary to precisely translate in any direction without loss of altitude.

From a stabilized hover, perform the following translations (constant altitude):
- longitudinally, both forward and aft
- laterally, both right and left
- diagonally, along the 45º angle in each of the four quadrants.

Test conditions shall include a stabilized hover at 50 ft above a ground reference line, and a range of gross weights and c.g. positions, including those associated with both minimum and maximum vertical landing weights.

Ground-based Vertical Landing

This task is intended to verify that the flight vehicle has the flying qualities and performance in the powered-lift terminal flight phase necessary to descend from a stabilized hover to a ground touchdown. From a stabilized hover, the pilot shall be able to establish a stabilized descent rate and divert his attention back to horizontal station keeping. Once the stabilized descent rate has been established, the pilot shall be able to maintain that descent rate with no substantial increase in the descent rate due to suckdown or loss of powered lift.

This task shall be accomplished using a fixed visual reference. Initiate a 4-ft/sec stabilized descent and maintain position over the desired touchdown point. Maintain the aircraft at the desired touchdown attitude. Set power lever to idle upon touchdown. Test conditions shall include a stabilized hover at 50 ft above the ground reference touchdown point, and both minimum and maximum vertical landing weights.

Shipboard Vertical Landing

This task is intended to verify that the flight vehicle has the flying qualities and performance in the powered-lift terminal flight phase necessary to descend from a stabilized hover to touchdown on a ship. From a stabilized hover, the pilot shall be able to translate along a 45º line to position the aircraft over the desired shipboard landing spot. Once the aircraft is stabilized above this point, a steady descent rate is initiated and the pilot shall be able to maintain that descent rate with no substantial increase in the descent rate due to suckdown or loss of powered lift.

This task shall be accomplished using a fixed visual reference from the ships' landing aids and ship features. From the initial point, perform a forward and right 45º translation and stabilize above the intended landing spot. Initiate a 4-ft/sec stabilized descent and maintain position over the desired touchdown point. Maintain the aircraft at the desired touchdown attitude. Set the power lever to idle upon touchdown. Test conditions shall include a stabilized hover at 100 ft above the flight deck and abeam the ship on a 45º bearing line from the intended touchdown point, and both minimum and maximum vertical landing weights.

Rolling Vertical Landing (Shore Based)

If the aircraft is intended to operate from austere sites with short runway lengths, this task is intended to verify that the flight vehicle has the flying qualities and performance to execute a rolling vertical landing at airspeeds well below the wingborne 1-g stall speed. Extreme accuracy

in touchdown dispersion is very important if the aircraft will be operating from short runway lengths.

Perform the approach as required to compensate for crosswinds. Adjust power, thrust vector, and AOA as appropriate for the given control system.

Waveoff from a Hover

This task is intended to verify that the flight vehicle has the flying qualities and performance in the powered-lift terminal flight phase such that just prior to touchdown the pilot can initiate a maximum performance waveoff during a descent to a vertical landing when the pilot has misjudged ship motion. From a stabilized descent rate, the pilot shall be able to determine that the effective descent rate is excessive or the aircraft is not properly positioned over the shipboard landing spot.

From a stabilized 4-ft/sec descent rate, command maximum thrust at a wheel height of 15 ft above the touchdown point. Maintain the aircraft at the desired waveoff attitude. The maneuver is complete when the aircraft passes 50 ft. Test conditions shall include a stabilized descent rate of 4 ft/sec, and both minimum and maximum vertical landing weights. For post-flight analysis, a step thrust command will be input at 12 ft above the touchdown point.

Accelerating Transition from Hover or Vertical Takeoff

This task is intended to verify that the flight vehicle has the flying qualities and performance to execute an accelerating transition from a vertical takeoff or hover. The pilot shall maintain a positive forward acceleration throughout the transition without losing any altitude.

With the aircraft stabilized in a hover at 50 ft, apply maximum power lever angle to allow the engine to spool up to maximum thrust. As altitude passes 100 ft, initiate the inceptor inputs required to begin the accelerating transition. Maintain optimum attitude for the accelerating transition with no altitude loss experienced during the transition. Perform conversion from STOVL configuration into a Cruise condition. Test conditions shall include a stabilized hover at 50 ft, and maximum vertical takeoff or maximum vertical landing weight, whichever is greater.

Decelerating Transition (no Crosswind)

This task is intended to verify that the flight vehicle has the flying qualities and performance to execute a decelerating transition from a wingborne flight condition at the maximum vertical landing weight. The pilot shall be able to execute a positive deceleration to a hover. To minimize the impact on recovery time in a mixed traffic pattern, the deceleration to a hover shall be accomplished as fast as possible without precise programming of cockpit inceptors.

The task starts at 1.5 miles from touchdown, stabilized at the approach speed on a 3.0º glideslope. The pilot shall perform the conversion from wingborne to powered-lift flight while maintaining optimum attitude for the decelerating transition, with no significant deviation from the glidepath during the transition, and then stabilize in a hover over the desired touchdown point. There are two tasks:

Task A: Straight-in Approach
Maintain power approach speed on the 3º glidepath until 3000 ft short of the touchdown point. At the 3000-ft point, begin scheduling a deceleration such that a hover condition will be achieved over the touchdown point.

Task B: VFR (Case III) Pattern
Maintain power approach speed until abeam the landing spot on the downwind leg. Begin scheduling a deceleration such that a hover condition will be achieved over the touchdown point.

Test conditions shall include environmental conditions from Common to Extraordinary (except no crosswind component of steady wind), starting at 1.5 miles from touchdown, stabilized at the approach speed on a 3.0º glideslope, and maximum vertical landing weight in STOVL configuration.

Decelerating Transition (with Crosswind)

This task is intended to verify that the flight vehicle has the flying qualities and performance to execute a decelerating transition from a wingborne flight condition at the maximum vertical landing weight. The pilot shall be able to execute a positive deceleration to a hover with crosswinds. There are two ways these tasks can be flown. The first is a crabbed approach and then kick out the crab angle to align the fuselage heading with the runway heading. The second method is to maintain groundtrack along the centerline while keeping sideslip equal to zero. This results in flying down the centerline and constantly changing the aircraft's heading to point into the relative wind such that the airplane's heading is directly into the wind at the hover condition.

The task starts at 1.5 miles from touchdown, stabilized at the approach speed, on a 3.0º glideslope. The pilot shall perform conversion from wingborne to powered-lift while maintaining optimum attitude for the decelerating transition, with no significant deviation from the glidepath during the transition, then stabilize in a hover over the desired touchdown point. There are two tasks:

Task A: Straight-in Approach
Maintain power approach speed on the 3º glidepath until 3000 ft short of the touchdown point. At the 3000-ft point, begin scheduling a deceleration such that a hover condition will be achieved over the touchdown point.

Task B: VFR Pattern
Maintain power approach speed until abeam the landing spot on the downwind leg. Begin scheduling a deceleration such that a hover condition will be achieved over the touchdown point.

Test conditions shall include starting at 1.5 miles from touchdown, stabilized at power approach speed, on a 3.0º glideslope, maximum vertical landing weight in the STOVL configuration.

Waveoff from a Decelerating Approach

This task is intended to verify that the flight vehicle has the flying qualities and performance in the powered-lift terminal flight phase to initiate a maximum performance waveoff during a deceleration to a vertical landing or a rolling vertical landing. This task shall demonstrate that the pilot can initiate a waveoff at a point on the approach to a hover or rolling vertical landing.

Follow nominal approach procedures and nominal deceleration schedule to a pre-determined airspeed. Initiate a maximum-performance waveoff once this airspeed is reached. Capture and maintain optimum waveoff attitude. Continue waveoff and transition until fully-wingborne flight is achieved. Test conditions shall include starting at 1.5 miles from touchdown, stabilized at approach speed, on a 3.5º glideslope, with the aircraft conversion completed, and maximum vertical landing weight.

Waveoff from a Performance-Deficit Condition

This task is intended to verify that the flight vehicle has sufficient pilot cueing to enable the pilot to recognize a "too heavy to hover" flight condition. The aircraft systems or flight control system shall prevent the pilot from decelerating into a performance-deficit condition. The pilot cues could be (but are not limited to) visual indications in the HUD, tactile indications such as power lever angle near maximum deflection, or a flight control limitation which will not allow the pilot to decelerate beyond the critical airspeed where the airplane has no excess performance to accelerate out of this condition.

Follow nominal approach procedures and nominal deceleration schedule to a point in which cues have become apparent to the pilot that there will be insufficient performance to hover. Initiate a maximum-performance waveoff once this condition is reached. Capture and maintain optimum waveoff attitude. Exchange of altitude is acceptable to assist in acceleration of the aircraft. Continue waveoff and transition until wingborne flight is achieved. Test conditions shall include starting at 1.5 miles from touchdown, stabilized at a specified airspeed, on a 3.5º glideslope, with the aircraft conversion completed, and a 1.1 Hover Weight Ratio (HWR = Gross Weight / Hover Weight).

5.2.7.1 V/STOL operations.
It shall be possible to operate the air vehicle from __(1)__.

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommended values for blank (1):
All air vehicles: "austere basing sites"
Ship-based air vehicles: "LHA, LHD, and CV type ships as well as austere basing sites."

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

See MIL-F-83300 and AFFDL-TR-70-88.

VERIFICATION

Verification shall be via the requirements of 5.2.7.1.1 through 5.2.7.1.9.

See 5.2.7 Verification.

5.2.7.1.1 Short takeoff (STO).
It shall be possible for the air vehicle to accomplish a maximum performance takeoff with minimal pilot compensation. Tracking centerline (or reference line) shall be easy for __(1)__.

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommended value for blank (1):
All air vehicles: "runway takeoffs"
Ship-based air vehicles: "both ship and runway takeoffs. It is desirable to have the same configuration for ship-based or land-based STO. Deck minimum endspeed for STO shall meet the constraints of the performance requirements of JSSG-2001."

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

It should be a goal to need only a single pilot input for lift-off.

See MIL-F-83300 and AFFDL-TR-70-88.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight, if possible), and flight test using appropriate mission-representative STOs.

See 5.2.7 Verification.

For ship-based air vehicles, "appropriate mission-representative STOs" includes STOs for both ship-borne and land-based decks/runways.

5.2.7.1.2 Vertical takeoff (VTO).
At weights up to a maximum vertical-takeoff weight, it shall be possible for the air vehicle to accomplish a vertical takeoff within the conical section defined as follows: during ascent from a maximum thrust vertical takeoff, the aircraft position shall be maintained over the takeoff point such that the c.g. remains within a conical section by circle with a diameter of __(1)__ at the surface and a concentric circle with a diameter of __(2)__ at a height of __(3)__. It shall be possible to maintain takeoff attitude throughout the ascent and stabilize in a hover at __(4)__. Attitude control and flying qualities shall be satisfactory from lift-off through the transition to wing-borne flight.

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommended values:
Blank (1): 5 ft
Blank (2): 20 ft
Blank (3): 50 ft
Blank (4): 100 ft

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

It is a goal that wind conditions cause no change in pilot technique, although workload may increase.

See MIL-F-83300 and AFFDL-TR-70-88.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight, if possible), and flight test using appropriate mission-representative VTOs.

See 5.2.7 Verification.

For ship-based air vehicles, “appropriate mission-representative VTOs” includes VTOs from both ship-borne and land-based spots.

5.2.7.1.3 Shipboard recovery pattern.
During approach and landing pattern tasks, the air vehicle shall satisfactorily integrate with existing shipboard recovery procedures, including compatibility with the Case I, II, and III recovery patterns. Deceleration capability shall be adequate to be able to slow to pattern airspeed following a high-speed overhead break turn. Airspeed and altitude control on the downwind leg shall be adequate to ensure the proper interval between aircraft and to increase the likelihood of initiating the final approach at the correct speed and glideslope. Bank control during the approach turn shall be adequate to minimize lineup errors at rollout.

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

This requirement applies to ship-based V/STOL air vehicles only.

REQUIREMENT LESSONS LEARNED

Shipboard recovery procedures are outlined in the LHA/LHD NATOPS Manual and the CV NATOPS Manual.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight, if possible), and flight test.

See 5.2.7 Verification.

5.2.7.1.4 Powered-lift landing.
It shall be possible for the air vehicle to accomplish a landing with ground roll for weights at which a vertical landing is not possible.

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale. The need for a requirement on powered-lift landing will be defined in the appropriate Operational Requirements Document (ORD).

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

See MIL-F-83300 and AFFDL-TR-70-88.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight, if possible), and flight test using appropriate mission-representative powered-lift landings.

See 5.2.7 Verification.

For ship-based air vehicles, “appropriate mission-representative powered-lift landings” includes both ship-borne and land-based powered-lift landings.

A suggested task is to establish the appropriate aircraft configuration at a height of 1,000 ft that yields the slowest approach speed consistent with precise flightpath and touchdown control. A glideslope from 3 to 8º is to be assumed with or without a flare depending on maximum gear sink rate limits. At touchdown, the pilot shall use control inputs and maximum braking to minimize ground roll. This task shall be performed at all weights from minimum to a maximum defined powered-lift landing weight. Further, all near-ground operation shall be at conditions calculated to be Foreign Object Damage (FOD)-free; i.e., the intent is that the aircraft have the flexibility to avoid FOD on austere surfaces by means of rolling takeoffs and landings.

5.2.7.1.5 Hover.
In normal operation, it shall be possible for the V/STOL air vehicle to ascend and descend to any nominal hover height and to maintain any chosen height in that range. For station keeping in normal operation, it shall be possible to maintain the touchdown reference point within the conical section defined in 5.2.7.1.2. It shall be possible to translate the aircraft at essentially constant height to acquire the touchdown point to the same accuracy. It shall also be possible to __(1)__.

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommended value for blank (1):

For ship-based air vehicles: "stabilize a hover abeam from a translating ship, and at any instant translate the aircraft laterally to acquire the touchdown point to the same accuracy described above, while maintaining small forward airspeed to keep up with the ship. This shall be accomplished while maintaining satisfactory attitude and height control with no undesirable coupling between any axes."

This requirement applies only to V/STOL air vehicles. If the air vehicle is not required to be ship-based, blank (1) is not applicable.

REQUIREMENT LESSONS LEARNED

It is a goal that wind conditions cause no change in pilot technique, although workload may increase.

See MIL-F-83300 and AFFDL-TR-70-88.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight, if possible), and flight test using appropriate mission-representative hover tasks.

See 5.2.7 Verification.

5.2.7.1.6 Vertical landing (VL).
It shall be possible for the air vehicle to start a vertical landing from a stabilized hover out of ground effect (nominally a height of 50 ft) and perform a controlled descent to the __(1)__ while maintaining thrust margin plus spatial and attitude control capabilities sufficient for safe operations. In normal operation at weights up to maximum vertical landing weight (MVLW), the vehicle shall be capable of maintaining a nominal sink rate of __(2)__ with minimal increase due to ground effects. This controlled descent shall be accomplished while maintaining satisfactory attitude control without any coupling into the vertical axis. It shall be possible to control touchdown reference point to the accuracy required to land on __(3)__. It shall be possible to accomplish a waveoff from a vertical landing with __(4)__ flying qualities from any point on a descent above __(5)__.

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommended value for blank (1):
All air vehicles: ground
Ship-based air vehicles: ground or ship deck

Recommended value for blank (2): 4 ft/sec

Recommended value for blank (3):
All air vehicles: “a V/STOL pad”
Ship-based air vehicles: “a shore-based V/STOL pad, or LHA, LHD, and CV type ships”

Fill in blank (4) with the required qualitative degree of suitability; i.e., specify either “Satisfactory” or “Tolerable.”

Fill in blank (5) with minimum altitude required for waveoff, or with “Minimum Descent Height.”

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

The AV-8B Harrier had problems with thrust response in the vicinity of the ground/deck environment during vertical landings. Specifically, there was a problem with bouncing up off of the ground/deck due to a time lag between commanded throttle cutback and actual thrust decrease.

It is a goal that wind conditions cause no change in pilot technique, although workload may increase.

See MIL-F-83300 and AFFDL-TR-70-88.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight, if possible), and flight test using appropriate mission-representative vertical landings to a land-based VL spot and, for ship-based air vehicles, a ship-borne VL spot.

See 5.2.7 Verification.

5.2.7.1.7 Ground handling.
Air vehicle __(1)__ handling shall be satisfactory. Specifically, there shall be no objectionable aircraft motions after __(2)__ contact, and controllability shall not be in question.

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommended value for blank (1):
All air vehicles: Ground
Ship-based air vehicles: Ground and deck

Recommended value for blank (2):
All air vehicles: ground
Ship-based air vehicles: ground and ship-deck

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

See Concept of Operations for USMC (including Shipboard Operating Bulletins for LHA/LHD/CV type ships), etc.

The AV-8B had problems in the nose wheel steering gains used for ground/deck taxiing and centerline alignment during short takeoffs and slow landings. Though the problems in nose wheel steering gains were not necessarily due to the V/STOL capabilities of the AV-8B, the problems do emphasize the importance of requiring satisfactory ground handling for V/STOL

aircraft, especially when any thrust component is directed downwards, decreasing the effective weight of the aircraft and reducing the effectiveness of the tires to create sufficient traction on the ground/deck.

VERIFICATION

Verification of ground handling shall be performed through appropriate use of analysis, demonstration, simulation, and ground handling and taxi tests. Adequate ground handling shall be verified through appropriate ground handling tests and simulations under the most extreme conditions.

See 5.2.7 Verification.

For ship-based air vehicles, the most extreme ground handling conditions shall include rolling, pitching, heaving, wet, low-visibility shipborne environments and ground handling and taxi tests shall include both land-based and ship-borne testing.

5.2.7.1.8 Transition/Conversion.
There shall be no objectionable air vehicle characteristics or aircraft limitations during the conversion process that would delay or adversely impact the normal launch and recovery processes, including loose formation flight. In addition, there shall be no need for precise programming by the pilot of engine power, attitudes, etc., in terms of speed or time. At maximum vertical takeoff gross weight, it shall be possible to accelerate from hover to wing-borne flight with minimal pilot compensation.

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

At any time during a transition in normal operation, including conversion, it is desirable for the pilot to be able to stop the maneuver quickly and safely and reverse its direction.

See MIL-F-83300 and AFFDL-TR-70-88.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight, if possible), and flight test using appropriate mission-representative transition tasks.

See 5.2.7 Verification.

5.2.7.1.9 Hovering translation.
The air vehicle shall have Satisfactory flying qualities and sufficient performance in the powered-lift terminal flight phase to precisely translate in any direction, without loss of altitude and without any adverse coupling effects between axes.

REQUIREMENT RATIONALE

It shall be easy to maneuver the aircraft while in a hover in any direction without loss of altitude.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

It is a goal that wind conditions cause no change in pilot technique, although workload may increase.

See MIL-F-83300 and AFFDL-TR-70-88.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight if possible), and flight test using appropriate mission-representative hovering translation tasks. For ship-based air vehicles, this includes both shipborne and shore-based operations.

See 5.2.7 Verification.

5.2.7.2 V/STOL dynamic response requirements.
Air vehicle dynamic response during V/STOL operations shall meet the requirements of 5.2.7.2.1 through 5.2.7.2.2.

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

See MIL-F-83300 and AFFDL-TR-70-88.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight if possible), and flight test using appropriate mission-representative tasks.

See 5.2.7 Verification.

Techniques for verification of flying qualities requirements in V/STOL operations can be found in MIL-F-83300, AFFDL-TR-70-88, NASA Technical Paper 3356, NASA TM 104021, and NADC-82146-60.

5.2.7.2.1 Flying qualities in hovering flight.
Without attaining excessive attitudes, it shall be possible to hover over a spot in steady winds of up to __(1)__ from any direction relative to the aircraft heading, except as limited by the boundaries of the ROTH.

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommended value for blank (1): 35 kts

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

The flying qualities and flight control system performance shall meet the intent of requirements 3.2 through 3.2.5.4 of MIL-F-83300. The following subsections contain specific additions, revisions and modifications (based on NASA Technical Paper 3356, NASA TM 104021, and NADC-82146-60) explicitly supersede the corresponding guidelines of MIL-F-83300. These additions, revisions, and modifications to MIL-F-83300's design guidelines pertain primarily to the areas of dynamic response and control power.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight if possible), and flight test using appropriate mission-representative tasks.

See 5.2.7 Verification.

5.2.7.2.1.1 Pitch axis response in hovering flight.
For modes with pitch rate or pitch attitude as the primary commanded variable, the bandwidth, phase delay, and subsidence ratio of the responses to the appropriate command input shall be within the limits defined in table XCI.

TABLE XCI. Pitch axis dynamic response limits.

Response Parameter	Command Type	Level		
		1	2	3
Minimum bandwidth (rad/sec)	Pitch rate			
	Pitch attitude			
Maximum phase delay (sec)	Pitch rate			
	Pitch attitude			
Maximum subsidence ratio	Pitch rate			
	Pitch attitude			

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommended values for table XCI:

Response Parameter	Command Type	Level		
		1	2	3
Minimum bandwidth (rad/sec)	Pitch rate	2.0	1.0	1.0
	Pitch attitude	3.0	1.8	1.8
Maximum phase delay (sec)	Pitch rate	0.10	0.17	0.20
	Pitch attitude	0.10	0.17	0.20
Maximum subsidence ratio	Pitch rate	0.31	0.44	0.44
	Pitch attitude	0.31	0.44	0.44

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

See 5.2.7.2.1 Requirement Lessons Learned.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight if possible), and flight test using appropriate mission-representative tasks.

See 5.2.7 Verification.

5.2.7.2.1.2 Roll axis response in hovering flight.
For modes with roll rate or bank angle as the primary commanded variable, the bandwidth, phase delay, and subsidence ratio of the responses to the appropriate command input shall be within the limits defined in table XCII.

TABLE XCII. Roll axis dynamic response limits.

Response Parameter	Command Type	Level		
		1	2	3
Minimum bandwidth (rad/sec)	Roll rate			
	Bank angle			
Maximum phase delay (sec)	Roll rate			
	Bank angle			
Maximum subsidence ratio	Roll rate			
	Bank angle			

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommended values for table XCII:

Response Parameter	Command Type	Level		
		1	2	3
Minimum bandwidth (rad/sec)	Roll rate	2.0	1.0	1.0
	Bank angle	3.0	1.8	1.8
Maximum phase delay (sec)	Roll rate	0.10	0.17	0.20
	Bank angle	0.10	0.17	0.20
Maximum subsidence ratio	Roll rate	0.31	0.44	0.44
	Bank angle	0.31	0.44	0.44

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

See 5.2.7.2.1 Requirement Lessons Learned.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight if possible), and flight test using appropriate mission-representative tasks.

See 5.2.7 Verification.

5.2.7.2.1.3 Yaw axis response in hovering flight.
For modes with yaw rate or heading rate as the primary commanded variable, the bandwidth and phase delay of the heading response to yaw controller input shall be within the limits defined in table XCIII.

TABLE XCIII. Yaw axis dynamic response limits.

Response Parameter	Level		
	1	2	3
Minimum bandwidth (rad/sec)			
Maximum phase delay (sec)			

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommended values for table XCIII:

Response Parameter	Level		
	1	2	3
Minimum bandwidth (rad/sec)	2.0	0.8	0.8
Maximum phase delay (sec)	0.1	0.17	0.2

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

See 5.2.7.2.1 Requirement Lessons Learned.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight if possible), and flight test using appropriate mission-representative tasks. See 5.2.7 Verification.

5.2.7.2.1.4 Vertical axis response in hovering flight.
For modes with vertical velocity as the primary commanded variable, the bandwidth and phase delay of the altitude response to the height controller input shall be within the limits defined in table XCIV.

TABLE XCIV. Vertical axis dynamic response limits.

Response Parameter	Level		
	1	2	3
Minimum bandwidth (rad/sec)			
Maximum phase delay (sec)			

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommended values for table XCIV:

Response Parameter	Level		
	1	2	3
Minimum bandwidth (rad/sec)	0.6	0.2	--
Maximum phase delay (sec)	0.3	--	--

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

See 5.2.7.2.1 Requirement Lessons Learned.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight if possible), and flight test using appropriate mission-representative tasks.

See 5.2.7 Verification.

5.2.7.2.1.5 Longitudinal translation response in hovering flight.
For modes with longitudinal translation as the primary commanded variable, the bandwidth and phase delay of the longitudinal position response to the longitudinal translation controller input shall be within the limits defined in table XCV.

TABLE XCV. Longitudinal translation dynamic response limits.

Response Parameter	Level		
	1	2	3
Maximum bandwidth (rad/sec)			
Minimum bandwidth (rad/sec)			
Maximum phase delay (sec)			

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommended values for table XCV:

Response Parameter	Level		
	1	2	3
Maximum bandwidth (rad/sec)	1.0	--	--
Minimum bandwidth (rad/sec)	0.3	0.1	--
Maximum phase delay (sec)	0.6	--	--

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

See 5.2.7.2.1 Requirement Lessons Learned.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight if possible), and flight test using appropriate mission-representative tasks.

See 5.2.7 Verification.

5.2.7.2.1.6 Lateral translation response in hovering flight.
For modes with lateral translation as the primary commanded variable, the bandwidth and phase delay of the lateral position response to the lateral translation controller input shall be within the limits defined in table XCVI.

TABLE XCVI. Lateral translation dynamic response limits.

Response Parameter	Level		
	1	2	3
Maximum bandwidth (rad/sec)			
Minimum bandwidth (rad/sec)			
Maximum phase delay (sec)			

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommended values for table XCVI:

Response Parameter	Level		
	1	2	3
Maximum bandwidth (rad/sec)	0.7	--	--
Minimum bandwidth (rad/sec)	0.37	0.15	--
Maximum phase delay (sec)	0.7	--	--

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

See 5.2.7.2.1 Requirement Lessons Learned.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight if possible), and flight test using appropriate mission-representative tasks.

See 5.2.7 Verification.

5.2.7.2.1.7 Cross-axis coupling in hovering flight.
The air vehicle shall have no objectionable adverse coupling, that is any unwanted response in an axis not commanded by the pilot which requires secondary compensation by the pilot using a different inceptor. Favorable coupling may be used if it is integrated into the primary commanded response (e.g. attitude change to augment translation). Vertical landings shall be possible with no objectionable cross-axis coupling between attitude and height (vertical) control.

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

See 5.2.7.2.1 Requirement Lessons Learned.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight if possible), and flight test using appropriate mission-representative tasks.

See 5.2.7 Verification.

5.2.7.2.1.8 Angular (moment-generating) control power in hovering flight.
For all Air Vehicle States appropriate to hovering flight (including Failure States), in steady winds, control power available shall be sufficient to produce at least the attitude changes of table XCVII within 1 sec from initiation of the relevant inceptor application. These values apply for angular rate and attitude control modes.

TABLE XCVII. Attitude change in 1 sec.

Flying Qualities Level		Attitude Change (deg)		
Single-Axis Control Application	Simultaneous Pitch, Roll and Yaw Control Application	Pitch	Roll	Yaw
1				
2	1			
3	2			

REQUIREMENT RATIONALE

These requirements for control power for single-axis control application and simultaneous roll/pitch/yaw control application are minimum requirements. They are meant to specify control power available for maneuvering, which is excess control power above and beyond any control power needed for VTO, STO, hover, station-keeping, glideslope maintenance, and attitude control in crosswinds, turbulence, Hot Gas Ingestion (HGI), and ground effects.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

Recommended values for table XCVII:

Flying Qualities Level		Attitude Change (deg)		
Single-Axis Control Application	Simultaneous Pitch, Roll and Yaw Control Application	Pitch	Roll	Yaw
1		3	6	4
2	1	2	3	3
3	2	2	2	2

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

Several AV-8As were lost as a result of having to use more than 50 percent of available roll control power to trim out a 30-kt crosswind. These requirements have been derived and validated through years of AV-8B/Harrier development, as well as independent simulation and analysis from NASA and other agencies.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight if possible), and flight test using appropriate mission-representative tasks. See 5.2.7 Verification.

For the three atmospheric disturbance intensities, land-based V/STOLs shall use the steady wind speeds defined in 5.1.2 Verification. Shipboard V/STOLs shall use the steady wind speeds defined in 5.2.6 Verification.

5.2.7.2.2 Flying qualities in the transition region.
For all possible configurations and conditions between hover and the maximum conversion speed, any control system or flying qualities parameters which are scheduled with flight condition (blending of command types as a function of airspeed, for example) shall exhibit a smooth variation with no objectionable transients and with no reversal in any direction of change.

REQUIREMENT RATIONALE

See 5.2.7 Requirement Rationale.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 through 3.3.11.1.3 in JSSG-2001.

There was no corresponding requirement in MIL-F-8785C or MIL-STD-1797A.

This requirement applies only to V/STOL air vehicles.

REQUIREMENT LESSONS LEARNED

See MIL-F-83300 and AFFDL-TR-70-88.

VERIFICATION

Verification shall be performed through appropriate use of analysis, demonstration, simulation (offline, piloted fixed-base, piloted motion-based, and piloted in-flight if possible), and flight test using appropriate mission-representative tasks.

See 5.2.7 Verification.

5.2.8 Characteristics of the primary flight control system

5.2.8.1 Transfer to alternate control modes.
The transient motions and trim changes resulting from any mode changes of the flight control system shall be such that dangerous flying qualities never result. These mode changes include intentional mode switches by the pilot, as well as any mode switches caused by the flight control system automatically, with or without the pilot's conscious intent. For Air Vehicle Normal States, with controls free, the motion transients resulting from intentional mode changes shall not exceed the limits of table XCVIII for at least __(1)__ following the mode change. Without retrimming, the changes in control forces required to maintain attitude and sideslip shall not exceed the limits of table XCIXfor at least __(2)__ following an intentional mode change.

TABLE XCVIII. Transient response limits to configuration or control mode change.

Axis of Response	Flight Envelope	Transient Limits
Pitch		
Roll		
Yaw		

TABLE XCIX. Control force limits for configuration or control mode change.

Axis of Response	Maximum Control Force Change
Pitch	
Roll	
Yaw	

REQUIREMENT RATIONALE

Transients due to mode switching are distracting, and if the transients are too large, pilots will object. Any unnecessary distractions and added workload can interfere with a pilot's mission performance. Mode switching of the flight control system shall never result in unusual or unreasonable demands on the pilot to retain control. Transients due to mode switching must not be excessive or cause excessive distraction.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.5.6 through 3.5.6.2
MIL-STD-1797: 4.1.12.7, 4.2.6.2, 4.2.8.6.6, 4.5.7.2, 4.5.9.5.7, 4.6.5.3, and 4.6.7.10.

Recommended values for table XCVIII:

Axis of Response	Flight Envelope	Transient Limits
Pitch	ROSH and ROTH	No more than ±.05 g normal acceleration at the pilot's station
Roll	ROSH	No more than ±3º/sec roll
	ROTH	No more than ±5º/sec roll
Yaw	ROSH and ROTH	No more than ±5º sideslip, but never to exceed the structural limit

Recommended value for blank (1): 2 sec

Recommended values for table XCIX:

Axis of Response	Maximum Control Force Change
Pitch	20 lbs
Roll	10 lbs
Yaw	50 lbs

Recommended value for blank (2): 5 sec

REQUIREMENT LESSONS LEARNED

This requirement is intended to apply both to pilot-initiated changes and to automatic changes initiated by selection of weapons, flap setting, etc. Since this requirement deals with intentional modification of the flight control system, it is implied that no failures have occurred, except where operating procedures call for the crew to switch modes upon experiencing a particular failure. Failures are covered explicitly by 5.2.1.5. Proper application of this requirement may be performed by careful design of the air vehicle augmentation systems. Mode switching shall assure that the new mode chosen does not have any large transients in initialization.

Trim transients following intentional pilot actions shall obviously be small enough not to produce significant distractions. Do not overlook such automatic changes as a switch to a different control mode when the pilot selects a particular weapon.

Since the intent of a flight control system is to improve the air vehicle response characteristics—whether measured by improved flying qualities or by increased mission effectiveness—any system which can be chosen by the pilot shall not cause objectionable transient motions. There has been some speculation as to whether some small transient motion is or is not desirable. The argument for an intentional transient is that inadvertent pilot switching of autopilot modes is less likely if accompanied by a noticeable transient motion.

Military Specification MIL-F-8785B allowed 0.05 g normal acceleration for pitch transients following mode changes. This was increased to 0.10 g in MIL-F-8785C, to allow (if not encourage) designers to provide some noticeable transient (see AFWAL-TR-81-3109). In AFWAL-TR-81-3109 an accident was cited wherein the pilot inadvertently bumped off the altitude hold mode (which automatically disengaged when a small force was applied to the control column). The flight recorder showed a 0.04 g transient which was unnoticed by the crew, who were deeply involved in trying to lower a malfunctioning landing gear. However, it is our contention that the undesirable features of transient motions due to mode switching are significant. Furthermore, a distracted crew would probably not notice a transient considerably larger than 0.04 g, especially if there were any turbulence at all. Therefore, use of the maximum allowable transient of 0.05 g specified in MIL-F-8785B is recommended. As in MIL-F-8785B and MIL-F-8785C, 2 sec is deemed a reasonable time to allow for the pilot to resume control.

“Dangerous flying qualities” need to be interpreted in a rational manner, so that it applies to feasible design options. The element of danger cannot be completely removed from flight.

VERIFICATION

Verification shall be by analysis and by simulation or flight test in the performance of the tasks of 5.1.1.1, 5.1.1.2, 5.1.1.3.2, and 5.1.1.3.3. These tasks shall be flown by test pilots at specific flight conditions throughout the ROSH and ROTH. The specific flight conditions to be evaluated shall be the most common operating conditions, any operating conditions critical to the mission of the air vehicle, and those flight conditions where transients due to mode change are predicted to be at their greatest. The mode changes to be evaluated shall include intentional mode switches by the pilot, as well as any mode switches caused by the flight control system automatically, with or without the pilot's conscious intent. Proof of compliance in these demonstration tasks shall consist of time histories of air vehicle response and pilot inputs, pilot comments, and C-H ratings. The comments and ratings shall indicate that the flying qualities are no worse than the required Level of flying qualities for each combination of Air Vehicle State and Flight Phase.

Verification normally will be a part of flight control system design and testing, and of flight envelope expansion. Evaluation pilots should be alert to potential operational problems in exploring the safe limits of the flight envelope. Compliance shall be evaluated at likely conditions for mode switching and at the most critical flight conditions. The critical flight conditions are highly dependent on the aerodynamic and flight control system configurations. Some factors which determine critical conditions are given in the discussion of 5.2.1.5. Critical conditions will usually be at the corners of the expected envelopes (e.g., a SAS for power approach should be switched at the highest and lowest airspeeds, at low altitudes). Limited analytical and ground-based simulation may be used to supplement actual flight testing, especially in early stages of development; but flight testing is ultimately required. The conditions examined shall be in the range of those encountered operationally.

Trim transients following intentional pilot actions with the flight control system shall obviously be small. This flying qualities requirement pertaining directly to the flight control system applies to all axes of control and response. Demonstration will involve mode switching at representative altitudes throughout the ROSH and ROTH, focusing on those areas of airspeed, altitude, and task that would produce the largest transients between the two modes involved. Configuration changes and control mode changes are to be made by normal means.

5.2.8.2 Augmentation systems.
Operation of stability augmentation and control augmentation systems and devices, including any performance degradation due to saturation, shall not introduce any objectionable flight or ground handling characteristics. Any performance degradation of stability and control augmentation systems due to saturation of components, rate limiting, or surface deflections, shall only be momentary, and shall not introduce any objectionable flight or ground handling characteristics.

5.2.8.2.1 Rate of control surface displacement.
The ability of the aircraft to perform the operational maneuvers required of it shall not be limited by control surface deflection rates in Common and Uncommon atmospheric disturbances. Control rates shall be adequate to retain stabilization and control in Extraordinary atmospheric disturbances.

REQUIREMENT RATIONALE

In certain flight conditions, turbulence intensities, and failure states, performance of augmentation systems can actually degrade the flying qualities. The purpose of these requirements is to insure that this effect is analyzed and minimized. Compliance with these paragraphs is especially important to vehicles employing relaxed static stability.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.5.2.3 and 3.5.4
MIL-STD-1797: 4.1.12.4 and 4.1.12.9.

These qualitative requirements result from experience and pertain directly to the flight control system. They apply to all axes of control and response. These requirements particularly apply for all Normal States and Failure States in the atmospheric disturbances of 5.1.2 and during maneuvering flight at the AOA, sideslip, and load factor limits of the RORH. They also apply to post-stall gyrations, spins, and recoveries with all systems, such as the hydraulic and electrical systems, operating in the state that may result from the gyrations encountered.

REQUIREMENT LESSONS LEARNED

Atmospheric disturbances in the form of gusts shall not prevent any maneuvering in the ROSH. This means that no limitations shall be imposed due solely to control travel. Since ability to counter gusts includes surface rate characteristics, these too are mentioned explicitly. While specific disturbances are listed, the evaluation remains somewhat qualitative. The control required for attitude regulation is in addition to that required for trim and maneuvering.

Another requirement on control surface rate capability, more explicit although still qualitative is 5.1.3. Whereas that requirement concerns loss of control, 5.2.8.2.1 applies more generally.

On the X-29, SCAS gains were originally a function of air data from the side-mounted probe. Large position errors transonically gave erroneous gains which lowered the systems phase margin.

A modicum of common sense is required in the application of these requirements. Operational maneuvering in thunderstorm turbulence is not normally required. It seems reasonable to require operational maneuvers in turbulence intensities up to Uncommon. For turbulence intensities greater than Uncommon it seems reasonable to require sufficient maneuver capability for loose attitude control.

Changes of mechanical gearings and stability augmentation gains in the primary flight control system are sometimes accomplished by scheduling the changes as a function of the settings of secondary control devices, such as flaps or wing sweep. This practice is generally acceptable, but gearings and gains normally shall not be scheduled as a function of trim control settings since pilots do not always keep air vehicles in trim.

VERIFICATION

Verification shall be by simulation or flight test in the performance of the tasks of 5.1.1.1 and 5.1.1.2 under various levels of atmospheric disturbances. These tasks shall be flown by test pilots at specific flight conditions throughout the ROSH and ROTH. The specific flight conditions to be evaluated shall be the most common operating conditions, any operating conditions critical to the mission of the air vehicle, and any conditions determined by analysis or simulation to be worse than the Level of flying qualities required in 5.1.1. For conditions which are considered too dangerous to test in flight, verification can be shown in a manned simulation. Proof of compliance in these demonstration tasks shall consist of pilot comments and C-H ratings. The comments and ratings should indicate that the flying qualities are no worse than the required Level of flying qualities for each combination of Air Vehicle State, Flight Phase, and level of atmospheric disturbance.

This group of flying qualities requirements pertaining directly to the flight control system applies to all axes of control and response. These generally qualitative requirements result from experience. Verification normally will be a part of flight control system design and testing, and of flight envelope expansion. For powered or boosted controls, the effect of engine speed and the duty cycle of both primary and secondary control together with the pilot control techniques shall be included.

Auxiliary hydraulic devices may use up significant portions of the available hydraulic power during critical phases of the mission. For example, actuation of landing gear, flaps, slats, etc., during the landing approach, when the engines are operating at relatively low power settings, could drain enough hydraulic power to make it difficult for the pilot to make a safe approach, especially in turbulence. In other flight conditions with less auxiliary demand or higher engine thrust, that same hydraulic system might be more adequate. Also, at high dynamic pressure high hinge moments may limit control surface rate and deflection.

In precision control tasks, such as landing approach and formation flying, it has been observed that the pilot sometimes resorts to elevator stick pumping to achieve better precision (see

AFFDL-TR-65-198, AFFDL-TR-66-2, and Boeing Report D6-10732 T/N). This technique is likely to be used when the short-period frequency is low or if the phugoid is unstable, but has been observed in other conditions also. Some important maneuvers, such as correcting an offset on final approach, call for simultaneous, coordinated use of several controls.

The required operational maneuvers are commensurate with the particular Level of flying qualities under consideration. The maneuvers required in Level 3 operation, for example, will normally be less precise and more gradual than for Level 1 and 2 operation. In some cases this may result in lower demands on control authority and rates for Level 3 operation. Note, however, that when the handling qualities of the air vehicle are near the Level 3 limits, increased control activity may occur, even though the maneuvers are more gradual.

The demands of various performance requirements and the rapid advancement of control system technology have resulted in the application of relaxed static stability in both the pitch and yaw axes. These systems provide excellent flying qualities until the limits of surface deflection or rate are reached. In this case, the degradation in flying qualities is rapid and can result in loss of control due to pilot-in-the-loop oscillations or divergence. It has been found, however, that momentarily reaching the rate or deflection limit does not always result in loss of control; the time interval that a surface can remain on its rate or deflection limit depends on the dynamic pressure, the level of instability of the vehicle, and other factors. A thorough analysis of the capability of the augmentation system shall be performed over the RORH and shall include variations in predicted aerodynamic terms, e.g., position and system tolerance. During flight at high AOA, operation of augmentation systems has caused departure, either because the aerodynamic characteristics of the surface have changed or the surface has reached its limit. During departures or spins, engines may flameout or have to be throttled back or shut down such that limited hydraulic or electric power is available to control the gyrations, recover to controlled flight, and restart the engine(s). The analysis of flying qualities shall take into account these degraded system capabilities.

Evaluation pilots shall be alert for potential operational problems in exploring the safe limits of the flight envelope. Critical conditions will usually be at the corners of the expected envelopes (e.g., a SAS for power approach should be switched at the highest and lowest airspeeds, at low altitudes). Limited analytical and ground-based simulation may be used to supplement actual flight testing, especially in early stages of development; but flight testing is ultimately required. The conditions examined shall be in the range of those encountered operationally.

For flight in turbulence, compliance may be shown principally through analysis of gust response characteristics using either an analytical model or a piloted simulation involving the gust models of 3.2 and a flight-validated model of the air vehicle. Such analysis must include not only the normal operational maneuvers involving pitch, roll, and yaw controls; but also the critical maneuvers (especially for hydraulic actuation systems) which may limit the responsiveness of the control surfaces. As mentioned previously, these might include extension of landing gear and high-lift devices on landing approaches, etc. Some evaluation shall be conducted by flying in real turbulence.

5.2.8.3 Cockpit controller characteristics.
The characteristics of the cockpit controllers shall meet the requirements of 5.2.8.3.1 through 5.2.8.3.7.

REQUIREMENT RATIONALE

Requirements 5.2.8.3.1 through 5.2.8.3.7 are concerned with those aspects of the primary flight control system which are directly related to the flying qualities. These requirements are in addition to the requirements of the applicable control system design specification. Some of the important mechanical characteristics of control systems (including servo valves or actuators) are: friction and preload, lost motion, flexibility, mass imbalance and inertia, nonlinear gearing, and rate limiting. Meeting separate requirements on these items, however, will not necessarily insure that the overall system will be adequate.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.5, 3.5.1, and 3.5.2
MIL-STD-1797: 4.1.12.

REQUIREMENT LESSONS LEARNED

Customarily, flight control system specifications refer to the flying qualities requirements. The overall flying qualities requirements pertaining to the flight control system are placed here because of their generality. These general qualitative requirements apply to all axes of control and response. They result from experience. Although the trim system is part of the flight control system, for convenience trim system requirements have been grouped separately under 5.2.9.1.

VERIFICATION

Although simulation may be helpful for design purposes, flight verification shall be of the flight hardware installed in the aircraft.

Throughout the flight test program, pilots should note any difficulty in meeting these qualitative requirements. Evaluation in flight shall be made over the operational load factor and airspeed ranges, at the minimum and maximum operational altitudes, and especially at high speeds, where required control surface deflections are small. Operationally-oriented fine tracking will be a crucial test.

5.2.8.3.1 Control force versus control deflection.
Dynamically, throughout the frequency range of pilot control inputs, the deflection of the pilot's controller shall not lead the control force.

REQUIREMENT RATIONALE

Both control force and control deflection provide pilot cues. This requirement is intended to assure consonance between the two.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.2.3.2
MIL-STD-1797: 4.2.8.4.

REQUIREMENT LESSONS LEARNED

The minimum values specified for dynamic F_s/n in 5.2.2.1.8.2 and ζ_{sp} in 5.2.2.1.1.2 equivalent systems criteria are not sufficient to prevent the occurrence of PIO. In fact, there are documented cases of PIO-prone aircraft with Level 1 values of $(F_s/n)_{min}$ and ζ_{sp}. The feel system, which allowed δ_s to lead F_s, was found to be responsible for these PIOs. The details of these cases provide valuable design guidance and are discussed at some length below.

An excellent treatise by Peter Neal on the historical development and analytical aspects of the dynamic requirement is given in AFFDL-TR-69-72 and "Influence of Bobweights on Pilot-Induced Oscillations." Basically, the requirement stems from and reflects experience with production bobweight-augmented elevator control systems. Such systems have the virtue of keeping F_s/n relatively constant, i.e., large changes with attitude and loading (c.g.) are prevented. Early versions, which involved manual control, featured elevators with nearly 100% aerodynamic balance. Later versions featured full-power hydraulics with spring and bellows feel systems. More recently, n_z feedback directly to the servo valve rather than to the control stick has given a response feel system without contributing phase shift between control force and control deflection. Both documents provide a feedback-control-analysis basis for studying the problem and fixes associated with bobweights in the specific context of PIO. Similar, earlier efforts by others are referenced; however, even earlier work on bobweight effects in unpowered elevator control systems, of NACA and RAE origin, is not cited. The main emphasis is to explain, by virtue of analysis, the particular problems encountered by a succession of example airplanes: P-63A (NACA Memo Report L6E20), A4D-2 (Douglas Aircraft Co. LB-25452), T-38A ("Investigation and Elimination of PIO Tendencies in the Northrop T-38A" and "Report of the T-38 Flight Control System PIO Review Board"), and F-4 (FTC-TR-67-19).

Pertinent conclusions reached by Neal ("Influence of Bobweights on Pilot-Induced Oscillations") are:

1. The use of a control-system bobweight without consideration of its effects on the aircraft's dynamics can lead to serious PIO problems.

2. Potential PIO problems due to a bobweight can be minimized by increasing the sensitivity of the bobweight to pitch acceleration, as by blending forward and aft bobweights, using the following rule of thumb:

$$\ell_b = g\,[\omega_{sp}^2/(n/\alpha)]^{-1} + \ell_{CR}$$

Taking the highest minimum $\omega_{sp}^2/(n/\alpha)$, for Category A (3.2.1.2 of MIL-F-8785C)

$$\ell_b = 115 + \ell_{CR}$$

where

- ℓ_b is the distance (in feet) of an equivalent point-mass bobweight ahead of the c.g.:

$$\ell_b = g\,\frac{\text{bobweight stick force due to unit } \ddot{\theta}}{\text{bobweight stick force due to unit } n_z}$$

- ℓ_{CR} is the distance (in feet) of the instantaneous center of rotation for control inputs ahead of the c.g.: $-Z_\delta/M_\delta$

3. When this criterion is satisfied, the contribution of the bobweight to stick force per g may still be limited by the fact that the closed-loop feel-system roots can be driven unstable. This problem can usually be improved by the use of a viscous stick damper.

4. The final control system design shall be checked against other longitudinal requirements. Such checks may in fact show the undesirability of using viscous stick damping because of associated lags in response to stick inputs.

VERIFICATION

Given some very basic instrumentation (a control position potentiometer and strain gauge to measure control force) it is a simple matter to obtain the phase relationship between control force and control position via a pilot-generated frequency sweep at each selected flight condition. The phase relationship between control position and control force can be obtained from the frequency sweep data via a Fast Fourier Transform computer program. As a general rule the frequency range of interest will be between 0.5 and 10 rad/sec.

Demonstration of compliance with this dynamic requirement requires a quantitative determination of the phase relationship between control of deflection and force. However, if the control forces clearly lead control deflection based upon pilot comments, the requirement shall be considered satisfied. It should be emphasized that due to actual control system effects such as friction, hysteresis, etc., ground-based simulation may not be adequate and therefore flight test results are needed.

5.2.8.3.2 Control centering.
Pitch, roll, yaw, direct lift, and direct side force controllers shall exhibit positive centering in flight at any normal trim setting. Although absolute centering is not required, the combined effects of centering, breakout force, stability, and force gradient, shall not produce objectionable flight characteristics, such as poor precision-tracking ability, or permit large departures from trim conditions with controls free.

REQUIREMENT RATIONALE

A discernible neutral point (or trim or equilibrium point) shall always be provided in manual pitch, roll, yaw, direct lift, and side force controllers. That is, if the pilot chooses to release a control it shall return toward a neutral or trim state. If no such cues are provided, the pilot will be forced to search manually for such a trim condition. This can lead to poor maneuvering control, or, in the extreme, to pilot-in-the-loop oscillations.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.5.2.1
MIL-STD-1797: 4.1.12.1.

REQUIREMENT LESSONS LEARNED

With absolute centering, a cockpit control will always return exactly to its trim position when released. Positive centering is a tendency to return; upon release, the control will move toward the trim position, but friction may prevent absolute centering.

Allowable levels of friction may preclude absolute centering, but generally this is acceptable. On the other hand, throttles generally have no feel except friction. This has been found acceptable for infrequent, trim-type usage, but it can cause difficulties if the pilot attempts continuous control about a trim point. For example, after moving the throttle, the pilot has no trim position reference.

VERIFICATION

Although simulation may be helpful for design purposes, final verification shall be of the flight hardware installed in the aircraft. Throughout the flight test program, pilots should note any difficulty in meeting this qualitative requirement. Evaluation in flight shall be made over the operational load factor and airspeed ranges, at the minimum and maximum operational altitudes,

and especially at high speeds where required control surface deflections are small. Operationally-oriented fine tracking will be a critical test.

5.2.8.3.3 Control free play.
The free play in each cockpit controller, that is, any motion of the cockpit controller which does not move the control surface in flight, shall not result in objectionable flight characteristics, particularly for small-amplitude control inputs.

REQUIREMENT RATIONALE

Some small amount of free play may be desirable to prevent over-sensitivity to unintended control motions. However, in normal operations, and especially in high-demand times such as turbulence penetration or air combat, free play can contribute to overcontrol, loss of accuracy, and rapid pilot fatigue.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.5.2.2
MIL-STD-1797: 4.1.12.2.

REQUIREMENT LESSONS LEARNED

No numerical value has yet been found that appears generally adequate. The allowable free play would seem to be a function of control-deflection sensitivity (angular acceleration per inch or degree of movement) and possibly control-force sensitivity as well.

VERIFICATION

Although simulation may be helpful for design purposes, final verification shall be of the flight hardware installed in the aircraft. Throughout the flight test program, pilots should note any difficulty in meeting this qualitative requirement. Evaluation in flight shall be made over the operational load factor and airspeed ranges, at the minimum and maximum operational altitudes, and especially at high speeds where required control surface deflections are small. Operationally-oriented fine tracking will be a critical test.

5.2.8.3.4 Control displacement limits.
For each cockpit controller, no physical interference with the pilot's body, seat, or cockpit structure shall prevent the pilot from obtaining full control authority.

REQUIREMENT RATIONALE

The allowable control displacement must be limited to allow proper control in both pitch and roll.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.5.9.1.

REQUIREMENT LESSONS LEARNED

The F-15 has an aileron-rudder interconnect and reduced differential tail/aileron authority for roll performance at high angle of attack. However, ability to use full lateral control stick with large aft stick displacement was limited by interference with the pilot's legs. Therefore, the lateral control system was mechanized such that partial lateral stick deflection provides full rudder and lateral control deflection with large aft stick deflections. (The reduced lateral control surface deflection and the aileron-rudder interconnect prevent roll reversal at high angle of attack.)

VERIFICATION

Stops are normally provided to limit travel. Normally, full wheel or stick throw will meet this requirement. Proof of compliance shall consist of demonstration that the pilot can achieve full control authority without striking any part of his body, the seat, or the cockpit structure with the controller.

5.2.8.3.5 Dynamic characteristics.
A linear or smoothly-varying aircraft response to cockpit control deflection and to control force shall be provided for all amplitudes of control input. The response of the control surfaces in flight shall not lag the cockpit control force inputs by more than the angles specified in table C, for frequencies equal to or less than the frequencies specified in table CI.

TABLE C. Control surface lags.

Level	Allowable Lag (deg)		
	Category A Flight Phases	Category B Flight Phases	Category C Flight Phases
1			
2			
3			

TABLE CI. Applicable frequencies for control surface lags.

Control	Upper frequency (rad/sec)
Pitch	The larger of equivalent ω_{sp} and 2.0
Roll & Yaw	The largest of equivalent ω_d, $1/T_R$, and 2.0

REQUIREMENT RATIONALE

This requirement pertains directly to the flight control system and applies generally to all axes of control and response. The generally qualitative requirement results from experience.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.5.3
MIL-STD-1797: 4.1.12.5.

Recommended values for table C:

Level	Allowable Lag (deg)	
	Category A & C Flight Phases	Category B Flight Phases
1	15	30
2	30	45
3	60	60

REQUIREMENT LESSONS LEARNED

In the Navy's experience the control surface lag requirements, which are not explicitly covered by the flight control system specification, provide additional guidance concerning the portion of the time delay which may be attributed to control surface lag. Such time delay can be an important source of pilot-in-the-loop oscillation tendencies.

If quantitative limits are desired on control surface lag, the limits from table C and table CI are recommended. While pilots do not normally observe surface motion due to stability augmentation, the time delay between pilot input and surface movement (here expressed as phase delay) can be disconcerting (see 5.2.2.1.5 guidance).

VERIFICATION

Verification normally will be a part of flight control system design and testing, and of flight envelope expansion.

5.2.8.3.6 Control system damping.
All control system oscillations apparent to the pilot shall be well damped, unless they are of such an amplitude, frequency, and phasing that they do not result in objectionable oscillations of the cockpit controls or the airframe on the ground, during the flight, and in atmospheric disturbances.

REQUIREMENT RATIONALE

This requirement pertains directly to the flight control system and applies generally to all axes of control and response. This generally-qualitative requirement results from experience.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.5.3.1
MIL-STD-1797: 4.1.12.6.

REQUIREMENT LESSONS LEARNED

Many airplanes, including airplanes using stability augmentation, have second-order control system modes which exhibit low damping. If these modes have natural frequencies considerably above the airframe natural frequencies, or if there are control system zeros near the poles, the low damping is often not even noticeable to the pilot. There are other cases, however, where the low control system damping is very objectionable or even dangerous.

Normal acceleration bobweights have a tendency to cause trouble in this area, especially if the control system employs an irreversible elevator actuator. In this situation, the basic control system usually has very low damping ($2\zeta\omega_n$). Without a bobweight, this is normally no problem because the natural frequency (ω_n) of the control system is much higher than ω_{sp}. When a bobweight is added, however, the natural frequency of the control system is significantly reduced because of the increased control system inertia. On several airplanes, the bobweight used was large enough to reduce the control system natural frequency to the point that rapid control movements caused objectionable high frequency control system oscillations.

Certain types of adaptive control systems employ modes having very low damping. If the natural frequency of such a mode is too low, objectionable control system oscillations may result.

VERIFICATION

Verification normally will be a part of flight control system design and testing, and of flight envelope expansion.

5.2.8.3.7 Direct force controllers.
Direct force controllers (such as in direct lift or lateral translation modes) which are separate from the attitude controllers shall have a direction of operation consistent with the sense of the air vehicle motion produced, be conveniently and accessibly located, comfortable to use, and compatible with pilot force and motion capabilities. Functions shall be provided in the control system that would only allow these modes to be engaged within their design flight regime or maneuvers. When used either by themselves or in combination with other controllers, flight safety and mission effectiveness shall not be degraded. These systems shall not defeat limiters that are necessary for stable and controlled flight, or for structural considerations.

REQUIREMENT RATIONALE

These requirements pertain to direct force control modes which are actuated by cockpit control manipulators that are separate and distinct from the normal cockpit controls (i.e., centerstick, sidestick, or wheel, rudder pedals, and throttle quadrant), as opposed to direct force control modes which are blended or integrated to modify the responses to the conventional cockpit control manipulators. These requirements apply to all axes of control and response. These requirements are written to insure that operation of the controllers is simple and straightforward. When implementing these controllers, it must be assumed that the pilot may elect to engage the device in the middle of a maneuver, or in conjunction with another mode.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.4.11
MIL-STD-1797: 4.1.12.11.

REQUIREMENT LESSONS LEARNED

If loss of control or structural damage could occur, an inhibitor shall be incorporated in the system such that it cannot be engaged, or if it is already engaged, that other modes with which it is not compatible cannot be engaged. Furthermore, operation of these devices shall not be able to defeat AOA limiters, sideslip limiters, or load factor limiters that are built into the basic flight control system to provide stable and controlled flight.

VERIFICATION

If the air vehicle has any direct force controllers, the set of evaluation tasks for 5.1.1.1 and 5.1.1.2 shall include some tasks which utilize these controllers. Verification of this requirement shall be by simulation or flight test in the performance of these tasks. These tasks shall be flown by test pilots at specific flight conditions throughout the ROSH and ROTH, except that these controllers need not be evaluated in flight regimes in which they cannot be engaged. The specific flight conditions to be evaluated shall be the most common operating conditions, any operating conditions critical to the mission of the air vehicle, and any conditions determined by analysis or simulation to be worse than the Level of flying qualities required in 5.1.1. For conditions which are considered too dangerous to test in flight, verification can be shown in a

manned simulation. Proof of compliance in these demonstration tasks shall consist of pilot comments and C-H ratings. The comments and ratings should indicate that the flying qualities are no worse than the required Level of flying qualities for each combination of Air Vehicle State, Flight Phase, and level of atmospheric disturbance.

According to AFWAL-TR-81-3027:

> "The "separate" manipulators can be used as trimming devices or for continuous tracking. In the first case the control is used only intermittently, to establish a new trim condition or operating point. It operates like a trim button; changes are "beeped in." In the second case the auxiliary manipulator is used continuously, as in tracking a target. Presumably the conventional controls are used to establish the operating point.
>
> If the system design is such that both conventional and auxiliary manipulators are used continuously, then it violates a pilot-centered requirement for frequency separation of controls. A pilot cannot easily coordinate more than two control axes continuously, simultaneously, and in the same frequency range of operation. He must time share his attention between the multiple controls.
>
> Even when conventional controls are used to establish an operating point, i.e., as "trim" controls, use of the auxiliary manipulators implies additional pilot workload relative to using the conventional controls alone. The increased workload is presumably traded for significant performance advantages obtainable only by using the auxiliary controllers. This implies careful tailoring of the response characteristics when using the separated manipulator to achieve flying task performance that is significantly better than that obtainable with conventional control responses."

Recommended tasks for the evaluation of flying qualities for direct force modes include open-loop tasks, air-to-air tracking, air-to-ground tracking, dive bombing, and approach and landing. Examples of tasks using direct force controls in both the trimming and continuous tracking roles are given below.

Open-loop tasks

Open-loop tasks with direct force control modes include step, ramp, pulse, and doublet inputs with the associated controllers, attitude captures, vertical and lateral displacements, and wind-up turns and pullups. Examples of the use of open-loop tasks to evaluate direct force control modes are included in AFFTC-TR-77-23, AFFDL-TR-78-9, AFFTC-TR-83-45, AFFTC-TR-83-46, and AFWAL-TR-84-3008, Vol 6, Part 1.

Air-to-air tracking

Air-to-air tracking has been found suitable for testing such direct force control modes as direct lift, pitch pointing, wings-level turn (direct side force), lateral translation, and yaw pointing.

Examples of the use of these modes in air-to-air tracking evaluation tasks can be found in AFFTC-TR-77-23, AFFDL-TR-78-9, AFWAL-TR-81-3027, AFFTC-TR-83-45, AFWAL-TR-84-3060, and AFWAL-TR-84-3008, Vol 6, Part 1.

Air-to-ground tracking

Air-to-ground tracking has been found suitable for testing such direct force control modes as direct lift, pitch pointing, wings-level turn, lateral translation, and yaw pointing. Examples of the use of these modes in air-to-ground tracking evaluation tasks can be found in AFFDL-TR-71-106, AFFDL-TR-72-120, AFFTC-TR-77-23, AFFDL-TR-78-9, AFFTC-TR-83-45, AFWAL-TR-84-3060, and AFWAL-TR-84-3008, Vol 6, Part 1.

Simulated bombing

Simulated bombing has been found suitable for testing such direct force control modes as direct lift, vertical translation, wings-level turn, lateral translation. Examples of the use of these modes in simulated bombing evaluation tasks can be found in AFFDL-TR-71-106, AFFDL-TR-72-120, AFFDL-TR-76-78, AFFTC-TR-77-23, AFFDL-TR-78-9, AFFTC-TR-83-45, AFWAL-TR-84-3060, and AFWAL-TR-84-3008, Vol 6, Part 1.

Approach and landing

Approach and landing has been found suitable for testing such direct force control modes as vertical translation, wings-level turn, lateral translation, and yaw pointing. Examples of the use of these modes in approach and landing evaluation tasks can be found in AFFDL-TR-73-2, AFFTC-TR-77-23, AFFDL-TR-78-9, and AFWAL-TR-84-3060.

Formation flying

Formation flying has been found suitable for testing such direct force control modes as direct lift, vertical translation, wings-level turn, and lateral translation. Examples of the use of these modes in formation flying evaluation tasks can be found in AFFTC-TR-77-23, AFFDL-TR-78-9, AFFTC-TR-83-45, and AFWAL-TR-84-3008, Vol 6, Part 1.

Aerial refueling

Aerial refueling has been found suitable for testing such direct force control modes as direct lift, vertical translation, wings-level turn, and lateral translation. Examples of the use of these modes in aerial refueling evaluation tasks can be found in AFFTC-TR-83-45 and AFWAL-TR-84-3008, Vol 6, Part 1.

5.2.8.4 Displays and instruments.
For those missions and tasks in which the pilot's use of displays and instruments is critical in performing the task, the characteristics of these displays and instruments shall not introduce any objectionable flight or ground handling characteristics.

REQUIREMENT RATIONALE

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: No corresponding requirement
MIL-STD-1797: 4.1.12.

REQUIREMENT LESSONS LEARNED

Cockpit automation is becoming increasingly prominent. It would not yet be prudent, however, to give more than a little general guidance on division of duties and the interface between human and automatic control. Time lags in displays will tend, of course, to be seen as effective delays in aircraft response.

It has been suggested the crew be responsible for head-up, eyes-out tasks, situation awareness, and executive functions: mission, tactics, environment, and contingencies. While some functions that might be automated are: mission planning, station keeping, monitoring, rendezvous, alerting, navigation (including air drop), routine functions.

As exemplified by the Integrated Fire and Flight Control (IFFC) advanced development program, there are cases where partial automation is the best task solution. In that program the HUD displayed a box, its size representing the range of operation of the IFFC system. The pilot would put this box on his air-to-air target, and IFFC would put the pipper on the target. It was found helpful to leave roll control to the pilot: he could easily track the target's bank angle and keeping involved aided the pilot to take over quickly after any failure in the high-bandwidth IFFC system.

VERIFICATION

5.2.9 Characteristics of secondary flight control systems

5.2.9.1 Trim system.

In straight flight, throughout the ROSH, the trim system shall be capable of reducing the steady-state control forces to the values given in table CII. Trim systems shall not defeat other features incorporated in the flight control system that prevent or suppress departure from controlled flight or exceedance of structural limits, or that provide force cues which warn of approach to flight limits.

TABLE CII. Untrimmed steady-state cockpit control forces.

	Pitch	Roll	Yaw
Level 1			
Level 2			
Level 3			

5.2.9.1.1 Trim system irreversibility.

All trimming devices shall maintain a given setting indefinitely unless changed by the pilot, or by a special automatic interconnect (such as to the landing flaps), or by the operation of an augmentation device. If an automatic interconnect or augmentation device is used in conjunction with a trim device, the trim device shall return to its initial trim position on removal of each interconnect or augmentation command.

5.2.9.1.2 Rate of trim operation.

Trim devices shall operate rapidly enough to enable the pilot to maintain low control forces under changing conditions normally encountered in service, yet not so rapidly as to cause oversensitivity or trim precision difficulties under any conditions. It shall be possible to trim the pitch control forces to maintain __(1)__ throughout:

a. Level-flight accelerations at maximum augmented thrust from 250 kts or $V_{R/C}$, whichever is less, to V_{max} at any altitude when the air vehicle is trimmed for level flight prior to initiation of the acceleration.

b. Dives to all attainable speeds within the ROTH when the air vehicle is trimmed for level flight at any altitude within the ROTH prior to initiation of the dive.

c. __(2)__.

5.2.9.1.3 Stalling of trim systems.

Stalling of a trim system due to aerodynamic loads during maneuvers shall not result in an unsafe condition. The entire trim system shall be capable of operating during dive recoveries from dives to all attainable speeds throughout the RORH, at any attainable, permissible load factor, at any possible position of the trimming device.

5.2.9.1.4 Trim for asymmetric thrust (air vehicles with more than two engines).
For Level 1, it shall be possible to trim the cockpit-control forces to __(3)__ in straight, level-flight with up to two engines inoperative following asymmetric loss of thrust from the most critical propulsive factors at speeds from the maximum-range speed for the engine(s)-out configuration to the speed obtainable with normal rated thrust on the functioning engine(s). Systems completely dependent on the failed engines shall also be considered failed.

5.2.9.1.5 Automatic trim system.
Automatic trimming devices shall not degrade or inhibit the action of response limiters.

REQUIREMENT RATIONALE

These paragraphs are included to insure adequate trim system operation. Establishing, maintaining, and changing the trim or operating point are basic factors in piloting. To ease pilot workload, it is necessary to specify the ability of the trim system to reduce control forces in operational flight and in the event of trim system failures or flight control failures affecting other control surfaces. Some air vehicles have features in the flight control system that are incorporated to prevent g overstress or departure, or provide force cues that air vehicle limits are being reached. The use or misuse of the trim system shall not degrade the protection afforded by these features.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.2.3.5 and 3.6.1 through 3.6.1.4
MIL-STD-1797: 4.1.13 through 4.1.13.3, 4.1.13.5, 4.1.13.6, 4.2.8.6.3, and 4.2.8.7.

Recommended values for table CII:

Level	Pitch	Roll	Yaw
1	0	0	0
2	0	0	0
3	10	5	20

Recommended value for blank (1) in 5.2.9.1.2:
For centerstick controllers: "less than 10 lbs push or pull"
For wheel-control controllers: "less than 20 lbs push or pull", however, in the event that operation of the trim system requires the removal of one hand from the wheel controller, the force limit shall be "less than 10 lbs push or pull"
For sidestick controller: "forces acceptable to the pilot"

Recommended value for blank (2) in 5.2.9.1.2, for combat air vehicles add:

c. Ground attack maneuvers required in normal service operation.

Recommended value for blank (3) in 5.2.9.1.4: zero

Requirement 5.2.9.1.4 does not apply for vehicles with less than three engines.

In addition to these requirements on the trim system, 5.2.9.2 limits transients and control force changes due to operation of other controls. In addition to the pitch trim change requirements of 5.2.9.2.1, it is stipulated no other control actions shall add significantly to pilot workload.

REQUIREMENT LESSONS LEARNED

The trim system is part of the flight control system, but the general trim requirements are grouped here for convenience. The purpose of a trim system is to reduce steady-state forces on cockpit controls, preferably to zero. Transient forces due to failures are similarly limited by 5.2.1.5.2. The ROSHs cover the design missions and also delineate the minimum requirement. It would be desirable, however, to have trim capability throughout the larger ROTH and perhaps even to the maximum permissible speed. Straight flight includes climbs and dives.

In normal operations, this requirement is very straightforward. If a trim system is provided, it must be effective. However, the more quick or powerful a trim system, the more catastrophic a trim failure can be. The difficulty in designing a trim system will be in assuring that extreme failures (trim hardover, sticking, etc.) are capable of being overcome by the pilot. Hence override or alternate trim mechanisms (e.g., dual trim systems) can be of prime importance.

Load factor or AOA limiting systems are often provided for relaxed static stability application. In some of these systems, however, trim system inputs are made downstream of the output from the primary system control laws. The result is that load factor or AOA limiting systems can be defeated by use of trim. In some cases, it has been found that autopilot inputs through the trim system can also defeat these AOA limiting systems. A similar problem may exist in yaw for air vehicles with sideslip limiters. In other cases, limitations on trim actuators have limited surface position and authority and reduced the effectiveness of angle-of-attack limiters. Finally, initial conditions on integrators used for trim can inhibit the response of the aircraft until the initial condition is removed.

The irreversibility requirement allows trim scheduling or interconnection with other control devices (e.g., flaps), but it specifically disallows float or drift.

It may be difficult to find a trim rate which will be good for all loadings in all mission phases. Slow trim rates will not keep up with rapidly changing flight conditions, and so will fatigue the pilot. Too rapid trim rates give oversensitivity, make trim difficult, and accentuate the effect of any runaway trim. 5.2.2.1.8.7 and 5.2.2.1.8.8 set specific limits on forces in dives and during rapid speed changes, while trimming to decrease the forces.

While the requirement on stalling of trim systems applies generally, the problem has been encountered with pitch trim by adjusting incidence of the horizontal tail. First, some of the available elevator capability goes to oppose the mistrimmed stabilizer and less is left to counter any adverse gust-induced pitching motions. Second, elevator forces will be increased and may complicate recovery from a high-speed dive. Third, and perhaps most significant, whenever the

elevator opposes the stabilizer, the aerodynamic hinge moment on the stabilizer may reach a level that is impossible for the trim actuator to overcome. See, for example, AIAA 64-353.

If, for example, nose-down trim is used to counter the air vehicle's pitch-up response to a vertical downdraft, the air vehicle will pitch down more sharply when the draft reverses in direction. Elevator will be used to counter the pitch-down motion, and the resulting aerodynamic load may be sufficient to stall the stabilizer actuator when nose-up retrim is attempted. As speed increases, the adverse effects increase, and the elevator may have insufficient effectiveness to counter the nose-down forces of the draft and the mistrimmed stabilizer. It is obvious that tuck effects may also complicate the picture, and it is significant that tuck effects can not be countered by a Mach trim system that is unable to move the stabilizer.

A Boeing 720B airliner encountered stalling of the pitch trim actuator during a turbulence upset over O'Neill, Nebraska, on 12 July 1993 (NASA CR-2677). The air vehicle was passing through 39,000 ft in a climb to 41,000 ft in IMC when severe turbulence was encountered. A large downdraft was penetrated and the air vehicle pitch attitude increased to +60º. This occurred despite application of full forward stick. The gust was reversed to a large updraft, putting the air vehicle into a severe dive with an estimated flightpath angle of about -35º. The pitch trim control was reported by the crew to be frozen in the dive. Recovery was made with power (pullout at 14,000 ft) and pitch trim control was restored.

Two other turbulence upsets occurred with commercial jet transports (another Boeing 720B and a DC-8), in which the wreckage of both air vehicles showed the trim actuator in the full nose-down position. The frequency of such turbulence upset accidents has been reduced drastically in recent years by pilot training to fly loose attitude control and essentially to ignore large airspeed excursions in severe turbulence. However, the possibility of entering a dive with full nose-down mistrim shall be considered in the design process.

Operational experience with early electrical trim systems running away included crashes not only due to loss or inadequacy of control, but also due to excessive pilot fatigue from having to hold high forces for an extended time until a landing could be made. KC-135, B-57, and other aircraft have been lost due to runaway trim, so that now elaborate precautions are commonly taken to preclude dangerous trim runaway, trim and control use of the same surface, or trimming by adjusting the null position of the feel spring through a limited range. Civil airworthiness regulations have long required ability to continue flight and land safely with maximum adverse trim.

Autotrim can be insidious. Several B-58s are thought to have been lost because the pitch autotrim would allow approach to stall AOA with no indication whatsoever to the pilot until very close to stall. Attitude hold stabilization has a similar effect with the pilot's hand lightly on the control. Pitch autotrim does not promote holding airspeed, and a number of trim and autostabilization mechanizations need the addition of some form of stall and overspeed limiters.

VERIFICATION

Verification shall be by demonstration in setting up on conditions for various tests throughout the flight test program. Flight conditions shall explore the boundaries of trim authority. Inducing

failures at these conditions may be impractical or judged too dangerous for flight. For conditions which are considered too dangerous to test in flight, verification can be shown by intentionally mistrimming the air vehicle and recovering from the mistrim, or by manned simulation. Proof of compliance shall consist of flight data records and pilot comments. Flight data records shall be used to verify compliance with 5.2.9.1, 5.2.9.1.1, 5.2.9.1.3, 5.2.9.1.4, and 5.2.9.1.5. Pilot comments shall be used to verify compliance with 5.2.9.1.2 and 5.2.9.1.3. For verification of 5.2.9.1 and 5.2.9.1.5, deliberate attempts shall be made to defeat limiters and departure prevention systems. For verification of 5.2.9.1.3, deliberate attempts shall be made to stall the trim system.

Trim capability must eventually be shown through flight testing at conditions covering the range of operational flight. Most flight verification will be accomplished during the normal course of the flight test program (all flight tests, not just flying qualities).

Flight conditions shall be chosen to explore the boundaries of trim authority; e.g., low altitude and high speed (high dynamic pressure), and high altitude and low speed (low dynamic pressure) at most forward and most aft c.g. Inducing failures at these conditions may be impractical or judged too dangerous for flight. In this case, compliance with the failure requirements may be demonstrated by intentionally mistrimming the air vehicle and recovering from the mistrim, or by simulation. Conditions which are the most critical will depend on the control feel. External stores can affect both stability and the zero-lift pitching moment as well as the air vehicle loading.

The failures to be considered in applying Level 2 and Level 3 requirements shall include trim sticking and runaway in either direction. It is permissible to meet Level 2 and 3 requirements by providing the pilot with alternate trim mechanisms or override capability.

Flight test crews shall monitor trim system characteristics throughout the program to note any discrepancies. Of special interest are extreme loadings and corners of the flight envelope, including sustained maneuvers at $n \neq 1$; e.g., dives and dive recoveries, pullups, wind-up turns, with the cockpit trim setting fixed throughout—and for trim rate, rapid speed changes, and configuration and thrust changes. Check at the highest trim system loadings, which may be the critical test of irreversibility.

It is clear that full nose-down mistrim should be accounted for in the dives. For example, a Boeing 720 with full nose-down trim at the dive entry will encounter stalling of the pitch trim drive in the dive if the pilot is manually attempting to pull out. Judgment will have to be applied to decide if the mission requirements and failure considerations such as runaway trim or trim actuation power failure should allow this type of abuse. Federal Aviation Administration Advisory Circular 25.253-1A gives guidance on design upset maneuvers for civil transport air vehicles.

5.2.9.2 Operation of secondary control devices and in-flight configuration changes.
The transients and steady-state trim changes for normal operation of secondary control devices (such as throttle, thrust reversers, flaps, slats, speed brakes, deceleration devices, dive recovery devices, wing sweep, and landing gear) shall not impose excessive control forces to maintain the desired heading, altitude, attitude, rate of climb, speed, or load factor without use of the trimmer control. This requirement applies to all in-flight configuration changes and combinations of changes made under service conditions, including the effects of asymmetric operations such as unequal operation of landing gear, speed brakes, slats, or flaps. In no case shall there be any objectionable buffeting or oscillation caused by such devices.

5.2.9.2.1 Pitch trim changes.
With the air vehicle trimmed, and without retrimming, the pitch trim changes caused by operation of secondary control devices shall not require a peak pitch control force in excess of the values of table CIII in order to maintain the trim condition for at least 5 sec following the initiation of the configuration change. The magnitude and rate of trim change subsequent to this time period shall be easily trimmable by use of the normal trimming devices.

TABLE CIII. Pitch control force limits following configuration changes.

Level	Maximum Control Force (lbs)
1	
2	
3	

REQUIREMENT RATIONALE

These requirements set limits on transients and control forces changes due to operation of secondary controls. Secondary control actions shall not add significantly to pilot workload. Frequently-encountered pitch trim changes, if too large, can add to pilot workload at critical times during a mission.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.1 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.6.3 and 3.6.3.1
MIL-STD-1797: 4.1.13.4 and 4.2.5.

Recommended values for TABLE CIII:

	Centerstick Controller	Wheel Controller	Sidestick Controller
Level 1	10	20	
Level 2	15	30	
Level 3	15	30	

REQUIREMENT LESSONS LEARNED

Crossfeeds and feedbacks in the stability and control augmentation system generally will reduce the magnitude of trim changes. Wing downwash and vertical placement of the engines are two of the determining factors. For thrust reversing, configuration-dependent aerodynamics play an important role. The direction of the pitch trim change can also be important, producing either helpful or unfavorable coupling. In any case, the magnitude should not be excessive.

More specific requirements on auxiliary dive recovery devices are contained in 5.2.9.3.

VERIFICATION

Table CIV gives some recommended initial trim conditions and configuration changes for determination of compliance with the requirement on pitch trim changes. These are the trim changes that, when larger than the limits specified, have been bothersome in the past. Generally, the conditions of table CIV will suffice for determination of compliance with this requirement. For aircraft with variable-sweep wings, additional requirements shall be imposed consistent with operational employment of the vehicle. Thrust vectoring and other special features also need to be considered.

TABLE CIV. Suggested conditions to evaluate pitch trim changes.

Flight Phase	Initial Trim Conditions[1]					Configuration Change	Parameter to be Held Constant[2]
	Altitude	Speed	Landing Gear	High-Lift Devices and Wing Flaps	Thrust		
Approach	$h_{o_{min}}$	Normal pattern entry speed	Up	Up	TLF	Gear Down	Altitude and airspeed[3]
Approach	$h_{o_{min}}$	Normal pattern entry speed	Up	Up	TLF	Gear Down	Altitude
Approach	$h_{o_{min}}$	Normal pattern entry speed	Down	Up	TLF	Extend high-lift devices and wing flaps	Altitude and airspeed[3]
Approach	$h_{o_{min}}$	Normal pattern entry speed	Down	Up	TLF	Extend high-lift devices and wing flaps	Altitude
Approach	$h_{o_{min}}$	Normal pattern entry speed	Down	Down	TLF	Idle thrust	Airspeed
Approach	$h_{o_{min}}$	$V_{o_{min}}$(PA)	Down	Down	TLF	Extend approach drag device	Airspeed
Approach	$h_{o_{min}}$	$V_{o_{min}}$(PA)	Down	Down	TLF	Takeoff thrust	Airspeed
Approach	$h_{o_{min}}$	$V_{o_{min}}$(PA)	Down	Down	TLF	Takeoff thrust plus normal clean up for waveoff (go-around)	Airspeed
Takeoff	$h_{o_{min}}$	$V_{o_{min}}$(TO)	Down	Takeoff	Takeoff thrust	Gear up	Pitch attitude
Takeoff	$h_{o_{min}}$	Minimum flap retract speed	Up	Takeoff	Takeoff thrust	Retract high-lift devices and wing flaps	Airspeed
Cruise	$h_{o_{min}}$ and $h_{o_{max}}$	Speed for level flight	Up	Up	MRT	Idle thrust	Pitch attitude
Cruise	$h_{o_{min}}$ and $h_{o_{max}}$	Speed for level flight	Up	Up	MRT	Actuate deceleration device(s)[4]	Pitch attitude
Cruise	$h_{o_{min}}$ and $h_{o_{max}}$	Speed for level flight	Up	Up	MRT	Maximum augmented thrust	Pitch attitude
Cruise	$h_{o_{min}}$ and $h_{o_{max}}$	Speed for best range	Up	Up	TLF	Actuate deceleration device(s)[4]	Pitch attitude
Air-to-air combat	$h_{o_{min}}$ and $h_{o_{max}}$	Speed for level flight	Up	Up	MRT	Idle thrust	Pitch attitude
Air-to-air combat	$h_{o_{min}}$ and $h_{o_{max}}$	Speed for level flight	Up	Up	MRT	Actuate deceleration device(s)[4]	Pitch attitude
Air-to-air combat	$h_{o_{min}}$ and $h_{o_{max}}$	Speed for level flight	Up	Up	MRT	Maximum augmented thrust	Pitch attitude
Air-to-air combat	$h_{o_{min}}$ and $h_{o_{max}}$	Speed for best range	Up	Up	TLF	Actuate deceleration device(s)[4]	Pitch attitude

NOTES: [1] Auxiliary drag devices are initially retracted and all details of configuration not specifically mentioned are normal for the light phase.

[2] The evaluation shall be made in the manner expected in operational practice, rather than necessarily holding everything else constant.

[3] Throttle setting may be changed during the maneuver.

[4] If power reduction is permitted in meeting the deceleration requirements established for the mission, actuation of the deceleration device shall be accompanied by the allowable power reduction.

5.2.9.3 Auxiliary dive recovery devices.
Operation of any auxiliary device intended solely for dive recovery shall always produce a positive increment of normal acceleration, but the total normal load factor shall never exceed __(1)__ with controls free.

REQUIREMENT RATIONALE

Special pitching-moment-producing devices are sometimes used for dive recovery on subsonic airplanes when Mach number effects severely reduce elevator effectiveness.

REQUIREMENT GUIDANCE

This requirement flows down from 3.3.11.1.3 in JSSG-2001.

The corresponding requirements in previous specifications are:
MIL-F-8785C: 3.6.4
MIL-STD-1797: 4.1.12.10.

The recommended value for Blank (1) is: 0.8 n_L

REQUIREMENT LESSONS LEARNED

This requirement is essentially the same as 3.3.17 of MIL-F-8785. The requirement seems reasonable for this type of device.

VERIFICATION

Verification shall be by analysis, simulation, and flight test.

6. NOTES

(This section contains information of a general or explanatory nature that may be helpful, but is not mandatory.)

6.1 Intended use.

This standard is intended to ensure flying qualities that provide adequate mission performance and flight safety for military air vehicles developed to perform combat and combat-support missions in environments unique to military weapon systems, regardless of design implementation or flight control system mechanization.

6.2 Acquisition requirements.

Acquisition documents should specify the following:

a. Title, number, and date of the standard.

6.3 Subject term (key word) listing.

Angle
Attitude
Axis
Bank
Carrier
Control
Crosswind
Cruise
Dynamics
Failure
Flight path
Force
Lateral
Longitudinal
Maneuver
Oscillation
Pitch
Recovery
Roll
Sideslip
Spin
Stall
Trim
Wind speed
Yaw

6.4 Changes from previous issue.

Marginal notations are not used in this revision to identify changes with respect to the previous issue due to the extent of the changes.

Custodians:
Army – AV
Navy – AS
Air Force – 11

Preparing activity:
Air Force – 11
(Project 15GP-0067)

Review activities:
Army – MI
Air Force – 99

Note: The activities listed above were interested in this document as of the date of this document. Since organizations and responsibilities can change, you should verify the currency of the information above using the ASSIST Online database at http://assist.daps.dla.mil.