AFWAL-TR-82-3081
VOLUME II

PROPOSED MIL STANDARD AND HANDBOOK — FLYING QUALITIES OF AIR VEHICLES

Volume II: Proposed MIL Handbook

Roger H. Hoh
David G. Mitchell
Irving L. Ashkenas
Richard H. Klein
Robert K. Heffley

Systems Technology, Inc.
Hawthorne, CA 90250

John Hodgkinson

McDonnell Aircraft Co.
St. Louis, MO 63166

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FLIGHT DYNAMICS LABORATORY
AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
AIR FORCE SYSTEMS COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO

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This report is published to elicit comments, and suggestions, for adopting or revising the material which the Government will incorporate into an official MIL Standard and Handbook. As it stands, the material is preliminary and occasionally controversial, does not in all cases represent the views of the Government or a particular Government organization, and should not be used for procurement. This report does not reflect our work in progress on flying qualities requirements for large airplanes, direct force controllers or STOL aircraft.

This report has been reviewed by the Office of Public Affairs (ASD/PA) and is releasable to the National Technical Information Service (NTIS). At NTIS, it will be available to the general public, including foreign nations.

This technical report has been reviewed and is approved for publication.

Stanley O. Fuller
STANLEY O. FULLER, Captain, USAF
Project Engineer
Control Dynamics Branch

R.O. Anderson
RONALD O. ANDERSON, Chief
Control Dynamics Branch
Flight Control Division

FOR THE COMMANDER

Ernest F. Moore
ERNEST F. MOORE, Colonel, USAF
Chief, Flight Control Division
Flight Dynamics Laboratory

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Design Criteria	MIL-F-8785	MIL Handbook

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

MIL-F-8785C, "Military Specification, Flying Qualities of Piloted Airplanes", has been revised into a MIL Standard and supporting Handbook. This report is a proposed version of the MIL Standard and Handbook developed by Systems Technology, Inc., with the McDonnell Aircraft Co. acting in a consulting role. This report is published to elicit comments, and suggestions, for adopting or revising the material which the Government will incorporate into an official MIL Standard and Handbook. As it stands, the material is preliminary and

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FOREWORD

The work reported herein was performed during the period from April 1980 to July 1982 under Contract F33615-80-C-3604 from the Air Force Wright Aeronautical Laboratories, Air Force Systems Command. Lieutenant Robert B. Crombie was the initial project engineer. This responsibility was later transferred to Captain Stanley G. Fuller. This work was completed under Program Element 61102F, Project 2403, Task 05, and Work Unit 40. The STI technical director was Mr. Irving L. Ashkenas. Mr. Roger H. Hoh served as STI project engineer.

The authors wish to express their gratitude to the many reviewers in industry and government who offered beneficial comments on various portions of this report. Special thanks are extended to Messrs. David J. Moorhouse and Robert J. Woodcock of AFWAL for their significant contributions.

We wish also to thank our co-workers at both STI and McDonnell Aircraft Company for their support.

Finally, appreciation [illegible] extended to the STI Editorial Department, and especially Mss. Linda [illegible]fman and Kay Wade and Mr. Charles Reaber, for their outstanding work in all phases of production of this report.

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SECTION - INTRODUCTION

1. SCOPE OF THE REPORT

MIL-F-8785C, Military Specification — Flying Qualities of Piloted Airplanes, has been reformatted into a MIL Standard and a supporting MIL Handbook. This report is a draft of the proposed MIL Standard, which has been developed by Systems Technology, Inc., with the McDonnell Aircraft Company acting in a consulting role. It is presented to industry and the United States armed forces for comments and proposed revisions. The responsibility for the legal MIL Standard and Handbook rests within the armed forces. This draft will be considered and form the basis for revisions, industry and government comments and a tri-service review in the process of developing the MIL Standard and Handbook. Suggested values and background information are contained in this volume.

MIL-F-8785C and the backup documents to both it and its predecessor, MIL-F-8785B, were reviewed extensively. Much of the material contained therein is still considered to be valid and relevant and has been retained in this document.

2. CONCEPT OF MIL STANDARD AND HANDBOOK

The MIL Standard is a skeleton document consisting of incomplete requirements in verbal form which are to be completed by the procuring activity using numerical criteria from the MIL Handbook. A custom MIL Standard will be developed for each new aircraft procurement or major modification of an existing aircraft, as follows:

1) Identify mission requirements.

2) Break down requirements into piloting tasks.

3) For each paragraph in the MIL Standard, select the most appropriate handling quality criterion from the MIL Handbook and insert into the Standard.

The procedure results in a customized handling quality specification for each new aircraft or modification of an existing aircraft. The purpose of this revised format is to facilitate tailoring a detailed handling quality specification to the particular mission requirements of the aircraft being acquired.

3. ORGANIZATION OF MIL HANDBOOK

The MIL Standard and Handbook criteria are presented in terms of aircraft response axes, which represents a significant change from MIL-F-8785C where the criteria were presented in terms of modes. This change is to better accommodate highly augmented airplanes, a primary objective of the contract to develop the new MIL Standard and Handbook. Also, allowance is made for responses in each axis to "primary," "secondary," and "other" controllers; and the manufacturer is free to select which controls are primary in each axis. For example, in the Advanced Medium STOL Transport (AMST) program, Douglas selected throttles as the primary flight path controller whereas Boeing selected pitch attitude.

Utilizing the above format results in some requirements for which there are, at this time, no criteria sufficiently developed to be included in the specification. In such cases a skeleton paragraph is given in the MIL Standard and the reasons for such a requirement are included in the MIL Handbook.

Many readers will want to locate corresponding paragraphs for MIL-F-8785C and the MIL Standard and Handbook. Tables 1 and 2 are given to simplify such correlations. Table 1 cross-references the MIL Standard to MIL-F-8785C, and Table 2 relates 8785C to the Standard. The corresponding MIL-F-8785C paragraphs are also referenced in each MIL Handbook item.

Paragraph numbers are always identical between the MIL Standard and MIL Handbook.

"Quality Assurance" was presented as Section 4 in MIL-F-8785C. We felt that inasmuch as quality assurance contains the acceptable methods

TABLE 1. NUMERICAL CROSS-INDEX FROM MIL STANDARD TO MIL-F-8785C

MIL STANDARD PARAGRAPH	RELATED MIL-F-8785C PARAGRAPH	MIL STANDARD PARAGRAPH	RELATED MIL-F-8785C PARAGRAPH
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3.1.7.2	3.4.1.2	3.3.1.1	New
3.1.8	3.1.11	3.3.1.2	3.2.1.3
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3.2.2.1	New	3.4.1.1	3.2.1.1.1
3.2.2.2	3.2.2.3	3.4.2	New
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TABLE 1. (Concluded)

MIL STANDARD PARAGRAPH	RELATED MIL-F-8785C PARAGRAPH	MIL STANDARD PARAGRAPH	RELATED MIL-F-8785C PARAGRAPH
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3.5.8.3	3.5.6, 3.5.6.1	3.6.6.2.1	3.3.2.5
3.5.8.4	3.4.6	3.6.6.2.2	3.3.2.6
3.5.8.5	3.4.7	3.6.6.2.3	3.3.5.1
3.5.9.1	3.3.4, 3.3.4.1, 3.3.4.1.1, 3.3.4.1.2, 3.3.4.2	3.6.6.2.4	3.3.7
		3.6.6.2.5	3.3.5.1.1
3.5.9.2	3.3.6.3.2	3.6.6.2.6	3.3.8
3.5.9.3	3.3.7.1, 3.3.7.2, 3.3.7.3, 3.3.9	3.6.6.2.7	3.3.5.2
		3.6.6.2.8	3.3.9.1, 3.3.9.2
3.5.9.4	3.3.9.2	3.6.6.2.9	3.5.5.2
3.5.9.5	3.3.8	3.6.6.2.10	3.5.6.2
3.5.9.6	3.4.6, 3.3.4.1.2	3.7.1	3.4.11
3.5.9.7	3.3.9.5	3.8.1	3.4.3
3.5.9.8	3.4.10	3.8.2	3.4.4
3.5.10.1	3.3.4.5	3.8.3	3.4.4.1
3.5.10.2	3.3.4.3	3.8.4	3.4.2
3.5.10.3	3.3.4.1.3	3.8.4.1	3.4.1.1
3.5.10.4	3.5.2.1	3.8.4.2	3.4.2.1
3.5.10.5	3.5.2.2	3.8.4.2.1	3.4.2.1.1, 3.4.2.1.1.1, 3.4.2.1.1.2
3.5.10.6.1	3.3.2.6		
3.5.10.6.2	3.3.8	3.8.4.2.2	3.4.2.1.2
3.5.10.6.3	3.3.7	3.8.4.2.3	3.4.2.1.3
3.5.10.6.4	3.3.7.1	3.8.4.2.4	3.4.2.1.3.1
3.5.10.6.5	3.3.9.2, 3.3.9.4	3.8.4.3	3.4.2.2
3.5.10.6.6	3.5.6.2	3.8.4.3.1	3.4.2.2.1
3.6.1.1.1	3.3.1.1	3.8.4.3.2	3.4.2.2.2
3.6.1.1.2	3.3.6, 3.3.6.1	3.9.1	3.8
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3.6.2.3	3.3.1.1	4.2.1	6.2.1
3.6.3.	3.3.7	4.2.2	6.2.2
3.6.4.1	3.3.9, 3.3.9.1, 3.3.9.2, 3.3.9.3, 3.3.9.4	4.2.3	6.2.3
		4.2.4	6.2.4
3.6.4.2	3.4.8, 3.4.9, 3.5.5.1	4.2.5	6.2.5
3.1.6.4.3	3.5.6, 3.5.6.1	4.2.6	6.2.6
3.6.4.4	3.4.6	4.2.7	6.2.7
3.6.4.5	3.4.7	4.2.8	6.2.8
3.6.5	3.3.5	4.3	6.4
		4.4	6.5
		4.5	6.6

TABLE 2. NUMERICAL CROSS-INDEX FROM MIL-F-8785C TO MIL STANDARD

MIL-F-8785C PARAGRAPH	MIL-F-8785C PARAGRAPH TITLE	CORRESPONDING MIL STANDARD AND HANDBOOK PARAGRAPHS
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1.2	Application	1.2
1.3	Classification of Airplanes	1.3
1.3.1	Land- or Carrier-Based Designation	1.3
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2.1	Issues of Documents	2.0
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3.1.2	Loadings	3.1.1
3.1.3	Moments and Products of Inertia	3.1.2
3.1.4	External Stores	3.1.3
3.1.5	Configurations	3.1.4
3.1.6	State of the Airplane	1.6
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3.1.6.2	Airplane Failure States	1.6.2
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3.1.8.3	Maximum Service Altitude	1.5.2
3.1.8.4	Service Load Factors	1.5.2
3.1.9	Permissible Flight Envelopes	1.5.3
3.1.10	Application of Levels	3.1.5, 3.1.6
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3.1.10.3.1	Ground Operation and Terminal Flight Phases	3.1.5.4
3.1.10.3.2	When Levels Are Not Specified	No Corresponding Requirement
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TABLE 2 (Continued)

MIL-F-8785C PARAGRAPH	MIL-F-8785C PARAGRAPH TITLE	CORRESPONDING MIL STANDARD AND HANDBOOK PARAGRAPHS
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3.2.2.1.2	Short-Period Damping	3.2.1.1
3.2.2.1.3	Residual Oscillations	3.2.3
3.2.2.2	Control Feel and Stability in Maneuvering Flight at Constant Speed	3.2.9.1, 3.2.10.2
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3.2.3.3.1	Longitudinal Control in Catapult Takeoff	No Corresponding Requirement
3.2.3.3.2	Longitudinal Control Force and Travel in Takeoff	3.2.9.7.1, 3.2.10.1
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3.3.2.2.1	Additional Roll Rate Requirement for Small Inputs	3.5.1.1.4
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3.3.2.6	Turn Coordination	3.5.10.6.1, 3.6.6.2.2

TABLE 2 (Continued)

MIL-F-8785C PARAGRAPH	MIL-F-8785C PARAGRAPH TITLE	CORRESPONDING MIL STANDARD AND HANDBOOK PARAGRAPHS
3.3.3	Pilot-Induced Oscillations	3.5.2, 3.6.2.2
3.3.4	Roll Control Effectiveness	3.5.9.1
3.3.4.1	Roll Performance for Class IV Airplanes	3.5.9.1
3.3.4.1.1	Roll Performance in Flight Phase CO	3.5.9.1
3.3.4.1.2	Roll Performance in Flight Phase GA	3.5.9.1, 3.5.9.6, 3.6.5.4
3.3.4.1.3	Roll Response	3.5.10.3
3.3.4.2	Roll Performance for Class III Airplanes	3.5.9.1
3.3.4.3	Roll Control Forces	3.5.10.2
3.3.4.4	Linearity of Roll Response	3.5.4
3.3.4.5	Wheel Control Throw	3.5.10.1
3.3.5	Directional Control Characteristics	3.6.5, 3.6.6
3.3.5.1	Directional Control with Speed Change	3.6.6.2.3
3.3.5.1.1	Directional Control with Asymmetric Loading	3.6.5.3, 3.6.6.2.5
3.3.5.2	Directional Control in Wave-Off (Go-around)	3.6.6.2.7
3.3.6	Lateral-Directional Characteristics in Steady Sideslips	3.5.6, 3.6.1.1.2, 3.6.6.1
3.3.6.1	Yawing Moments in Steady Sideslips	3.6.1.1.2, 3.6.6.1
3.3.6.2	Side Forces in Steady Sideslips	3.5.6
3.3.6.3	Rolling Moments in Steady Sideslips	3.5.6
3.3.6.3.1	Exception for Wave-Off (Go-around)	3.5.6
3.3.6.3.2	Positive Effective Dihedral Limit	3.5.6, 3.5.9.2
3.3.7	Lateral-Directional Control in Crosswinds	3.5.7, 3.5.10.6.3, 3.6.3, 3.6.6.2.4
3.3.7.1	Final Approach in Crosswinds	3.5.9.3, 3.5.10.6.4, 3.6.5.1
3.3.7.2	Takeoff Run and Landing Rollout in Crosswinds	3.5.9.3, 3.6.5.1
3.3.7.2.1	Cold- and Wet-Weather Operation	3.6.5.1
3.3.7.2.2	Carrier-Based Airplanes	3.6.5.1
3.3.7.3	Taxiing Wind Speed Limits	3.5.9.3, 3.6.5.1
3.3.8	Lateral-Directional Control in Dives	3.5.9.5, 3.5.10.6.2, 3.6.6.2.6
3.3.9	Lateral-Directional Control with Asymmetric Thrust	3.5.9.3, 3.6.4.1
3.3.9.1	Thrust Loss During Takeoff Run	3.5.9.4, 3.5.10.6.5, 3.6.4.1, 3.6.6.2.8
3.3.9.2	Thrust Loss After Takeoff	3.5.9.4, 3.5.10.6.5, 3.6.4.1, 3.6.6.2.8
3.3.9.3	Transient Effects	3.5.8.1, 3.6.4.1
3.3.9.4	Asymmetric Thrust-Yaw Controls Free	[illegible] 0.6.5, 3.6.4.1
3.3.9.5	Two Engines Inoperative	3.6.5.2
3.4.1	Dangerous Flight Conditions	3.1.7
3.4.1.1	Warning and Indication	3.1.7.1, 3.8.4.1

TABLE 2 (Continued)

MIL-F-8785C PARAGRAPH	MIL-F-8785C PARAGRAPH TITLE	CORRESPONDING MIL STANDARD AND HANDBOOK PARAGRAPHS
3.4.1.2	Devices for Indication, Warning, Prevention, Recovery	3.1.7.2
3.4.2	Flight at High Angle of Attack	3.8.4
3.4.2.1	Stalls	3.8.4.2
3.4.2.1.1	Stall Approach	3.8.4.2.1
3.4.2.1.1.1	Warning Speed for Stalls at 1 g Normal to the Flight Path	3.8.4.2.1
3.4.2.1.1.2	Warning Range for Accelerated Stalls	3.8.4.2.1
3.4.2.1.2	Stall Characteristics	3.8.4.2.2
3.4.2.1.3	Stall Prevention and Recovery	3.8.4.2.3
3.4.2.1.3.1	One-Engine-Out Stalls	3.8.4.2.4
3.4.2.2	Post-Stall Gyrations and Spins	3.8.4.3
3.4.2.2.1	Departure from Controlled Flight	3.8.4.3.1
3.4.2.2.2	Recovery from Post-Stall Gyrations and Spins	3.8.4.3.2
3.4.3	Cross-Axis Coupling in Roll Maneuver	3.8.1
3.4.4	Control Harmony	3.8.2
3.4.4.1	Control Force Coordination	3.8.3
3.4.5	Buffet	3.2.7.6
3.4.6	Release of Stores	3.2.7.4, 3.3.3.1, 3.4.3, 3.5.8.4, 3.5.9.6, 3.6.4.4, 3.6.5.4
3.4.7	Effects of Armament Delivery and Special Equipment	3.2.7.5, 3.3.3.1, 3.4.3, 3.5.8.5, 3.6.4.5
3.4.8	Transients Following Failures	3.2.7.2, 3.3.3.2, 3.5.8.2, 3.6.4.2
3.4.9	Failures	3.2.7.2, 3.3.3.2, 3.5.8.2, 3.6.4.2
3.4.10	Control Margin	3.2.8.5, 3.5.9.8, 3.6.5.5
3.4.11	Direct Force Controls	3.6.1.2.1, 3.7.1
3.5.1	General Characteristics	No Corresponding Paragraph
3.5.2	Mechanical Characteristics	No Corresponding Paragraph
3.5.2.1	Control Centering and Breakout Forces	3.2.9.5, 3.3.5, 3.4.5, 3.5.10.4
3.5.2.2	Cockpit Control Free Play	3.2.9.6, 3.5.10.5
3.5.2.3	Rate of Control Displacement	3.2.10.3
3.5.2.4	Adjustable Controls	No Corresponding Requirement
3.5.3	Dynamic Characteristics	3.2.1.1, 3.5.1.1.5
3.5.3.1	Damping	No Corresponding Requirement
3.5.4	Augmentation Systems	No Corresponding Requirement
3.5.5	Failures	No Corresponding Requirement (See 3.2.7.2, 3.5.8.2, 3.6.4.2)

TABLE 2 (Continued)

MIL-F-8785C PARAGRAPH	MIL-F-8785C PARAGRAPH TITLE	CORRESPONDING MIL STANDARD AND HANDBOOK PARAGRAPHS
3.5.5.1	Failure Transients	3.2.7.2, 3.5.8.2, 3.6.4.2
3.5.5.2	Trim Changles Due to Failures	3.2.9.7.6, 3.6.6.2.9
3.5.6	Transfer to Alternate Control Modes	3.2.7.3, 3.5.8.3, 3.6.4.3
3.5.6.1	Transfer Transients	3.2.7.3, 3.5.8.3, 3.6.4.3
3.5.6.2	Trim Changes	3.2.9.7.7, 3.5.10.6.6, 3.6.6.2.10
3.6.1	Trim System	3.2.9.8
3.6.1.1	Trim for Asymmetric Thrust	No Corresponding Requirement
3.6.1.2	Rate of Trim Operation	3.2.9.7.3, 3.2.9.8.1
3.6.1.3	Stalling of Trim Systems	3.2.9.8.2
3.6.1.4	Trim System Irreversibility	3.2.9.8.3
3.6.2	Speed and Flight Path Control Devices	3.2.5, 3.3.2
3.6.3	Transients and Trim Changes	No Corresponding Requirement (See 3.2.7.1)
3.6.3.1	Pitch Trim Changes	No Corresponding Requirement
3.6.4	Auxiliary Dive Recovery Devices	No Corresponding Requirement (See 3.2.5)
3.7.1	Form of the Disturbance Models	3.9.2
3.7.1.1	Turbulence Model (von Karman Form)	3.9.2
3.7.1.2	Turbulence Model (Dryden Form)	3.9.2
3.7.1.3	Discrete Gust Model	3.9.2
3.7.2	Medium/High-Altitude Model	3.9.2
3.7.2.1	Turbulence Scale Lengths	3.9.2
3.7.2.2	Turbulence Intensities	3.9.2
3.7.2.3	Gust Intensities	3.9.2
3.7.2.4	Gust Magnitudes	3.9.2
3.7.3	Low-Altitude Disturbance Models	3.9.2
3.7.3.1	Wind Speeds	3.9.2
3.7.3.2	Wind Shear	3.9.2
3.7.3.3	Vector Shear	3.9.2
3.7.3.4	Turbulence	3.9.2
3.7.3.5	Gusts	3.9.2
3.7.4	Carrier-Landing Disturbance Model	3.9.2
3.7.4.1	Free-Air Turbulence Components	3.9.2
3.7.4.2	Steady Component of Carrier Airwake	3.9.2
3.7.4.3	Periodic Component of Carrier Airwake	3.9.2
3.7.4.4	Random Component of Carrier Airwake	3.9.2
3.7.5	Application of the Disturbance Model In Analysis	3.9.3
3.8	Requirements for Use of the Disturbance Models	3.9.1
3.8.1	Use of Disturbance Models	3.9.1

TABLE 2 (Concluded)

MIL-F-8785C PARAGRAPH	MIL-F-8785C PARAGRAPH TITLE	CORRESPONDING MIL STANDARD AND HANDBOOK PARAGRAPHS
3.8.2	Qualitative Degrees of Suitability	3.9.1
3.8.3	Effects of Atmospheric Disturbances	3.1.5.1, 3.1.5.2
3.8.3.1	Requirements for Airplane Normal States	3.1.5.1, 3.1.5.2
3.8.3.2	Requirements for Airplane Failure States	3.9.4
4.1	Compliance Demonstration	3.1.10.1, 3.1.10.2
4.1.1.1	Effects of Failure States	3.1.6
4.1.1.2	Effects of Atmospheric Disturbances	1.7, 3.9.1, 3.9.4
4.1.1.3	Computational Assumptions	3.1.6.1
4.1.2	Simulation	3.1.10.1
4.1.3	Flight Test Demonstration	3.1.10.1
4.2	Airplane States	1.6.1
4.2.1	Weights and Moments of Inertia	3.1.10.2
4.2.2	Center-of-Gravity Positions	3.1.10.2
4.2.3	Thrust Settings	No Corresponding Requirement
4.3.1	Altitudes	3.1.10.2
4.3.2	Special Conditions	3.1.10.2
4.4	Tests at Specialized Facilities	No Corresponding Requirement
6.1	Intended Use	4.1
6.2	Definitions	4.2
6.2.1	General	4.2.1
6.2.2	Speeds	4.2.2
6.2.3	Thrust and Power	4.2.3
6.2.4	Control Parameters	4.2.4
6.2.5	Longitudinal Parameters	4.2.5
6.2.6	Lateral-Directional Parameters	4.2.6
6.2.7	Atmospheric Disturbance Parameters	4.2.7
6.2.8	Terms Used in High Angle of Attack Requirements	4.2.8
6.3	Interpretation of F_s/n Limits of Table V	3.2.9.1
6.4	Gain Scheduling	4.3
6.5	Engine Considerations	4.4
6.6	Effects of Aeroelasticity, Control, Equipment and Structural Dynamics	4.5
6.7	Application of Levels	1.7
6.7.1	Level Definitions	1.7
6.8	Related Documents	2.0
6.9	Marginal Indicia	No Corresponding Paragraph

for demonstrating compliance with the requirement, it belongs in the Requirements section along with all other pertinent definitions, i.e., allowable Levels, Failure States, configurations, etc.

4. LEVELS OF FLYING QUALITIES

Two definitions of flying qualities which are related but not exactly equivalent are given in the Handbook — the definitions from MIL-F-8785C and the Cooper-Harper scale. The Cooper-Harper scale is included as an alternative to account for the fact that all existing quantitative boundaries are based on fairing the 3-1/2 (= Level 1) and 6-1/2 (= Level 2) ratings. It therefore seems reasonable to use the same scale when establishing flying quality levels by flight or simulator demonstrations. Additionally, use of the Cooper-Harper scale forces a definition of the specific flying quality task elements for each mission. The existing MIL-F-8785C adjectival definitions of Levels has been retained as an alternative. These definitions can be applied interchangeably at the discretion of the procuring activity. However, use of the Cooper-Harper scale is strongly encouraged because it will result in:

- More rigorous task definitions by the testing activity.
- A data base generated for each new aircraft which is usable to develop new, and refine existing, criteria.
- More meaningul comments in defense of the ratings.

5. ATMOSPHERIC DISTURBANCES

It is well known that atmospheric disturbances play a major role in distinguishing between good and bad flying qualities. We have attempted to account for this by developing a standardized model of "Moderate" turbulence to be used when demonstrating compliance via simulation. This model consists of a specific rms random gust component, a set of critical wind shears, and steady crosswinds for landing. It is a relatively simple model that is specifically oriented toward exposing handling quality deficiencies. Detailed information is given for mechanizing the model on a digital computer.

A new scale is proposed for identifying allowable degradations in handling qualities in specifically defined atmospheric disturbances. Finally, a scale is also given as a guideline to allow test pilots to specify the magnitude of turbulence encountered in flight.

6. REVISED AND NEW CRITERIA

The scope of this effort did not allow a significant amount of criteria development. In fact, the guidelines were to utilize only existing criteria in the literature with minor refinements as required. It was clearly necessary to carefully pick and choose what areas deserved the most attention. It was decided to concentrate on development of criteria that would be specifically applicable to highly augmented aircraft, reflected in part by the results of an informal industry survey taken at the 1981 AIAA Atmospheric Flight Mechanics Conference. This survey indicated unanimous agreement that one of the major weaknesses of MIL-F-8785C is that many of the requirements are not easily applied to highly augmented aircraft.

The following paragraphs include significant revisions to the MIL-F-8785C requirements that appear in the Standard and Handbook either in terms of new criteria, revised criteria, or simply the addition of guidance for application.

MIL Standard Paragraph	Title
3.1.5	Allowable Levels for Aircraft Normal States
3.1.6	Allowable Levels for Aircraft Failure States
3.2.1	Pitch Attitude Response to Pitch Controller
3.2.2	Pilot-Induced Pitch Oscillations
3.2.9.4	Control Force Versus Control Deflection
3.5.1	Roll Response to Roll Controller
3.6.1.2	Yaw Axis Bandwidth Requirements
3.6.2.	Yaw Response to Roll Controller

3.7.1	Bandwidth Requirement for Lateral Translation
3.8.4	Flight at High Angle of Attack
3.9	Handling Quality Requirements in Atmospheric Disturbances
3.9.1	Allowable Flying Quality Degradations in Atmospheric Disturbances
3.9.2	Definition of Atmospheric Disturbance Form
3.9.3	Application of Disturbance Models in Analyses

7. LOWER-ORDER EQUIVALENT SYSTEM FITTING PROGRAM

The use of lower-order equivalent systems in the MIL Handbook requires a computer program to fit the lower-order form to the higher-order system. In fact, Paragraph 3.2.1.1 refers to such a program as appearing in "Appendix A." The program being referred to was written at the McDonnell Aircraft Company in St. Louis and also exists in-house at AFWAL/FIGC and NADC. While it was originally intended to include a listing of this program in Appendix A, further consideration deemed this to be impractical, primarily due to the problems associated with compatibility of software on various computers. As an alternative, we have elected to provide a reasonably detailed description of the basic algorithms required to perform fitting of lower-order equivalent systems.

8. EXAMPLE MIL STANDARD

An example MIL Standard for a representative airplane appears in Appendix B. This Standard consists of the proposed MIL Standard of Volume I, with the blanks filled in using recommended Handbook requirements. It is not meant to represent a procurement document for any existing or proposed airplane, but is included to illustrate the procedures involved in applying the MIL Standard and Handbook concept.

SECTION 1

1. SCOPE AND OPERATIONAL OBJECTIVES

DISCUSSION

This section of the specification has been used to define a general framework that permits tailoring each requirement according to:

1) The kind of aircraft (Mission Class)

2) The job to be done (Flight Phase and Envelope)

3) How well the job must be done (State and Level)

The framework then comprises these paragraphs:

1.1 Scope

1.2 Application

1.3 Classification of Aircraft (Mission)

1.4 *Flight Phase Categories*

1.5 Flight Envelopes

1.6 Aircraft States

1.7 Levels of Flying Qualities

In the following paragraphs each of these elements is defined and discussed.

1.1 SCOPE

A. REQUIREMENT

1.1 Scope. This specification contains the requirements for the flying and ground handling qualities of a U.S. military aircraft. It is intended to assure flying qualities for adequate mission performance and flight safety regardless of the design implementation or flight control system augmentation.

B. RELATED MIL-F-8785C REQUIREMENT

1.1

C. DISCUSSION

The scope is essentially unchanged from that of MIL-F-8785C. The requirements are not aimed at unconventional aircraft such as helicopters, V/STOL, or re-entry vehicles, but many of the requirements may be found to apply reasonably well to those aircraft in specific instances. Separate flying qualities specifications are being prepared for these vehicles. The emphasis is now to be on the complete aircraft, including flight control system augmentation.

1.2 APPLICATION

A. REQUIREMENT

1.2 Application. The flying qualities of the aircraft proposed or contracted for shall be in accordance with this specification. The requirements are written in terms of the axis of vehicle motion and include all aspects of control for that axis, as well as vehicle responses to other inputs, e.g., turbulence, store release, etc. This approach therefore includes requirements for other (i.e., secondary) methods of control for a given axis (DLC, speed brakes, etc.). The requirements apply, as stated, to the combination of airframe and related subsystems. This includes stability augmentation and flight control systems (automatic and/or manual), when provided.

B. RELATED MIL-F-8785C REQUIREMENT

1.2

C. DISCUSSION

The flying qualities of the proposed or contracted aircraft shall be in accordance with the provisions of the MIL Standard. The requirements apply as stated to the combination of airframe and related subsystems. Stability augmentation and control augmentation are specifically to be included when provided in the aircraft. The change here pertains to the way in which the requirements are now written, i.e., by axis as opposed to the flight controls.

Additional or alternative requirements imposed by the procuring activity are contained in the Standard.

1.3 AIRCRAFT CLASSIFICATION AND OPERATIONAL MISSIONS

A. REQUIREMENT

1.3 Aircraft Classification and Operational Missions. For the purpose of this Standard, the aircraft specified in this requirement is to accomplish the following missions: ________________. The aircraft thus specified will be a Class ___ aircraft.

B. RELATED MIL-F-8785C REQUIREMENTS

1.3, 1.3.1, 3.1.1

C. DISCUSSION/APPLICATION

1. Mission

Unfortunately, the word "mission" is used in several contexts not only in this specification, but throughout the writings pertinent to acquiring a new weapon system. In the broadest sense, "operational missions" applies to classifying the aircraft as fighter, bomber, reconnaissance, etc., or as in "accomplishing the mission" of bombing, strafing, etc. In 1.3 the object is to introduce to the designer in general terms the function of the vehicle he is to design. It should be sufficient for the procuring activity to refer to those paragraphs of the Handbook to define the overall performance requirements, the operational requirements, employment and deployment requirements.

The operational missions considered should not be based on just the design mission profiles. 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 missions.

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

The foregoing discussion serves to emphasize the importance of the intended use of the aircraft and the impact this has on the configurations, loadings, and Operational Flight Envelopes for which the aircraft is to be designed. Once the intended uses or operational missions are defined, a Flight Phase analysis of each mission must be conducted. With the Flight Phases established, the configurations and loading states which will exist during each Phase can be defined.

2. Aircraft Classification and Operational Missions

a. Classification of Aircraft

An aircraft is placed in one of the following Classes:

Class I: Small light aircraft such as:

- Light utility
- Primary trainer
- Light observation

Class II: Medium weight, low-to-medium maneuverability aircraft 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
- Trainer for Class II

Class III: Large, heavy, low-to-medium maneuverability aircraft such as:

Heavy transport/cargo/tanker
Heavy bomber
Patrol/early warning/electronic countermeasures/airborne command, control, or communications relay
Trainer for Class III

Class IV: High-maneuverability aircraft such as:
Fighter-interceptor
Attack
Tactical reconnaissance
Observation
Trainer for Class IV

The procuring activity will assign an aircraft to one of these Classes, and the Handbook requirements for that Class are meant to apply. When no Class is specified in the requirement, the requirement is meant to apply to all Classes. When operational missions so dictate, an aircraft of one Class should be required by the procuring activity to meet selected requirements ordinarily specified for aircraft of another Class.

The letter -L following a Class designation identifies an aircraft as land-based; carrier-based aircraft are similarly identified by -C. When no such differentiation is made in a requirement, the requirement is meant to apply to both land-based and carrier-based aircraft.

The classification scheme simplifies mission definition. Basically, the 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 in 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 craft, should be designed for high limit load factor. These vehicles tend to group in the weight range from 5000 to 100,000 lb. There are a few small, lightweight 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 should be on the basis of more detailed information about the

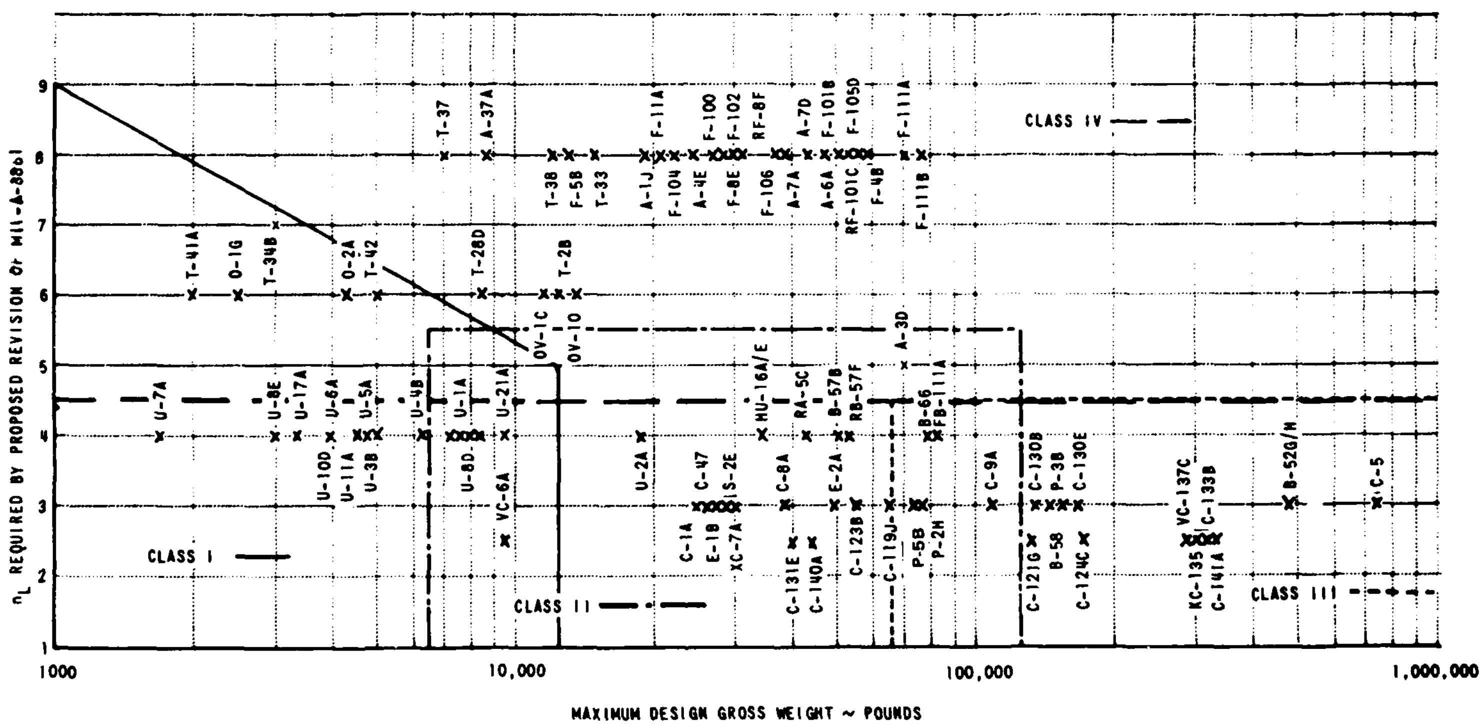


Figure 1 (1.3). Classification of Aircraft (from Reference 11) intended use; or, alternatively, the detail specification should be composed of requirements selected from those stated for both of these Classes in the MIL Handbook. Figure 1 also illustrates that all other aircraft are required to be designed for a limit load factor of less than 4 g, and that current aircraft span the weight range from 1000 to almost 1,000,000 lb. 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 the motions and riding qualities. If the effect 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.

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 desirable 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 experience is required in this area.

1.4 FLIGHT PHASE CATEGORIES

A. REQUIREMENT

1.4 Flight Phase Categories. To accomplish the mission requirements the following general Flight Phase categories are involved: ______________________________. Special Flight Phases to be considered are: ______________________________.

B. RELATED MIL-F-8785C REQUIREMENT

1.4

C. DISCUSSION/APPLICATION

Experience with airplane 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 Aircraft Normal State associated with it (e.g., flaps and gear down for landing approach and up for cruising flight). In many instances, therefore, the flying qualities specification should state requirements as a function of mission Flight Phase. This degree of breakdown gives the designer additional guidance in optimizing his design so that each Aircraft State has adequate flying qualities for the tasks to be performed while the aircraft is in that State.

In flight and simulator evaluations, pilots generally rate a set of flying qualities on suitability for a given mission segment like one of these Flight Phases. The pilots assign an overall rating, based on ability, or 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; and of flight in turbulence. These considerations have led naturally to identifying flying qualities requirements in terms of the three Flight Phase categories given below:

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. Weapon 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)

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 (CL)
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)

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. Approach (PA)
d. Wave-off/go-around (WO)
e. Landing (L)

When necessary, recategorization or addition of Flight Phases or delineation of requirements for special situations, e.g., zoom climbs, will be specified by the procuring activity.

These Flight Phases shall 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, requirements are 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 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 current use of the T-37, T-28, etc.).

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. First, the possible Flight Phases were divided into two groups on the basis of terminal and nonterminal operation. Then nonterminal flight was further divided into two groups based primarily on the degree of maneuverability and/or precision of control required. The requirements of Section 3 are generally stated in terms of these three Flight Phase Categories. However, a number of the requirements are directed at specific Flight Phases; those requirements apply only to the specific Flight Phase stated.

Not all of these Flight Phases apply to a given aircraft. Those that are appropriate to design operational missions and emergencies will be chosen for each design. The list cannot be exhaustive because new mission requirements continue to be generated. Thus the procuring activity may delete some Phases and add others. Responsibility for choosing applicable Flight Phases has been defined contractually in the Standard, i.e., the procuring activity should have prepared the initial listing of Flight Phases. The contractor is therefore made contractually responsible for assuring that this listing is inclusive and exhaustive (for the stated primary and alternate missions), and for suggesting necessary additions so that the intent of the Flight Phase concept will be accomplished (i.e., there will be no gap between successive phases of every flight, and transition between phases of each flight will be smooth). It is the procuring activity'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 cannot be measured in flight, and the effectiveness of aerodynamic controls is necessarily low at low dynamic pressure. Lacking enough data to formulate general requirements for these cases, it is left for the procuring activity to provide specific requirements as specific mission needs dictate.

1.5 FLIGHT ENVELOPES

DISCUSSION

Three envelopes are to be defined. One, the Operational Envelope, is set down by the procuring activity (in Paragraph 1.3) for primary and alternate missions, including maneuverability over the speed-altitude range, and represents the minimum requirements. At this stage the Flight Phases will also be known (from Paragraph 1.4). In response to these and other requirements the contractor can then design the aircraft and:

- Further define the Operational Flight Envelope for each Flight Phase, based on the associated Aircraft Normal States,
- Construct the larger Service Flight Envelope for the Aircraft Normal State associated with each Flight Phase, and
- Similarly construct portions of the Permissible Flight Envelope boundaries, beyond which operation is not allowed.

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

The envelopes are described by the specification of a two-dimensional (speed and load factor) figure representing the conditions where the requirements apply. An example that defines terms for the Operational and Service Envelopes is shown in Figure 1. The load factor, n, denotes maneuverability without regard to thrust available, i.e., the flying qualities specification places no requirements on load-factor capability in constant-speed level flight. These Envelopes are defined at various altitudes corresponding to the Flight Phases; thus they could be considered to be three-dimensional. The aircraft flying qualities in all three Envelopes will always be "acceptable," or better.

Some Flight Phases of the same Category will involve the same, or very similar, Aircraft Normal States; so one set of Flight Envelopes may represent several Flight Phases. Each Flight Phase will involve a

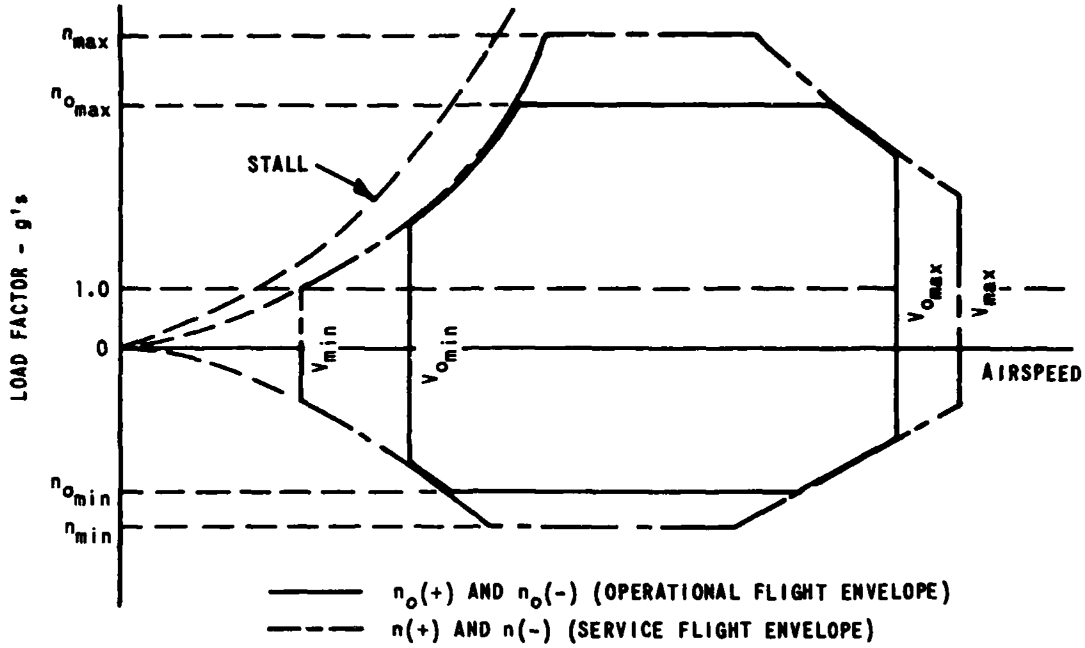


Figure 1 (1.5). Definition of Flight Envelope Terms

range of loadings. Generally it will be convenient to represent this variation by superimposing boundaries for the discrete loadings of Paragraph 3.1.1, or possibly by bands denoting extremes. If different external store complements affect the Envelope boundaries significantly, it may be necessary for the contractor to construct several sets of Envelopes for each Flight Phase, each set representing a family of stores. A manageably small total number of Envelopes should result. It is apparent that the Flight Envelopes must and can 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) Flight Envelopes. Generally, flight tests will cover the Service Flight Envelope, with specific tests (stalls, dives, etc.) to the Permissible limits. The same test procedures usually apply in both Service and Operational Envelopes; only the numerical requirements and qualitative levels differ. If, for example, speed and altitude are within the Operational Flight Envelope but normal load factor is between the Operational and Service Flight Envelope boundaries, the requirements for the Service Flight Envelope apply. Ideally, the flight test program should also lead to definition of Flight Envelopes depicting Level 1 and Level 2 boundaries. These Level boundaries should aid the using commands in tactical employment, even long after the procurement contract has been closed out.

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

It should also be noted that the boundaries of these envelopes should not be set by ability to meet the flying qualities requirements. The flying qualities requirement should be met within the boundaries which normally are set by other factors unless specific deviations are granted. The only exception is control power, which may set some boundaries if the requirements on the Operational Flight Envelope are still met. The rationale for each type of Envelope is presented below in the discussion of each subparagraph.

1.5.1 Operational Flight Envelopes

A. REQUIREMENT

1.5.1 Operational Flight Envelopes. The Operational Flight Envelopes define the boundaries in terms of speed, altitude and load factor within which the aircraft must be capable of operating in order to accomplish the missions of Paragraph 1.3. Envelopes for each applicable Flight Phase are as follows: __________. In the absence of the above, the contractor shall use the representative conditions of Table 1 of the Handbook for the applicable Flight Phases.

B. RELATED MIL-F-8785C REQUIREMENT

3.1.7

C. DISCUSSION/APPLICATION

Operational Flight Envelopes 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 that the aircraft can be controlled without undue concentration. The required size of the Operational Flight Envelopes for a particular aircraft has been given in Paragraph 1.3; however, this can further be delineated by using Table 1 for each flight phase category. Additional boundaries will be provided by the contractor.

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

a) The Operational Flight Envelope for a given Flight Phase should initially be considered to be as large a portion of the associated Service Flight Envelope as possible, to permit the greatest freedom of use of the aircraft.

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

TABLE 1 (1.5.1)

OPERATIONAL FLIGHT ENVELOPE

FLIGHT PHASE CATEGORY	FLIGHT PHASE	AIRSPEED		ALTITUDE		LOAD FACTOR	
		$V_{o_{min}}$ ($M_{o_{min}}$)	$V_{o_{max}}$ ($M_{o_{max}}$)	$h_{o_{min}}$	$h_{o_{max}}$	$n_{o_{min}}$	$n_{o_{max}}$
A	AIR-TO-AIR COMBAT (CO)	$1.4\ V_S$	V_{MAT}	MSL	Combat Ceiling	-1.0	n_L
	GROUND ATTACK (GA)	$1.3\ V_S$	V_{MRT}	MSL	Medium	-1.0	n_L
	WEAPON DELIVERY/LAUNCH (WD)	V_{range}	V_{MAT}	MSL	Combat Ceiling	.5	*
	AERIAL RECOVERY (AR)	$1.2\ V_S$	V_{MRT}	MSL	Combat Ceiling	.5	n_L
	RECONNAISSANCE (RC)	$1.3\ V_S$	V_{MAT}	MSL	Combat Ceiling	*	*
	IN-FLIGHT REFUEL (RECEIVER) (RR)	$1.2\ V_S$	V_{MRT}	MSL	Combat Ceiling	.5	2.0
	TERRAIN FOLLOWING (TF)	V_{range}	V_{MAT}	MSL	10,000 ft.	.0	3.5
	ANTISUBMARINE SEARCH (AS)	$1.2\ V_S$	V_{MRT}	MSL	Medium	0	2.0
	CLOSE FORMATION FLYING (FF)	$1.4\ V_s$	V_{MAT}	MSL	Combat Ceiling	-1.0	n_L
B	CLIMB (CL)	$.85\ V_{R/C}$	$1.3\ V_{R/C}$	MSL	Cruising Ceiling	.5	2.0
	CRUISE (CR)	V_{range}	V_{NRT}	MSL	Cruising Ceiling	.5	2.0
	LOITER (LO)	$.85\ V_{end}$	$1.3\ V_{end}$	MSL	Cruising Ceiling	.5	2.0
	IN-FLIGHT REFUEL (TANKER) (RT)	$1.4\ V_S$	V_{MAT}	MSL	Cruising Ceiling	.5	2.0
	DESCENT (D)	$1.4\ V_S$	V_{MAT}	MSL	Cruising Ceiling	.5	2.0
	EMERGENCY DESCENT (ED)	$1.4\ V_S$	V_{max}	MSL	Cruising Ceiling	.5	2.0
	EMERGENCY DECELERATION (DE)	$1.4\ V_S$	V_{max}	MSL	Cruising Ceiling	.5	2.0
	AERIAL DELIVERY (AD)	$1.2\ V_S$	200 kt	MSL	10,000 ft	0	2.0
C	TAKEOFF (TO)	Minimum Normal Takeoff Speed	V_{max}	MSL	10,000 ft.	.5	2.0
	CATAPULT TAKEOFF (CT)	Minimum Catapult End Airspeed	$V_{o_{min}}$ +30 kt	MSL	—	.5	n_L
	APPROACH (PA)	Minimum Normal Approach Speed	V_{max}	MSL	10,000 ft.	.5	2.0
	WAVE-OFF/GO-AROUND (WO)	Minimum Normal Approach Speed	V_{max}	MSL	10,000 ft.	.5	2.0
	LANDING (L)	Minimum Normal Landing Speed	V_{max}	MSL	10,000 ft.	.5	2.0

*Appropriate to the operational mission.

c) The Operational Flight Envelopes should 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 should 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. FAR Part 25 (Reference 118) specifies 1.3 V_s for landing-distance calculations; while Part 23 (Reference 161) specifies approach at 1.5 V_s for these calculations when required.

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 in the large Envelopes of Items a-d, above, consideration should be given to restricting the Operational Flight Envelope toward the minimum consistent with the requirements of the Flight Phase of the operational mission under consideration.

1.5.2 Service Flight Envelopes

A. REQUIREMENT

1.5.2 Service Flight Envelopes. For each Aircraft Normal State the contractor shall establish, subject to the approval of the procuring activity, Service Flight Envelopes showing combinations of speed, altitude, and normal acceleration derived from aircraft limits as distinguished from mission requirements. For each applicable Flight Phase and Aircraft Normal State, the boundaries of the Service Flight Envelopes can be coincident with or lie outside the corresponding Operational boundaries. The boundaries of the Service Flight Envelopes shall be based on considerations discussed in the Handbook.

B. RELATED MIL-F-8785C REQUIREMENT

3.1.8, 3.1.8.1, 3.1.8.2, 3.1.8.3, 3.1.8.4

C. DISCUSSION

The Service Flight Envelope encompasses the Operational Flight Envelopes for the same Flight Phase and Aircraft Normal State. Its larger volume denotes the extent of flight conditions that can be encountered without fear of exceeding aircraft limitations (safe margins should be determined by simulation and flight test). At least Level 2 handling qualities are required for normal operation. This allows a pilot to accomplish the mission Flight Phase associated with the Aircraft Normal State although mission effectiveness or pilot workload, or both, may suffer somewhat.

This Envelope is also intended to insure that any deterioration of handling qualities will be gradual as flight progresses beyond the limits of the Operational Flight Envelope. 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 Operational Flight Envelope.

The boundaries of the Service Flight Envelopes shall be based on the speed, altitude, and load factor considerations discussed below.

1. Maximum Service Speed

The maximum service speed, V_{max} or M_{max}, for each altitude is 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.

V_{max} need be no greater than that for which recovery is possible at the corresponding altitude and dive angle, with a 2000 ft margin above Mean Sea Level. In setting the maximum service speed, the designer 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 foot clearance above sea level while remaining within the Service Flight Envelope.

2. Minimum Service Speed

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

a) $1.1\ V_s$

b) V_s + 10 knots 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 specifications.

3. Maximum Service Altitude

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

4. Service Load Factors

Maximum and minimum service load factors, n(+) [n(−)], shall be established as a function of speed for several significant altitudes. The maximum [minimum] service load factor, when trimmed for 1 g flight at a particular speed and altitude, is the lowest [highest] algebraically of:

a) The positive [negative] structural limit load factor.

b) The steady load factor corresponding to the minimum allowable stall warning angle of attack (3.8.4.2).

c) The steady load factor at which the pitch control is in the 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.

1.5.3 Permissible Flight Envelopes

A. REQUIREMENT

1.5.3 Permissible Flight Envelopes. The contractor shall define Permissible Flight Envelopes which encompass all regions in which operation of the aircraft is both allowable and possible, and which the aircraft is capable of safely encountering. These Envelopes define boundaries in terms of speed, altitude, and load factor.

B. RELATED MIL-F-8785C REQUIREMENT

3.1.9

C. DISCUSSION

Basically, the Permissible Flight Envelope is designed such that, from all points within it, it shall be possible to readily and safety return to the Service Flight Envelope without exceptional pilot skill or technique, regardless of component or system failures. The requirements on stall, spin, and dive characteristics, on dive recovery devices, and on approach to dangerous flight conditions shall also apply.

In this regard, the maximum permissible speed in dives or level flight, 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 must, as a minimum, define the boundaries given below.

1. Maximum Permissible Speed

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.

Maximum permissible speed need be no greater than that for which recovery is possible at the corresponding altitude and dive angle, with a 2000 ft margin above Mean Sea Level. To allow for inadvertent excursions beyond placard speed, some margin should be provided between the maximum permissible speed and the high-speed boundaries of the Operational and Service Flight Envelopes. Such a margin is not specified because no satisfactory general requirement could be formulated. However, for specific designs, the procuring activity should consider 1.1 V_H (commonly used for structural specification) or the upset requirements of FAR Part 25 (Reference 118) and Advisory Circular AC 25.253-1A (Reference 162).

2. Minimum Permissible Speed

For aircraft where maximum lift determines minimum speed, the minimum permissible speed in 1 g flight is V_S as defined in 6.2.2. 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 Permissible Flight Envelope, subject to the approval of the procuring activity. This defined minimum permissible speed shall be used as V_S in all applicable requirements, and no requirements will apply below this speed.

1.6 STATE OF THE AIRCRAFT

DISCUSSION

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

1.6.1 Aircraft Normal States

A. REQUIREMENT

1.6.1 Aircraft Normal States. The contractor shall define and tabulate all pertinent items to describe the Aircraft Normal States (no component or system failure) associated with each of the applicable Flight Phases. This tabulation shall be in the format of Table 1 and shall use the nomenclature specified in 4.2. Certain items, 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 will be treated as specific States, and which include the most critical values and the extremes encountered during the Flight Phase in question.

B. RELATED MIL-F-8785C REQUIREMENTS

3.1.6.1, 4.2

C. DISCUSSION

These paragraphs introduce the Aircraft State terminology for use in the requirements. The contractor is required to define the Aircraft Normal States for each applicable Flight Phase, in the format of Table 1. If the position of any particular design feature can affect flying qualities independently of the items in Table 1, its position should be tabulated as well. Initially, variable parameters should be presented in discrete steps small enough to allow accurate interpolation to find the most critical values or combinations for each requirement; then those critical cases should be added. As discussed under 3.1.1 through 3.1.3, center-of-gravity positions that can be attained only with prohibited, failed, or malfunctioning fuel sequencing need not be considered for Aircraft Normal States.

TABLE 1 (1.6.1)

AIRCRAFT 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	CL													
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													

1.6.2 Aircraft Failure States

A. REQUIREMENT

1.6.2 Aircraft Failure States. The contractor shall define and tabulate all Aircraft Failure States, which consist of Aircraft Normal States modified by one or more malfunctions in aircraft components or systems; for example, a discrepancy between a selected configuration and an actual configuration. Those malfunctions that result in center-of-gravity positions outside the center-of-gravity envelope defined in 3.1.1 shall be included. Each mode of failure shall be considered. Failures occurring in any Flight Phase shall be considered in all subsequent Flight Phases.

B. RELATED MIL-F-8785C REQUIREMENT

3.1.6.2

C. DISCUSSION

A low level of confidence in the calculation of failure probabilities emerged from the flying qualities community during the development of the F-16. This confidence is based upon the method rather than the aircraft in question. It appears, in fact, that the SPOs generally allow a deviation from this requirement. The more practical approach, which appears to be in general use, is to define specific failures for each configuration and to consider the level of flying qualities which the aircraft degrades to when these failures occur. Such an approach has been referred to as "generic failure design." That is, no matter what happens, flying qualities must remain within some specific level as defined by the procuring activity.

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 itself 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.

1.6.3 Aircraft Special Failure States

A. REQUIREMENT

1.6.3 Aircraft Special Failure States. Certain components, systems, or combinations thereof may have extremely remote probabilities of failure during a given flight. These failure probabilities may, in turn, be very difficult to predict with any degree of accuracy. Special Failure States of this type need not be considered in complying with the requirements of Section 3 if justification for considering them as Special Failure States is submitted by the contractor and approved by the procuring activity.

B. RELATED MIL-F-8785C REQUIREMENT

3.1.6.2.1

C. DISCUSSION

In most cases, a considerable amount of engineering judgment will influence the procuring activity's decision to allow or disallow a proposed Aircraft Special Failure State. Probabilities that are extremely remote are exceptionally difficult to predict accurately. Judgments will weigh consequences against feasibility of improvement or alternatives, and against projected ability to keep high standards throughout design, qualification, production, use, and maintenance. Meeting other pertinent requirements -- MIL-F-9490, MIL-A-8860, etc. -- should be considered, as should experience with similar items. Generally, Special Failure States should be brought to the attention of those concerned with flight safety.

Several categories of Special Failure States can be distinguished. Certain items might be approved 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 should remain after failure of all engines, provided by accumulators or an auxiliary power source as appropriate.

Note that the required approval of Aircraft Special Failure States, in conjunction with certain requirements that must be met regardless of component or equipment status, can be used as desired to require a level of stability for the basic airframe, limit use of stick pushers to alleviate pitch-up, disallow rudder-pedal shakers for stall warning, rule out fly-by-wire control systems, require an auxiliary power source, force consideration of vulnerability, etc. The procuring activity should state those considerations they wish to impose, as completely as they can, at the outset; but it is evident that many decisions must be made subjectively and many will be influenced by the specific design.

1.7 LEVELS OF FLYING QUALITIES

A. REQUIREMENT

1.7 <u>Levels of Flying Qualities</u>. The acceptability of the handling characteristics of an aircraft are quantified herein in terms of "Levels" that are defined as ____________. Where possible, the requirements of Section 3 are stated in terms of three limiting values of one or more flying quality parameters. Each value, or combination of values, represents a <u>minimum</u> condition necessary to meet one of the three "Levels" of acceptability.

In some cases sufficient simulation or flight test data do not exist to allow the specification of numerical values of a flying quality parameter. In such cases it is not possible to explicitly define the lower boundary of each Level. These cases are handled by stating the required "Level" of flying qualities for specified piloting tasks, which require compliance by demonstration in flight or via piloted simulation.

It is expected that flying qualities will degrade with increasing atmospheric disturbances and/or Aircraft Failure States. To account for this, the Levels will be adjusted as a function of turbulence magnitude and failures. These adjustments to the definition of flying quality Levels are to be used for those requirements where numerical values are not specifically stated. The adjusted Level definitions should not be construed as a recommendation to degrade flying qualities with increasing values of atmospheric disturbances.

The requirements for aircraft Levels as a function of flight envelopes and failure states are presented in Paragraph 3.1.5. The effect of atmospheric disturbances on Levels is given in Paragraphs 3.9.1 and 3.9.4.

B. RELATED MIL-F-8785C REQUIREMENT

1.5

C. DISCUSSION

Where possible, the requirements of Section 3 have been stated in terms of three values of the stability or control parameter being specified. Each 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. The levels are:

Level 1 Flying qualities clearly adequate for the mission Flight Phase. Aircraft is satisfactory without improvement.

Level 2 Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists. Aircraft deficiencies warrant improvement.

Level 3 Flying qualities such that the aircraft can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate, or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed. Aircraft deficiencies require improvement.

In actual practice, the flying quality boundaries 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 in Figure 1 and the level definitions. Typically, a Cooper-Harper pilot rating of 1 to 3-1/2 defines Level 1, a Cooper-Harper rating between 3-1/2 and 6-1/2 defines Level 2, and a Cooper-Harper rating between 6-1/2 and 8 defines Level 3.

Utilization of the Cooper-Harper Pilot Rating Scale as the levels of flying qualities has the added benefit of more precise definitions which are related to pilot workload and task as well as making the pilot rating correlations consistent with the Level 1, 2, and 3 criterion boundaries in the flying quality standard. Hence the procuring activity is encouraged to utilize this method. It should be noted that the two definitions can be used in a single standard. However, this seems a likely source of confusion.

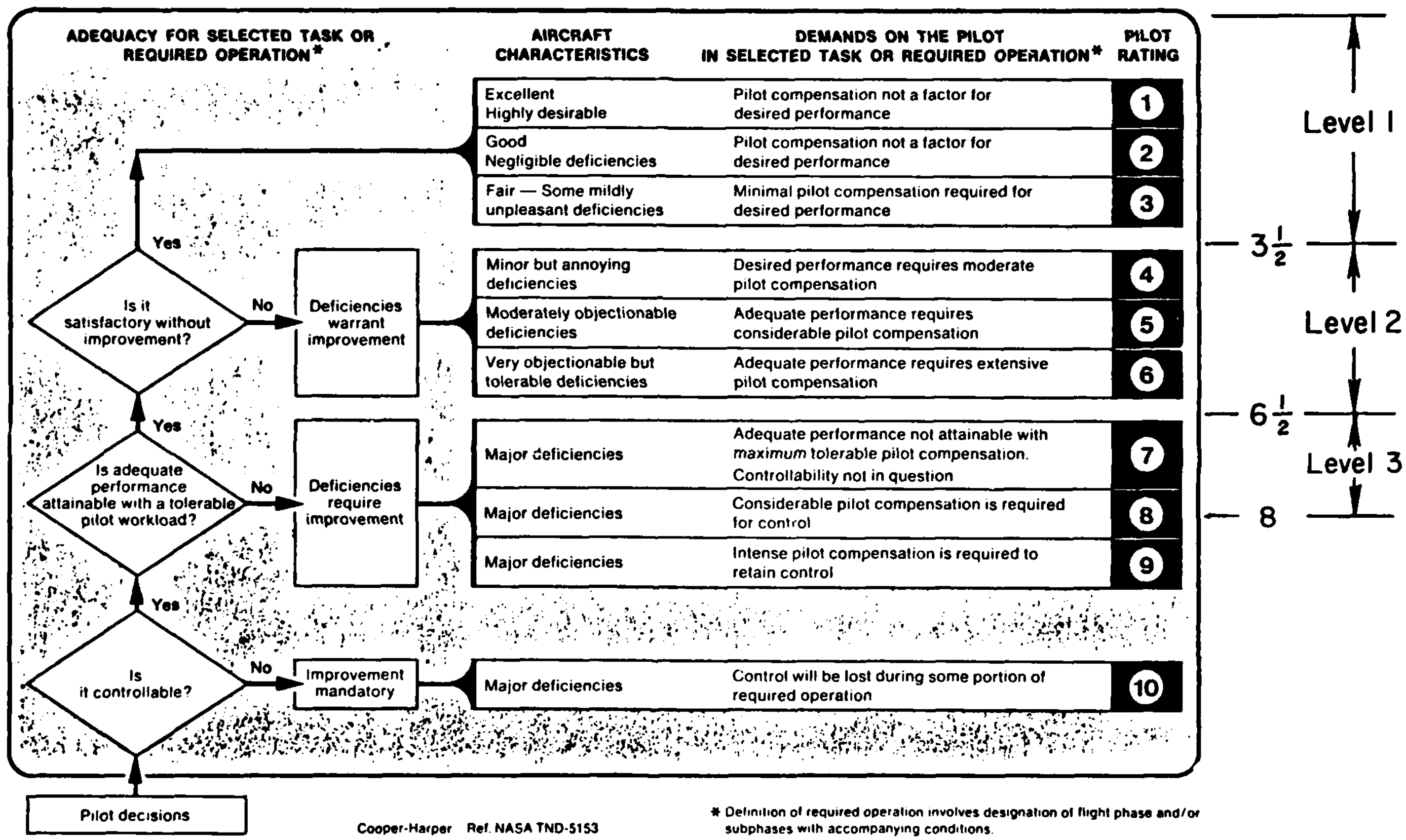


Figure 1 (1.7). Definition of Flying Quality Levels

SECTION 2

2. APPLICABLE DOCUMENTS

A. REQUIREMENT

2. APPLICABLE DOCUMENTS. The following specifications and standards, of the issue in effect on the date of invitation for bids or request for proposal, form a part of this specification to the extent specified herein. Copies of specifications and standards required by contractors in connection with specific procurement functions should be obtained from the procuring activity or as directed by the contracting officer.

Specifications: ______________________________

Standards: ___________________________________

B. RELATED MIL-F-8785C REQUIREMENTS

2., 2.1, 6.8

C. DISCUSSION

Note the phase "to the extent specified herein." Only those documents specifically referred to in filling out the blanks in the requirements are to be listed here. Recommended documents are listed below.

1. Recommended Documents

Specifications

MIL-A-8861	Airplane Strength and Rigidity — Flight Loads
MIL-D-8708	Demonstration Requirments for Airplanes
MIL-F-9490	Flight Control Systems — Design, Installation and Test of, Piloted Aircraft, General Specification for
MIL-C-18244	Control and Stabilization Systems, Automatic, Piloted Aircraft, General Specification for
MIL-F-18372	Flight Control Systems, Design, Installation and Test of, Aircraft (General Specification for)

MIL-F-83300 Flying Qualities of Piloted V/STOL Aircraft

MIL-S-83691 Stall/Post-Stall/Spin Flight Test Demonstration Requirements for Airplanes

MIL-W-25140 Weight and Balance Control Data (for Airplanes and Rotorcraft)

Standard

MIL-STD-756 Reliability Prediction

2. Related Documents

Specifications

MIL-C-5011 Charts; Standard Aircraft Characteristics and Performance, Piloted Aircraft

MIL-M-7700 Manual, Flight

MIL-A-8860 Airplane Strength and Rigidity — General Specification for

MIL-A-8871 Airplane Strength and Rigidity Flight and Ground Operations Test

MIL-G-38478 General Requirements for Angle-of-Attack-Based Systems

Standard

MIL-STD-882 Systems Safety Program for Systems and Associated Subsystems and Equipment: Requirement for

Publications

AFSC Design Handbooks:
DH 1-0 General
DH 2-0 Aeronautical Systems

AFFDL Technical Report:
TR 69-72 Background Information and User Guide for MIL-F-8785B, Military Specification — Flying Qualities of Piloted Airplanes, August 1969

AFWAL Technical Report:
TR-81-3109 Background Information and User Guide for MIL-F-8785C, Military Specification — Flying Qualities of Piloted Airplanes, July 1982

SECTION 3

3. REQUIREMENTS

3.1 GENERAL REQUIREMENTS

3.1.1 Loadings

A. REASON FOR REQUIREMENT

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

B. RELATED MIL-F-8785C REQUIREMENT

3.1.2

C. STATEMENT OF REQUIREMENT

3.1.1 Loadings. The envelope of center of gravity and weight for each flight phase shall be specified by the contractor. In addition, the contractor shall specify the maximum c.g. excursion attainable through failure in systems or components for each flight phase.

D. RATIONALE BEHIND REQUIREMENT

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 loading(s) with respect to the requirement.

Only permissible center-of-gravity positions need be considered for Aircraft Normal States. Fuel sequencing, transfer failures or malperformance, and mismanagement that might move the center of gravity outside the established limits are expressly to be considered as Aircraft Failure States. The worst possible cases that are not approved Special Failure States (1.6.3) must be examined.

Since the requirements apply over the full range of service loadings, effects of fuel slosh and shifting should 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.

The procuring activity 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. Peculiarities of configuration or possible alternative mission tasks may lead to the specification of additional loadings.

E. GUIDANCE FOR APPLICATION

Once the specific loadings are defined, application of this requirement is straightforward.

F. DEMONSTRATION OF COMPLIANCE

The procuring activity will check the material submitted for completeness. Eventually, weight and balance measurements will be made to confirm the estimates. The requirements apply to the actual flight weights and centers of gravity.

G. SUPPORTING DATA

None.

H. 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 asymmetry, for example, at prolonged small sideslip in cruising flight.

3.1.2 *Moments and Products of Inertia*

A. REASON FOR REQUIREMENT

Inertial characteristics of the aircraft affects its flying qualities, so the contractor must define the inertias for all expected loadings.

B. RELATED MIL-F-8785C REQUIREMENT

3.1.3

C. STATEMENT OF REQUIREMENT

3.1.2 Moments and Products of Inertia. The contractor shall define the moments and products of inertia of the aircraft associated with all loadings of 3.1.1. The requirements of this specification shall apply for all moments and products of inertia so defined.

D. RATIONALE BEHIND REQUIREMENT

The need for such a requirement should be self-evident.

E. GUIDANCE FOR APPLICATION

In any preliminary design of an aircraft, inertias must be estimated in order to determine the dynamic flying qualities; there should not be any difficulty in meeting this requirement.

F. DEMONSTRATION OF REQUIREMENT

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 oscillations using equipment such as that at the Air Force Flight Test Center) or parameter estimation from flight test data.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.1.3 External Stores

A. REASON FOR REQUIREMENT

Once the procuring activity has specified the stores to be considered, the contractor must assure that evaluation of the aircraft with these store combinations covers all operational flight conditions.

B. RELATED MIL-F-8785C REQUIREMENT

3.1.4

C. STATEMENT OF REQUIREMENT

3.1.3 External Stores. The external stores and store combinations to be considered are as follows: ______________________. The requirements of this Standard shall apply to these store conditions. The effects of external stores on the weight, moments of inertia, center-of-gravity position, and aerodynamic characteristics of the aircraft shall be considered for each mission Flight Phase. When the stores contain expendable loads, the requirements of this Standard apply throughout the range of store loadings.

D. RATIONALE BEHIND REQUIREMENT

Specification of stores and stores combinations affects the overall ability of the aircraft to meet its mission requirements.

E. GUIDANCE FOR APPLICATION

In determining the range of store loadings to be specified in the contract, the procuring activity should 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 excessive design penalties. Since such limitations are restrictive operationally, the procuring activity may be reluctant to approve them.

The designer should attempt to assure that there are no restrictions on store placement on the aircraft within the range of design stores.

However, it is 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 should be made physically impossible to violate necessary store loading restrictions. If this too should not be practicable, the contractor should submit both an analysis of the effects on flying qualities of violating the restrictions and an estimate of the likelihood that the restrictions will be exceeded.

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 amounts of inertia increase and instability that can be tolerated. Store separation is a prime concern.

F. DEMONSTRATION OF COMPLIANCE

The wording of the requirement makes compliance straightforward. Often the large number of possible stores combinations will, from a practical standpoint, limit flight demonstration to a few cases. A careful analysis before flight testing will assure that the most critical combinations (from a flying qualities perspective) are being evaluated.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

High-angle-of-attack testing conducted on the F-15 (Reference 251) shows a degradation in flying qualities and departure resistance with external stores. Stores tests with the F-16 (Reference 252) show 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."

3.1.4 Configurations

A. REASON FOR REQUIREMENT

This requirement is intended to assure that all expected aircraft configurations are considered, and that the conditions for compliance are sufficiently called out.

B. RELATED MIL-F-8785C REQUIREMENT

3.1.5

C. STATEMENT OF REQUIREMENT

3.1.4 Configurations. The requirements of this specification shall apply for all configurations required or encountered in the applicable Flight Phases of Section 1.4. 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 selected configurations to be examined must consist of those required for performance and mission accomplishment. Additional configurations to be investigated are defined as follows: ____________________. Control positions which activate stability augmentation necessary to meet the requirements of this standard are considered to be always ON unless otherwise specified.

D. RATIONALE BEHIND REQUIREMENT

All aircraft configurations either necessary or likely to be encountered must be evaluated.

E. *GUIDANCE FOR APPLICATION*

The designer must define the configuration or configurations which his 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 be those required for performance 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 Flight Phases, rather than for aircraft configurations. The flying qualities should 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.

F. DEMONSTRATION OF COMPLIANCE

This is defined by the requirement itself, and by any specific requirements from the procuring activity.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.1.5 Allowable Levels for Aircraft Normal States

3.1.5.1 Within Operational Flight Envelopes

3.1.5.2 Within Service Flight Envelopes

3.1.5.3 Within Permissible Flight Envelopes

3.1.5.4 For ground operation

A. REASON FOR REQUIREMENTS

Levels of flying qualities as indicated in 1.7 are employed to specify the minimum allowable handling qualities for an aircraft operating in a normal, i.e., unfailed state.

Considered as a group, this set of requirements specifies the flying qualities Levels to be attained in all areas of operations with and without turbulence.

B. RELATED MIL-F-8785C REQUIREMENTS

3.1.10, 3.1.10.1, 3.1.10.3.1, 3.1.10.3.3, 3.8.3, 3.8.3.1

C. STATEMENT OF REQUIREMENTS AND RECOMMENDED LEVELS

3.1.5 Allowable Levels for Aircraft Normal States

3.1.5.1 Within Operational Flight Envelopes. The minimum required flying qualities for the Aircraft Normal State within the Operational Flight Envelope will be Level ____. To account for degradation in handling qualities due to atmospheric disturbances the requirements will be adjusted as a function of disturbance magnitude according to the requirements of Paragraph 3.9.1.

3.1.5.2 Within Service Flight Envelopes. The minimum required flying qualities for the Aircraft Normal State within the Service Flight Envelope but outside the Operational Flight Envelope will be Level ____.

3.1.5.3 Within Permissible Flight Envelopes. From all points in the Permissible Flight Envelopes and outside the Service Flight Envelope, it shall be possible readily and safely to return to the Service Flight Envelope without exceptional pilot skill or technique. The requirements on flight at high angle of attack, dive characteristics, dive recovery devices and dangerous flight conditions shall also apply.

3.1.5.4 For ground operation. Some requirements pertaining to taxiing involve operation outside the Operational, Service, and Permissible Flight Envelopes, as at V_s or on the ground. When requirements are stated at conditions such as these, the Levels shall be applied as if the conditions were in the Operational Flight Envelope.

Recommended levels: The recommended levels are given in Table 1.

TABLE 1 (3.1.5)

RECOMMENDED LEVELS FOR AIRCRAFT NORMAL STATES

WITHIN OPERATION FLIGHT ENVELOPE	WITHIN SERVICE FLIGHT ENVELOPE
Level 1	Level 2

D. RATIONALE BEHIND REQUIREMENTS

These requirements are taken directly from MIL-F-8785C with a minor addition referencin the allowable degradations due to atmospheric disturbances as given in Paragraph 3.9.1. These Normal States represent the usual modes of piloted flight. The rationale is discussed under 1.5.2 and 1.5.3.

E. GUIDANCE FOR APPLICATION

Aircraft Normal States include both all-up operation and degradations/failures that are sufficiently probable to be considered Normal. See 3.1.6.1 for guidance on the latter.

F. DEMONSTRATION OF COMPLIANCE

Does not apply.

G. SUPPORTING DATA

None

H. LESSONS LEARNED

None.

3.1.6 Allowable Levels for Aircraft Failure States

DISCUSSION

Higher performance of aircraft 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 directed at the achievement of adequate flying qualities without imposing undue requirements that could lead to unwarranted system complexity or decreased flight safety. For example, an airplane 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 MIL Standard requires:

- High probability of good flying qualities where the aircraft 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 aircraft no matter what failures occur.
- A process to assure that all the ramifications of reliance on powered controls, stability augmentation, etc., receive proper attention.

Two options are presented to allow the procuring agency to quantitatively specify the allowable degradation in flying qualities due to failure states. The first option is unchanged from MIL-F-8785C. It involves the following failure probability calculations:

1) Identify those Aircraft Failure States which have a significant effect on flying qualities (3.1.6.2).

2) Calculate the probability of encountering various Aircraft Failure States, per flight.

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

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

The second option assumes that certain failures and/or combinations of failures will occur regardless of their probability (which is <u>not</u> calculated). As in Option 1, the degraded flying qualities for each selected Failure State are then evaluated. This approach is referred to as Generic Failure Analysis. The generic failure analysis (Option 2) is provided to allow a formal Handbook requirement that reflects current industry practice. The procuring activity may in fact require probability calculations for certain axes and/or system components and a generic failure analysis for others. The generic failure analysis therefore encompasses the requirements for specific failures of MIL-F-8785C (3.1.10.2.1), Reference 4.

3.1.6.1 Probability Calculations

A. REASON FOR REQUIREMENT

This requirement is included to provide a sound analytical method for accounting for the effects of failures.

B. RELATED MIL-F-8785C REQUIREMENT

3.1.10.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.1.6.1 Probability Calculation. When Aircraft Failure States exist (1.6.2), a degradation in flying qualities is permitted only if the probability of encountering a lower Level than specified in Para. 3.1.5 is sufficiently small. The contractor shall determine, based on the most accurate available data, the probability of occurrence of each Aircraft Failure State per flight hour within the Operational and Service Flight Envelopes. Each specific failure is assumed to be present at whichever point in the Flight Envelope being considered is most critical (in the flying qualities sense). From these Failure State probabilities and effects, the contractor shall determine the overall probability, per flight hour, that one or more 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 values shown in Table 1.

TABLE 1 (3.1.6.1)

LEVELS FOR AIRCRAFT FAILURE STATES

PROBABILITY OF ENCOUNTERING	WITHIN OPERATIONAL FLIGHT ENVELOPE	WITHIN SERVICE FLIGHT ENVELOPE
Level 2 after failure	< ___ per flight hr	
Level 3 after failure	< ___ per flight hr	< ___ per flight hr

Recommended Levels are given in Table 2.

TABLE 2 (3.1.6.1)

RECOMMENDED LEVELS FOR AIRCRAFT FAILURE STATES

PROBABILITY OF ENCOUNTERING	WITHIN OPERATIONAL FLIGHT ENVELOPE	WITHIN SERVICE FLIGHT ENVELOPE
Level 2 after failure	$< 2.5 \times 10^{-3}$ per flight hr	
Level 3 after failure	$< 2.5 \times 10^{-5}$ per flight hr	$< 2.5 \times 10^{-3}$ per flight hr

D. RATIONALE BEHIND REQUIREMENT

The similar MIL-F-8785C (Reference 4) requirement specified failure probabilities as a function of number of flights, rather than flight hours. As discussed in "Supporting Data," a typical flight time of four hours was used for the 8785C numbers. The Table 2 recommended probabilities are now a function of flight hour, simply dividing the 8785C numbers by 4. This assures that the requirements are constant with operational mission time, where in the past the requirements were easier to meet for aircraft with very short operational flight times and harder to meet for aircraft with very long flights.

Limited degradation of flying qualities (e.g., Level 1 to Level 2) is acceptable if the combined probability of such degradation is small. If the probability is high, then no degradation beyond the Level required for Normal States is acceptable after the failure occurs. Another way of stating this is that in the Operational Envelope the probability of encountering Level 2 any time at all on a given four-hour flight should not, for example, according to the Table 2 recommendations, exceed 10^{-2}, and the probability of encountering Level 3 on any portion of the flight should not exceed 10^{-4}. Somewhat reduced requirements should also be imposed for flight within the Service Flight

Envelope, for both Normal and Failure States. Outside the Service Flight Envelope, most of the requirements of the MIL Standard do not apply. There is, however, a qualitative requirement to be able to return to the Service Flight Envelope after a failure (i.e., Para. 3.1.5.3).

E. GUIDANCE FOR APPLICATION

The discussions that follow are taken from Reference 11.

Implementation of the Level concept involves both reliability analyses (to predict failure probabilities) and failure effect analyses (to insure compliance with requirements). Both types of analyses are in direct accord with, and in the spirit of, MIL-STD-756A (reliability prediction) and MIL-S-38130A (safety engineering). These related specifications are, in turn, mandatory for use by all Departments and Agencies of the Department of Defense. Implementation of the flying qualities specification is, for the most part, a union of the work required by these related specifications with normal stability and control analysis.

Failure States influence the aircraft 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. 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.

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.

Note that specific Special Failure States (1.6.3) may be approved; these Failure States need not be considered in determining the probability of encountering degradation to Level 3. This allows each catastrophic failure possibility to be considered on its own. Requiring approval for each Special Failure State gives the procuring activity an opportunity to examine all the pertinent survivability and vulnerability aspects of each design. Survivability and vulnerability are important considerations, but it has not yet been possible to relate any specific flying qualities requirements to them.

A typical approach (but not the only one) for the system contractor is outlined below.

1. Initial Design

The basic airframe is designed for a Level 1 "target" in respect to most flying qualities in the Operational Flight Envelope. 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 altitude); in those 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 Service Flight Envelope. 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 Operational Flight Envelope and to Level 2 in the Service Flight Envelope.

2. Initial Evaluation

The reliability and failure mode analyses are next performed to evaluate the nominal system design evolved above. All aircraft 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 and approval as necessary, or specific values provided by the procuring agency. Prediction methods used will 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 Operational Flight Envelope, Level 3 in the Operational Envelope, and Level 3 in the Service Envelope; and c) recommended airframe/equipment changes to improve flying qualities or increase subsystem reliability to meet the specification requirements. It should be noted that the flying qualities/flight safety requirements are concerned with failure mode effects, while other specifications provide reliability requirements per se (all failures regardless of failure effects). In the event of a conflict, the most stringent requirement should apply.

3. Re-Evaluation

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

The results of the analyses of vehicle flying qualities/flight safety may be used directly to: a) establish flight test points that are critical and should 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.

F. DEMONSTRATION OF COMPLIANCE

The failure modes and effects analysis 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 of the assumptions, such as independence of failure modes, etc., should be firmly established and mutually agreed upon by the contractor and SPO. The combined effects of turbulence and failures should also be considered. It is recommended that the boundaries given in Fig. 1 serve as guidelines for these combined effects.

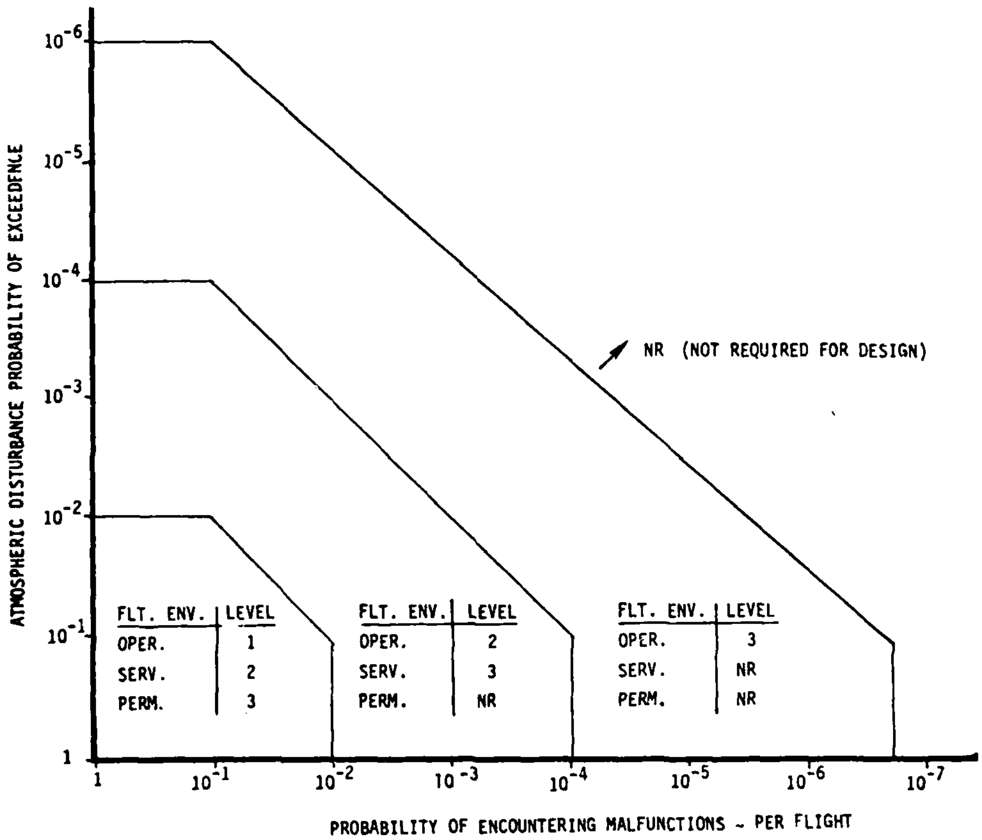


Figure 1 (3.1.6.1). Definition of Levels Which Include Atmospheric Disturbances as well as Failures

The combined effects of failures and turbulence should be validated in a manned simulation. Multiple-axis failures should also be simulated, especially where the flying quality parameters result in pilot ratings near the applicable Level lower limits.

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 final design phases and during flight test, and any data from these or other test programs should 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. Finally, the results of all analyses and tests will be subject to normal procedures of procuring agency approval.

G. SUPPORTING DATA

The numerical values in Table 1 should reflect specific requirements for a given weapon system. The procuring activity engineer should, as a matter of course, confer with both the using command representative and the reliability engineers to assure that the probabilities associated with the Levels are *consistent with the overall design goals.* However, the recommended values of Table 2 are reasonable, based on experience with past aircraft. To illustrate this, the following table presents actual control system failure information for several piloted aircraft:

Reference	System	Mean Time Between Malfunctions (MTBM)
29	F-101B	86 hours
29	F-104	300 hours
29	F-105D (Flight Control plus Electronics)	14 hours
29	E-1B	185 hours
30	B-58	20 hours

Unfortunately the flying qualities effects of the reported failures are not given along with the above data. Reference 31 indicates, however, that the mean time between "critical" failures is about five times the MTBM. If "critical" failures are ones that degrade one or more flying qualities to Level 2, then for a typical average flight time of four hours:

$$P(\text{Level 2}) = \text{Probability of encountering Level 2 flying qualities during a single flight}$$

$$= 1 - e^{-4/[5(MTBM)]}$$

$$\doteq \frac{4}{5(MTBM)}$$

This yields:

System	P(Level 2)
F-101B	0.0093
F-104	0.0027
F-105D	0.057
E-1B	0.0043
B-58	0.040

These data indicate that all systems, with the exceptions of the F-105D (where electronic components represented in the data might not degrade flying qualities upon failure) and the B-58, meet the requirement for P(Level 2) $< 10^{-2}$ (or one out of a hundred flights). Numbers of roughly the same magnitude have been used for both American (Reference 32) and Anglo-French (Reference 33) supersonic transport design.

A more significant analysis was conducted on the F-4 by the former Air Force Flight Dynamics Laboratory (Reference 239). The level of degradation used in this report was based on whether or not the failure resulted in an abort. Failures without abort were considered degraded to Level 2, and those which caused an abort were considered degraded to Level 3. The results showed that the F-4 handling qualities, in an average 2.57 hour flight, will be degraded to Level 2 on an average of 0.043/flight, and to Level 3 a maximum of 0.0021/flight (21×10^{-4}).

A similar comparison can be made between accident loss rates and the requirement for P(Level 3) < 2.5×10^{-5}/flight hour. It should be emphasized that Level 3 as defined in Para. 1.7 and in the requirements represents a safe aircraft to fly. However, due to a lack of knowledge in some instances, especially when many flying qualities are degraded at once, the Level 3 boundaries are, while not necessary totally safe, at least "safety related." Reference 34 indicates the following aircraft accident loss rates during 1967. Also shown is the probability of aircraft loss, per 4-hour flight, for an assumed exponential loss distribution.

Aircraft	1967 Loss Rate (Losses/100,000 Hr)	Probability of Loss During 4-Hour Flight
F-101	15	6×10^{-4}
F-104	23	9.2×10^{-4}
F-105	17	6.8×10^{-4}
F-106	10	4×10^{-4}
F-4	14.1	5.64×10^{-4}
F-102	9	3.6×10^{-4}
F-100	10	4×10^{-4}
	Avg 14	Avg 5.6×10^{-4}

If Level 3 represented a safety problem, which it conservatively does not, then the allowable 10^{-4} probability of encounter per 4-hour flight would account for about 1/4 to 1/9 of the total probability of aircraft loss. That is, flying-qualities-oriented losses would represent about 1/4 to 1/9 of all losses. Other losses could be due to engine failures, etc. Based on experience therefore, the recommended, Table 2 value is reasonable.

As a final note, Reference 35 indicates an Army aircraft accident rate of 22.2/100,000 hours which is very close to the previously cited experience with a number of Air Force aircraft.

H. LESSONS LEARNED

The following excerpts were taken from written comments made by ASD regarding lessons learned utilizing Para. 3.1.10 of MIL-F-8785B.

F-16:	Levels were applied to failures without calculating probabilities; assumed that if a failure could occur, it would eventually (i.e., generic failure analyses)
F-15, F-16, F-105:	Low confidence in failure probability calculations; better to consider individual failures (i.e., generic failure analyses)
B-1/AMST:	See ASD TR-78-13 (Reference 36) for approach to failure states taken on B-1 and AMST. Probability analysis was used. B-1 experience with a 10-hour mission indicated that the probability of encountering Level 3 of 10^{-4}/flight [as required in 8785B] was extremely difficult and was not met.
F-15, A-10:	Very hard to determine realistic probabilities; recommend defining special failure states from past experience (i.e., generic failure analyses)
A-10:	Only specific failures were investigated (i.e., generic failure analyses); most of front section of spec not really used.
F-5E:	Flight outside the Operational Flight Envelope should not be considered abnormal; Para. 3.1.10.2 of MIL-F-8785B deleted as useless.

3.1.6.2 Generic Failure Analysis

A. REASON FOR REQUIREMENT

In accordance with the foregoing "lessons learned," 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.

B. RELATED MIL-F-8785C REQUIREMENT

3.1.10.2.1

C. STATEMENT OF REQUIREMENT

3.1.6.2 Generic Failure Analysis. The allowable Flying Quality Levels for each of the Failure States in Paragraph 1.6.2 are defined as follows: ______________________.

D. RATIONALE BEHIND REQUIREMENT

This approach assumes that a given component, or series of components, will fail. Furthermore, it is assumed the failures will occur in the most critical flight condition; for example, a yaw damper failure at the maximum service ceiling in turbulence. Based on the comments made by users of MIL-F-8785B (see "Lessons Learned" in Para. 3.1.6.1), this approach is a reflection of the way things are sometimes being done.

E. GUIDANCE FOR APPLICATION

The selection of failures to be considered is based on preliminary estimates of handling quality 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 and/or simulation to determine the resulting degradation in flying qualities.

Because the selection of failure modes is highly dependent on the details of the design, close coordination between the contractor and the procuring activity will be required when identifying failure modes to be analyzed. Indeed, this is currently standard practice.

F. DEMONSTRATION OF COMPLIANCE

In most cases, demonstration of compliance will consist of showing that the flying qualities parameters in question fall within the prescribed boundaries for specified Levels. Where such boundaries are not available, either ground-based or in-flight simulation will be required. Failures in more than one axis that cause the specified flying quality parameters to fall near the lower boundaries should also be simulated. Finally, the combined effects of failures and turbulence should be investigated utilizing a piloted simulation.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

This approach has been utilized on the F-15, F-16, A-10, and F-5E with reasonable success.

3.1.7 Dangerous Flight Conditions

3.1.7.1 Warning and indication

3.1.7.2 Devices for indication, warning, prevention, recovery

A. REASON FOR REQUIREMENT

Approach to any dangerous flight condition must be clearly indicated 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.

B. RELATED MIL-F-8785C REQUIREMENTS

3.4.1, 3.4.1.1, 3.4.1.2

C. STATEMENT OF REQUIREMENTS

3.1.7 Dangerous Flight Conditions. Dangerous conditions may exist where the aircraft should not be flown. When approaching these flight conditions, it shall be possible by clearly discernible means for the pilot to recognize the impending dangers and take preventive action.

3.1.7.1 Warning and indication. Warning and indication of approach to a dangerous condition shall be clear and unambiguous. For example, a pilot must be able to distinguish readily among stall warning (which requires pitching down or increasing speed), Mach buffet (which may indicate a need to decrease speed), and normal aircraft vibration (which indicates no need for pilot action).

3.1.7.2 Devices for indication, warning, prevention, recovery. It is intended that dangerous flight conditions be eliminated and the requirements of this specification met by appropriate aerodynamic design and mass distribution, rather than through incorporation of a special device or devices. As a minimum, these devices shall perform their function whenever needed but shall not limit flight within the Operational Flight Envelope. Neither normal nor inadvertent operation of such devices shall create a hazard to the aircraft. For Levels 1 and 2, nuisance operation shall not be possible. Functional failure of the devices shall be indicated to the pilot.

D. RATIONALE BEHIND REQUIREMENTS

Paragraph 3.1.7 (and its subparagraphs) were deemed necessary to insure that the pilot is properly warned when approaching dangerous flight conditions, particularly near the extremes of the flight envelopes. 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.

Paragraph 3.1.7.1 is designed to discourage prevention devices that create more problems than they solve.

E. GUIDANCE FOR APPLICATION

Application of these requirements, in principle, is straightforward; difficulties may arise only when deciding if a flight condition is "dangerous."

F. DEMONSTRATION OF COMPLIANCE

Ground testing will be necessary to assure that functional failure of any warning devices is indicated to the pilot. Ultimately, flight testing will be required (see, for example, MIL-S-83691).

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.1.8 Interpretation of Subjective Requirements

A. REASON FOR REQUIREMENT

This statement is included to clarify the authority of the procuring activity in determining compliance with subjective requirements.

B. RELATED MIL-F-8785C REQUIREMENT

3.1.11

C. STATEMENT OF REQUIREMENT

3.1.8 Interpretation of Subjective Requirements. In several instances throughout the specification subjective terms, such as objectionable flight characteristics, realistic time delay, normal pilot technique and excessive loss of altitude or buildup of speed, have been employed where insufficient information exists to establish absolute quantitative criteria. Final determination of compliance with requirements so worded will be made by the procuring activity.

D. RATIONALE BEHIND REQUIREMENT

While subjective requirements permit wide latitude for the contractor in the early stages, the focus in the flying qualities specifications has been, and will continue to be, on quantifying all requirements for which sufficient data exists. The procuring activity should always have final power in accepting compliance with subjective requirements.

E. GUIDANCE FOR APPLICATION

No discussion is necessary.

F. DEMONSTRATION OF COMPLIANCE

The procuring activity will rely heavily upon comments by evaluation pilots in determining compliance with subjective requirements.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.1.9 Interpretation of Quantitative Requirements

A. REASON FOR REQUIREMENT

Equivalent system approximations to aircraft response characteristics are allowed for comparison with the quantitative requirements.

B. RELATED MIL-F-8785C REQUIREMENT

3.1.12

C. STATEMENT OF REQUIREMENT

3.1.9 Interpretation of Quantitative Requirements. The numerical requirements of this specification generally are stated in terms of a linear mathematical description of the aircraft. Certain factors, for example flight control system nonlinearities and higher-order characteristics or aerodynamic nonlinearities, can cause the aircraft response to differ significantly from that of the linear model. The contractor shall determine equivalent classical systems which have responses most closely matching those of the actual aircraft. Then those numerical requirements of Section 3 which are stated in terms of linear system parameters (such as frequency, damping ratio and modal phase angles) apply to the parameters of that equivalent system rather than to any particular modes of the actual higher-order system. The adequacy of the response match between equivalent and actual aircraft shall be agreed upon by the contractor and the procuring activity.

D. RATIONALE BEHIND REQUIREMENT

This requirement was implemented in MIL-F-8785C in acknowledgment of the increasing complexity of aircraft control systems. The BIUG for that document, Reference 122, discusses the rationale behind this requirement very succinctly, as follows.

In the past, both operational experience and flying qualities research were largely limited to aircraft which behaved in the classical manner: response to control and disturbance inputs characterized by transfer functions of familiar form. The effects of additional dynamics introduced through the flight control system were recognized at the time MIL-F-8785B was written, but limited knowledge prevented adequate treatment. Still, aircraft design developments continue to emphasize equalization to "improve" aircraft response. In Reference 80, Stapleford discusses both good and bad possibilities. Certainly one would expect that failure to consider one or more dynamic modes in the frequency range of pilot control would give erroneous results. Prime examples include the F-14 (Reference 164) and the YF-17 (Reference 165) designs. The F-14's stability augmentation system was designed to increase the low short-period frequency. At one stage it did that well in landing approach, but it also introduced higher-order dynamics which resulted in an overall "effective short-period frequency" little changed from augmentation-off. In a flight evaluation of predicted YF-17 characteristics using the AFWAL-Calspan NT-33 Variable Stability Airplane, pilots rated the short-period response poor to bad. (The equivalent system approach may not have been used to improve the response.) However, it is pertinent that a configuration intended to have good flying qualities got "good" pilot ratings in flight only after the flight control system compensation had been simplified.

Boothe, et al. (Reference 160) suggest several simple mechanizations that augment stability without increasing the order of the system response. However, prefilters, forward-loop compensation, crossfeeds, etc., are legitimate design tools that are being used on many current aircraft and indeed seem to be the norm. These artifacts do increase system order and we need to be able to account for their effects in the requirements. Thus, with modern flight control and stability augmentation systems, there is considerable confusion regarding the "proper" selection of modal parameters such as short-period frequency and damping. Correlation of Level 1 flying qualities with characteristics of the bare airframe is certainly not valid for augmented aircraft in general. Stability and control augmentation frequently introduce additional dynamics *in the frequency range of pilot control*, thereby invalidating any interpretation of the requirements in terms of particular roots of a transfer function. Although these fallacies have been pointed out many times, misinterpretations continue. The feeling is not uncommon that some requirements just do not apply. To clarify application of the requirements to flying qualities in general this paragraph, 3.1.9, has been added.

In reality, we are only interested in pilots' opinions as to whether the actual aircraft dynamics enable the appropriate tasks to be performed well enough with acceptable workload. We now require, therefore, that the active dynamics be approximated by the responses of transfer functions of classical form. The appropriate parameters of this equivalent transfer function must meet the modal requirements of the specification. This so-called "equivalent system" approach allows continued use of the familiar test data base for a broad range of mechanizations. It has been advocated strongly by Hodgkinson and others (References 80, 81, 85, 86).

F. GUIDANCE FOR APPLICATION

The preceding discussion should not be taken to imply that there is little problem with applying the specification requirements to equivalent system parameters. For configurations which exhibit conventional-appearing dynamics, application is indeed straightforward. It also appears to be true at present that pilots are most comfortable with response dynamics that are "natural," i.e., like the classical modes. Certainly, additional prominent modes result in a more complicated dynamic response. As we consider configurations with dynamics that depart more and more from the classical order or form, then more and more judgment will be required in defining the appropriate equivalent system parameters and assessing compliance with the requirements. Hodgkinson has suggested that flying qualities will be poor if no equivalent system can be found to give a "good" fit to the actual response. Success of the equivalent system approach in applying or defining the Level 2 and 3 boundaries is not definite at this time. There are also questions which remain to be answered. Is the equivalent system solution unique? (Not universally, it seems.) Can the equivalent system parameters be juggled until compliance is indicated? (In limited observations, some tendency toward equivalent results from different techniques has been noted.) Are requirements necessary for either the amount or the quality of the mismatch? (To date this has not been a major problem.) In spite of the qualifying remarks and the above questions, this approach is a way to apply known requirements to advanced configurations with high-order dynamic responses. We preserve the validated data base of MIL-F-8785B and the experience in its use. At the same time the equivalent systems are to be defined by matching an appropriate aircraft response to pilot control input. We therefore focus attention on the quality of the actual overall response perceived by the pilot, rather than imply consideration of a dominant mode as may be inferred (however incorrectly) from MIL-F-8785B. We also believe that the use of the equivalent system approach is responsive to the needs of designers. Failure of an equivalent system parameter to meet the requirement then indicates the characteristic of the system (e.g., damping, delay or lag, etc.) that must be improved. We acknowledge that the use of equivalent systems is not a magic solution to good flying qualities; however, properly used it is a good tool for designing or evaluating advanced configurations which are becoming indiscriminately complex.

F. DEMONSTRATION OF COMPLIANCE

In order to demonstrate compliance with the modal requirements of MIL-F-8785C, equivalent systems must first be defined to approximate the actual aircraft dynamics whether predicted analytically or obtained from flight test. Considerations for specific axes are discussed elsewhere following the appropriate requirement. In general, however, it is necessary to add a term representing a time delay to the "classical form" of the transfer functions. This term allows a closer match of the higher-frequency content of most advanced systems considered to date. The time delay has been correlated with pilot opinion ratings and has yielded new requirements.

The requirement as written is intended to allow the contractor to use any reasonable method of determining the equivalent aircraft systems. However, the procuring activity may require some other method for final compliance demonstration.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.1.10 Quality Assurance

3.1.10.1 Compliance demonstration

A. REASON FOR REQUIREMENTS

These requirements are included to indicate acceptable methods for demonstrating compliance with the handling qualities criteria in the MIL Standard.

B. RELATED MIL-F-8785C REQUIREMENTS

4.1, 4.1.2, 4.1.3

C. STATEMENT OF REQUIREMENTS

3.1.10.1 Compliance demonstration. Compliance with the quantitative requirements of Section 3 shall be demonstrated through analysis. In addition, compliance with many of the requirements will be demonstrated by simulation, flight test, or both. The methods for demonstrating compliance shall be established by agreement between the procuring activity and the contractor. Representative flight conditions, configurations, external store complements, loadings, etc., shall be determined for detailed investigations in order to restrict the number of design and test conditions. The selected design points must be sufficient to allow accurate extrapolation to the other conditions at which the requirements apply.

a) Analysis. The analytical methods, procedures, assumptions, etc., applied shall be made available to the procuring activity. In some instances (e.g., control power) compliance may be demonstrated partially or wholly by analysis when the analytical model is validated with flight test data and approved by the procuring activity. In other instances (e.g., control in turbulence) analysis will provide information on specific test conditions requiring simulation, flight test, or both.

b) Simulation. The danger, extent or difficulty of flight testing may dictate simulation rather than flight test to evaluate some conditions and events, such as the influence of Severe disturbances, events close to the ground (except 3.2.8.4 shall be demonstrated in flight), combined Failure States and disturbances, etc. In addition, by agreement with the procuring activity, piloted simulation shall be performed before first flight of a new aircraft design in order to demonstrate the suitability of the handling qualities, and also to demonstrate compliance with qualitative requirements in atmospheric disturbances. Where simulation is the ultimate method of demonstrating compliance for a requirement, the simulation model shall be validated with flight test data.

c) Flight test. The required flight tests will be defined by operational, technical, and safety considerations as decided jointly by the procuring activity, the test agency, the contractor, and other involved agencies using results from 3.1.10.1a and 3.1.10.1b. It is expected that flight test demonstration of the requirements in calm air and selected requirements in at least Moderate turbulence will be accomplished to insure that flying quality degradations are not excessive.

D. RATIONALE BEHIND REQUIREMENTS

An attempt has been made to provide direction as to the analyses, simulations, and fight tests that should normally be done. In addition, since it is required to demonstrate compliance with all the quantitative requirements analytically at some stage of the design, explicit direction is also given as to which items will not normally apply to flight test. It is expected that the methods of determining compliance for any particular aircraft will be defined by the procuring activity.

The paragraph on simulation has been included to provide some guidance on the use of piloted simulation for compliance demonstration rather than engineering development. Specifically, piloted simulation is required before first flight of a new design. Reference 165, for instance, documents the benefits of in-flight simulation before first flight. In addition, it is suggested that piloted simulation could be the primary means of demonstrating compliance with certain requirements.

The last sentence of 3.1.10.1(c) is intended to indicate that in a normal flight test program considerable effort should be expended to insure that some encounters with real atmospheric disturbances are accomplished. A chase aircraft, parameter identification or just the available weather information (with some assumptions) would be used to give estimates of probable disturbance intensities. These encounters may then afford the opportunity to check the qualitative requirements informally.

E. GUIDANCE FOR APPLICATION

The procuring activity, the test agency, and the contractor will jointly agree on which tests are hazardous and what analyses, simulations, and buldup maneuvers are needed to assure safety.

In order to call for flight testing in Severe disturbances, the procuring activity and other involved agencies should specify at the outset of a design what is expected. For example, design for a terrain-following mission might require acceptable ride and flying qualities over specified terrain in specified turbulence. While neither that terrain nor that turbulence may be encountered in flight test, the actual terrain may be known or measured and the actual turbulence deduced from flight records by parameter identification techniques or more subjectively from Table 3 (4.3). That would furnish direct evidence of acceptability, while serving to validate the analysis and simulation that use the specified turbulence.

At the 1978 AFFDL flying qualities workshop (Reference 163), anxiety was expressed over requirements for which flight testing to demonstrate compliance would be extremely difficult or time-consuming. Requirements related to atmospheric disturbances were of particular concern. However, neither past practice, nor present procedures, nor foreseeable future demands show such difficulty. Flight testing has always been a most pragmatic occupation. That certainly holds with flying qualities. The following discussion attempts to show what reasonably can be expected.

Currently flight test costs are up, flying hours are down, and emphasis has shifted from engineering evaluation to investigation of conditions approximating operational use. In this climate we must seek optimized flight test techniques to extract the greatest quantity of most-needed flying qualities data in the available flight test time. There is no hope of a flight handling evaluation of the type and scope of AFFTC's Phase IV evaluations of former years. This change is not all bad.

To a large extent the traditional techniques are being supplanted by parameter identification from dynamic flight records. As AFFTC has shown, using appropriate control inputs, data can be accumulated quickly over a large flight envelope for reduction by computer to transfer functions or stability derivatives. Twisdale (Reference 112) describes a means of extracting such data from air combat tracking related to the manner in which fighter aircraft are intended to be used. From accurate, well-documented results the aircraft designer's stability and control predictions can be corrected to obtain a validated analytical model. Thoroughness of documentation is as critically essential as accuracy. Where those flight tests do not themselves generate the values of many motion parameters needed to determine compliance, an engineer can use the validated model to investigate any aspect of specification compliance at will. With this procedure, however, there are many more chances for error along the way. For meaningful results a good deal of coordination is necessary among all those involved in design, testing, evaluation, and procurement.

F. DEMONSTRATION OF COMPLIANCE

Experience has shown that it is sometimes possible to have a Level 1 aircraft that does not meet certain quantitative requirements. This in fact is an expected occurrence for certain criteria where the supporting data are sparse or nonexistent (for example, stick force gradients with normal acceleration or displacement). In such cases the manufacturers should request an exception. Allowance of such exceptions should be made on the basis of demonstration by flight test or piloted simulation. The scenarios utilized to conduct such demonstrations should involve the actual piloting tasks for the appropriate missions and mission phases. Additionally, the critical disturbance environment should be defined. For example, if attitude response characteristics in landing are in question, the task should involve precision touchdowns from an offset maneuver to insure that the pilots are forced to close a tight attitude loop. Such aggressive tracking has been shown to expose handling quality deficiencies that are completely unrecognizable for nonprecision tasks. Also, precision landings should be required in moderate turbulence. If a simulation is used, the critical wind shears defined in

Paragraph 3.9 should be employed. A minimum of three evaluation pilots is required to establish the level of flying qualities. Where pilot ratings are taken it is recommended that strict adherence to the Cooper-Harper scale be maintained and that all ratings be justified by supporting commentary.

G. SUPPORTING DATA

Not applicable.

H. LESSONS LEARNED

Controversy over the level of handling qualities of an aircraft for a specific task is generally a result of poorly structured test procedures or informal evaluation techniques. An example is the F-16 in the landing configuration. Many pilot ratings and commentaries, particularly from pilots experienced in the aircraft, indicate that the F-16 is Level 1 for landing. Full Scale Development Test and Evaluation at AFFTC, on the other hand, indicated that "F-16 flying qualities in the PA configuration should be improved to reduce pilot workload during the landing task" (Reference 125). The aircraft has Level 2 values of time delay (see Paragraph 3.2.1.1) and marginal equivalent values of short-period frequency in the landing configuration. Full Scale Development F-16s also employed an isometric sidestick which was in the Level 2/3 region of the test data shown in Paragraph 3.2.9.4. Upon closer investigation one finds that the Level 1 ratings apply to landing in non-critical conditions. When queried, most F-16 pilots agree that the aircraft must be landed basically open loop for the last 50 ft. Attempts to make last minute corrections frequently result in pilot-induced oscillations or pitch bobbling as a minimum. The lesson to be learned is that compliance by demonstration requires a very carefully planned experiment. It is of utmost importance that the task be well defined and that the pilots be forced into an aggressive tracking situation. In general, the task must be as demanding as any likely to be encountered in operational use. Atmospheric disturbances should be included as a marginal Level 1 aircraft will often degrade to a solid Level 2 in moderate turbulence.

3.1.10.2 Design and test conditions

A. REASON FOR REQUIREMENT

In order to facilitate analysis, design, and testing, a comprehensive list of critical flight conditions is presented.

B. RELATED MIL-F-8785C REQUIREMENTS

4.1, 4.2.1, 4.2.2, 4.3.1, 4.3.2

C. STATEMENT OF REQUIREMENT

3.1.10.2 Design and test conditions. Table 1 specifies general guidelines, but the peculiarities of the specific aircraft design may require additional or alternate test conditions.

a) Terms specified in Table 1 such as "heaviest weight" and "greatest moment of inertia" mean the heaviest and greatest consistent with 3.1.1 and 3.1.2. When a critical center-of-gravity position is identified, the aircraft weight and associated moments of inertia shall correspond to the most adverse service loading in which that critical center-of-gravity position is obtained.

b) Terms specified in Table 1 such as "most forward c.g." and "most aft c.g." mean the most forward or most aft consistent with 3.1.1. When a critical weight or moment of inertia is identified, the center-of-gravity position shall correspond to the most adverse service loading in which that critical weight or moment of inertia is obtained.

c) For terminal Flight Phases, it will normally suffice to examine the selected Aircraft States at only one altitude below 10,000 feet (low altitude). For nonterminal Flight Phases, it will normally suffice to examine the selected Aircraft States at one altitude below 10,000 feet or at the lowest operational altitude (low altitude), the maximum operational altitude ($h_{o_{max}}$), and one intermediate altitude. When the maximum operational altitude is above 40,000 feet or when stability or control characteristics vary rapidly with altitude, more intermediate altitudes than specified in Table 1 shall be investigated. When the Service Flight Envelope extends far above or below the Operational Flight Envelope, the service altitude extremes must be considered.

TABLE 1 (3.1.10.2) DESIGN AND TEST CONDITION GUIDELINES

REQUIREMENT NUMBER	TITLE	CRITICAL LOADING	LOAD FACTOR	ALTITUDE	SPEED	FLIGHT PHASE
Section 3.2	HANDLING QUALITY REQUIREMENTS FOR PITCH AXIS					
3.2.1	Response to Pitch Controller	Most forward c.g. † and most aft c.g. §	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	*,CR,RT,PA L,CT
3.2.2	Pilot-Induced Oscillations	--	Minimum permissible to maximum permissible	↓	↓	↓
3.2.3	Residual Oscillations	↓	1.0	↓	$V_{o_{min}}$ to $V_{o_{max}}$	*,PA
3.2.7.2	Response to failures	↓	All	$h_{o_{min}}$ and $h_{o_{max}}$	V_{min} to V_{max}	--
3.2.7.3	Response to configuration or control mode change	↓	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	↓	↓
3.2.7.4	Response to stores release	↓	$n_o(-)$ to $n_o(+)$	↓	$V_{o_{min}}$ to $V_{o_{max}}$	CO,GA,WD, AD
3.2.7.5	Response to armament delivery	↓	↓	↓	↓	*,RT
3.2.7.6	Response to buffet	↓	↓	↓	↓	*
3.2.8.1	Control power in unaccelerated flight	Most forward c.g.	1.0	↓	V_{min} to V_{max}	--
3.2.8.2	Control power in maneuvering flight	Most forward c.g. †	As required	↓	$V_{o_{min}}$ to $V_{o_{max}}$	CO,GA,AR, TF,CR,PA
3.2.8.3	Control power for takeoff	Most forward c.g. (nose-wheel), most aft c.g. (tail-wheel aircraft)	1.0	Low	As required	TO
3.2.8.4	Control power for landing	Most forward c.g.	1.0	Low	$V_S(L)$ or geometric limit	L
3.2.8.5	Control power for other conditions	--	All	$h_{o_{min}}$, medium, $h_{o_{max}}$	All	--
3.2.9.1	Steady-state control force per g	Most forward c.g. † and most aft c.g. §	n(-) to n(+)	↓	V_{min} to V_{max}	*,RT,CR,PA L,CT
3.2.9.2	Transient control force per g	Most aft c.g. §	1.0	↓		--
3.2.9.3	Control force variations during rapid speed changes	--	As required	↓	$V_{o_{min}}$ to $V_{o_{max}}$ and transonic	CO,GA,DE
3.2.9.4.1	Control force vs. deflection -- steady-state gradient	Most forward c.g. †	$n_o(-)$ to $n_o(+)$	↓	V_{min} to V_{max}	*,RT,CR,PA, L,CT
3.2.9.4.2	Transient control force vs. deflection	Most aft c.g. §	1.0	↓	↓	--
3.2.9.5	Control centering and breakout forces	--	$n_o(-)$ to $n_o(+)$	$h_{o_{min}}$ and $h_{o_{max}}$	↓	↓
3.2.9.6	Free play	↓	↓	↓	↓	↓
3.2.9.7.1	Force limits -- takeoff	Most forward c.g. and most aft c.g.	As required	Low	0 to V_{max}(TO)	TO,CT

† Combined with heaviest weight. § Combined with lightest weight.

* *All applicable Category A Flight Phases.* -- *No general guidance can be provided.*

"As required" -- flight conditions are specified in requirement or are determined by nature of test maneuver.

TABLE 1 (3.1.10.2). (Continued)

REQUIREMENT NUMBER	TITLE	CRITICAL LOADING	LOAD FACTOR	ALTITUDE	SPEED	FLIGHT PHASE
3.2.9.7.2	Force limits -- landing	Most forward c.g.	1.0	Low	$V_S(L)$ or geometric limit	L
3.2.9.7.3	Force limits -- dives: SFE	Most forward c.g. † and most aft c.g. §	As required	2000 ft MSL to h_{max}	V_{min} to V_{max}	D,ED,CO, CR,GA
	PFE	↓	↓	As required	V_{MAT} to maximum permissible	↓
3.2.9.7.4	Force limits -- sideslips	--	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	CO,CR,PA,L
3.2.9.7.6	Force limits -- failures	--	All	$h_{o_{min}}$ and $h_{o_{max}}$	V_{min} to V_{max}	--
3.2.9.7.7	Force limits -- configuration or control mode charge	↓	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	↓	↓
3.2.9.8	Trim systems	Most forward c.g. and most aft c.g.	↓	↓	↓	↓
3.2.9.8.1	Trim systems -- rate of operation	--	↓	As required	As required	D,ED,CO, GA
3.2.9.8.2	Trim systems -- stalling of trim systems	Most forward c.g. †	As required	↓	Start of dive recovery to V_{max}	D,ED,CO, CR
3.2.9.8.3	Trim systems -- irreversibility	--	1.0	MSL to h_{max}	V_{min} to V_{max}	--
3.2.10.1	Control displacements -- takeoff	Most forward c.g. and most aft c.g.	As required	Low	0 to $V_{max}(TO)$	TO,CT
3.2.10.2	Control displacements -- maneuvering	↓	$n(-)$ to $n(+)$	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	*,RT,CR,PA, L,CT
3.2.10.3	Control displacements -- gust regulation	--	$n_o(-)$ to $n_o(+)$	$h_{o_{min}}$ and $h_{o_{max}}$	↓	--
SECTION 3.3	HANDLING QUALITY REQUIREMENTS FOR VERTICAL FLIGHT PATH AXIS					
3.3.1.2.1	Response to attitude change -- steady-state response	--	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	$V_{o_{min}}$ and $V_{o_{min}}$ - 5 kt	PA
SECTION 3.4	HANDLING QUALITY REQUIREMENTS FOR LONGITUDINAL AXIS					
3.4.1	Response to Attitude Changes	Most aft c.g.	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	CO,RR,FF,CR LO,RT,All Category C
3.4.1.1	Relaxation in transonic flight	↓	↓	↓	Transonic	CO
SECTION 3.5	HANDLING QUALITY REQUIREMENTS FOR ROLL AXIS					
3.5.1	Roll Response to Roll Controller	--	1.0 and $n_o(+)$	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	*,CL,CR,LO, RT,DE,PA,L

† Combined with heaviest weight. § Combined with lightest weight.

* All applicable Category A Flight Phases. -- No general guidance can be provided.

"As required" -- flight conditions are specified in requirement or are determined by nature of test maneuver.

TABLE 1 (3.1.10.2). (Continued)

REQUIREMENT NUMBER	TITLE	CRITICAL LOADING	LOAD FACTOR	ALTITUDE	SPEED	FLIGHT PHASE
3.5.2	Pilot-Induced Oscillations		Minimum permissible to maximum permissible	MSL to h_{max}	V_{min} to V_{max}	--
3.5.4	Linearity of Roll Response to Roll Controller	Greatest rolling moment of inertia	As required (not above $0.8n_L$)	$h_{o_{min}}$, medium, $h_{o_{max}}$		CO,GA,TF,CL,CR,TO,CT
3.5.6	Roll Response to Yaw Controller	Lightest weight	1.0			CO,CR,PA,L
3.5.7	Roll Control for Takeoff and Landing in Crosswinds	--		Low	As required	Taxi,TO,L
3.5.8.1	Response to asymmetric thrust	Lightest weight	1.0	All	V_{min} to V_{max}	CO,GA,TF,CL,CR,TO,CT
3.5.8.2	Response to failures	--	All	$h_{o_{min}}$ and $h_{o_{max}}$		--
3.5.8.3	Response to configuration or control mode change	--	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	--
3.5.8.4	Response to stores release		$n_o(-)$ to $n_o(+)$		$V_{o_{min}}$ to $V_{o_{max}}$	CO,GA,WD,AD
3.5.8.5	Response to armament delivery					*,RT
3.5.9.1	Control power - response to roll control inputs	Greatest and smallest rolling moments of inertia	As required (not above $0.8\ n_L$)	$h_{o_{min}}$, medium, $h_{o_{max}}$		As required
3.5.9.2	Control power - steady sideslips	Lightest weight	1.0			CO,CR,PA,L
3.5.9.3	Control power - cross-winds	--	As required	Low	As required	TO,L,PA
3.5.9.4	Control power - engine failures	Lightest weight	1.0	$h_{o_{min}}$	Down to $V_{min}(TO)$	TO,CT
3.5.9.5	Control power - dives and pullouts	--	As required	2000 ft MSL to h_{max}	V_{MAT} to V_{max}	D,ED
3.5.9.6	Control power - stores release	--	$n_o(-)$ to $n_o(+)$	$h_{o_{min}}$, medium, $h_{o_{max}}$	$V_{o_{min}}$ to $V_{o_{max}}$	CO,GA,WD,AD
3.5.9.7	Control power - two engines inoperative	Lightest weight	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{range} (1 and 2 engines out)	--
3.5.9.8	Control power for other conditions	--	All		All	--
3.5.10.1	Wheel control displacements	Greatest rolling moment of inertia	As required (not above $0.8n_L$)		V_{min} to V_{max}	CO,GA,AR,TF,CR,GA,L
3.5.10.2	Forces to achieve required roll rates	Greatest and smallest rolling moments of inertia				
3.5.10.3	Sensitivity	Smallest rolling moment of inertia				
3.5.10.4	Breakout and centering forces	--	$n_o(-)$ to $n_o(+)$	$h_{o_{min}}$ and $h_{o_{max}}$	V_{min} to V_{max}	--

† Combined with heaviest weight.

§ Combined with lightest weight.

* All applicable Category A Flight Phases.

-- No general guidance can be provided.

"As required" -- flight conditions are specified in requirement or are determined by nature of test maneuver.

TABLE 1 (3.1.10.2). (Continued)

REQUIREMENT NUMBER	TITLE	CRITICAL LOADING	LOAD FACTOR	ALTITUDE	SPEED	FLIGHT PHASE
3.5.10.5	Free play	--	$n_o(-)$ to $n_o(+)$	$h_{o_{min}}$ and $h_{o_{max}}$	V_{min} to V_{max}	--
3.5.10.6.1	Force limits - steady turns	↓	As required	$h_{o_{min}}$, medium, $h_{o_{max}}$	$V_{o_{min}}$	CO,CR,LO,PA
3.5.10.6.2	Force limits - dives and pullouts	↓	↓	2000 ft MSL to h_{max}	V_{MAT} to V_{max}	D,ED
3.5.10.6.3	Force limits - crosswinds	↓	1.0	Low	As required	TO,L
3.5.10.6.4	Force limits - steady sideslips	↓	↓	↓	V_{min} to V_{max}	↓
3.5.10.6.5	Force limits - engine failures after takeoff	Lightest weight	↓	$h_{o_{min}}$, medium, $h_{o_{max}}$	$V_{min}(TO)$ to $1.4V_{min}$	CR,TO,CT
<u>SECTION 3.6</u>	HANDLING QUALITY REQUIREMENTS FOR YAW AXIS					
3.6.1.1.1	Equivalent systems requirement - transient response	Greatest rolling moment of inertia	1.0 and $n_o(+)$	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	*,CR,RT,PA,L
3.6.1.1.2	Equivalent systems requirement - steady-state response	Lightest weight	1.0	↓	↓	CO,CR,PA,L
3.6.2.1	Yaw response to roll controller - coordination in turn entry and exit	Greatest yawing and rolling moments of inertia	↓	↓	↓	*,CR,PA,L
3.6.2.2	Pilot-induced oscillations	--	Minimum permissible to maximum permissible	MSL to h_{max}	↓	--
3.6.3	Yaw Control for Takeoff and Landing in Crosswinds	--	1.0	Low	As required	TO,L,Taxi
3.6.4.1	Response to asymmetric thrust	Lightest weight	1.0	$h_{o_{min}}$	0 to $V_{max}(TO)$	TO,CT
		↓	↓	All	V_{min} to V_{max}	CO,GA,TF,CR CL,TO,CT
		↓	↓	$h_{o_{min}}$, medium, $h_{o_{max}}$	$1.4V_{min}$	CR
3.6.4.2	Response to failures	--	All	↓	V_{min} to V_{max}	--
3.6.4.3	Response to configuration or control mode change	--	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	--
3.6.4.4	Response to stores release	↓	$n_o(-)$ to $n_o(+)$	↓	$V_{o_{min}}$ to $V_{o_{max}}$	CO,GA,WD,AD
3.6.4.5	Response to armament delivery	--	↓	↓	↓	*,RT
3.6.5.1	Control power - takeoff, landing, and taxi	--	As required	Low	0 to $V_{max}(TO)$	PA,TO,L,Taxi
3.6.5.2	Control power - two engines inoperative	Lightest weight	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{range} (1 and 2 engines out)	--

* All applicable Category A Flight Phases. -- No general guidance can be provided.

"As required" -- flight conditions are specified in requirement or are determined by nature of test maneuver.

TABLE 1 (3.1.10.2). (Concluded)

REQUIREMENT NUMBER	TITLE	CRITICAL LOADING	LOAD FACTOR	ALTITUDE	SPEED	FLIGHT PHASE
3.6.5.3	Control power - asymmetric loading	--	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	CO,GA,CR,D, PA,L
3.6.6	Yaw Axis Control Forces	↓	$n_o(-)$ to $n_o(+)$	↓	↓	*,CR,PA,L
3.6.6.1	Force linearity	Lightest weight	1.0	↓	↓	CO,CR,PA,L
3.6.6.2.1	Force limits - rolling maneuvers	Greatest rolling moment of inertia	As required	↓	↓	CO,GA,AR,TF, CR,PA,L
3.6.6.2.2	Force limits - steady turns	--	↓	↓	$V_{o_{min}}$	CO,CR,LO,PA
3.6.6.2.3	Force limits - speed changes	↓	1.0	↓	V_{min} to V_{max}	CO,GA,CR,D, PA,L
3.6.6.2.4	Force limits - cross-winds	--	1.0	Low	As required	TO,L
3.6.6.2.5	Force limits - asymmetric loading	↓	↓	$h_{o_{min}}$, medium, $h_{o_{max}}$	$V_{o_{min}}$ to $V_{o_{max}}$	--
3.6.6.2.6	Force limits - dives and pullouts	↓	As required	2000 ft MSL to h_{max}	V_{MAT} to V_{max}	D,ED
3.6.6.2.7	Force limits - go-arounds	Lightest weight	1.0	Low	V_{min}(PA) or landing speed	WO
3.6.6.2.8	Force limits-asymmetric thrust	↓	↓	$h_{o_{min}}$	0 to V_{max}(TO)	TO,CT
3.6.6.2.9	Force limits - failures	--	All	$h_{o_{min}}$ and $h_{o_{max}}$	V_{min} to V_{max}	--
3.6.6.2.10	Force limits - configuration or control mode changes	↓	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	↓	↓
<u>SECTION 3.8</u>	HANDLING QUALITY REQUIREMENTS FOR COMBINED AXES					
3.8.1	Cross-Axis Coupling in Roll Maneuvers	--	0 to $0.8n_L$	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	CO,GA,AR,TF
3.8.2	Crosstalk Between Pitch and Roll Controllers	↓	$n_o(-)$ to $n_o(+)$	↓	↓	--
3.8.3	Control Harmony	See MIL-S-83691 or MIL-D-8708, whichever is applicable for flight demonstration. More severe conditions generally will be investigated by analysis and model testing.				
3.8.4	Flight at High Angle of Attack	↓	↓	↓	↓	↓

* All applicable Category A Flight Phases. -- No general guidance can be provided.

"As required" flight conditions are specified in requirement or are determined by nature of test maneuver.

d) In addition to the flight conditions indicated in Table 1, speed-altitude combinations that result in the following shall all be investigated, where applicable:

- Maximum normal acceleration response per degree of controller deflection.
- Maximum normal acceleration response per pound of control force.
- Highest dynamic pressure and highest Mach number.

D. RATIONALE BEHIND REQUIREMENT

Table 1 was presented in MIL-F-8785C as a guideline for design and testing. The format has been revised to be consistent with the current Standard.

E. GUIDANCE FOR APPLICATION

The procuring activity should, in accordance with the compliance demonstration requirements of 3.1.10.1, define those tests and conditions necessary for analysis, simulation, and flight test.

F. DEMONSTRATION OF COMPLIANCE

The contractor will be expected to investigate all conditions specifically on contract. In addition, areas that might be considered critical but that were not specified should be included. Exceptions (e.g., cases where the contractor feels flight testing is not necessary) must be well supported by valid analysis.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.2 HANDLING QUALITY REQUIREMENTS FOR PITCH AXIS

3.2.1 Pitch Attitude Response to Pitch Controller

DISCUSSION

Two alternative criteria are presented for this requirement.

The first criterion (Para. 3.2.1.1) consists of the basic boundaries previously defined in MIL-F-8785C for classical unaugmented airplanes. The criterion is extended to include augmented airplanes via the use of lower-order equivalent systems. An optional set of boundaries is presented to allow the use of $1/T_{\theta_2}$ in lieu of n/α if desired.

The second criterion (Para. 3.2.1.2) is oriented toward highly augmented aircraft as well as aircraft with unconventional flight modes generated with direct force controls (DFC). This criterion is based on the pilot's ability to do tight closed-loop tracking and is termed the bandwidth criterion. It is recommended that this criterion be utilized when the mismatch between the lower-order equivalent and higher-order systems is large, or when the pitch axis augmentation results in non-classical responses, e.g., for attitude command systems. Guidelines for deciding when this is a problem are given in "Rationale Behind Requirement" under 3.2.1.1.

3.2.1.1 Pitch axis lower-order equivalent systems requirements

A. REASON FOR THIS REQUIREMENT

Pitch control of conventional aircraft 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 in landing).

The use of lower-order equivalent systems allows an interpretation of the responses of the most complex vehicles in familiar terms utilized to characterize the response of classical aircraft. This requirement incorporates many of the MIL-F-8785C phugoid and short-period mode parameters and sets detailed guidelines on application of equivalent systems parameters.

B. RELATED MIL-F-8785C REQUIREMENTS

3.1.12, 3.2.1.2, 3.2.2.1, 3.2.2.1.1, 3.2.2.1.2, 3.5.3.

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.1.1 Pitch axis lower-order equivalent systems requirements. The equivalent parameters describing the responses of pitch rate and normal load factor (at the center of rotation) to a pitch control force input shall have the following characteristics: ________________.

Recommended Values

1. Short term response - For the equivalent pitch rate and normal load factor transfer function defined below, recommended requirements on the equivalent parameters are given in the following table.

$$\frac{\dot{\theta}}{F_s} = \frac{K_\theta[s + (1/T_{\theta_2})]e^{-\tau_{e_\theta} s}}{s^2 + 2\zeta_{sp}\omega_{sp}s + \omega_{sp}^2}$$

$$\frac{n_z'}{F_s} = \frac{K_{n_z}e^{-\tau_{e_n} s}}{s^2 + 2\zeta_{sp}\omega_{sp}s + \omega_{sp}^2}$$

Notes:

a) The equivalent systems are to be obtained from simultaneous matching of the $\dot{\theta}/F_s$ and n_z/F_s higher order system responses over a frequency range of approximately 0.1 to 10 rad/sec.

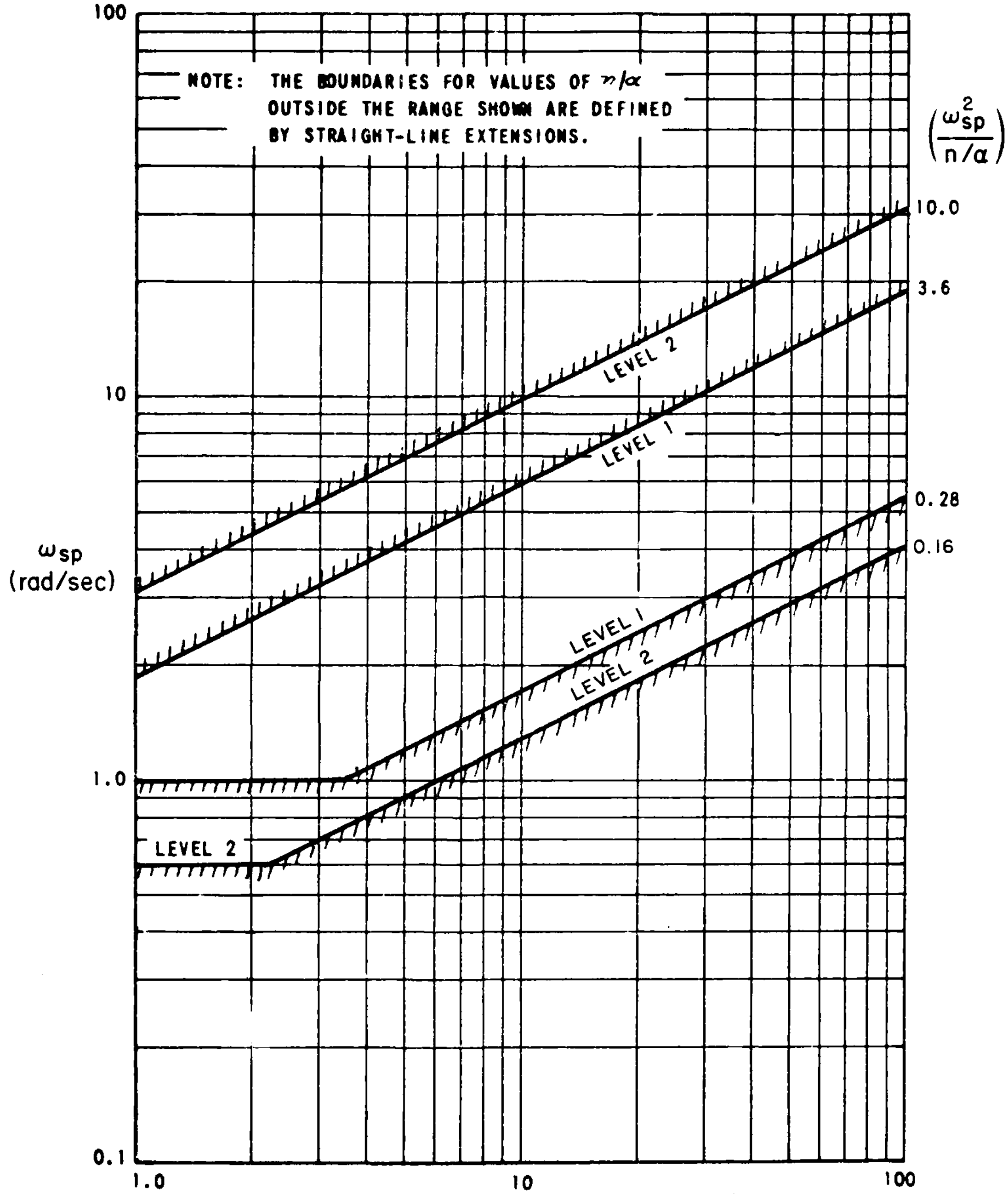
b) n_z' is normal acceleration as measured at the aircraft center of rotation.

2. Long term response

$$\frac{\dot{\theta}}{F_s} = \frac{K_\theta[s + (1/T_{\theta_1})]}{s^2 + 2\zeta_p\omega_p s + \omega_p^2}$$

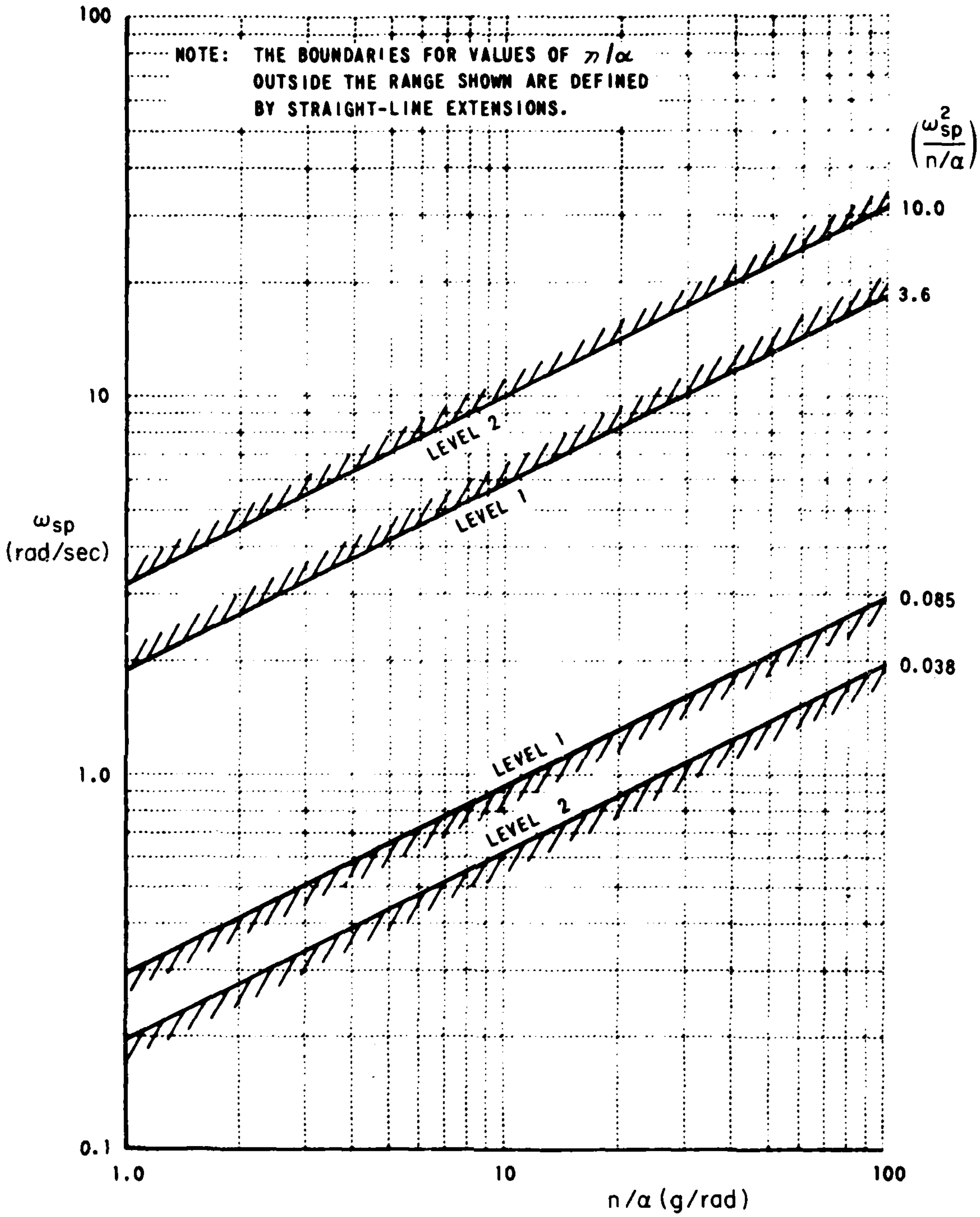
Note: While a lower order equivalent system match could be used, the parameter ζ_p is the only one specified and it can be generally calculated directly from a time response.

PARAMETER	ω_{sp} vs. n/α	$\omega_{sp}T_{\theta_2}$ vs ζ_{sp}
ζ_p	Table 1	Table 1
ω_p	No Requirement	No Requirement
ζ_{sp}	Table 2	Figure 2
ω_{sp}	Figure 1	Figure 2
n/α	Figure 1	—
$1/T_{\theta_1}$	At least greater than Zero	At least greater than Zero
$1/T_{\theta_2}$	—	Figure 2
τ_{e_θ}, τ_{e_n}	Table 3	Table 3



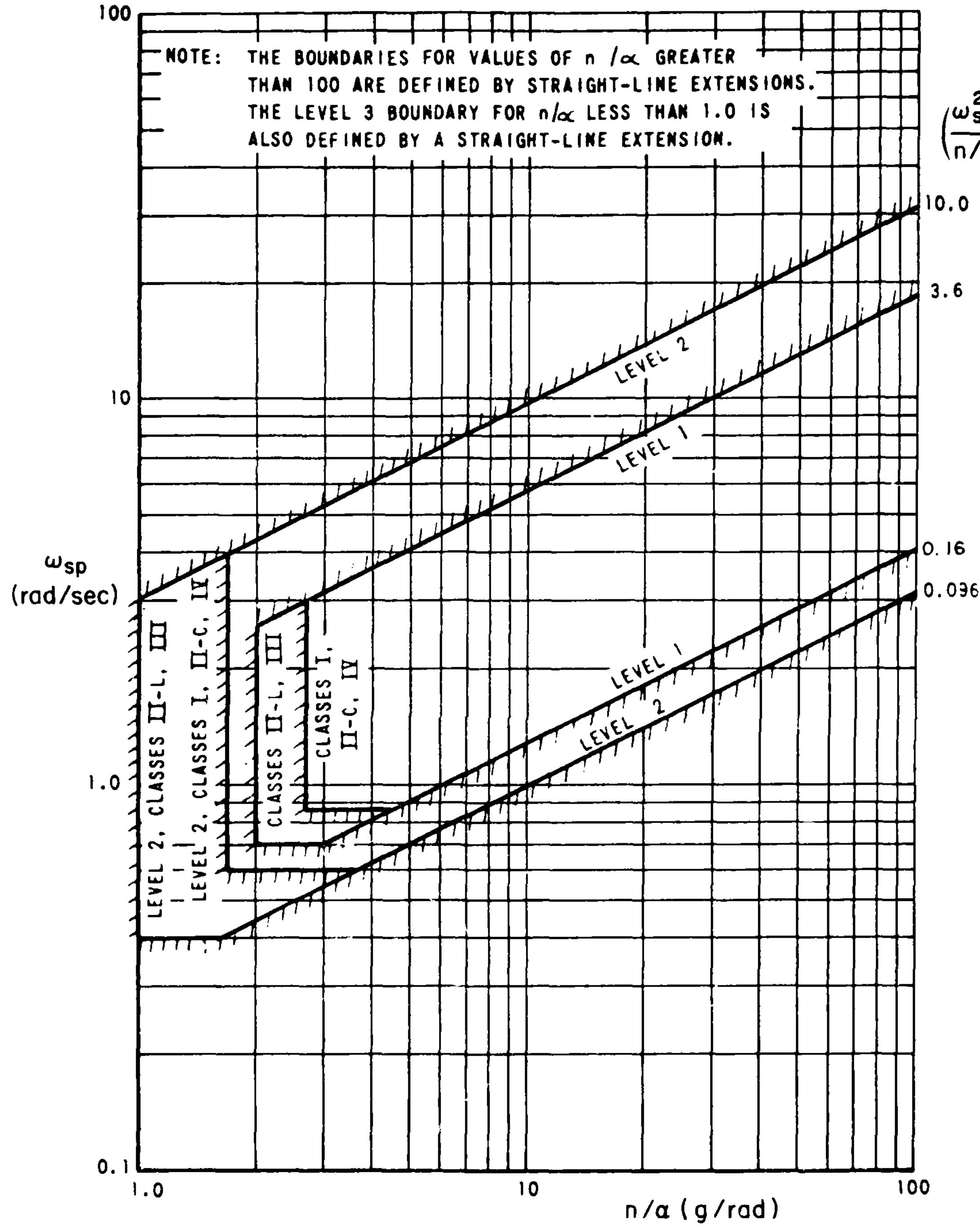
a) Category A Flight Phases

Figure 1 (3.2.1.1). Requirements for Short-Term Pitch Response to Pitch Controller (ω_{sp} vs. n/α)



b) Category B Flight Phases

Figure 1 (3.2.1.1). Continued



c) Category C Flight Phases

Figure 1 (3.2.1.1). Concluded

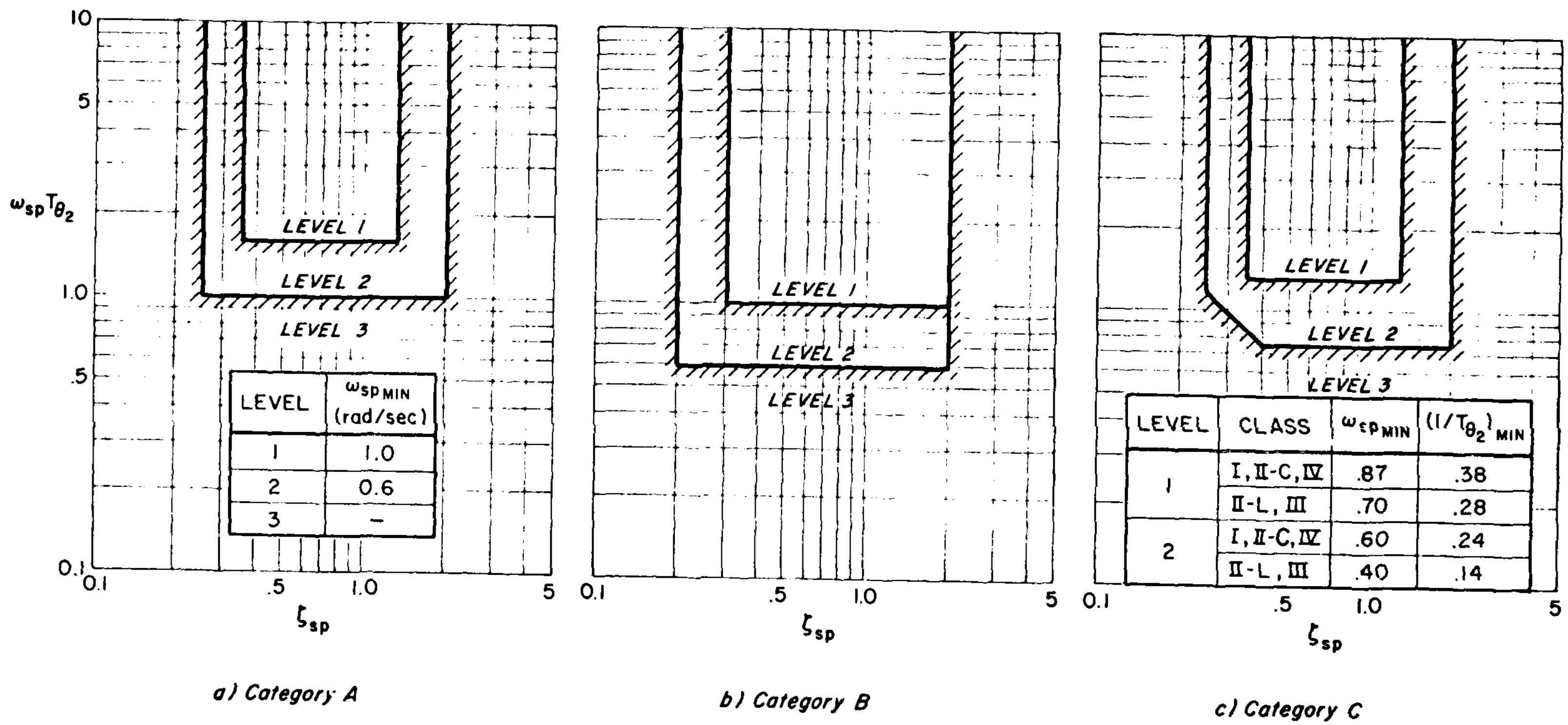


Figure 2 (3.2.1.1). Requirements for Short-Term Pitch Response to Pitch Controller ($\omega_{sp}T_{\theta_2}$ vs. ζ_{sp})

TABLE 1 (3.2.1.1)

EQUIVALENT PHUGOID DAMPING RATIO LIMITS

Level 1	ζ_p at least 0.04
Level 2	ζ_p at least 0
Level 3	T_2 at least 55 sec ($T_2 = -0.693/\zeta_p\omega_p$) and $\lvert\zeta_p\rvert < 1.0$

TABLE 2 (3.2.1.1)

EQUIVALENT SHORT-PERIOD DAMPING RATIO LIMITS

LEVEL	CATEGORY A AND C FLIGHT PHASES		CATEGORY B FLIGHT PHASES	
	MINIMUM	MAXIMUM	MINIMUM	MAXIMUM
1	0.35	1.30	0.30	2.00
2	0.25	2.00	0.20	2.00
3	$T_2 > 6$ sec*	--	$T_2 > 6$ sec*	--

*In the presence of one or more other Level 3 flying qualities, ζ_{sp} shall be at least 0.05 unless flight safety is otherwise demonstrated to the satisfaction of the procuring activity. T_2 applies to the value of an unstable first-order short-period root

TABLE 3 (3.2.1.1)

LIMITS ON AIRCRAFT RESPONSE DELAY, τ_e

LEVEL	ALLOWABLE DELAY (sec)
1	0.10
2	0.20
3	0.25

τ_e is the greater of τ_{e_θ} or τ_{e_n}

D. RATIONALE BEHIND REQUIREMENTS

The use of lower-order equivalent systems allows us to retain boundaries generated from classical airplane data, i.e., *MIL-F-8785C*, Reference 4. More specifically, requirements for pitch axis control have set boundaries on the classical modal parameters, ζ_p, $\zeta_p \omega_p$, $1/T_{\theta_2}$, ζ_{sp} and ω_{sp}, in

$$\frac{\theta}{\delta} = \frac{M_\delta (1/T_{\theta_1})(1/T_{\theta_2})}{[\zeta_p;\ \omega_p][\zeta_{sp};\ \omega_{sp}]}$$

This expression is a linearized, reduced-order model of the actual aircraft response. It assumes any higher-order terms due to structural and flight control effects are sufficiently separated from $1/T_{\theta_1}$, $1/T_{\theta_2}$, ω_p and ω_{sp} to allow order reduction by restricting the frequency range of analysis. In most cases the phugoid and short period modes are sufficiently separated that further order reduction is possible as follows:

$$\frac{-M_\delta/M_\alpha (1/T_{\theta_1})}{[\zeta_p;\ \omega_p]} \quad \text{and} \quad \frac{M_\delta (1/T_{\theta_2})}{[\zeta_{sp};\ \omega_{sp}]}$$

Phugoid | Short Phugoid

These expressions are universally recognized as pitch models of phugoid and short-period dynamics, respectively, within appropriate respective frequency ranges.

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 additional modes in many cases are no longer well separated in frequency from $1/T_{\theta_1}$, $1/T_{\theta_2}$, ω_p, and ω_{sp}. In fact, there are commonly a number of high-order system characteristic roots in the short-period frequency range. While no specific guidance is offered, the phugoid and short period are generally separated by at least a factor of 10. The assumption of a widely separated phugoid and short period mode breaks down at low values of static stability (i.e., $M_\alpha \doteq 0$) such as for aircraft with extreme aft center of gravity locations and on most STOL configurations. When the phugoid and short period are no longer identifiable as widely separated modes, the short term response cannot be characterized as second order in nature. Inasmuch as the data utilized to generate the LOES parameter boundaries were representative of classical second order short-term responses, these boundaries (Figures 1 and 2) are not valid for the higher-order short term responses under consideration (i.e. ω_{sp} not $\gg \omega_p$). It is recommended that the bandwidth criterion be utilized for these cases (Para. 3.2.1.2).

One approach to reducing the order of these systems for specification compliance was to choose a suitable short-period pair of roots from the high-order array. This was done either by picking "dominant roots" (the most suitable pair) or by "root tracking" (identifying the locus of aircraft short-period roots as the various feedback gains were introduced). There has been considerable experience in recent years to indicate that these approaches will lead to unsatisfactory handling qualities. A clear responsibility of the MIL Standard is therefore to discourage this particular type of order reduction. Considerable research has been devoted to order reduction by matching frequency responses to obtain low-order equivalent systems. Since this approach has the advantage of including the effects of adjacent modes, is easily related to previous specifications, and identifies important equivalent delay effects, it is included in the specification.

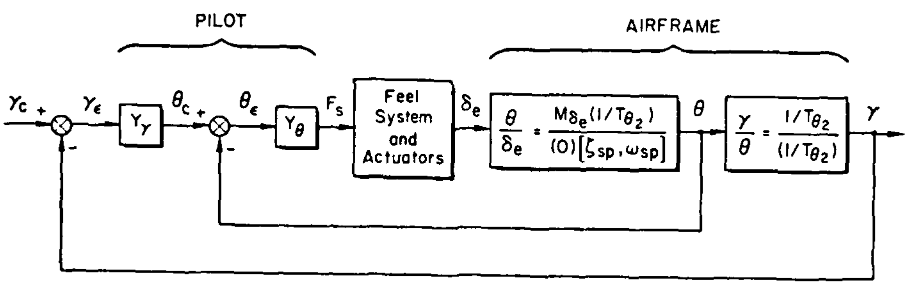
1. Phugoid Requirements

These are taken directly from Reference 4. No phugoid research data have been generated since that earlier specification. Modern flight control systems commonly do not exhibit an identifiable phugoid.

The pitch response is a suitable response for identifying the phugoid, since pitch residue is normally high in the phugoid mode (when it is not, because of stability and control augmentation, the phugoid requirements are greatly exceeded). Very frequently an accurate estimate of the phugoid can be obtained from a narrow frequency range which excludes the short period.

2. Short-Term Pitch Response Requirements Using ω_{sp} vs. n/α

For conventional aircraft, without direct lift control, the piloting task can be viewed as an inner- and outer-loop, single-controller task as indicated in Figure 3. This figure points out that $1/T_{\theta_2}$ plays a role in the short-term attitude dynamics and also defines the short-term flight path (or load factor) response. The boundaries in Figure 1 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 below.



$$\left.\frac{\theta}{F_s}\right|_e = \frac{A_\theta (1/T_{\theta_2}) e^{-\tau_{e_\theta} s}}{(0)\left[\zeta_e, \omega_e\right]}$$

$$\left.\frac{\gamma}{F_s}\right|_e = \frac{A_\gamma e^{-\tau_{e_\gamma} s}}{(0)\left[\zeta_e, \omega_e\right]}$$

where

$$(a) = (s + a)$$

$$\left[\zeta, \omega\right] = \left[s^2 + 2\zeta\omega s + \omega^2\right]$$

Figure 3 (3.2.1.1). Pilot Control of Pitch Attitude and Flight Path

a. Control Anticipation Parameter (CAP)

Bihrle in Reference 113 defines the Control Anticipation Parameter (CAP) as

$$\text{CAP} \equiv \frac{\ddot{\theta}_o}{\Delta n_{z_{ss}}} \approx \frac{\omega_{sp}^2}{\frac{V}{g}\left(1/T_{\theta_2}\right)}$$

where $(V/g)\ (1/T_{\theta_2})$ can be approximated by n/α for aircraft with negligible control system dynamics and tail lift effect, as is common. Because of the time lapse in reaching the steady state, a pilot needs an earlier indication of the response to control inputs -- and the initial and final responses must be neither too sensitive nor too insensitive to commanded flight-path change.

b. Frequency Response Interpretation

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 4).

c. MIL-F-8785B BIUG Interpretation

A closely related analysis in Reference 11 arrives at $(F_s/n)M_{F_s}$ 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_s frequency response of Figure 4.

d. Static Stability Interpretation

Because $n/\alpha \propto C_{L_\alpha}$ and $\omega_{sp}^2 \propto C_{m_\alpha}$, $\omega_{sp}^2/(n/\alpha)$ is widely recognized as being related to static margin, dC_m/dC_L. In fact,

$$\frac{n}{\alpha} \cong \frac{C_{L_\alpha} qS}{W}$$

and

$$\omega_{sp}^2 \cong -\frac{qS\bar{c}}{I_y} C_{L_\alpha}\left(\frac{C_{m_\alpha}}{C_{L_\alpha}} + \frac{\rho V S}{4m} C_{m_q}\right)$$

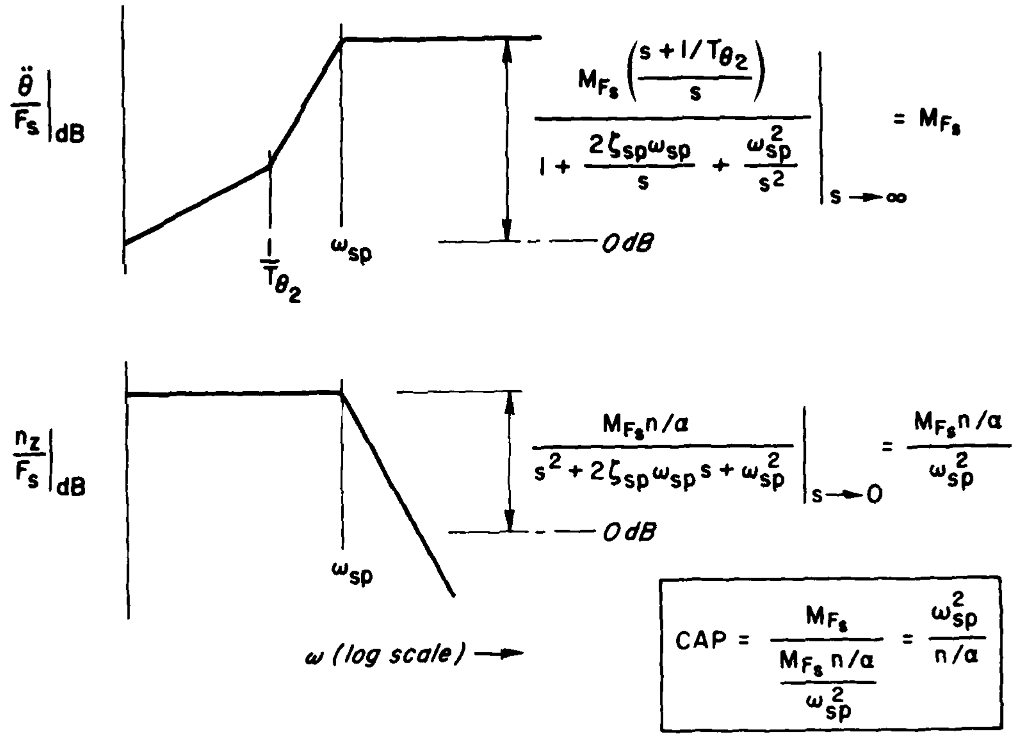


Figure 4 (3.2.1.1). Definition of CAP From Frequency Response Asymptotes

Therefore, neglecting C_{m_q},

$$\frac{\omega_{sp}^2}{n/\alpha} \cong -\frac{C_{m_\alpha}}{C_{L_\alpha}} \bar{c} \cdot \frac{W}{I_y}$$

$$= y_{sm} \frac{g}{k_y^2}$$

where y_{sm} is the static margin in the same units as $\bar{c}$ (i.e., y_{sm} is the absolute distance of the neutral point ahead of the c.g.), and k_y^2 is the pitch radius of gyration (with C_{m_q} considered, it is common to speak of "maneuver margin" rather than "static margin.")

For many aircraft, k_y is about 17 percent of the aircraft length ℓ, so $\omega_{sp}^2/(n/\alpha) \approx 1100\ (y_{sm}/\ell^2)$.

For the F-4 aircraft, using the values of 64 ft for its length and 16 ft for $\bar{c}$, the specified Level 1 value of 0.28 for $\omega_{sp}^2/(n/\alpha)$ reduces to a stick-fixed static margin requirement of 5 percent. Thus this requirement is comparable to the earlier static margin requirement in U.S. Air Force Specification 1815B/Navy Specification SR-119B (Reference 57).

3. Short-Term Pitch Response Requirement Using $\omega_{sp}T_{\theta_2}$ vs. ζ_{sp}

As was mentioned previously, the short-period frequency requirement of 8785C was based upon the premise that the normal acceleration response to attitude changes is a primary factor affecting the pilot's perception of the minimum allowable ω_{sp} [i.e., limits are placed on $\omega_{sp}^2/(n/\alpha)$]. The physical interpretation of the so-called "Control Anticipation Parameter" [CAP = $\omega_{sp}^2/(n/\alpha)$] assumes that the dominant concern for a pilot pitch control input is normal acceleration response. It is, of course, also true that the pitch attitude response to pitch control inputs is of paramount importance. Whether the appropriate correlating parameter is n/α or $1/T_{\theta_2}$ is a moot point in that data that are correlatable with $1/T_{\theta_2}$ will generally also correlate with n/α. This issue was studied in Reference 114, where it was observed that the product $\omega_{sp}T_{\theta_2}$ provided a slightly better correlation than CAP. Physically, $\omega_{sp}T_{\theta_2}$ represents the separation in phase between path aircraft responses in path and pitch attitude. Thus $\omega_{sp}T_{\theta_2}$, in combination with ζ_{sp}, also defines the shape of the attitude frequency response, with "desirable values" yielding a K/s shape in the frequency range of primary interest (see Reference 114).

A useful criterion, therefore, is the product $\omega_{sp}T_{\theta_2}$, based on the separation in frequency between $1/T_{\theta_2}$ and ω_{sp}. If these two are close in frequency, the aircraft responses in attitude and flight path to elevator deflection occur almost simultaneously, resulting in abrupt heave responses to the pitch controller. This produces pilot comments such as "trim hard to find" and "pilot effort produces oscillations."

4. Use of Lower-Order Equivalent Systems

Both of the above requirements involve classical flying quality parameters. The concept of lower-order equivalent systems allows us to retain equivalents of these parameters for highly augmented aircraft. The rationale utilized in the Handbook is that these equivalent parameters can be compared with well established boundaries set by classical airplane data. This concept was originally developed more than ten years ago and was first reported in Reference 80. It has been refined considerably since that time (for example, see References 81-89). A key issue during the development of the lower-order equivalent system (LOES) approach was whether to fix or free $1/T_{\theta_2}$ during the fitting process.

When $1/T_{\theta_2}$ is allowed to be free it can take on very large (or small) values. If the physical significance of $1/T_{\theta_2}$ were related purely to attitude control it would be appropriate to utilize the freed value of $1/T_{\theta_2}$ when making comparisons with the criterion boundaries. However, there is considerable evidence to indicate that the role of $1/T_{\theta_2}$ in the correlations of classical airplanes is more related to the lag from the response in attitude to the response in path. If this is indeed the case, the ω_{sp} vs. n/α boundaries are to be physically interpreted as a specification on attitude (ω_{sp}) as well as path control (n/α). The appropriate value of $1/T_{\theta_2}$ to plot on the criterion would then be the fixed (real) one. With a single control surface (e.g., no DLC) no augmentation can change the dynamic relationship of pitch to heave motion.

An example of the differences between the effective parameter values with $1/T_{\theta_2}$ fixed and free is illustrated in Table 4 (taken from Reference 83 fits of the Reference 12 data). From this table it can be seen that substantial differences in all the effective parameters exist between the $1/T_{\theta_2}$ fixed and free fits. Hence the dilemma is not a trivial one.

TABLE 4 (3.2.1.1)

EXAMPLES OF VARIATIONS IN LOES PARAMETERS WITH $1/T_{\theta_2}$ FIXED AND FREE

CONFIGURATION	$1/T_e$		ω_e		ζ_e		τ_e	
	FIXED	FREE	FIXED	FREE	FIXED	FREE	FIXED	FREE
1A	1.25	0.43	3.14	2.54	0.39	0.65	0	0.020
1G	1.25	176.	0.78	1.55	0.74	1.07	0.185	0.043
2H	1.25	4.08	2.55	3.80	0.80	0.52	0.126	0.098
4D	1.25	5.25	3.47	4.61	0.58	0.23	0.169	0.111

In general, for a single controller the proper relationship between attitude and flight path will always be maintained if the following two transfer functions are matched simultaneously.*

$$\frac{\dot{\theta}}{F_s} = \frac{K_1(s + 1/T_{\theta_2})e^{-\tau_{e_\theta} s}}{s^2 + 2\zeta_{sp}\omega_{sp}s + \omega_{sp}^2} \tag{1}$$

$$\frac{n_z'}{F_s} = \frac{K_2 e^{-\tau_{e_n} s}}{s^2 + 2\zeta_{sp}\omega_{sp}s + \omega_{sp}^2} \tag{2}$$

where n_z' refers to the normal acceleration at the instantaneous center of rotation (at $x_{cr} = Z_{\delta_e}/M_{\delta_e}$ the initial n_z response to a step control input is zero. It is assumed that measurements of $\dot{\theta}$ and n_z' are in level, symmetrical flight, so that $\dot{\theta} \equiv q$). Conventional subscripts on the equivalent dynamics are retained for consistency.

*This discussion assumes that the phugoid mode is well-separated in frequency from the short-period mode, as is normally the case.

This conclusion was reached as a result of lengthy discussions centered on whether $1/T_{\theta_2}$ should be fixed or free during the fitting process. Proponents of holding $1/T_{\theta_2}$ fixed argued that n/α in the specification is centered about path control. Holding $1/T_{\theta_2}$ fixed at the value determined from the lift curve slope $[1/T_{\theta_2} \doteq -Z_w = (\rho S U_o/m)(C_{L_\alpha} - C_D)]$ preserves the known path to attitude relationship, i.e.,

$$\frac{\gamma}{\theta} = \frac{1}{T_{\theta_2}s + 1}$$

Those in favor of allowing $1/T_{\theta_2}$ to be free noted that it tends to "gallop" to large values for aircraft with known deficiencies, thereby revealing the existence of a problem. For example, a current high-performance fighter is known to be rated "excessively sluggish" (Level 2) in the power approach flight condition. Figure 5 shows the characteristics with $1/T_{\theta_2}$ fixed and free. For $1/T_{\theta_2}$ fixed, the sluggish response is manifested as excessive effective time delay (0.15 sec) whereas for $1/T_{\theta_2}$ free the deficiency is manifested as an n/α which falls on the lower specification boundary. Finally, utilizing the $1/T_{\theta_2}$-free fit to ω_{sp}, but plotting the fixed value of n/α, actually predicts an airplane with excessive abruptness (plots above the upper limit in Figure 5). In this case, as in most such instances, either method predicts the same <u>Level</u> of flying qualities, but the <u>causes</u> manifest very differently.

It should be noted that a "perfect fit" using both the attitude and flight path transfer functions will <u>always</u> (for a single pitch controller) yield the <u>fixed</u> value of $1/T_{\theta_2}$. However, if there are lags, such as a stick prefilter, introduced at frequencies in the middle of the fitting region, the fit may be marginal: the lower-order equivalent system (LOES) of the $\dot{\theta}/F_s$ transfer function is a first over <u>second</u> order, whereas the dominant modes of the higher order system turn out to be a first over <u>third</u> order system.

The problem can be approached in two ways: 1) we can ignore the mismatch and use the LOES model in Eqs. 1 and 2; or 2) we can utilize LOES modes more appropriate to the controlled element rather than being

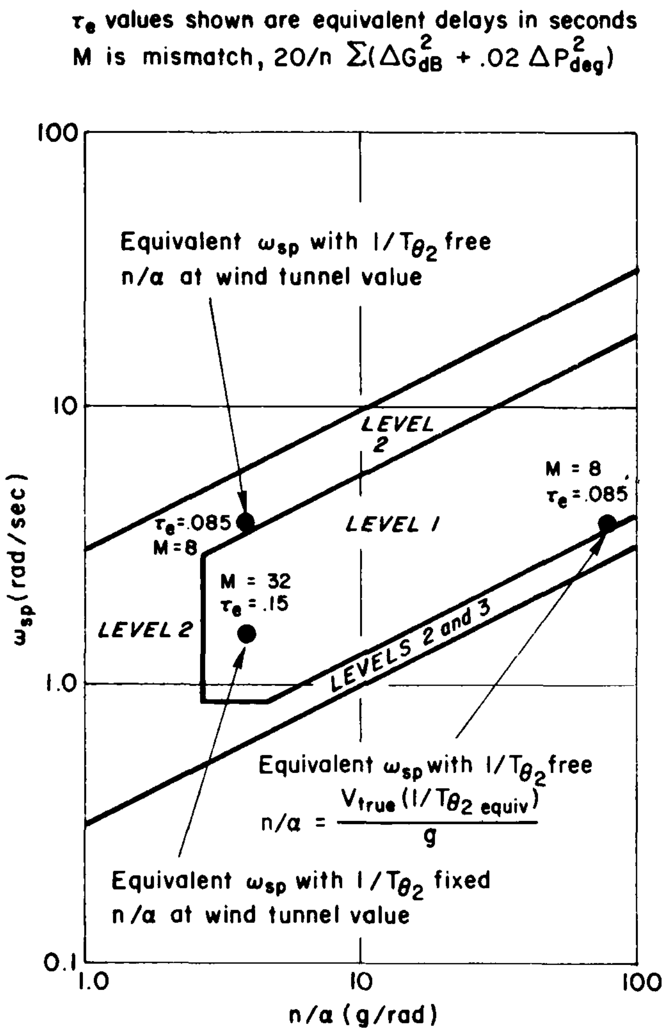


Figure 5 (3.2.1.1). Effect of Fitting with $1/T_{\theta_2}$ Fixed and Free, Category C Requirements

constrained to a first over second order. For example, aircraft with stick prefilters are better approximated by a first over third order, as noted above. The problem with the second alternative is that our data base is for classical unaugmented airplane data. The consequence of the first alternative is that the fitting routine could come up with parameter values which are not physically meaningful. The consensus was to accept the mismatch (Alternative No. 1), rather than attempt to expand the criterion.

5. Level 3

A first-order divergence (T_2 = 6 sec) is allowed for the Level 3 pitch attitude dynamics. This is consistent with the Level 3 static stability requirement in Paragraph 3.4.1.

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. Reference 181, for example, indicated a Level 2 boundary with T_2 (based on the unstable aperiodic root) of 2.5 seconds in "light" turbulence and 4.25 seconds in "moderate" turbulence. Pilot ratings were fairly constant at 5 to 6 until the time to double amplitude was reduced below 6 seconds, when significant deterioration began.

E. GUIDANCE FOR APPLICATION

1. Alternate Requirements

Two alternate requirements have been included for specifying pitch control through the use of classical parameters or their equivalents, i.e., ω_{sp} vs. n/α (Figure 1) and $\omega_{sp}T_{\theta_2}$ vs. ζ_{sp} (Figure 2). For the ranges of T_{θ_2} normally encountered in either augmented or unaugmented aircraft, these requirements are identical.

2. Equivalent Systems

The equivalent lower-order parameters for this section may be obtained by any means mutually agreeable to the procuring agency and contractor. The equivalent system matching routine documented in Appendix A is provided as guidance to indicate the expected level of sophistication in the matching procedure.

$$\frac{\dot{\theta}}{F_s} \simeq \frac{K_\theta(1/T_{\theta_2})e^{-\tau_{e_\theta}s}}{[\zeta_{sp};\ \omega_{sp}]}$$

$$\frac{n_z'}{F_s} \simeq \frac{K_{n_z}e^{-\tau_{e_n}s}}{[\zeta_{sp};\ \omega_{sp}]}$$

This representation is not intended to place any restrictions on the form of the match. For instance, the denominator modes may be composed of two first-order roots rather than an oscillatory pair. In this case, the snort-period roots are:

$$(1/T_{sp_1})(1/T_{sp_2}) \equiv \zeta_{sp}\omega_{sp} \mp \omega_{sp}\sqrt{\zeta_{sp}^2 - 1}$$

The parameters of the above equations should be obtained by matching the high-order pitch response and the normal load factor response from ω_1 to ω_2, with the frequencies defined as follows:

ω_1 Normally 0.1 rad/sec

ω_2 Normally 10 rad/sec, but not less than a priori estimates of ω_{sp} or $1/T_{\theta_2}$.

A priori estimates can be based on early analytical studies, wind-tunnel results, or extrapolations from experience with similar configurations. The purpose is to assure that the dynamics of the equivalent airframe are adequately defined, without requiring unusually low- or high-frequency end points in the match.

In using reduced-order models in a MIL Standard the question of allowable mismatch is important. There is currently insufficient data to place definitive requirements on mismatch between the HOS and LOES. It should be noted, however, that the question of mismatch is inherent in any n-dimensional representation of an m-dimensional response, when $n < m$.

Mismatch is defined as:

$$M = \Sigma(\Delta G)^2 + K\Sigma(\Delta\Phi)^2$$

$$= \Sigma(G_{HOS} - G_{LOS})^2 + K\Sigma(\Phi_{HOS} - \Phi_{LOS})^2$$

where

G is the amplitude in dB

Φ is the phase in radians

$(\Delta G)^2$ and $(\Delta\Phi)^2$ are calculated at discrete frequencies between ω_1 and ω_2 evenly spaced on a logarithmic scale and may be compared with the envelopes in Figure 6.

A brief NT-33 landing approach simulation tackled the question of mismatch (References 82 and 91). High-order systems and simulations of their low-order equivalents were flown. The experiment indicated that very large mismatches proved unnoticeable to pilots (a sum-of-squares mismatch around 200 in the frequency range of $0.1 < \omega < 10$ rad/sec compared to the previous arbitrary values of 10).

Reference 83 offers a theory to explain the large mismatches. By examining pilot rating differences between pairs of configurations in previous NT-33 data (References 5 and 12), frequency response envelopes were derived. Each pair of configurations consisted of an unaugmented, low-order airplane response and a high-order system formed by adding terms to the low-order response; Figure 6 shows the envelopes that were derived. Some 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 6 envelopes would be cause to suspect that the equivalent parameters may not accurately predict pilot opinion. In such cases it is recommended that the bandwidth criterion of Para. 3.2.1.2 also be checked.

Additional comments on the use of lower-order equivalent systems may be found in References 92 and 158.

The influence of mid-frequency added dynamics on LOES was discussed in Reference 158, where it was shown that a series of (possibly unrealistic) lead/lag combinations evaluated in the Neal-Smith in-flight

*The basic aircraft dynamics were modified via equalization networks. Modifications that resulted in 1 pilot rating change were defined as "maximum unnoticeable added dynamics."

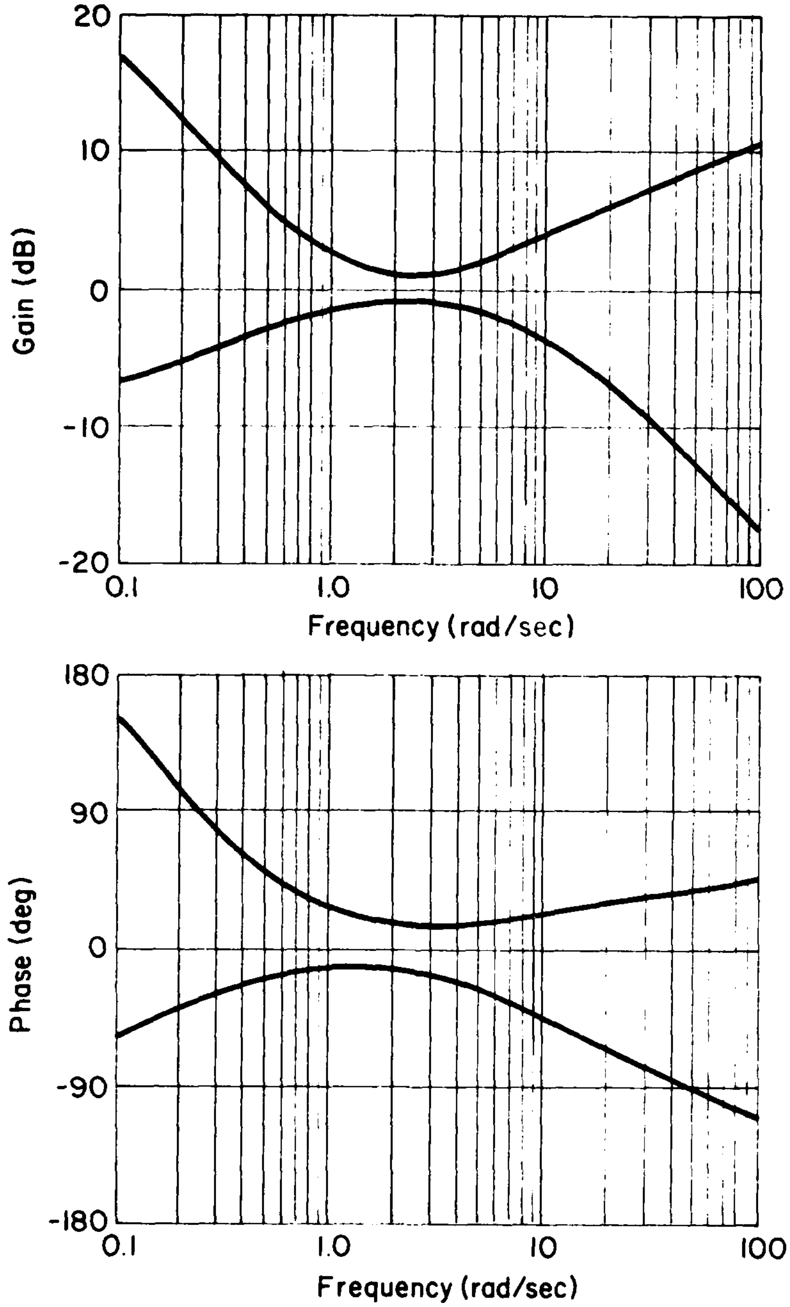


Figure 6 (3.2.1.1). Envelopes of Maximum Unnoticeable Added Dynamics simulation (Reference 12) produces LOES parameters which are not necessarily equivalent to their classical counterparts. Of the 51 configurations flown in the Neal-Smith program, 41 were either unaugmented or included only lags (first- or second-order). The flying qualities of about two-thirds of these are correctly predicted by the requirements. The remaining ten configurations had combination lead/lag dynamics, and only five of these are predicted accurately. For the five that failed, the equivalent dynamics (ζ_e, ω_e, τ_e) predicted Level 1 flying qualities but they were rated Level 2 by the pilots. Table 5 lists the dynamics of the HOS and LOES for these configurations.

Review of Table 5 reveals some interesting similarities in the low-order equivalent systems. With the exception of Configuration 1C, all have $\zeta_e < 0.5$ (though still greater than 0.35). Three have $\tau_e = 0$. All but 1C have a first-order lag near the short-period frequency; 1C has a first-order <u>lead</u> near ω_{sp}.

TABLE 5 (3.2.1.1). LEAD/LAG CONFIGURATIONS WITH LEVEL 1 LOES AND LEVEL 2 PILOT RATINGS

CONF.	HOS						LOES			PILOT RATINGS	
	$1/T_{\theta_2}$	ζ_{sp}	ω_{sp}	$1/T_1$	$1/T_2$	ω_3	ζ_e	ω_e	τ_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 (Reference 12); LOES from MCAIR (Reference 83)
2. Equivalent dynamics are Level 1 using MIL-F-8785C limits.

Figures 7 and 8 more clearly show the effects of the added lead/lag combinations on these configurations. In Figure 7 the Bode amplitude asymptotes of the basic vehicle dynamics ($1/T_{\theta_2}$, ω_{sp}) of Configurations 1, 2, and 7 are sketched, and the added dynamics are shown in broken lines. The first-order lead, in all cases, rotates the amplitude ratio counterclockwise by 20 dB/decade at frequencies above the lead frequency, while the first-order lag (near ω_{sp}) serves to rotate the amplitude ratio back. The net effect is an apparent "hump" around ω_{sp}, characterized in the LOES match by a low "equivalent" damping ratio (Table 5). This occurs because the lower order form has no other way to match a "hump" in the amplitude plot.

Similar effects are seen in the phase angle, Figure 8: the "humps" appear as phase lead (since, for the basic configurations, $\tau_e \equiv 0$). In fact, Figure 8 shows that an LOES match over the frequency range of 0.1-10 rad/sec would produce $\tau_e < 0$ (if negative time delays were allowed) for Configurations 1A, 2A, and 7A. The small positive τ_e for Configurations 1C and 2B results from the relatively low-frequency second-order lag (ω_3) for these cases, 16 rad/sec as opposed to 63 rad/sec.

There are two potential methods for dealing with lead/lag systems like those of Table 5; 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_e

As will be shown in "Supporting Data," a redefinition of the damping ratio limits of MIL-F-8785C would improve correlations for the Neal-Smith configurations. That is, if $\zeta_{e_{min}}$ for Level 1 were increased from 0.35 to 0.50, four configurations in Table 5 would fit the requirements (ignoring Configuration 1C, for which none of this discussion is applicable).

A change in the damping ratio requirements would mean that either: a) we restrict unaugmented vehicles as well; this is not appealing

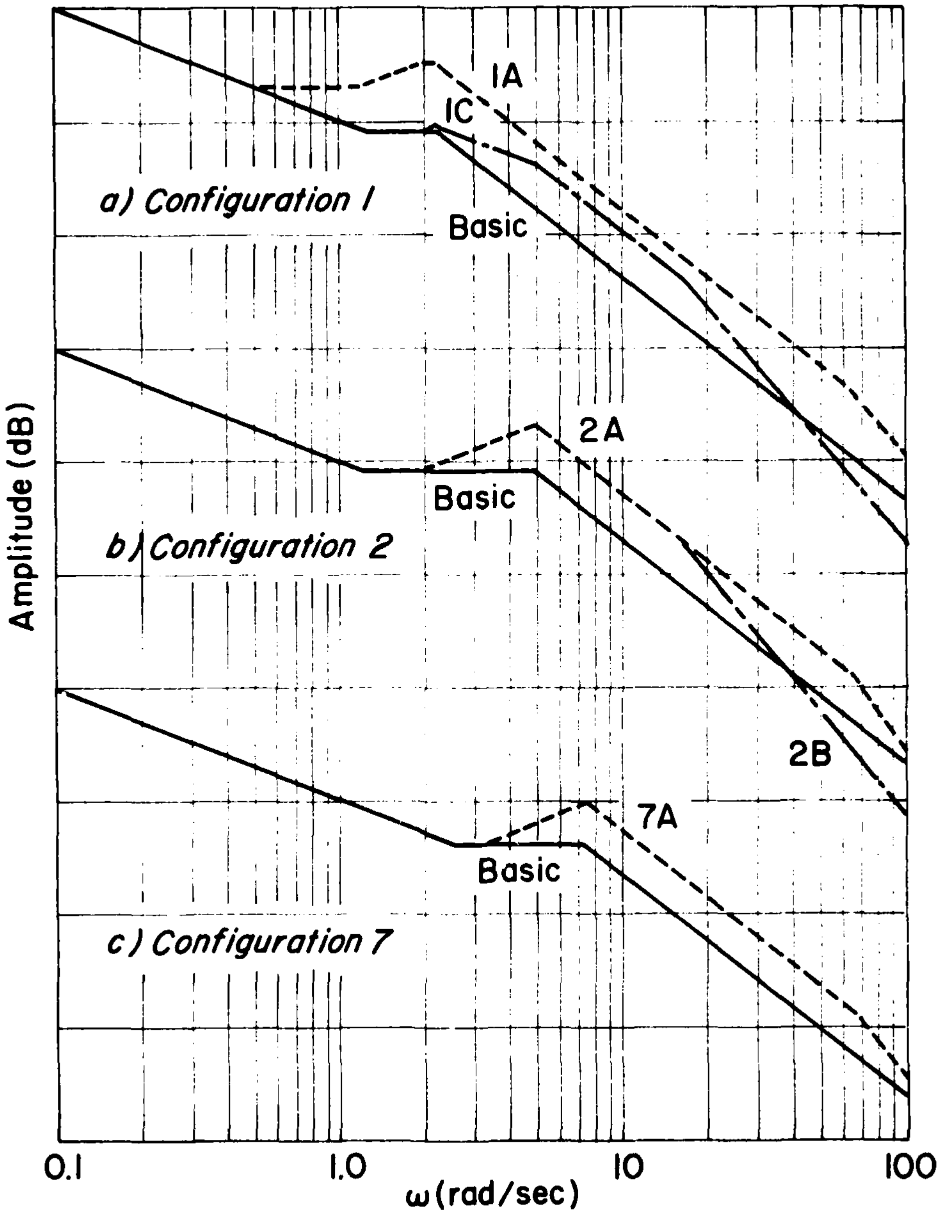


Figure 7 (3.2.1.1). Comparison of Bode Amplitude Asymptotes for Basic and Augmented Configurations of Table 5

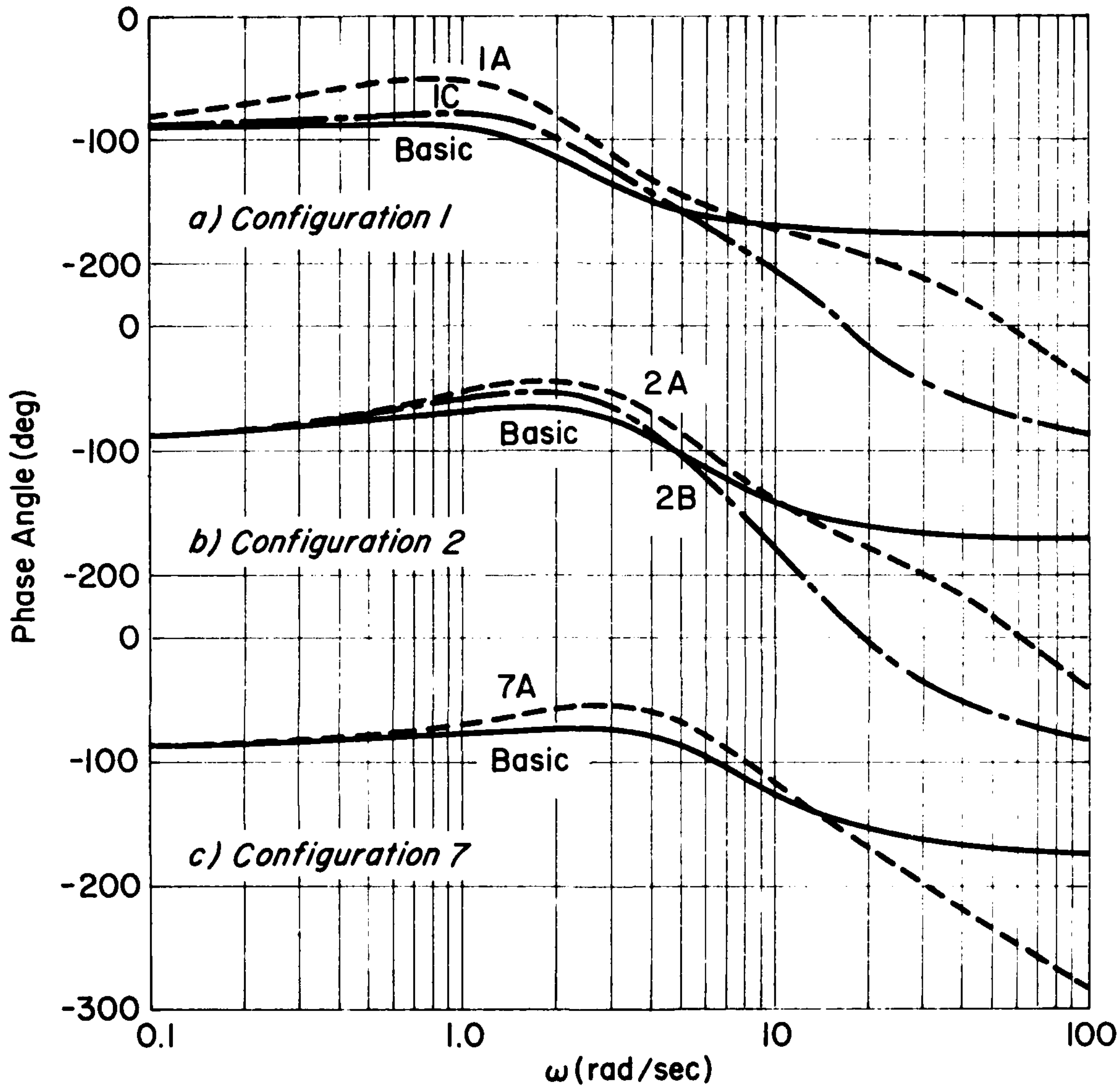


Figure 8 (3.2.1.1). Comparison of Bode Phase Angles For Basic and Augmented Configurations of Table 5 since $\zeta_{sp_{min}}$ is very well-supported by flight test data for classical aircraft; or b) we specify two sets of requirements -- one for unaugmented aircraft, and one for augmented aircraft. The latter is especially unattractive, since this is tacit admission that ζ_e is not equivalent to ζ_{sp}, and that "equivalent systems" is a misnomer. Additionally, it presents the problem of defining an augmented vs. unaugmented aircraft; e.g., should addition of a simple high-frequency stick filter (whose only major effect is to increase τ_e, Figure 9) suddenly mean that the aircraft must meet a more stringent damping requirement? In fact, the problem with Configurations 1A, 2A, 2B and 7A is directly traceable from pilot commentary to overabruptness and apparently has nothing to do with damping ratio at all. Looking at the frequency response plots in Figure 7, the "hump" at high frequency would indeed be expected to lead to an abrupt response. However, the low-order form has no provision for a hump in the amplitude without a correspondingly low ζ_{sp}. It seems unlikely that a flight control system designer would ever suggest equalization that would produce such humps in the frequency response. If for some unforeseen reason this should occur in the real (not simulated) world, it is suggested that the bandwidth criterion be utilized. Four of the five configurations of Table 5 fit the bandwidth requirement [see Figure 8 (3.2.1.2)].

b. Redefine τ_e

As mentioned above, three of the four low-ζ_e violators of Table 5 also have $\tau_e = 0$. As Figure 8 suggests, a better LOES fit is obtained for these three cases if τ_e is allowed to be less than zero. Specifically, negative time delays can be found in an LOES match to be as follows:

- Configuration 1A -- $\tau_e = -0.004$ sec
- Configuration 2A -- $\tau_e = -0.008$ sec
- Configuration 7A -- $\tau_e = -0.014$ sec

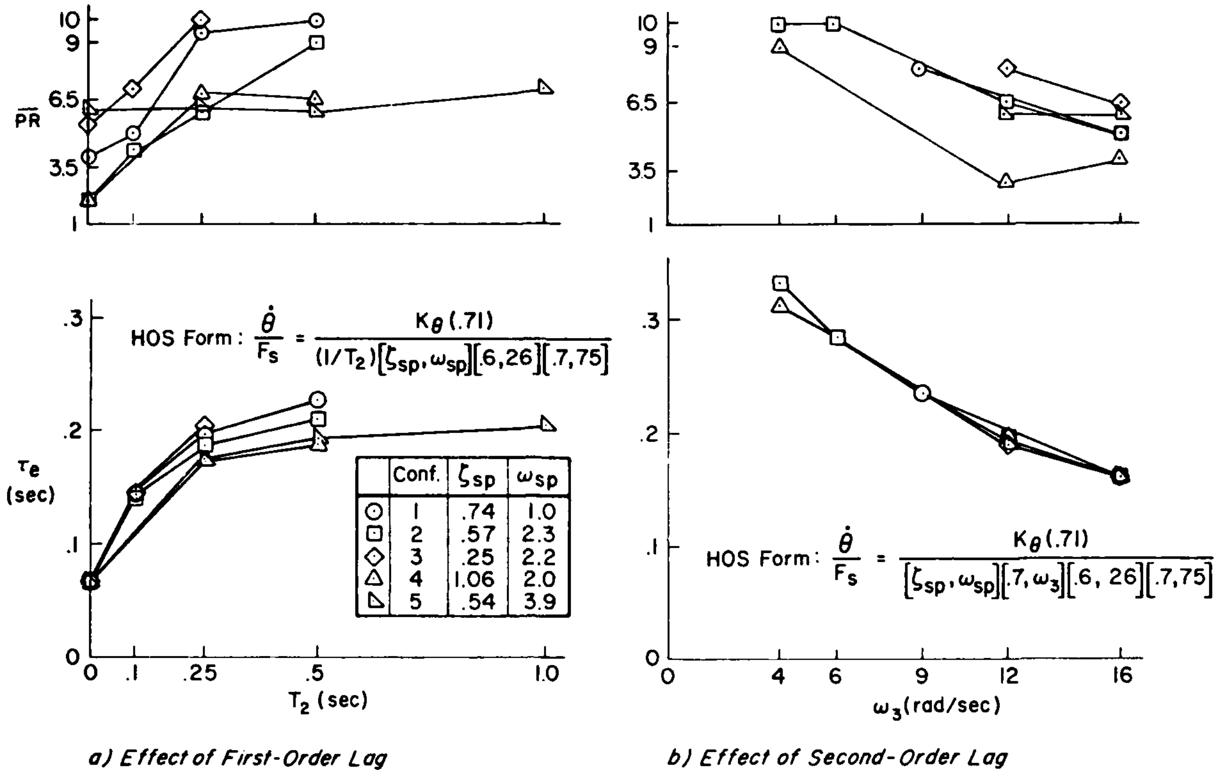


Figure 9 (3.2.1.1). Effect of First- and Second-Order Lags on Equivalent Time Delay and Pilot Rating: LAHOS Configurations (Reference 5)

The problem, clearly, is in interpreting the significance of negative time delay, or time lead. Physically, it might be considered to represent a HOS which is too abrupt (i.e., if $\tau < 0$, the system responds to an input τ seconds before the input is made or has finite magnitude at zero time) more or less in keeping with the above-noted pilot commentary.

3. Phugoid Dynamics

The long-term or phugoid mode is usually easily identifiable as a low-frequency lightly damped oscillation. When this is the case, it is reasonable to measure the damping ratio directly from the pitch rate or airspeed time response to a pulse pitch control input and utilize the result to check for compliance with Table 1. Equivalent system matches over the low frequency range will not be necessary in these cases. Additionally if the phugoid mode is suppressed by the flight control system to the point where it cannot be identified, the requirements of Table 1 are satisfied.

For compliance with Level 3, the time to double amplitude must be obtained utilizing the perturbation values of pitch attitude (or airspeed) away from trim.

F. DEMONSTRATION OF COMPLIANCE

As the design progresses, analysis, simulation, and finally flight test will be appropriate demonstration means.

1. Analysis

Construction of frequency responses for matching is conveniently performed by linearizing the high-order system for all possible input amplitudes, and matching the frequency response using the computer program of Appendix A. 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 (as discussed below).

2. Simulation

Fast Fourier analysis of real-time or non-real-time simulations of the aircraft is best performed using responses to a stick force input with wide frequency content.

3. Flight Test

References 84, 93, 94, and 115 describe fast Fourier reduction of flight data. Reference 93 describes AFFTC experience with the method.

Reference 94 discusses use of an electronically generated frequency sweep which worked adequately, and Reference 84 shows that FFT can work adequately even when the test condition is theoretically least suited to the method. Reference 115 shows a pilot-generated frequency sweep that worked very well. A typical frequency sweep and the resulting Bode plot (for a direct side force control configuration from Ref. 115) are shown in Figures 10 and 11, respectively. The instrumentation required to obtain these data was minimal, consisting of a yaw rate gyro and a pedal position transducer.

4. Piloting Aspects of Flight Test for Augmented Aircraft

Reference 84 discusses the piloting aspects of flight test for augmented aircraft, from which the following is extracted:

> Figure [12] illustrates a landing time history of a configuration with 0.17 seconds actual delay in the longitudinal command path. The landing is reasonably routine. Figure [13] shows the same configuration, with the same pilot, on a different landing. A pilot induced oscillation, with virtual loss of control, is evident. As discussed by Smith [Reference 91], a high rate of descent had developed which forced the pilot to control the aircraft more urgently....The pilot awarded a rating of 5 presumably on the basis that the aircraft had been landed routinely, with some deficiencies, on two occasions, and control was almost lost on one landing due to one of those momentary aberrations which afflict pilots for reasons unknown.
>
> The pilot in question proved himself during the simulation to be adaptable to widely different dynamics, whereas the main evaluation pilot in the same program, for example, registered a more consistently progressive deterioration in rating as the dynamic flying qualities parameters of the aircraft were degraded. The two pilots, though both highly skilled, therefore demonstrated a contrast in piloting technique. This contrast is significant because both adaptability and consistency are qualities which are needed, and therefore commonly exhibited, by many development test pilots. The adaptive technique, however, presents more of a challenge to the flying qualities engineer. He must pay particular attention to pilot briefing and to choice of piloting task.
>
> Pilot Briefing - Augmented dynamics possess potential problems which might not appear unless the pilot adheres to the properly chosen demanding task. Therefore, the briefing should

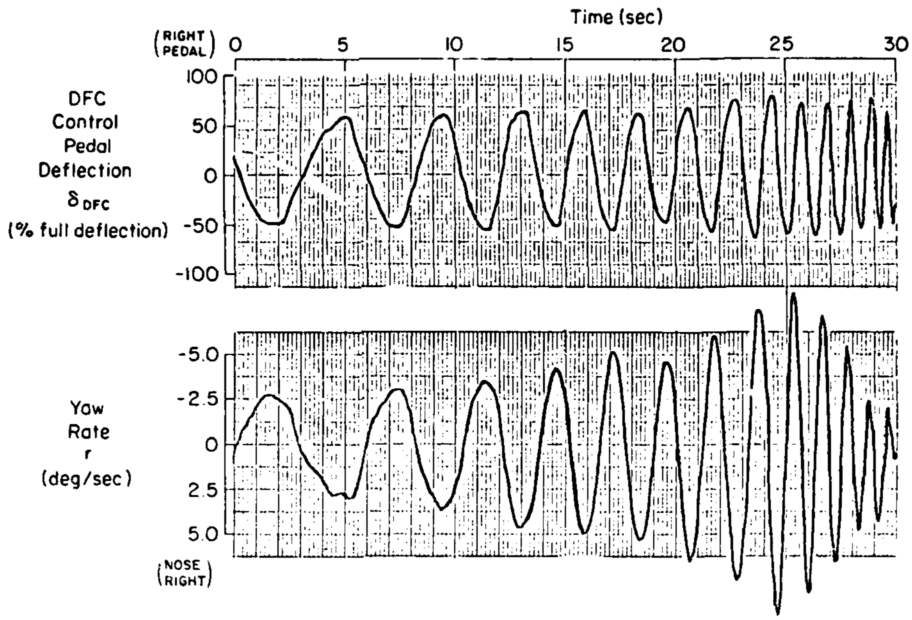


Figure 10 (3.2.1.1). Typical DFC Control "Frequency Sweep" and Response for Configuration Identification (Configuration WLT 2 from Reference 115)

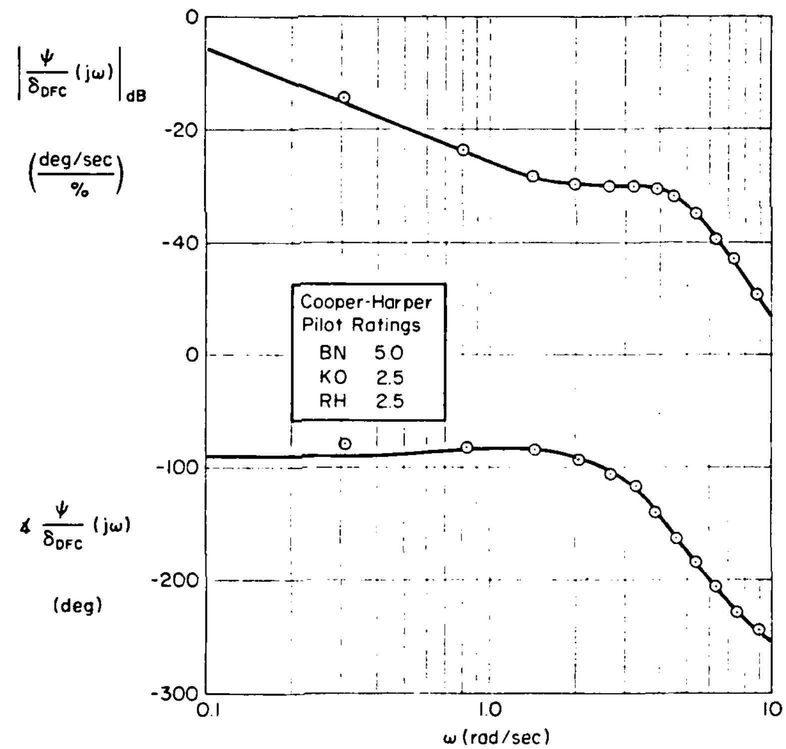


Figure 11 (3.2.1.1). Fourier Transformed Heading Response Resulting From Frequency Sweep Shown in Figure 10

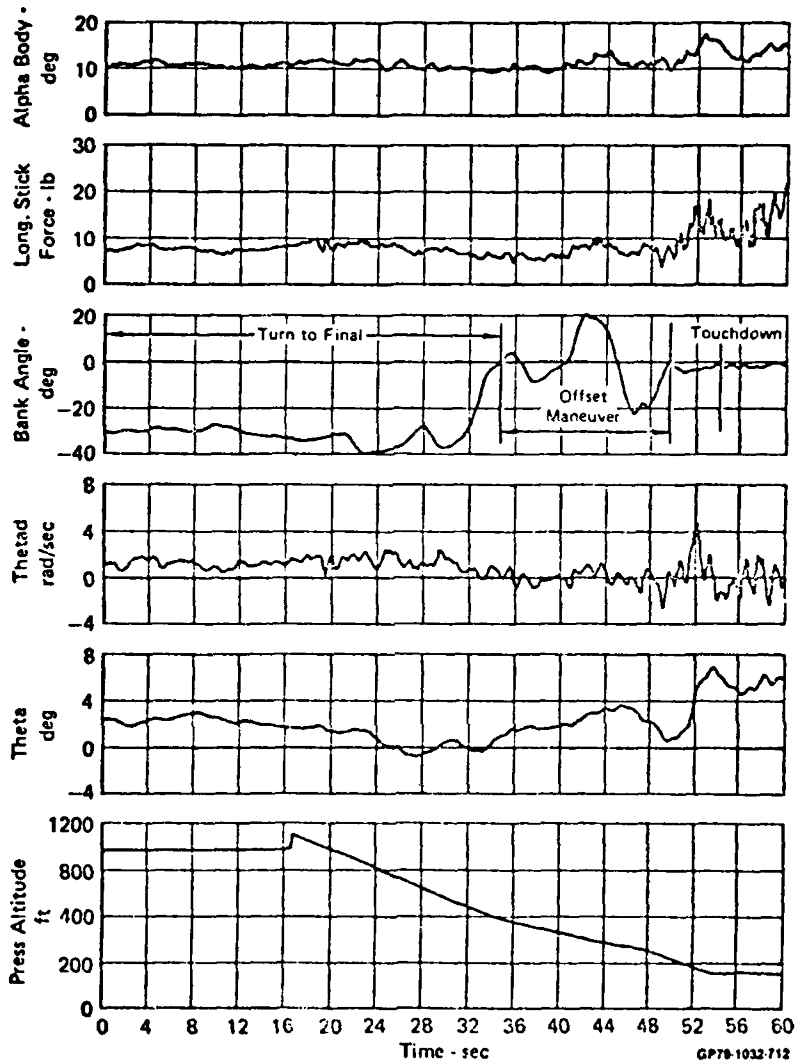


Figure 12 (3.2.1.1). Approach and Landing, No Pilot-Induced Oscillation, Configuration P12 of Reference 91, Medium Offset Approach (75 ft Lateral, 50 ft Vertical), Landing No. 1 (from Reference 84)

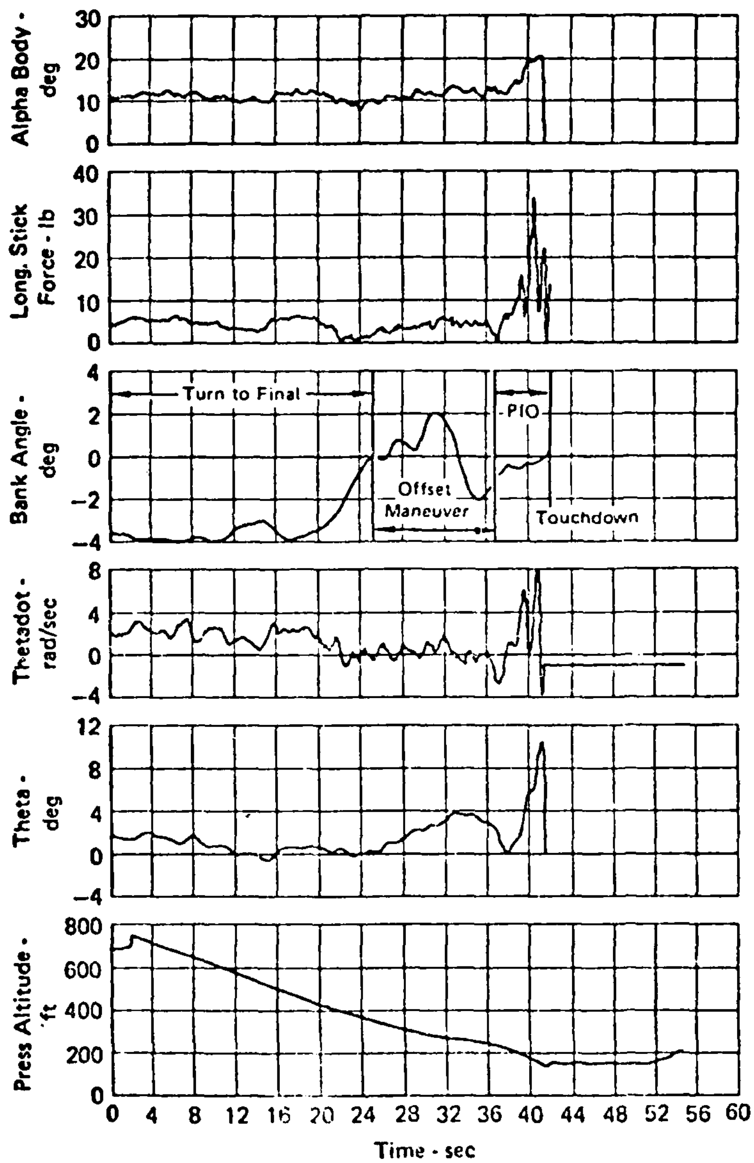


Figure 13 (3.2.1.1). Pilot-Induced Oscillation at Touchdown, Configuration P12 of Reference 91, Medium Offset Approach (75 ft Lateral, 50 ft Vertical), Landing No. 2 (from Reference 84) encourage the pilot to tackle the task aggressively but realistically. If the pilot is not aware of Smith's discussion of flying qualities "cliffs," the briefing should include it [Reference 5]. The classic cliff example is the peculiarity of lags in augmented dynamics, which can produce excellent flying qualities in loosely defined tasks, but pilot-induced oscillations in tightly-defined tasks. Therefore, the pilot should be encouraged to demand much of the airplane.

Piloting Task - A demanding but realistic task must be flown to expose potential flying qualities problems. An offset precision touchdown has proved very suitable for exploring longitudinal landing dynamics, for example. However, this is not necessarily the critical task for lateral dynamics. Task selection is difficult because pilot's perceptions of difficulty are sometimes misleading: the approach is commonly considered more difficult than flare and touchdown, for example, whereas the touchdown phase can clearly be critical [Reference 5]....There is an obvious need for operational realism in tasks, though there is some evidence that deliberately unrealistic tasks such as handling qualities during tracking (HQDT), might conveniently predict...difficulties in other more realistic tasks [Reference 112].

G. SUPPORTING DATA

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 subsections.

1. 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. There is a considerable amount of existing data which do not support the boundaries in Figures 1 and 2 (see Reference 11). However, a close review of the data reveals that most of the scatter was due to secondary effects. For example, in some cases the stick force per g (F_s/n) was outside the Level 1 limits. In other cases the tests were performed with an extremely low load factor limit ($n_z < 2.0$ g), or with the short-period frequency near a wing structural mode. There is evidence in the references that in these cases the extraneous factors may be influencing pilot ratings.

A careful review of References 9, 10, 22, 26, 28, and 95-102 was undertaken to cull out inappropriate data. Those reports which were complete enough to allow a thorough analysis of the test conditions and results were reviewed in detail. Others were considered to raise too many questions to be analyzed with confidence. (This is <u>not</u> meant to imply that some of the reports are invalid, but that they were not complete enough to gain sufficient insight into the causes for expected or unexpected pilot ratings.)

In particular, valid and usable reports were those which contained at least the following: 1) characteristics of short-period mode(s); 2) description of aircraft actuators, feel system, etc.; 3) description of maneuvers; 4) flight conditions; 5) pilot opinion rating scales used; and 6) <u>pilot comments or discussion of pilot comments</u>. The last factor especially reduced the number of reports retained for analysis. Pilots' descriptions of motions, responses, flight conditions, and control forces were considered essential to justify any pilot ratings which were inconsistent with other test data or with expectations.

References 22, 26, 95, 97, 100 and 101, all rather old, did not contain sufficient pilot commentary, if any, to be useful in the above context. In addition, high Mach number data in Reference 96 were not used because pilots considered the attitude display of the aircraft (an XB-70) to be inadequate when operating at high speeds. Low-n/α tests of Reference 9 ($n/\alpha \doteq 16.9$ g/rad) were subject to a buffet-onset load factor limit of $n_z = 2$ g -- low for evaluating a fighter-type aircraft. It was also noted in Reference 9 (page 41) that:

> Airplane sensitivity was more erratic and difficult to control when the structural modes of the airplane were excited. The primary structural mode excited was wing bending, which occurred at frequencies between 17 and 21 rad/sec (2.7 to 3.3 cps). These bending frequencies were observed in the oscillograph record of a wing tip mounted accelerometer and are a function of the fuel remaining in the tip tanks. Both pilots commented on the varying degree of

structural excitation that occurred when the airplane undamped frequencies varied from approximately 8 to 11.5 rad/sec (approximately half the structural frequencies). The erratic nature of the pilot ratings and pilot-selected stick forces in this region are also understandable. The pilots were obviously correcting and interpreting sensivity due to structural factors as well as the inherent airplane sensitivity.

Based on this evidence all data of References 9, 13, and 102 (which are T-33-based experiments) with $\omega_{sp} > 8$ rad/sec were deleted. Ratings data from Reference 99 (taken in a B-26, simulating a fighter configuration) showed greater scatter and overall better (lower) ratings than any of the other reports. This led to an evaluation of the reference, and to the conclusion that the tasks of Reference 99 were not sufficiently demanding to provide a good basis for evaluation of closed-loop handling qualities. Hence the data were not used.

In summary, References 9, 28, and 102 provided good short-period data for Category A; Reference 96 contained the only Category B data; References 10 and 98 contained Category C information.

a. Category A

1) ω_{sp} vs. n/α Criterion

Figure 14 shows the short-period frequency boundaries for the Category A Flight Phases. The applicable data (with Level 1 F_s/n) from References 9, 28, and 102 are compared with the boundaries. These data represent 52 separate ω_{sp} - n/α combinations flown and rated by six pilots. Eight configurations which fell within the Level 1 boundaries were rated Level 2 or worse. 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. Note that most of the violations occur at large n/α (as at high speed).

The data in Figure 14 represent those cases for which ζ_{sp} and F_s/n were within the present Level 1 boundaries. Therefore, the ratings shown can be assumed to be due solely to short-period frequency and n/α influences. For some experiments it could be argued that even Level 2 F_s/n should be plotted, since the pilots were allowed to select the

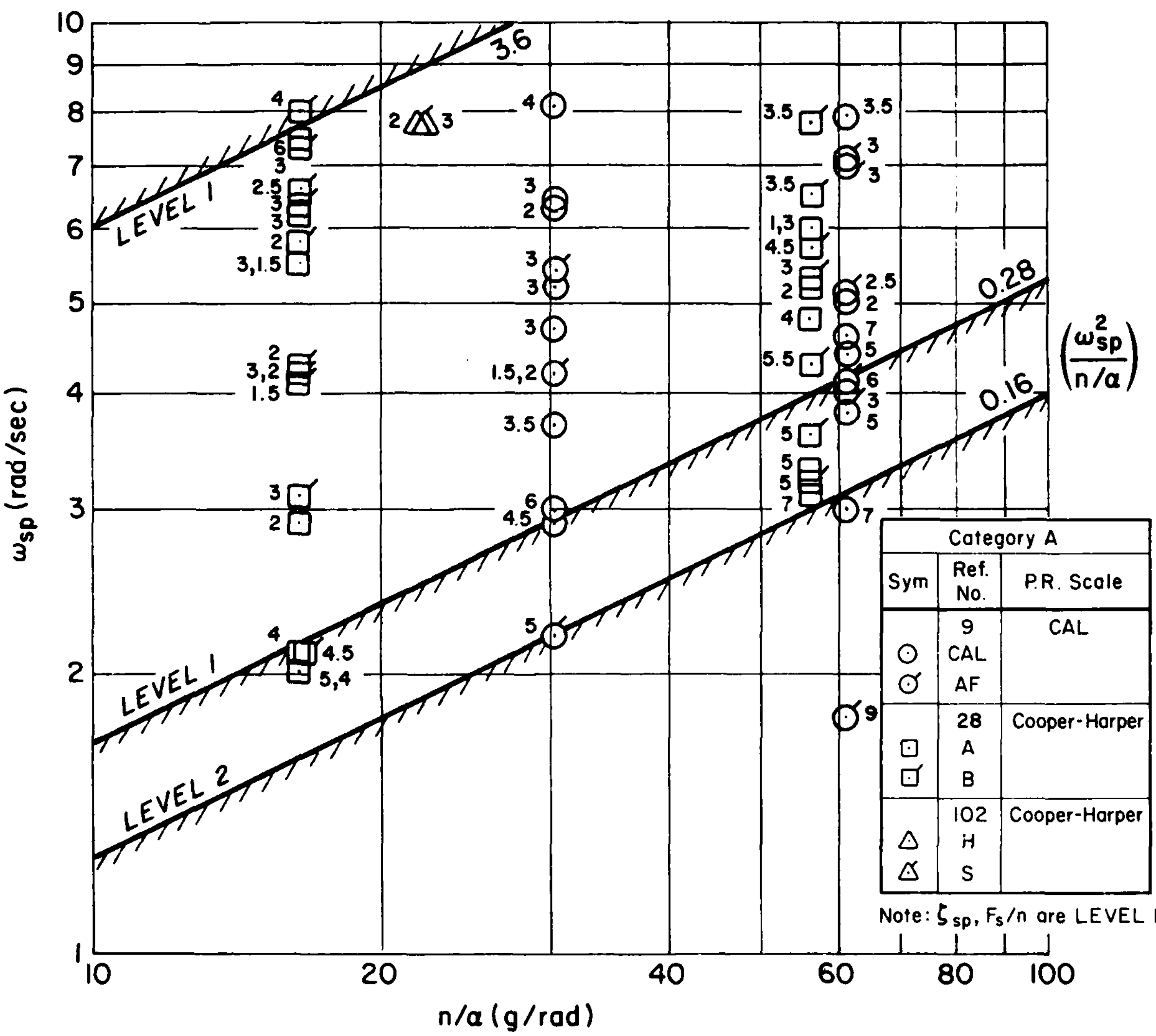


Figure 14 (3.2.1.1). Comparison of Pilot Ratings With Category A Short-Period Frequency Requirements optimum value. We have taken a somewhat conservative approach by eliminating these data. Our reasoning was that Levels 2 and 3 are boundaries for an off-nominal or failed state, and that pilots will not have a chance to optimize F_s/n after a failure. It should be noted that Level 2 values of F_s/n are usually selected by pilots to account for a basic flying quality deficiency. For example, a pilot would desire a very low F_s/n after a failure which results in a statically unstable airframe, requiring pulse-like control inputs.

Short-period damping boundaries are shown in Figure 15. Since this criterion presents ζ_{sp} as an independent parameter, it has been plotted directly against pilot rating. The other variables (ω_{sp}, n/α, and F_s/n) are Level 1 values. The resulting data (48 individual ratings) show definite trends in correlation with the boundaries. However, several points for which damping is good ($\zeta_{sp} \doteq 0.67$–0.74) are rated Level 2 to 3. The low-damping data from Reference 9 suggest that the ζ_{sp} lower limits could be reduced. Seven ratings are worse (higher in value) than predicted, all occurring within the Level 1 boundaries; only one is better than predicted.

Any possible interdependence between ζ_{sp} and $\omega_{sp}^2/(n/\alpha)$ can be taken into account by replotting the criterion boundaries on a grid of ζ_{sp} vs. $\omega_{sp}^2/(n/\alpha)$ as in Figure 16.

The authors of the 8785B BIUG (Reference 11) also noticed that ζ_{sp} lower limits were too restrictive; however, they decided that since most evaluations had been conducted in minimal turbulence the data-supported limits would not account for effects of turbulence. Thus, the limits of 8785B (Reference 13) were chosen, somewhat higher.

The upper $\omega_{sp}^2/(n/\alpha)$ limits of 8785C (Reference 4) are difficult to confirm based on the Figure 16 plot because the high-frequency ($\omega_{sp} > 8$ rad/sec) data have been removed from the data base, as noted above. Some of the data may be usable, however, since the structural bending mode was reported to be most pronounced at low speed with a high fuel load. For this Handbook all high-frequency data have been removed; and time did not permit further review of the low-fuel and high-speed cases

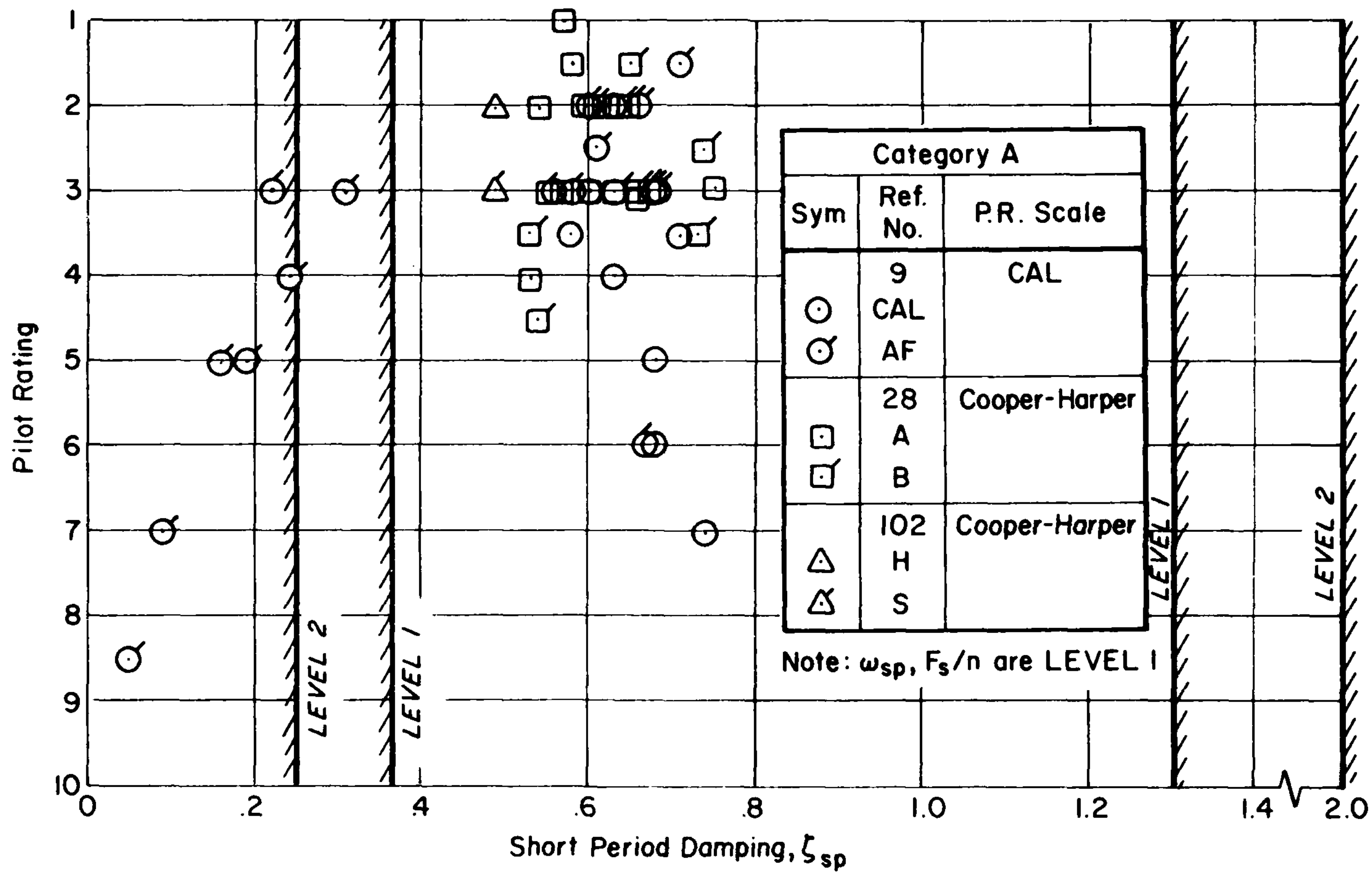


Figure 15 (3.2.1.1). Comparison of Pilot Ratings With Category A Short-Period Damping Requirements

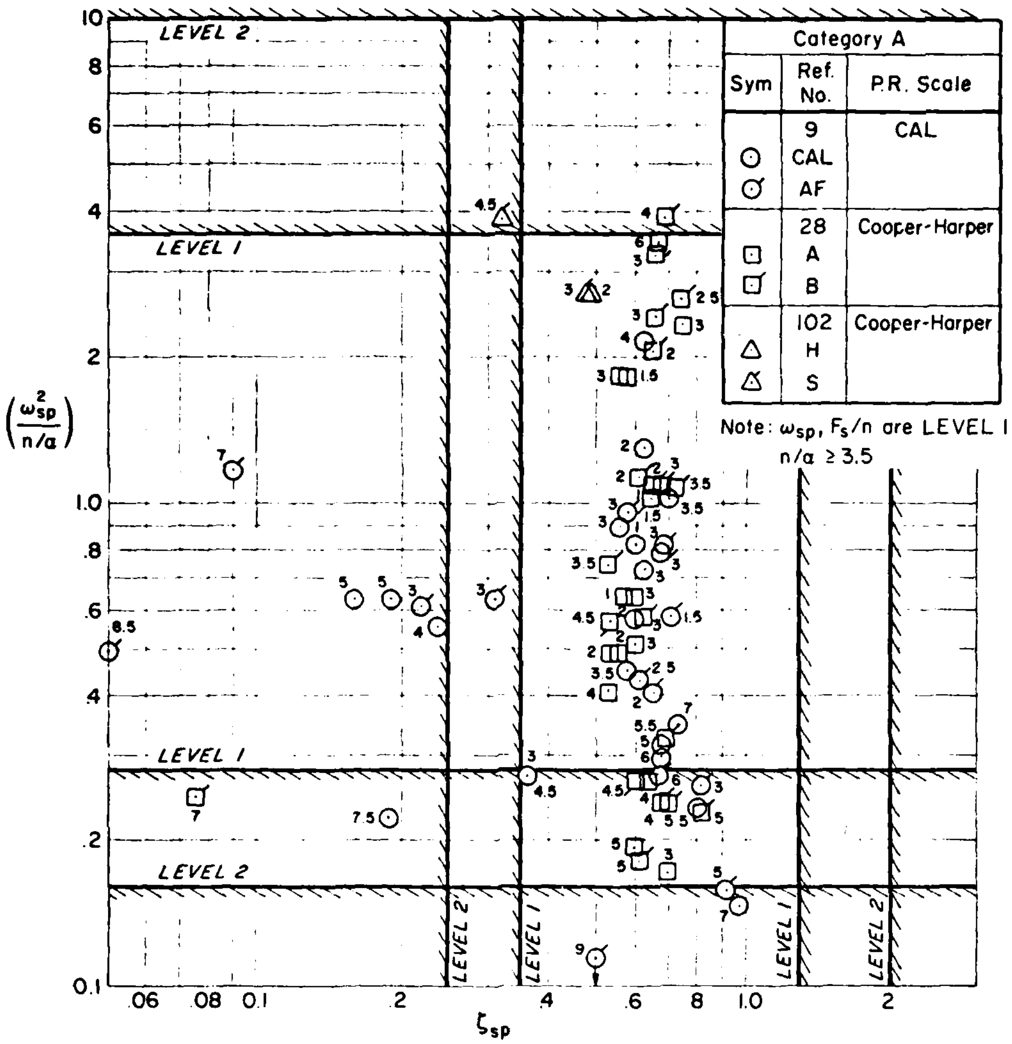


Figure 16 (3.2.1.1). Comparison of Pilot Ratings with Category A Short-Period Frequency and Damping Ratio Requirements for possible suitability. Thus, a better analysis of the CAP upper boundaries could perhaps be accomplished using any such valid data as may exist in References 9, 28, and 102. In the meantime, the upper boundary on $\omega_{sp}^2/(n/\alpha)$ has been retained.

2) $\omega_{sp}T_{\theta_2}$ vs. ζ_{sp} Criterion

Figure 17 shows the data used to support the requirements based on $\omega_{sp}T_{\theta_2}$ and ζ_{sp}. The damping limits are not supported by the pilot ratings but are consistent with those shown in the previous discussion. The absolute lower limits on ω_{sp} utilized in the ω_{sp} vs. n/α criterion have been retained in the $\omega_{sp}T_{\theta_2}$ vs. ζ_{sp} requirement for the lack of any better data. They are presented in a table in Figure 2a.

More work needs to be done to define the upper limits on $\omega_{sp}T_{\theta_2}$ for Category A.

b. Category B

1) ω_{sp} vs. n/α Criterion

Applicable pilot ratings from Reference 96 (XB-70) are compared with the ω_{sp} limits in Figure 18 and with the ζ_{sp} boundaries in Figure 19. The data do not conflict with the boundaries, but there are not enough data to judge the appropriateness of the criteria.

2) $\omega_{sp}T_{\theta_2}$ vs. ζ_{sp} Criterion

Since there are insufficient data to propose boundaries, the Category B limits have been made compatible with the Category A and C limits, so $\omega_{sp}T_{\theta_2} = 1.0$ for Level 1 and $\omega_{sp}T_{\theta_2} = 0.58$ for Level 2. Figure 20 illustrates the criterion, and compares the Reference 96 data.

c. Category C

1) ω_{sp} vs. n/α Criterion

The Category C flight test data of References 10 and 98 (B-367-80 and T-33, respectively) are compared with the short-period frequency requirements in Figure 21 and damping requirements in Figure 22.

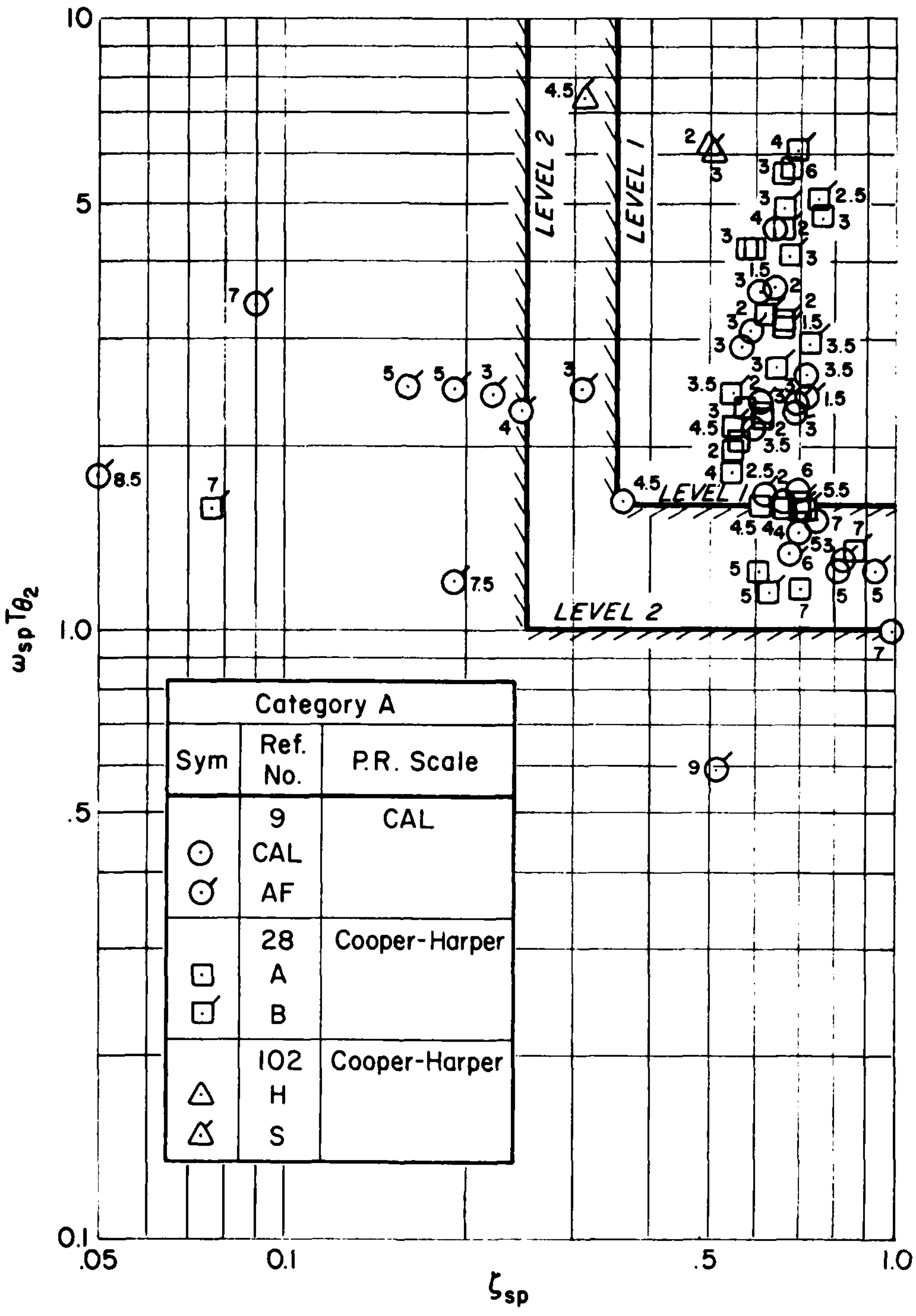


Figure 17 (3.2.1.1). Alternate Category A Flying Qualities Requirements for Short-Period Pitch Response

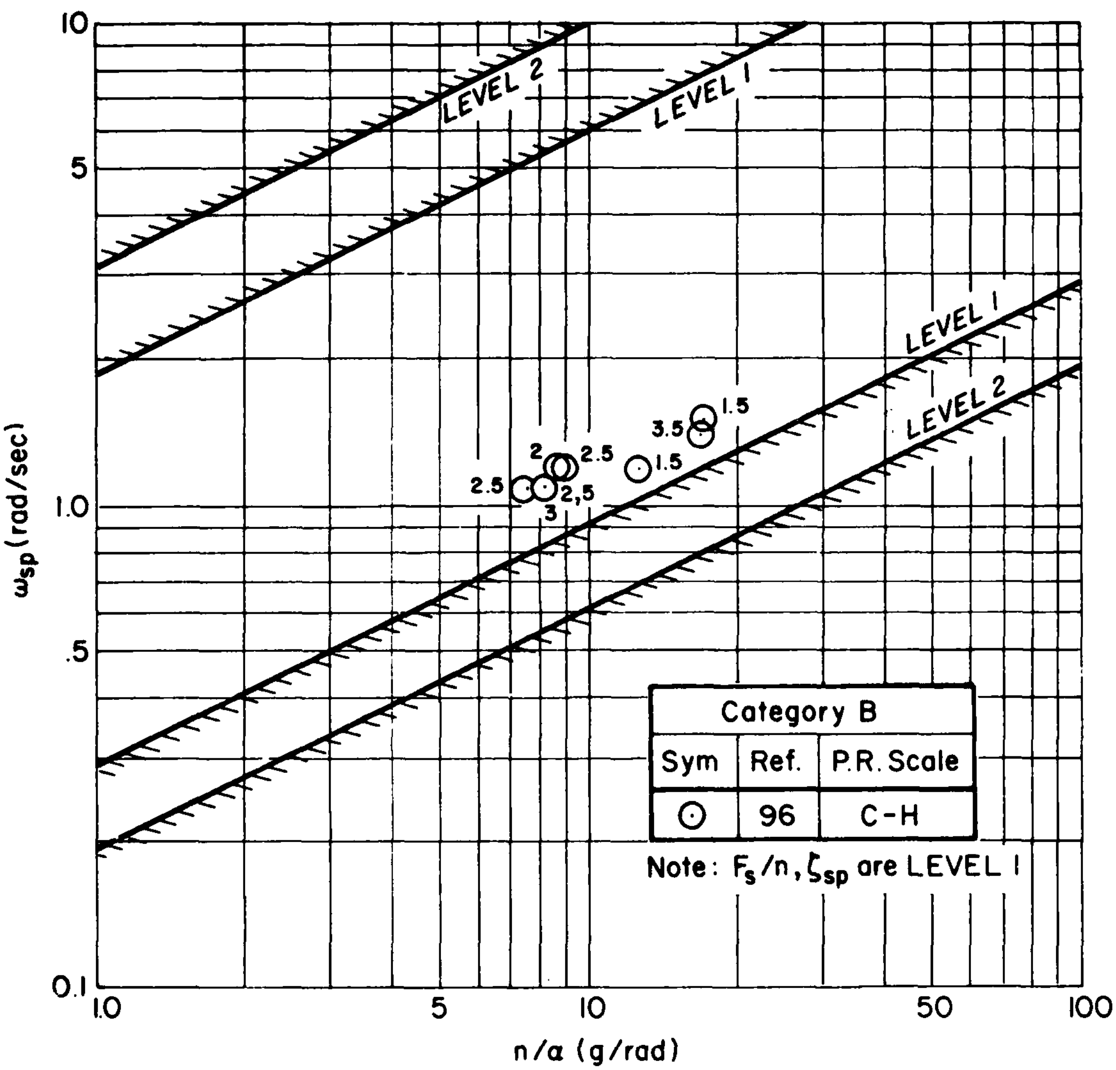


Figure 18 (3.2.1.1). Comparison of Pilot Ratings with Category B Short-Period Frequency Requirements

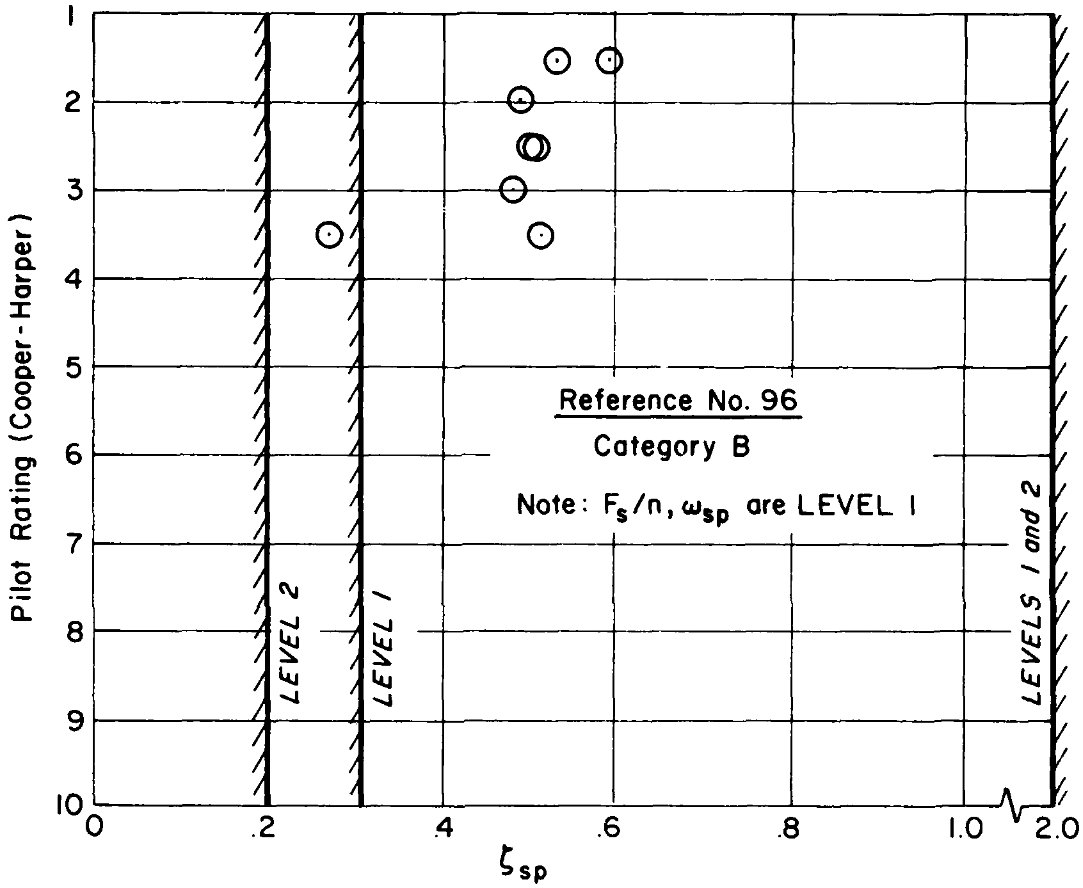


Figure 19 (3.2.1.1). Comparison of Pilot Ratings with Category B Short-Period Damping Requirements

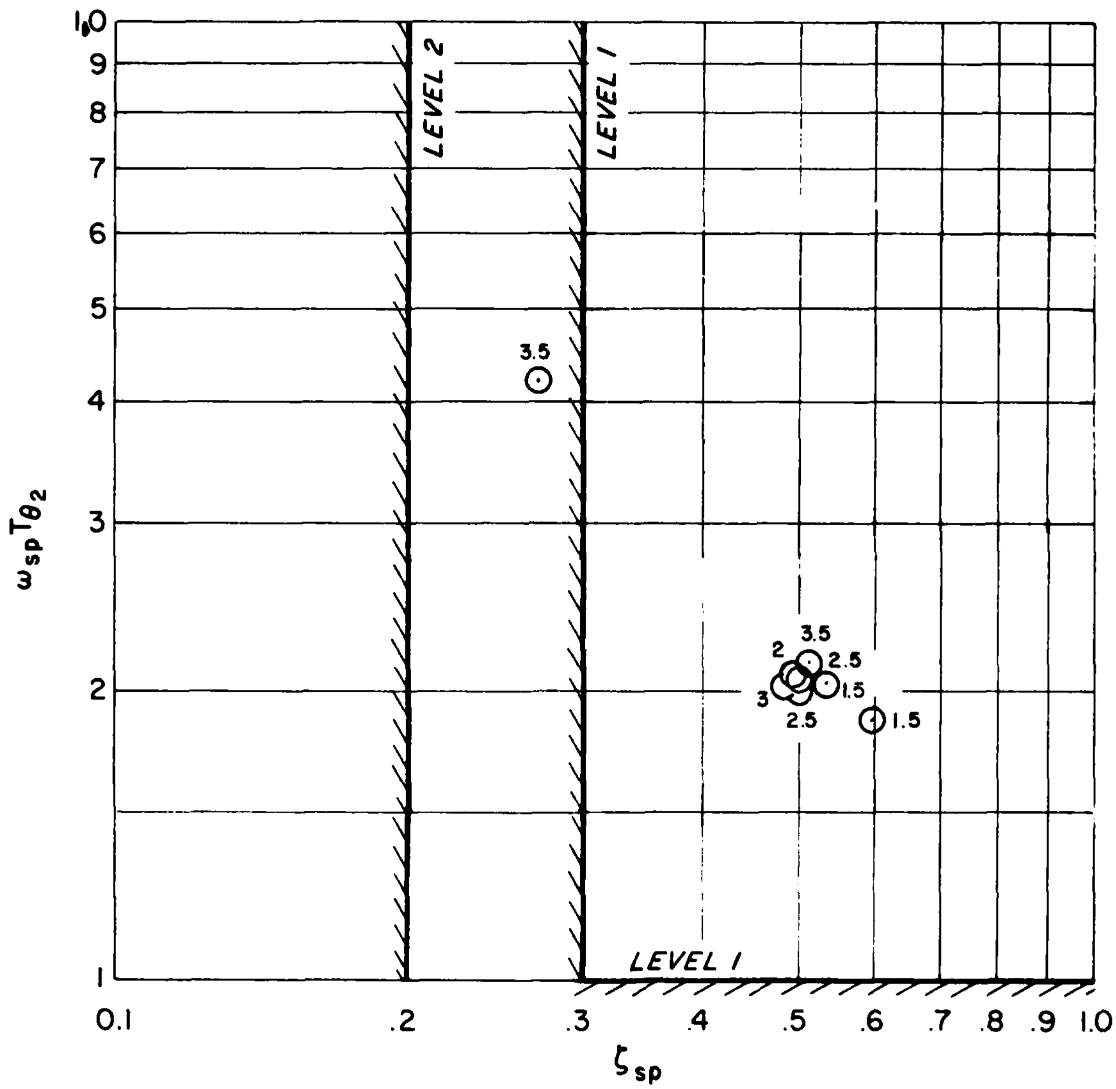


Figure 20 (3.2.1.1). Alternate Category B Short-Period Flying Qualities Requirements (Reference 96 data, Level 1 F_s/n)

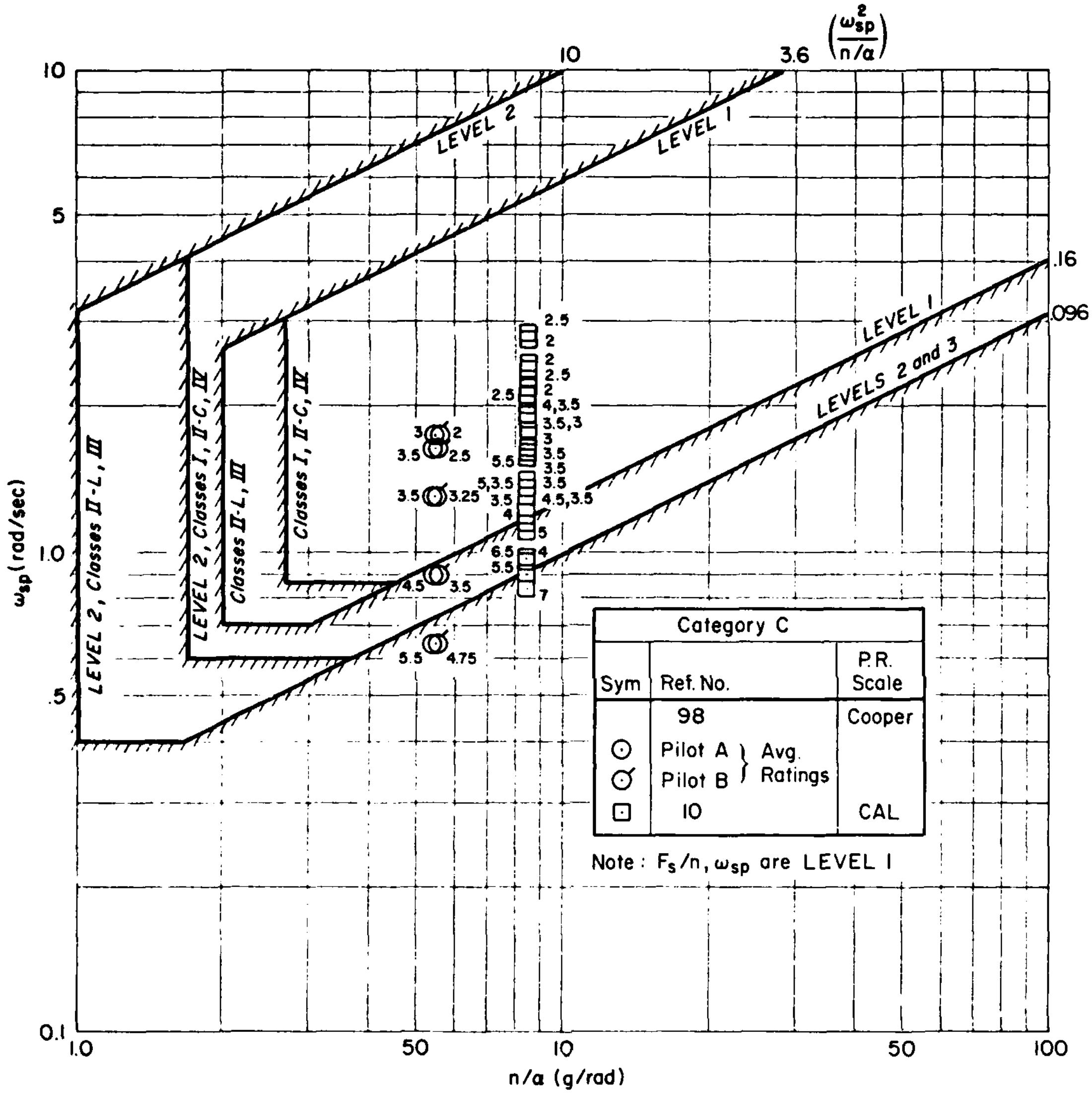


Figure 21 (3.2.1.1). Comparison of Pilot Ratings with Category C Short-Period Frequency Requirements

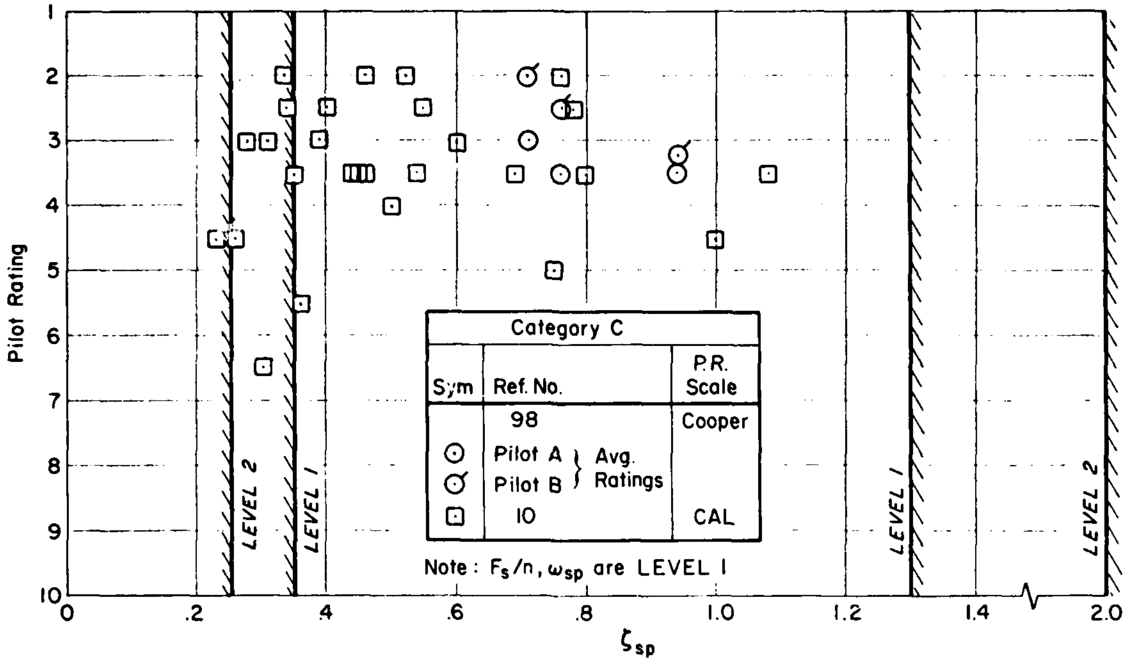


Figure 22 (3.2.1.1). Comparison of Pilot Ratings with Category C Short-Period Damping Requirements

The frequency data fit the boundaries very well, with a success rate of about 81 percent. This is comparable to the 80 percent for the Category A data, but there are far fewer Category C ratings, over a much smaller range of n/α. Damping predictions are worse, 73 percent — identical to that for Category A. The data support reduction in the minimum ζ_{sp} for all levels, similar to those suggested by the Category A data. However, as for the Category A data, these tests were conducted in minimal turbulence, so the MIL-F-8785C damping ratio limits have been retained.

2) $\omega_{sp}T_{\theta_2}$ vs. ζ_{sp} Criterion

Preliminary, straight-line boundaries of $\omega_{sp}T_{\theta_2}$ and ζ_{sp} are shown in Figure 23 (solid lines). The data fit these limits with a confidence level of about 82 percent, so the boundaries seem to work well. An even better fit is given by the dashed lines in Figure 23, which correlate

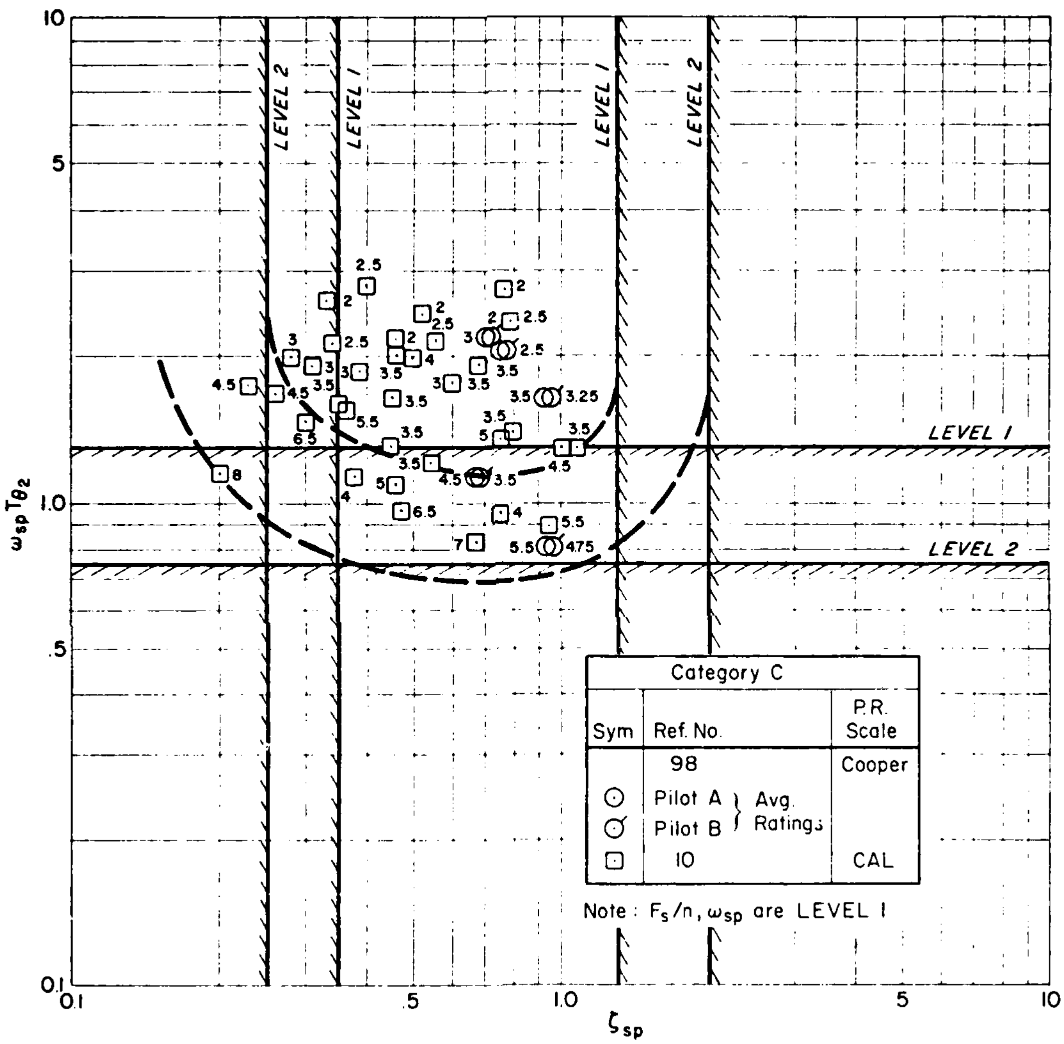


Figure 23 (3.2.1.1). Alternate Category C Short-Period Flying Qualities Requirements with more than 90 percent of the ratings. Note that these latter boundaries tend to eliminate combinations of low damping and low frequency. We have therefore elected to set the criterion boundaries based on the dashed lines in Figure 23. The minimum levels of ω_{sp} (independent of $\omega_{sp}T_{\theta_2}$) have been taken directly from the ω_{sp} vs. n/α criterion and are presented in a table in Figure 2c. The minimum levels of $1/T_{\theta_2}$ are based on the n/α limits in Figure 1c by assuming an approach speed of 135 kt and noting that $1/T_{\theta_2} \doteq (g/V)(n/\alpha)$.

2. Supporting Data — Augmented Airplanes

Feedbacks, crossfeeds, and feedforwards utilized for modification of aircraft response characteristics frequently result in higher-order non-classical control input transfer functions. At present there is no universally accepted method to evaluate higher-order system (HOS) effects on flying qualities. The first attempt to account for HOS effects in a specification (MIL-F-8785C, Reference 4) was the insertion of the word "equivalent" for classical short-period damping and frequency requirements, as well as addition of a maximum phase lag on the $n_{z_{pilot}}/F_s$ frequency response and inclusion of an allowable response time delay. Reference 80 analyzed a configuration whose flight control system shifted the short-period mode to a new location which was within Level 1. However, a low-order match to the high-order response indicated a sluggish system with an effective frequency well below the high-order root, which in fact was identical to the unaugmented short-period mode.

The variable-stability NT-33 results of DiFranco (Reference 103) and Neal and Smith (Reference 12) also clearly show that identifying a single mode from a high-order response is unsuitable.

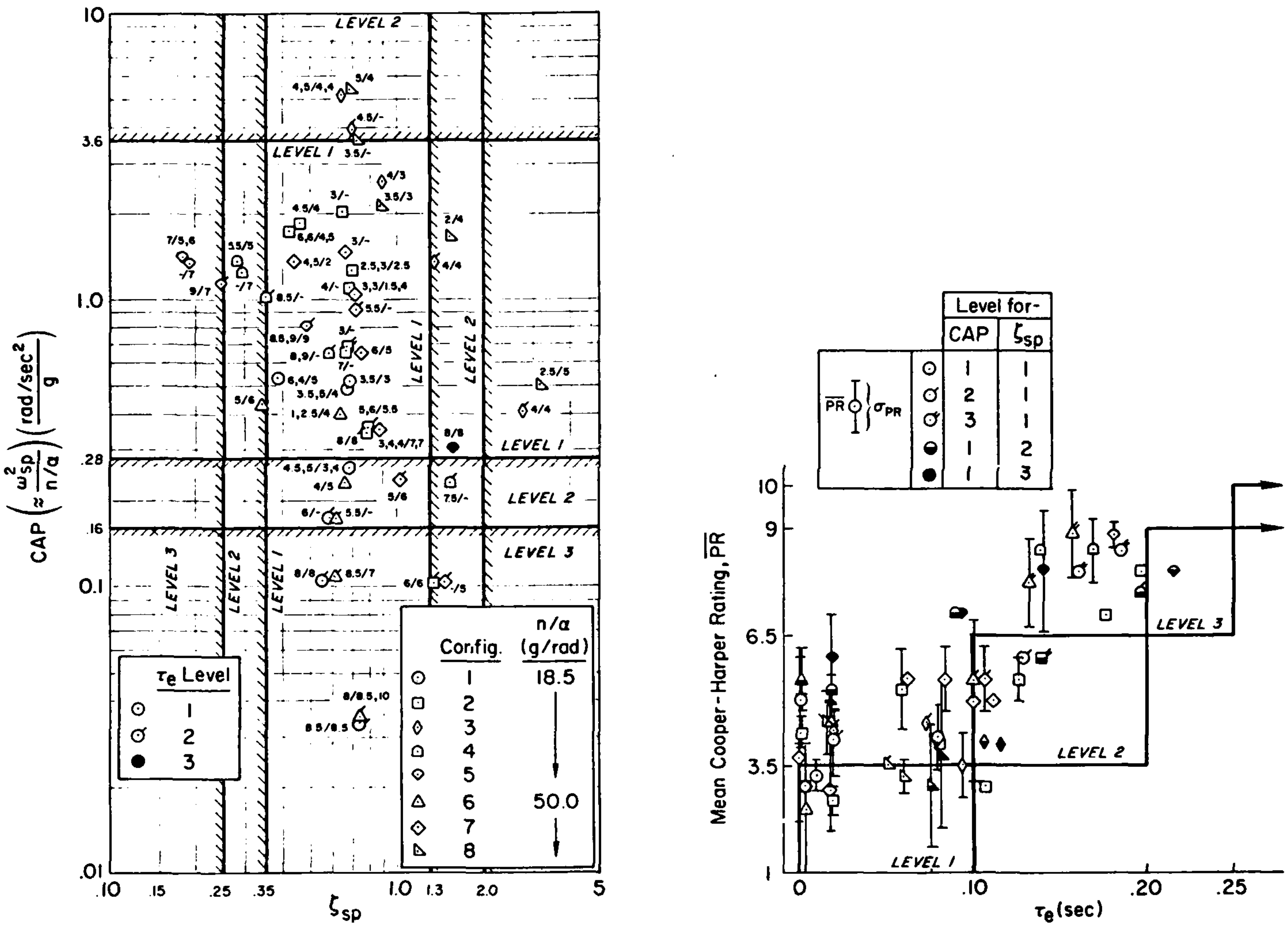
In this section, HOS and LOES will be examined and compared with the specification boundaries of Figure 1. The sources of the HOS data are two Calspan research efforts, by Neal and Smith (Reference 12) for Category A flight phases and by Smith (Reference 5) for Category C.

a. Neal-Smith Data (Reference 12)

The in-flight NT-33 experiments conducted by Neal and Smith represent a first look at generic variations typical of highly augmented aircraft. For evaluation of HOS characteristics and criteria, 51 separate FCS/short-period configurations were flown on the USAF/Calspan NT-33. Of these, some require qualifications: tests conducted at 250 kt ($n/\alpha \doteq 18.5$) were limited to a load factor of 2.5 g, due to buffet onset; Neal and Smith reported that the pilots did not fly these tests as aggressively as they did the high-speed (350 kt) tests.

Figure 24 shows the equivalent dynamics of the 51 Neal-Smith configurations (from Reference 83) and corresponding Cooper-Harper ratings. The frequency and damping ratio limits have been cross-plotted to facilitate presentation of the data. Figure 24a includes actual pilot ratings for each pilot; for those cases which have $\tau_e < 0.1$ (Level 1), correlation is quite good. There is clearly a relationship between τ_e and PR (Figure 24b), though the τ_e limits appear to be too lenient since many of the configurations which are predicted to have Level 1 ζ_{sp} and ω_{sp} have higher (poorer) ratings than predicted by τ_e alone. In Figure 24b only the mean pilot rating and standard deviation have been plotted, to reduce the number of data points.

A point-by-point comparison of the LOES data in Figure 24 shows that the flying qualities Levels are accurately predicted for about two-thirds of the configurations. (This requires some liberal interpretation; e.g., if a PR change of one-half rating would improve correlation, the configuration is assumed to fit the criterion. Such a PR variation is well within the range of normal ratings variations.) This correlation rate is not outstanding, but is close to that found for flying qualities data in general (for example, the data used in Reference 11 to define the ζ_{sp} and ω_{sp} boundaries of MIL-F-8785B). Several Level 2 rated configurations lie well within the Level 1 boundaries. However, the four configurations with $\zeta_e \doteq 0.4$ indicate that if the damping ratio lower limits were <u>increased</u> correlation would improve (see "Guidance for Application"). This is in contradiction with the results



a) ζ_{sp} and ω_{sp} Requirements

b) τ_e Requirements

Figure 24 (3.2.1.1). Comparison of LOES Dynamics With Short-Period Requirements; Category A, Neal-Smith (Reference 12) Configurations, MCAIR (Reference 83) Matches of the previous section, where a decreased ζ_{sp} limit would improve correlation for classical aircraft.

Not surprisingly, the $\omega_{sp}T_{\theta_2}$ criterion does not correlate the Reference 12 data any better (Figure 25). Correlation would be improved if the minimum damping for Level 1 were increased to 0.5 and if $\omega_{sp}T_{\theta_2}$ were increased to a value around 1.85. Both changes would be incompatible with the data base for classical unaugmented aircraft. The inconsistency between the unaugmented and augmented airplane data base needs to be further studied to determine if it is a fundamental characteristic of equivalent systems (see "Guidance for Application" for specific examples). Until such analyses can be conducted, we have elected to utilize the original MIL-F-8785C boundaries which are based on classical airplane data. However, for the purposes of guidance, according to the data the equivalent frequency and damping for Level 1 augmented airplanes should meet the following criteria for Category A flight conditions.

$$\omega_{sp}T_{\theta_2} > 1.85 \quad \text{or} \quad 3.6 > \frac{\omega_{sp}^2}{n/\alpha} > 0.37$$

$$\zeta_{sp} > 0.50$$

b. LAHOS Data (Reference 5)

A systematic evaluation comparable to the Neal-Smith study was conducted for HOS effects in landing approach (Category C). The Landing Approach Higher Order System (LAHOS) study (Reference 5) provides a good set of data for comparing LOES with Category C requirements.

Figure 26a compares LOES matches with $\omega_{sp}^2/(n/\alpha)$, and the LAHOS data are plotted against the allowable time delay (τ_e) requirements in Figure 26b. The LAHOS data correlate well with the boundaries. In fact, the flying qualities of about 85 percent of the LAHOS configurations are accurately predicted. The only area of poor correlation in Figure 26 involves those configurations which should have Level 1 flying qualities, but are rated by the pilots as Level 2. This may be in part a

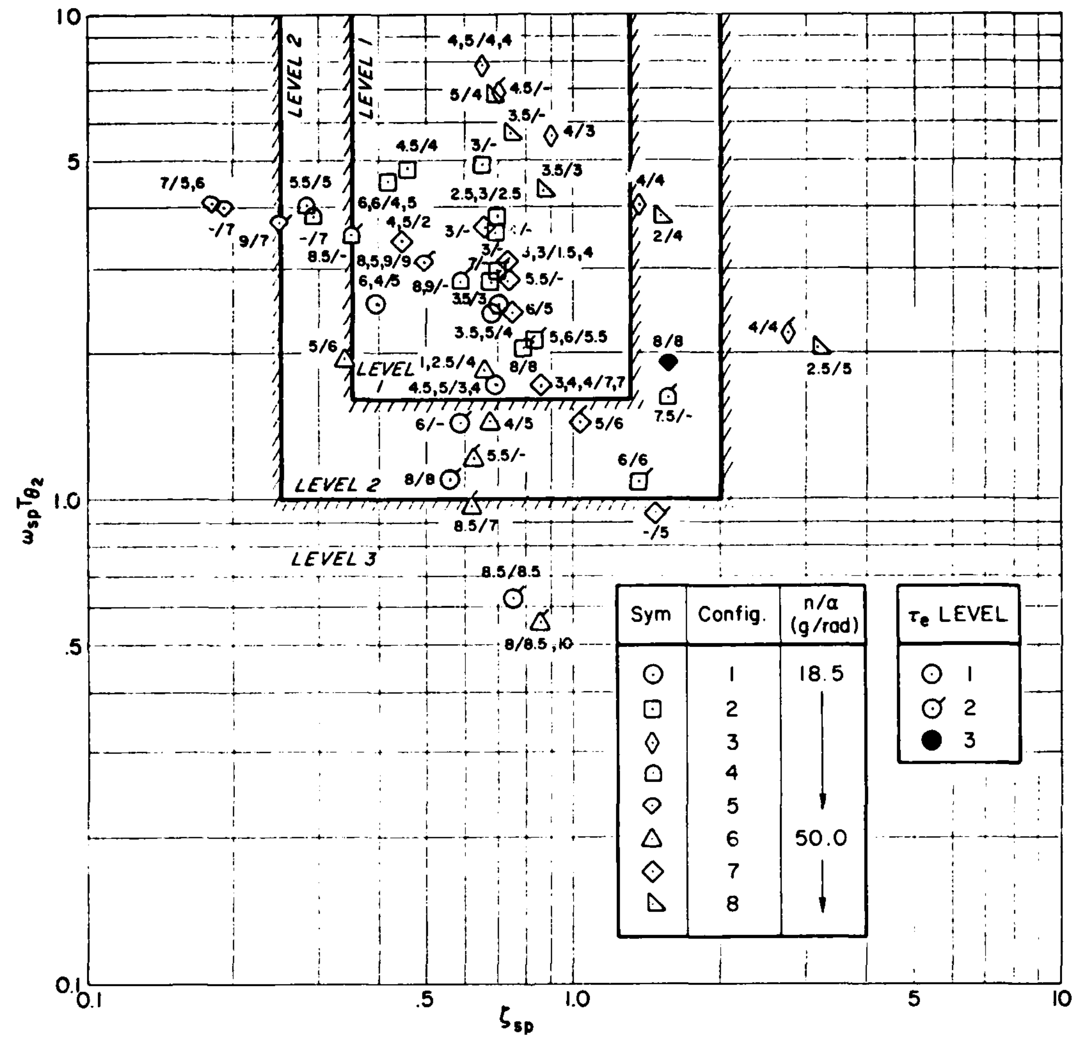
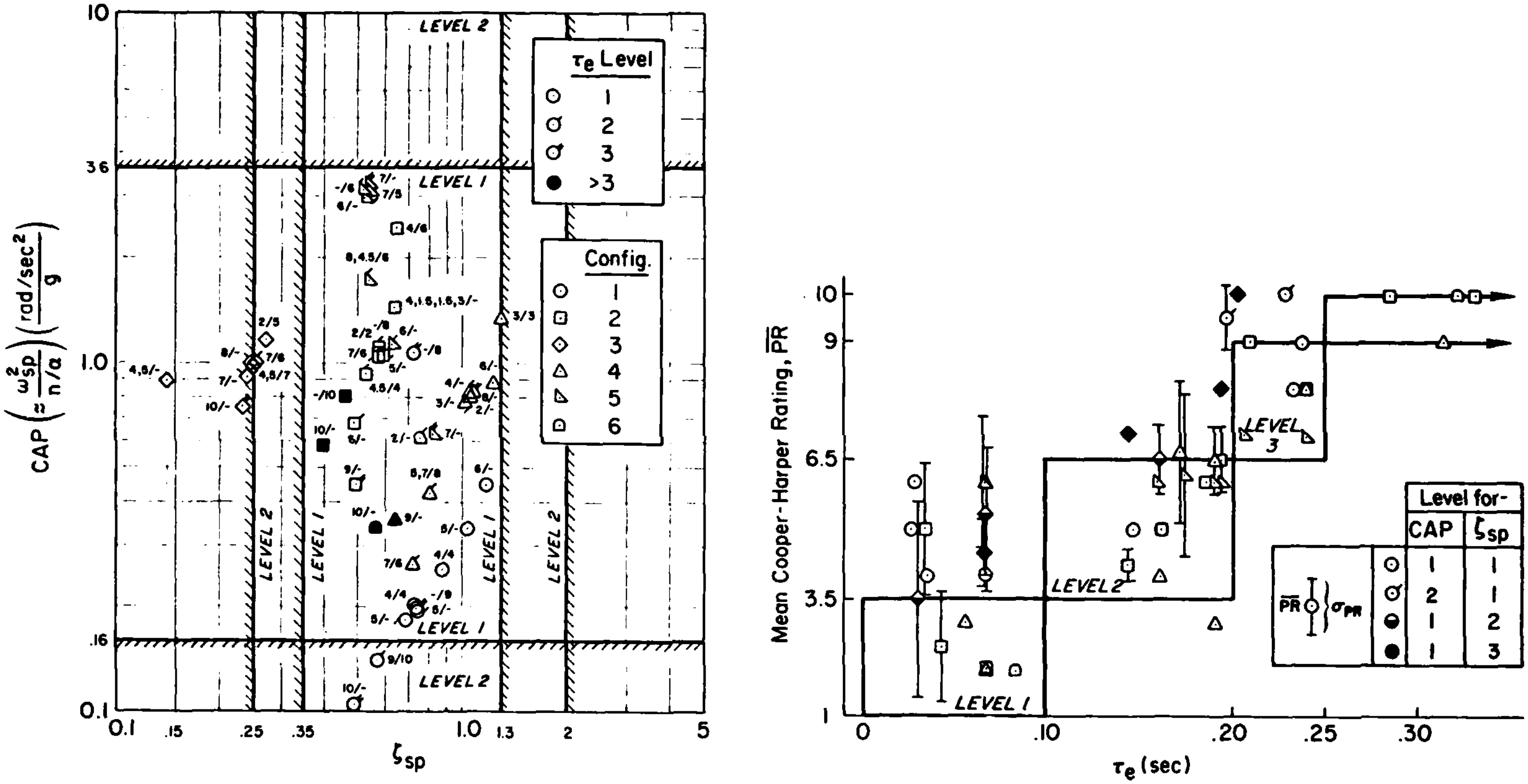


Figure 25 (3.2.1.1). Comparison of Neal-Smith LOES Characteristics with $\omega_{sp}T_{\theta_2}$ vs. ζ_{sp}



a) ζ_{sp} and ω_{sp} Requirements

b) τ_e Requirements

Figure 26. (3.2.1.1). Comparison of LOES Dynamics With Short-Period Requirements; Category C, LAHOS (Reference 5) Configurations, MCAIR (Reference 88) Matches

function of the fidelity of the tests and the realism of the tasks: a combination of instrument and visual approaches through touchdown and landing, or with intentional go-around maneuvers. Most of the classical data upon which the short-period requirements are based (see Reference 11) were generated for approach and go-around tasks only, seldom including actual landing, which is normally the most critical area. The LAHOS data may therefore be more representative of flying qualities in the terminal phases of flight. Reference 5 discusses this at some length.

One shortcoming of LAHOS is that the equivalent systems do not cover a wide range of ζ_{sp} and ω_{sp} (Figure 26a); these are Level 2 or worse for only nine of the 46 configurations. LAHOS is primarily an exercise of the τ_e limits (Figure 26b). (This is <u>not</u> a shortcoming of the LOES approach, but an artifact of the range of HOS evaluated in the LAHOS program).

Not surprisingly, $\omega_{sp}T_{\theta_2}$ vs. ζ_{sp} is very similar (Figure 27). As with the Category A data (Neal-Smith), considerable improvement in the correlations would be possible by increasing the ζ_{sp} limit to 0.50 and $\omega_{sp}T_{\theta_2}$ to 1.85. There is one data point to suggest a possible increase in $\omega_{sp}T_{\theta_2}$ to 2.2.

3. Large Airplanes (Classes II and III)

There have been frequent suggestions by manufacturers of large aircraft that the Level 1 lower requirements on ω_{sp} vs. n/α should be lowered in the landing approach. Calspan, in Reference 104, did not recommend this relaxation because data in the BIUG, Reference 11, and in Calspan's proposal to modify MIL-F-8785B, Reference 59, substantiate the original boundary quite well. Informal discussions with pilots of large aircraft indicate that current large airplanes possess generally comfortable bandwidth for routine use. However, when the task difficulty is increased due to weather conditions, for example, hard landings and go-arounds are common. In view of the very demanding landing conditions being imposed on large military aircraft such as the YC-14, YC-15, and C-X, relaxation of the requirement seems imprudent until more substantiating data become available.

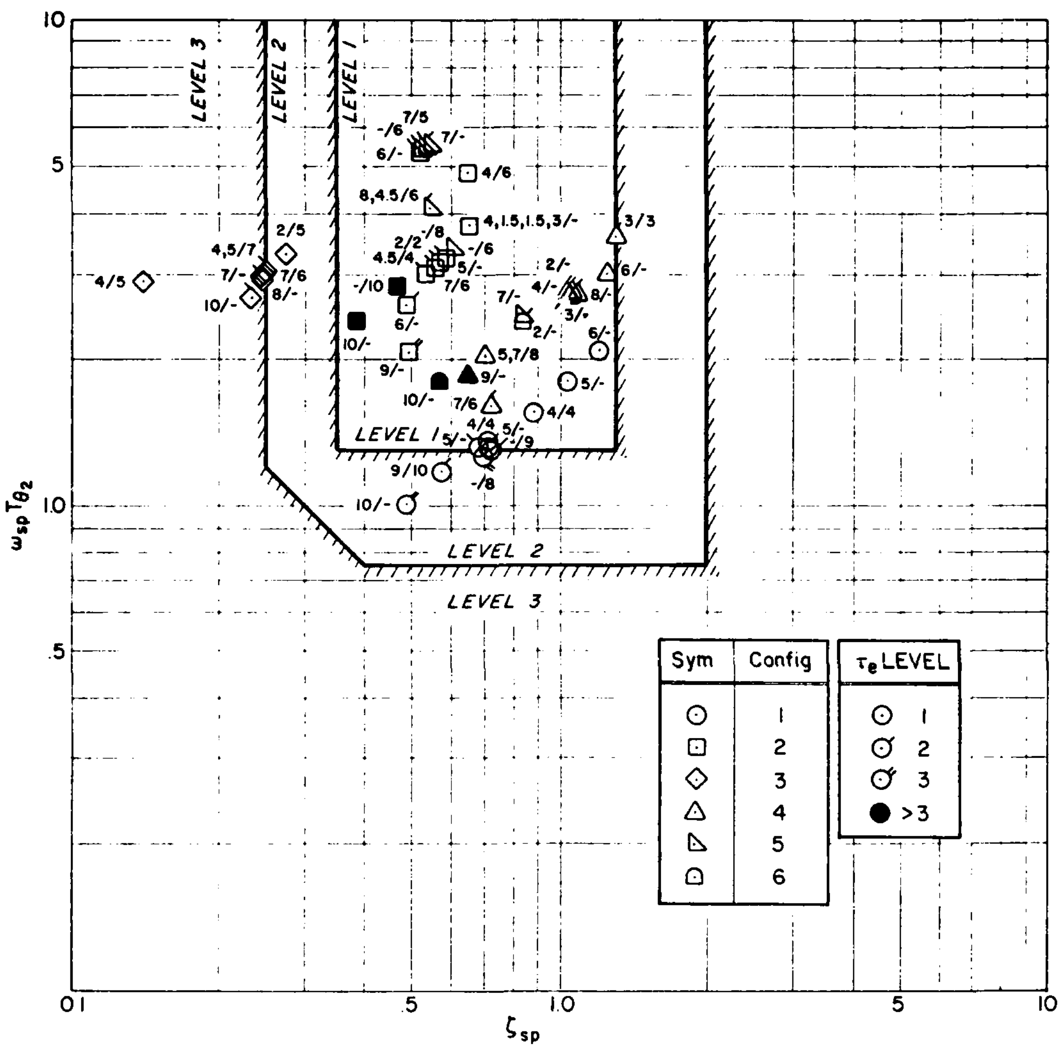


Figure 27 (3.2.1.1). Comparison of LAHOS LOES Characteristics with $\omega_{sp} T_{\theta_2}$ vs. ζ_{sp}

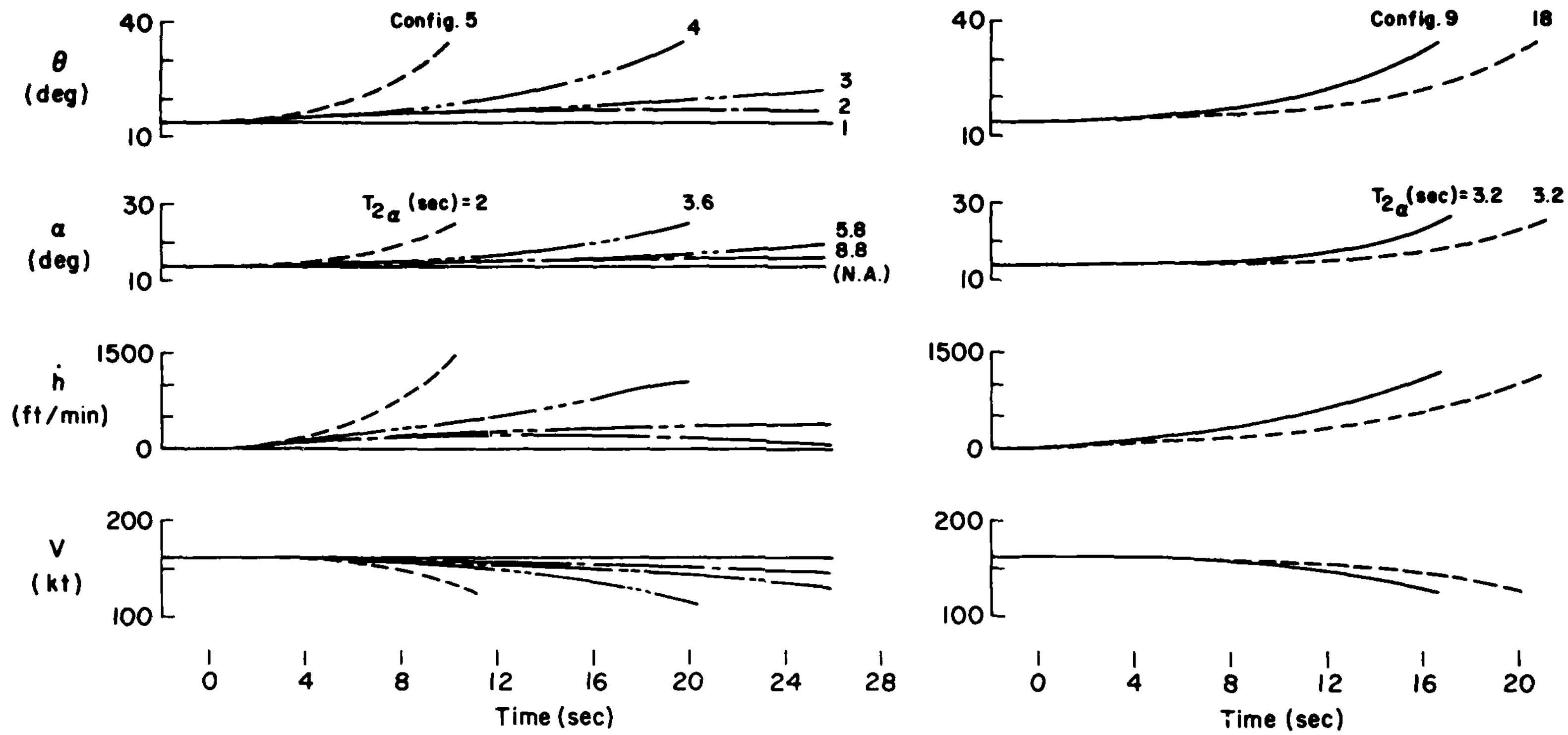
4. Level 3

An aperiodic divergence with a time to double amplitude of 6 sec is allowed for Level 3 flying qualities. Substantiation for the requirement was given in Reference 122. The essence of that discussion is repeated below.

In response to a pulse control input, stable aircraft reach new steady values of α, $\dot{h}$ and V; unstable aircraft have the same initial response, then diverge, as illustrated by Figure 28 (from Reference 178). For a supersonic transport design, impulse responses are shown for various degrees of static instability as $C_m(\alpha)$ was varied. Also shown is the response of a configuration having much more static instability, with time to double amplitude reduced by a pitch damper. Evaluation pilots rated both of these configurations unacceptable, but termed the latter's characteristics insidious. From Reference 179, commenting on an F9F-2 airplane with static instability ameliorated by a pitch damper to give about 6 seconds to double amplitude:

> The rate of divergence of the airspeed was scarcely noticeable to the pilots in normal flying. However, this degree of instability might be objectionable for flight operations where accurate control of airspeed is required.

From Reference 95, pilot tolerance of aperiodic instability is much greater than of oscillatory instability (Figure 29). In that variable-stability YF-86D evaluation, an aperiodic divergence was not considered safe with less than 1 sec to double amplitude: "there was a dangerous situation in that a short distraction of the pilot's attention could allow the unstable vehicle to diverge to the point that it was difficult to recover." For statically stable configurations "the unacceptable boundary is close to the zero damping boundary over most of the frequency range...in the very low-frequency and very high-frequency ranges a small amount of positive damping is required to remain within the acceptable region." Commenting on this different tolerance, Taylor and Day (Reference 182) state:



a) Comparison of Linear $C_m(\alpha)$ Configurations 1-5

b) Effect of Increased Pitch Damping with Same T_2. ($C_{m_q} + C_{m_{\dot{\alpha}}}$) for Config. 18 was 5 Times That of Config. 9

Figure 28 (3.2.1.1). Comparison of Effects of Various Stability Characteristics on Airplane Response to Elevator Pulse (-5 deg for 0.2 sec at t=0) (Reference 178)

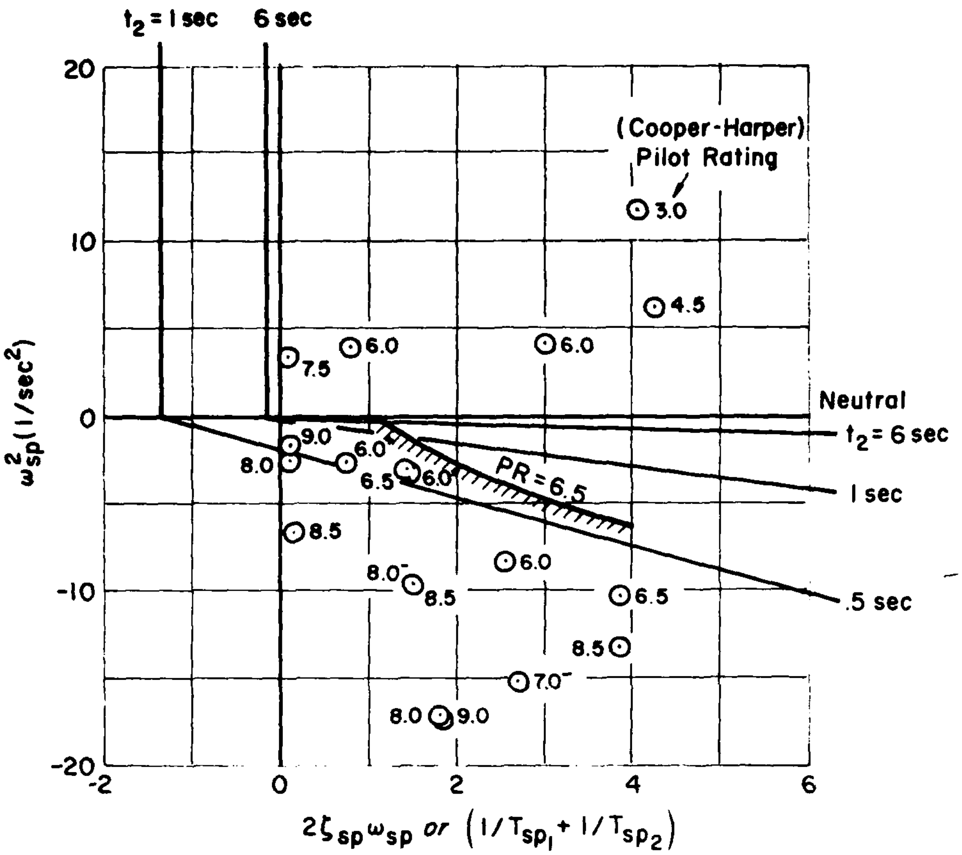


Figure 29. (3.2.1.1). Contours of Constant Pilot Opinion in Statically Unstable Region; Constant Stick-to-Stabilizer Gearing (Reference 95)

> At the higher frequencies, the technique for controlling the motion was not learned as quickly.... Controlling the pure divergence in the region of a static instability was more natural and less tiring than controlling the oscillatory airplane motions, inasmuch as the pilot need only to counteract the angle-of-attack divergence without leading the motion to stabilize the aircraft.

The unchanged phugoid requirement, $T_2 > 55$ seconds for Level 3, still limits the low-frequency tolerable oscillatory instability (the α, q, and n_z feedbacks used in these variable-stability airplanes would not suppress the phugoid mode in the region of low short-period frequency and damping). Higher-frequency oscillatory instabilities are unlikely, requiring considerable negative aerodynamic damping; the limit of 6 seconds to double amplitude would fit the Level 3 boundary of Reference 95 for $0 < \omega_n < 6$ rad/sec.

For aperiodic instability, Reference 180 shows that the boundary of acceptability "for emergency condition" (Cooper 6.5) was insensitive to the value of lift-curve slope, or $1/T_{\theta_2}$ or n/α, for positive lift-curve slopes. This boundary value was 2 seconds to double amplitude.

Reference 181 demonstrates that at least at low speeds, the "short-period" approximation can give a grossly incorrect value of T_2. The T_2 obtained from the angle-of-attack trace matched the theoretical value well when C_{m_α} was actually linear. References 178 and 181 both elaborate on the range of values for time to double amplitude obtained by different means: calculation from three-degree-of-freedom equations and various simplifications, measurement from α, θ or V responses. M_α nonlinearities gave different results for nose-up and nose-down perturbations; of course the worst direction would govern, for all reasonable magnitudes. Most of the evaluations gave some consideration to turbulence. The Reference 181 baseline configuration had a Level 2 value of $d\gamma/dV$, but zero values were included in the evaluation -- with a little improvement in rating, but less noticeable in turbulence. The evaluations considered both visual and instrument flight.

On the basis of all these considerations, 6 seconds to double amplitude seems a reasonable, safe limit. Operators may be well advised to give pilots of potentially unstable airplanes some flight simulator experience with such instability. It should be noted that pitch attitude and airspeed will double in amplitude (with respect to their trim values) at approximately the same rate since $\dot{u} \doteq g\theta$. Hence the allowable divergence in attitude is the same as airspeed response to attitude (Para. 3.4.1).

It is desirable, though impractical at this time, to make the allowable instability a function of time. Clearly an instability in cruise, where it might be hours before a runway is available, could be very tiring to the pilot.

H. LESSONS LEARNED

Three major lessons have emerged from recent work on equivalent systems:

1) There are sufficient parameters in the equivalent system models to allow correlation with flying qualities problems of the very high-order systems which have so far been designed for operational aircraft.

2) Of these equivalent parameters, large equivalent delays are highly correlated with pilot-induced oscillation tendencies.

3) Succumbing to the temptation to add complexity to the flight control system can easily degrade, rather than improve, the handling qualities.

The second lesson, though evident in the in-flight simulation data of DiFranco, Neal and Smith, and LAHOS (References 103, 12, and 5 respectively) has also been learned the hard way. The Tornado experience described by Gibson in Reference 105 was discussed in Reference 106 as follows:

> [The Tornado description] is a rare example of a type of paper that should be encouraged. In this paper the airplane designer admits that his airplane, equipped though it is with a full authority fly-by-wire flight control system, turned out to have serious flying qualities problems that required solutions. The example is rare not because problems occurred, but because the designer was willing to report on the experience. In fact, similar problems (pitch PIO in landing caused by control system phase shift and roll PIO caused by high roll control gain) have been experienced in highly augmented aircraft designed in the USA such as the YF-17, YF-16, F-18, and Space Shuttle.

In the Tornado example, the problem was related to excessive pitch command gains and high-frequency filters (i.e., large τ_e). Richards and Pilcher (Reference 107) give a frank discussion of PIOs (lateral in this case) encountered when the demanding task of shipboard landing was first evaluated with an early F-18 version containing excessive equivalent delay.

An important lesson learned from both the Tornado and F-18 experience is that the pilot-induced oscillations due to equivalent delay, or phase shift, though pronounced, can be very isolated. Lateral PIOs occurred in two of the 49 carrier landings performed with the F-18. Considerable flight experience had been accumulated on the Tornado before the hard landing reported by Gibson.

Differences between ground-based simulation and in-flight characteristics appear inherent in experience with the above aircraft. Presumably all these aircraft were simulated on ground equipment during the design, and their problems only appeared later in flight. The differences, seen in the early results of DiFranco (Reference 103) and Parrag (Reference 108) have also been the subject of some recent study. Figure 30 illustrates some differences between pilot ratings for various equivalent delays in various simulations. The figure is from Reference 84, which summarizes the lessons learned:

> Pilot rating degradation due to equivalent delays is often far more serious in flight than on a ground-based simulator.... Most of the data show a threshold in pilot rating degradation due to delay followed by a fairly linear increase in the rating.
>
> The Navion in-flight results [References 109, 110] form both extremes of the data, i.e., producing the most immediate degradation due to delay (for lateral dynamics) and also the least ultimate degradation (for longitudinal dynamics).
>
> The MCAIR ground-based data are similar to the F-8 low stress landings of Berry, et al. [Reference 111]. The F-8 high stress landing data closely approach the NT-33 longitudinal landing data [References 5, 91] and the NT-33 lateral landing data [References 82, 91]. A general trend of rating versus delay can be inferred....However, there is much to be learned about lags and equivalent delay effects.

In the "Supporting Data" section an example of the value of equivalent systems was shown for an augmented airplane. A more recent application of LOES provides even stronger support. An emergency backup control system, for the USAF AFTI/F-16 power approach and landing, was designed with a pitch rate feedback to the horizontal tail. Figure 31 illustrates the control system, and shows the θ/δ_e transfer function for

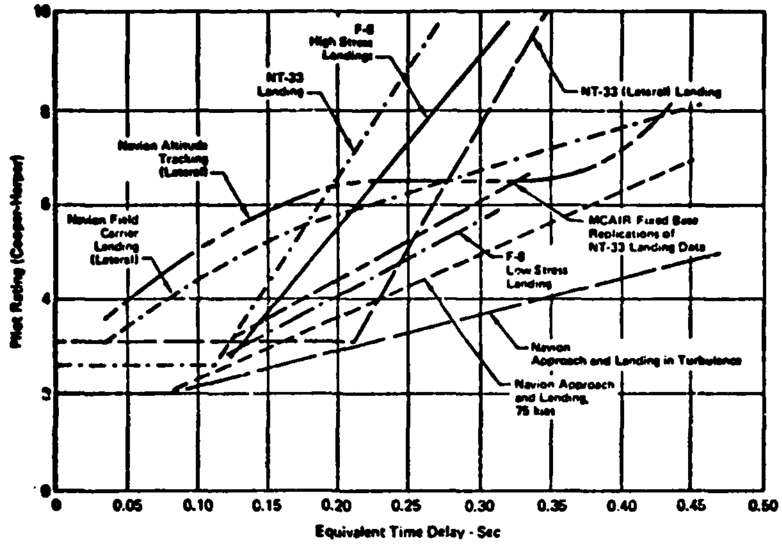


Figure 30 (3.2.1.1). Comparison of Equivalent Delay Effects in Pitch or Roll Rate Response to Stick Force for Different Simulations (from Reference 84)

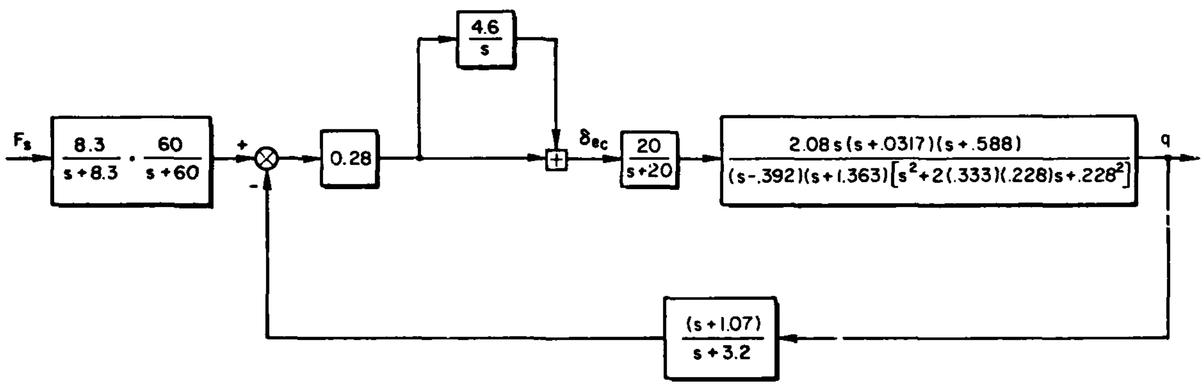


Figure 31 (3.2.1.1). AFTI/F-16 Independent Back-Up Pitch Rate Feedback Block Diagram the unaugmented AFTI/F-16 in the power approach (129 kt, 13.2 deg α). The feedback is intended to stabilize the short-period, which consists of two first-order modes for the basic airplane. As Figure 32 shows, the augmented airplane has two well-damped second-order modes, and the "short-period" mode (resulting from the coupled pitch rate lag and $1/T_{sp_2}$) is well within Level 1 limits for damping ratio and frequency.

However, when this system was simulated and flown on the USAF/Calspan NT-33, it received Cooper-Harper pilot ratings of 8 and 9, and was considered extremely sluggish with very heavy control forces.

An equivalent system match of the θ/F_s transfer function (see Figure 33) clearly shows why the airplane was Level 3: equivalent ω_{sp} = 0.685 rad/sec (with n/α = 3.9) is on the boundary between Levels 2 and 3 on Figure 1c; and τ_e = 0.186 is Level 2 by Table 3. What appeared to be an adequate augmentation (Figure 31) results in an airplane that is not much better

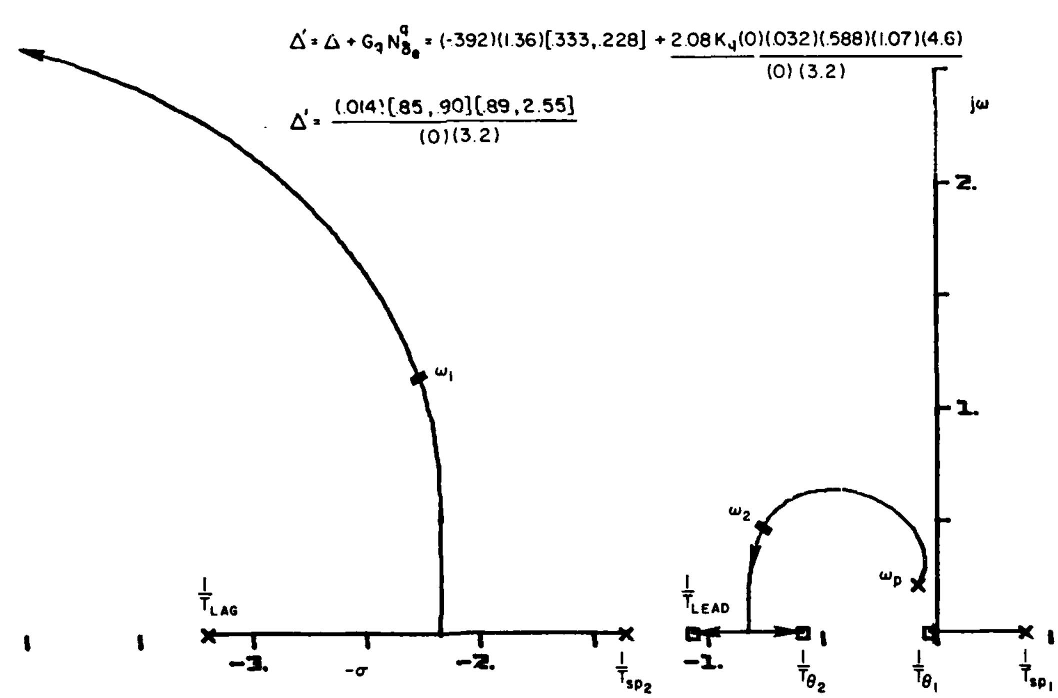


Figure 32 (3.2.1.1). AFTI/F-16 $q \rightarrow \delta_e$ Feedback (IBU)

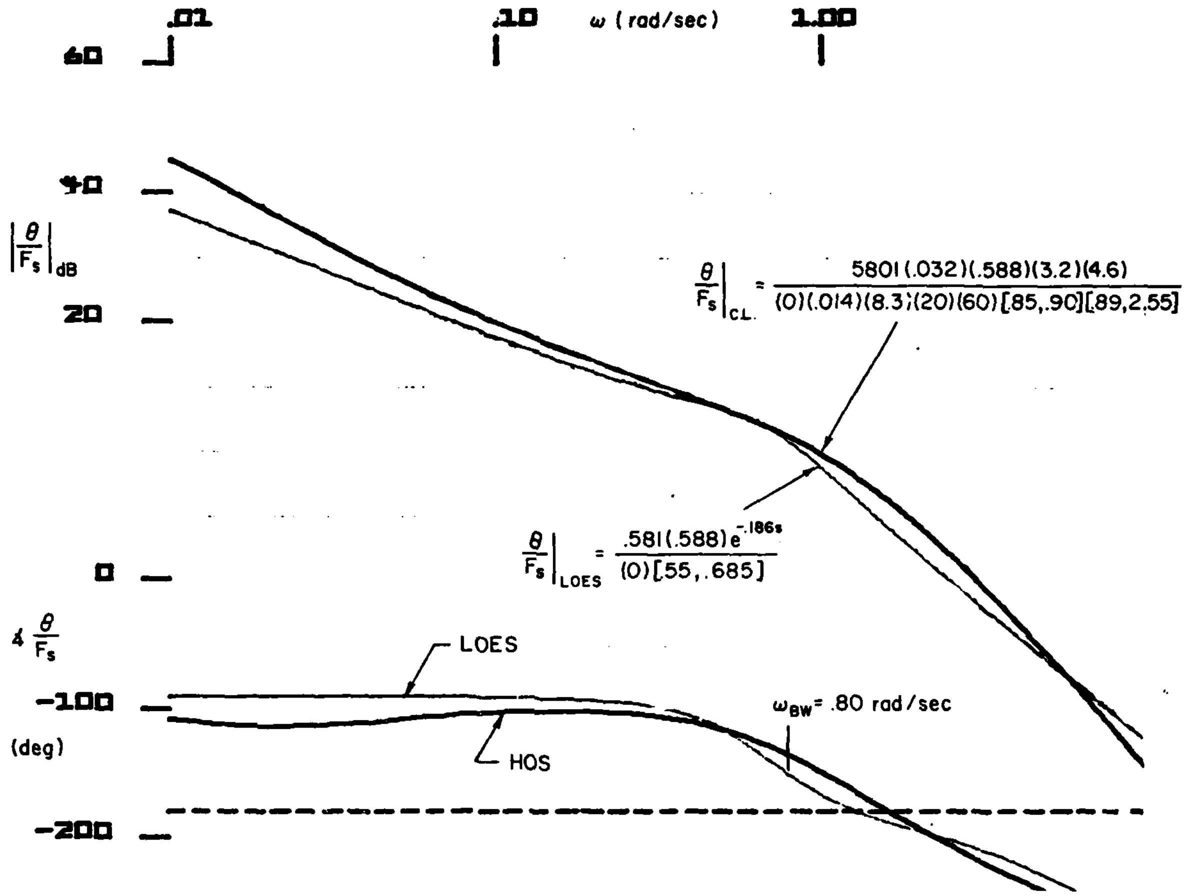


Figure 33 (3.2.1.1). AFTI/F-16 $\theta \div F_s$ for IBU
($q \div \delta_e$ Closed)

than the basic F-16. Note that the equivalent short period is lower than either ω_1 or ω_2 in Figure 32 — an illustration of the fact that it is incorrect to pick a "dominant root" to plot on the Figure 1 boundaries.

Possible Modifications to CAP

As noted by Bischoff (Reference 268), the control anticipation parameter must be redefined for aircraft with effective time delay since $\ddot{\theta}(0) = 0$ in this case. Following DiFranco (Reference 9), Bischoff defines, on the basis of a unit step stick force input, a more general control anticipation parameter, CAP′, as

$$CAP' \equiv \frac{\ddot{\theta}_{max_{HOS}}}{n_{z_{ss}}} \tag{24}$$

where the maximum pitch acceleration, $\ddot{\theta}_{max_{HOS}}$, will occur sometime after the force input as shown in Figure 34. CAP' is further extended to the short period lower order equivalent system model by defining

$$CAP'_e \equiv \left(\frac{\omega_{sp}^2}{n/\alpha}\right)_e \frac{\ddot{\theta}_{max_{HOS}}}{\left(\ddot{\theta}_{LOES}\right)_{t=\tau_e}} \tag{25}$$

where the first factor in parentheses is defined from the LOES parameters. This factor alone does not give a good approximation to CAP' because the short period LOES model will not generally be accurate in the high frequency region which largely determines the initial pitch acceleration history. Thus the second factor is required where $\left(\ddot{\theta}_{LOES}\right)_{t=\tau_e} = A_\theta$ (from the LOES model) and $\ddot{\theta}_{max_{HOS}}$ is determined numerically from the HOS response (such as in Figure 34).

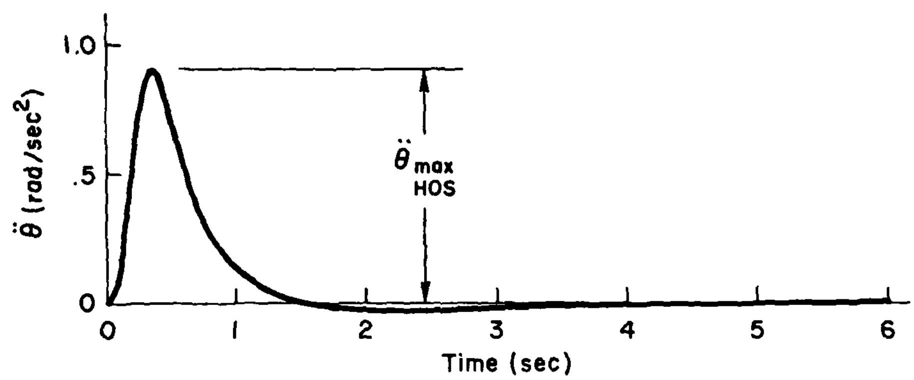


Figure 34 (3.2.1.1). Pitch Acceleration Response to a Unit Step Force Input

Reference 268 accounts for time delay explicitly by defining flying qualities Levels in the CAP′ - τ plane, as shown in Figures 35 and 36. The boundaries shown for each flying quality Level were defined by correlations of data from DiFranco (Ref. 9), Neal and Smith (Ref. 12), and the LAHOS study (Ref. 5). These boundaries do seem to correlate the data slightly better than is achieved using the present requirement based on CAP (compare Figures 24 and 26). However, the CAP'_e parameter is subject to all the limitations for equivalent systems noted under "Guidance for Application" in this section. Hence most of the points that do not correlate with CAP or $\omega_{sp}T_{\theta_2}$ will also be missed by CAP'_e, and are missed by CAP′. The bandwidth specification of Para. 3.2.1.2 appears to do a somewhat better job than CAP′.

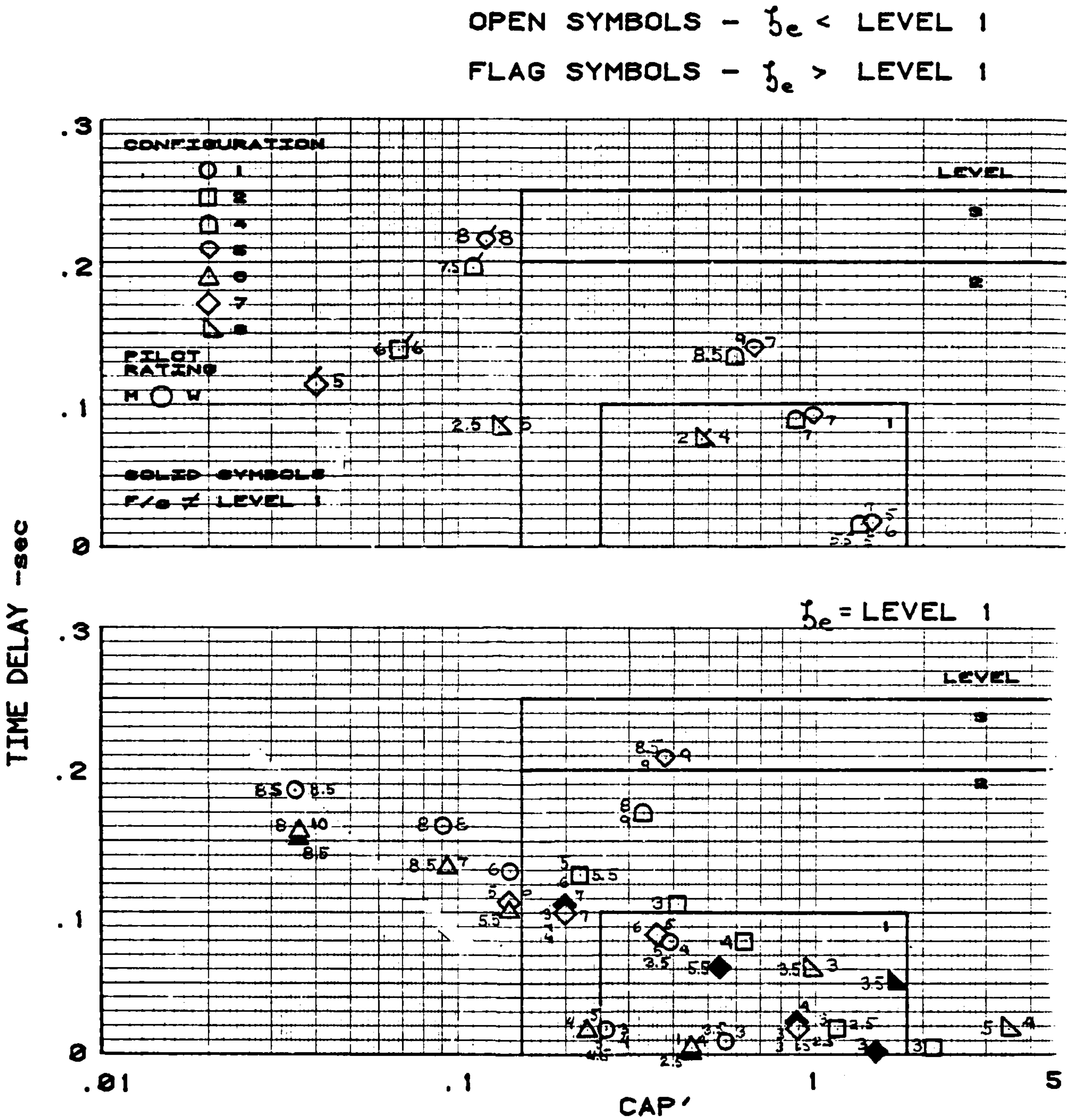


Figure 35. Time Delay versus CAP' - Neal-Smith Data
(from Reference 268)

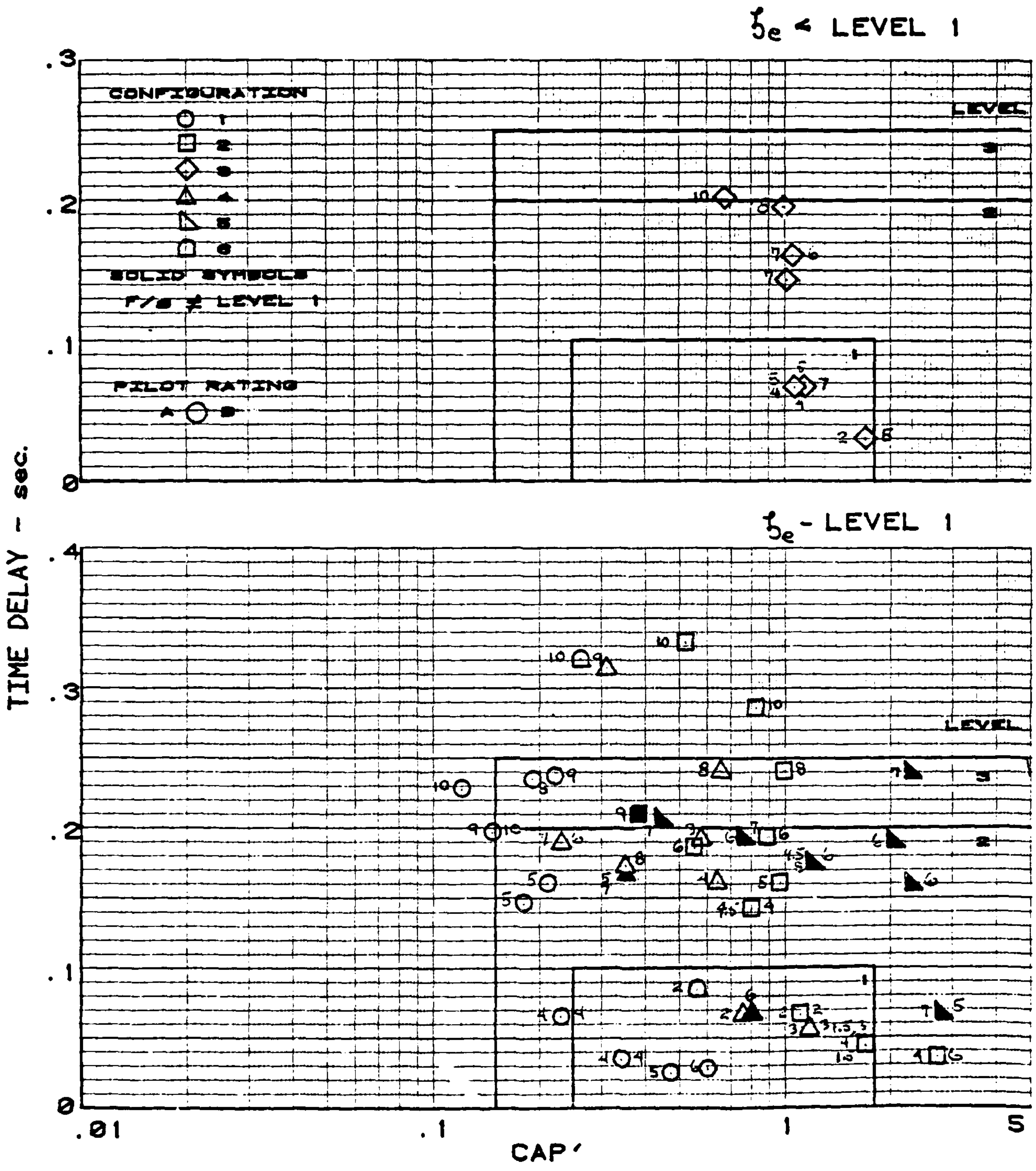


Figure 36. Time Delay versus CAP' - LAHOS Data
(from Reference 268)

3.2.1.2 Pitch axis bandwidth requirements

A REASON FOR REQUIREMENT

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 value of bandwidth will have superior performance when regulating against disturbances. A bandwidth criterion is especially useful for highly augmented aircraft where 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 should be utilized when the mismatch between the lower-order and higher-order systems exceeds the values defined in Figure 6 (3.2.1.1). No assumption of pilot dynamics is necessary in applying this requirement, since any such assumption would simply shift the boundaries.

B. RELATED MIL-F-8785C REQUIREMENT

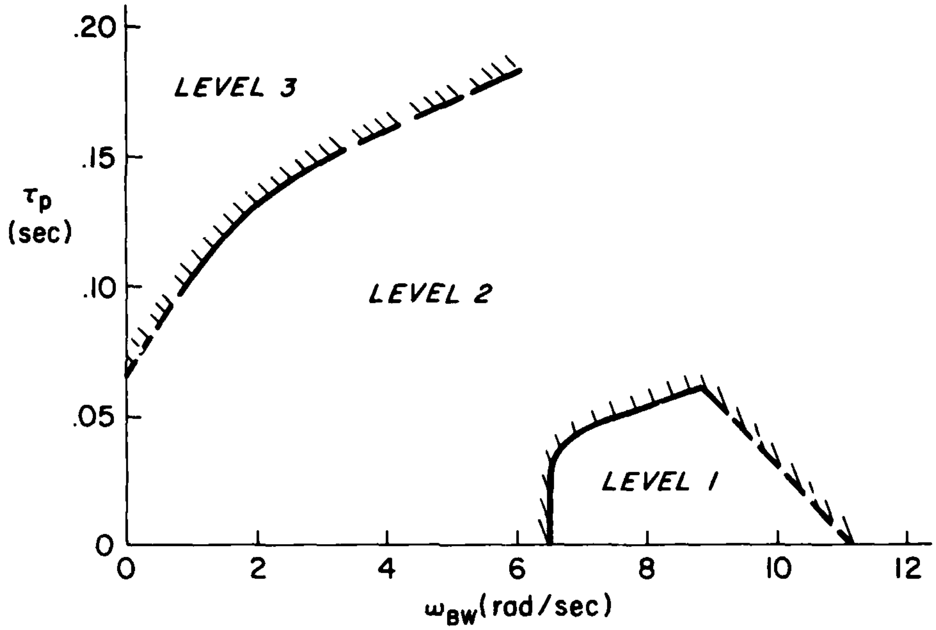
None.

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

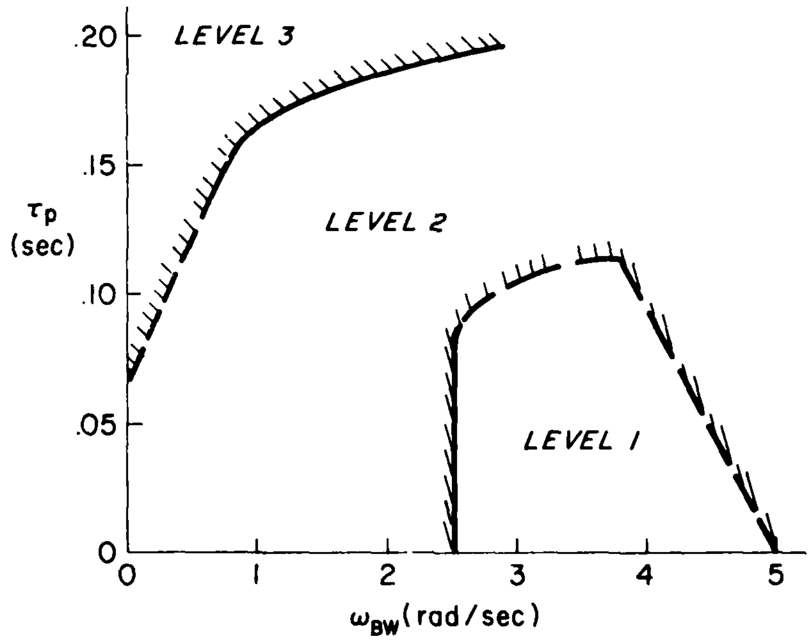
3.2.1.2 Pitch axis bandwidth requirements. The bandwidth of the open-loop pitch attitude response to pitch controller shall have the following characteristics: ________________.

Recommended limits for the pitch attitude bandwidth are given as a function of the parameter τ_p (defined in "Rationale Behind Requirement") in Figure 1 for Categories A and C. No recommendations for Category B are made at this time.

In addition, any long-period mode (corresponding to the classical phugoid) should meet the equivalent phugoid requirements of 3.2.1.1.



a) Category A Flight Phases



b) Category C Flight Phases

Figure 1 (3.2.1.2). Bandwidth Requirements

D. RATIONALE BEHIND REQUIREMENT

The concept of using bandwidth is not new. A recent utilization of bandwidth was in the Neal-Smith criterion (see Reference 12). This criterion consists of an empirical plot of the closed-loop pitch attitude resonance $|\theta/\theta_c|_{max}$ vs. pilot equalization for a piloted closure designed to achieve a <u>specified</u> bandwidth. Experience with this criterion has shown that the results can be sensitive to the selected value of closed-loop bandwidth. The criterion developed herein utilizes the maximum value of bandwidth achievable without threatening stability, thereby removing the necessity for selecting a value for ω_{BW} a pri[illegible].

Another criterion utilizing bandwidth was suggested in [illegible]erence 141. This criterion also selected a fixed value of ban[illegible]th (1 rad/sec for power approach). It utilized the phase margin, ϕ [illegible] slope of the phase curve, $d\phi/d\omega$, at the selected bandwidth freque[illegible] as correlating parameters. Again, experience has shown that the fixed value of bandwidth limited application of the criterion.

Most, if not all, familiar handling quality metrics are, in fact, related to bandwidth. However, these metrics tend to apply for classical aircraft which can be characterized by lower-order systems. For example, the short-term pitch response of a classical aircraft is well represented by the familiar short-period approximation

$$\frac{\theta}{F_{es}} \doteq \frac{M_{F_{es}}\left(s + 1/T_{\theta_2}\right)}{s\left(s^2 + 2\zeta_{sp}\omega_{sp}s + \omega_{sp}^2\right)} \tag{1}$$

It is easily shown for this (and similar) transfer function(s) 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 (unstable) plane. As illustrated by the generic sketches in Figure 2 for an idealized pilot supplying only 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 Figures 2a and 2b). The

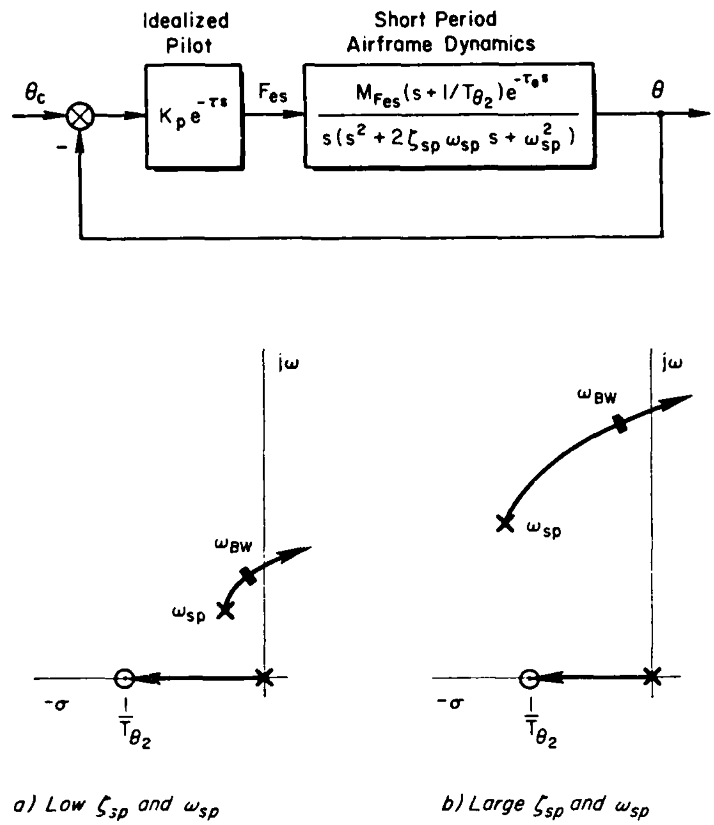


Figure 2 (3.2.1.2). Simplified Pilot Vehicle Closure for Pitch Control

aircraft of Figure 2 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 in Figure 2). It can be seen from Figure 2 that ω_{BW} depends uniquely on ω_{sp}, ζ_{sp}, $1/T_{\theta_2}$, and τ_e. Hence, these familiar flying quality metrics are, in fact, a measure of bandwidth. Again we make the point that bandwidth is <u>not</u> a new idea.

The present impetus for using ω_{BW} as a criterion evolved from attempts to develop a flying quality specification for aircraft utilizing unconventional response modes with direct force controls (wings-level turns, pitch pointing, etc.) (Reference 115). 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 involves θ/θ_c, a closed-loop describing function of pilot/vehicle response. Here, however, there is no assumption about a pilot model, and "bandwidth" is specified in terms of the aircraft-alone gain and phase margins.

The bandwidth as defined for handling quality criterion purposes is the frequency at which the phase margin is 45 deg or the gain margin is 6 dB, whichever frequency is lower (Figure 3). Referring to Figure 2, this describes the pilot's ability to double his gain or to add a time delay or phase lag without causing an instability. In order to apply this definition, one first determines the frequency for neutral stability from the phase portion of the Bode plot (ω_{180}). The next step is

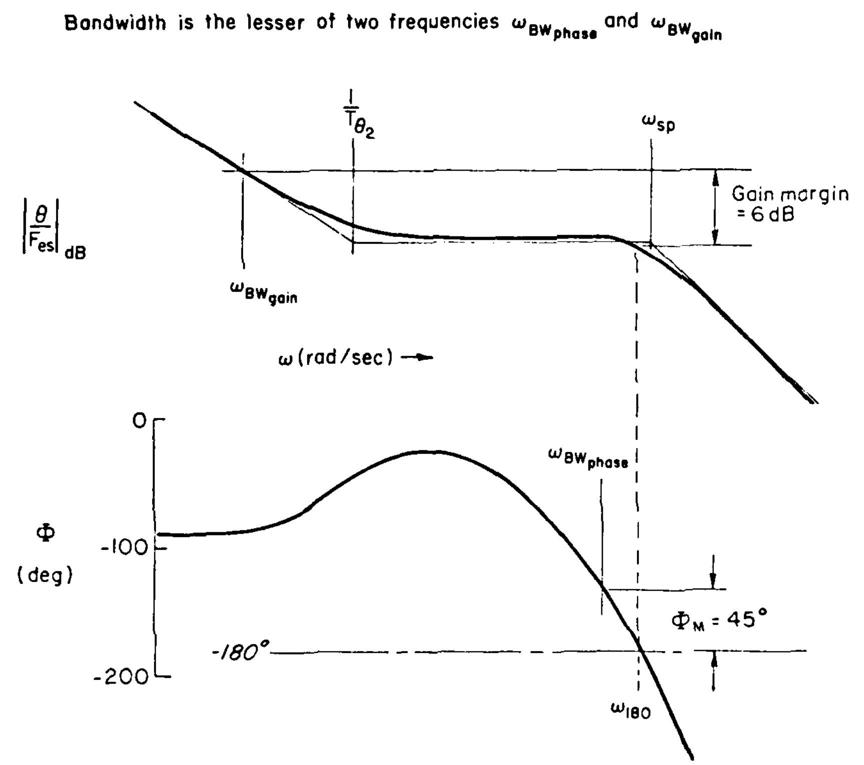


Figure 3 (3.2.1.2). Definition of Bandwidth Frequency, ω_{BW} From Open Loop Frequency Response to note the frequency at which the phase margin is 45 deg (ω_{135}). This is the bandwidth frequency as defined by phase, $\omega_{BW_{phase}}$. Finally, note the amplitude corresponding to ω_{180} and add 6 dB. Find the frequency at which this value occurs on t: amplitude curve; 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 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 in Figure 3.

Several sets of data were correlated with bandwidth using the above definition. A typical result is shown in Figure 4 utilizing the data from Reference 12. While there is a definite pilot rating trend with ω_{BW}, the scatter for bandwidths between 2 and 6 rad/sec does not allow a

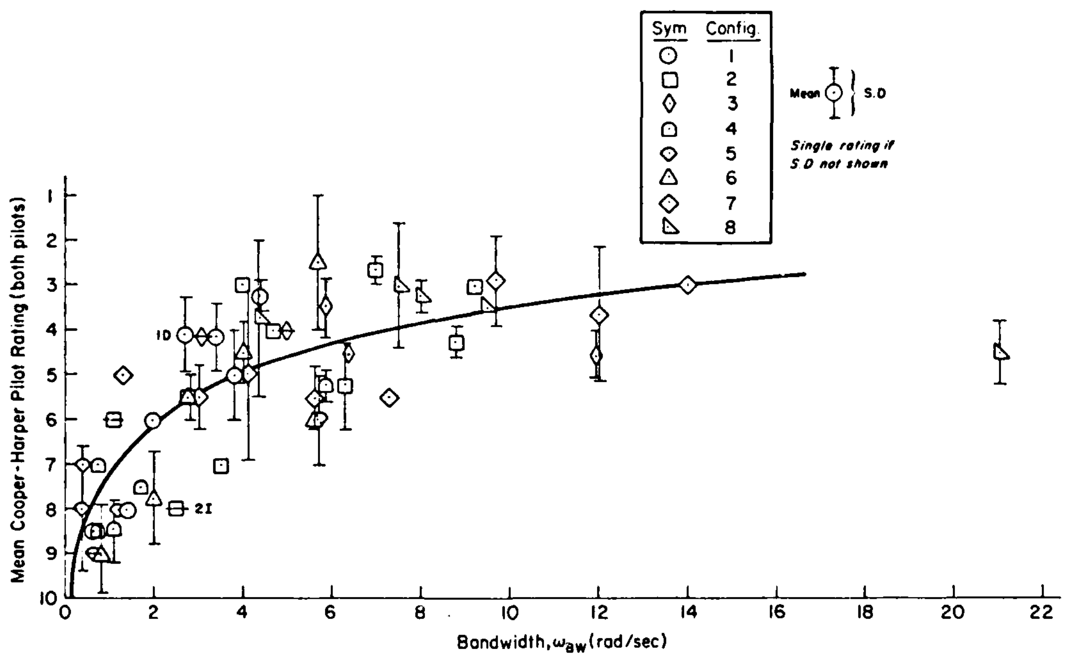


Figure 4 (3.2.1.2). Comparison of Neal-Smith Data (Ref. 12) With Bandwidth (Mean Ratings) quantitative definition of flying quality levels. A detailed analysis of the pilot/vehicle closure characteristics was made for Configurations 1D and 2I. This was done to determine why these two configurations with nearly equal ω_{BW} would have such a large difference in pilot ratings (4 and 8 respectively). The detailed pilot/vehicle closures are shown in Figures 5a and 5b. The value of bandwidth is seen to be about the same for both cases. However, if the pilot were to track very aggressively by further increasing his gain (to operate at frequencies above ω_{BW}), Configuration 1D would only be unstable for very high pilot gains, whereas 2I would rapidly become unstable (compare the root loci in Figures 5a and 5b). This behavior is predictable from the phase curves. In particular, Configuration 1D has a phase curve that rolls off very gradually at large values of frequency, whereas the phase for 2I drops off rapidly as the frequency is increased above ω_{BW}. It is not surprising that this case (2I) received a poor pilot rating (PR = 8) considering that attempts at aggressive tracking result in a closed-loop divergence. Hence, we have evidence that the ability of the pilot to attain good closed-loop regulation without threatening stability depends not only on

1) The value of bandwidth, ω_{BW},

but also on

2) The shape of the phase curve at frequencies above ω_{BW}.

rapid rolloffs in phase are well represented by a pure time delay, $e^{-j\omega\tau}$. Accordingly, both of the key factors noted above will be accounted for by plotting pilot rating data on a grid of ω_{BW} vs. τ. This is done for the Reference 12 data (which were plotted versus ω_{BW} alone in Figure 4) as shown in Figure 6. The scatter is seen to be considerably reduced and the data are reasonably well separated into Level 1, 2, and 3 regions. The values of τ used in this plot were obtained from lower-order equivalent system fits of the higher-order system

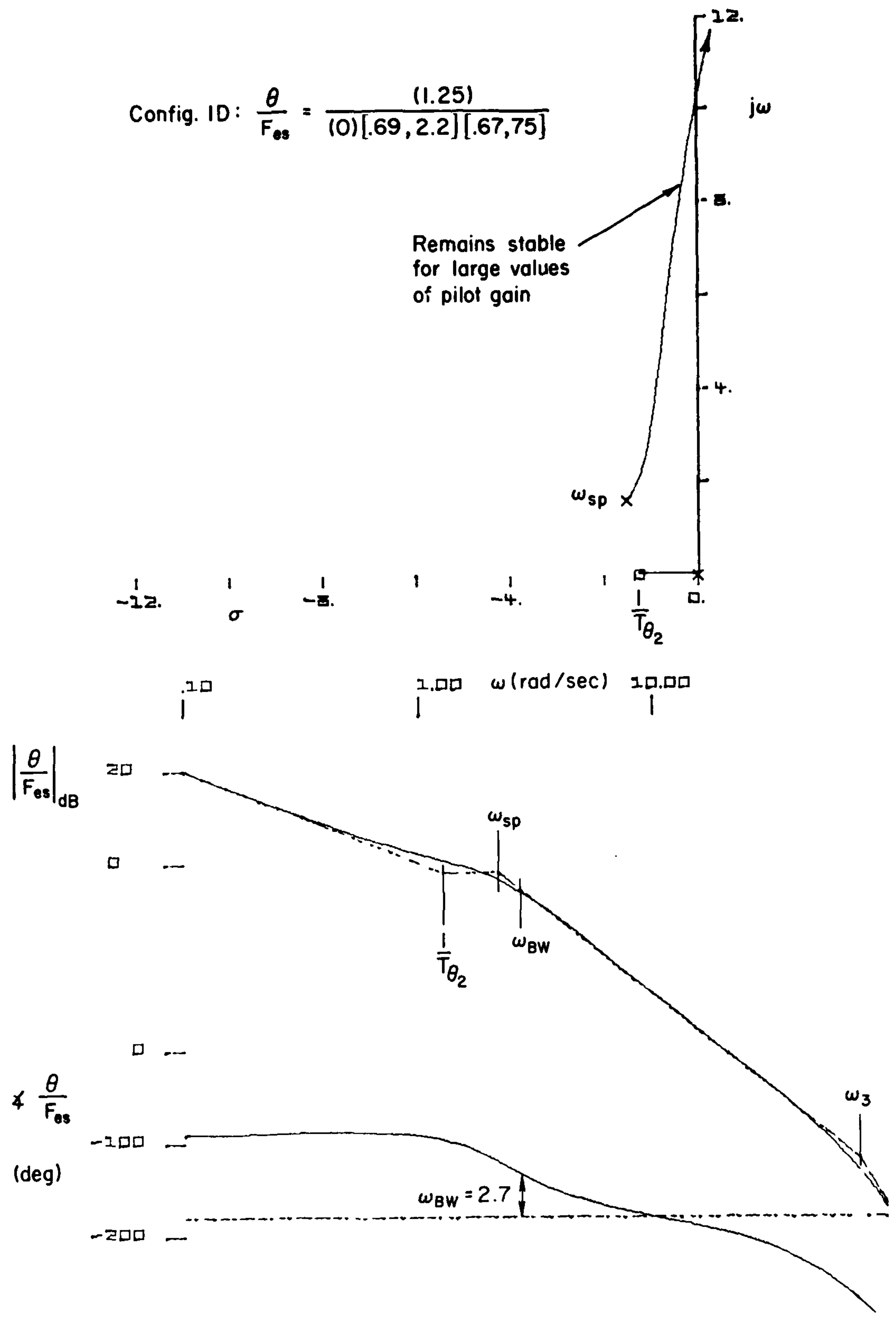


Figure 5a (3.2.1.2). Level 1/2 System of Neal-Smith (1D):
ω_{BW} = 2.7 rad/sec, Mean PR = 4.1

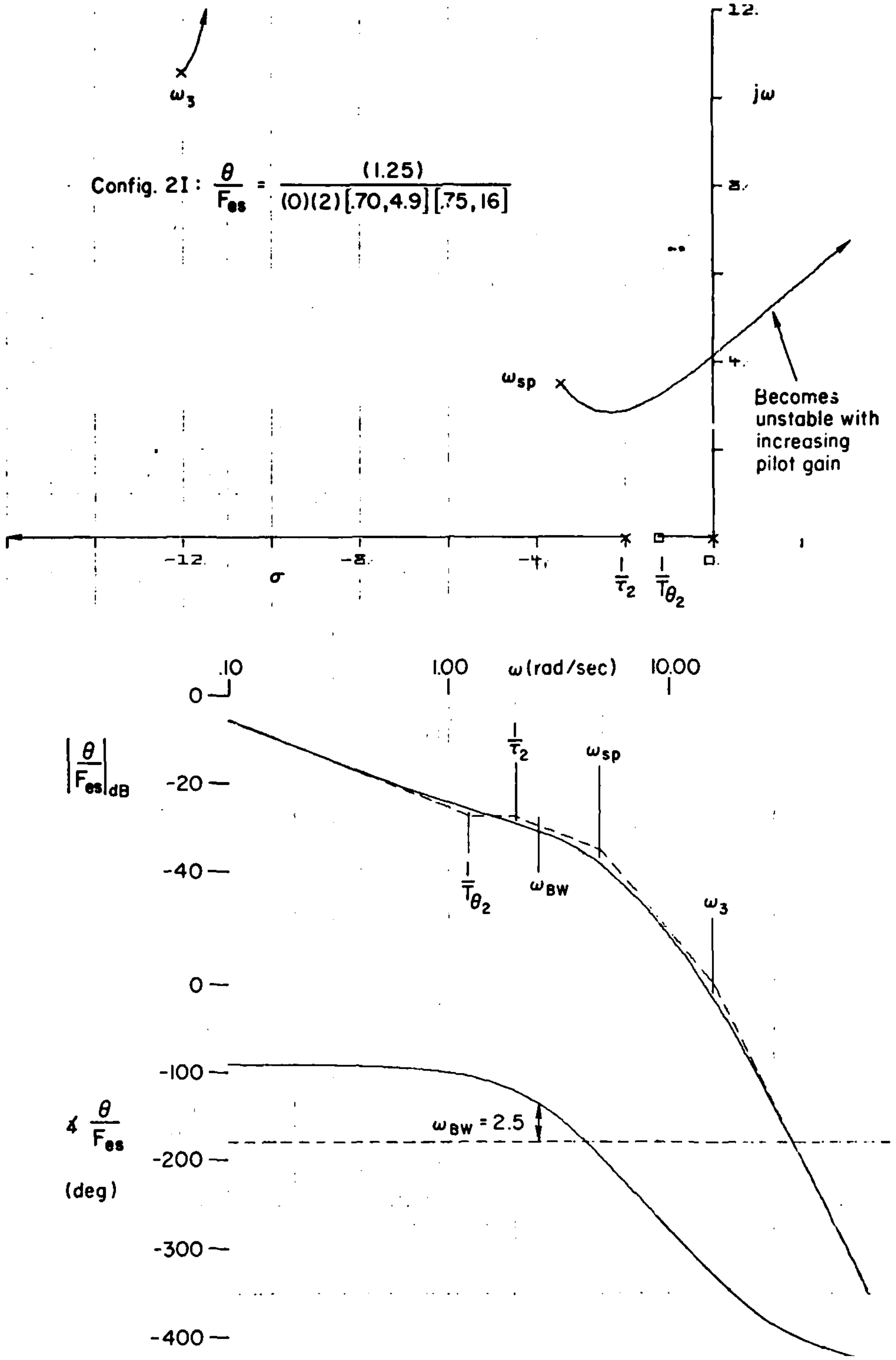


Figure 5b (3.2.1.2). Level 3 System of Neal-Smith (2I); ω_{BW} = 2.5, Mean PR = 8.0

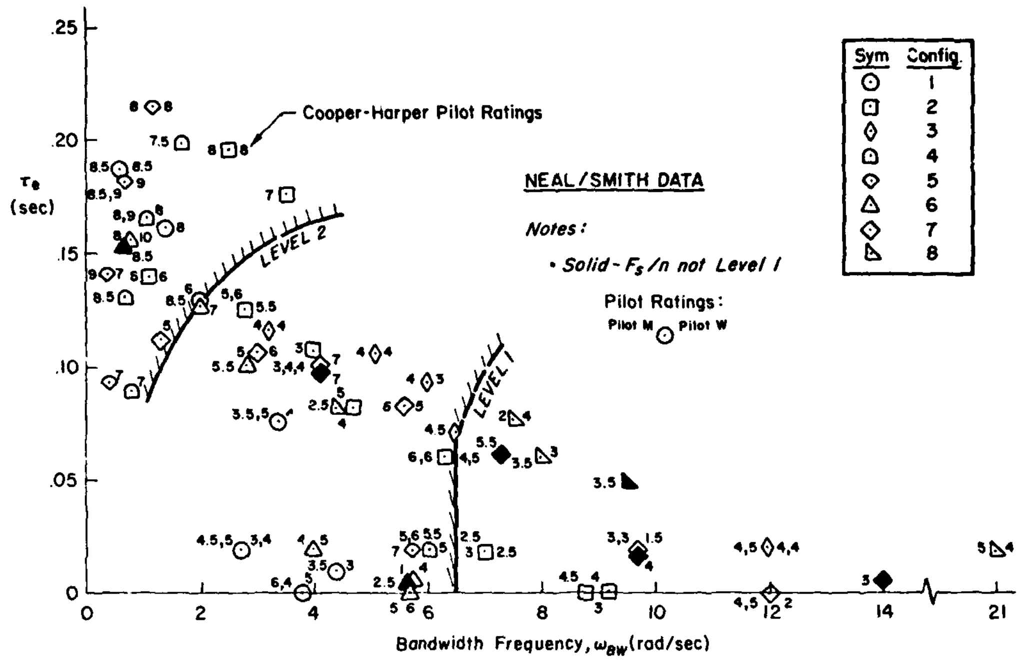


Figure 6 (3.2.1.2). Correlation of Pilot Ratings with ω_{BW} and τ_e (Ref. 12 Data)

transfer functions (Reference 83). The lower-order equivalent system form was:

$$\frac{\dot{\theta}}{F_s} = \frac{(s + 1/T_{\theta_2})e^{-\tau_e s}}{s^2 + 2\zeta_e\omega_e s + \omega_e^2} \tag{2}$$

The zero $1/T_{\theta_2}$ was fixed at the aircraft value (see discussion in 3.2.1.1). But this criterion is intended to avoid the need for an equivalent system match, so a workable and much simpler approach is to note that the change in phase due to a time delay is a linear function of frequency, i.e., $\Delta\phi = \tau\omega$. To the extent that the rolloff in phase beyond -180 deg can be attributed to τ_e in Eq. 2, we can estimate τ_e in the vicinity of some frequency ω_1 (and associated phase ϕ_1) from:

$$\tau_p = -\frac{\phi_1 + 180^\circ}{57.3\omega_1} \tag{3}$$

where ω_1 is some frequency greater than the frequency for neutral stability* and the symbol τ_p represents the estimate of τ_e. Correlations between τ_e and τ_p for the combined References 12 and 5 data resulted in a correlation coefficient of 0.96. Thus, there is very good evidence that τ_p can be used in place of τ_e in Figure 6, as will be shown in "Supporting Data."

E. GUIDANCE FOR APPLICATION

ω_{BW} and τ_p are easily obtained when the frequency responses are available. However, the frequency responses themselves must be obtained from analysis, simulation or flight test data -- as in the case of the Reference 115 flight test of Direct Force Control modes. In that program it was found that excellent frequency responses could be obtained by Fast Fourier Transforming flight test data. In particular, pilot-generated frequency sweeps worked very well. A typical frequency sweep and the resulting Bode plot are shown in Figures 10 (3.2.1.1) and 11 (3.2.1.1), respectively. The instrumentation required to obtain this data was minimal, consisting of a yaw rate gyro and a pedal position transducer. Nonetheless, the data must be manipulated (via a Fast Fourier Transform computer program), which is in a sense less desirable than reading parameters off a time response.

Responses that are gain-margin-limited tend to have shelf-like amplitude plots as shown in Figure 7. With such systems a small increase in pilot gain results in a large change in crossover frequency and a corresponding rapid decrease in phase margin. The decrease in phase margin becomes critical for attitude control when τ_p is moderately large (of order 0.1 to 0.2). The two configurations shown in Figure 7 are taken from the Reference 5 experiment. Applying the previously discussed definition of bandwidth, we find that both Configurations 5-6 and

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*ω_1 was taken as twice the neutral stability frequency, i.e., $\omega_1 = 2\omega_{180}$. Hence,

$$\tau_p = -(\phi_{2\omega_{180}} + 180^\circ)/(57.3 \times 2\omega_{180})$$

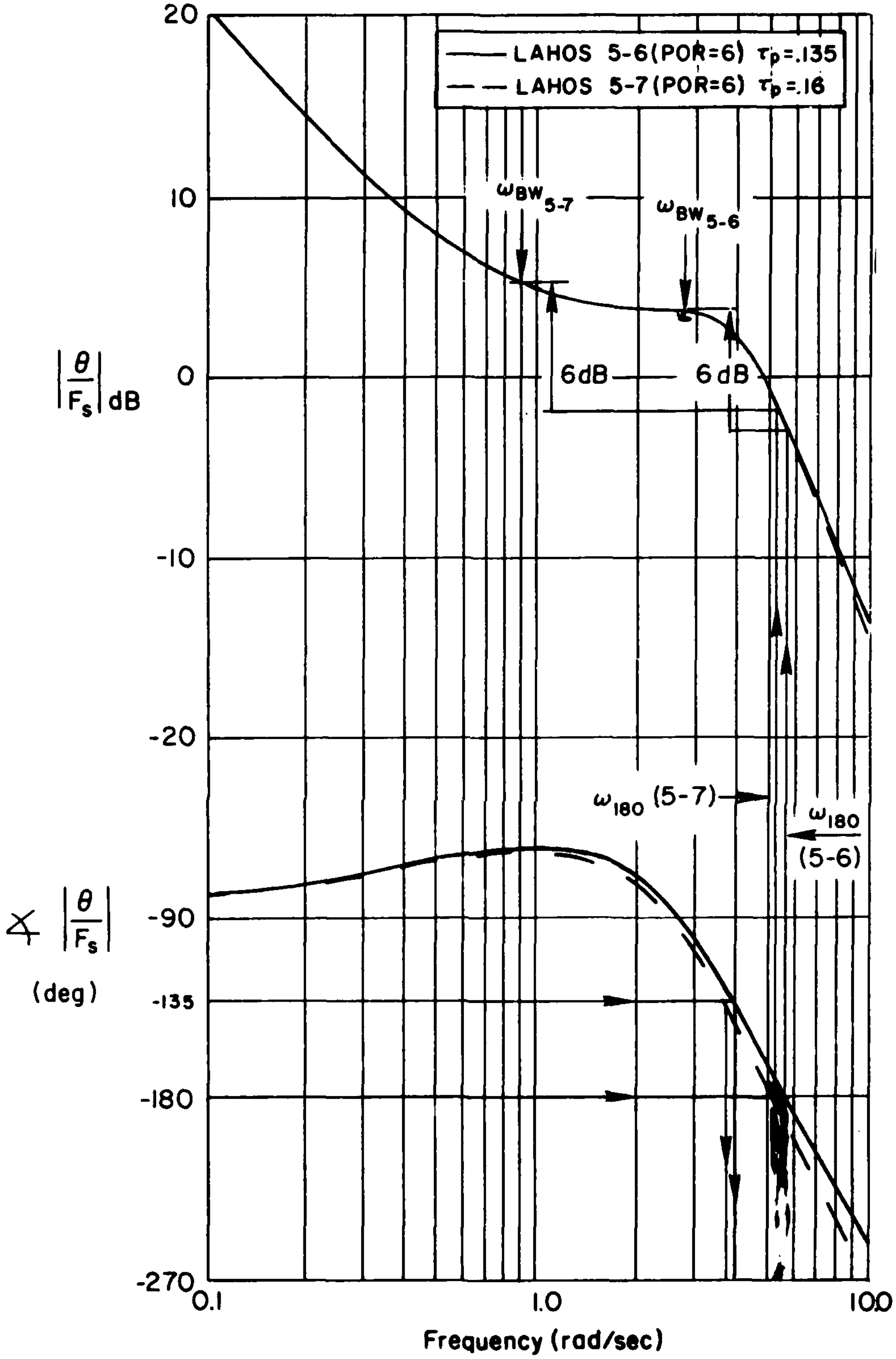


Figure 7 (3.2.1.2). Large Difference in Bandwidth Due to Shelf in Amplitude Plot Combined with Moderate Values of τ_p (configurations of Reference 5)

5-7 are gain-margin-limited. Both configurations suffer from the same deficiency, i.e., moderate values of τ_e combined with a shelf-like amplitude curve that results in a very rapid decrease in phase margin with small changes in pilot gain. However, the 6 dB limit selected to define $\omega_{BW_{gain}}$ does not "catch" Configuration 5-6. While this configuration is correctly predicted to be Level 2 (PR = 6) on the basis of τ_p, the value of ω_{BW} is in the Level 1 region. Had a slightly higher value of gain margin been picked to define ω_{BW}, the bandwidths for Configurations 5-6 and 5-7 would be approximately equal. However, because of the nature of shelf-like frequency responses, there will always be a case which can "fool" the criterion. An experienced handling qualities engineer would immediately recognize the shelf-like shape and moderate τ_p as a significant deficiency. However, the purpose of a criterion is to eliminate such judgment calls. Nonetheless, it is not expected that this idiosyncrasy will result in problems with correlating or predicting pilot rating data inasmuch as moderate (Level 2) values of τ_p are required to get misleading values of ω_{BW} (i.e., rapid phase rolloff in a frequency region where the amplitude curve is flat must occur to get the effect shown in Figure 7).

F. DEMONSTRATION OF COMPLIANCE

The values of τ_p and ω_{BW} required to demonstrate compliance with the Figure 1 boundaries are obtained from open-loop frequency responses of pitch attitude such as those shown in Figures 3, 5, and 7. These plots may be obtained from analyses (Figure 5) or from Fourier-transformed flight test or simulator data such as was shown in Figure 10 (3.2.1.1). 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 should be accomplished with different input magnitudes. Data taken during operational tasks will implicitly account for non-linear effects if technically good data can be obtained.

G. SUPPORTING DATA

The data from Neal-Smith (Reference 12) are compared with the bandwidth Category A requirements in Figure 8. These results are reasonably encouraging, though there are a number of Level 2 ratings at high values of bandwidth. The abbreviated pilot comments (taken from References 12 and 160) indicate that abruptness and oversensitivity become a problem when ω_{BW} is large. This was especially true of the Reference 160 pilot ratings (given in parentheses in Figure 8). A possible boundary on ω_{BW} is shown in Figure 8 to account for this problem. This boundary is

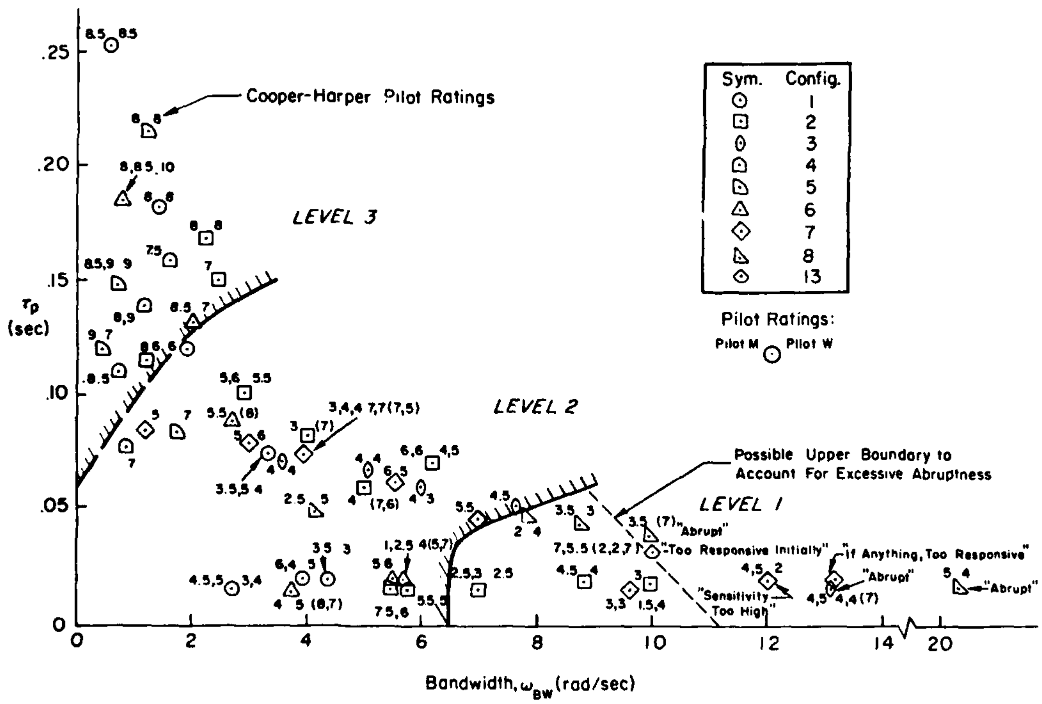


Figure 8 (3.2.1.2). Correlation of Pilot Ratings with ω_{BW} and τ_p for Neal-Smith Data (Category A) (Data from Reference 12, Ratings in Parentheses from Reference 160) considered tentative because the issue of overresponsiveness is not completely understood at this time. A broader data base is felt to be necessary to verify the results concerning an upper limit on ω_{BW}, so this is indicated by a broken line on Figure 1.

The evaluation maneuvers performed in the Neal-Smith study included a pitch-bar tracking test but did not have an actual air-to-air tracking task, and there is some suggestion of acceptance of abruptness when tracking a target aircraft. For example, Configuration 13 in the Reference 12 experiments was rated 7 and 5.5 due to "excessive sensitivity." However, in a follow-on experiment (Reference 160) with a target aircraft, Configuration 13 was rated a 2 on two separate evaluations. At first glance this would seem to be an idiosyncrasy of different pilots in a different experiment. However, the target aircraft was removed during a repeat experiment and the rating went from 2 back up to 7 (see ◊ in Figure 8).

The data correlations in Figure 8 represent up-and-away flight and are appropriate for generating boundaries for Category A. Data for Category C (approach and landing) may be found in Reference 5. These data are correlated with ω_{BW} and τ_p in Figure 9. The upper boundary on ω_{BW} for Level 1 is considered tentative for the reasons discussed above.

The bandwidth criterion was developed for highly augmented airplanes, and the data shown in its support have been for high-order systems. Figures 10 and 11 compare bandwidths of classical (unaugmented) airplanes with pilot ratings obtained in flight simulations. For References 9, 28, 101, and 102, the test vehicle was the USAF/Calspan T-33, for which $\tau_p \approx 0.07$ sec (due to actuation and feel systems); τ_p for the Reference 98 data, a Boeing 367-80, is not known but is assumed to be about the same.

The classical-airplane data agree rather well with the Level 2 and 3 boundaries, but for both Categories A and C the Level 1 boundary of Figure 1 appears too stringent. (For example, in Category A flight, Figure 1a does not allow τ_p greater than about 0.06 sec for Level 1; therefore all the Figure 9 data (for $\tau_p \approx 0.07$) should be rated Level 2 or worse.

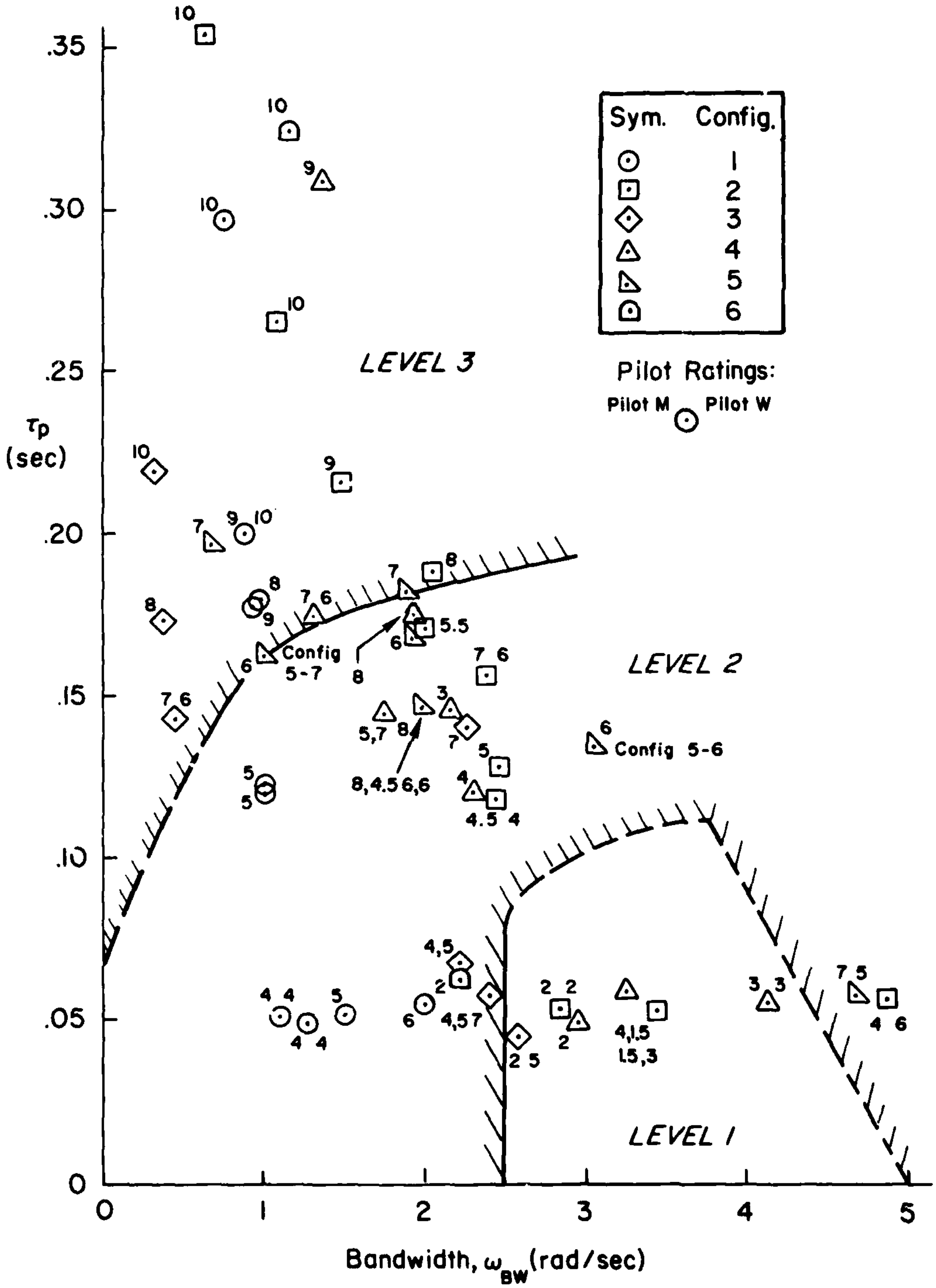


Figure 9 (3.2.1.2). Correlation of Pilot Ratings with ω_{BW} and τ_p for Approach and Landing (Reference 5 Data)

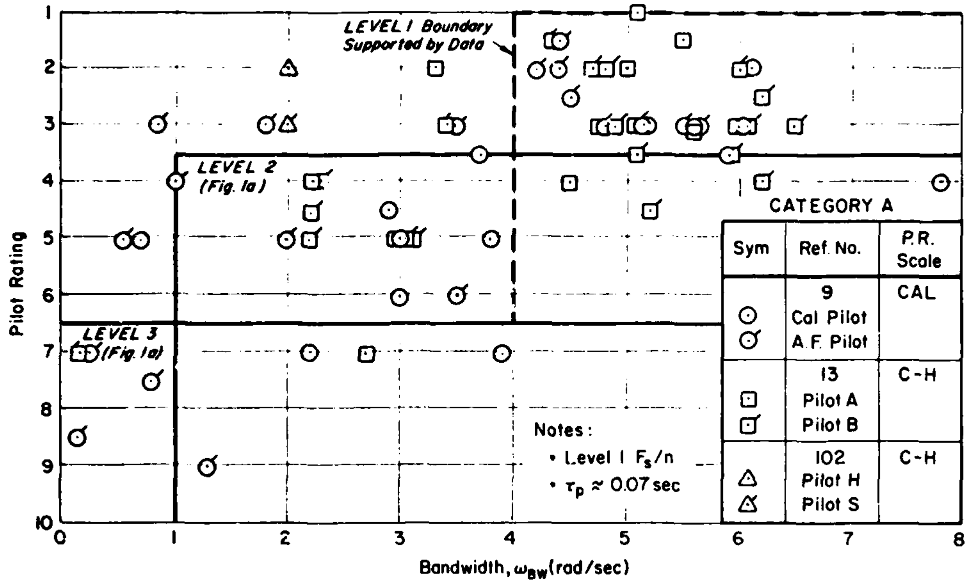


Figure 10 (3.2.1.2). Comparison of Pilot Ratings for Category A Short-Period Configurations with Bandwidth (Classical Airplanes)

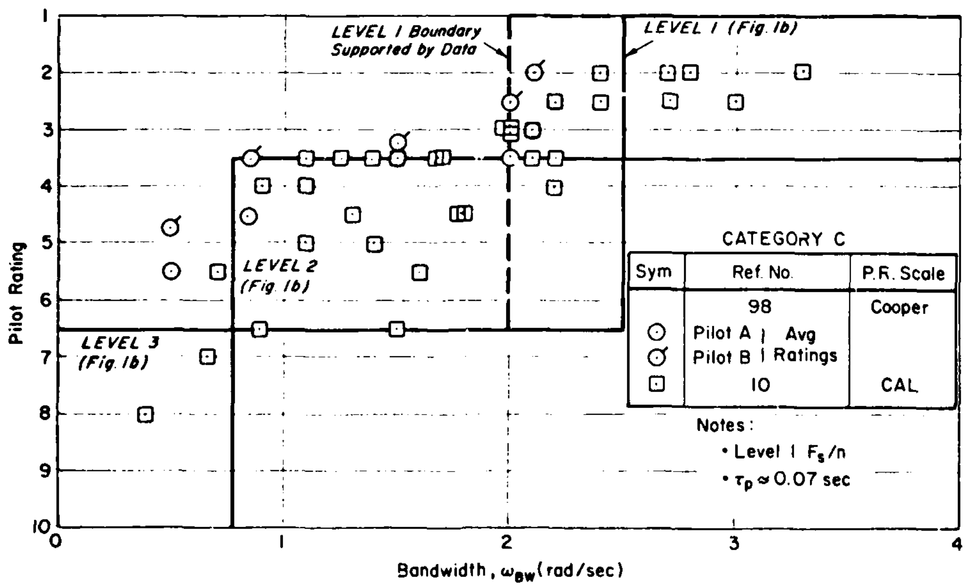


Figure 11 (3.2.1.2). Comparison of Pilot Ratings for Category C Short-Period Configurations with Bandwidth (Classical Airplanes)

The data, however, tend to support a Level 1 boundary at $\omega_{BW} = 4$ rad/sec, as shown by the dashed line. No rating worse than 4-1/2 was given for $\omega_{BW} > 4$.] The reasons for this disagreement have not been resolved, though the tasks used for evaluation, as discussed in 3.2.1.1, may not have been tight enough to promote pilot objections to response abruptness or to excessive time delays.

No supporting data are available at this time to establish Category B boundaries.

Somewhat unstable configurations, with no bandwidth at all, can be flown quite safely (see discussion of Level 3 requirements in "Supporting Data" for Para. 3.2.1.1). Therefore this bandwidth criterion is not sensitive to statically unstable aircraft, and 3.2.1.1 should be applied.

H. LESSONS LEARNED

In its application to direct force control modes, Reference 115, the bandwidth criterion was found to work in areas where conventional criteria are inappropriate. This is discussed in more detail in Para. 3.6.1.2, "Yaw axis bandwidth requirements."

3.2.2 Pilot-Induced Pitch Oscillations

DISCUSSION

Two alternative requirements are presented for this paragraph. The first, a new criterion based upon the PIO studies of Reference 235, is given in Paragraph 3.2.2.1. It is the first specification attempt at quantifying PIO proneness. The second requirement (Paragraph 3.2.2.2) is a qualitative statement prohibiting PIO tendencies, taken intact from MIL-F-8785C. It may be most useful for the procuring activity to specify adherence to the quantitative requirement of 3.2.2.1 during preliminary analysis and design, and to apply the qualitative statement in 3.2.2.2 during simulation and flight testing.

3.2.2.1 Pilot-induced pitch oscillations due to phase lag

A. REASON FOR REQUIREMENT

The purpose of this requirement is to insure that aggressive tracking behavior will not result in instabilities of the closed-loop pilot/aircraft system.

B. RELATED MIL-F-8785C REQUIREMENT

None.

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.2.1 Pilot-induced pitch oscillations due to phase lag. The total phase angle by which normal acceleration measured at the pilot's location lags the pilot's pitch control force input at a criterion frequency, ω_R, must be less than ________________.

Recommended value: The specified phase angle should be less than 180-14.3 ω_R in degrees where ω_R is in radian/second. The criterion frequency ω_R is defined to be any frequency within the range $1 < \omega_R \leq 10$ rad/sec at which lightly damped (resonant) oscillations in pitch attitude can result from turbulence inputs or from piloting control of the aircraft when used in the intended operational manner. This requirement should be waived at the discretion of the procuring activity for those flight conditions for which the ratio of normal acceleration measured at the pilot's location to pitch rate, evaluated at the criterion frequency, is less than 0.012 g/deg/sec.

D. RATIONALE BEHIND REQUIREMENT

We are currently in somewhat of a quandary regarding a specific requirement for PIO. It would, in fact, seem that the equivalent systems and bandwidth requirements (3.2.1.1, 3.2.1.2) as well as the transient F_s/n criterion (3.2.9.2) were specifically formulated to insure that piloted closed-loop tracking in the pitch axis would be satisfactory. Hence, this requirement seems redundant. However, it is included in this draft version of the proposed MIL Handbook for industry review and comment.

The following discussion of the proposed criterion, originally presented in Reference 235, is taken from Reference 122.

The PIO theory of Reference 235 postulates that if the pitch loop is resonant at frequency ω_R, then the pilot may at some time (which cannot necessarily be predicted) attempt to control normal acceleration a_{z_p} to the exclusion or near exclusion of θ. According to Reference 235, a PIO may occur when the normal-acceleration response $n_z(j\omega)/n_{z_e}(j\omega)$ (the subscript e denotes the error sensed by the pilot) is "subjectively predictable": concentrated about some resonant frequency within the pilot's bandwidth of control, with a magnitude there above a threshold value. This situation may arise during pitch target tracking or as a result of the pitching response to a large, abrupt control input, failure transient or gust. A pilot attempting to control normal acceleration at that frequency will incite a PIO if no phase margin exists there; that is, if the phase angle of the $n_z(j\omega)/n_{z_e}(j\omega)$ transfer function is more negative than -180 deg at the resonant frequency. Using a pure 0.25 sec time delay plus gain to model the pilot, the stated phase requirement for the airplane is evolved. Violation of the phase criterion implies that if the pilot switches to a_{z_p} control, the acceleration loop will be dynamically unstable and a PIO will be initiated. This paragraph provides the flight control system engineer with a quantitative criterion for minimum required dynamic performance of feel and control systems. The amplitude criterion of this paragraph is proposed as a quantitative guide for preliminary identification in the design process (airframe or flight control system) of those flight conditions for which longitudinal PIO is probably not a realistic possibility. A combined threshold is postulated of maximum acceptable rms pitch rate in tracking and minimum a_z consciously felt by the pilot. More data should be collected from in-flight simulation to establish the validity of this response ratio; the number selected, 0.012 g/deg/sec, conforms to past cases of longitudinal PIO (Reference 235). The frequency ω_R is, in disguise, a closed-loop, pilot/vehicle parameter. Fortunately it is also a very physical parameter (pitch loop resonant frequency) that is readily understood and accepted. No method is given in the proposed specification for its selection; methods for doing so are contained in Reference 235. The frequency ω_R can be readily identified from flight test.

The existence of a *significant* resonance in closed-loop pitch attitude control indicates that the pilot has closed the loop with very little phase or gain margin. It is difficult to conceive how such closures would occur on aircraft that meet the Level 1 *equivalent system* or bandwidth boundaries (Paragraphs 3.2.1.1 and 3.2.1.2, respectively).

E. GUIDANCE FOR APPLICATION

It would probably be an easy matter for SPO engineers to ascertain compliance with this paragraph without relying on pilot/vehicle analysis methods. For example, ω_R and the specified phase lag can easily be obtained from simulator or in-flight time histories. Nonetheless, analytical estimates can, and should, be made by the airframe manufacturer as part of the design evolution.

F. DEMONSTRATION OF COMPLIANCE

The user should refer to Chapters IV, V, and VI of Reference 235 when applying this requirement.

G. SUPPORTING DATA

Reference 235 illustrated several examples of PIO-prone aircraft. One example is similar to the YF-17 as simulated on the USAF/Calspan variable stability T-33. The θ and a_{z_p} transfer functions are given as follows*:

$$\frac{\theta}{F_s} = \frac{K_\theta(2)(2.3)[.44,\ 11.]}{(0)[.89,\ 1.98][.7,\ 4.0](5)} \qquad (1)$$

$^*(1/T) \rightarrow [s + (1/T)];\ [\zeta,\ \omega] \rightarrow [s^2 + 2\zeta\omega s + \omega^2]$

$$\frac{a_{z_p}}{F_s} = \frac{K_{a_{z_p}}(2)(2.3)[.08,\ 5.04][.44,\ 11]}{(.9)(5.)[.89,\ 1.98][.7,\ 4]} \tag{2}$$

The following discussion is quoted from Reference 235.

Figure 1 is a Bode plot of the airplane's pitch attitude dynamics $\theta/F_s(j\omega)$. If we assume that the crossover frequency will lie between 2-4 rad/sec, then it is clear that the aircraft dynamics are roughly of the form K/s^2 in this region. As a rule, dynamics of this sort will lead to lightly damped closed-loop oscillations and degraded pilot opinion ratings. An inspection of the data base of Reference 238 and a modicum of iteration suggest that a reasonable model for pilot dynamics in pitch tracking would be

$$Y_p(j\omega) = K_p(2.5j\omega + 1)e^{-.385j\omega}$$

A Bode plot of the open-loop system dynamics $Y_{OL} = Y_p(\theta/F_s)$ is also shown in Figure 1. Figure 1 indicates that the absolute maximum crossover frequency with this $Y_p(j\omega)$ is 3.3 rad/sec. Accordingly, $\omega_c = 2.9$ was selected and is assumed to be consistent with what would be measured in actual flight; this yields a small phase margin (about 16 deg). Obviously, even small increases in pilot gain will rapidly degrade system stability. This result appears to be consistent with the evaluation pilots' comments about the poor pitch handling qualities of this configuration in flight tests (Reference 165).

The corresponding closed-loop dynamics $\theta/\theta_c = Y_{CL}$ are shown in Figure 1 for $\omega_c = 2.9$. Obviously, the closed-loop system is extremely resonant at this condition. It is evident by inspection that the resonant peak of θ/θ_c will dominate the a_{z_p} power spectrum. The corresponding damping ratio for this mode is approximately 0.03. Thus, by the simplified criterion for subjective predictability, it must be concluded that PIO cannot be ruled out on the basis of pitch control handling qualities. The resonance frequency $\omega_R = 3.0$ rad/sec for the given $Y_p(j\omega)$. More pilot lead and higher gain would increase ω_R somewhat.

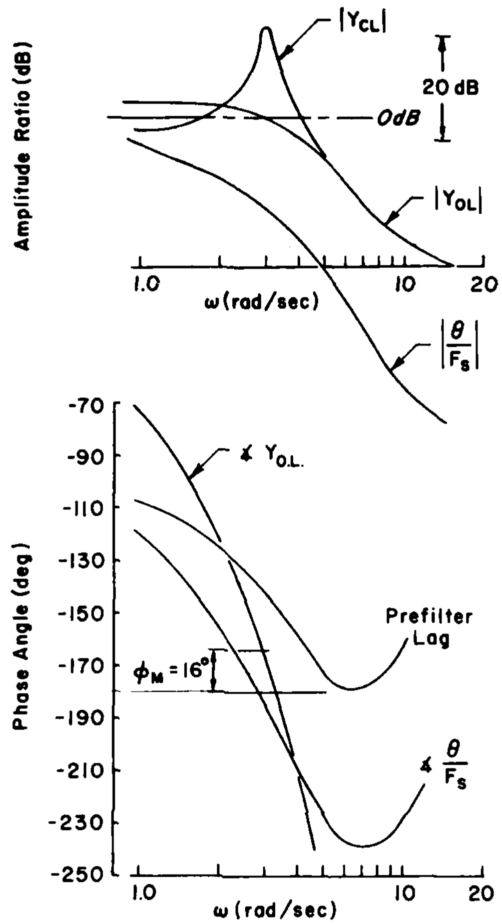


Figure 1 (3.2.2.1). YF-17 Pitch Attitude Dynamics
(from Reference 235)

By the assessment rules of Reference 235, the analysis must now proceed to an investigation of stability of the $a_{z_p} \rightarrow F_s$ loop when the pilot's gain is adjusted to make $\omega_c = \omega_R$. The total $a_{z_p} \rightarrow F_s$ system phase (ϕ) versus frequency is plotted in Figure 2 in accordance with the rules of the PIO theory. The pilot time delay was assumed to be 0.25 seconds. At $\omega = 3.0$ we see that $\phi = -205$ deg, $180 + \phi = -25$ deg (the system phase margin), and we see that the acceleration closed loop is unstable. Thus, longitudinal PIO can be initiated provided that the pilot attempts to control a_{z_p}.

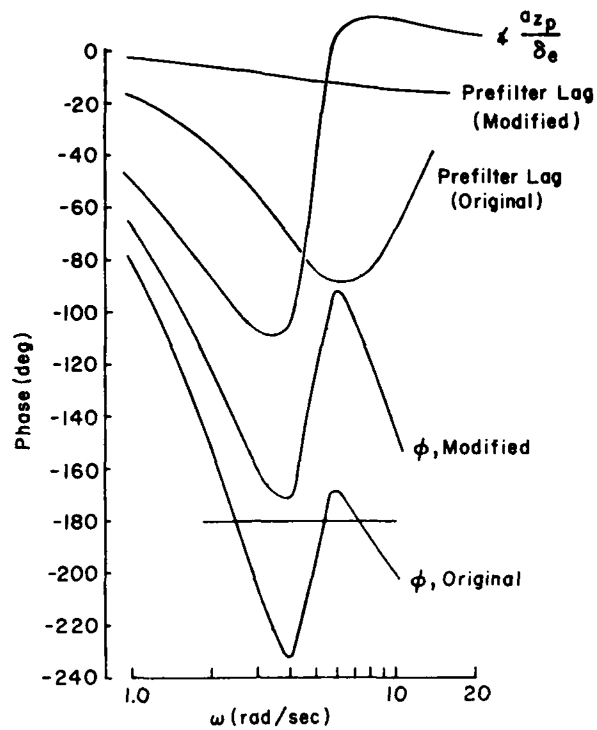


Figure 2 (3.2.2.1). YF-17 Acceleration Control System Dynamics (from Reference 235)

The ratio $|a_{z_p}/\dot{\theta}(3.0j)| = 0.031$ g/deg/sec. Thus, by present theory we would be justified in concluding that PIO would be likely with this airplane and control system.

The actual normal acceleration dynamics simulated with the NT-33A yield $|a_{z_p}/\dot{\theta}(3.0j)| = 0.0213$ g/deg/sec. This is about twice the criterion value of 0.012; on that basis it can be concluded that errors in the simulation of a_{z_p} motion amplitude were probably of no consequence.

The PIO frequency and amplitude obtained with the NT-33A simulation are unpublished. It is known from informal communication between the writer of Reference 235 and Calspan staff members that the PIO frequency occurred at approximately 1/2 cps. It may therefore be concluded that this analysis (and, as a consequence, the present theory) is supported by the flight test results.

H. LESSONS LEARNED

The example given in the "Supporting Data" showed that the criterion successfully predicted a PIO. But what if we checked the pitch dynamics against the equivalent systems or bandwidth criteria of Paragraphs 3.2.1.1 and 3.2.1.2? A lower-order equivalent system was not run for the Equation 1 dynamics. However, defining the short-period damping as 0.89 (as is done in Reference 235) may not be appropriate considering the significant number of higher-order modes that exist. The bandwidth criterion can be checked directly from Figure 1 (θ/F_s) with the following results:

$$\omega_{BW} \doteq 1.3 \text{ rad/sec}$$

$$\tau_p = 0.16 \text{ sec}$$

Plotting these values on Figure 1 (3.2.1.2) shows that the aircraft is very close to the Level 3 region of the flying qualities boundary. Hence, the conclusion that the aircraft is PIO prone is not surprising. In fact, the resonant peak in $|Y_{CL}|$ of Figure 1 is a direct consequence of the above values of ω_{BW} and τ_p. Nonetheless, it may be desirable to retain the criteria to emphasize the notion that a_{z_p} may well be a key parameter for identifying PIO-prone aircraft. Also it may be possible for a configuration to pass the lower-order equivalent system or bandwidth criterion and be caught by the PIO criterion.

3.2.2.2 Pilot-induced pitch oscillations -- qualitative requirement

A. REASON FOR REQUIREMENT

There should be no tendency toward closed-loop oscillations. Any such tendency will destroy mission effectiveness and likely will be dangerous. This paragraph is simply a qualitative statement to cover the obvious undesirability of PIO tendencies. This requirement is considered as an alternative to 3.2.2.1.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.2.3

C. STATEMENT OF REQUIREMENT

3.2.2.2 Pilot-induced pitch oscillations -- qualitative requirement. There shall be no tendency for pilot-induced oscillations, that is, sustained or uncontrollable oscillations resulting from the efforts of the pilot to control the aircraft. The pitch attitude response dynamics of the airframe plus control system shall not change abruptly with the motion amplitudes of pitch, pitch rate or normal acceleration unless it can be shown that this will not result in a pilot-induced oscillation.

D. RATIONALE BEHIND REQUIREMENT

The qualitative requirement of MIL-F-8785C is retained in view of uncertainties in the state of the art of flight control system design. This paragraph is a tacit recognition of the complexity of the PIO problem and an admission that no detailed specification is, at this time, a guarantee against building a PIO-prone airframe/flight-control-system combination. The requirement precludes PIO, PIO tendencies or general handling qualities deficiencies resulting from amplitude-dependent changes in aircraft 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 automatic control system. PIO has occurred in the T-38A, A-4D, and YF-12 due to such abrupt changes.

E. GUIDANCE FOR APPLICATION

This paragraph should be used to supplant the more quantitative, but preliminary, requirement of 3.2.2.1 when piloted simulation or flight testing are to be conducted.

F. DEMONSTRATION OF COMPLIANCE

Pilot comments during flight testing will be the primary source of compliance demonstration. PIO tendency is most liable to appear in the most demanding tasks.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

A very good summary report on PIOs is given in Reference 225. The following paragraphs from that reference discuss the causes of PIOs:

> There are several ways of looking at the causes of a PIO. One is to catalog all the PIO situations ever recorded, including all the necessary subsystem details, etc., and then to say that each combination of vehicle and subsystems when combined with the pilot was the cause of a PIO. Another way is to note that certain system phenomena, such as stick-force-to-control-deflection hysteresis, often lead to PIO when other conditions are right and can thus cause PIO. A third way, and one which seems to transcend the difficulties of the previous two, is to say that certain inherent human physical limitations are the basic cause for any PIO. This is not to degrade the human pilot's role but, instead, to emphasize it, because it is unlikely that any black-box could be devised which is as clever and effective in coping with unmanageable controlled elements as a skilled pilot. Were it not for the pilot's versatile gain adaptability, many flight conditions would be unstable. But there is a limit to the rapidity with which the human can adapt, and this can sometimes lead to a PIO.
>
> When referred to the pilot, then, the basic causes of PIO seem to fall into the following categories:

1. Incomplete pilot equalization

   a. Incomplete training

   b. Inappropriate transfer of adaptation (i.e., carryover of improper techniques from another aircraft)

2. Excessive demands on pilot adaptation

   a. Required gain, lead, or lag lie outside the range of normal capabilities

   b. Rate of adaptation is too slow to preclude oscillation

   c. Inadequate capability to cope with system nonlinearities

3. Limb-manipulator coupling

   a. Impedance of neuromuscular system (including limb) on control stick or pedals changes feel system dynamics

   b. Motion-induced limb force feedback (e.g., arm becomes a bobweight)

Table 1, from Reference 225, lists some known PIO cases and their causes for then-current (early 1960s) aircraft. The causes are equally relevant for modern aircraft, and the lessons learned from the cases listed are valuable in preventing PIOs. The reader is referred to Reference 225 for additional information on PIOs.

TABLE 1 (3.2.2.2). CLASSIFICATION OF SOME KNOWN PIO CASES
(from Reference 225)

Examples shown as; SPECIES (Aircraft; 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 $\zeta\omega_{sp}$ led to zero ζ_{sp} when regulating large disturbances. GCA-INDUCED PHUGOID (C-97); D, v; 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 aircraft 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, T-33VSA, 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 oscillations.	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 = Augmenter (damper)

**Critical Flight Conditions:

a = Low altitude, near-sonic Mach

b = Landing approach or takeoff

c = Cruise

3.2.3 Residual Pitch Oscillations

A. REASON FOR REQUIREMENT

The primary purpose of the requirement is to prevent limit cycles in the control system or structural oscillations that might compromise tactical effectiveness, cause pilot discomfort, etc.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.2.1.3

C. STATEMENT OF REQUIREMENT

3.2.3 Residual Pitch Oscillations. Any sustained residual oscillations in calm air shall not interfere with the pilot's ability to perform the tasks required in service use of the aircraft. For Levels 1 and 2, oscillations in normal acceleration at the pilot's station greater than ± 0.02 g will be considered excessive for any Flight Phase. These requirements shall apply with the pitch control fixed and with it free.

D. RATIONALE BEHIND REQUIREMENT

This requirement may be considered a relaxation of the requirement in 3.2.1 for positive damping at all magnitudes of oscillation. Its intent is to recognize thresholds below which damping is immaterial.

E. GUIDANCE FOR APPLICATION

None required.

F. DEMONSTRATION OF COMPLIANCE

Measurements of normal acceleration at pilot's station should be made in the course of test flight to meet the other flying quality requirements.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

Allowable normal acceleration oscillations have been decreased to ±0.02 g from the ±0.05 g of MIL-F-8785C. This is based on flight test experience with the B-1 (Reference 253), 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 Reference 253, "Flying qualities were initially undesirable due to this limit cycle." Normal acceleration transients in cruise were about ±0.05 - 0.12 g, as Figure 1 shows. The limit cycle was eliminated by installation of a mechanical shaker (dither) vibrating at 20 Hz.

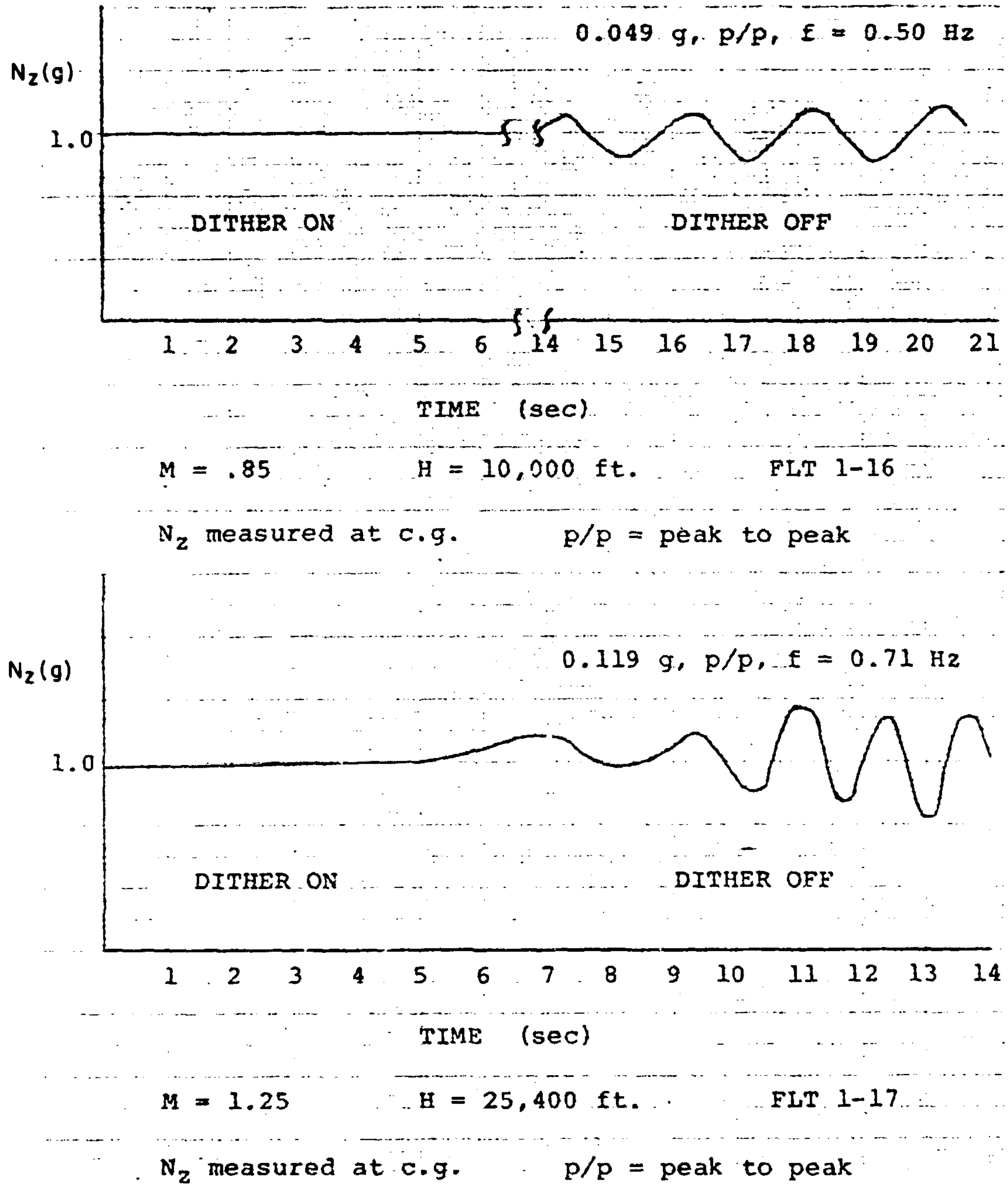


Figure 1 (3.2.3). Effect of Dither on B-1 Limit Cycle Oscillations (from Reference 253)

3.2.4 Vertical Acceleration at Pilot Station

A. REASON FOR REQUIREMENT

The level of vertical acceleration response at the pilot station to the pitch controller should not be objectionably large or of a confusing nature in terms of the pilot's perception of pitch rate response to a pitch controller input.

B. RELATED MIL-F-8785C REQUIREMENT

None.

C. STATEMENT OF REQUIREMENT

3.2.4 Vertical Acceleration at Pilot Station. Vertical acceleration at the pilot station due to pitch control inputs shall have the following characteristics: ________________.

D. RATIONALE BEHIND REQUIREMENT

This is a new requirement whose need is apparent, though insufficient information exists to formulate recommended criteria. Unusual pilot locations can adversely affect handling qualities. A prominent example is the Space Shuttle, where confusing acceleration cues played a part in pilot-induced oscillations encountered during approach and landing tests (Reference 241).

E. GUIDANCE FOR APPLICATION

F. DEMONSTRATION OF COMPLIANCE

G. SUPPORTING DATA

H. LESSONS LEARNED

There is not enough information available for these areas at this time.

3.2.5 <u>Pitch Axis Response to Secondary Controllers</u>

A. **REASON FOR REQUIREMENT**

Operation of controllers intended for flight path or speed control should not cause objectionable pitch response characteristics.

B. **RELATED MIL-F-8785C REQUIREMENT**

3.6.2

C. **STATEMENT OF REQUIREMENT**

3.2.5 <u>Pitch Axis Response to Secondary Controllers</u>. The pitch attitude response to a rapid change in secondary cockpit flight control (throttle, DLC, etc.) shall not exceed the following: ________________.

D. **RATIONALE BEHIND REQUIREMENT**

This requirement is intended to prevent objectionable coupling between controllers designated for the regulation of airspeed or flight path and pitch attitude. It is recognized 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.

There is currently insufficient data to write a quantitative requirement.

E. **GUIDANCE FOR APPLICATION**

F. **DEMONSTRATION OF COMPLIANCE**

G. **SUPPORTING DATA**

H. **LESSONS LEARNED**

Discussions for these areas should be added as data become available.

3.2.6 [Reserved]

3.2.7 Pitch Axis Response to Other Inputs

3.2.7.1 Pitch axis response to auxiliary controls

A. REASON FOR REQUIREMENT

This requirement places a limit on the allowable pitch excursions due to operation of auxiliary controls.

B. RELATED MIL-F-8785C REQUIREMENT

3.6.3

C. STATEMENT OF REQUIREMENT

3.2.7.1 Pitch axis response to auxiliary controls. The maximum allowable pitch response to any auxiliary control shall not exceed ________________.

D. RATIONALE BEHIND REQUIREMENT

There is a limit to the amount of pitch excursions a pilot is willing to accept when operating auxiliary controls. In this case, "auxiliary" refers to controls used for relatively open-loop operations. This includes landing gear extension/retraction, thrust reversers, flaps, etc. — devices which would primarily cause long-term trim changes. There is not enough information available at this time to recommend limits. However, such limits may be written in terms of the control activity required to counter the disturbance or as an open-loop response with controls free.

E. GUIDANCE FOR APPLICATION

F. DEMONSTRATION OF COMPLIANCE

G. SUPPORTING DATA

H. LESSONS LEARNED

No information is available at this time.

3.2.7.2 Pitch axis response to failures

A. REASON FOR REQUIREMENT

Even though an aircraft may be flyable in a failed condition, the transient between the normal and failed state could result in further flying quality degradation.

B. RELATED MIL-F-8785C REQUIREMENTS

3.4.8, 3.4.9, 3.5.5.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.7.2 Pitch axis response to failures

a) Closed-Loop: The pitch attitude motions following sudden aircraft system or component failures shall be such that dangerous conditions can be avoided by pilot corrective action. A time delay of at least ____ sec between the failure and initiation of pilot corrective action shall be incorporated when determining compliance. No single failure of any component or system shall result in Level 3 pitch-axis flying qualities; Special Failure States (1.6.3) are excepted. The crew member concerned shall be provided with immediate and easily interpreted indications whenever failures occur that require or limit any flight crew action or decision.

b) Open-Loop: With controls free, the aircraft motions due to partial or complete failure of the augmentation system shall not exceed the following limits: ______________, for at least ___ seconds following the failure.

Recommended values for the above requirements are as follows:

a) Minimum time delay: 1 second.

b) Transient motions (within first 2 seconds following failure):

Levels 1 and 2 (after failure): ±0.5 g incremental normal acceleration at the pilot's station, except that neither stall angle of attack nor structural limits shall be exceeded. In addition, for Category A, vertical excursions of 5 feet.

Level 3 (after failure): No dangerous attitude or structural limit is reached, and no dangerous alteration of the flight path results from which recovery is impossible.

D. RATIONALE BEHIND REQUIREMENT

Part a) of the requirement is taken directly from Paragraphs 3.4.8 and 3.4.9 of MIL-F-8785C except that a minimum value for time delay is called out in place of stating that a "realistic time delay" shall be incorporated. A recommended minimum value of 1 second is consistent with Para. 3.3.9.3 in MIL-F-9490D. The FAA is more conservative with hardover failures of autopilot servos, requiring 3 seconds before pilot takeover is assumed. This time delay should 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.

Part b) places quantitative limits on the attitude motions. These limits were taken from Paragraph 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.

The revision to MIL-F-8785C followed the recommendations of Reference 234. In particular, 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 as readily accept the larger responses. 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 led General Dynamics/Ft. Worth to suggest a minimum allowable transient (according to Reference 234). This has not been incorporated into the Handbook, but should be a consideration in the design process.

E. DEMONSTRATION OF COMPLIANCE

Worst-case flight conditions should be identified and tested. It is expected that hardover failures occurring at V_{max} and low altitude (maximum dynamic pressure) will be most critical in terms of exceeding the specified limits. As a minimum, Failure States (1.6.2) or Special Failure States (1.6.3) must be tested. This must include engine failures.

F. SUPPORTING DATA

None available.

G. LESSONS LEARNED

As noted above, General Dynamics/Ft. Worth suggested a minimum allowable transient to cue the pilot that a failure has occurred. This was based on experience with the B-58.

3.2.7.3 Pitch axis response to configuration or control mode change

A. REASON FOR REQUIREMENT

Pitch transients due to intentional mode switching must not be excessive.

B. RELATED MIL-F-8785C REQUIREMENTS

3.5.6, 3.5.6.1.

C. STATEMENT OF REQUIREMENTS AND RECOMMENDED VALUES

3.2.7.3 Pitch axis response to configuration or control mode change. The transient motions and trim changes resulting from the intentional engagement or disengagement of any portion of the primary flight control system by the pilot shall be such that dangerous flying qualities never result. With controls free, the motion transients resulting from these situations shall not exceed the following limits for at least ____ seconds following the transfer: ________________. These requirements apply only for Aircraft Normal States (1.6.1).

Recommended values: Transient motions (within first 2 sec following transfer):

Within the Operational Flight Envelope: ±0.05 g normal acceleration at the pilot's station.

Within the Service Flight Envelope: ±0.05 g at the pilot's station.

D. RATIONALE BEHIND REQUIREMENT

Since the intent of a flight control system is to improve the aircraft response characteristics — whether measured by improved flying qualities or by increased mission effectiveness — any system which can be chosen by the pilot should not cause noticeable transient motions. There has been some speculation as to whether a 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. MIL-F-8785B allowed

0.05 g normal acceleration. This was increased to 0.10 g in MIL-F-8785C in an apparent effort to encourage designers to allow some noticeable transient (see Reference 122). In the Background Information and User Guide for 8785C (Reference 122) 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 went 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 is any turbulence at all. Therefore, we are recommending that the maximum allowable transient of 0.05 g used in MIL-F-8785B be utilized in standards developed from this Handbook.

G. GUIDANCE FOR APPLICATION

No specific guidance is offered except that tests should be conducted at the most critical flight conditions.

F. DEMONSTRATION OF COMPLIANCE

Critical conditions will usually be the corners of the expected operational envelopes (e.g., a SAS for power approach should be switched at the highest and lowest expected airspeeds, at low altitudes). Limited analytical and ground-based simulation may be used to supplement actual flight testing, especially in the early stages of development. But flight testing is ultimately required.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.2.7.4 <u>Pitch axis response to stores release</u>

A. REASON FOR REQUIREMENT

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

B. RELATED MIL-F-8785C REQUIREMENT

3.4.6

C. STATEMENT OF REQUIREMENT

3.2.7.4 <u>Pitch axis response to stores release</u>. The intentional release of any stores shall not result in objectionable flight characteristics for Levels 1 and 2. However, the intentional release of stores shall never result in dangerous or intolerable flight characteristics. This requirement applies for all flight conditions and store loadings at which normal or emergency store release is structurally permissible.

D. RATIONALE BEHIND THE REQUIREMENT

This paragraph is unchanged from MIL-F-8785C. It is a necessary catch-all requirement. Because of the variety of possibilities, it must be left qualitative.

E. GUIDANCE FOR APPLICATION

Evaluation of this criterion should occur as a natural part of operational flight testing.

F. DEMONSTRATION OF COMPLIANCE

Operational flight test will be necessary for final demonstration.

G. SUPPORTING DATA

None available.

H. LESSONS LEARNED

None available.

3.2.7.5 Pitch axis response to armament delivery

A. REASON FOR REQUIREMENT

This requirement is included to insure that armament delivery will not have an adverse effect on flying qualities that could impair mission effectiveness.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.7

C. STATEMENT OF REQUIREMENT

3.2.7.5 Pitch axis response to armament delivery. Operation of movable parts such as bomb bay doors, cargo doors, armament pods, refueling devices, and rescue equipment, or firing of weapons, release of bombs, or delivery or pickup of cargo shall not cause buffet, trim changes, or other characteristics which impair the tactical effectiveness of the aircraft under any pertinent flight conditions. These requirements shall be met for Levels 1 and 2.

D. RATIONALE BEHIND THE REQUIREMENT

This paragraph has remained unchanged in MIL-F-8785C and in the MIL Standard. The slight difference in tone between 3.2.7.5 and 3.2.7.4 is the result of design and operational experience.

E. GUIDANCE FOR APPLICATION

This requirement is similar to 3.2.7.4.

F. DEMONSTRATION OF COMPLIANCE

Operational flight test should be required.

G. SUPPORTING DATA

None available.

H. LESSONS LEARNED

None available.

3.2.7.6 Buffet

A. REASON FOR REQUIREMENT

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

B. RELATED MIL-F-8785C REQUIREMENT

3.4.5

C. STATEMENT OF REQUIREMENT

3.2.7.6 Buffet. Within the boundaries of the Operational Flight Envelope, there shall be no objectionable buffet which might detract from the effectiveness of the aircraft in executing its intended missions.

D. RATIONALE BEHIND REQUIREMENT

It is not entirely clear whether this response occurs in heave or pitch, or both. It has been included here since buffet usually represents a pitch controller cue to the pilot. A requirement on buffet must be qualitative because of the varied sources of, and pilot opinions on, buffeting. For a fighter pilot, buffet may be defined (Reference 236) as "a vibration which is perceptible to the pilot to a degree that intrudes into his concentration on his manoeuvring task and may interfere with the precision of his control." In this regard, the cause of the buffeting is unimportant.

E. GUIDANCE FOR APPLICATION

Clearly, in those cases where buffet is a signal to the pilot of approach to a dangerous flight condition (3.1.7.1), some buffet is desirable. Reference 236 contains a concise discussion on 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 instru-

ments. Of the many different buffet level criteria to be found...the following is a summary which smooths out the variations. The "g" values quoted are [maximum excursions about trim:]

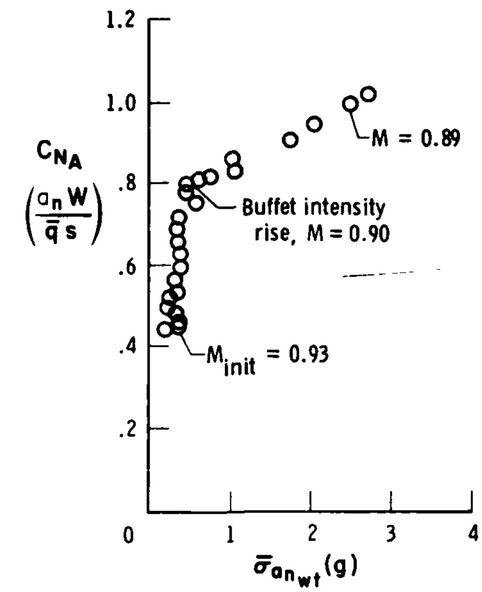
Onset	$\pm .035$ to	$.1\ g_z$	perception depends on workload/ normal g
Light	$\pm .1$ to	$.2\ g_z$	definitely perceptible
Moderate	$\pm .2$ to	$.6\ g_z$	annoying
Severe	$\pm .6$ to	$1.0\ g_z$	intolerable for more than a few seconds

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 manoeuvring, 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 or 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.

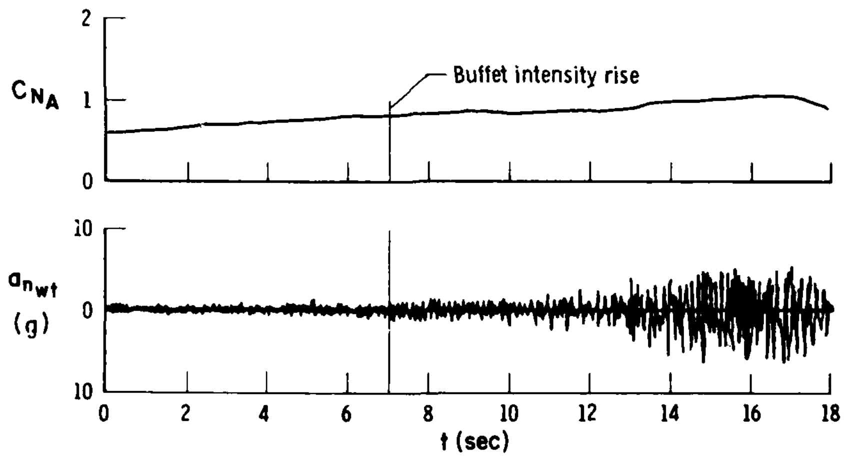
The significance of buffet in air combat depends upon the task. If flight in buffet gives a performance improvement then pilots will use this region during the tactical phase of combat. Tracking will also take place at quite high buffet levels, even with guns; but when the low frequency, high amplitude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

F. DEMONSTRATION OF COMPLIANCE

Flight testing at elevated angles of attack and load factors will reveal any buffeting tendencies. A windup turn maneuver while tracking a target can be especially useful in identifying buffet regions. In flight, buffet intensity rise can be measured with a wingtip accelerometer. Figure 1 (from Reference 237) illustrates methods of determining the region of buffet intensity rise from (a) normalized rms values of wingtip normal acceleration, and (b) estimations based on time history data.



a) Normalized RMS Values of Wingtip Normal Acceleration



b) Time History of Wingtip Acceleration

Figure 1 (3.2.7.6). Buffet Intensity Rise Determination (from Reference 237)

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.2.8 Pitch Axis Control Power

3.2.8.1 Pitch axis control power in unaccelerated flight

A. REASON FOR REQUIREMENT

This requirement is intended to insure that the pilot can maintain equilibrium flight throughout the flight envelope.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.3.1.

C. STATEMENT OF REQUIREMENT

3.2.8.1 Pitch axis control power in unaccelerated flight. In steady 1 g flight at all service altitudes, the attainment of all speeds between V_S and V_{max} shall not be limited by the effectiveness of the longitudinal control or controls.

D. RATIONALE BEHIND THIS REQUIREMENT

Controllability at speeds down to the 1 g stall speed is generally deemed necessary for safety of maneuvering aircraft such as the military use. V_{max}, the high-speed boundary of the Service Flight Envelope, must be at least $V_{O_{max}}$; beyond that, it may be set by the contractor -- who then must deliver on his promise.

E. GUIDANCE FOR APPLICATION

It is important to explore all corners of the flight envelope. For example, adequate pitch axis control during high-speed dives can be critical due to combined aeroelastic and Mach number effects.

F. DEMONSTRATION OF COMPLIANCE

Operational flight test will reveal any deficiencies in pitch control power.

G. SUPPORTING DATA

None required.

H. LESSONS LEARNED

None.

3.2.8.2 <u>Pitch axis control power in maneuvering flight</u>

A. REASON FOR REQUIREMENT

This requirement is included to insure that the pitch axis controller is sufficiently powerful to produce an adequate range of load factors for maneuvering.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.3.2.

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.8.2 <u>Pitch axis control power in maneuvering flight</u>. Within the Operational Flight Envelope, it shall be possible to develop, by use of the pitch control alone, the following range of load factors: ______________. 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 percent or ±50 kt equivalent airspeed (except where limited by the boundaries of the Operational Flight Envelope).

<u>Recommended range</u> of load factors:

Levels 1 and 2: $n_o(-)$ to $n_o(+)$

Level 3: $n = 0.5$ g to the lower of:

a) $n_o(+)$

b) $n = 2.0$ for $n_o(+) < 3$ g

$0.5\,[n_o(+) + 1]$ for $n_o(+) > 3$ g

D. RATIONALE BEHIND REQUIREMENT

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.

E. GUIDANCE FOR APPLICATION

This requirement is restricted in application to the Operational Flight Envelope with relaxed requirements for infrequent Failure States. Outside the Operational Flight Envelope, whatever falls out of the design is now acceptable, as long as the other control requirements are met. The Level 3 requirement assures modest nose-down and nose-up control capability for stabilization as well as for altering equilibrium and maneuvering.

F. DEMONSTRATION OF COMPLIANCE

Operational flight test should be required.

G. SUPPORTING DATA

None required.

H. LESSONS LEARNED

None available.

3.2.8.3 Pitch axis control power in takeoff

A. REASON FOR REQUIREMENT

This requirement is intended to regulate against aircraft that exhibit no apparent pitch response to commands during the takeoff roll until the flying speed is reached (V_{min}). These aircraft tend to "pop off," resulting in overrotation and a necessity for immediate control reversal to avoid stall.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.3.3.

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.8.3 Pitch axis control power in takeoff. The effectiveness of the pitch control shall not restrict the takeoff performance of the aircraft. Satisfactory takeoffs shall not be dependent upon use of the trim controller during takeoff or on complicated control manipulation by the pilot. It shall be possible to obtain and maintain the following attitudes ________________________ during the takeoff roll.

The recommended attitudes are as follows:

- For nosewheel aircraft it should be possible to obtain at 0.9 V_{min} the pitch attitude that will result in a liftoff at V_{min}.
- For tailwheel aircraft it should be possible to maintain any pitch attitude up to that for a level thrust line at 0.5 V_S for Class I aircraft and at V_S for Classes II, III, and IV.

These requirements should be met on hard surface runways. In the event that the aircraft has a mission requirement for operation from unprepared fields, these requirements should be specified to be met on such fields.

D. RATIONALE BEHIND REQUIREMENT

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 flying qualities.

E. GUIDANCE FOR APPLICATION

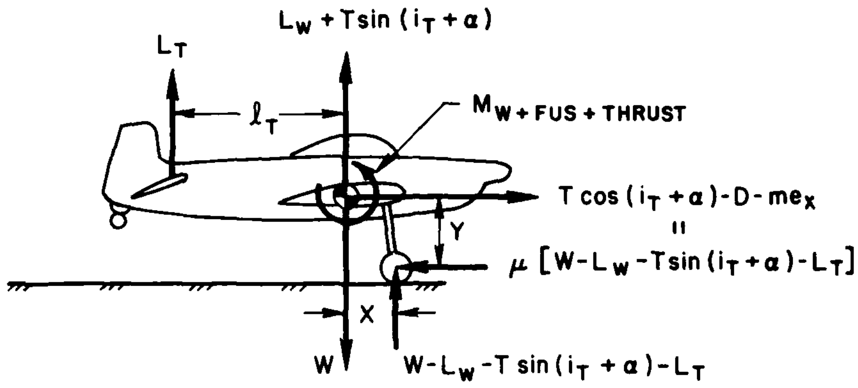
This requirement is more important for single-engine aircraft or any multi-engine aircraft which have V_{MCA} equal to or less than V_{min}. For multi-engine aircraft where $V_{MCA} > V_{min}$, the requirement could be relaxed to 0.9 V_{MCA}.

F. DEMONSTRATION OF COMPLIANCE

The ability to comply with this requirement should be obvious during operational flight test. Special emphasis should be placed on short-field takeoffs at the maximum forward center-of-gravity limit.

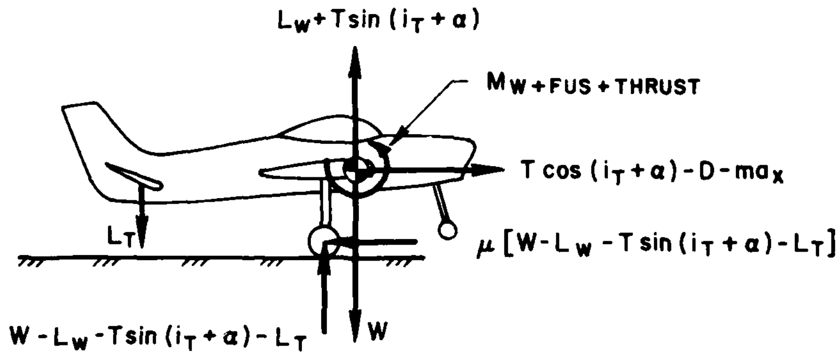
The requirement for takeoff from unprepared fields is included on the basis of a rational analysis. All aircraft will have to operate from hard-surface runways, and therefore hard surfaces were used as the basic requirement. An increased coefficient of friction, however, such as occurs with unprepared fields, increases the elevator effectiveness required for nose-wheel aircraft but decreases the effectiveness required of tail-wheel aircraft, as can be seen from the sketches in Figure 1.

Assume first that the tails of both aircraft are adequately sized to achieve the takeoff attitude on a hard-surfaced runway (low μ). Then on a soft runway (higher μ), the increased rolling friction force gives a nose-down pitching moment about the aircraft c.g. which helps lift a tail wheel but hinders lifting a nose wheel. Nose-wheel lift-off speed will increase monotonically with increasing μ, approaching the speed for takeoff in the ground attitude. But tail-wheel lift-off speed will decrease with increasing μ until just the application of takeoff thrust will rotate the aircraft at zero speed. Then a different technique would be required.



From ΣM_{CG},

$$L_T = \frac{M_{W+F+TH} + (X - \mu Y)[W - L_W - T\sin(i_T + \alpha)]}{\ell_T + X - \mu Y}$$



$$L_T = \frac{-M_{W+F+TH} + (X + \mu Y)[W + L_W + T\sin(i_T + \alpha)]}{\ell_T + X - \mu Y}$$

Figure 1 (3.2.8.3). Dynamics of Takeoffs for Tailwheel vs. Nosewheel Aircraft

C. SUPPORTING DATA

None required.

H. LESSONS LEARNED.

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-tail aircraft than in an identical aircraft with a conventional horizontal tail. This has been attributed to delayed liftoff due to inability to rotate to the takeoff attitude prior to V_{min}. The root cause of the problem lies in the fact that the horizontal tail is out of the propeller wake. Multi-engine airplanes 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.

The requirement is important for turbojet aircraft where relatively large pitch attitudes are required for liftoff.

3.2.8.4 Pitch axis control power in landing

A. REASON FOR REQUIREMENT

This requirement insures that the aircraft 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.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.3.4.

C. STATEMENT OF REQUIREMENT

3.2.8.4 Pitch axis control power in landing. The pitch control shall be sufficiently effective in the landing Flight Phase in close proximity to the ground so that ________________.

It is recommended that the following requirements be placed on pitch control during landing flare and rollout with the aircraft trimmed for the minimum recommended approach speed not to exceed 1.3 $V_S(L)$:

- The geometry limited touchdown attitude can be achieved at touchdown.
- The guaranteed minimum landing speed [$V_{min}(L)$] can be achieved when flaring from shallow ($\gamma \approx -3^o$) and steep ($\gamma \approx -6^o$) approaches.
- The nosewheel can be gently lowered to the ground at speeds down to 0.9 $V_{min}(L)$.
- For tailwheel aircraft, the tailwheel can be gently lowered to the ground at 0.5 $V_{min}(L)$ for Class I and 0.75 $V_{min}(L)$ for Classes II, III, and IV.

D. RATIONALE BEHIND REQUIREMENT

This requirement is to insure adequate pitch control during flare and rollout in ground effect. Elevator effectiveness can be severely degraded in ground effect due to a decrease in downwash caused by 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. flight conditions).

E. GUIDANCE FOR APPLICATION

No particular guidance is deemed required except to note that the maximum forward c.g. (regardless of weight) defines the critical flight condition.

F. DEMONSTRATION OF COMPLIANCE

Demonstration of compliance should be concentrated at the extreme forward c.g. loadings.

G. SUPPORTING DATA

None required.

H. LESSONS LEARNED

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 be a result of the resulting high shock environment.

3.2.8.5 Pitch axis control power for other conditions

A. REASON FOR REQUIREMENT

This catch-all specification is intended to assure adequate pitch control power in any situation not already covered in the Standard.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.10

C. STATEMENT OF REQUIREMENT

3.2.8.5 Pitch axis control power for other conditions. Control authority, rate and hinge moment capability shall be sufficient to assure safety throughout the combined range of all attainable angles of attack (both positive and negative) and sideslip. This requirement applies to the prevention of loss of control and to recovery from any situation for all maneuvering, including pertinent effects of factors such as regions of control-surface-fixed instability, inertial coupling, fuel slosh, the influence of symmetric and asymmetric stores, stall/post-stall/spin characteristics, atmospheric disturbances and Aircraft Failure States (maneuvering flight appropriate to the Failure State is to be included). Consideration shall be taken of the degrees 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.

D. RATIONALE BEHIND REQUIREMENT

The other paragraphs falling under 3.2.8, as well as 3.8.4 and 3.8.5, cover all normal, anticipated situations for pitch control power. This paragraph is added to cover any unusual or unspecified conditions that might be encountered in flight. Hinge moments affect actuator rate, and can even limit control deflection. And they are hard to estimate accurately. Without careful attention in design, these limitations can result in unwanted feedback to the cockpit controls. Control problems are compounded on a surface used in more than one axis, e.g., collective stabilizer for pitch, differential stabilizer for roll. At the high dynamic pressure responsible for high hinge moments, aeroelasticity may be a factor.

E. GUIDANCE FOR APPLICATION

Due to its broad generality the requirement should be applied for all phases of analysis, simulation, and flight test. Excessive stability, as well as excessive instability of the basic airframe, is of concern with respect to available control authority and rate.

F. DEMONSTRATION OF COMPLIANCE

Conditions which the procuring activity considers too dangerous to investigate in flight should be investigated by analysis, model testing and simulation as appropriate. For example, aircraft not required to be designed structurally for spinning should not be spun, but models of them might profitably be spun in free flight or a spin tunnel; certainly wind-tunnel studies should investigate deep stall characteristics, whether or not flight demonstration will proceed past a full stall. The scope of analysis, simulation and testing needs careful consideration at the outset of a program. Then the design must be monitored for possible additional troubles with controllability, and any necessary changes made in the demonstrating program.

It is <u>not</u> the intent of this requirement to add unreasonably to the risk of a flight test program. It <u>is</u> the intent to assure that dangerous conditions are found before the aircraft gets into operational use.

G. SUPPORTING DATA

None required.

H. LESSONS LEARNED

None required.

3.2.9 Pitch Axis Control Forces

DISCUSSION

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 the force requirements of this section:

- 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.

- The requirements should be considered to be stringently applicable to 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.

- Maximum forces specified appear in most cases to be quite large for weaker male pilots for continuous operation.

- 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 (Reference 6) discussed the effect of stick location on push and pull capabilities (Figure 1), and the effect of upper arm angle on push and pull strength for the 5th and 95th percentile male (Figure 2). The data shown in these figures are from Reference 256 for male Air Force personnel in the sitting position. As described in Reference 6 these data show that:

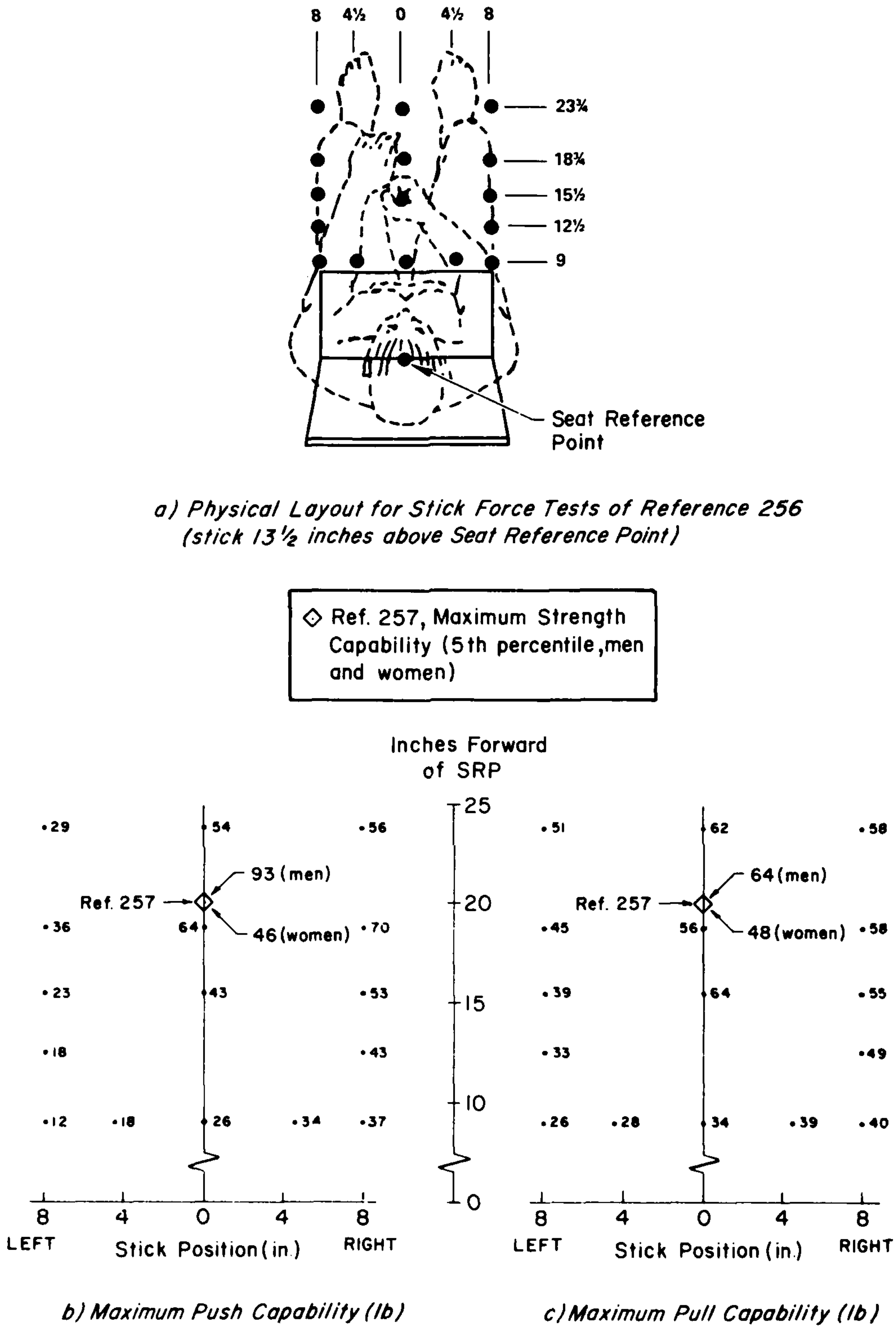
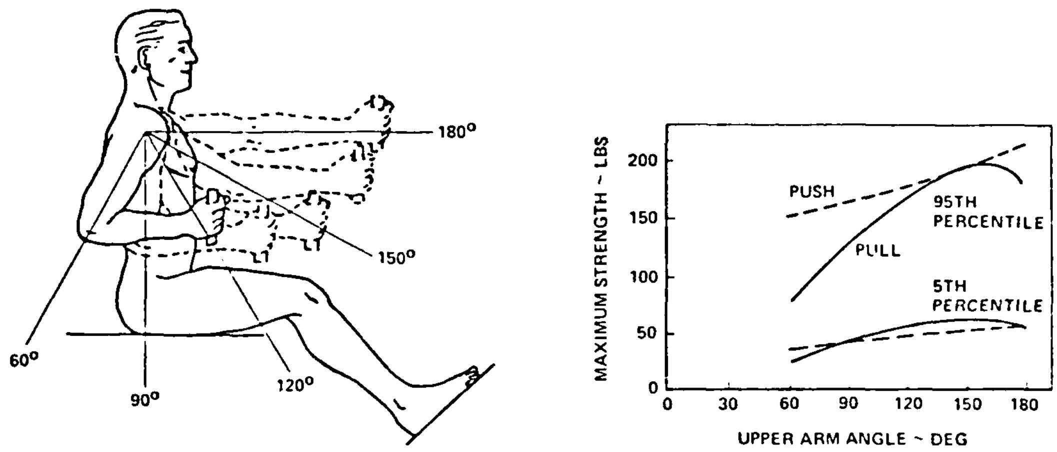


Figure 1 (3.2.9). Effect of Arm/Stick Geometry on Maximum Push and Pull Capability by the Right Arm for the 5th Percentile Male (Reference 256)



a) Physical Layout

b) Upper Arm Angle vs. Maximum Strength

Figure 2 (3.2.9). Effect of Upper Arm Angle on Pull and Push Strength for the 5th and 95th Percentile Male (Reference 256)

> ...one's maximum force capability is not symmetric left and right and varies by 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 instance, the F-5A stick deflection is greater than that of the A-7D by more than a factor of 2. This places the stick in a different location in the cockpit for maximum deflection.

A more recent study, Reference 257, presents a comparison between male and female strength characteristics for operating an aircraft control stick. Table 1 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 in Figure 1 for comparison.

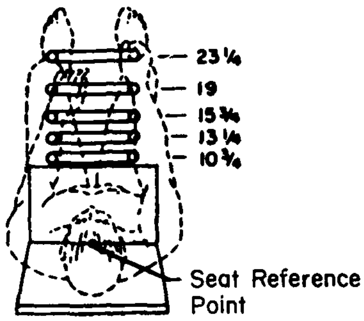
TABLE 1 (3.2.9). MAXIMUM FORCES EXERTED ON AIRCRAFT CONTROL STICK (LBS) BY MEN AND WOMEN (REFERENCE 257)

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

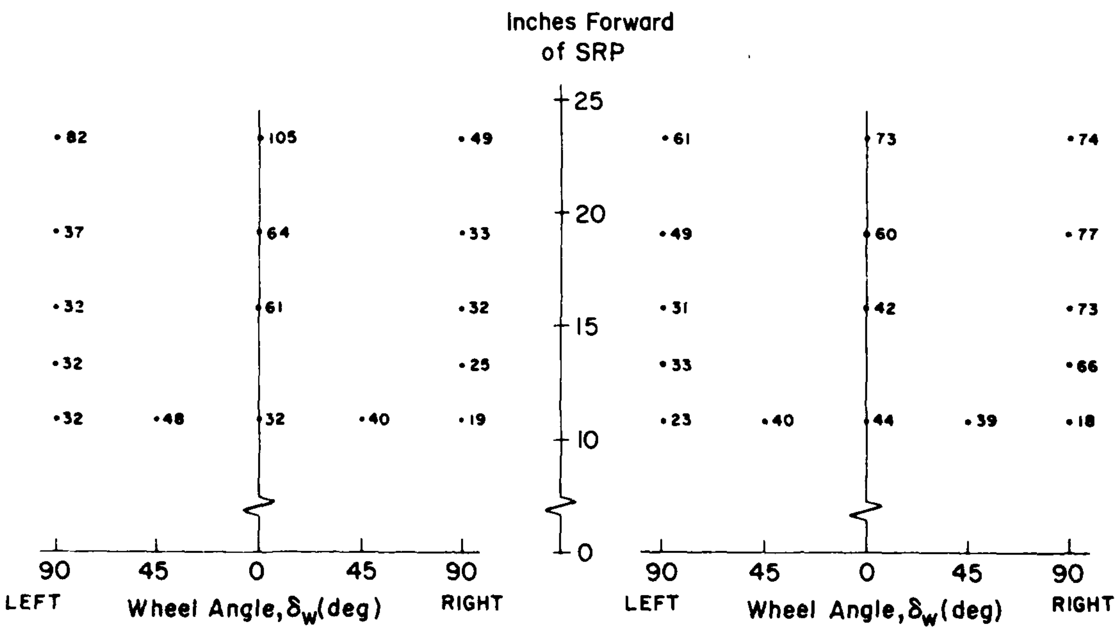
It can be seen that the strength capabilities of men are greater than that of women: i.e., the 95th percentile woman has approximately the same performance as the 5th percentile man. As Reference 257 observes, the force limits in this Handbook "may not be consistent with the capabilities of pilots."

Figure 3 shows data from Reference 256 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 3a 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 pilots' entire arm is used. This can be seen in Figure 3b where the push capability at 23-1/4 inches from the SRP is approximately twice that obtained when the control wheel is at its closest at 10-3/4 inches from the SRP. Similarly the maximum pull capability varies almost by a factor of 2 in Figure 3c depending on the control wheel angle and position.

It must be stressed that these are <u>maximum</u> forces in single applications; clearly, continuous operation (such as would be expected in meeting any of the force requirements in the MIL Standard) would produce much lower maximum forces. In a discussion of some general principles



a) Physical Layout for Wheel Force Tests of Reference 256 (wheel 18 inches above seat reference point)



b) Maximum Push Capability (lb)

c) Maximum Pull Capability (lb)

Figure 3 (3.2.9). Effect of Arm Position and Wheel Angle on Maximum Push and Pull Capability by the Right Arm for the 5th Percentile Male (Reference 256) of control design, and one- vs. two-handed operations, Reference 256 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 shoulder, pull is approximately doubled, push is not greatly increased except at close distances...

3.2.9.1 Pitch axis control forces — steady-state control force per g

A. REASON FOR REQUIREMENTS

These requirements relate to the classical "stick-free" static maneuvering stability (stick force per g, F_s/n) at approximately constant speed. The basic premise is that F_s/n represents a necessary tactile cue for elevated values of load factor. Low values of F_s/n result in excessive sensitivity with a tendency toward exceeding the airplane structural limits. High values lead to pilot fatigue during maneuvering.

B. RELATED MIL-F-8785C REQUIREMENTS

3.2.2.2, 3.2.2.2.1

C. STATEMENT OF REQUIREMENTS AND RECOMMENDED VALUES

3.2.9.1 Pitch axis control forces -- steady-state control force per g.

a) Control Feel and Stability in Maneuvering Flight at Constant Speed. In steady turning flight and in pullups and pushovers at constant speed, for Levels 1 and 2 there shall be no tendency for the aircraft pitch attitude or angle of attack to diverge aperiodically with controls fixed or with controls free. For the above conditions, the incremental control force 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 those required to initiate the change. These requirements apply for all local gradients throughout the range of service load factors defined in 1.5.2.

b) Control Forces in Maneuvering Flight. 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 following load factor ranges:

CLASS	MIN.	MAX.
I, II & III	0.5	$0.5[n_o(+) + 1]$ or 3
IV	0	Lesser of Class I, II, and III maximums

> Outside this range, a departure from linearity resulting in a local gradient which differs from the average gradient for the maneuver by more than 50 percent is considered excessive, except that larger increases in force gradient are permissible at load factors greater than 0.85 n_L. The local force gradients shall be: ________________. In addition, F_s/n should be near the Level 1 upper boundaries of these gradients for combinations of high frequency and low damping. The term gradient does not include that portion of the force versus n curve within the breakout force.
>
> For side stick controllers, the contractor shall show that the control force gradients will produce suitable flying qualities.

Recommended limits are given in Table 1.

D. RATIONALE BEHIND REQUIREMENTS

The requirements for control forces in maneuvering flight are unchanged from MIL-F-8785C.

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 adequately described 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 (References 3 and 24) that F_s/n at very low n/α can or should be higher than at high 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.

TABLE 1 (3.2.9.1). PITCH MANEUVERING FORCE GRADIENT LIMITS

a) Center Stick Controllers

LEVEL	MAXIMUM GRADIENT $(F_s/n)_{max}$, lb/g	MINIMUM GRADIENT $(F_s/n)_{min}$, lb/g
1	$240/(n/\alpha)$ but not more than 28.0 nor less than $56/(n_L-1)$ *	The greater of $21/(n_L - 1)$ and 3.0
2	$360/(n/\alpha)$ 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

*For $n_L < 3$, $(F_s/n)_{max}$ is 28.0 for Level 1, 42.5 for Level 2.

b) Wheel Controllers

LEVEL	MAXIMUM GRADIENT $(F_s/n)_{max}$, lb/g	MINIMUM GRADIENT $(F_s/n)_{min}$, lb/g
1	$500/(n/\alpha)$ but not more than 120.0 nor less than $120/(n_L-1)$	The greater of $35/(n_L - 1)$ and 6.0
2	$775/(n/\alpha)$ 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

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, as can be seen from the following constant-speed relation:

$$\frac{F_s}{\alpha} = \frac{F_s}{n} \frac{n}{\alpha}$$

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.

However, there is some question as to the significance of the References 3 and 24 tests. These references are discussed in detail under "Supporting Data."

E. GUIDANCE FOR APPLICATION

To illustrate the use of the gradient limits of Table 1, Figures 1 and 2 show possible boundaries for two representative aircraft. Figure 1 is for a centerstick controller with $n_L = 7.0$; Figure 2 is for a wheel controller with $n_L = 3.0$. Similar plots may be constructed for any aircraft using the Table 1 formulas. However, such plots, while representing the Table 1 suggested limits, do not convey the entire picture, as illustrated by the following considerations.

Effects of stick/wheel position on acceptability of F_s/n are not covered by these requirements. But it seems intuitively obvious that there must be an interrelationship of force and deflection gradients with control location. See the discussion for 3.2.9 for more information on this subject.

Another item for consideration is the allowance in Table 1 for considerably higher values of F_s/n when n/α is low. The data found to both support and refute the change in dependence from n_L to n/α is discussed in detail under "Supporting Data." However, in terms of applying the

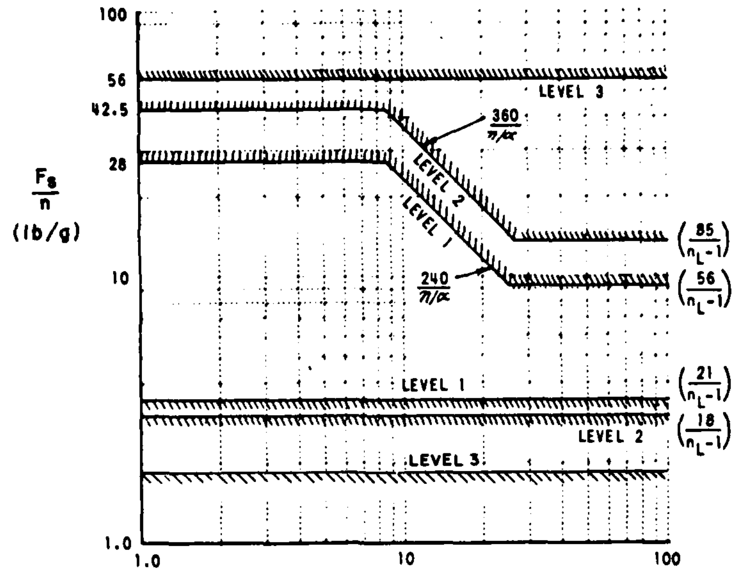


Figure 1 (3.2.9.1). Elevator Maneuvering Force Gradient Limits: Center-Stick Controller, $n_L = 7.0$

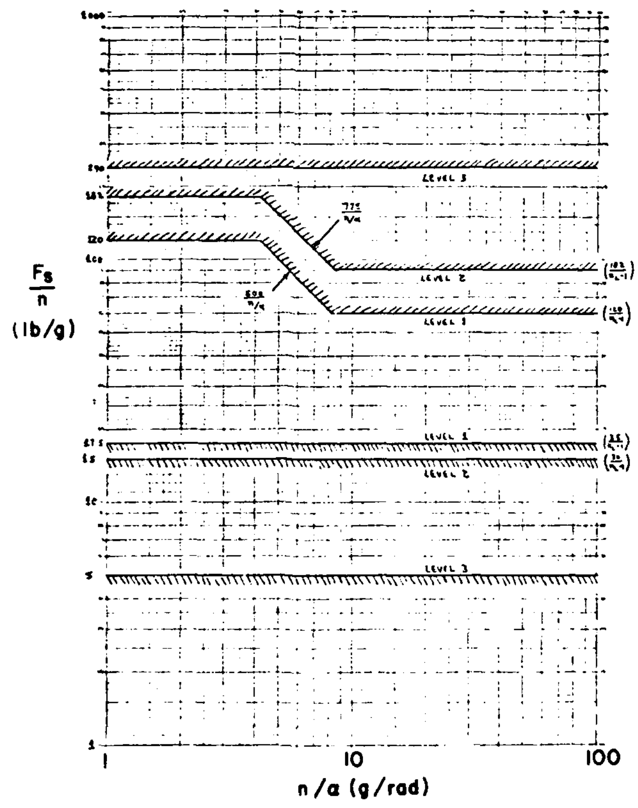


Figure 2 (3.2.9.1). Elevator Maneuvering Force Gradient Limits: Wheel Controller, $n_L = 3.0$

requirement at low speed this question is really academic. When operating in low-speed, high-angle-of-attack flight, the limiting factor may be not the gradient but merely the maximum stick force at $C_{L_{max}}$. For example, Figure 3 (from Reference 7) illustrates the variation of F_s/n with airspeed for the OV-10A aircraft. This variation is due to the effects of the elevator spring tab. The gradients shown are well within the Level 1 requirements of Table 1. However, as airspeed decreases, maximum attainable load factor (n) also decreases. So the apparently large and rapid change in F_s/n actually results in fairly constant maximum stick force at stall (Figure 4). Therefore, while the high-speed criterion for specifying F_s/n is structural protection (n_L), at low speeds normal load factor will produce stall before n_L is exceeded. It follows that the incremental force between the trimmed airspeed and stall (or at the boundary of the operational flight envelope) becomes the important factor at speeds below "maneuvering speed" (V_A). Maneuvering speed is, of course, defined when stall occurs at the limit load factor (see Figure 3).

A more basic consideration relative to Table 1 is the complete absence of a force gradient specification for sidestick controllers, reflecting in part the limited data base. However, References 8 and 166 based on a series of flight tests conducted by students of the USAF Test Pilot School at the Air Force Flight Test Center, Edwards AFB, give some insight into preferred gradients. These data are plotted in Figure 5 and are based on an air-to-air tracking task. The test aircraft was the USAF/Calspan variable stability T-33 with a T-38A utilized as the target. The pilot ratings shown are the average over three pilots, all with fighter experience. An approximate Level 1 boundary is suggested in Figure 5. In general, the F_s/n range is comparable to that of Table 1, i.e., 2-14 lb/g. The relatively weak frequency dependence may also exist for centerstick controllers, although there are no data to support this.

Note that the F_s/n gradients in Figure 5 are the <u>initial</u> values; the mechanization was such that the slope at larger deflections was half the initial slope. The breakout forces were 1/2 lb. Reference 8 suggests that the sidestick neutral position

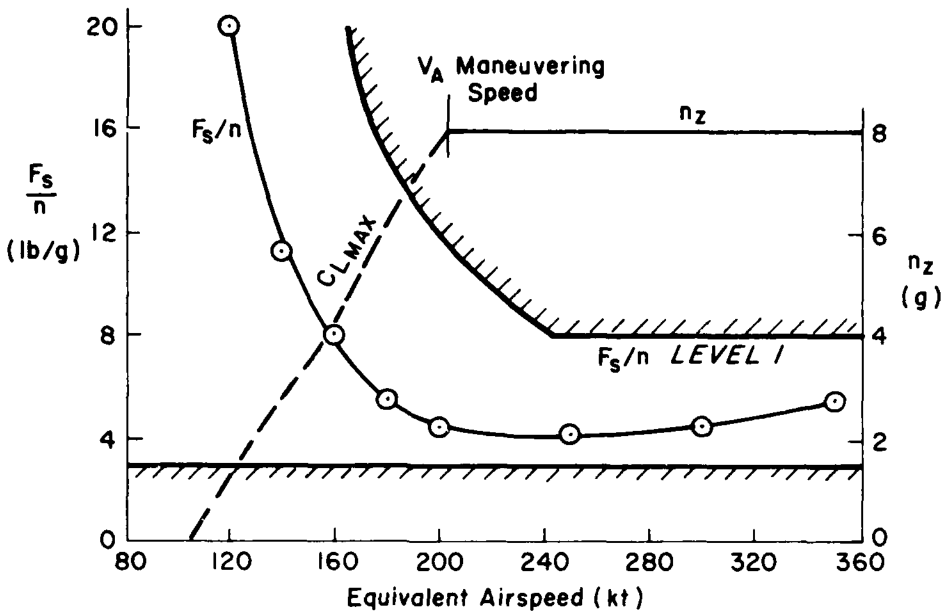


Figure 3 (3.2.9.1). OV-10A Maneuvering Control
(Reference 7)

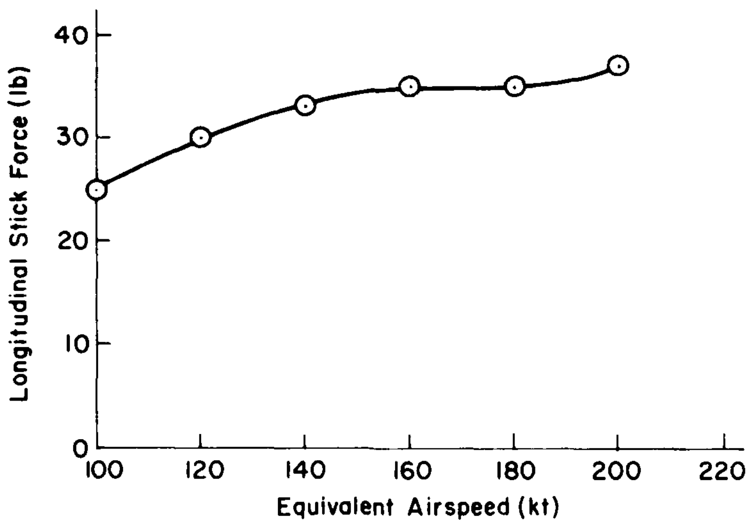


Figure 4 (3.2.9.1). Longitudinal Stick Force at Stall
(Reference 7)

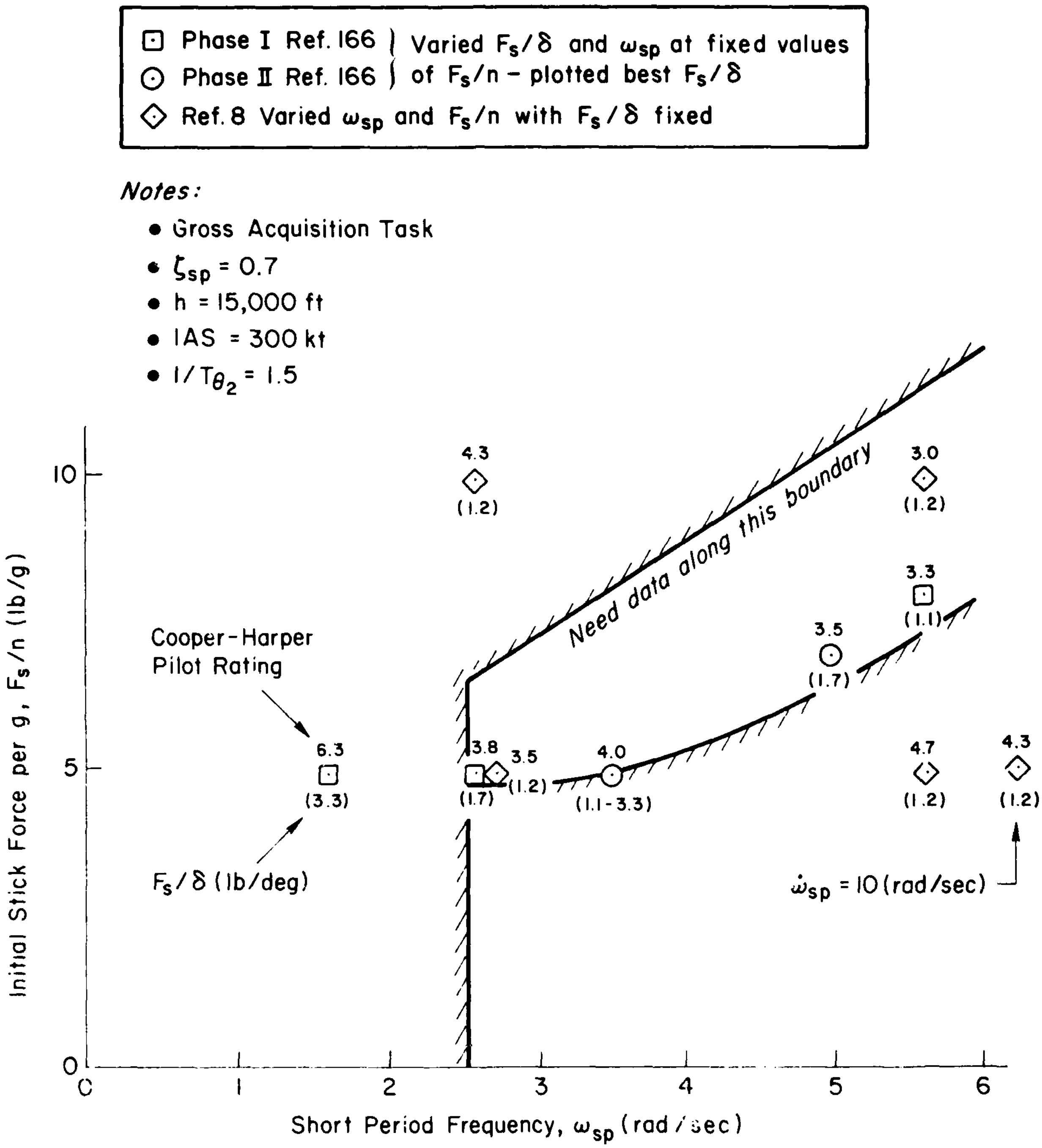


Figure 5 (3.2.9.1). Short Period Frequency vs. Longitudinal Stick Force Per g (F_s/δ Separately Optimized)

...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.

F. DEMONSTRATION OF COMPLIANCE

The following discussion is concerned with the flight testing aspects of determining control force characteristics. However, at the discretion of the procuring activity, a valid moving-base piloted simulation may be used in proof-of-concept stages.

There are several methods for obtaining the required control force data. The best method to use depends primarily on the speed range under consideration. A 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" should be interpreted as "Mach No" since that then 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. An alternate version of this method is to stabilize the airplane holding various amounts of out-of-trim control force in straight flight, then suddenly release the control and measure the resulting normal acceleration increment.

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 should be left at their trim settings throughout the maneuver to minimize the possibilty of introducing extraneous pitching moments. The gradients obtained in this manner will not be quite as linear as with the symmetric pullup method because of the difference in pitch rate between pullups and turns (see, e.g., Reference 2). But, with the possible exception of a more stable slope near 1 g in the turns, the differences are generally small enough to be within the measurement errors.

A third method that is sometimes used involves a windup turn. 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 should 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. Also, it is impossible to obtain data for n less than 1.0 using turn methods.

G. SUPPORTING DATA

The data base for verifying control force requirements is sparse, and limited entirely to Class IV airplanes. The most thorough data sets are derived from USAF/Calspan T-33 flight tests in which pilots chose optimum values of F_s/n for varying short-period characteristics. However, the useful information is basically a byproduct, since the intent of these tests was the study of short-period frequency and damping. So, while specific conditions can be found for which ζ_{sp} and ω_{sp} fell within Level 1 boundaries, they were not held constant, and subsequent pilot ratings could reflect an interrelationship of ζ_{sp}, ω_{sp}, and F_s/n. In addition, control force breakout may not have been within the Level 1 limits of MIL-F-8785C for the tests; breakout is not documented in any of the supporting references except in Reference 12, where it is reported to be zero.

References 9 and 10 contain data used in Reference 11 to support the F_s/n limits of MIL-F-8785B. More recent tests (References 5 and 12) add to the data base for centerstick controllers. Reference 13 provides some insight into requirements for wheel-type controllers.

Figure 6 compares values of "optimum" F_s/n from Reference 9 with the requirements of Table 1. In this test program gradients were selected before performing evaluation tasks, and were held constant throughout each evaluation. The external parameters (ζ_{sp}, ω_{sp}, $1/T_{\theta_2}$, τ_e) are all within Level 1 limits, but may vary widely. While there is considerable scatter, pilot ratings increase (degrade) as F_s/n increases. However, a much lower $(F_s/n)_{max}$ (≈ 6.5 for Level 1) than the specification upper limit is indicated.

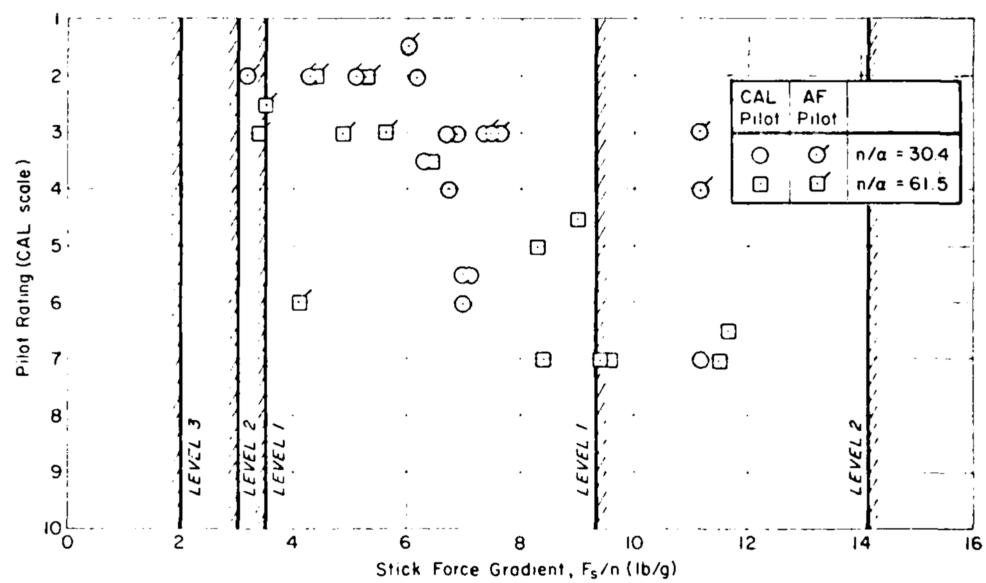


Figure 6 (3.2.9.1). Comparison of Optimum F_s/n with Limits of Table 1 (Reference 9, Category A; n_L = 7 g)

Data from Reference 10 (for "front-side" and "bottom" operations only) are shown in Figure 7. A much higher range of F_s/n was chosen for these (Category C) tests -- again, by the pilots at the start of each evaluation. These data were used as support for the adoption of n/α-dependence on $(F_s/n)_{max}$ in Reference 11. However, the goal of the experiments was again to investigate short-period dynamics and not F_s/n influences, so there is no single "constant" in the data. In addition, the task consisted of an instrument landing approach until 2 miles from the runway; visual approach to the threshold; and wave-off at 25-100 ft. There is evidence (e.g., Reference 5) that requiring a full approach through landing (wheels on runway) can produce quite different results than with a waveoff and go-around. This may be the reason that, of the 18 data points on Figure 7, only three have PR > 4 — generally less scatter than one might expect. (This will become clear when the Reference 5 data are introduced.)

The objective of the flight test program of Reference 12 was also an analysis of effects of short-period variations (in this case, through addition of higher-order lead/lag networks). Again, choice of "optimum" F_s/n was up to the pilots, and was specified before the rest of the evaluation task was performed. The data of Figure 8 are for only those configurations where ζ_e, ω_e, $1/T_{\theta_e}$, and τ_e are Level 1, based upon equivalent system matches and requirements. The data support the lower limits, but again suggest a smaller upper limit. Generally, the pilot ratings are Level 2 for $F_s/n > 6$ lb/g.

The LAHOS program of Reference 5 consisted of the most stringent set of tasks flown. Pilots were required to: a) fly an instrument approach to within 200 ft of the runway, followed by a visual landing; b) fly two visual landings with an intentional offset on close final; and c) land precisely at a marked location on the runway. These are clearly tight tasks requiring aggressive control actions by the evaluation pilots. A key difference between the Reference 5 program and that of Reference 10 was pilot selection of F_s/n: initial selection was made at the start of a run, but the gradient could be changed at any time during the run at the pilot's request. The range of F_s/n chosen by the

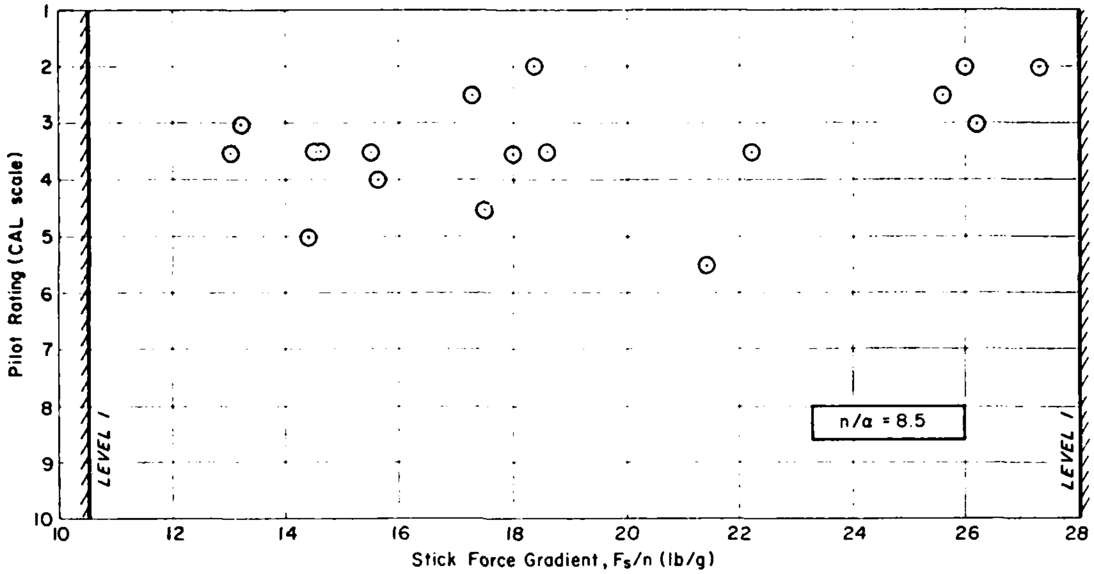


Figure 7 (3.2.9.1). Comparison of Optimum F_s/n with Limits of Table 1 (Reference 10, Category C; n_L = 3 g)

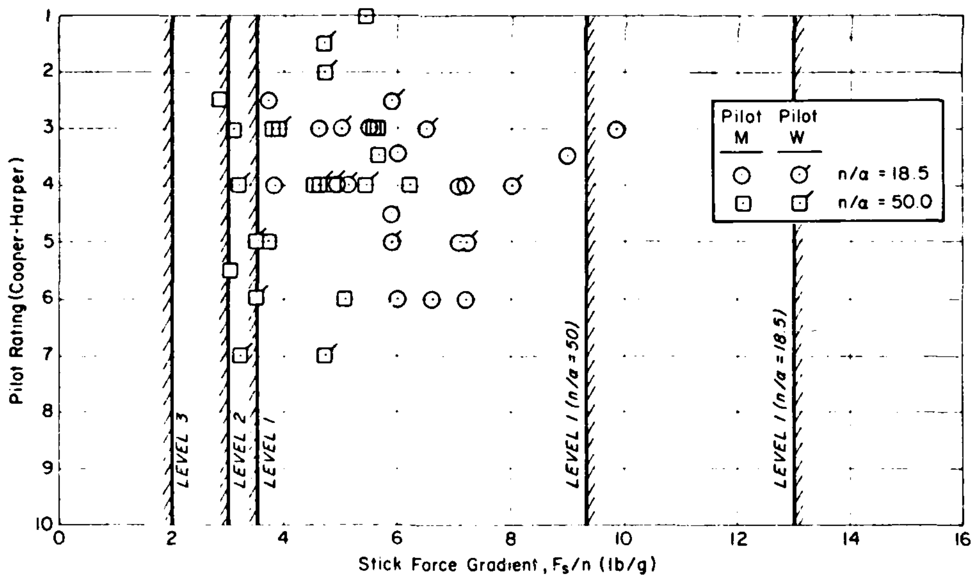


Figure 8 (3.2.9.1). Comparison of Optimum F_s/n from Data of Neal and Smith (Reference 12) with Limits of Table 1 (Category A; n_L = 7 g) pilots (Figure 9) is similar to that of Figure 7. (As before, only those data for which ζ_e, ω_e, $1/T_{\theta_e}$, and τ_e are Level 1 are shown.)

The pilot rating data of Figures 6, 7, 8, and 9 are compiled in Figure 10. (For clarity, the ratings have been averaged over 1 lb/g "slices" of F_s/n, to reduce the number of points shown. Standard deviations are also indicated.) Very few "optimum" gradients less than 3 lb/g were chosen. The gradients chosen tend to coalesce around 3-7 lb/g -- except those for References 5 and 10, which are for Category C operations. As discussed in "Guidance for Application," the real issue at very low n/α may not be F_s/n anyway, but $(F_s)_{max}$ at stall.

At this point the only conclusion to be drawn from Figure 10 is that there is a definite preference for low stick force gradients, between about 3 and 4 lb/g in Class IV airplanes. In addition, the overall range of selected gradients is small (except as noted above for References 5 and 10), 4-12 lb/g. This of course could be a function of other factors, such as short-period frequency and damping or stick displacement or location.

The wheel-controller data, from Reference 28, are shown in Figure 11. Both fixed and pilot-selected gradients were tested on the USAF/Calspan T-33, but there appear to be no rating differences. While the data are sparse, they indicate mild support for the Table 1 upper limit for Level 1. Within the Level 1 limits, 20 points out of 27 are rated 3-1/2 or better; outside the upper limit, 5 out of 6 have ratings greater than 3-1/2.

Two other data sources, References 3 and 24, were studied briefly for any additional information on an n/α-dependence. Reference 3 is a fixed-base simulator study utilizing a sidestick (modeled after the X-15 sidestick) with nonlinear deflection characteristics. The tasks generally were low-demand (including pilot-initiated disturbances) or required flying through rough air. It is felt that pilot preference in a fixed-base simulator may not reflect the real-world situation where the pilot is more conscious of the potential of overstressing the airplane.

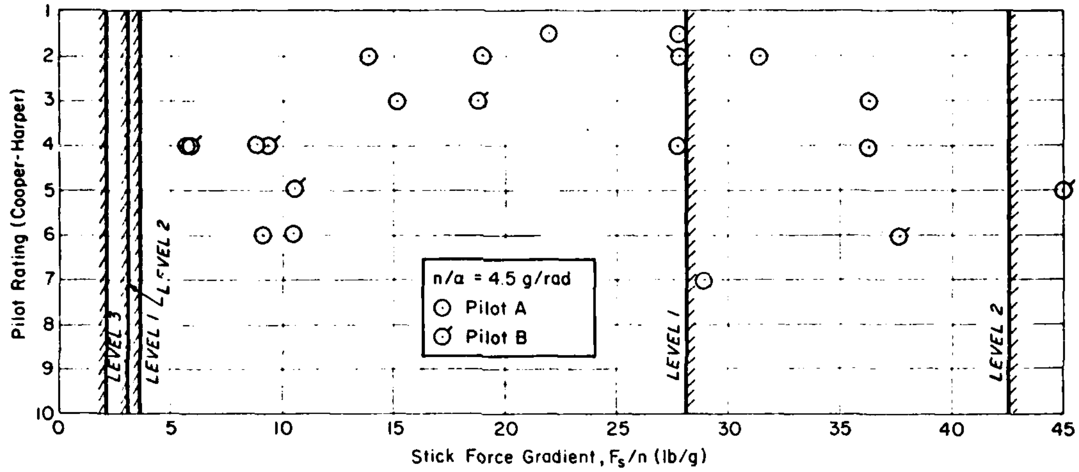


Figure 9 (3.2.9.1). Comparison of Optimum F_s/n from LAHOS (Reference 5) Data with Limits of Table 1 (Category C; n_L = 7 g)

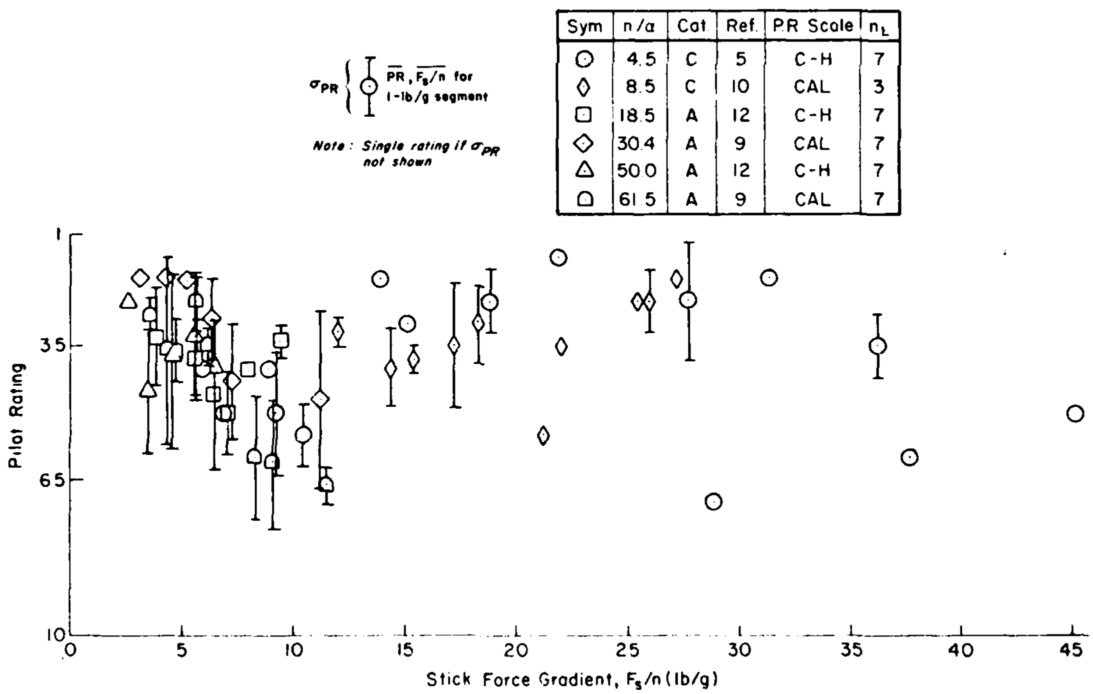


Figure 10 (3.2.9.1). Average Pilot Ratings for 1-lb/g Segments of F_s/n

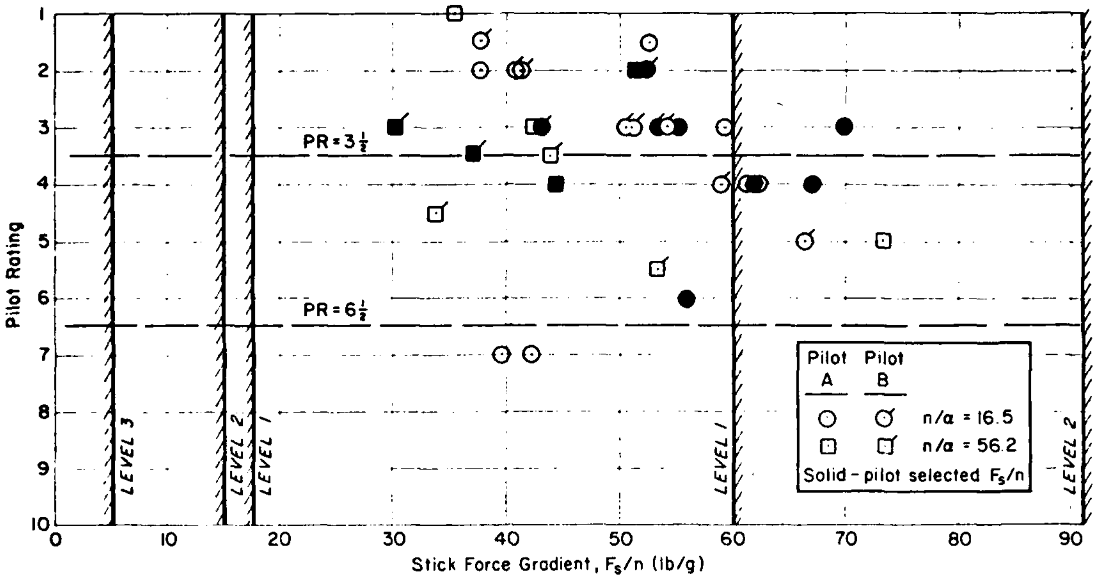


Figure 11 (3.2.9.1). Comparison of F_s/n with Limits of Table 1, Wheel Controllers (Reference 28, Category A; n_L = 3 g)

Reference 24 involved simulation of the C-5A with a B-26 in-flight simulator. Wheel travel was considered by the three evaluation pilots to be excessive. As a result, pilot ratings for both the unaugmented and augmented vehicle were poor, as summarized in the following conclusion:

> On the basis of the three-pilot sample as a whole, there is little conclusive evidence as to the relative desirability of the two stick forces per g evaluated..
> ..Pilot A most clearly indicated the desirability of the higher value (158 lb/g), particularly in the unaugmented case. Pilot B felt that this value was a bit high, but acceptable; Pilot C preferred the low value of 106 lb/g.

The bottom line of this discussion is: the supportive data necessary to fully validate a set of requirements for F_s/n do not seem to exist. It is felt that the requirements of Table 1, which are unchanged from MIL-F-8785C and little changed from MIL-F-8785B, will serve as a preliminary guide for controller design since nothing better is available. Ultimately a set of criteria might be devised in which displacements, gradients, and locations of the controllers are all interdependent -- as they must be in real life.

H. LESSONS LEARNED

Much of the available information on existing vehicles suggests that the lower limits of F_s/n for wheel controllers may be too high. A number of aircraft fail to meet the requirements; for some that do, extended operations lead to pilot fatigue. For example, Figure 12 (from Reference 14) summarizes elevator control force gradient characteristics for the C-5A, C-141A, YC-141B, and the L-1011 for Category B at forward and aft center of gravity conditions. This summary shows that the L-1011 complies with the maximum and minimum control force gradient requirements but that the C-5A and C-141A/YC-141B do not. The C-5A at forward c.g. compares favorably with the Level 1 maximum values; however, the gradients at aft c.g. fall below the Level 1 and Level 2 boundaries. 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. However, C-5A comments do not support the aft c.g. minimum boundary. The minimum boundary for Level 1 requirements appears to be too high in this instance.

Results from simulation and flight tests of different stick force gradients on the B-1 bomber (Reference 15) are summarized in Table 2. The pertinent findings were: 1) that a minimum F_s/n of 7-8 lb/g for Level 1 was acceptable for an airplane with $n_L = 3$ g [where, by Table 1, $(F_s/n)_{min} = 10.5$ lb/g]; and b) that F_s/n as low as 17 lb/g was too high for terrain-following flight, based upon pilot fatigue. The conclusions suggest 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 value 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.

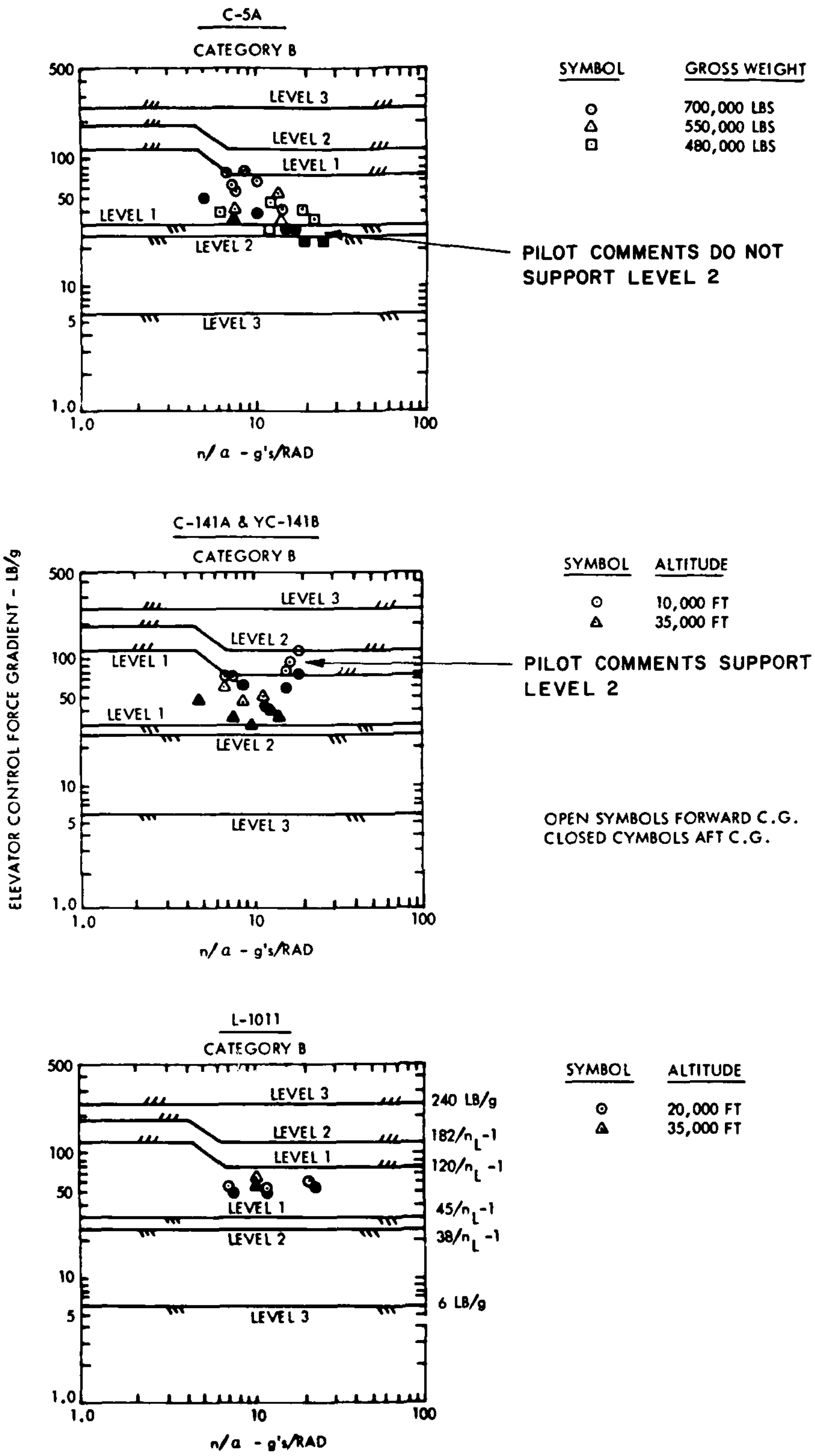


Figure 12 (3.2.9.1). Elevator Control Force Gradients for Transport Aircraft (from Reference 14)

TABLE 2 (3.2.9.1)

B-1 EXPERIENCE WITH STICK FORCE GRADIENTS
(From Reference 15)

Consolidated pilot comments - based on flight simulator tests

- 21 lb/g Level 1 minimum is too high for 2 g aircraft.
- 7-8 lb/g Level 1 minimum is good design guide for 3 g aircraft.
- 3 lb/g minimum should be maintained for failure modes

B-1 terrain following flight pilot comments

- 17 lb/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 - 2 pilots sharing task
  - Short task - 30 minutes
  - Medium task - 1 hour
  - Long task - 2 hours

Many Class I (general aviation) aircraft tend to fall around and below the minimum F_s/n requirements. Figure 13 compares various aircraft in landing configurations (gear and flaps extended) with the wheel-controller requirements of Table 1. The data were obtained from References 16 and 254. 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 13 meet the n_L = 2.0 $(F_s/n)_{min}$ requirement at aft c.g. The seven-aircraft study of Reference 16 resulted in a range of $19 < F_s/n < 45$ lb/g for acceptable gradients for Normal category

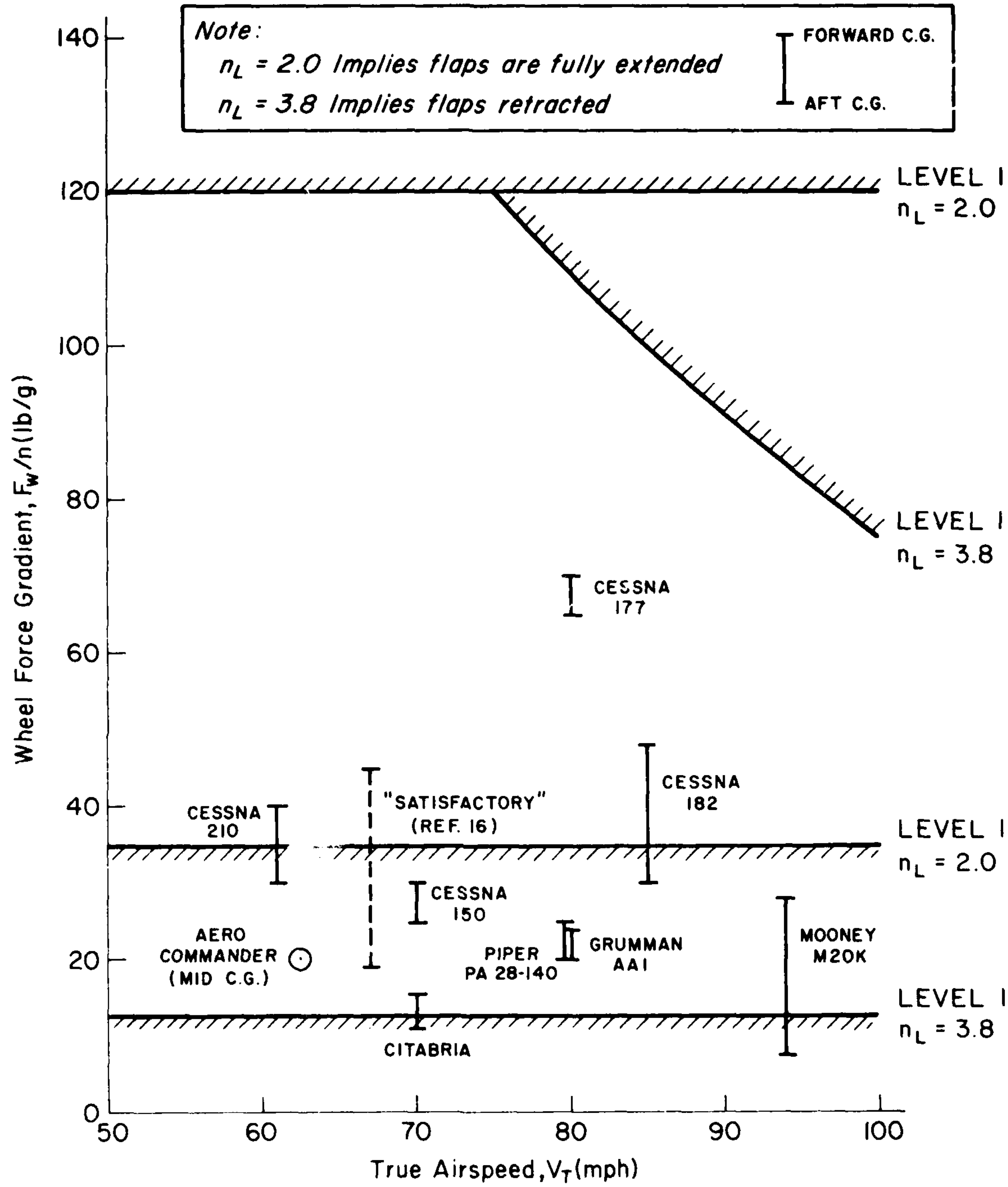


Figure 13 (3.2.9.1). Comparison of F_s/n for Various Class I Aircraft in Landing Approach (Category C) with Limits of Table I operations (see Figure 13). No information is available on what was considered unsatisfactory, though 5 lb/g was considered to be too light: "these low gradients allowed the limit load factor to be attained too easily."

3.2.9.2 Pitch axis control forces — transient control force per g

A. REASON FOR REQUIREMENT

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 Para. 3.2.9.1. Such reduced values of F_s/n may lead to pilot-induced oscillations. This requirement is intended to prevent such an occurrence.

B. RELATED MIL-F-8785C REQUIREMENTS

3.2.2.3.1, 3.2.2.3.2.

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.9.2 Pitch axis control forces -- transient control force per g. The buildup of control forces during maneuver entry must not lag the buildup of normal acceleration at the pilot's location. In addition, the frequency response of normal acceleration at the pilot station to pitch control force input shall have the following characteristics: ________________________.

The following values are recommended:

MINIMUM F_s/n AT ANY FREQUENCY GREATER THAN 1 RAD/SEC (Units are lb/g)

	Level 1	Level 2	Level 3
One-handed controllers	$\frac{14}{n_L - 1}$	$\frac{12}{n_L - 1}$	$\frac{8}{n_L - 1}$
Two-handed controllers	$\frac{30}{n_L - 1}$	$\frac{25}{n_L - 1}$	$\frac{17}{n_L - 1}$

D. RATIONALE BEHIND REQUIREMENT

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 n 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. If, in addition, 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 ω_{sp}, ζ_{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 pilot-induced oscillations. Consider, for example, an aircraft that meets the Level 2 requirements on ω_{sp}, ζ_{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 pilot-induced oscillation (PIO) tendencies serious enough to make it unacceptable. The requirements listed above are designed to prevent this situation, by setting an upper limit on the frequency-response amplitude of $n(s)/F_s(s)$ [expressed as a lower limit on $F_s(s)/n(s)$]. This has the effect of increasing the minimum F_s/n requirements of 3.2.9.1 for low values of ζ_{sp} (stick-free), as can be seen by examination of Figure 1. 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}). This functional relationship is shown in Figure 2. From this figure 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 $(F_s/n)_{min}$.

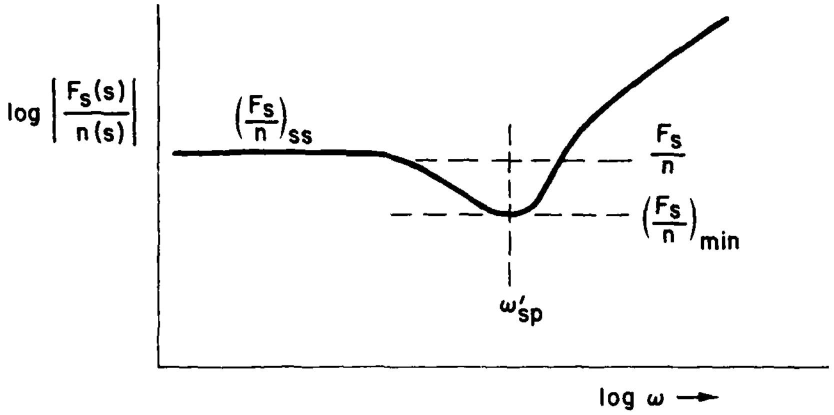


Figure 1 (3.2.9.2). Illustration of Resonance Dip in F_s/n Due to Low ζ_{sp}

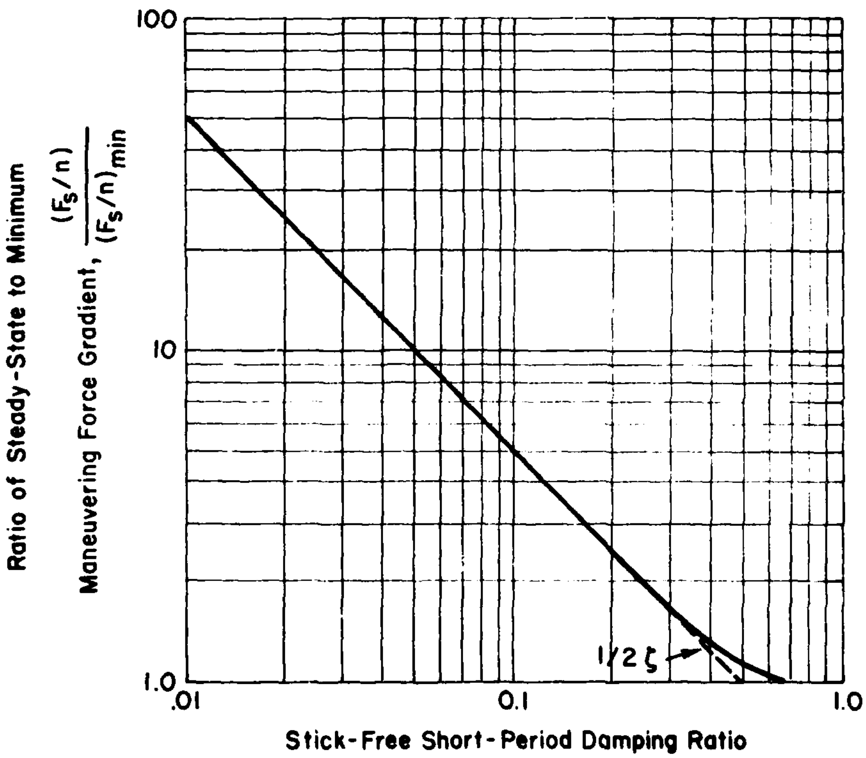


Figure 2 (3.2.9.2). The Ratio $(F_s/n)/(F_s/n)_{min}$ vs. ζ_{sp}

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 usually the case, the resonance dip can be accentuated by the control system mode, as can be seen from Figure 3. In this situation, an equivalent ζ_{sp} (stick-free) can be obtained from Figure 2 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 $(F_s/n)_{min}$ should be used.

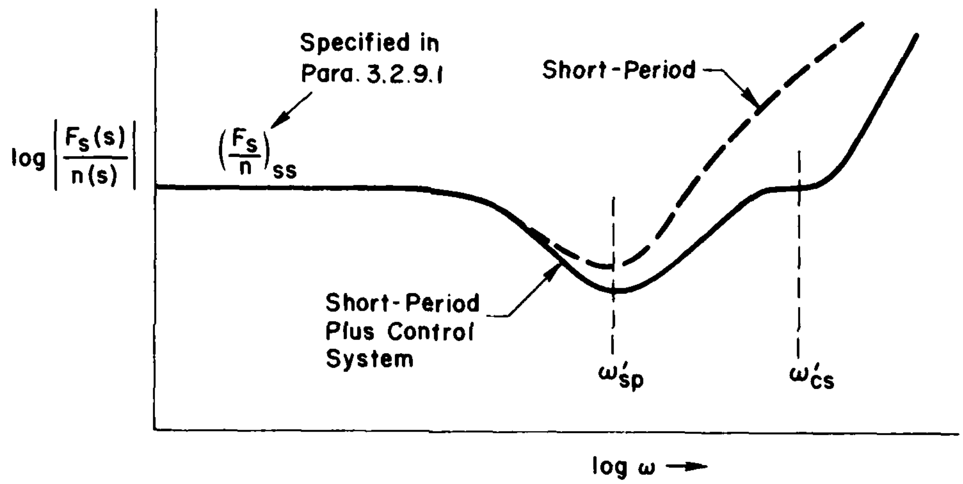


Figure 3 (3.2.9.2) Sketch of Effect of Control System on Resonant Dip

E. GUIDANCE FOR APPLICATION

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 2 is defined by F_s/n and ζ_{sp} when the control system natural frequency ω'_{cs} is well above the short-period frequency ω'_{sp}. Using the Figure 2 relationship between ζ_{sp} and $(F_s/n)/(F_s/n)_{min}$, the Level 1, 2, and 3 boundaries of $(F_s/n)_{min}$ have been superimposed on the F_s/n and ζ_{sp} requirements from Paragraphs

3.2.1.1 and 3.2.9.1 for a 7 g airplane. The shaded areas in Figure 4 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 control system or structural modes that occur near the short-period frequency will of course increase the influence of the $(F_s/n)_{min}$ requirement, i.e., the $(F_s/n)_{min}$ boundaries in Figure 4 will have a tendency to shift to the right.

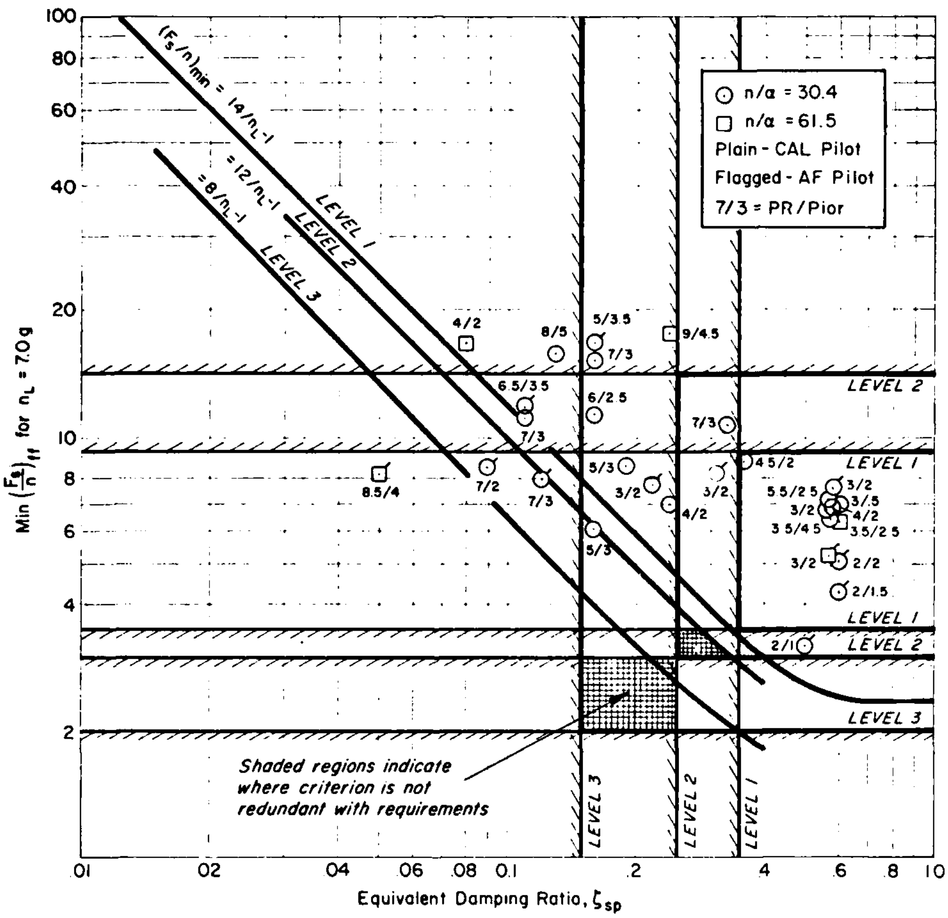


Figure 4 (3.2.9.2). Comparison of $(F_s/n)_{min}$ Boundaries with $(F_s/n)_{ss}$ and ζ_{sp} for Cases Where $\omega_{sp} << \omega_{cs}$ (Reference 9)

The T-33 test program of Reference 9 involved pilot assessments of various short-period configurations in Category A Flight Phases. The pilots were allowed to select optimum F_s/n values before evaluating each configuration. These data are plotted on Figure 4, which shows there was some preference for increasing F_s/n as ζ_{sp} decreased. [These data are discussed in more detail below under "Supporting Data."] The $(F_s/n)_{min}$ boundaries plotted in Figure 4 tend to correlate this rating behavior.

F. DEMONSTRATION OF COMPLIANCE

Testing is required with the aircraft configured for most aft c.g., since this is the condition for lowest F_s/n. For meaningful analysis or simulation, the linear approximation of the flight control system and airplane must be accurate. In the end, however, PIO tendencies need to be evaluated in flight. Ground-based simulator evaluations may be of little value. To obtain flight data, it will be necessary for the pilot to pump the stick sinusoidally at various frequencies. Several techniques have been employed to aid the pilot in this task. One method is described in Reference 17, 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.

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 airplane response for the least effort.

G. SUPPORTING DATA

Little is available in the way of supporting data. The Calspan flight test programs using the variable-stability T-33 provide the only significant data base. Those tests in which pilots chose "optimum" F_s/n for the short-period configurations under consideration allow some insight into the applicability of Para. 3.2.9.2.

Figure 4 shows data from Reference 9. Only those cases for which ω_{sp} is Level 1 are plotted. The controller characteristics such as breakout and friction are also Level 1. Low-speed data from the reference are not included because a 2 g n_L buffet limit may have influenced pilot ratings. The ratings given in Figure 4 are based on assessments of PIO tendencies and include both the handling qualities ratings (PR, from the CAL 10 point scale) and PIO ratings (PIOR, from Figure 5). The PIO ratings are closely correlated with the Cooper-Harper pilot ratings, as one would expect, since the PIO scale is worded in terms of closed-loop pilot control.

DESCRIPTION	NUMERICAL RATING
NO TENDENCY FOR PILOT TO INDUCE UNDESIRABLE MOTIONS	1
UNDESIRABLE MOTIONS TEND TO OCCUR WHEN PILOT INITIATES ABRUPT MANEUVERS OR ATTEMPTS TIGHT CONTROL. THESE MOTIONS CAN BE PREVENTED OR ELIMINATED BY PILOT TECHNIQUE.	2
UNDESIRABLE MOTIONS EASILY INDUCED WHEN PILOT INITIATES ABRUPT MANEUVERS OR ATTEMPTS TIGHT CONTROL. THESE MOTIONS CAN BE PREVENTED OR ELIMINATED BUT ONLY AT SACRIFICE TO TASK PERFORMANCE OR THROUGH CONSIDERABLE PILOT ATTENTION AND EFFORT.	3
OSCILLATIONS TEND TO DEVELOP WHEN PILOT INITIATES ABRUPT MANEUVERS OR ATTEMPTS TIGHT CONTROL. PILOT MUST REDUCE GAIN OR ABANDON TASK TO RECOVER.	4
DIVERGENT OSCILLATIONS TEND TO DEVELOP WHEN PILOT INITIATES ABRUPT MANEUVERS OR ATTEMPTS TIGHT CONTROL. PILOT MUST OPEN LOOP BY RELEASING OR FREEZING THE STICK.	5
DISTURBANCE OR NORMAL PILOT CONTROL MAY CAUSE DIVERGENT OSCILLATION. PILOT MUST OPEN CONTROL LOOP BY RELEASING OR FREEZING THE STICK.	6

Figure 5 (3.2.9.2). PIO Tendency Rating Scale

As Figure 4 indicates, there is a preference for high F_s/n when ζ_{sp} is low. This is not surprising, since very large gradients would tend to inhibit pilot overcontrol and reduce the tendency to PIO. However, from the small amount of data at low ζ_{sp} in Figure 4 it is difficult to conclude that the F_s/n requirement of Para. 3.2.9.2 is necessary. There is no clear degradation in ratings at low ζ_{sp} as F_s/n is reduced, nor is there data at low ζ_{sp} <u>and</u> low F_s/n to show that the pilot would consider this condition to be worse.

Figure 6 supports the conclusions from Figure 4, i.e., there is a preference for large gradients at low damping but little to support the

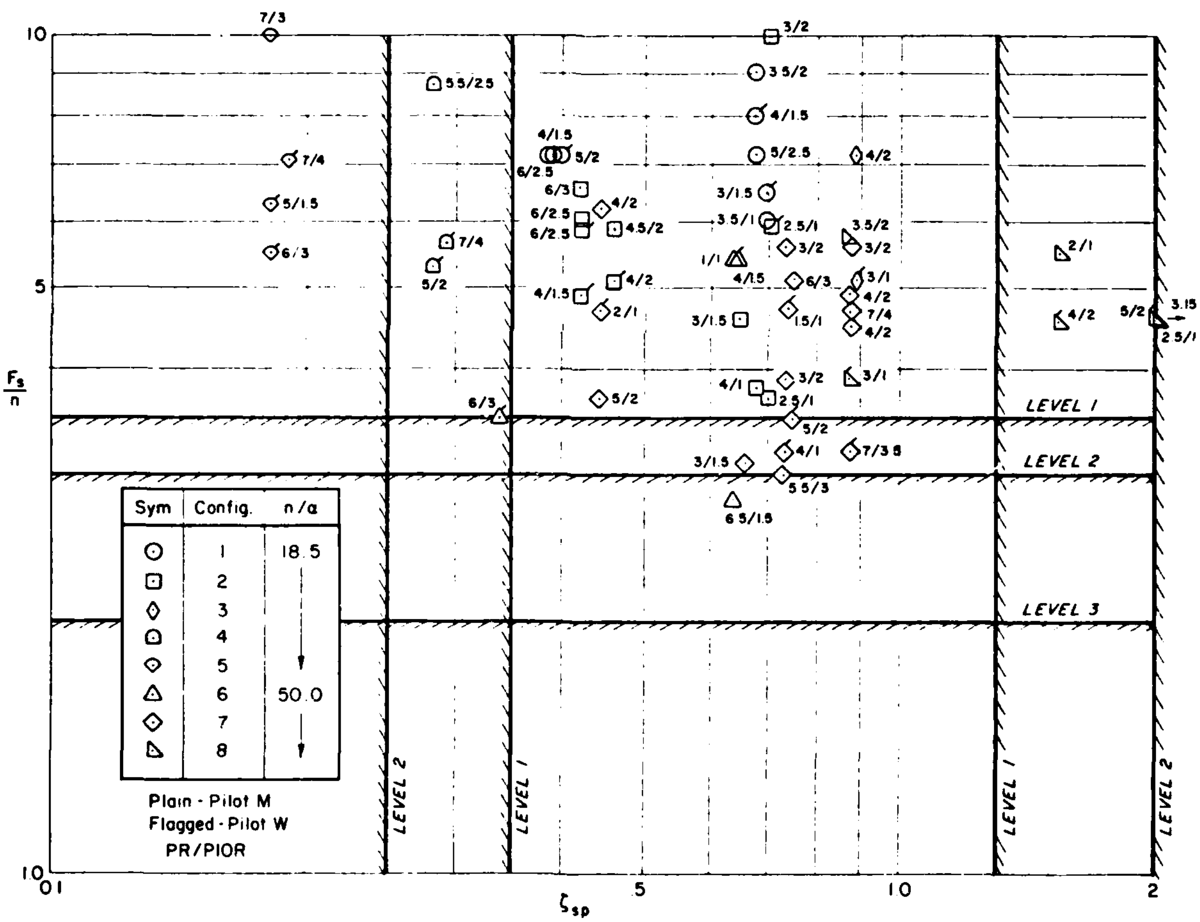


Figure 6 (3.2.9.2). T-33 Data from Reference 12
(Equivalent ω_{sp}, τ_e are Level 1)

need for the dynamic F_s/n requirement. These data, from the tests of Neal and Smith (Reference 12), include those configurations for which both equivalent ω_{sp} and τ_e were Level 1 in value. The pilot ratings are based on the Cooper-Harper scale, and the PIO ratings on the scale of Figure 5. Thus, the basis for this requirement remains theoretical with the additional thought that it may catch PIO tendencies in some higher-order systems that might otherwise escape detection until extensive flight experience has accumulated.

H. LESSONS LEARNED

The data used in Reference 11 are reexamined here. Figures 7 and 8 are reproduced from Reference 11.

Figure 7 is a good illustration of lessons learned regarding the need for control system modification. In each case the airplane with the original control system exhibited strong PIO tendencies in the high-speed, low-altitude flight regime. A modified control system was tried in each airplane, which significantly improved the situation. The majority of the points in Figure 7 are for the A4D-2 (Reference 18). The T-38A and F-4C data are from flight test (References 19 and 20, respectively). With the exception of the T-38A, there are no pilot ratings or detailed pilot comments available. It is only known that the shaded points of Figure 7 are associated with strong PIO tendencies. The solid line for $(F_s/n)_{min} = 1.4$ lb/g divides the data very nicely.

Some additional data on PIO tendencies are presented in Figure 8, taken from Reference 21. The points are again rather crudely divided into those cases that exhibited PIO tendencies and those that did not. Since little is known about the severity of the PIO problems associated with these airplanes, Figure 8 is used only to establish trends. As can be seen from the figure, a line of constant $(F_s/n)_{min}$ also fits these data very well.

The lines of constant $(F_s/n)_{min}$ in Figures 7 and 8 were obtained from Figure 4. Note, however, that while the $(F_s/n)_{min}$ lines fit the data, so do lines for $\zeta_{sp} \approx 0.15$ and $F_s/n \approx 3.0$. And, as was stated

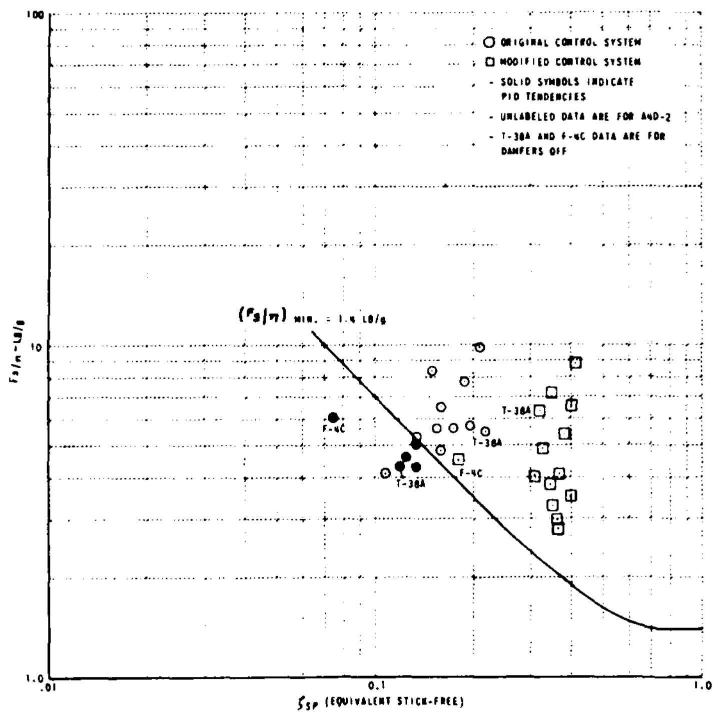


Figure 7 (3.2.9.2). PIO Characteristics of A4D-2, T-38A, and F-4C (References 18, 19, 20)

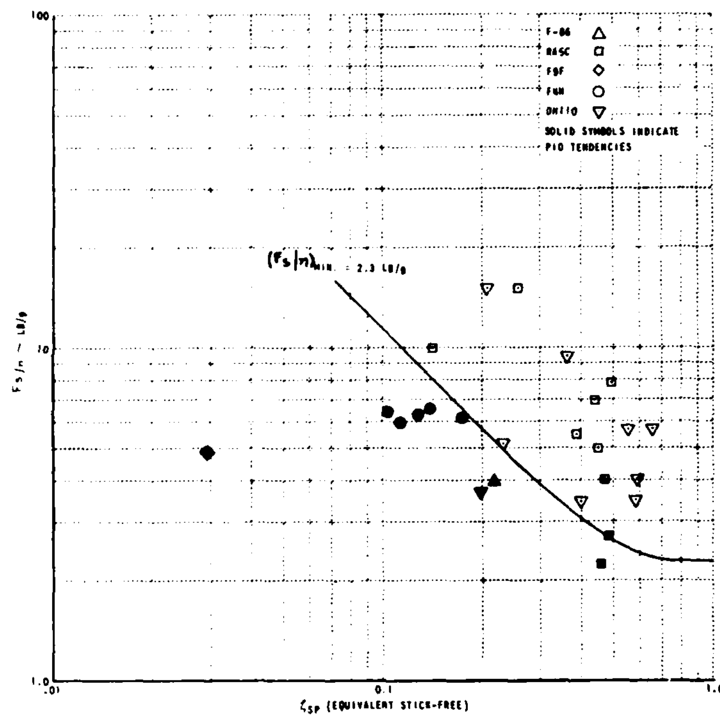


Figure 8 (3.2.9.2). PIO Characteristics of Airplanes Described in Reference 21 earlier, many PIO tendencies are characteristically due to low ζ_{sp}. Therefore the data of Figures 7 and 8 do not reveal any requirement for a $(F_s/n)_{min}$ specification. Such a requirement would be supported by obtaining test data in the shaded regions of Figure 4 or by introducing lightly damped modes that influence the equivalent ζ_{sp}, i.e., which make $\zeta_e < \zeta_{sp}$.

3.2.9.3 Pitch axis control forces -- control force variations during rapid speed changes

A. REASON FOR REQUIREMENT

This is intended to prevent unduly large pitch control force gradients with speed, requiring excessive trimming or high steady control force.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.1.1.2.

C. STATEMENT OF REQUIREMENT

3.2.9.3 Pitch axis control forces -- 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.

D. RATIONALE BEHIND REQUIREMENT

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.

If the c.g. is allowed to be farther aft at supersonic speeds than at subsonic speeds, an adequate rate of c.g. shift should be provided for rapid transonic deceleration.

E. GUIDANCE FOR APPLICATION

Application of the requirement should be straightforward, except that its wording is sufficiently general that it becomes a purely subjective specification. 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.

F. DEMONSTRATION OF COMPLIANCE

Early analysis can determine transonic pitching and control forces to maintain 1 g flight as well as normal acceleration with fixed controls.

Flight or simulator testing, covering the operational speed range (and the transonic speed range, if applicable), utilizing maneuvers mentioned in Para. 3.2.9.1 should be conducted.

G. SUPPORTING DATA

No supporting data are available for this requirement.

H. 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.

3.2.9.4 Pitch axis control forces -- control force vs. control deflection

3.2.9.4.1 Steady-state control force/deflection gradient

A. REASON FOR REQUIREMENT

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.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.2.2.2.

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.9.4.1 Steady-state control force/deflection gradient. The average gradient of pitch-control force per unit of pitch-control deflection at constant speed shall be within the following range: ____________________.

RECOMMENDED ALLOWABLE VALUES (FOR CATEGORY A FLIGHT PHASES)

	MINIMUM	MAXIMUM
Wheel and centerstick controllers	5 lb/in.	No data
Sidestick controllers	1 lb/in.	2.5 lb/in.

D. RATIONALE BEHIND REQUIREMENT

Paragraphs 3.2.9.1 and 3.2.9.2 set limits on allowable values of pitch control force. This requirement extends the force requirement to set limits on controller deflections required to attain a given force. The "Supporting Data" show evidence of a need for such a requirement, and intuitive logic indicates that some interrelationship exists between control forces, deflections, and positions [see also 3.2.9.1, Control force per g].

E. GUIDANCE FOR APPLICATION

Limited flight test data show strong support for a lower limit on F_s/δ_s (indeed, Reference 22 suggests that F_s/δ_s should be 25 lb/in. or higher for centerstick controllers). The specific limits are directly related to F_s/n and control location (see 3.2.9.1). However, they are not well defined at this time.

Some guidance for designing sidestick controllers may be gained from Figure 1 (reproduced with minor changes from Reference 23). A sidestick evaluation performed by the USAF Test Pilot School, Edwards AFB, using the variable-stability T-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 1 to maintain control harmony for the air-to-air evaluations (consisting of 2 g bank-to-bank and 3.5 g wind-up turns). If the pilot comments indicated 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 1 yielded the best results, both in pilot ratings and comments. Pilots indicated that control motions were noticeably large but not uncomfortable. 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

Initial Longitudinal Control Force/Response Gradient (lb/g)	1.1	1.4	2.0	5.0
Very Light (3.0)	13 No pitch bobble tendency but imprecise positioning. Avg of CH 3.7	9 Pitch and lateral are both too sensitive. Avg of CH 4.4	5 Pitch and lateral both a little too sensitive. Avg of CH 5.1	1 Pitch extremely sensitive. Lateral fair. Avg of CH 6.7
Light (4.0)	14 Pitch and lateral steady and responsive. Motion noticeably large. Avg of CH 2.9	10 Pitch a little sensitive. Lateral bobble. Avg of CH 4.3	6 Slight pitch bobble. Better at higher g's. Lateral sluggish (cont. harmony). Avg of CH 4.5	2 Pitch too sensitive. Lateral wandering and sensitive. Avg of CH 6.0
Medium (5.0)	15 Motion noticeably large. No pitch bobble, slightly sluggish. Avg of CH 3.3	11 Very slight pitch bobble tendency, but good. Large lateral corrections difficult Forces high & bobble Avg of CH 4.4	7 Pitch steady once on tgt. Lateral forces high (cont. harmony) Avg of CH 3.85	3 Pitch a little sensitive. Lateral slow to respond. Avg of CH 5.0
Heavy (8.6)	16 A/C very sluggish and forces uncomfortable. Avg of CH 5.0	12 A/C sluggish but stable. Forces heavy. Avg of CH 4.5	8 Pitch steady but forces too heavy. Lateral forces to neavy. Tiring. Avg of CH 4.3	4 Pitch very stable at higher g's, but forces tiring. Avg of CH 4.1

Control Force/Deflection Gradient (lb/deg)

Figure 1 (3.2.9.4.1). Pilot Comments for Air-to-Air Tasks with Standard Harmony (from Reference 23)

TABLE 1 (3.2.9.4.1). CATEGORY A CONTROL CONFIGURATION FOR T-33 SIDESTICK EVALUATIONS (REFERENCE 23)

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	Very 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 boundaries were not completely determined, these comments imply that even heavier force gradients would be unacceptable.

Configurations 1 and 2 were rated the poorest. They were 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 1/2 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 T-33 sidearm controller varied the force/deflection gradient F_s/δ and short-period frequency, ω_{sp}. The stick force per g (F_s/n) was 7 or 8 lb/g for the high ω_{sp} and 5 lb/g for the lower values of ω_{sp} in accordance with earlier results [see Figure 7 (3.2.9.1)] as shown in Table 2. A summary of average pilot ratings (3 pilots) and commentary is given in Figure 2 for the gross acquisition task and in Figure 3 for the fine tracking maneuver.

The poor pilot ratings for the low short-period frequency cases, ω_{sp} = 1.6 rad/sec, are expected based on Para. 3.2.1.1. However, the ratings for the lower values of F_s/δ_s are worse than for Level 2, indicating that failure modes should be a consideration when contemplating light force/deflection gradients.

For ω_{sp} in the Level 1 region (ω_{sp} > 2.55), 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 3).

The "optimum" value of F_s/δ is seen to be about 1.7 lb/deg until ω_{sp} > 5 rad/sec, at which time the data indicate that decreasing F_s/δ is desirable (see Figure 2).

F. DEMONSTRATION OF COMPLIANCE

Flight testing performed to demonstrate compliance with 3.2.9.1, "Control force per g," should include measurements of δ_s/n. For discussion of flight test techniques, see 3.2.9.1.

G. SUPPORTING DATA

There is little or no recent data available for analysis. The following discussion is taken directly from Reference 11, with a few words added at the end. Data for sidestick controllers was introduced under "Guidance for Application."

TABLE 2 (3.2.9.4.1). 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 (lb/deg)	LONGITUDINAL FORCE/RESPONSE (lb/g)	LATERAL FORCE/DEFLECTION (lb/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	.95
E	2.6	1.7	5	.95
F	2.6	1.1	5	.95
G	1.8	3.3	5	.95
H	1.8	1.7	5	.95
I	1.8	1.1	5	.95

PHASE II; AIR-TO-AIR TEST POINTS
(13,000 FT MSL, 300 KIAS)

CONFIGURATION	ω_{sp} (rad/sec)	LONGITUDINAL FORCE/DEFLECTION (lb/deg)	LONGITUDINAL FORCE/RESPONSE (lb/G)	LATERAL FORCE/DEFLECTION (lb/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	.95

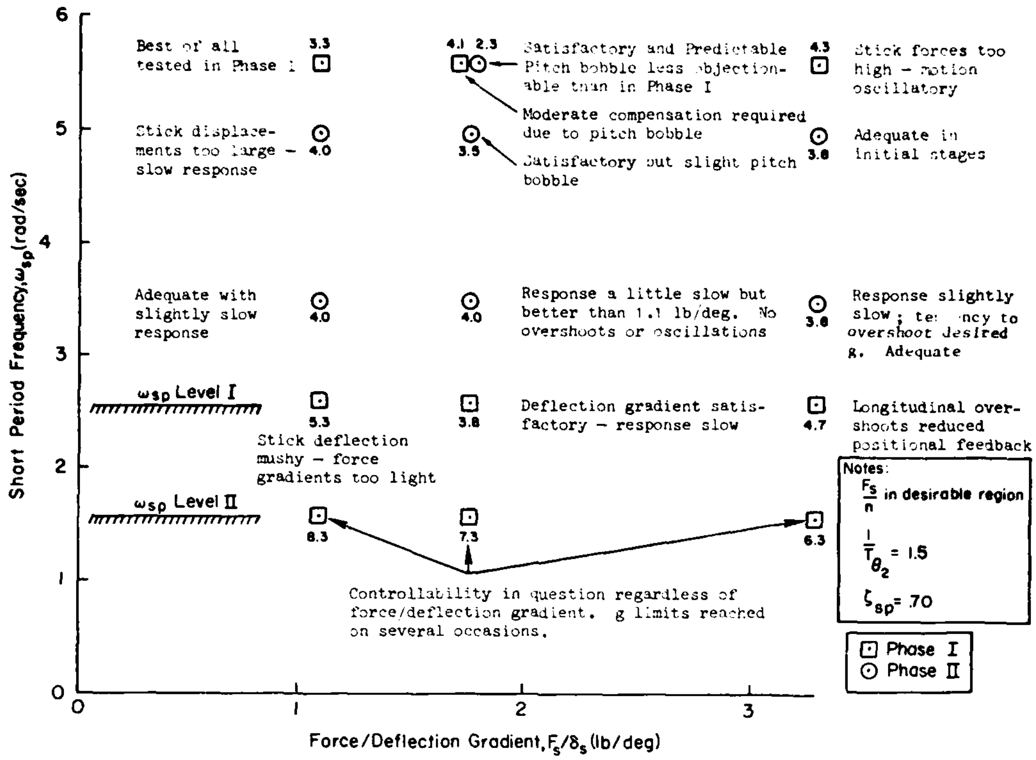


Figure 2 (3.2.9.4.1). Average Pilot Ratings for Gross Acquisition Task

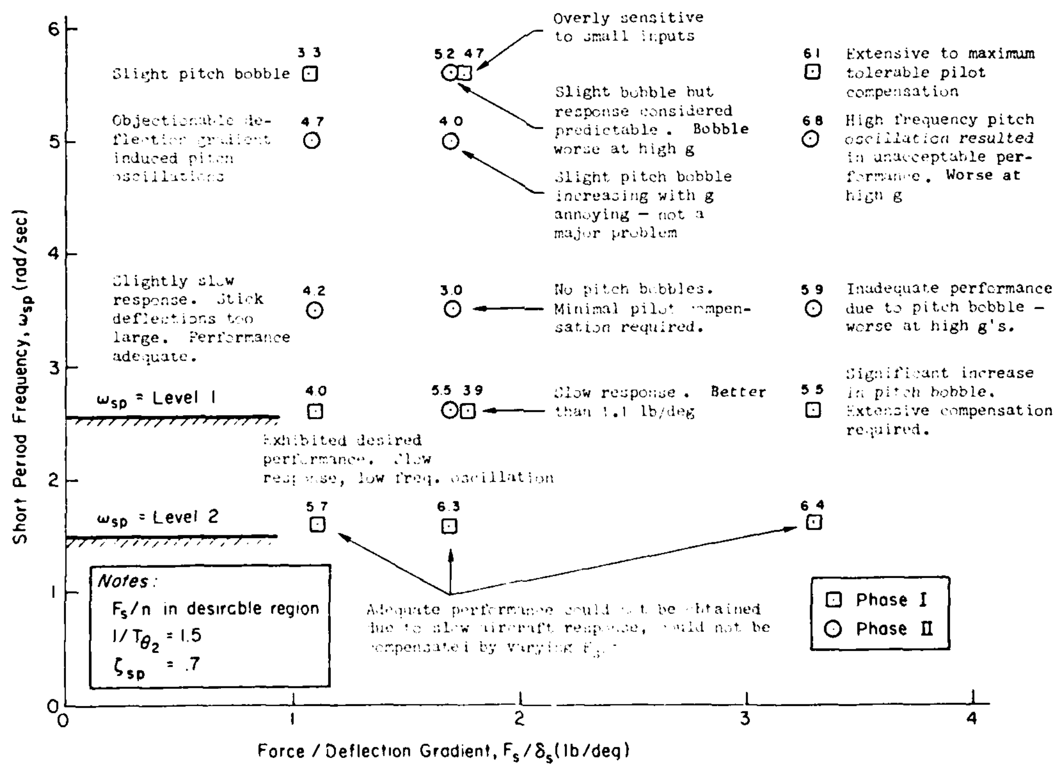


Figure 3 (3.2.9.4.1). Average Pilot Ratings For Fine Tracking Tasks

The flying qualities investigations of References 22, 24-28 all included variations of control position per g as well as control force per g. References 24 and 25 deal with the landing approach flight phase, while all the others are for Category A Flight Phases.

Both References 24 and 25 indicate unfavorable pilot comments when the control motions required to maneuver the airplane become too large. Since these investigations were specific simulations of some C-5A configurations, the short-period natural frequency was below the minimum 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 4. 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 References 22 and 26 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 in Figure 12 were reduced to 5 lb/in. Examples of "good" operational airplanes were produced with indicated gradients as low as 5 lb/in. The requirement for a force/deflection gradient of at least 5 lb/in. has been retained as a recommended lower limit from Paragraph 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.

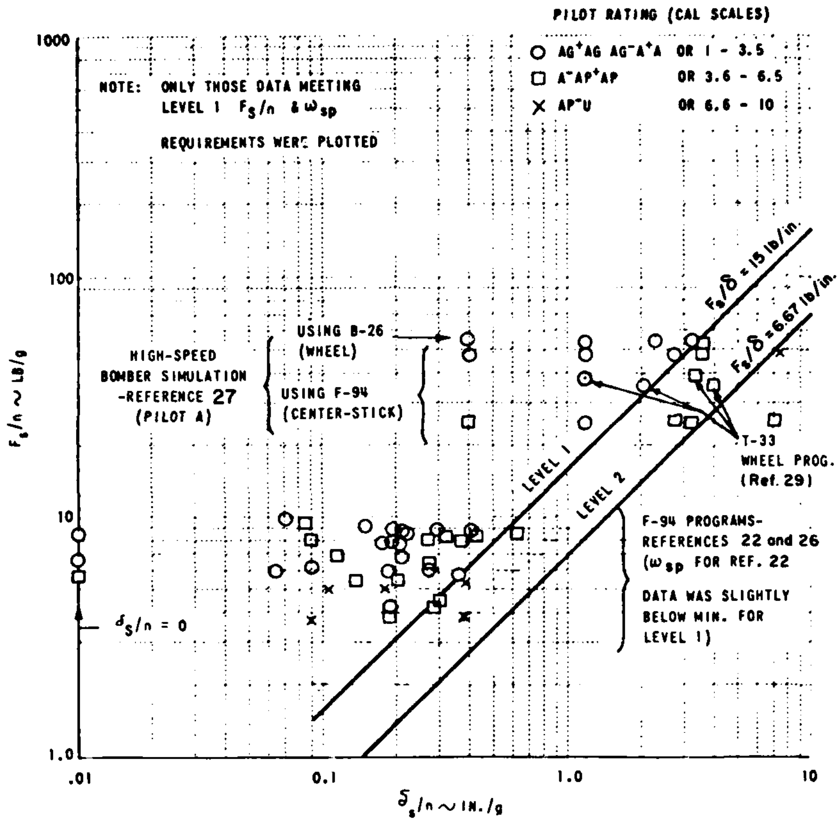


Figure 4 (3.2.9.4.1). Control Force Per Control Displacement Category A Flight Phases — Centerstick (from Reference 11)

H. LESSONS LEARNED

The F-16 employs an essentially fixed sidearm controller ($F_s/\delta \approx \infty$), which clearly would not meet the requirements of this section. F-16 pilots seem to have adapted to the essentially fixed sidestick. However, before accepting such a controller as being Level 1, it should be noted that these pilots had no alternative but to adapt. The T-33 subject pilots (Figure 1) did not feel that extreme force gradients were desirable when given the opportunity to compare with gradients across the spectrum.

3.2.9.4.2 Transient control force vs. deflection

A. REASON FOR REQUIREMENT

Experience has shown that a strong tendency for PIO exists when control deflection leads control force. This requirement is included to eliminate this problem.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.2.3.2.

C. STATEMENT OF REQUIREMENT

3.2.9.4.2 Transient control force vs. deflection The deflection of the pilot's control must not lead the control force throughout the frequency range of pilot control inputs. In addition, the peak pitch control forces developed during abrupt maneuvers shall not be objectionably light.

D. RATIONALE BEHIND REQUIREMENT

The minimum values specified for F_s/n in Paragraph 3.2.9.2 and ζ_{sp} in Paragraph 3.2.1.1 are not sufficient to prevent the occurrence of a PIO. In fact, there are documented cases of PIO-prone aircraft with Level 1 values of $(F_s/n)_{min}$ and $(\zeta_{sp})_{min}$. 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 in some length in the "Lessons Learned" subsection.

E. GUIDANCE FOR APPLICATION

Given some very basic instrumentation,* 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.

*A control position potentiometer and strain gauge to measure control force are required.

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.

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.

F. DEMONSTRATION OF COMPLIANCE

Demonstrating compliance with this requirement requires a quantitative determination of the phase relationship between control deflection and force. However, if the control forces clearly lead control deflection based upon pilot comments from simulation and/or flight test, the requirement should be considered as having been 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 highly desirable.

G. SUPPORTING DATA

The supporting data for this requirement are based on experience where PIOs occurred. Accordingly, the background material for this requirement is presented in the "Lessons Learned" subsection.

H. LESSONS LEARNED

An excellent treatise by Peter Neal on the historical development and analytic aspects of this requirement is given in References 11 and 183. 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., presenting large changes with attitude and loading (c.g.). Early versions, involving manual control, featured elevators with near 100 percent aerodynamic balance. Later versions featured full-power hydraulics with artificial (springs and/or 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. References 11 and 183 provide a feedback-control-analysis basis for studying the problems 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, i.e., P-63A (Reference 135), A4D-2 (Reference 18), T-39A (References 19, 140) and F-4 (Reference 20).

Pertinent conclusions reached by Neal (Reference 183) are:

1) The use of a control-system bobweight without consideration of its effects on the airplane'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, using the following criterion:

$$l_b = 115 + l_{cr}$$

where:

l_b = distance (in feet) of equivalent point-mass bobweight ahead

of c.g. $= g \dfrac{\text{bobweight stick force due to unit } \ddot{\theta}}{\text{bobweight stick force due to unit } n_z}$

l_{cr} = distance in (feet) of elevator center of rotation ahead

of c.g. $= \left(Z_{\delta_e}/M_{\delta_e}\right)$

3) When the previous 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 should 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.

3.2.9.5 Pitch axis control forces — control centering and breakout forces

A. REASON FOR REQUIREMENT

Controls should have clearly defined neutral positions and should always tend to return to neutral when released by the pilot. In addition, some effort should be necessary to deflect a control out of its neutral position. These in combination serve as the major initial pilot cues of control motion.

B. RELATED MIL-F-8785C REQUIREMENTS

3.5.2.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.9.5 Pitch axis control forces — control centering and breakout forces. Longitudinal controls should 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. Breakout forces, including friction, preload, etc., shall be within the following limits: ____________________. These values refer to the cockpit control force required to start movement of the control surface.

TABLE 1 (3.2.9.5). RECOMMENDED PITCH AXIS BREAKOUT FORCES (LB)

CONTROL	CLASSES I, II-C, IV		CLASSES II-L, III	
	MINIMUM	MAXIMUM	MINIMUM	MAXIMUM
Centerstick	1/2	3	1/2	5
Wheel	1/2	4	1/2	7
Sidestick	1/2	1	1/2	1

Values for Levels 1 and 2 (Upper Limits Doubled for Level 3)

D. RATIONALE BEHIND REQUIREMENT

A discernible neutral point (or trim or equilibrium point) should always be provided in manual pitch, roll, or yaw controllers. That is, if the pilot chooses to release a control it should return to a neutral or trim state. If no cues are provided, the pilot will be forced to manually search for such a trim condition. This can lead to poor maneuvering control or, in the extreme, to pilot-induced oscillations. The sidestick breakout forces are based on recommendations of Reference 23.

E. GUIDANCE FOR APPLICATION

While this paragraph does not specify absolute centering, the tendency for positive centering should always be detectable. 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.

Several studies (References 45 and 126) have substantiated the upper limits on breakout forces in Table 1. Design for the lower values is recommended, since operation with large breakout forces could contribute to pilot fatigue.

F. DEMONSTRATION OF COMPLIANCE

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.

G. SUPPORTING DATA

None available.

H. LESSONS LEARNED

None.

3.2.9.6 <u>Pitch axis control forces -- free play</u>

A. REASON FOR REQUIREMENT

While some small amount of free play may be desirable to reduce sensitivity to inadvertent control motions, the hysteresis resulting from combined free play and friction will create generally deleterious phase lags. Free play by itself (threshold) has no associated phase lag but rather its quasilinear describing function is a reduced gain.

B. RELATED MIL-F-8785C REQUIREMENT

3.5.2.2

C. STATEMENT OF REQUIREMENT

3.2.9.6 <u>Pitch axis control forces -- free play</u>. The free play (and possible associated hysteresis) in the longitudinal controller shall not result in objectionable flight characteristics, especially for small amplitude inputs. Hysteresis and free play should be within the following boundaries: ________________.

D. RATIONALE BEHIND REQUIREMENT

The requirement is designed to prevent unacceptable dead zones in the pitch controller. In normal operations, and especially in high-demand tasks such as turbulence penetration or air combat, free play can contribute to overcontrol and rapid pilot fatigue; associated possible hysteresis can contribute to reduced damping and possible limit cycle operation.

No numerical values have yet been found that appear 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; also of the friction level and resulting hysteresis loop. As of this writing the Air Force has the McDonnell Aircraft Company under contract to obtain more definitive data on controllers.

E. GUIDANCE FOR APPLICATION

This requirement is not intended to eliminate all free play (or associated hysteresis). Free play is sometimes designed into a controller to provide a dead zone, and some amount of basic free play is inherent without special measures, e.g., the use of preload.

F. DEMONSTRATION OF COMPLIANCE

Evaluation of free play (and hysteresis) in flight should 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.

G. SUPPORTING DATA

None available.

H. LESSONS LEARNED

None available.

3.2.9.7 Pitch axis control force limits

3.2.9.7.1 Pitch axis control force limits — takeoff

A. REASON FOR REQUIREMENT

Limits on maximum push and pull forces required for takeoff should be lower than those allowed for other operations, i.e., by the F_s/n values of 3.2.9.1. This is to assure that the pitch control input for takeoff will not be abrupt or require two-handed operation.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.3.3.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.9.7.1 Pitch axis control force limits -- takeoff. With the trim setting optional but fixed, the pitch-control forces required during all types of takeoffs for which the aircraft is designed, including short-field takeoffs and assisted takeoffs such as catapult or rocket-augmented, shall be within the following limits: ____________.

RECOMMENDED VALUES FOR AIRCRAFT WITH CENTERSTICK OR WHEEL CONTROLLERS

Nose-wheel and bicycle-gear airplanes

Classes I, IV-C:	20 pounds pull to 10 pounds push
Classes II-C, IV-L:	30 pounds pull to 10 pounds push
Classes II-L, III:	50 pounds pull to 20 pounds push

Tail-wheel airplanes

Classes I, II-C, IV:	20 pounds push to 10 pounds pull
Classes II-L, III:	35 pounds push to 15 pounds pull

For sidestick controllers the force should not be objectionable to the pilot. 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 in both the presence and absence of ground effect. Takeoff power should 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).

D. RATIONALE BEHIND REQUIREMENT

The force limits for this paragraph are intended to insure adequate control force characteristics at takeoff. It is obvious that pitch forces in takeoff should not place unreasonable demands on the pilot, either in the form of large pull forces (requiring two-handed operation and possibly causing abrupt responses) or large push forces (resulting in unnatural motions). The limits are strictest for small, highly maneuverable (Class I) aircraft and relaxed for large (Class III) aircraft. For tailwheel airplanes the push forces required to raise the tail may be larger than pull forces. At this time there is 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.

E. GUIDANCE FOR APPLICATION

The force limits are specified as absolute requirements. Therefore, for centerstick and wheel controllers, longitudinal control forces on takeoff must at all times be within the specified limits; for sidestick controllers, in the absence of definitive guidance, the forces must not be objectionable to the pilot.

F. DEMONSTRATION OF COMPLIANCE

The final proof of compliance will be flight testing. Such testing will include not only normal takeoffs, but any takeoffs which may be peculiar to the airplane mission requirements. Tests should be conducted at the conditions for most forward and most aft center-of-gravity positions, and will cover the velocity range from 0 to V_{max} (TO).

Analysis and simulation may be used for early guidance, but these will be only as good as the initial estimates of elevator control power and airplane performance characteristics.

G. SUPPORTING DATA

There are no known systematic flight tests of these requirements.

H. LESSONS LEARNED

As with supporting data, little information can be found for existing flight vehicles. Reference 126, a validation of MIL-F-8785B using a Class III-L airplane (P-3B), indicates pilot support for the limits for Class III, nosewheel-equipped airplanes.

3.2.9.7.2 pitch axis control force limits — landing

A. REASON FOR REQUIREMENT

The forces required in landing should always be natural (i.e., pull for flare) and should be small enough to allow one-handed operation without placing excessive demands on the pilot.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.3.4.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.9.7.2 Pitch axis control force limits -- landing. The pitch control forces for landing shall be less than ________________ for the recommended approach speed and fixed trim settings. This applies in both presence and absence of ground effect.

RECOMMENDED VALUES FOR CENTERSTICK
OR WHEEL CONTROLLERS

Classes I, II-C, IV:	35 pounds pull
Classes II-L, III:	50 pounds pull

For sidestick controllers the forces should not be objectionable to the pilot.

D. RATIONALE BEHIND REQUIREMENT

The limits for aircraft with centerstick or wheel controllers are intended to prevent unreasonable demands on the pilot at landing, where one-handed operation is almost mandatory.

Trimmed flight prior to landing must be done in approach, after extension of gear/flaps, etc., but before landing flare. The recommendations explicitly include landing conditions in and out of ground (taken to include carrier deck, sea, etc.) effect.

These force limits are more restrictive than the FAA Part 25 (Reference 118) requirement of 75 lbs and Part 23 (Reference 161) of 60 lbs for stick or 75 lbs for wheel controllers. However, operational experience shows support for lower limits for continuous maneuvering.

E. GUIDANCE FOR APPLICATION

The suggested force limits for centerstick and wheel controllers are absolute requirements, independent of flying quality levels or failure states. As such they serve as simple pass-fail criteria. The procuring activity may choose, however, to apply flying quality Level requirements. For sidestick controllers the limits must be assessed qualitatively.

F. DEMONSTRATION OF COMPLIANCE

Ultimately, compliance must be proven through flight test. Analysis and simulation are subject to the accuracy of aerodynamic estimates in ground effect. The flight test should 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, should be performed, to assure that power setting does not adversely affect the force characteristics.

G. SUPPORTING DATA

There are no existing data for use in evaluating this requirement.

H. LESSONS LEARNED

A lack of commentary on the recommended limits suggests that their implementation is acceptable.

3.2.9.7.3 Pitch axis control force limits — dives

A. REASON FOR REQUIREMENTS

As a frequently used but peculiar flight operation, the dive is subject to separate force limit requirements.

B. RELATED MIL-F-8785C REQUIREMENTS

3.2.3.5, 3.2.3.6, 3.6.1.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.9.7.3 Pitch axis control force limits -- dives.

Service Flight Envelope. With the aircraft trimmed for level flight at speeds throughout the Service Flight Envelope, the control forces in dives to all attainable speeds within the Service Flight Envelope shall not exceed __(a)__. In similar dives, but with use of trim following the dive entry, it shall be possible with normal piloting techniques to maintain the forces within the following limits: __(b)__.

Permissible Flight Envelope. With the aircraft trimmed for level flight at V_{MAT} but with use of trim optional in the dive, it shall be possible to maintain the pitch control force within the following limits in dives to all attainable speeds within the Permissible Flight Envelope: __(c)__. The force required for recovery from these dives shall not exceed: __(d)__. Trim and deceleration devices, etc., may be used to assist in recovery if no unusual pilot technique is required.

Note: Letters in blanks correspond to recommended values in Table 1.

TABLE 1 (3.2.9.7.3)

RECOMMENDED FORCE LIMITS FOR DIVES AND RECOVERY FROM DIVES

REQUIREMENT NUMBER	CONTROLLER	FORCE (lb)	
		PUSH	PULL
(a)	Centerstick	50	10
	Sidestick	*	*
	Wheel	75	15
(b)	Centerstick	10	10
	Sidestick	*	*
	Wheel	20**	20**
(c)	Centerstick	50	35
	Sidestick	*	*
	Wheel	50	35
(d)	Centerstick	120	120
	Sidestick	*	*
	Wheel	120	120

*Limits for sidestick controllers have not been established. However, the forces must be acceptable to the *pilot*.

**Two-handed operation. In event that operation of the trim system requires removal of one hand the force limits should be as for centerstick.

D. RATIONALE BEHIND REQUIREMENT

Push and pull force limits must be stated to assure that demands on the pilot are not unreasonable. Since dives may be performed throughout both the Service and Permissible Flight Envelopes, separate requirements are given for each. Operation of manual trim systems implies that the dive is a sustained (rather than momentary) maneuver, so the trim should be effective in substantially reducing pitch forces.

E. GUIDANCE FOR APPLICATION

Since the force limits are absolute, application of the requirements consists of verifying that the aircraft falls within the force limits stated for the Flight Envelope under consideration.

F. DEMONSTRATION OF COMPLIANCE

Compliance must ultimately be proven through flight test. Analysis may be used for initial verification. The tests and analyses should be conducted over the following range of aircraft and flight conditions: a center of gravity range from most forward (combined with heaviest aircraft weight) to most aft (combined with lightest aircraft weight); for the Service Flight Envelope altitudes from 2000 ft above MSL to the maximum service altitude, for the range of minimum to maximum service speeds; for the Permissible Flight Envelopes altitudes as required by the procuring activity or the ranges of the Permissible Flight Envelope, over the speed range from V_{MAT} to the maximum permissible.

G. SUPPORTING DATA

There is no known set of data with which to verify these requirements.

H. LESSONS LEARNED

The limited dive testing of Reference 126, using a Class III airplane (P-3B), supports the one-handed wheel push force limit of 50 pounds.

3.2.9.7.4 Pitch axis control force limits — sideslips

A. REASONS FOR REQUIREMENT

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.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.3.7

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.9.7.4 Pitch axis control force limits — sideslips. With the aircraft trimmed for straight, level flight with zero sideslip, the pitch-control force required to maintain constant speed in steady sideslips with up to _____ pounds of pedal force in either direction, or in sideslips as specified in the Operational Flight Envelope, 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: _________________. If a variation of pitch-control force with sideslip does exist, it is preferred that increasing pull force accompany increasing sideslip, and that the magnitude and direction of the force change be similar for right and left sideslips. For Level 3 there shall be no uncontrollable pitching motions associated with the sideslips discussed above.

RECOMMENDED VALUES
(For pedal force of 50 pounds or less)

Centerstick controllers:	10 pounds pull to 3 pounds push
Sidestick controllers:	Acceptable to the pilot
Wheel controllers:	15 pounds pull to 10 pounds push

D. RATIONALE BEHIND REQUIREMENT

Any pitching moments due to sideslip — whether intentional or unintentional — should not require heavy compensation by the pilot. This keeps pilot workload in crosswinds (or in failures, such as one-engine-out operation on multi-engine airplanes) to a minimum.

E. GUIDANCE FOR APPLICATION

For operations at Level 1 or 2 conditions, the limits are straightforward: either the requirements are met or they are not. 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.

F. DEMONSTRATION OF COMPLIANCE

A flight test program is essential, especially given the difficulty in analytically defining aircraft characteristics in asymmetric flight. The program should 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 the Level 3 state. Such conditions, if required, should be specified in the Operational Flight Envelope by the procuring activity.

G. SUPPORTING DATA

The existence of systematic data is not known at this time.

H. LESSONS LEARNED

Reference 126, using a Class III aircraft (P-3B), showed support for the pull limit of 15 pounds for wheel controllers.

3.2.9.7.5 [Reserved]

3.2.9.7.6 Pitch axis control force limits — failures

A. REASON FOR REQUIREMENT

Augmentation system failures should not cause abrupt or severe changes in the trim state of the aircraft. The ability to retain reasonable control is measured in terms of demands on the pilot to maintain trim conditions.

B. RELATED MIL-F-8785C REQUIREMENT

3.5.5.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.9.7.6 Pitch axis control force limits — failures. The change in longitudinal control force required to maintain trim pitch attitude following complete or partial failure of the augmentation system shall not exceed the following limits: ______________________.

It is recommended that for at least 5 seconds following the failure the change in pitch force not exceed 20 pounds.

D. RATIONALE BEHIND REQUIREMENT

The purpose of a requirement in this area is to insure that the short-term response of the aircraft to an augmentation system failure does not get out of hand before the pilot can react. The requirements of 3.2.7.2 describe allowable transient responses. However, it is felt to be necessary to also have limits on the control forces required to minimize these responses.

E. GUIDANCE FOR APPLICATION

Testing of failure modes — in flight or simulation — should always include consideration of demands on the pilot to manually retrim. The emphasis for applying this requirement, however, is in verifying that the requirements of 3.2.7.2, "Pitch axis response to failures," are met; i.e., test data for verification with this requirement can be obtained at the same time.

F. DEMONSTRATION OF COMPLIANCE

Testing required by 3.2.7.2 should be designed so that the control force requirements are also evaluated.

G. SUPPORTING DATA

At this time no supporting data are available.

H. LESSONS LEARNED

None.

3.2.9.7.7 Pitch axis control force limits — configuration or control mode change

A. REASON FOR REQUIREMENT

Intentional engagement or disengagement of any portion of the flight control system should never result in unusual or unreasonable demands on the pilot to retain control.

B. RELATED MIL-F-8785C REQUIREMENT

3.5.6.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.9.7.7 Pitch axis control force limits -- configuration or control mode change. The control force changes resulting from the intentional engagement or disengagement of any portion of the primary flight control system by the pilot shall not exceed the following limits: ______ ______________.

It is recommended that for at least 5 seconds following the mode change the change in pitch force not exceed 20 pounds.

D. RATIONALE BEHIND REQUIREMENT

Trim transients following intentional pilot actions with the flight control system should obviously be small.

Since this requirement deals with intentional modification of the flight control system, it is implied that no failures have occurred. Failures are covered explicitly by 3.2.9.7.6.

E. GUIDANCE FOR APPLICATION

Proper application of this requirement may be performed by careful design of the aircraft augmentation systems. Mode switching should assure that the new mode chosen does not have any large transients in initialization.

F. DEMONSTRATION OF COMPLIANCE

This requirement is effectively a subset of 3.2.7.3. Simulation, analysis, or flight test demonstrations for that paragraph should include force response tests.

G. SUPPORTING DATA

At this time no supporting data are available.

H. LESSONS LEARNED

None.

3.2.9.8 Pitch axis trim systems

A. REASON FOR REQUIREMENT

A requirement is necessary that specifies the characteristics of the pitch trim in reducing control forces, in operational flight and in event of trim system failures. This paragraph is included to insure satisfactory trim system operation.

B. RELATED MIL-F-8785C REQUIREMENT

3.6.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.9.8 Pitch axis trim systems. In straight flight, throughout the Operational Flight Envelope the trimming system shall be capable of reducing the steady-state control forces to ______________. The failures to be considered in applying Level 2 and 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.

RECOMMENDED FORCE LIMITS

Level 1 or 2:	Zero
Level 3:	No greater than 20 pounds, push or pull

D. RATIONALE BEHIND REQUIREMENT

The purpose of a trim system is to reduce steady-state forces on cockpit controls, preferably to zero. By placing the requirements on steady-state forces we have implicitly allowed transient forces to exceed the limits in this paragraph. This is consistent with the normal usage of trim controls wherein transient forces are handled with the primary cockpit controllers with trim utilized to remove steady-state forces.

E. GUIDANCE FOR APPLICATION

In normal operations, this requirement is very straightforward. If a pitch trim is provided, it must be effective. However, the more powerful a trim, 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) are of prime importance.

F. DEMONSTRATION OF COMPLIANCE

Trim capability must be shown through flight testing, at conditions covering the range of operational flight. The flight conditions should be devised to exercise the envelope 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. Failure operations at these conditions may be impractical; therefore, compliance with the failure requirement may be demonstrated by intentionally mistrimming the aircraft and recovering from the mistrim, or by simulation.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None available.

3.2.9.8.1 Pitch axis trim systems — rate of operation

A. REASON FOR REQUIREMENT

Trimming devices must be capable of operating rapidly enough to be of value to the pilot (especially in maneuvers which involve large sustained pitch-control inputs), but not so rapidly that their operations produce abrupt aircraft motions.

B. RELATED MIL-F-8785C REQUIREMENT

3.6.1.2

C. STATEMENT OF REQUIREMENT

3.2.9.8.1 Pitch axis trim systems — rate of 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.

D. RATIONALE BEHIND REQUIREMENTS

If a trim system is to function as desired, i.e., relieve steady-state pitch control forces, it must operate rapidly. Slow trim rates will: a) fatigue the pilot; and b) not be helpful since trim conditions may change more quickly. However, rapid trim motion could cause abrupt attitude changes and oversensitivity — making it impossible in practice to achieve desired trim.

E. GUIDANCE FOR APPLICATION

It may be difficult to specify a trim rate which will be applicable for all mission phases of the aircraft. It is more likely that the procuring agency will simply require that the trim operation rate not be objectionable, and that the contractor will use past experience as a guide. In any case, acceptability of the trim system will be a function of the operational missions to be flown, and of the untrimmed control forces. Specific missions should be considered individually in this light; in MIL-F-8785C this requirement included specific force limits that may be obtained in dives and during rapid speed changes, using trimming devices to decrease the forces.

F. DEMONSTRATION OF COMPLIANCE

At the discretion of the procuring activity, compliance may be proven through static ground testing of the trim system. However, if it is felt that the trim operation might be influenced by flight variables (e.g., dynamic pressure, electrical or hydraulic system load), flight testing is mandatory. Obviously, demonstration of the force requirements of 3.2.9.7 will involve trim operation, so it will be easy to demonstrate compliance with this requirement.

The conditions to be tested should include, as a minimum: a) dives and ground attack maneuvers required in normal service operation and b) 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 aircraft is trimmed for level flight prior to initiation of the maneuver.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.2.9.8.2 Pitch axis trim systems — stalling of trim systems

A. REASON FOR REQUIREMENT

Large control hinge moments are characteristic of flight in the transonic speed regime. It is therefore possible that the trim system would become ineffective. In addition, aircraft using an adjustable stabilizer for pitch trim are susceptible to stalling of the trim actuator under certain mistrim conditions. That is, if sufficient mistrim exists, it is possible for resulting aerodynamic loads to exceed the drive capability of the stabilizer actuator.

B. RELATED MIL-F-8785C REQUIREMENT

3.6.1.3

C. STATEMENT OF REQUIREMENT

3.2.9.8.2 Pitch axis trim systems — stalling of trim systems. Stalling of a trim system due to aerodynamic loads during maneuvers shall not result in an unsafe condition. Specifically, the longitudinal trim system shall be capable of operating during the dive recoveries of 3.2.9.7.3 at any attainable permissible n, at any possible position of the trimming device.

D. RATIONALE BEHIND REQUIREMENT

The basic rationale behind this requirement is obvious. Not so obvious is the effect of mistrimming. The following discussion, taken from Reference 173, describes the problem.

Mistrimming will occur in the event of a trim runaway, although a runaway is an unlikely possibility. Mistrim is more probable if trimming is attempted in severe turbulence. Proper trim is normally established by trimming to a zero stick force reference. In the rapidly changing conditions of severe turbulence, no stable stick force reference is available and trimming attempts are likely to result in mistrim.

Once mistrim exists, some of the elevator's pitching moment contribution must go to oppose the pitching moment developed by the mistrimmed stabilizer. This will have several adverse effects. 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 load on the stabilizer may reach a level that is impossible for the trim actuator to overcome.

If, for example, nose-down trim is used to counter the airplane's pitch-up response to a vertical downdraft, the airplane 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. As speed increases, the adverse effects increase, and the elevator may have insufficient effect 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 cannot be countered by a Mach trim system that is unable to move the stabilizer.

E. GUIDANCE FOR APPLICATION

In addition to high-speed dives, the effect of mistrim should be considered when utilizing an adjustable stabilizer for pitch trim.

F. DEMONSTRATION OF COMPLIANCE

Demonstrating compliance is a matter of placing the aircraft in a dive at the maximum permissible speed and checking operation of the pitch trim system. However, it is clear that full nose-down mistrim should be accounted for in these 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 should allow this type of abuse. See "Lessons Learned" for more discussion of this.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

A United Airlines Boeing 720B encountered stalling of the pitch trim actuator during a turbulence upset over O'Neill, Nebraska, on 12 July 1963 (Reference 255). The aircraft was passing through 39,000 ft in a climb to 41,000 ft in IMC conditions when severe turbulence was encountered. A large downdraft was encountered and the aircraft pitch attitude increased to +60 deg! This occurred despite application of full forward stick. The gust then reversed to a large updraft, putting the aircraft into a severe dive with an estimated flight path angle of about negative 35 deg. 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) where the wreckage of both aircraft 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 to essentially ignore large airspeed excursions in severe turbulence. However, the possibility of entering a dive with full nose-down mistrim should be considered in the design process.

3.2.9.8.3 Pitch axis trim systems — irreversibility

A. REASON FOR REQUIREMENT

Pilot workload increases significantly if the pitch trim system drifts or changes position. This requirement is intended to preclude such undesirable characteristics.

B. RELATED MIL-F-8785C REQUIREMENT

3.6.1.4

C. STATEMENT OF REQUIREMENT

3.2.9.8.3 Pitch axis trim systems -- 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, provision shall be made to ensure the accurate return of the device to its initial trim position on removal of each interconnect or augmentation command.

D. RATIONALE BEHIND REQUIREMENT

The bottom line for a trim device is: it must make the pilot's life easier. Unintentional changes in trim position will serve to complicate his job. This requirement allows for trim scheduling or interconnecting with other control devices, but it specifically disallows float or drift. In addition it assures that removal of the interconnect or scheduling does not leave it locked in an undesirable position.

E. GUIDANCE FOR APPLICATION

Application of this requirement is simple, i.e., design the trim device with a mechanical linkage that will move only when directed.

F. DEMONSTRATION OF COMPLIANCE

While flight testing must be conducted, there are no special maneuvers required to prove compliance. The flight test maneuvers flown for the requirements of 3.2.9 in general will be sufficient. Of especial importance are sustained maneuvers at $n \neq 1$, e.g., dives and dive recoveries, pull-ups, wind-up turns, with the cockpit trim setting fixed throughout.

G. SUPPORTING DATA

None available or required.

H. LESSONS LEARNED

None available.

3.2.10 Pitch Axis Control Displacements

3.2.10.1 Pitch axis control displacements — takeoff

A. REASON FOR REQUIREMENT

This requirement insures that there is reserve pitch control power during takeoff to allow regulation against gusts or pilot abuses.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.3.3.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.2.10.1 Pitch axis control displacements — takeoff. With the trim setting optional but fixed, the pitch-control travel during all types of takeoffs for which the aircraft is designed shall not exceed ____ percent of the total travel, stop-to-stop. Here the term takeoff includes ground run, rotation and liftoff, the ensuing acceleration to V_{max} (TO), and the transient caused by assist cessation. 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).

The recommended maximum pitch control displacement is 75 percent of total travel for all controller types.

D. RATIONALE BEHIND REQUIREMENT

This requirement is intended to insure that under no condition will it be necessary to use full control to meet the operational takeoff performance requirements. The obvious motivation for this is to provide adequate control margin during takeoff for regulation against atmospheric disturbances or pilot abuses. 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.

E. GUIDANCE FOR APPLICATION

No particular guidance is felt to be necessary.

F. DEMONSTRATION OF COMPLIANCE

Because of the well-recognized problems in modeling aerodynamics in ground effect, simulation results may not be adequate for compliance with this requirement. Flight testing performed to demonstrate compliance with 3.2.9.7.1 should include measurement of control displacements.

G. SUPPORTING DATA

None available.

H. LESSONS LEARNED

None available.

3.2.10.2 Pitch axis control displacements — maneuvering

A. REASON FOR REQUIREMENT

There is evidence that stick deflection characteristics, while secondary to force characteristics, are still important to the pilot when maneuvering.

B. RELATED MIL-F-8785C REQUIREMENTS

3.2.2.2, 3.2.2.2.2

C. STATEMENT OF REQUIREMENT

3.2.10.2 Pitch axis control displacements -- maneuvering. For all types of pitch controllers, the control motions in maneuvering flight shall not be so large or so small as to be objectionable. In steady turning flight and in pullups at constant speed, the incremental 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 those required to initiate the change.

D. RATIONALE BEHIND REQUIREMENT

For maneuvering flight it is necessary that control motions and displacements be comfortable and natural to the pilot. In this regard the above requirements are intended to assure that control motion and airplane motion are in harmony. Controller force/deflection characteristics are specified in Paragraph 3.2.9.4.

E. GUIDANCE FOR APPLICATION

Proper design of the pitch control power (3.2.8) and forces (3.2.9) for maneuvering should result in the above requirements being met. However, these requirements function as final tests of the adequacy of the controller specified by 3.2.8 and 3.2.9.

F. DEMONSTRATION OF COMPLIANCE

Flight testing should provide proof of compliance with these requirements, at conditions with most forward and most aft c.g., over the load factor range of the Service Flight Envelope. The subjective requirement for control motions to be "not objectionable" may also be interpreted as "not cause a degradation in Flying Quality Level," e.g., control motions alone should not degrade a Level 1 airplane to Level 2, etc.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

The isometric sidearm controller used in the YF-16 and F-16 was unacceptable to many pilots (see References 124 and 125). Their primary objection was the lack of tactile cues as to when the controller was at its limit. A small amount of motion was incorporated into the controller that has been adopted.

3.2.10.3 Pitch axis control displacements — gust regulation

A. REASON FOR REQUIREMENT

Control authority (measured in this section by displacement and surface deflection rates) must be sufficient for the aircraft to perform all required tasks in the Operational Flight Envelope in the presence of atmospheric disturbances.

B. RELATED MIL-F-8785C REQUIREMENT

3.5.2.3

C. STATEMENT OF REQUIREMENT

3.2.10.3 Pitch axis control displacements — gust regulation. The ability of the aircraft to perform operational maneuvers required of it shall not be limited in the ___________ atmospheric disturbances defined in 3.9 by control displacement or control surface deflection rates. 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 when establishing compliance with this requirement.

Recommended disturbance level: Moderate

D. RATIONALE BEHIND REQUIREMENT

Atmospheric disturbances in the form of gusts should not result in limitations in maneuvering in the Operational Flight Envelope. In terms of displacements this means that no limitations should be imposed due solely to control travel. Since ability to counter gusts includes surface rate characteristics, an explicit requirement is stated for deflection rates.

The last statement was included to point out that 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, however, that same hydraulic system might be more than adequate. Also, at high dynamic pressure, high hinge moments may limit control surface rate and deflection.

In precision control tasks such as the landing approach and formation flying it has been observed that the pilot sometimes resorts to elevator stick pumping to achieve better precision (see References 113, 120, and 121). This technique is likely to be used when the short-period frequency is less than the minimum specified or if the phugoid is unstable.

While specific disturbances are listed, the evaluation remains somewhat qualitative.

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 characteristics of the airplane are near the Level 3 limits, increased control activity may occur, even though the maneuvers are more gradual.

E. GUIDANCE FOR APPLICATION

Control system design should naturally include consideration of gust effects. For thorough design, the turbulence models of 3.9 should be applied. By this point in the design process the pitch control gearing, surface effectiveness, etc., would probably be well defined, and application of this requirement would be straightforward.

F. DEMONSTRATION OF COMPLIANCE

It is clearly impractical to demand flight testing to demonstrate compliance with this requirement. Instead, at the discretion of the procuring activity, compliance may be shown through analysis of gust response characteristics using either an analytical model or a piloted simulation, involving the gust models of 3.9. Such analysis must include not only the normal operational maneuvers involving pitch controls, but also the critical maneuvers (especially for hydraulic actuation systems) which may limit the responsiveness of the pitch control surface. As mentioned in Paragraph D, these might include extension of landing gear and high-lift devices on landing approach, etc.

A modicum of common sense is required in the application of this requirement. The specific levels of atmospheric disturbance to be applied are not specified. Yet Paragraph 3.9 contains turbulence up to the thunderstorm level. We do not normally require operational maneuvering in thunderstorm turbulence. It would seem reasonable to require operational maneuvering in turbulence intensities up to "moderate." For turbulence intensities greater than moderate it seems reasonable to require sufficient maneuver capability for loose attitude control.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.3 HANDLING QUALITY REQUIREMENTS FOR VERTICAL FLIGHT PATH AXIS

3.3.1 Vertical Axis Response to Attitude Change

3.3.1.1 Vertical axis response to attitude change — transient response

A. REASON FOR REQUIREMENT

This requirement is included to provide a separate and independent criterion for flight path response to pitch attitude changes. Two criteria are necessary: one for conventional aircraft where pitch attitude is the primary means for flight path control; the second for STOL aircraft where pitch attitude plays a secondary role in path control and/or is used to control speed.

B. RELATED MIL-F-8785C REQUIREMENT

None.

C. STATEMENT OF REQUIREMENT

3.3.1.1 Vertical axis response to attitude change — transient response.

a) The short-term flight path response to attitude changes shall have the following characteristics: ____________________.

b) If a designated controller other than attitude is the primary means of controlling flight path, the flight path response to an attitude change can be degraded to the following: ________________.

c) In all cases the pitch attitude response must lead the flight path angle by ________________ and must have a magnitude equal to or greater than the flight path angle.

Recommended values: Insufficient data available.

D. RATIONALE BEHIND REQUIREMENT

1. Frontside Operation

Aircraft operating on the front side of the power-required curve utilize pitch attitude to control flight path. In fact, the primary motivation for the limits set in 3.2.1 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 in Figure 3 (3.2.1.1). As shown in that block diagram, the short-term flight path response is related kinematically to the aircraft pitch attitude change by

$$\frac{\gamma}{\theta} = \frac{1}{T_{\theta_2}s + 1}$$

The long-term response is related to $d\gamma/dV$, which of course depends on where the operating point is on the power-required curve (see Para. 3.3.1.2).

The equivalent system requirements for pitch attitude control (see Para. 3.2.1.1) involve $1/T_{\theta_2}$ directly ($\omega_{sp}T_{\theta_2}$ limits) or indirectly $[\omega_{sp}$ vs. n/α where $n/\alpha \doteq (U_o/g)(1/T_{\theta_2})]$. 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 independent variation of speed and $1/T_{\theta_2}$ (basically all NT-33 data), it is not possible to determine whether the boundaries do indeed account for path as well as pitch. The lack of availability of such data also makes it impossible to establish a quantitative requirement for this paragraph. However, for design guidance, $1/T_{\theta_2}$ should be at least greater than the values specified in Figure 2 (3.2.1.1) for Category C.

The bandwidth criterion (3.2.1.2) clearly is a specification on attitude control only and therefore requires a separate specification on short-term path response, i.e., minimum value of $1/T_{\theta_2}$. Again the values specified in Figure 2 (3.2.1.1) provide reasonable guidance for

Category C. These limiting values are repeated in Table 1 for reference. The values of $(1/T_{\theta_2})_{min}$ in Table 1 are simply the lower boundaries on n/α from Figure 1 (3.2.1.1) at an approach speed of 135 kt. Generally speaking, $1/T_{\theta_2}$ is large enough to be of no concern for CTOL aircraft in the Category A and B flight phases, and hence no data are available to establish lower limits.

TABLE 1 (3.3.1.1)

GUIDANCE FOR LOWER LIMIT ON $1/T_{\theta_2}$

LEVEL	CLASS	$(1/T_{\theta_2})_{min}$
1	I, II-C, IV	0.38
	II-L, III	0.29
2	I, II-C, IV	0.24
	II-L, III	0.14

2. Backside Operation

An aircraft operating well on the back side of the power-required curve ($d\gamma/dV$ positive) must rely on thrust or thrust vectoring for path control. Such STOL aircraft usually have sufficiently fast engine response characteristics (or some type of blended DLC) that allow precision flight path tracking with the throttles (or other designated flight path controller). The control of pitch attitude becomes much less critical and hence some relaxation in the Level 1 limits should be allowed. One objective of the currently ongoing STOL amendment work will be to provide an estimate of the allowable relaxation in the attitude criterion as a function of the quality of the short-term flight path response to throttle (or designated flight path controller).

3. Attitude/Path Consonance

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 in Reference 269. These results indicate that an appropriate criterion parameter would be the phase angle between path and attitude at the short-period frequency, i.e.,

$$\phi(\gamma/\theta)|_{\omega=\omega_{sp}}$$

Noting that $\phi(\gamma/\theta)|_{\omega=\omega_{sp}} = \tan^{-1} \omega_{sp}T_{\theta_2}$, the criterion on $\omega_{sp}T_{\theta_2}$ (Para. 3.2.1.1) can be easily converted to $\phi(\gamma/\theta)|_{\omega=\omega_{sp}}$ with the results shown in Table 2. The advantage of using $\phi(\gamma/\theta)|_{\omega=\omega_{sp}}$ 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 2 are all based on the same NT-33 flight test data as the LOES boundaries in 3.2.1.1. Until more data can be obtained to indicate pilot rating trends and $1/T_{\theta_2}$ is varied at constant ω_{sp}, it is felt that Table 2 should be kept in the category of guidance.

Handbook items E through H have been left blank until more data can be obtained to upgrade the recommendation to a *requirement*.

TABLE 2 (3.3.1.1)

CONVERSION OF $\omega_{sp}T_{\theta_2}$ TO A PHASE ANGLE CRITERION

CATEGORY	LEVEL	$(\omega_{sp}T_{\theta_2})_{min}$	MAXIMUM ALLOWABLE $\phi(\gamma/\theta)|_{\omega=\omega_{sp}}$ (deg)
A	1 2	1.6 1.0	-58 -45
B	1 2	1.0 0.58	-45 -30
C	1 2	1.3 0.75	-52 -37

E. GUIDANCE FOR APPLICATION

F. DEMONSTRATION OF COMPLIANCE

G. SUPPORTING DATA

H. LESSONS LEARNED

Not enough information is available for discussion in these areas.

3.3.1.2 Vertical axis response to attitude change — steady-state response

A. REASON FOR REQUIREMENT

The accepted piloting technique for conventional aircraft is to adjust flight path with pitch attitude. This requirement is included to insure that the long-term flight path response to pitch attitude changes is acceptable to the pilot.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.1.3

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.3.1.2 Vertical axis response to attitude change — steady-state response. For aircraft without a designated secondary flight path control the steady-state path response to attitude inputs shall be as follows: ____________.

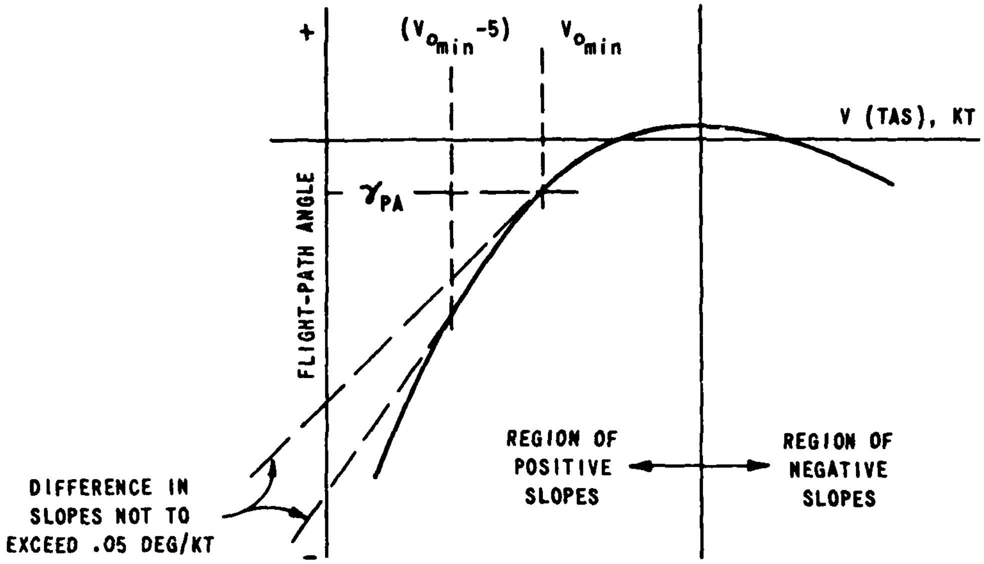
Recommended values: Flight-path stability is defined in terms of flight-path-angle change where the airspeed is changed by the use of pitch control only (throttle setting not changed by the crew). For the landing approach Flight Phase, the curve of flight-path angle versus true airspeed shall have a local slope at $V_{o_{min}}$ that is negative or less positive than:

a. Level 1: 0.06 degrees/knot

b. Level 2: 0.15 degrees/knot

c. Level 3: 0.24 degrees/knot

The thrust setting shall be that required for the normal approach glide path at $V_{o_{min}}$. The slope of the curve of flight-path angle versus airspeed at 5 knots slower than $V_{o_{min}}$ shall not be more than 0.05 degrees per knot more positive than the slope at $V_{o_{min}}$, as illustrated by the sketch below.



D. RATIONALE BEHIND REQUIREMENT

Discussions for this section, as well as the "Supporting Data" section, are taken from the MIL-F-8785B BIUG, Reference 11.

Operation on the "backside" of the drag curve in the landing approach leads to problems in airspeed and flight-path control. References 150, 151, 152, and 153 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 instability, even when the open-loop (unattended airplane) phugoid motion is stable, is caused by an "unstable" zero in the h/δ_e airplane transfer function. Specifically, Reference 150 uses closed-loop analyses 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, Reference 10 shows that $1/T_{h_1}$ is closely approximated by the ratio D/C, where D and C are defined implicitly as follows:

$$\frac{\dot{h}(s)}{\delta_e(s)} = \frac{As^3 + Bs^2 + Cs + D}{\left(s^2 + 2\zeta_p\omega_p s + \omega_p^2\right)\left(s^2 + 2\zeta_{sp}\omega_{sp}s + \omega_{sp}^2\right)}$$

The additional assumption that C is approximately equal to

$$\left[V\left(Z_{\delta_e}M_w - M_{\delta_e}Z_w\right)\right]$$

is generally valid, so that:

$$\frac{1}{T_{h_1}} \doteq \frac{D}{V\left(Z_{\delta_e}M_w - M_{\delta_e}Z_w\right)}$$

The climb-angle-to-elevator transfer function is as follows:

$$\frac{\gamma(s)}{\delta_e(s)} = \frac{1}{V}\frac{\dot{h}(s)}{\delta_e(s)} = \frac{1}{V}\frac{As^3 + Bs^2 + Cs + D}{\left(s^2 + 2\zeta_p\omega_p s + \omega_p^2\right)\left(s^2 + 2\zeta_{sp}\omega_{sp}s + \omega_{sp}^2\right)}$$

Applying the limit value theorem, for a step δ_e (δ_e/s) the slope of the steady-state γ versus δ_e curve is equal to the value of this transfer function when s approaches zero, so that:

$$\frac{d\gamma}{d\delta_e} = \left.\frac{\gamma(s)}{\delta_e(s)}\right|_{ss} = \frac{1}{V}\frac{D}{\omega_p^2\omega_{sp}^2}$$

In a similar manner, the slope of the steady-state u versus δ_e curve is obtained:

$$\frac{du}{d\delta_e} = \left.\frac{u(s)}{\delta_e(s)}\right|_{ss} = -\frac{g\left(Z_{\delta_e}M_w - M_{\delta_e}Z_w\right)}{\omega_p^2\omega_{sp}^2}$$

Therefore, the slope of the steady-state γ versus u curve for elevator inputs is

$$\frac{d\gamma}{du} = \frac{d\gamma/d\delta_e}{du/d\delta_e} = -\frac{1}{g}\frac{D}{V\left(Z_{\delta_e}M_w - M_{\delta_e}Z_w\right)}$$

Using the expression for $1/T_{h_1}$, we finally obtain

$$\frac{d\gamma}{du} = -\frac{1}{g}\frac{1}{T_{h_1}}$$

The $d\gamma/du$ limits, therefore, set limits on $1/T_{h_1}$.*

The limit on $d\gamma/du$ at 5 knots slower than $V_{o_{min}}$ was added to assure that the airplane remains tractable at commonly encountered off-nominal speeds.

For design purposes, $d\gamma/du$ can be estimated from the dimensional stability derivatives as follows:

$$\frac{d\gamma}{du} = \frac{1}{g}\left\{X_u - \left(X_w - \frac{g}{V}\right)\left[\frac{Z_u - \left(Z_{\delta_e}/M_{\delta_e}\right)M_u}{Z_w - \left(Z_{\delta_e}/M_{\delta_e}\right)M_w}\right] - \frac{X_{\delta_e}}{M_{\delta_e}}\left[\frac{M_wZ_u - M_uZ_w}{-Z_w + M_w\left(Z_{\delta_e}/M_{\delta_e}\right)}\right]\right\}$$

or

$$\frac{d\gamma}{du} = \frac{1}{g}\left\{X_u - \left(X_w - \frac{g}{V}\right)\left[\frac{Z_u - \left(Z_{\delta_e}/M_{\delta_e}\right)M_u}{1/T_{\theta_2}}\right] - \frac{X_{\delta_e}}{M_{\delta_e}}\left[\frac{\omega_p^2\omega_{sp}^2}{g\left(1/T_{\theta_2}\right)}\right]\right\}$$

For M_u and X_{δ_e} small, the following approximation is valid except for very-short-tailed airplanes:

$$\frac{d\gamma}{du} = \frac{1}{g}\left[X_u - \left(X_w - \frac{g}{V}\right)\frac{Z_u}{Z_w}\right]$$

---

*The earlier Reference 233 contains a similar approximate analysis of $1/T_{h_1}$.

It is possible to violate this requirement by operating well on the back side of the power-required curve ($d\gamma/du >> 0$) and still have a Level 1 airplane as long as some other means of controlling flight path is provided (usually power). Naturally the "secondary flight path controller" must have acceptable characteristics. For example, if thrust is designated as the flight path controller, good flight path response to changes in thrust $(\gamma/\delta_T)_{ss}$ 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., Reference 150). Requirements on γ/δ_T are specified in 3.3.2 based on STOL aircraft research.

E. GUIDANCE FOR APPLICATION

Since backside operation (defined when $d\gamma/du > 0$) is most critical during landing approach, this requirement is oriented toward that Flight Phase. It is also troublesome for takeoff, cruise, and high-altitude maneuvering, but it will probably not be as critical as for the 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 to (hopefully) improve the flight path response, this requirement would not be applied. The reader should instead consult 3.3.1.2.1, "Relaxation for aircraft with designated flight path controller."

F. DEMONSTRATION OF COMPLIANCE

The climb-angle-versus-airspeed data used to demonstrate compliance can be obtained during the stabilized-airspeed tests for static stability at low airspeeds.

By the nature of the way in which the climb-angle-versus-airspeed criterion was developed, the climb angle to be measured is the climb angle relative to the air, not the ground. This fact is mentioned for the benefit of flight-test engineers contemplating use of Doppler radar or ground-based tracking equipment to obtain the data. If such methods are used, the wind must be calm, or at least constant and accurately measured.

The most straightforward method 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 equal to

$$\sin^{-1}\left[\frac{\text{Vertical speed}}{\text{True airspeed}}\right]$$

Still air is necessary in any case, to minimize data scatter. Because of thrust effects 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 speed. The trim flight-path angle can have a marked effect on the results; the range of glide slopes expected in the operational and training mission should be tested.

G. SUPPORTING DATA

The $1/T_{h_1}$ data used to set numerical limits on $d\gamma/du$ are given in References 120, 154-157.

It is apparent from Figures 1-3 (from Reference 120) that pilot ratings of $1/T_{h_1}$ are dependent on the value of ζ_p. For Level 1, 3.2.1.1 requires $\zeta_p > 0.04$; greater damping might result from autothrottle or similar augmentation. Therefore the positive ζ_p data of Figure 1 were used to establish the Level 1 requirement for $1/T_{h_1}$ or $d\gamma/dV$. (The data from Figures 2-4 are obviously too conservative for Level 1. The configurations for Figure 2 had ω_{sp} marginally close to the lower Level 1 boundary; while those for Figure 4 were downrated because of the pitch response to horizontal gusts caused by M_u.) For Levels 2 and 3, the zero-ζ_p data seem appropriate:

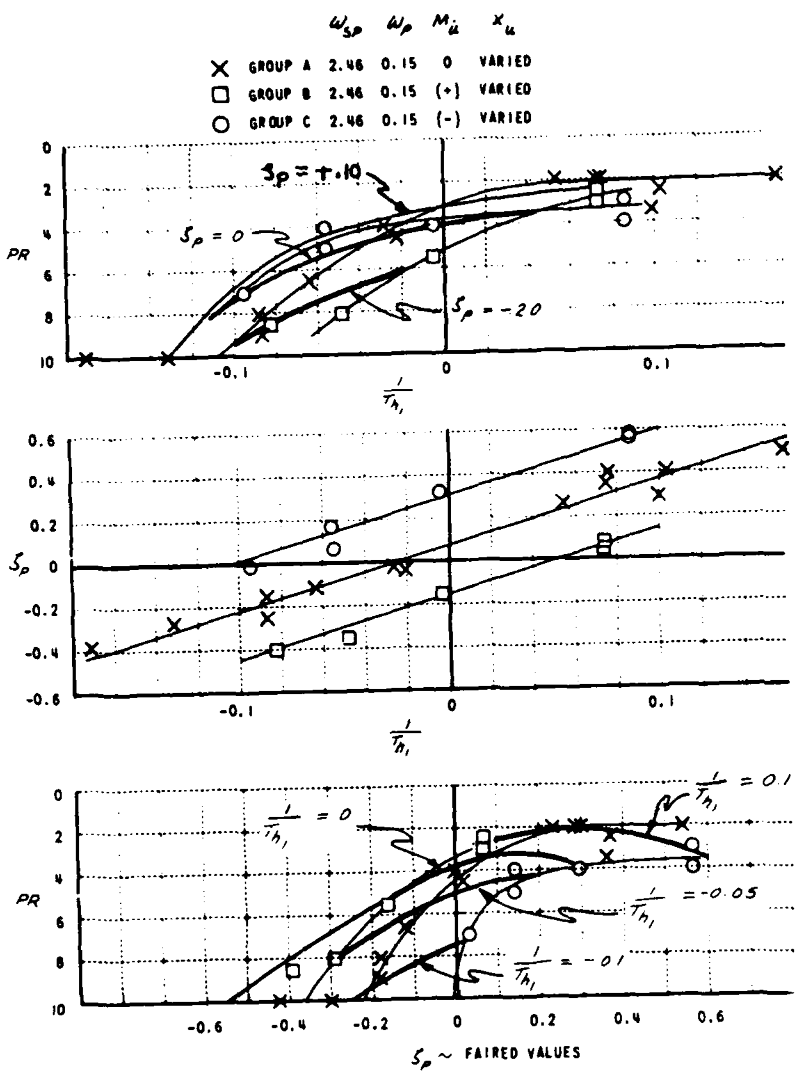


Figure 1 (3.3.1.2). Landing Approach
(T-33, Reference 120)

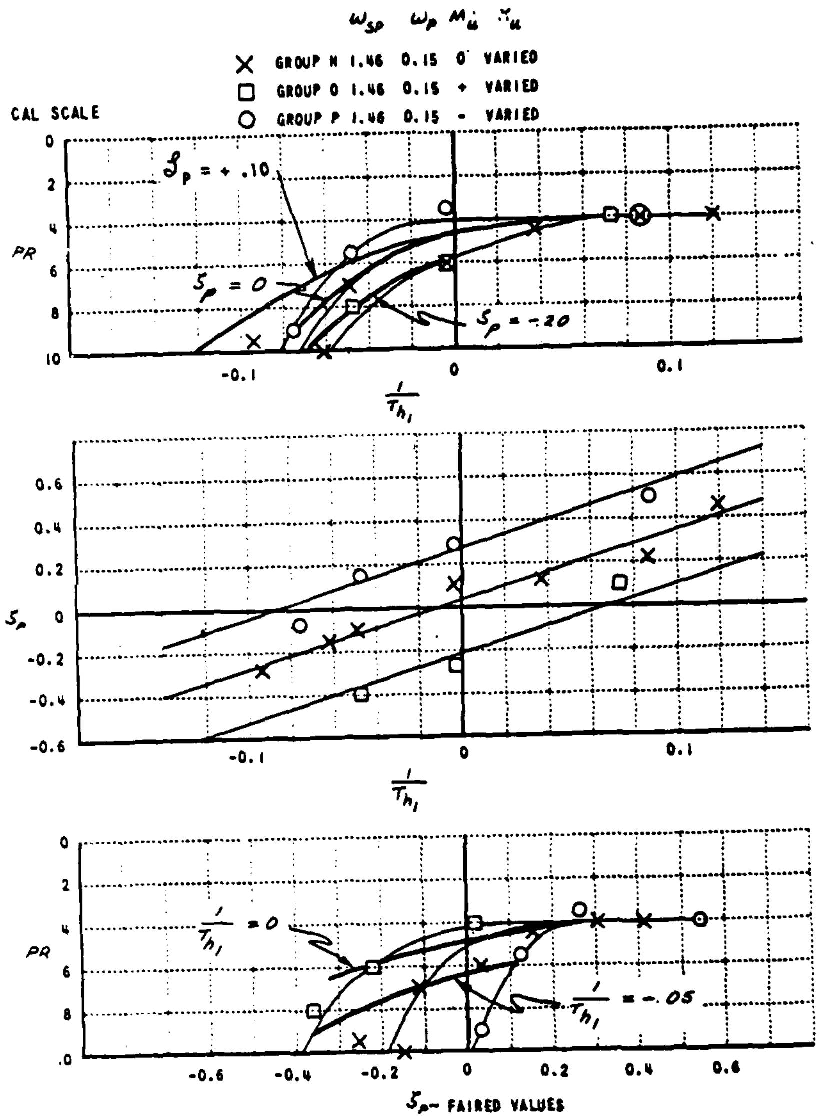


Figure 2 (3.3.1.2). Landing Approach
(T-33, Reference 120)

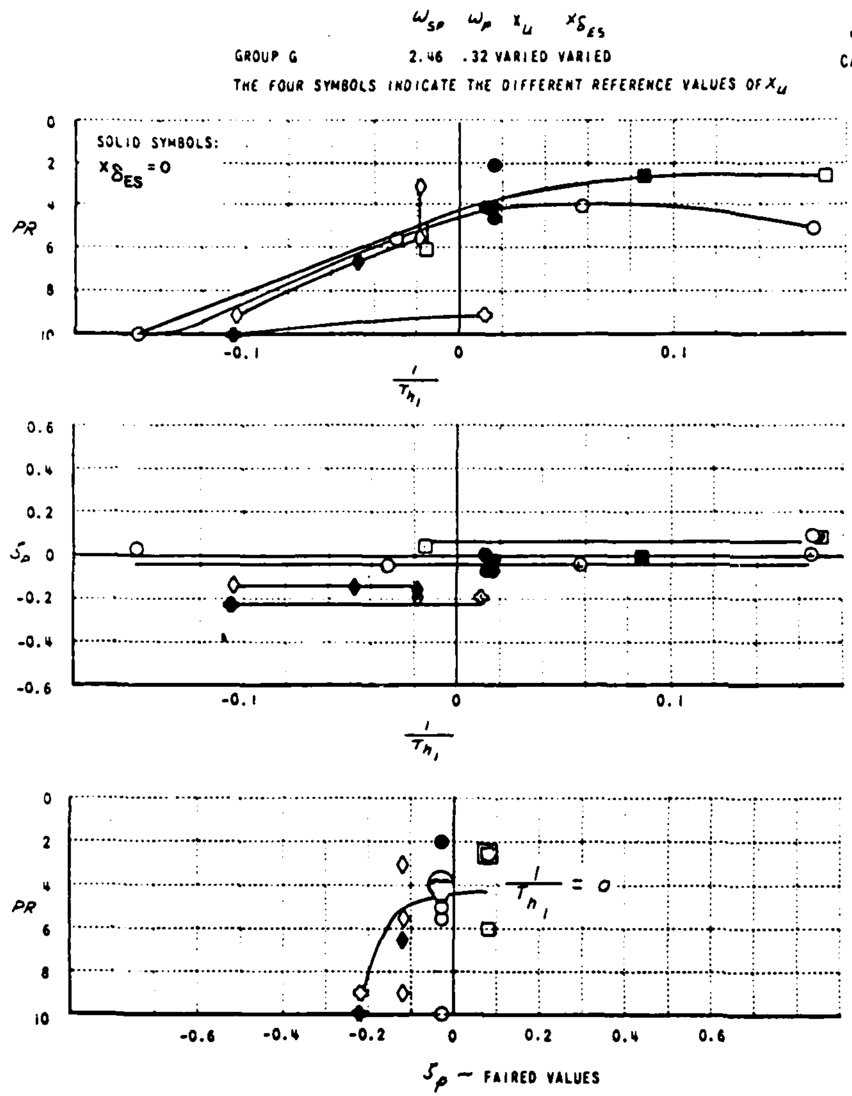


Figure 3 (3.3.1.2). Landing Approach
(T-33, Reference 120)

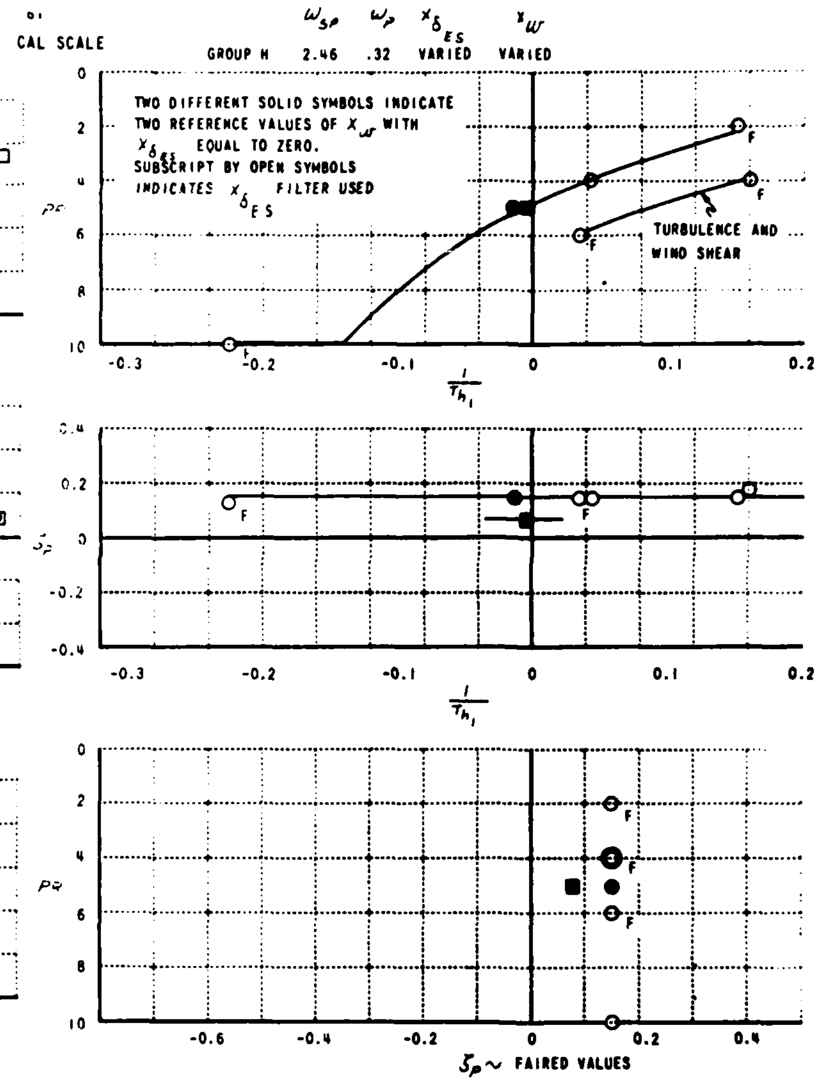


Figure 4 (3.3.1.2). Landing Approach
(T-33, Reference 120)

Figure	Level 2	Level 3
1	$1/T_{h_1} > -0.08$	$1/T_{h_1} > -0.12$
2	$1/T_{h_1} > -0.05$	$1/T_{h_1} > -0.08$

From Figure 3, with near-zero ζ_p:

Level 2	Level 3
$1/T_{h_1} > -0.05$	$1/T_{h_1} > -0.12$

From Figure 4, with high ζ_p but in turbulence:

Level 2	Level 3
$1/T_{h_1} > -0.05$	$1/T_{h_1} > -0.12$

Combinations of Level 2 or 3 values of $1/T_{h_1}$ with low ζ_p, ω_{sp}, or both appear worse than cases with high ζ_p and ω_{sp}. With these considerations in mind, $1/T_{h_1} = -0.02$ was chosen for the Level 1 boundary, -0.05 for Level 2, and -0.08 for Level 3. These values of $1/T_{h_1}$ correspond to the $d\gamma/dV$ values specified: multiply $1/T_{h_1}$ by $-(57.3)(1.689)/(32.2) = -3$.

The ground simulator experiment of Reference 155 altered $1/T_{h_1}$ by changing X_w and X_{δ_e}, and also considered the influences of thrust-line inclination and thrust-line offset on the flying qualities. There are very limited data for thrust-line offset, and the decision was made to assume that designers will take reasonable steps to keep the offset as small as possible. The data for zero thrust-line offset are presented in Figure 5 for different values of thrust-line inclination. The data do seem to indicate that some thrust-line inclination is desirable, but the variations in rating due to inclination are well within the scatter of the data considered as a whole.

The data from ground simulator experiments of References 154 and 157 are presented in Figure 6. It should be mentioned that only the data for the highest static margin in Reference 154 are presented because the lower static margins result in values of ω_{sp} that are too low for Level 1.

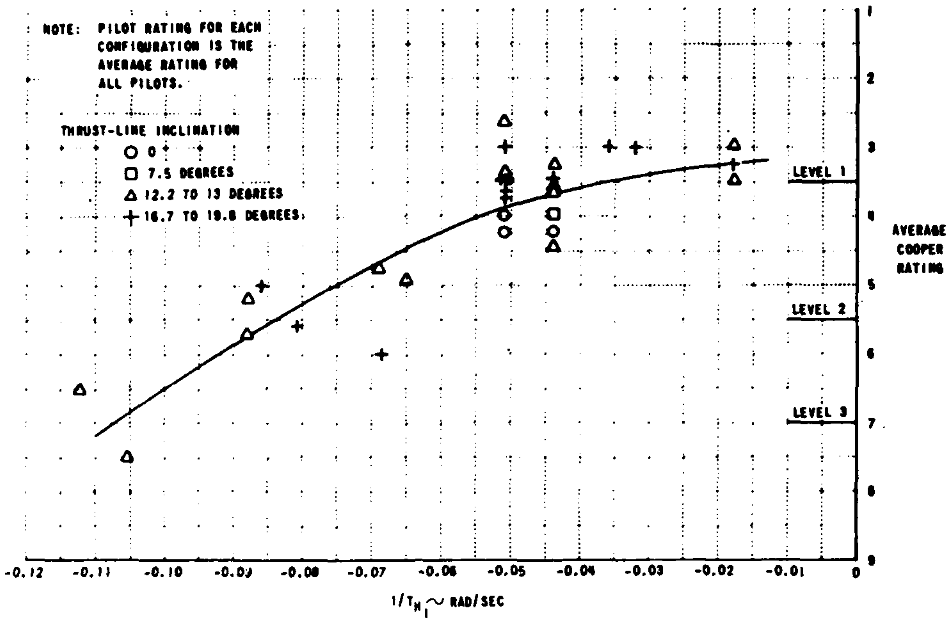


Figure 5 (3.3.1.2). Carrier Approach (Ground Simulator Experiment, Reference 155)

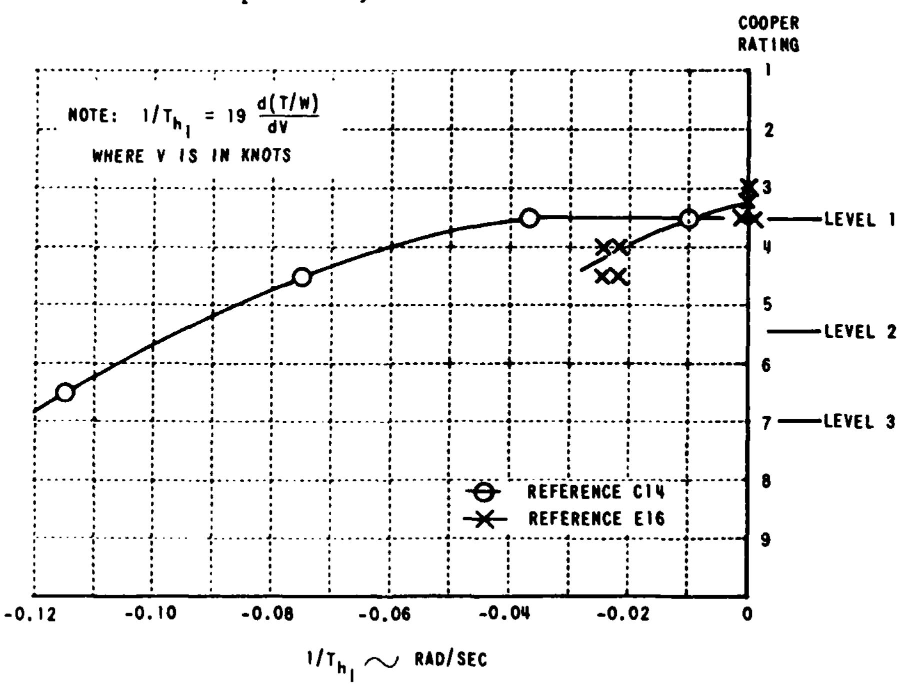


Figure 6 (3.3.1.2). SST Landing Approach (Ground Simulator Experiments, References 154 and 157)

The data from the in-flight experiment of Reference 156 are presented in Figure 7. There are several factors that influence interpretation of this data. First, the pilot rating scale used is a modified version of the Cooper scale and is rather difficult to interpret. Second, the "speed stability" was changed by altering $\partial T/\partial V$ as well as $\partial T/\partial \alpha$, which means that unstable values of "speed stability" were accompanied by negative values of phugoid damping. Since the "speed stability" was altered in this experiment by using engine thrust, the pilot could use the engine noise as an airspeed cue. The final (and probably most significant) factor is that most of the approaches were flown VFR, with a ground controller supplying continuous flight-path information by radio using a theodolite. Reference 156 states that this type of technique resulted in very tight control of flight path. A few approaches were made using precision-approach radar; these were much more difficult for the pilot to successfully accomplish. The relationship between the "speed stability" parameter $1/T_2$ of Figure 7 and $1/T_{h_1}$ is as follows:

$$1/T_{h_1} = 0.693\ (1/T_2)$$

A comparison of the requirements derived from Figures 1 through 4 and the data from Figures 5 through 7 is presented below.

	$1/T_{h_1}$ for Level 1	Level 2	Level 3
Requirement of 3.3.1.2.1	-0.02	-0.05	-0.08
Figure 5 (Reference 155)	-0.035	-0.084	-0.107
Figure 6 (Reference 157)	-0.020 to -0.035	-0.095	-1.121
Figure 6 (Reference 154)	-0.010	—	—
Figure 7 (Reference 156, no thrust lag)	+0.010	-0.190	-0.360
Figure 7 (Reference 156, thrust lag)	+0.017	-0.060	-0.125

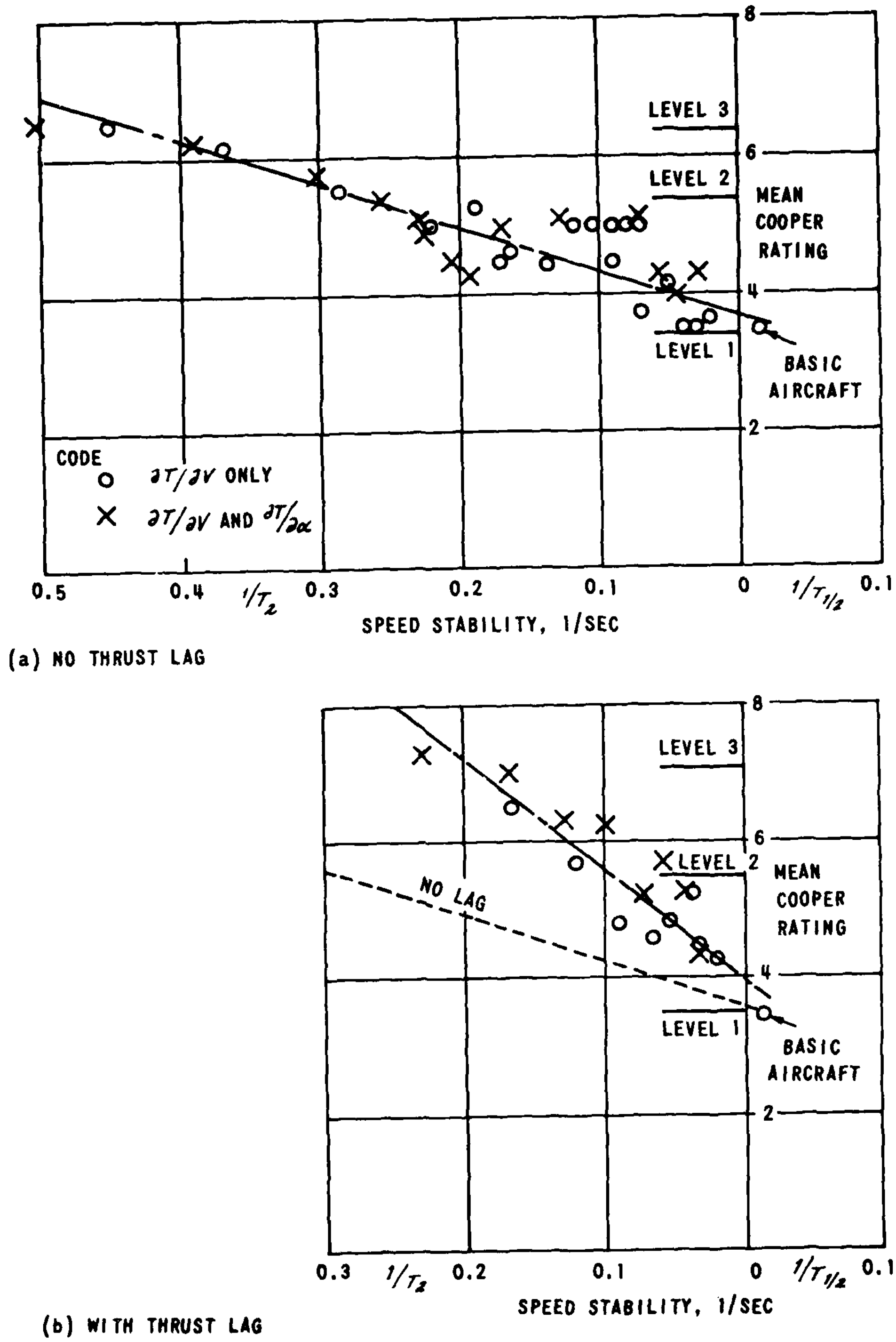


Figure 7 (3.3.1.2). Landing Approach
(AVRO 707, Reference 156)

The primary problem with Figure 7 seems to be that the majority of the data points are for VFR approaches with unusually good flight-path information available to the pilot (see Reference 156).

H. LESSONS LEARNED

There have been numerous aircraft that have been judged as unacceptable because of backside characteristics. The most recent of these is the F-16, which has notable deficiencies (Reference 125) in the landing approach flight condition. These deficiencies were specifically attributed to "flight path instabilities." $d\gamma/dV$ at the approach angle of attack (13 deg) was 0.15 (Level 2). It was also noted that pitch attitude control was imprecise, which compounded the problem.

3.3.1.2.1 Relaxation for aircraft with designated flight path controller

A. REASON FOR REQUIREMENT

This requirement represents a relaxation to 3.3.1.2 for aircraft equipped with a specific flight path controller. Such a relaxation is warranted since the primary control of long-term flight path can be accomplished with some control other than pitch attitude.

B. RELATED MIL-F-8785C REQUIREMENT

3.6.2

C. STATEMENT OF REQUIREMENT

3.3.1.2.1 Relaxation for aircraft with designated flight path controller. For aircraft with a designated secondary flight path control the required flight path response to attitude changes is ______________.

D. RATIONALE BEHIND REQUIREMENT

For most conventional aircraft this requirement is not applicable. For such aircraft as STOLs, where primary control of flight path is not with pitch attitude, a relaxation of 3.3.1.2 (i.e., operation well on the back side) should be allowed. A STOL amendment, currently under development, will address requirements such as this.

E. GUIDANCE FOR APPLICATION

F. DEMONSTRATION OF COMPLIANCE

G. SUPPORTING DATA

H. LESSONS LEARNED

No information is available at this time.

3.3.2 Vertical Axis Response to Designated Flight Path Controller

3.3.2.1 Vertical axis response to designated flight path controller — transient response

3.3.2.2 Vertical axis response to designated flight path controller — steady-state response

A. REASON FOR REQUIREMENTS

These requirements are intended to be the primary flight path control criteria for STOL aircraft. These aircraft operate well on the back side of the power-required curve and therefore use a designated controller other than pitch attitude (such as throttle) to control flight path.

B. RELATED MIL-F-8785C REQUIREMENT

3.6.2

C. STATEMENT OF REQUIREMENTS

3.3.2.1 Vertical axis response to designated flight path controller — transient response. When used as a primary controller the short-term flight path response to designated flight path controller inputs shall have the following characteristics: __________.

3.3.2.2 Vertical axis response to designated flight path controller — steady-state response. At all flight conditions the flight path controller will produce flight path motions in the same direction as the applied control and which are of the same sign as the steady-state values.

D. RATIONALE BEHIND REQUIREMENTS

There is a large body of data for STOL flight path control with thrust and DLC devices. These data will be incorporated into these paragraphs during development of the STOL amendment.

E. GUIDANCE FOR APPLICATION

F. DEMONSTRATION OF COMPLIANCE

G. SUPPORTING DATA

H. LESSONS LEARNED

None.

3.3.3 Vertical Axis Response to Other Inputs

3.3.3.1 Vertical axis response to auxiliary controls, stores release, and armament delivery

3.3.3.2 Vertical axis response to failures

A. REASON FOR REQUIREMENTS

Changes in the flight path to infrequent but anticipated inputs must not be objectionable to the pilot.

B. RELATED MIL-F-8785C REQUIREMENTS

3.4.6, 3.4.7, 3.4.8, 3.4.9

C. STATEMENT OF REQUIREMENTS

3.3.3 Vertical Axis Response to Other Inputs

3.3.3.1 Vertical axis response to auxiliary controls, stores release, and armament. There shall be no objectionable transients in flight path response due to the use of other auxiliary controls, or stores or armament release.

3.3.3.2 Vertical axis response to failures. No single failure of any component or system shall result in objectionable flying qualities.

D. RATIONALE BEHIND REQUIREMENTS

Requirements are needed in the vertical axis that correspond to similar statements in the other axes. Ultimately a more thorough set of requirements should be developed.

E. GUIDANCE FOR APPLICATION

F. DEMONSTRATION OF COMPLIANCE

G. SUPPORTING DATA

H. LESSONS LEARNED

Supportive discussions for these areas should be developed as more information becomes available.

3.3.4 <u>Flight Path Control Power</u>

3.3.4.1 <u>Control power for designated primary flight path controller</u>

3.3.4.2 <u>Control power for designated secondary flight path controller</u>

3.3.5 <u>Flight Path Controller Characteristics</u>

A. REASON FOR REQUIREMENTS

This set of requirements defines the effectiveness and cockpit characteristics of the designated flight path controller. They are oriented toward STOL aircraft.

B. RELATED MIL-F-8785C REQUIREMENTS

3.5.2.1

C. STATEMENT OF REQUIREMENTS

3.3.4 <u>Flight Path Control Power</u>

3.3.4.1 <u>Control power for designated primary flight path controller</u>. If a separate control is provided for direct lift or flight path it shall be capable of producing the following changes in flight path ____________________ following full actuation of the controller. This shall be accomplished with pitch attitude held fixed and the speed trimmed for __________.

3.3.4.2 <u>Control power for designated secondary flight path controller</u>. The secondary controller shall be sufficient to produce the following changes in flight path: ______________________.

3.3.5 <u>Flight Path Controller Characteristics</u>. The breakout, centering, and force gradient characteristics of the designated flight path controller shall be within the following limits:

Breakout: ± ___________ lb

Centering: ± ___________ %

Force gradient: ______________

D. RATIONALE BEHIND REQUIREMENTS

These requirements are grouped together because of their similar purposes, and because there is not enough data to recommend values for them. A STOL amendment for the MIL Standard will provide guidance.

E. GUIDANCE FOR APPLICATION

F. DEMONSTRATION OF COMPLIANCE

G. SUPPORTING DATA

H. LESSONS LEARNED

No information is available at this time.

3.4 HANDLING QUALITY REQUIREMENTS FOR LONGITUDINAL (SPEED) AXIS

3.4.1 Speed Response to Attitude Changes

A. REASON FOR REQUIREMENT

This requirement is intended to insure that the aircraft will not diverge in attitude and speed during intermittent periods of unattended pilot operation.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.1.1

C. STATEMENT OF REQUIREMENT

3.4.1 Speed Response to Attitude Changes.

a. The correlation between airspeed and pitch attitude shall be as follows: ________.

b. For Levels 1 and 2 there shall be no tendency for the airspeed to diverge aperiodically when the aircraft pitch attitude is disturbed from trim by any means. This requirement shall be considered satisfied if the gradient of pitch control force with airspeed is negative. Demonstration of positive phugoid damping in Paragraph 3.2.1 shall also be accepted as evidence of compliance.

c. For Level 3, the airspeed divergence characteristics must be within the following limits: ____.

Recommendation for Requirement a:

Transient Response. For rapid attitude changes the short-term airspeed change should be in the same direction as the final value.

Steady State Response. For a fixed positive change in attitude from trim, airspeed should not increase. This applies over a speed range of ±15 percent about trim or ±50 kt, whichever is less.

Recommendation for Requirement c:

> It is recommended that the time for airspeed to double amplitude following a pitch attitude disturbance from trim be not less than 6 seconds. Additionally, airspeed divergences should not be allowed in the presence of one or more other Level 3 flying qualities unless the flight safety of that combination of characteristics can be demonstrated.

D. RATIONALE BEHIND REQUIREMENT

This requirement insures positive static stability for Levels 1 and 2 and limits the amount of negative static stability for Level 3. It also requires that the airspeed track the pitch attitude in the conventional way (decreasing airspeed decreases with increasing pitch attitude) both in the long- and short-term response.

Static stability means that restoring pitching moments are generated when the airspeed is disturbed from trim. Airspeed is easily measured inasmuch as it is always available as a cockpit display. Furthermore, in most circumstances airspeed is more meaningful to most pilots than angle of attack.

From a piloting standpoint, pitch attitude is the primary longitudinal control variable. Pilots quickly learn that good control of pitch attitude 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 rate command/attitude hold augmentation, which has zero static stability in the classical sense (airspeed stability). In fact, for all but the most unconventional aircraft, this requirement is redundant with the requirements on pitch attitude in Paragraph 3.2.1. It is retained in this standard primarily to insure acceptable airspeed response characteristics for augmented aircraft that may have unconventional airspeed responses to changes in pitch attitude.

E. GUIDANCE FOR APPLICATION

Aircraft that meet the equivalent phugoid and short-period requirements of Paragraph 3.2.1.1 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 it is conceivable that the autothrottle could result in increasing airspeed with increasing pitch attitude. Such undesirable characteristics would be disallowed by this requirement.

F. DEMONSTRATION OF COMPLIANCE

This requirement will be considered satisfied if the gradient of pitch control force with airspeed is negative, i.e., the aircraft will return to its trim angle of attack after a disturbance. There seems no reason to require a stable pitch control <u>displacement</u> gradient with airspeed inasmuch as a stable force gradient assures a convergent airspeed response. Moreover, insisting on a stable control position gradient results in significant restriction in the aft c.g. limit, which of course limits aircraft utility. Downsprings and bobweights are utilized to augment the stick force gradient when the stick position gradient is zero on many successful aircraft, although overdoing that can induce a dynamic instability.

It is well recognized that the stick force gradient with airspeed is an important flying quality metric. Unfortunately, there seems to be no general agreement on what constitutes the lower boundary. Hence we require only that the gradient be less than zero.

A succinct description of measurement techniques for compliance with this requirement appeared as Appendix IVA of Reference 11. It is appropriate to reprint that description here.

The obvious method for determining the stick force gradient is to first trim the aircraft and 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 least. 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 a popular method because it is the quickest. After trimming, 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 provides an almost infinite number of data points, because 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 from the above two methods will be essentially equal. At high speeds, the gradients will differ by a factor that depends on altitude and airspeed. The difference is primarily due to the fact that large altitude changes accompany small airspeed changes during high-speed flight at constant throttle. This means that air density and the speed of sound will vary appreciably during static stability tests using the stabilized method, but not during acceleration-deceleration tests. 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 for high-speed flight conditions. For this reason, the acceleration-deceleration method is generally preferable for testing at high speeds.

A possible source of error, which can accompany the acceleration-deceleration tests, should be mentioned. This error is often present because 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. Unless the engine thrust and slipstream effects due to changing throttle and airspeed are known before the test, it is obvious that the control force and position data must be obtained with the throttles at their trim settings.

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 (say $M > 0.4$) where the 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 a reasonable indication of throttle-fixed stability, the following procedure should be used. After trimming the aircraft, reduce throttle and allow the airplane to decelerate at constant altitude to the low end of the desired speed range, taking no data. When the desired speed is reached, advance the throttle to the trim setting and hold normal acceleration as close to 1.0 g as is possible without use of abrupt control movements. The reverse procedure should be used for speeds above the trim speed. Data should only be taken during the acceleration and deceleration runs where the throttle is at the trim setting.* For climbing or descending Flight Phases, other appropriate throttle settings should 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 additional trim points will be needed. If an unstable region is found far from the trim point, however, the test should be repeated with the aircraft trimmed closer to the unstable region; the airplane may or may not be stable within the specified speed range about the new trim point.

Aircraft having certain types of SAS, 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 Paragraph 3.2.1.1 should be utilized to show compliance with this paragraph.

G. SUPPORTING DATA

The data supporting this requirement are the same as the data supporting the phugoid requirement in Paragraph 3.2.1.1.

H. LESSONS LEARNED

The requirement for a convergent airspeed response for acceptable flying qualities is well recognized. Recent experience in testing modified control laws on the F-16 have shown that some level of speed stability is necessary in the approach flight condition. The augmentation insures good attitude stability with or without angle of attack feedback. However, pilots indicated that the speed cue (provided

*The combined effect of thrust and acceleration can be seen by comparing acceleration and deceleration data, but for showing specification compliance only the data for the trim throttle setting are pertinent.

by the angle of attack feedback) was necessary to avoid inadvertent stalls near the ground.

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 seconds to double amplitude.

3.4.1.1 Speed response to attitude changes -- relaxation in transonic flight

A. REASON FOR REQUIREMENT

Aircraft naturally exhibit local instabilities in the transonic region. Considering that operation in this region is almost always transient in nature, it seems reasonable to allow limited levels of static instability.

B. RELATED MIL-F-8785C REQUIREMENT

3.2.1.1.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.4.1.1 Speed response to attitude changes -- relaxation in transonic flight. The requirements of 3.4.1 may be relaxed in the transonic speed range as follows: ______________.

Recommended values (provided any divergent aircraft motions with speed are gradual and not objectionable to the pilot):

a. Levels 1 and 2: For centerstick controllers, no local force gradient should be more unstable than 3 pounds per 0.01 M nor should the force change exceed 10 pounds in the unstable direction. The corresponding limits for wheel controllers are 5 pounds per 0.01 M and 15 pounds, respectively.

b. Level 3: For centerstick controllers, no local force gradient should be more unstable than 6 pounds per 0.01 M nor should the force ever exceed 20 pounds in the unstable direction. The correspondings limits for wheel controllers are 10 pounds per 0.01 M and 30 pounds, respectively.

This relaxation should not be applied to Level 1 for any Flight Phase which requires prolonged transonic operation.

D. RATIONALE BEHIND REQUIREMENT

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 as that where a relaxation is necessary; such a definition would leave essentially no requirement for stability. For aircraft that do not have supercritical wings, the lower end of the transonic region might be taken as the drag-rise Mach number. 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. In any case the relaxation is not meant to apply at any flight condition at which an operational mission requires prolonged operation.

E. GUIDANCE FOR APPLICATION

Since phugoid oscillations involve speed changes, all speeds at which operational missions might require prolonged flight should be reasonably far removed from the region of transonic trim changes. Otherwise, normally encountered disturbances would cause divergence.

A statement should be included in the detailed system definition for each procurement delineating if the relaxation is to be applied and for which Flight Phases.

F. DEMONSTRATION OF COMPLIANCE

See discussion for 3.4.1.

G. SUPPORTING DATA

None required.

H. LESSONS LEARNED

None required.

3.4.2 Speed Response to Speed Controller

3.4.2.1 Speed response to speed controller — transient response

3.4.2.2 Speed response to speed controller — steady-state response

3.4.3 Speed Axis Response to Other Inputs

3.4.4 Speed Axis Control Power

3.4.5 Speed Axis Controller Characteristics

A. REASON FOR REQUIREMENTS

These requirements delineate responses in the speed axis, both to the designated speed controller and to other inputs. They are oriented toward STOL aircraft.

B. RELATED MIL-F-8785C REQUIREMENTS

3.4.6, 3.4.7, 3.4.8, 3.4.9, 3.5.2.1

C. STATEMENT OF REQUIREMENTS

3.4.2 Speed Response to Speed Controller

3.4.2.1 Speed response to speed controller — transient response. The short-term airspeed response to the designated speed controller shall have the following characteristics: ____________________.

3.4.2.2 Speed response to speed controller -- steady-state response. The steady-state airspeed response to a step change of the designated speed controller shall have the following characteristics: ____________________.

3.4.3 Speed Axis Response to Other Inputs. There shall be no airspeed responses due to use of other controls, stores or armament release, configuration changes, or failures of any system or subsystem that result in objectionable flying qualities.

3.4.4 Speed Axis Control Power. The speed controller shall be capable of providing the following range of speeds throughout the Operational Flight Envelope: ____________________.

3.4.5 <u>Speed Axis Controller Characteristics</u>.

a) Breakout forces shall not exceed ____________.

b) Friction shall be adjustable from ___ lb to ___ lb.

c) *Displacements* shall be sufficient to provide from idle to full thrust and shall not be so large or so small as to be objectionable. The average control gradient shall not be less than __________.

D. RATIONALE BEHIND REQUIREMENTS

This block of requirements covers a large percentage of the speed axis requirements. They have been presented as a set here because, while there is clear justification for all of them, there is not enough data collected to date to discuss any of them in detail. The requirements are oriented toward STOL aircraft such as the YC-14, which had a separate speed controller. It is expected that some information in this area will be uncovered during work on the STOL amendment.

E. GUIDANCE FOR APPLICATION

F. DEMONSTRATION OF COMPLIANCE

G. SUPPORTING DATA

H. LESSONS LEARNED

The YC-14 experience should be investigated for lessons learned in this area. Especially relevant would be the speed control characteristics of the Coanda flaps in wind shear.

3.5 HANDLING QUALITY REQUIREMENTS FOR ROLL AXIS

3.5.1 Roll Response to Roll Controller

3.5.1.1 Roll axis lower-order equivalent system requirements

3.5.1.1.1 Roll mode

A. REASON FOR REQUIREMENT

This requirement is directed at precision of control in the roll axis. For aircraft that exhibit classical spiral (large spiral time constant) and dutch roll (low $|\phi/\beta|$) characteristics, the equivalent roll mode time constant (T_R) describes the airplane roll damping. For airplanes with a roll rate response that is not easily approximated by a first order, an alternative specification such as bandwidth may be in order.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.1.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.1.1.1 Roll mode. The equivalent roll mode time constant, T_R, *shall be no greater than the following:* ________________.

Recommended values: For the equivalent roll rate transfer function defined as follows, the maximum equivalent roll mode time constant, T_R, is given in Table 1:

$$\frac{p}{F_{as}} = \frac{K_p(0)[\zeta_\phi,\ \omega_\phi]e^{-\tau_{e_p}s}}{(1/T_s)(1/T_R)[\zeta_d,\ \omega_d]} \qquad (1)$$

The equivalent system should be fit to the higher-order system using algorithms similar to those specified in Appendix A, over the frequency range from 0.1 rad/sec to 10.0 rad/sec.

TABLE 1 (3.5.1.1.1)

RECOMMENDED MAXIMUM ROLL-MODE TIME CONSTANT (Seconds)

FLIGHT PHASE CATEGORY	CLASS	LEVEL 1	LEVEL 2	LEVEL 3
A	I, IV II, III	1.0 1.4	1.4 3.0	
B	All	1.4	3.0	10
C	I, II-C, IV II-L, III	1.0 1.4	1.4 3.0	

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 (see, for example, Reference 11, Appendix VB).

D. RATIONALE BEHIND REQUIREMENT

There are considerable data to show that pilot rating is a function of roll damping, for example Figure 1 (from Reference 37). 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.

E. GUIDANCE FOR APPLICATION

The considerable advances made in modeling higher-order aircraft responses with equivalent systems have, largely, been only for the pitch axis. Similar work is clearly necessary in the roll and yaw axes. It is expected that application of this requirement will involve some sort of reduced-order matching, whether it be by frequency (Appendix A) or time (Reference 11) domain techniques. At this time inadequate information exists to supply significant guidance for applying equivalent systems to meet this requirement.

Limits for the other parameters in the p/F_{as} transfer function, Equation 1, are given in other paragraphs in this Handbook.

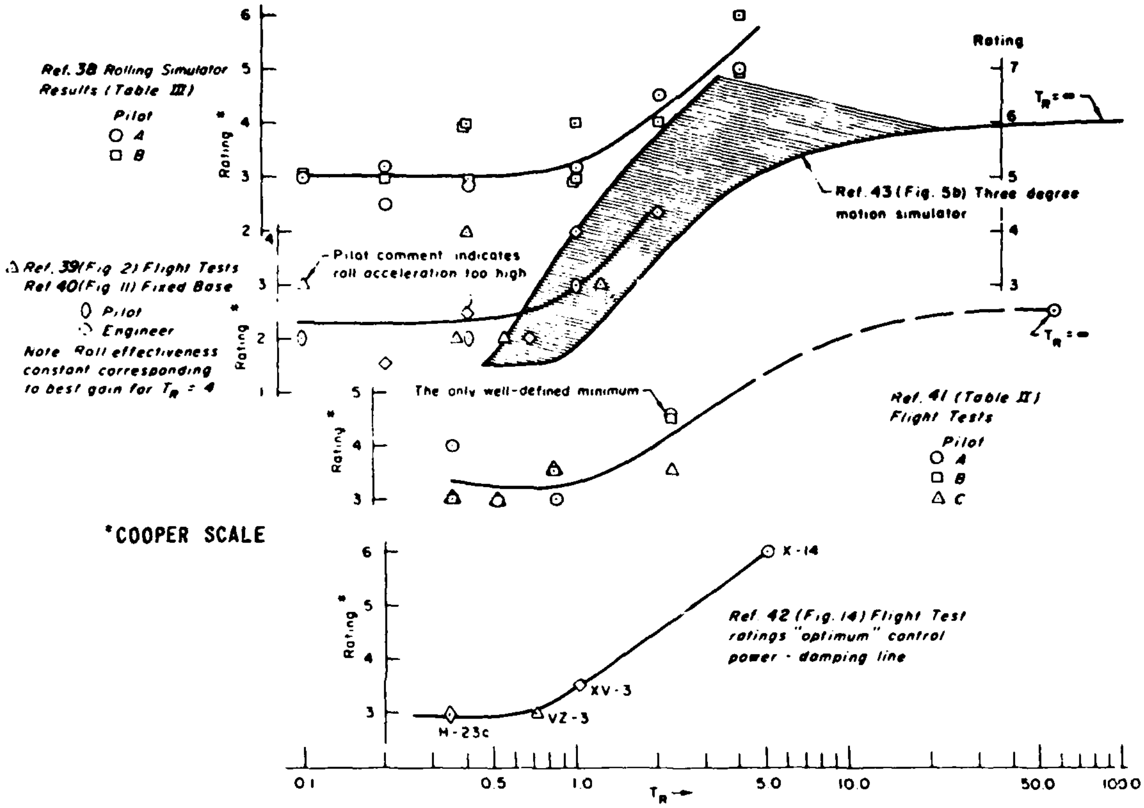


Figure 1 (3.5.1.1.1). Ratings Versus Roll Damping — Flight Test, Moving-Base, Fixed Base with Random Input (from Reference 37)

F. DEMONSTRATION OF COMPLIANCE

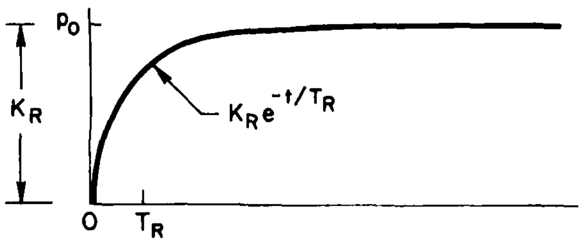
Compliance is easily demonstrated through flight testing at the minimum and maximum specified operational altitudes, over the range of service speed, with the airplane configured for maximum rolling moment of inertia.

The roll rate response to a step roll control input for airplanes with conventional response is usually made up of three distinct modes: the roll mode, the spiral mode, and the dutch roll mode. 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. 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.

The p/F_{as} response function, given in Equation 1, can be transformed to the time domain. Assuming τ_{e_p} is small, the roll rate time history following a step roll control input is given by:

$$\left.\frac{p(t)}{\delta_a}\right|_{step} = K_s e^{-t/T_s} + K_R e^{-t/T_R} + K_d e^{-\zeta_d \omega_d t} \cos\left[\omega_d \sqrt{1 - \zeta_d^2}\, t + \psi_p\right]$$

For a normal airplane, the roll mode, characterized by the first-order time constant, T_R, takes on the following form following a step aileron input:



Methods of extracting values of T_R from flight test data are given in Reference 11.

G. SUPPORTING DATA

The MIL-F-8785B BIUG (Reference 11) contained a concise description of data available for development and support of the recommended values of Table 1. The following discussion is primarily taken from Reference 11.

1. **Level 1 Requirements**

The starting point for specification of the criteria was the recommendation pertaining to roll mode time constant given in References 37 and 38. Both references report on extensive surveys of roll flying qualities and so were directly applicable to this effort. Reference 38 proposes a maximum T_R = 1.3 seconds for Class IV airplanes and T_R = 1.5 seconds for all other classes (Figure 2). From theoretical considerations and from analysis, Reference 37 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 (References 16, 44, and 46) indicate that, for Class I and IV airplanes performing precision tasks, even lower values of T_R are required to obtain satisfactory flying qualities.

Reference 44 (Figures 3, 4, and 5) shows that maximum satisfactory T_R for fighter aircraft for a carrier approach is approximately 1 second. Reference 46 (Figure 6) shows that with a T_R = 1.3 seconds, 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 fighter aircraft should have a T_R = 0.6 to 0.8 seconds. Reference 16 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.

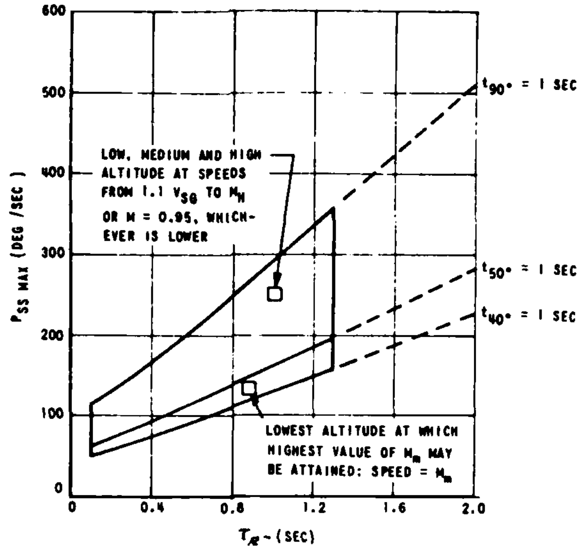


Figure 2 (3.5.1.1.1). Proposed Roll Performance Requirements (MIL-F-8785) for Class III Aircraft (from Reference 45)

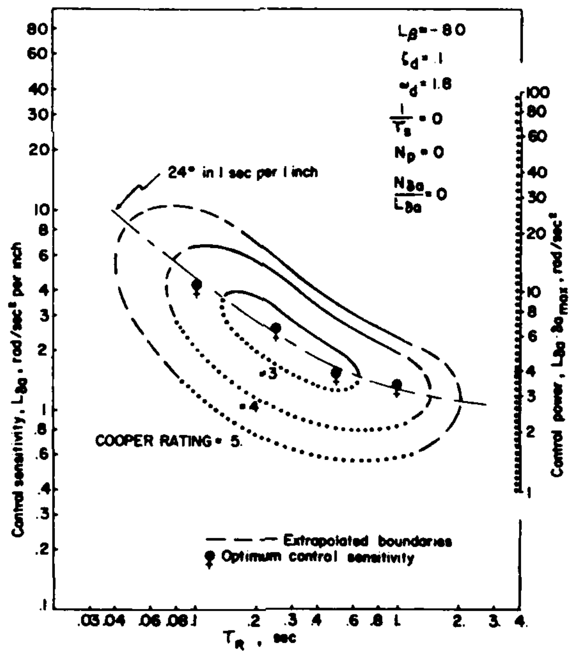


Figure 3 (3.5.1.1.1). Lateral Control Boundaries (from Reference 44)

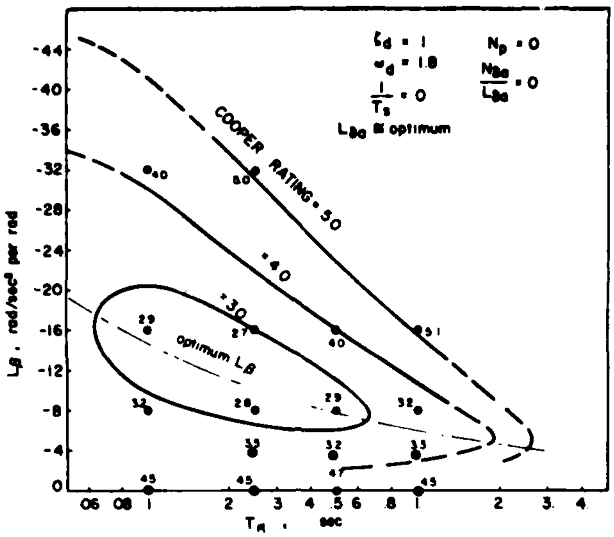


Figure 4 (3.5.1.1.1). Lateral Flying Qualities Boundaries (L_β vs. T_R, ζ_d = 0.1) (from Reference 44)

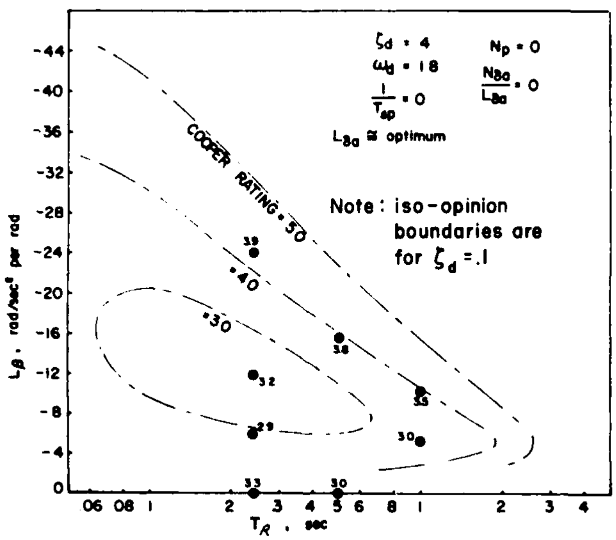


Figure 5 (3.5.1.1.1). Lateral Flying Qualities Boundaries (L_β vs. T_R, ζ_d = 0.4) (from Reference 44)

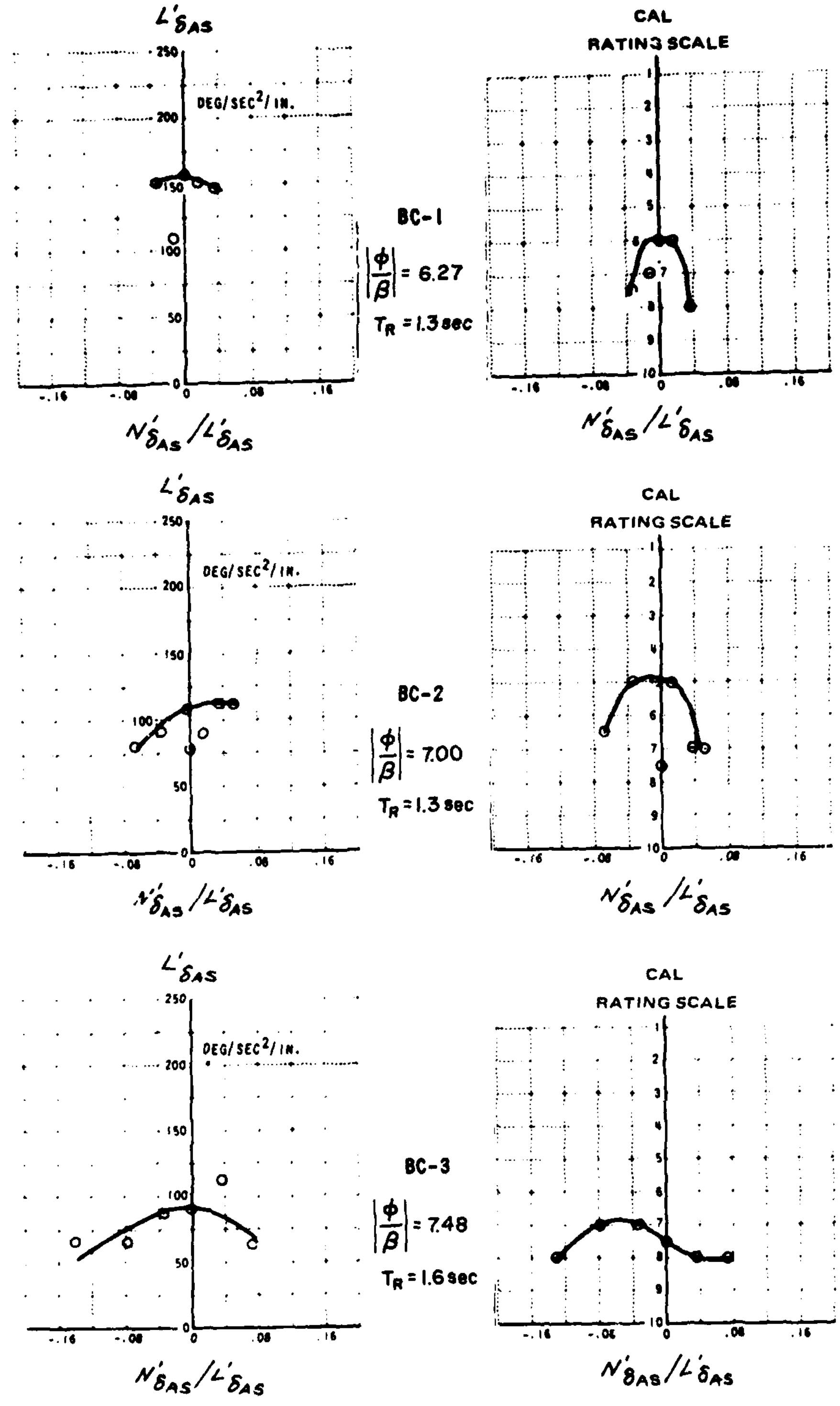


Figure 6 (3.5.1.1.1). Pilot Ratings and Optimum Aileron Sensitivity (Medium $|\phi/\beta|_d$, Long T_R) (from Reference 46)

The data of Reference 38 (Figures 7 through 11) have been widely referenced and interpreted, as for example in References 37 and 45. It should be noted, however, that the in-flight evaluations in Reference 38 were all for T_R less than 0.8 seconds (Figure 10) and any conclusions about roll mode time constants longer than 0.8 seconds would be based on the ground simulation data only. In general, the in-flight ratings of Reference 38 were worse than for the single-degree-of-freedom ground simulation ratings (Figure 11). This indicates that the presented one-degree-of-freedom data (Figure 9) may be a little optimistic. This difference in pilot ratings was discussed in Reference 38:

> The principal 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 10] 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 (Figures 7 and 8) may be considered to represent "ideal" aircraft.

Since, in general, a knee occurs in most of the data at approximately $T_R = 1$ second (Figure 1), and since $T_R = 1$ second is at least consistent with all pertinent data, this value of T_R has been selected as the recommended Level 1 limit for Class I, II-C, and IV airplanes for Flight Phase Category C, and for Class I and IV airplanes for Flight Phase Category A.

For Class I and IV airplanes performing Flight Phase Category B tasks, and for Class II-L and III airplanes performing all tasks, available data support a maximum value of $T_R = 1.3$ to 1.5 seconds; an average value of $T_R = 1.4$ seconds was selected. Ground simulator data in Reference 43 tend to support this value for large aircraft (cross-hatched

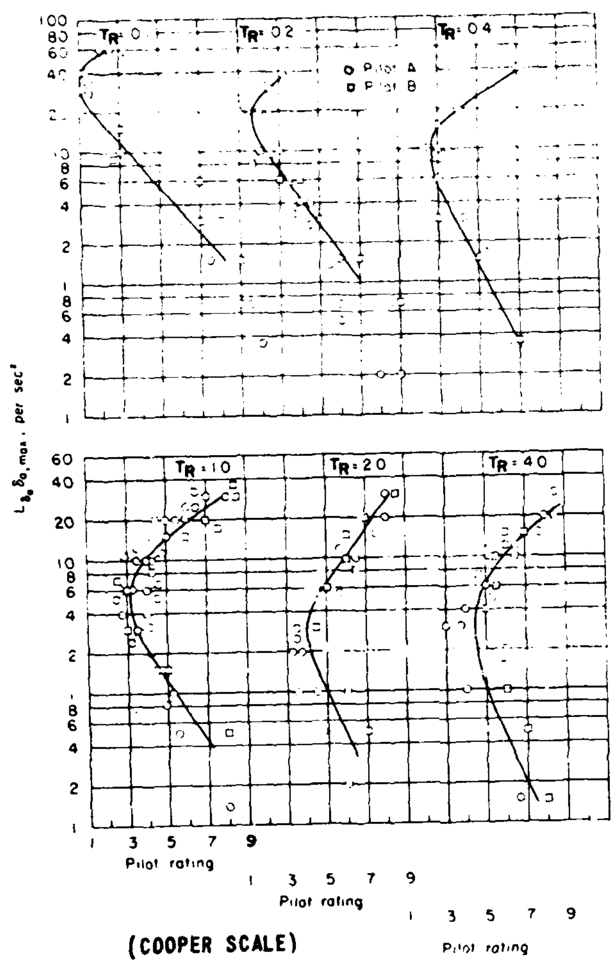


Figure 7 (3.5.1.1.1). Variation of Pilot Opinion with $L_{\delta_a}\delta_{a_{max}}$, For Constant Values of T_R as Obtained from the Stationary Flight Simulator (from Reference 38)

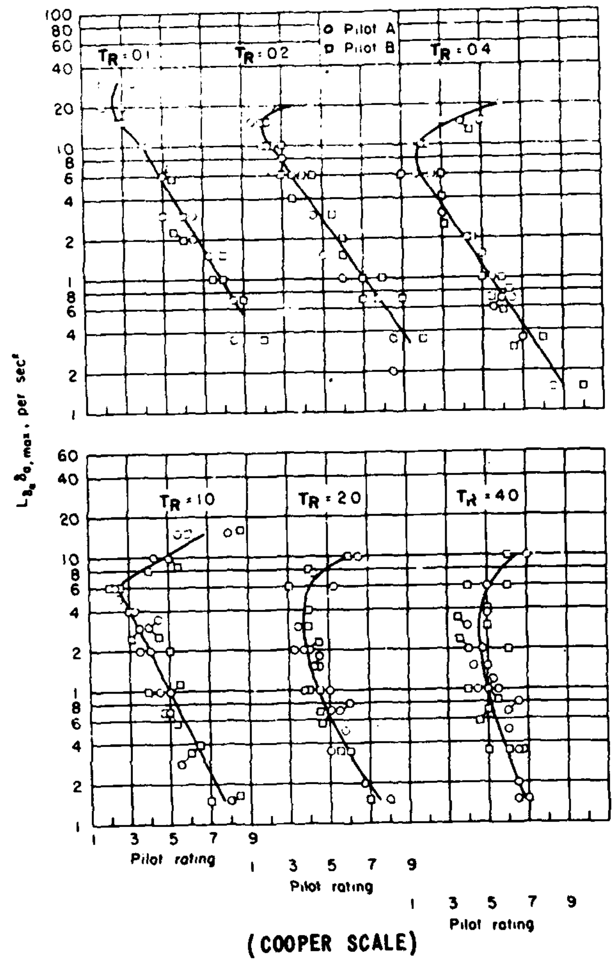


Figure 8 (3.5.1.1.1). Variation of Pilot Opinion with $L_{\delta_a}\delta_{a_{max}}$, for Constant Values of T_R as Obtained from the Moving Flight Simulator (from Reference 38)

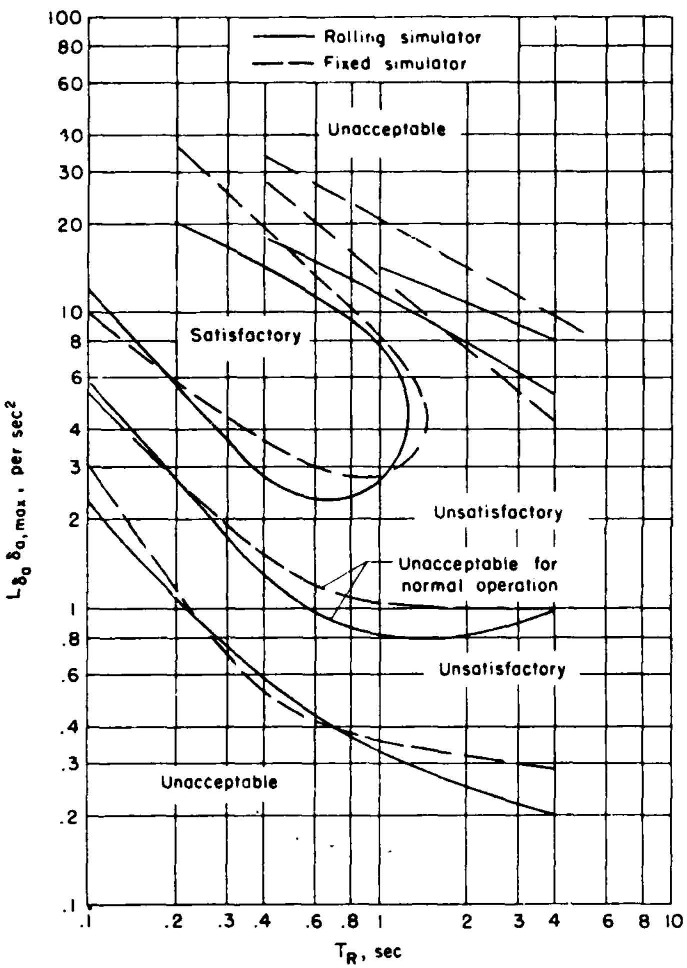


Figure 9 (3.5.1.1.1). Comparison of Pilot Opinion Boundaries Obtained from the Fixed and Moving Flight Simulators. (From Reference 38)

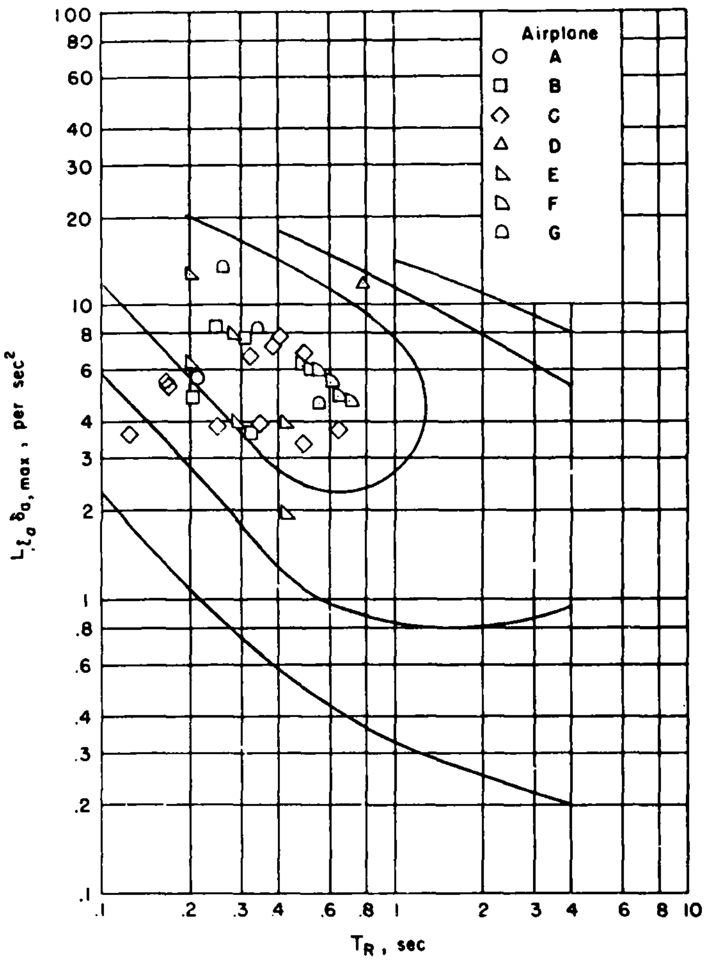


Figure 10 (3.5.1.1.1). Range of Parameters $L_{\delta_a}\delta_{a_{max}}$, and T_R Covered in the Flight Investigation, Shown in Comparison with the Motion Simulator Derived Boundaries (from Reference 38)

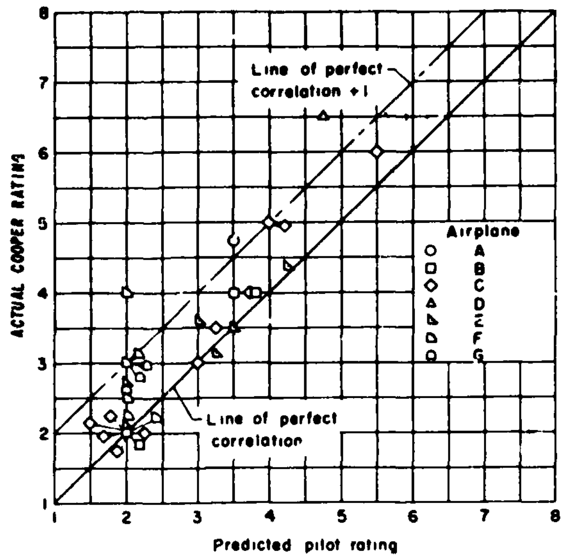


Figure 11 (3.5.1.1.1). Comparison of In-Flight Pilot-Opinion Rating with Those Predicted from Flight Simulator Boundaries (from Reference 38)

curves in Figure 1); and in-flight data in Reference 39 for small Class II airplanes, Flight Phase Category B (Figure 12), support a T_R at least greater than 1.2 seconds.

An additional consideration that is demonstrated by much of the data, for example References 44 and 49, 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 Reference 44 (Figures 4 and 5) show this dependence directly. In the opinion of the author of Reference 49, the main reason for the differences between the data of Reference 49 and the data to which it is compared (see Figure 13) is that the Reference 49 ground simulator data were based on a much larger value of $|\phi/\beta|_d$ response ratio. In addition, the lack of an adequate flight path display for the simulated high-speed condition ($M = 1.2$) of Reference 49 may have contributed. The pilot ratings of both the Reference 49 and 44 data are degraded because of the response to atmospheric disturbances. This phenomenon is discussed in the substantiation for the paragraph covering the response to atmospheric disturbances.

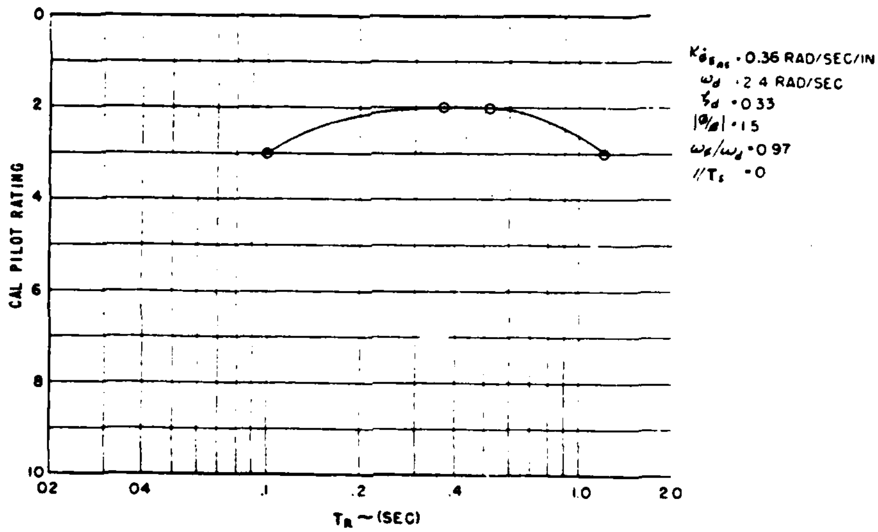


Figure 12 (3.5.1.1.1). Pilot Rating Versus Roll Mode Time Constant (from Reference 39)

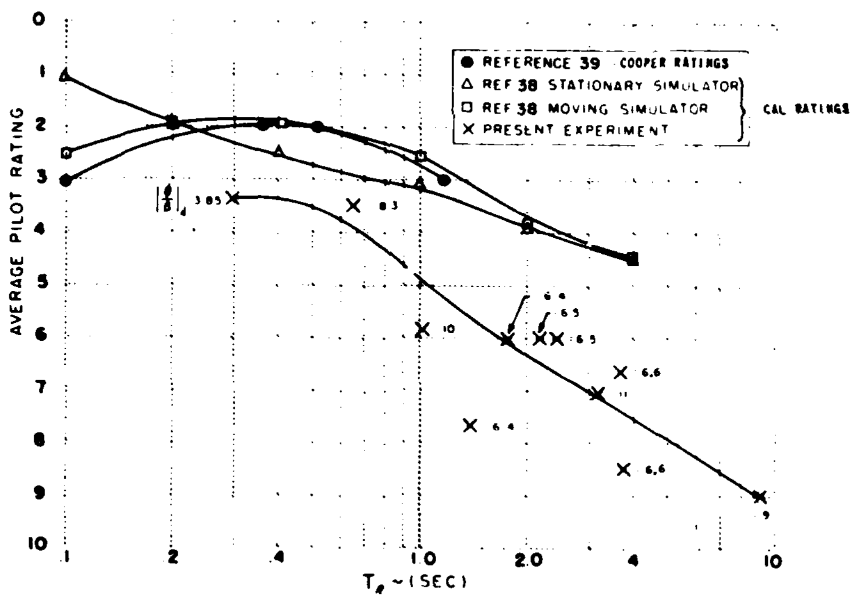


Figure 13 (3.5.1.1.1). Average Pilot Rating of Roll Mode Time Constant (from Reference 49)

2. Level 2 Requirements

References 37 and 45 do not make recommendations that pertain to 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 1 (from Reference 37), which summarizes data from References 38 and 43, shows that for a change in pilot rating from 3-1/2 to 5 or 5-1/2, T_R goes from approximately 1.3 to 3 seconds. Thus, even though the Reference 38 data are based on a fighter mission, the data do indicate the gradient of pilot rating with T_R over the above noted ranges. Reference 46 indicates, from in-flight evaluations, that for fighter aircraft performing precision and maneuvering tasks, the pilot ratings degraded to marginally acceptable for T_R values of 1.3 to 1.6 seconds. For large airplanes, Reference 47 suggests T_R values of 2.3 seconds for the satisfactory level, and 6 seconds for the acceptable level; however, these levels are probably associated with somewhat poorer flying qualities than are Cooper-Harper Levels 1 and 2. The Level 2 recommendations were selected from these considerations

3. Level 3 Requirement

A Level 3 value of T_R = 10 seconds is relatively arbitrary but is based on data of Reference 49 (Figure 13) for fighter aircraft. While the selected value of T_R 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 on wingless vehicles.

H. LESSONS LEARNED

A comprehensive series of flight tests was recently conducted on the USAF/Calspan variable stability NT-33 to investigate the effect of higher-order system dynamics on lateral handling qualities ("LATHOS" for lateral high-order systems, Reference 258). The effect of roll mode time constant obtained in LATHOS is given in Figure 14. Values of T_R

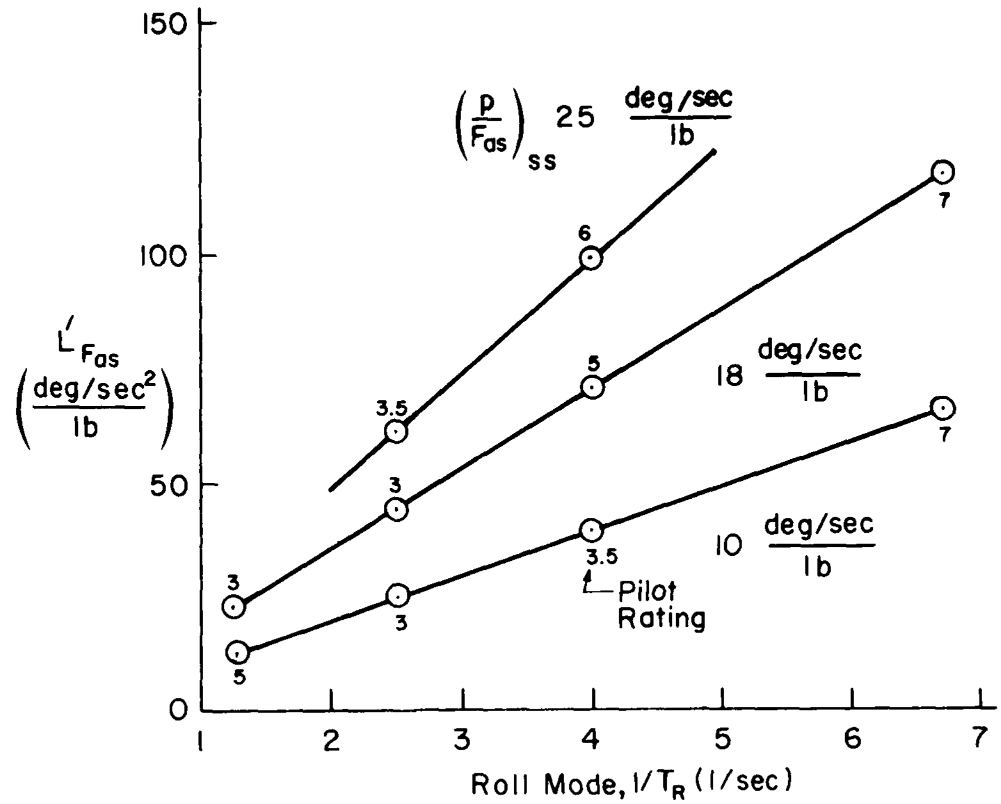


Figure 14 (3.5.1.1.1). Effect of Roll Mode -- LATHOS (Reference 258), Category A

existing Level 1 boundary were tested. However, the data for $1/T_R$ greater than 1 supports the current boundary ($T_R < 1.0$ sec) up to a value of $1/T_R \doteq 3$ ($T_R \doteq 0.33$). For $1/T_R$ greater than 3 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 Para. 3.5.10.3. The following will examine only the effects of low values of T_R.

Several examples of ratcheting are shown in Figures 10, 12, and 13 of Para. 3.5.10.3. The phenomenon is characterized by near-limit-cycle oscillations at frequencies between 2 and 3 cycles per second (12 and 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 (Reference 266) that the root cause of ratcheting is related to 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.

Another possible explanation for 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 that of a "bobweight" which would feed back to the aircraft. This phenomenon has been related to longitudinal pilot-induced oscillations (Reference 225). Experiments conducted at the Air Force Aerospace Medical Research Laboratory (AMRL) investigated pilot control performance while experiencing sinusoidal lateral vibrations (Reference 267). A simple roll-bar-tracking maneuver with a well-behaved controlled element was utilized. Figure 15 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 second (12 rad/sec) — near the frequencies of the observed ratcheting oscillations in the LATHOS experiment.

Several solutions to the problem of excessive sensitivity have been found. These are discussed in conjunction with the roll sensitivity discussion in Para. 3.5.10.3. Those solutions are to: 1) decrease the stick sensitivity around neutral; 2) minimize the augmented aircraft value of $1/T_R$; and 3) add a low-frequency stick prefilter with a break frequency of at least 10 1/sec.

Reduction of stick sensitivity for CAS-equipped aircraft is relatively straightforward. Use of nonlinear gradients (p/F_{as}) on LATHOS reduced the sensitivity only slightly but improved pilot ratings from 7

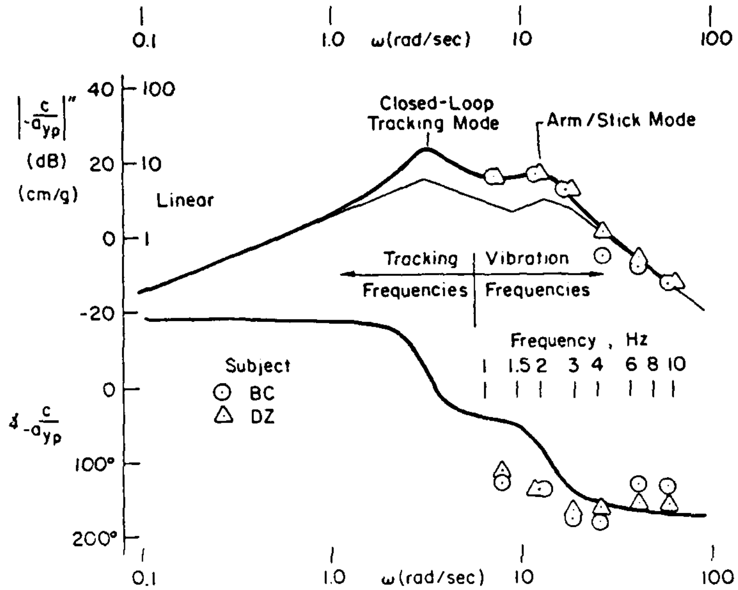


Figure 15 (3.5.1.1.1). Comparison of Models and Data for Closed-Loop Stick Deflection Responses Under Lateral Vibration (Reference 267)

to 4 [see Figures 14 (3.5.10.3) and 15 (3.5.10.3)]. 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 (Reference 265). 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 practical 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. In the longitudinal axis, a first-order lag of 3 rad/sec adds about 0.1 sec to overall τ_e [see Figure 9a (3.2.1.1)]. For the T-33 LATHOS experiment, where basic τ_e due to actuators was small (0.028 sec), this 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 in Figure 1 (3.5.1.1.5).

In summary, a large value of $1/T_R$ appears to result in excessive gain at high frequencies (see Figure 16) which seems to be the root cause of roll ratcheting. This can be alleviated to some extent by reducing the stick gain for small inputs, i.e., most high-frequency control activity occurs with low magnitude [see Figure 2 (3.5.10.3)]. 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 2 (3.5.10.3)].

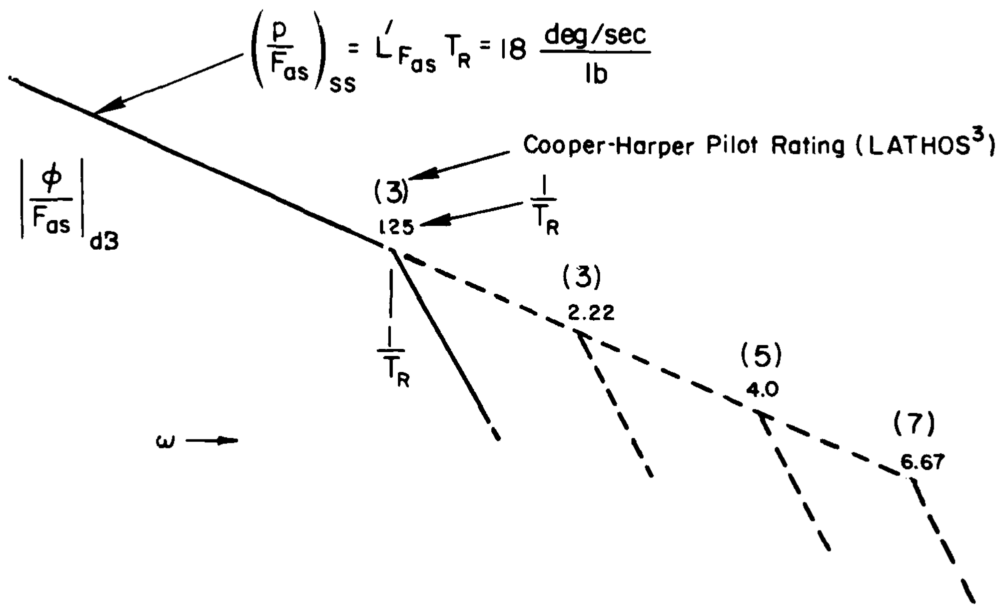


Figure 16 (3.5.1.1.1). Effect of $1/T_R$ on High Frequency Gain

3.5.1.1.2 Spiral stability

A. REASON FOR REQUIREMENT

The requirements on spiral stability are aimed primarily at insuring that the airplane will not diverge too rapidly in bank from a wings-level condition during periods of pilot inattention.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.1.3

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.1.1.2 Spiral stability. The combined effects of spiral stability, flight-control-system characteristics and rolling moment change with speed shall be such that the bank angle response shall have the following characteristics: ____________ following a disturbance in bank of up to 20 degrees. This requirement shall be met with the airplane trimmed for wings-level, zero-yaw-rate flight with the cockpit controls free.

TABLE 1 (3.5.1.1.2)

SPIRAL STABILITY — RECOMMENDED MINIMUM TIME TO DOUBLE AMPLITUDE

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

D. RATIONALE BEHIND REQUIREMENT

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 dangercus 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 (discussed in "Supporting Data"). Indeed, a coupled roll-spiral mode (3.5.1.1.3) is sometimes desirable.

E. GUIDANCE FOR APPLICATION

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 system fit of the p/F_{as} response, described in Para. 3.5.1.1.1 for the roll mode.

F. DEMONSTRATION OF COMPLIANCE

Bearing in mind that friction and asymmetric loadings must be accounted for, no special flight test techniques are required.

G. SUPPORTING DATA

Reference 45 recommended retention of the existing $T_2 > 20$ seconds requirement and also proposed a requirement, $T_{1/2} > 10$ seconds, on the degree of positive spiral stability permitted. The recommended 'lowable instability of Table 1 is similar, in that for Flight Phase Category B (analogous to the cruise configuration) the time to double amplitude is $T_2 > 20$ seconds. But for Flight Phase Categories A and C, where the pilot is generally closing a tight attitude loop, a less stringent value of $T_2 > 12$ seconds 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 airplane 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 Reference 50. 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 References 51 and 52, it is recommended that the Level 2 limit on T_2 be 8 sec. Even this limit is a conservative interpretation of the data in Reference 51, 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 Reference 53 (Figure 1) tend to suggest, however, that a higher value of T_2 might be justified for Level 2.

For Level 3, a value of $T_2 > 4$ seconds was selected as a compromise between what is flyable and what is controllable if the pilot cannot

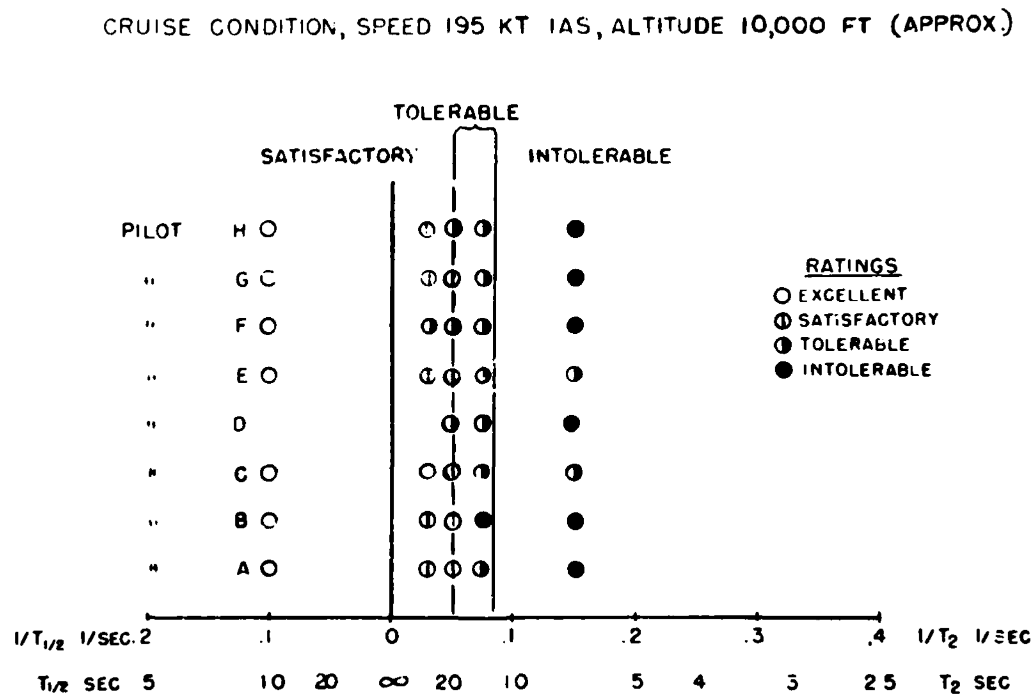


Figure 1 (3.5.1.1.2). Limits of Satisfactory and Tolerable Rates of Spiral Divergence (from Reference 53) devote full attention to flying the aircraft. This subject was discussed, as follows, in Reference 53:

> 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 (Reference [57]). 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 airplane 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 manuals, 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 Reference 45, a limit of $T_{1/2} > 10$ seconds on the degree of spiral stability was recommended primarily from consideration of References 54 and 55. Reference 54 stated that "the maximum desired spiral stability appears to be a time to half amplitude of 10 seconds" and, based on closed-loop analysis, Reference 55 suggested that $T_{1/2}$ less than approximately 7 to 14 seconds would generally cause a degradation of pilot opinion. If the experimental in-flight data of Reference 53 (Figure 1) and Reference 54 (Figure 2) are examined, however, it can be seen that good pilot ratings are obtained for $T_{1/2} \approx 10$ seconds and that the flying qualities do not begin to degrade appreciably until $T_{1/2} \approx 5$ seconds.

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 Reference 56, a wings-leveling device was installed

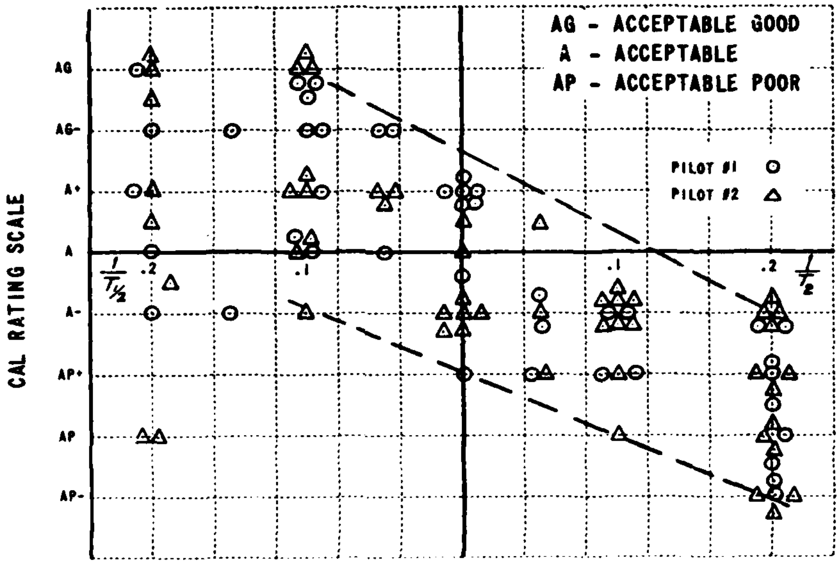


Figure 2 (3.5.1.1.2). Data for All Types of Flying -- Pilot Opinion Versus Spiral Damping (from Reference 54)

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 to not recommend a requirement on positive T_s or $T_{1/2}$ at this time, but more direct requirements on other factors associated with convergent spirals; that is, aileron forces in turns and roll maneuverability 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.

H. LESSONS LEARNED

This requirement is well established and no specific discussion is necessary.

3.5.1.1.3 Coupled roll-spiral oscillation

A. REASON FOR REQUIREMENT

Existence of a coupled roll-spiral oscillation (also called a "lateral phugoid") results in poor bank angle control and excessive lateral stability for maneuvering. This requirement disallows the existence of a coupled roll-spiral mode for all but very benign maneuvering flight conditions.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.1.4

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.1.1.3 Coupled roll-spiral oscillation. A coupled roll-spiral mode will be permitted provided it has the following characteristics: _____________________________.

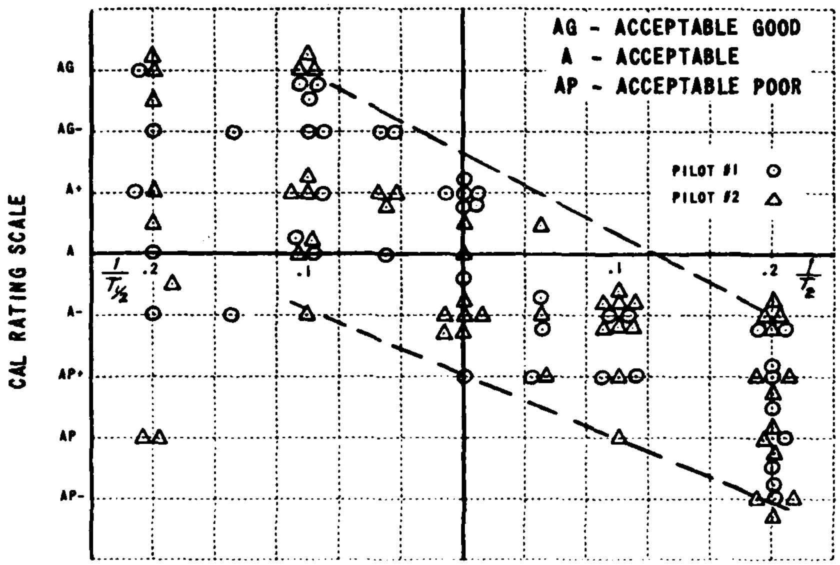
TABLE 1 (3.5.1.1.3)

RECOMMENDED MAXIMUM VALUES FOR ROLL-SPIRAL DAMPING COEFFICIENT, $\zeta_{RS}\omega_{RS}$

LEVEL	CATEGORY A	CATEGORIES B AND C**
1	*	0.5
2	*	0.3
3	*	0.15

*The aircraft shall not exhibit a coupled roll-spiral mode in Category A Flight Phases.

**The aircraft shall not exhibit a coupled roll-spiral oscillation in Category C Flight Phases requiring rapid turning maneuvers such as short approaches.



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C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

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LEVEL	CATEGORY A	CATEGORIES B AND C**
1	*	0.5
2	*	0.3
3	*	0.15

*The aircraft shall not exhibit a coupled roll-spiral mode in Category A Flight Phases.

**The aircraft shall not exhibit a coupled roll-spiral oscillation in Category C Flight Phases requiring rapid turning maneuvers such as short approaches.

D. RATIONALE BEHIND REQUIREMENT

The primary objections to a coupled roll-spiral mode are the lack of roll control effectiveness and high steady forces in turning flight (see Reference 59, page 138). Roll spiral coupling can arise from unusual values of L'_β, L'_p, N'_p, and N'_r, or from augmentation where bank angle feedback is employed (see Reference 66). 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 the experience was a V/STOL control system blending study conducted on a moving-base simulator (Reference 259). 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 Reference 59 comments, the pilots found that the lack of roll control authority and high steady forces in the turns were unacceptable.

In cases where the coupled roll spiral results from unusually large values of L_β, N'_p/L'_p, and N'_r, a low L'_p may result in controllability problems. This was the case for the M2-F2 lifting body (Reference 61).

The values in Table 1 were taken directly from MIL-F-8785C.

E. GUIDANCE FOR APPLICATION

Existence of a coupled roll-spiral mode should be allowed only during those flight conditions which do not require rapid bank angle maneuvering. Most control wheel steering (CWS) modes use roll rate command augmentation and hence do not have a coupled roll-spiral mode. However, it is conceivable that a bank angle command CWS mode would be proposed. In the event that a deviation from the requirement is requested, the flight tasks would be those involving rapid bank angle maneuvering.

F. DEMONSTRATION OF COMPLIANCE

Flight testing through the operational speed and altitude envelopes should reveal any potential problems. Analysis based upon wind tunnel estimates can predict potential areas where a lateral phugoid might exist; flight testing should be concentrated in these regions.

We do not expect any problem in the determination of equivalent system parameters for comparison with this requirement. 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.

G. SUPPORTING DATA

References 48, 49, and 60 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-body-type vehicles: large effective dihedral, low roll damping, and positive yaw acceleration due to roll rate.

The data of Reference 48 are for an in-flight simulation of a reentry vehicle using the USAF/Calspan variable-stability T-33. Figures 1, 2, and 3 present results of the simulation utilizing spiral descent and landing approach (Figure 1) and up-and-away flight (Figures 2 and 3) maneuvers. Four pilots evaluated the configurations, and inter-pilot variations in ratings were small for most cases. Figure 1 shows that in smooth air and slight proverse yaw due to ailerons, $(\omega_\phi/\omega_d)^2 = 1.344$, a lateral phugoid is acceptable but unsatisfactory (ratings are 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 3 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 Figures 2 and 3, ζ_d

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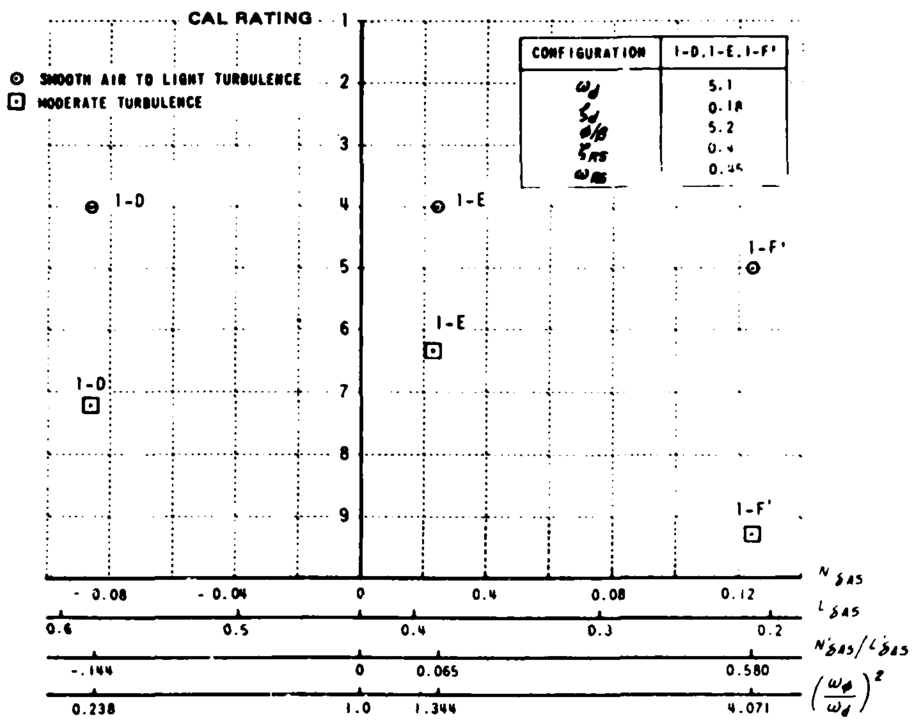

Figure 1 (3.5.1.1.3). Composite Pilot Ratings for Spiral Descent cf Simulated Reentry Vehicle (from Reference 48)

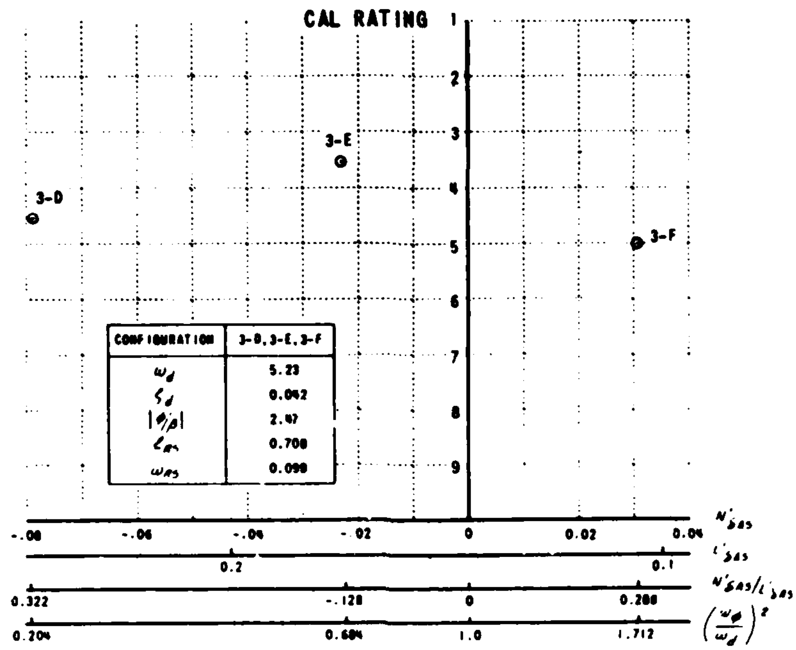

Figure 2 (3.5.1.1.3). Composite Pilot Ratings for Up-and-Away Flight; Moderate $|\phi/\beta|_d$ (from Reference 48)

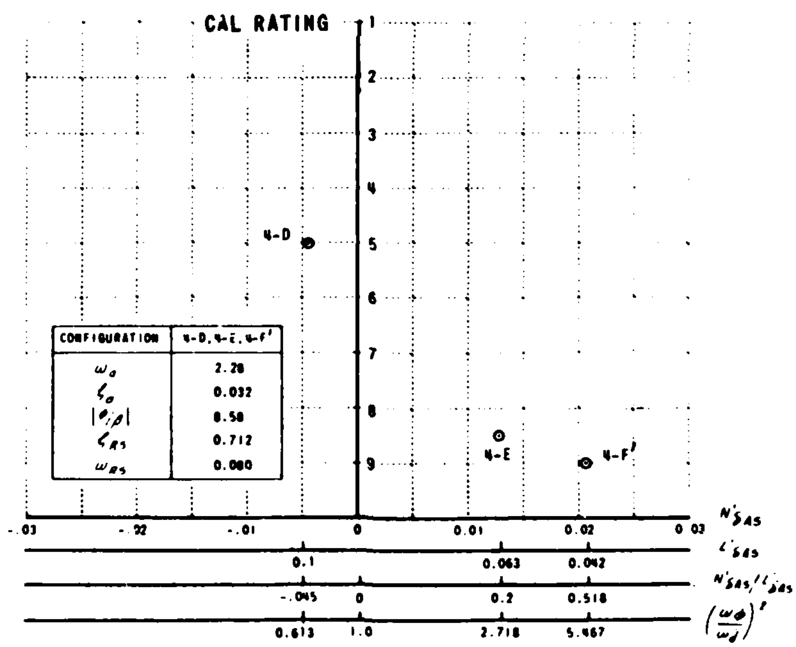

Figure 3 (3.5.1.1.3). Composite Pilot Ratings for Up-and-Away Flight; Large $|\phi/\beta|_d$ (from Reference 48) is Level 2 in value) for the tasks considered to be low-demand (Category B) in nature.

The ground simulation of Reference 60 involved cruise and low-speed conditions, including several ILS approaches. These also, with the exception of the ILS approaches, are Category-B-type maneuvers. However, the approach ratings were reported to correlate with the low-speed (general all-around flying) ratings, so these could be considered to be Category C data. Results are shown in Figure 4. The boundaries of Table 1 are shown for comparison. In general, though the dutch roll characteristics (ζ_d, ω_d, $|\phi/\beta|_d$) and adverse aileron yaw $[(\omega_\phi/\omega_d)^2]$ are varied, the data show definite trends with ζ_{RS} and ω_{RS}. They are also in agreement with the in-flight data of Figures 1-3, but they do not show strong support for the Table 1 boundaries. 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 percent to almost 80 percent. However, the lack of any pilot commentary or of detailed descriptions of the piloting tasks, somewhat reduces the credibility of the data. In addition, Reference 49 has data which disagrees entirely with both Reference 60 and Reference 48.

The ground simulation of Reference 49 shows a much more pessimistic view of the lateral phugoid (Figure 5). 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 covered climbing, diving, and level turns and both slow and rapid entries into 30 deg and 60 deg banks. Both smooth and simulated rough air were used. The open-loop task required that the pilots copy a standard IFR clearance.

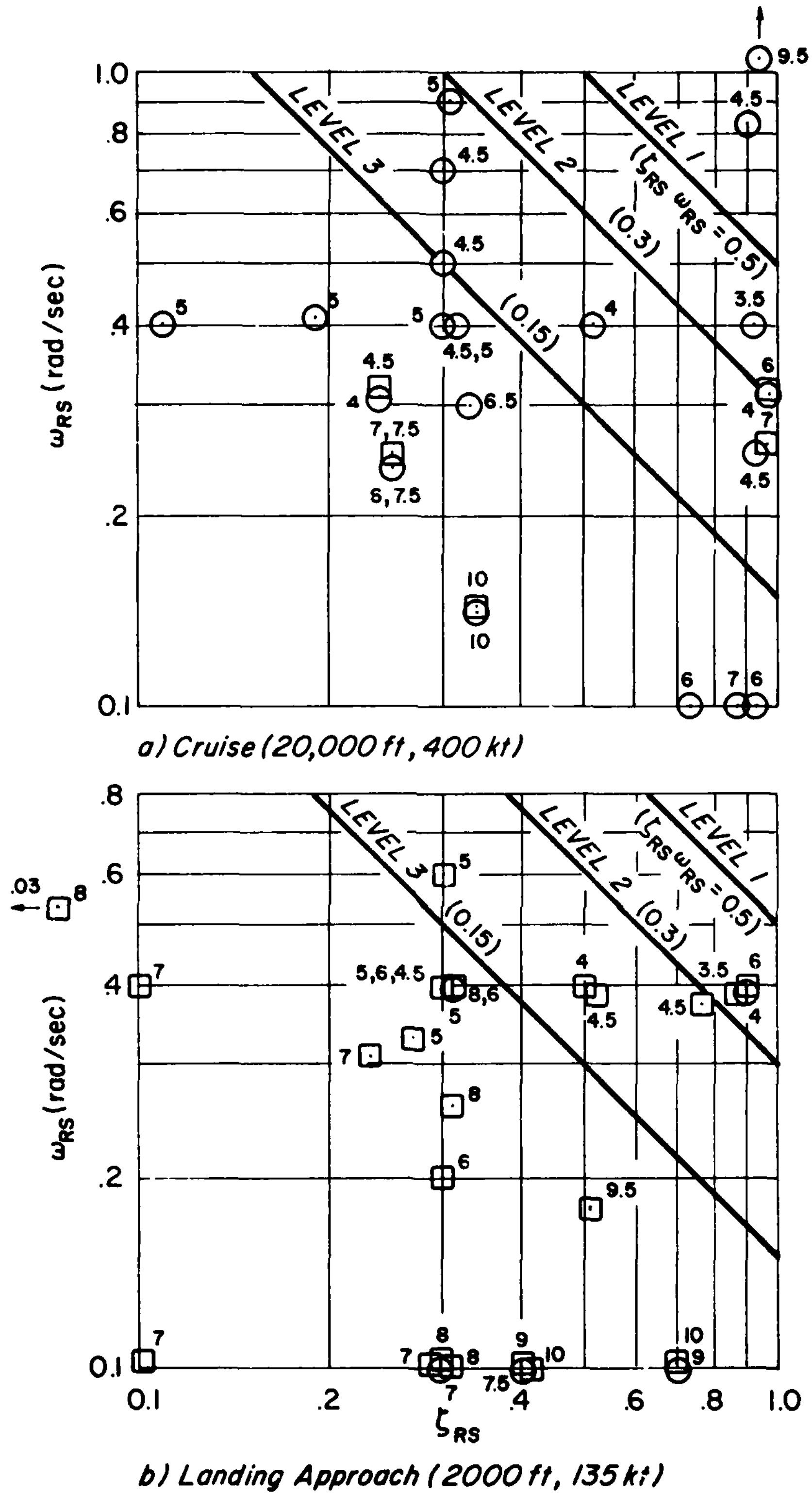

Figure 4 (3.5.1.1.3). Pilot Ratings for Ground Simulation of Reference 60 (Dutch Roll Characteristics Vary)

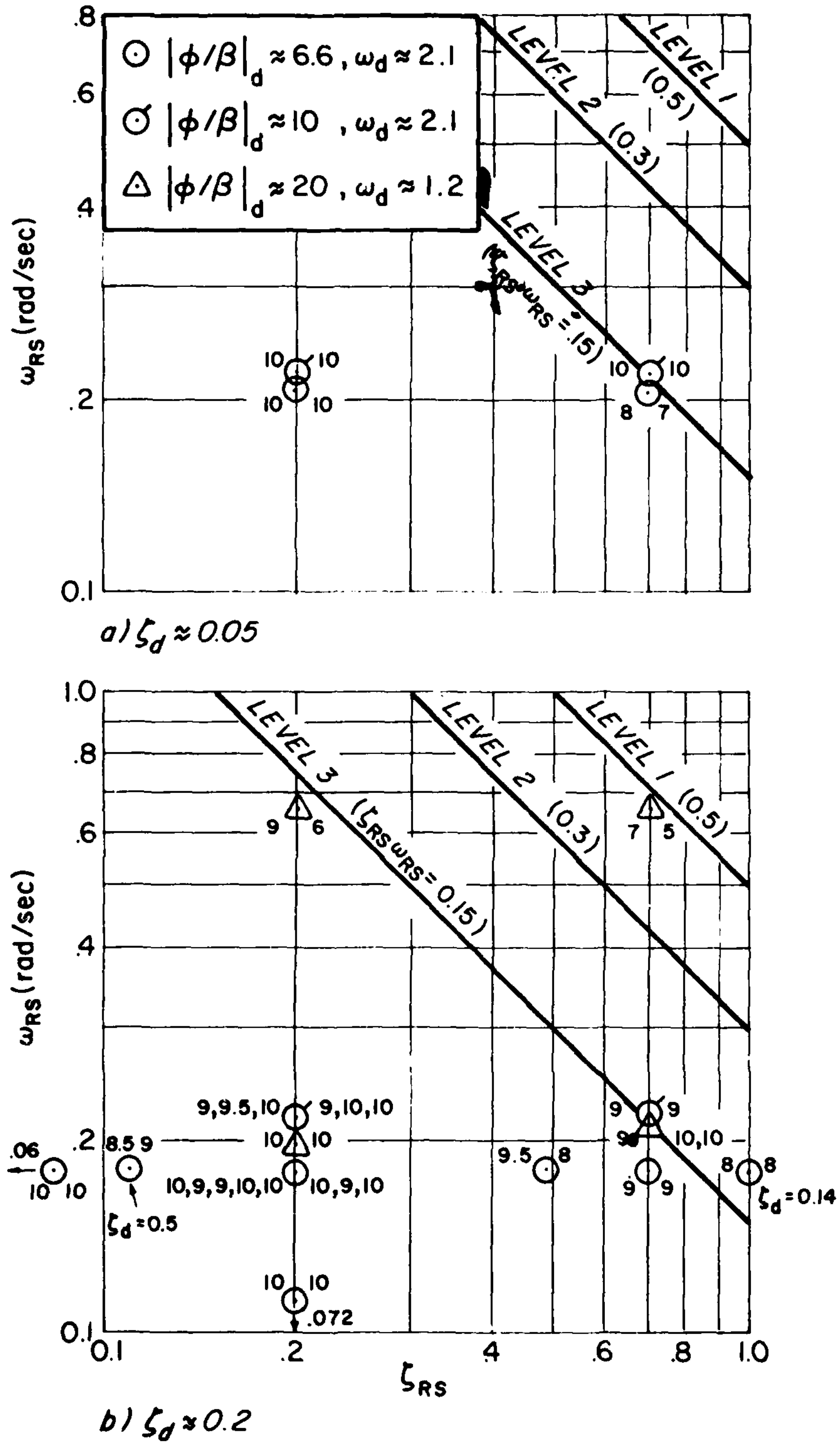

Figure 5 (3.5.1.1.3). Pilot Ratings for Ground Simulation of Reference 49 $[(\omega_\phi/\omega_d)^2 \approx 0.64 - 1.10]$

It is not clear why the data of Reference 49 differ so dramatically from both the ground simulation of Reference 60 and the flight data of Reference 48.* It is possible that the high ratio of $|\phi/\beta|_d$ (ranging from 6.1 to 26.5) caused the degradation; from Reference 49,

> The most obvious conclusion is that the complex roll-spiral mode configurations that were investigated represent poor to very bad tactical airplanes, primarily because of the lack of roll damping and the resultant "rolly" characteristics.

However, Reference 48 also contains some high $|\phi/\beta|_d$ cases (Figure 3), and for $(\omega_\phi/\omega_d)^2$ near 1 the average rating was 5 ($\zeta_{RS}\omega_{RS}$ = 0.057). The details of the simulated turbulence used in Reference 49 are not known; but it is possible that this had a major effect on the ratings (see Figure 1 and Reference 58).

H. LESSONS LEARNED

Experience with the M2-F2 lifting body (Reference 61) shows support for a strict requirement on the lateral phugoid, and illustrates the insidious nature of the lateral phugoid mode. Figure 6 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 occurs, through p and r feedbacks, but it is near in frequency and damping to the lateral phugoid. Low-frequency washout zeros 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 Reference 61 showed these oscillations to be due to the coupled roll-spiral mode. Time histories in Reference 61

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*The data of Reference 49 using real roll and spiral modes are also in disagreement with other such data, see Figure 16 of Paragraph 3.5.1.1.1, "Roll mode."

suggest that the M2-F2 became uncontrollable at $\alpha \approx -2$ deg; this coincides (Figure 6) with $\zeta_{RS}\omega_{RS} \approx 0$. Addition of a center fin (the M2-F3) improved primarily the yawing characteristics of the vehicle. As Figure 6 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 -- $\zeta > 1$ -- until $\alpha \approx 2$ deg). Flight tests of the M2-F3 (SAS ON) supported the prediction of good lateral flying qualities at negative angles of attack.

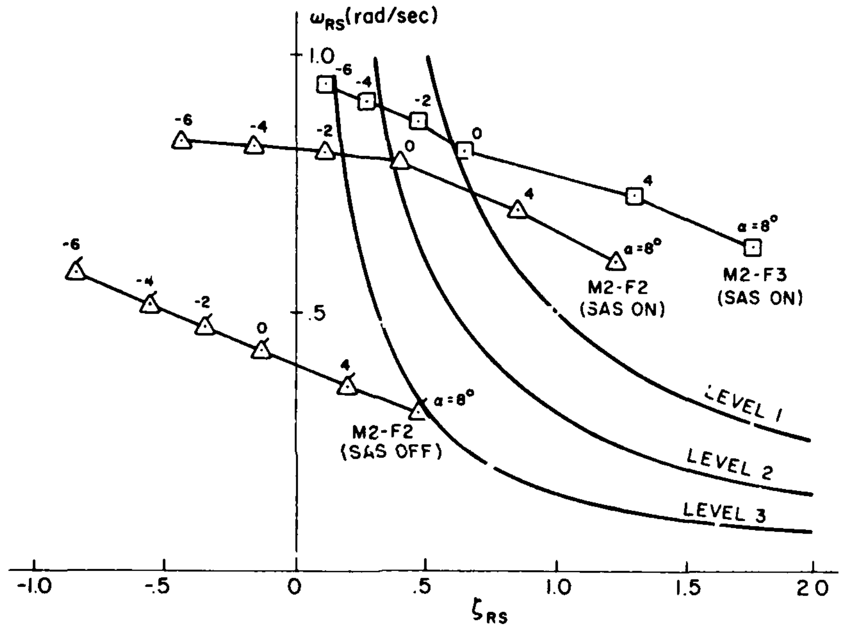


Figure 6 (3.5.1.1.3). Coupled Roll-Spiral Mode Characteristics for M2-F2 and M2-F3 Lifting Bodies (from Reference 61)

3.5.1.1.4 Roll rate oscillations

A. REASON FOR REQUIREMENT

This requirement is directed at precision of control of airplanes with moderate to high $|\phi/\beta|_d$ response ratios combined with marginally low dutch roll damping.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.2.2.1

C. STATEMENT OF REQUIREMENTS AND RECOMMENDED VALUES

3.5.1.1.4 Roll rate oscillations. The value of the parameter p_{osc}/p_{av} following a yaw-control-free step roll command shall be within the following limits: ____________. This requirement applies for step roll commands up to the magnitude that causes a 60 degree bank angle change in $1.7T_d$ seconds.

Configurations that meet the appropriate Category A dutch roll damping requirement (Paragraph 3.6.1.1) should be considered to meet this requirement as long as ω_ϕ/ω_d is within the following limits: __________.

Recommended limits for the parameter p_{osc}/p_{av} are specified by Figure 1.

D. RATIONALE BEHIND REQUIREMENT

This requirement was developed to regulate against unacceptable handling qualities due to large roll-sideslip coupling $|\phi/\beta|_d$ combined with a low dutch roll damping ratio ζ_d. 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 should be kept to an absolute minimum for all but the most undemanding tasks.

The existence of roll rate oscillations is directly traceable to the relative locations of the ω_ϕ and ω_d zeros in the p/F_{as} transfer function:

$$\frac{p}{F_{as}} = \frac{L_{F_{as}} s(s^2 + 2\zeta_\phi\omega_\phi s + \omega_\phi^2)e^{-\tau_{ep}s}}{(s + 1/T_s)(s + 1/T_R)[s^2 + 2\zeta_d\omega_d s + \omega_d^2]} \quad (1)$$

When the complex roots cancel ($\omega_\phi = \omega_d$ and $\zeta_\phi = \zeta_d$) the roll rate response is not contaminated by sideslip excursions in the dutch roll mode. When they do not cancel, the dutch roll contamination occurs primarily in the yaw axis 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 quite large and ζ_d is low. Note that for all Category A operations the required Level 1 value of ζ_d in Paragraph 3.6.1.1 effectively eliminates the need for this requirement. In the MIL-F-8785 specification series, approach and landing were considered to be low-gain, loose tracking tasks. However, experience has shown that this is not the case. Hence, the p_{osc}/p_{av} requirement is limited to cruise and some non-precision takeoff Flight Phases for all practical purposes.

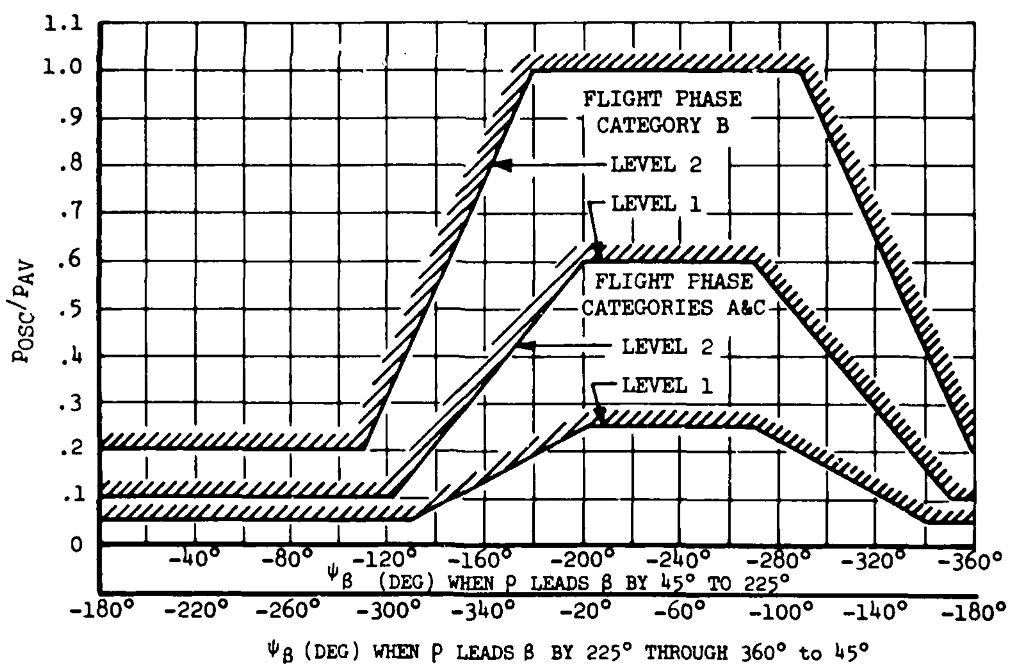
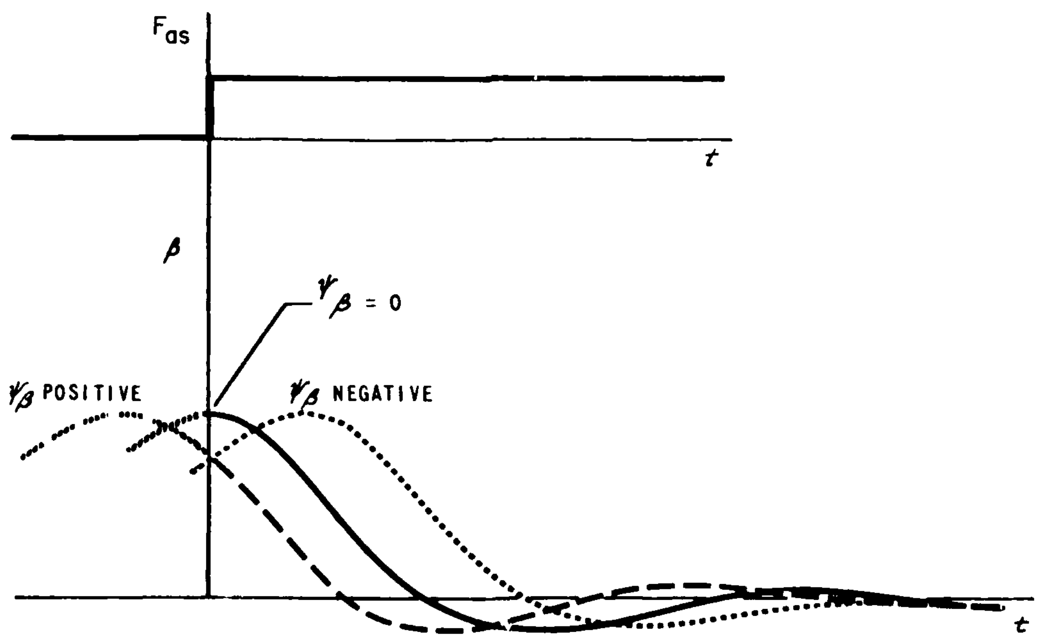


Figure 1 (3.5.1.1.4). Roll Rate Oscillation Limitations

An extensive description of the derivation of p_{osc}/p_{av} and ψ_β is given in Reference 11. Because of the decreased role of these parameters in the current requirements we have chosen to provide only enough background to utilize the Figure 1 boundaries. p_{osc}/p_{av} is a measure of the oscillatory component of roll rate to the average component of roll rate rollowing a rudder-pedals-free step lateral control input. Examples of measurements of p_{osc}/p_{av} are given under "Guidance for Application." The parameter ψ_β is shown in Reference 11 to be a measure of the relative location of the dutch roll pole and ω_ϕ zero (Equation 1). It defines the phasing of the dutch roll component of the sideslip response following a step lateral control input, i.e.,

$$\frac{\beta_d}{F_{as}} = C_d e^{-\zeta_d \omega_d t} \cos\left(\omega_d \sqrt{1 - \zeta_d^2\, t + \psi_\beta}\right) \tag{2}$$

This is illustrated graphically in the sketch below (from Reference 11).



The parameters p_{osc}/p_{av} and ψ_β have been used to specify the criterion as a function of Flight Phase Category and Level (Figure 1). It should be noted that Figure 1 has two ψ_β scales, one for positive dihedral (p leads β by 45 deg to 225 deg) and the other for negative dihedral (p leads β by 225 deg through 360 deg to 45 deg). Dihedral is defined by the parameter $\measuredangle p/\beta$, since dihedral as currently used in flying qualities work seems to be an ambiguous and ill-defined parameter. In this context, "positive dihedral" normally means negative $L'_\beta + Y_v L'_r \approx L'_\beta$. Note that

$$L'_\beta \equiv \frac{L_\beta + (I_{xz}/I_x)N_\beta}{1 - (I_{xz}^2/I_x I_z)}$$

It should also be noted that the value, or even the sign, of L'_β cannot always be determined from steady rudder-pedal-induced sideslips. Not only are product of inertia effects absent in steady sideslips, but also the control deflections are affected by control cross-coupling derivatives. On the other hand, $\measuredangle p/\beta$ generally is a good discriminator of the sign of dihedral.

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 its physical implications and significance to the piloting of an airplane. Very simply, ψ_β can be considered as an indicator of those airplane closed-loop stability characteristics that are related to the lateral-directional coupling derivatives. From Figure 1 it can be seen that the ratio of roll rate oscillation to steady-state roll rate can be much greater for some values of ψ_β than for others. Specifically, the specified values of p_{osc}/p_{av} for $0^\circ > \psi_\beta > -90^\circ$ are far more stringent than for $-180^\circ > \psi_\beta > -270^\circ$. There are at least two reasons why this is so:

a) Differences in closed-loop stability.

b) Differences in average roll rate.

From the root locus analysis in Figure 2a it can be shown that when the zero of the p/δ_{as} transfer function lies in the lower quadrant with respect to the dutch roll pole ($-180^o > \psi_\beta > -270^o$), 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 ($0^o > \psi_\beta > -90^o$), the closed-loop damping decreases when the pilot applies aileron inputs proportional to bank angle error (Figure 2b). In this case a pilot's tolerance of p_{osc}/p_{av} tends to decrease. Finally, when ζ_d becomes large the effect of the pole-zero location is diminished (Figure 2c), i.e., the variation in damping due to ω_ϕ/ω_d effect is small relative to the augmented damping. The connection between pole/zero location and the p_{osc}/p_{av} boundaries is shown in Figure 3, where the Level 1 and Level 2 boundaries in Figure 1 are mapped into ω_ϕ zero locations for several values of ω_d and ζ_d. Note that when ζ_d meets the Level 1 requirement ($\zeta_d > 0.19$) the acceptable region for ω_ϕ is very large in the region to the left of and below ω_d. However, there is always a low tolerance for $\omega_\phi > \omega_d$ because the closed-loop damping decreases. There is still a relatively tight limit on $\omega_\phi > \omega_d$ for a ζ_d of 0.25. This of course reflects the decrease in damping that occurs (note that $\zeta_d = 0.25$ is not much greater than the Level 1 limit of 0.19, see Figure 2c). 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 as ζ_d and ω_d increase.

An alternative method of specifying roll rate oscillations, recommended by Calspan (Reference 59), was considered. The proposed revision would involve extracting the effect of the spiral mode, T_s, from the roll rate response. This would get rid of the present significant effect T_s can have on p_{osc}/p_{av}, as shown in Figure 4. A new parameter, $\hat{p}_{osc}/\hat{p}_1$, would be used, where the hat (^) represents the spiral-less roll rate response. Then $\hat{p}_{osc}/\hat{p}_1 = (\hat{p}_1 + \hat{p}_3 - 2\hat{p}_2)/2\hat{p}_1$. Reference 59 also recommended that the parameter ψ_β be replaced with ψ_p, i.e., measure ψ from the roll rate response.

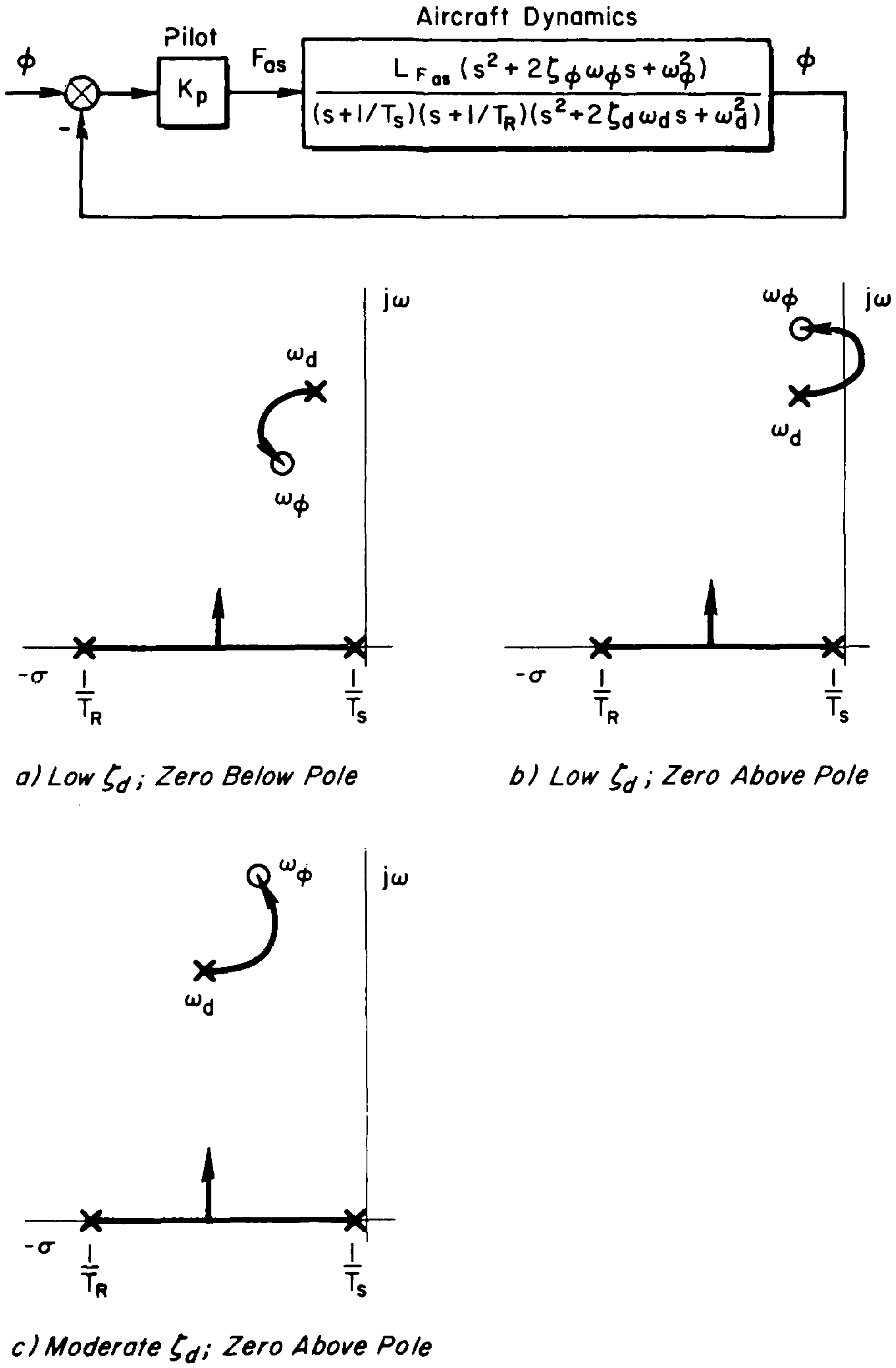


Figure 2 (3.5.1.1.4). Effect of Relative Pole/Zero Location On Piloted Control of Bank Angle

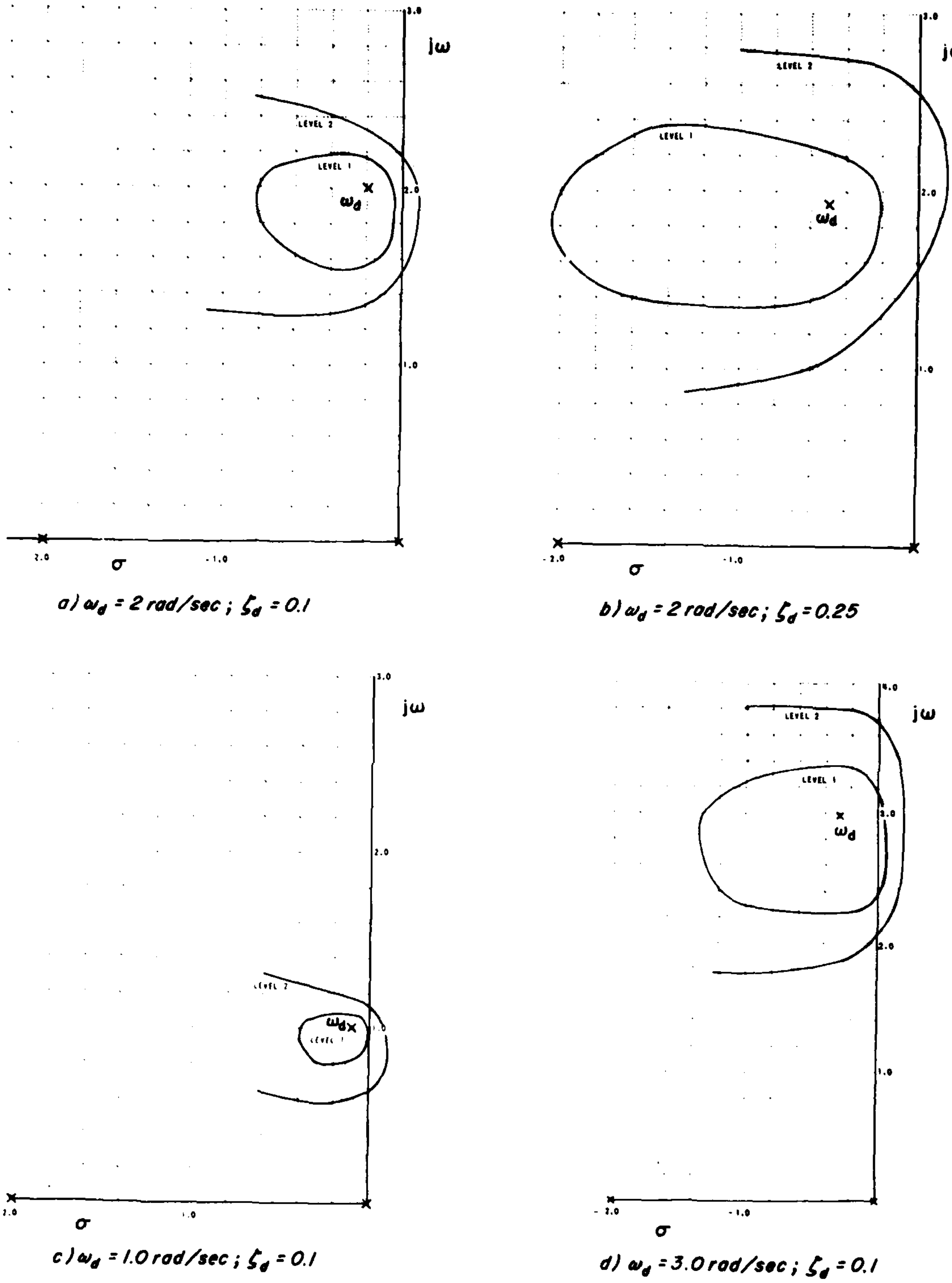


Figure 3 (3.5.1.1.4). Locations of ω_ϕ Zero Corresponding to Category A and C and Level 1 and 2 Boundaries in Figure 1 (T_R = 0.5 sec, T_s = ∞) (From Reference 11)

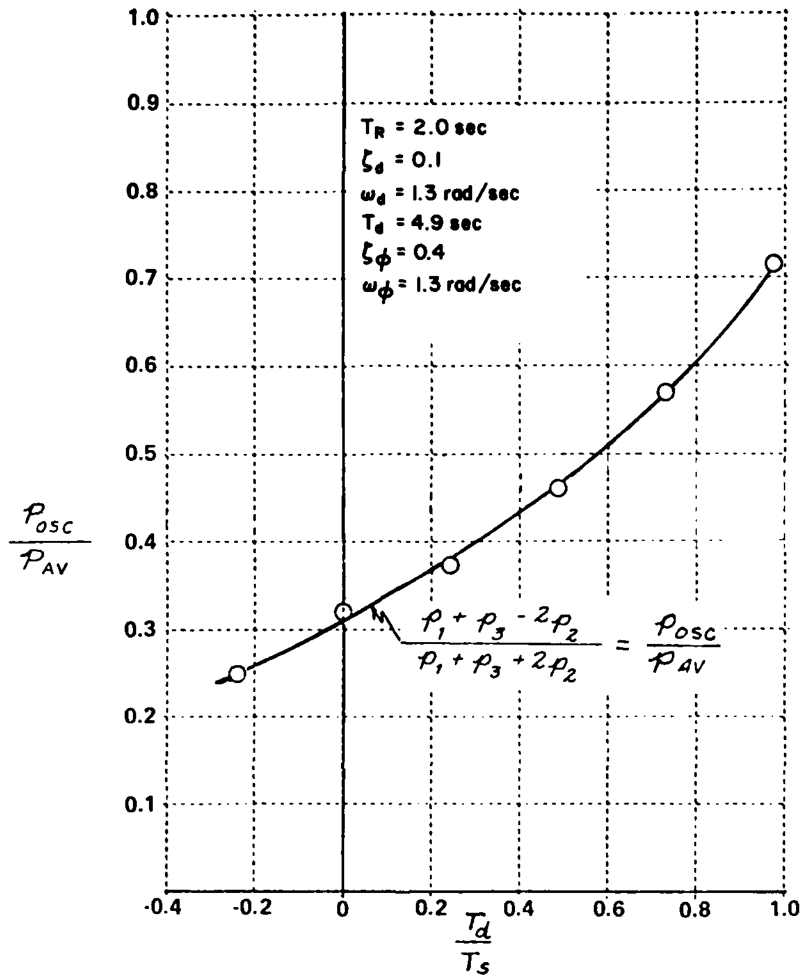


Figure 4 (3.5.1.1.4). p_{osc}/p_{av} as a Function of the Ratio of Dutch Roll Period and Spiral Root Time Constant (from Reference 59)

Data comparisons with the Reference 59 $\hat{p}_{osc}/\hat{p}_1$ vs. ψ_p and p_{osc}/p_{av} vs. ψ_β do not justify any change at this time. For example, the Category A data of Reference 46 show only a 40 percent correlation with the Reference 59 limits, and 61 percent correlation with Figure 1; the Category C data of Reference 69 have exactly the same correlation with both criteria, 72 percent. Overall, only about half the data used in Reference 59 to support $\hat{p}_{osc}/\hat{p}_1$ vs. ψ_β agree with the proposed requirement. This was not felt to be sufficient to justify a change.

E. GUIDANCE FOR APPLICATION

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 a number of guidelines and qualifications to be considered before using the requirement.

- The roll, spiral and dutch roll mode requirements (3.5.1.1.1, 3.5.1.1.2, and 3.6.1.1) should first be met. If T_s is very small, the requirement may result in misleading values of p_{osc}/p_{av} (see, e.g., Figure 4).

- For airplanes with very small L'_β amd very large L'_r (Reference 50), $\angle p/\beta|_d$ can be between 180 deg and 270 deg. As shown in Reference 11, 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.*

- 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.

- The requirement is of most value when ζ_d is near the Level 1 boundary (0.1-0.2).

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*The test results of Reference 50 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 p/β are still of interest.

F. DEMONSTRATION OF COMPLIANCE

Flight testing to demonstrate compliance with this requirement may, at the request of the procuring activity, be limited 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.

The parameters p_{osc}/p_{av} and ψ_β are defined in Section 6.2.6.

G. SUPPORTING DATA

The most complete set of Category A supporting data (Figures 5 and 6), from flight tests of Reference 46 using the variable-stability T-33, show conflicting results. The data of Figure 5, for moderate $|\phi/\beta|_d$ ratios, agree quite well with the boundaries. Likewise, the Figure 6

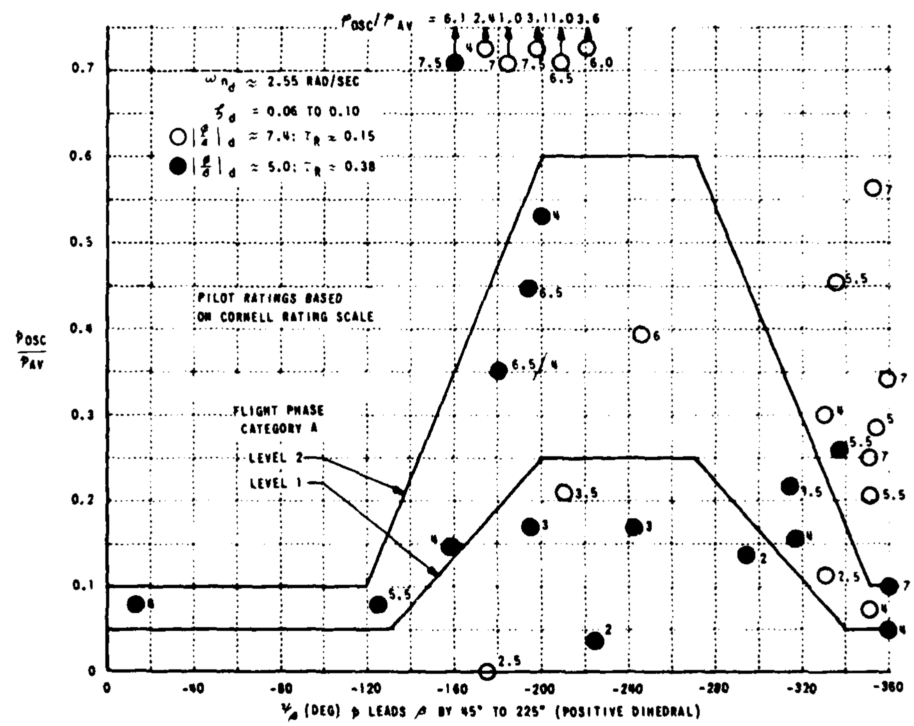


Figure 5 (3.5.1.1.4). Flight Phase Category A Data, Moderate $|\phi/\beta|_d$ (from Reference 46)

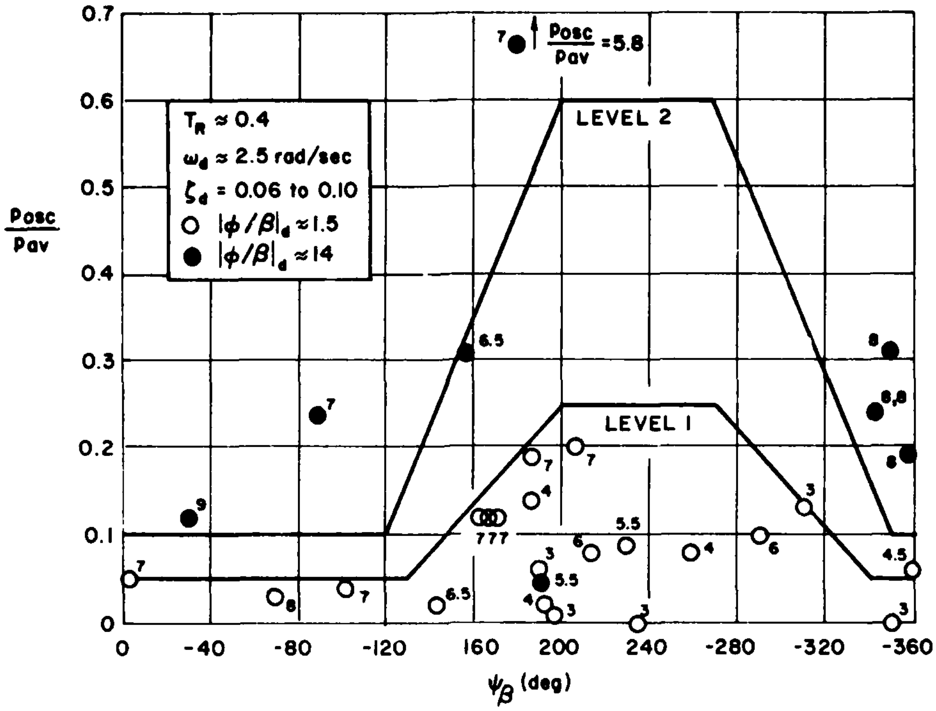


Figure 6 (3.5.1.1.4). Flight Phase Category A Data, Large and Small $|\phi/\beta|_d$ (from Reference 46)

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. (Note that the dutch roll damping requirements (3.6.1.1) account for these large $|\phi/\beta|_d$ cases already, predicting them to be worse than Level 3.) However, the cases for low $|\phi/\beta|_d$ (1.5) in Figure 6 show extremely poor correlation. The reasons for this have not been resolved, though 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 very poor.

The Category B data (Figures 7 and 8) show good correlation, but there are really only about ten data points with which to evaluate the Levels 2 and 3 regions (that is, data for which p_{osc}/p_{av} is large). Likewise, the Category C Levels 2 and 3 boundary (Figure 9) is not well defined by the data.

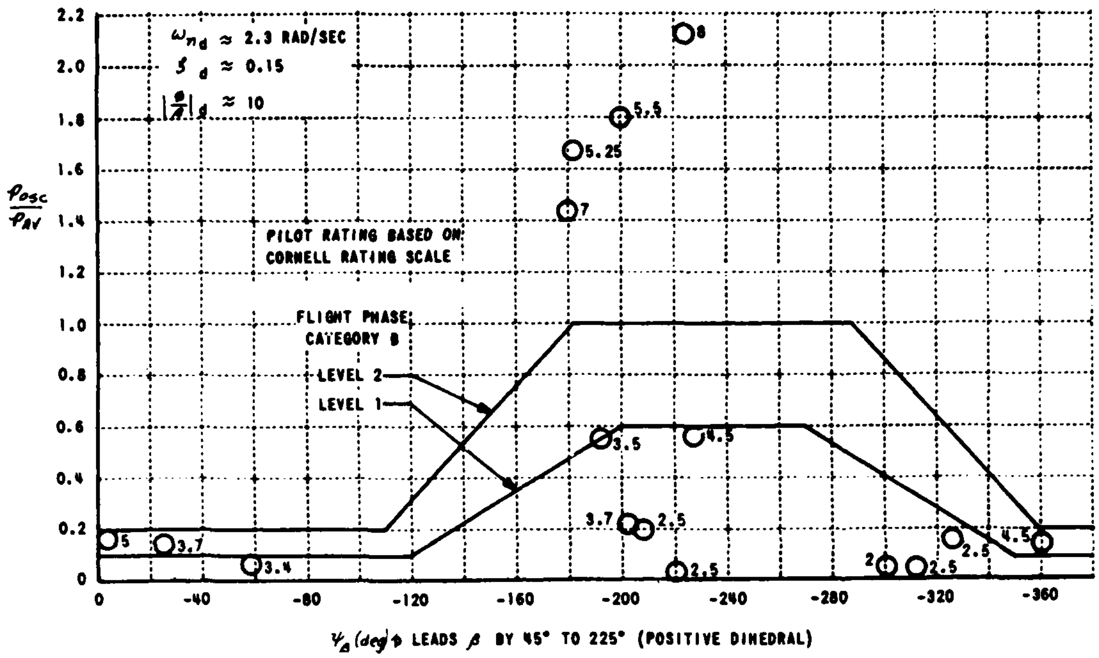


Figure 7 (3.5.1.1.4). Flight Phase Category B Data
(from Reference 48)

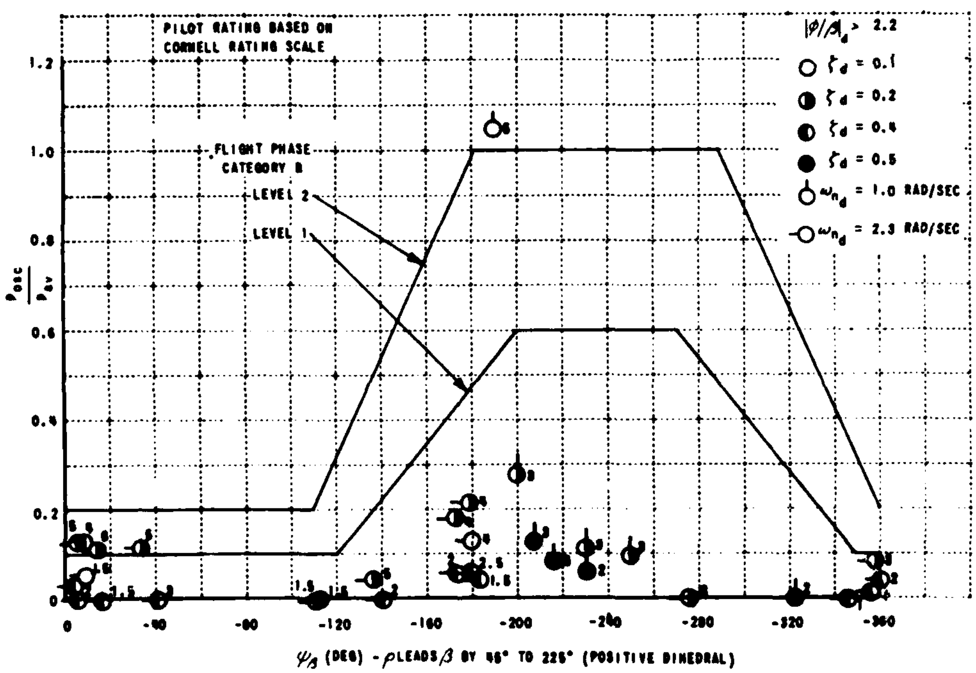


Figure 8 (3.5.1.1.4). Flight Phase Category B Data
(from Reference 39)

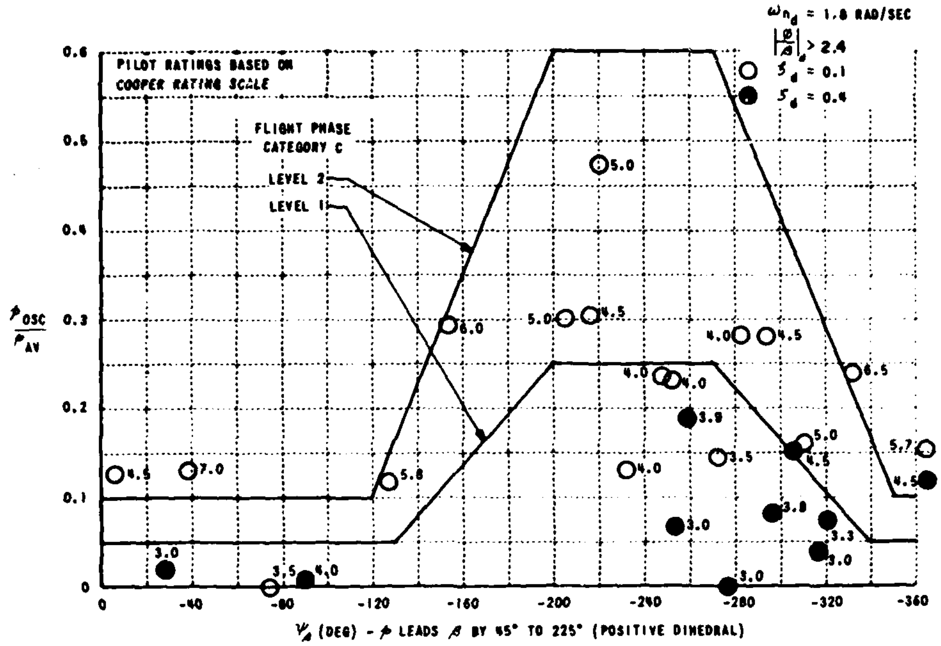


Figure 9 (3.5.1.1.4). Flight Phase Category C Data
(from Reference 44)

Similar data (Reference 69) show support for the requirements (Figure 10). Again, this is Category C data, though the test programs of References 44 and 69 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 Reference 69, simulated crosswinds) which would be expected to increase pilot workload.

The thorough test matrix of Reference 69 produced an abundance of data with which to draw some guidance for applying p_{osc}/p_{av} (given in "Guidance for Application"):

- Paragraph 3.5.1.1.4 need not be applied if $|\phi/\beta|_d$ is small (from Figure 10, $|\phi/\beta|_d < 1.5$ generally produces good ratings and low p_{osc}/p_{av}; though the ratings of Figure 6 for $|\phi/\beta|_d \approx 1.5$ are poor, p_{osc}/p_{av} is low).

- The criterion is most useful when either ζ_d, ω_d, $\zeta_d\omega_d$, or T_R is near the Level 1/2 limits. For example, there are 21 cases in Figure 10 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, ψ_β = -180 deg, PR = 8).

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. Reference 70 did provide some data, however, which are presented in Figure 11 for comparison with the roll rate oscillation requirement. The program of Reference 70 investigated lateral-directional instabilities relating to the X-15. In the course

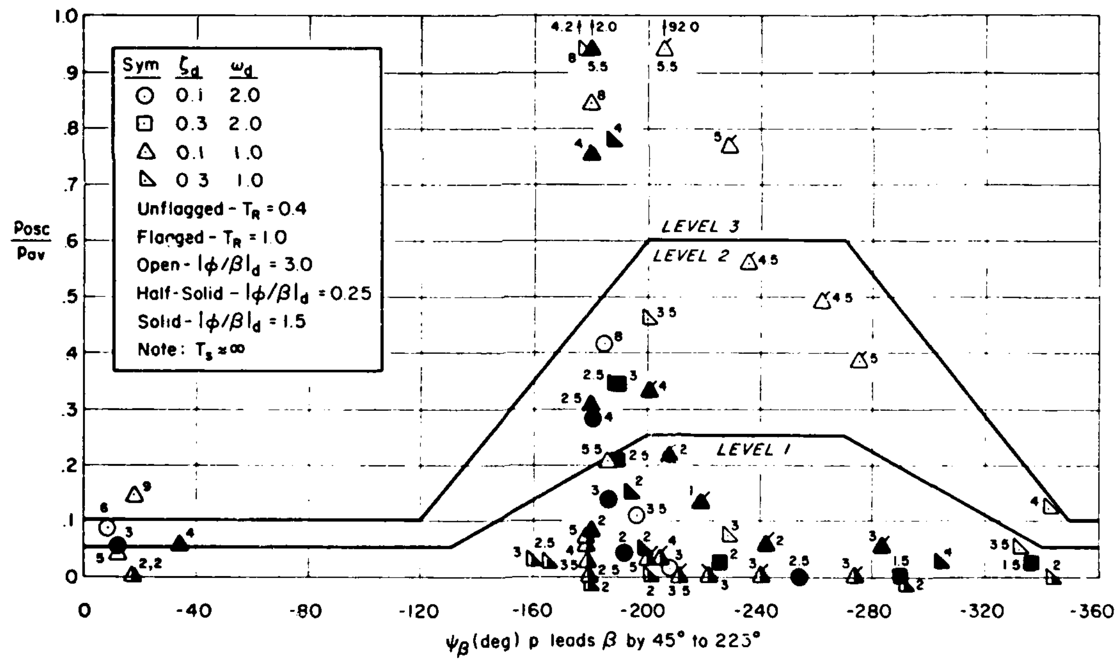


Figure 10 (3.5.1.1.4). Category C Data (Approach and Wave-Off); Cooper-Harper Pilot Ratings (from Reference 69)

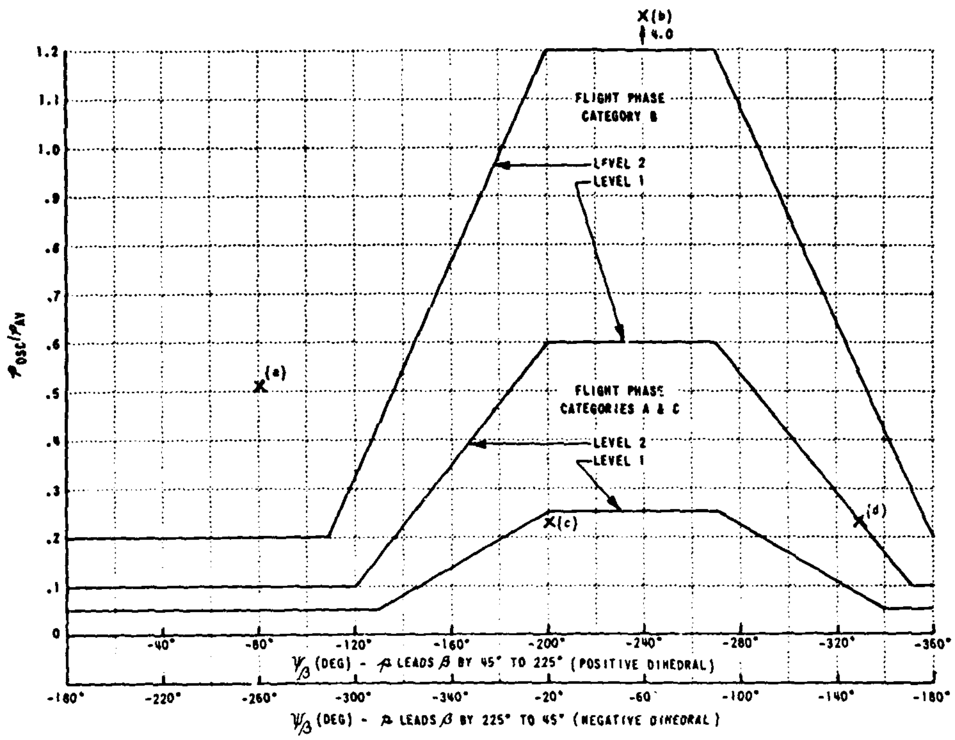


Figure 11 (3.5.1.1.4). Positive and Negative Dihedral
Data of Reference 70

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.

These configurations, which all had very light dutch roll damping and large $|\phi/\beta|_d$ response ratios, are plotted in Figure 11. The parameters p_{osc}/p_{av}, ψ_β, and $\measuredangle p/\beta$ were obtained from time histories of the responses to step aileron inputs. Configuration a, which falls well outside the Level 2 boundary of Figure 11, 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 11, 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 falls in the "good" area of Figure 11, 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 acceptability on Figure 11, was uncontrollable because of pilot-induced 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 11. 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 11 where the amount of allowable roll rate oscillation changes rapidly with ψ_β would indicate that the flying qualities of the configuration are sensitive to small changes in ψ_β. For example, if ψ_β were only 30 degrees greater (or if the peaks on the time histories presented differed by only 0.2 seconds from those of the configuration flown), the roll-sideslip coupling characteristics as indicated by Figure 11 could be completely compatible with the pilot comments.

H. LESSONS LEARNED

As the following table reflects, correspondence with users of MIL-F-8785B and -8785C shows that the requirements for roll rate and bank angle oscillations (Paragraphs 3.3.2.2, 3.3.2.2.1, and 3.3.2.3 of those specifications) have been generally ignored for current airplanes. This is in part the reason for deleting the requirements for roll rate oscillations and bank angle oscillations, 3.3.2.2 and 3.3.2.3 of MIL-F-8785C, and retaining only the single criterion, p_{osc}/p_{av}. (Paragraph numbers in the table refer to MIL-F-8785B.)

ASD COMMENTS ON MIL-F-8785B PARA. 3.3.2, LATERAL-DIRECTIONAL DYNAMIC RESPONSE CHARACTEISTICS

F-16: Paragraphs 3.3.2.1, 3.3.2.2, 3.3.2.2.1, and 3.3.2.3 were deleted in F-16 spec. 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-26, C-141: Paragraphs 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 ($\zeta_d\omega_d$) is set at sufficient value; these paragraphs add little to evaluation of F-15

AMST, B-1: Paragraph 3.3.2.2 values seem too low based on DC-10 and B-1 data; paragraphs 3.3.2.2.1 and 3.3.2.3 are redundant and only 3.3.2.3 should be retained.

Paragraph 3.3.2.2 of MIL-F-8785C set limits on roll at the first minimum following the first peak in response to a step roll control input. No data have been found that show this to give results different from p_{osc}/p_{av}. In fact, they are directly related, since (from Reference 11),

> The numerical values of the roll rates 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 airplanes 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 airplanes with "proverse yaw."

Likewise, a requirement based on ϕ_{osc}/ϕ_{av} and ψ_β (3.3.2.3 of MIL-F-8785C) would be expected to give results similar to p_{osc}/p_{av}. For these reasons, only the p_{osc}/p_{av} requirement of MIL-F-8785C has been retained.

3.5.1.1.5 Time delay

A. REASON FOR REQUIREMENT

This requirement is intended tc insure that the combined delay contributions of prefilters, SAS, servos, etc., do not degrade roll control.

B. RELATED MIL-F-8785C REQUIREMENT

3.5.3.

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.1.1.5 Time delay. The value of the equivalent time delay, τ_{e_p}, shall be no greater than the following: ________________.

Recommended values of τ_{e_p}, defined in 3.5.1.1.1, are given in Table 1.

TABLE 1 (3.5.1.1.5)

RECOMMENDED ALLOWABLE EQUIVALENT DELAY

LEVEL	ALLOWABLE DELAY (sec)
1	0.10
2	0.20
3	0.25

D. RATIONALE BEHIND REQUIREMENT

Based upon extensive research into the effect of time delays in the pitch axis, the important contribution of delays is well known. This requirement extends to the roll axis τ_e limits imposed on the pitch axis in Para. 3.2.1.1. For lack of substantive data, the limits remain unchanged from MIL-F-8785C. As data become available, these limits should be evaluated and adjusted if necessary.

E. GUIDANCE FOR APPLICATION

Paragraph 3.5.1.1.1, which sets roll mode limits, discusses the use of equivalent system matching of the roll rate response to roll controller to extract T_s, T_R, and τ_{e_p}. That section should be consulted for more information.

F. DEMONSTRATION OF COMPLIANCE

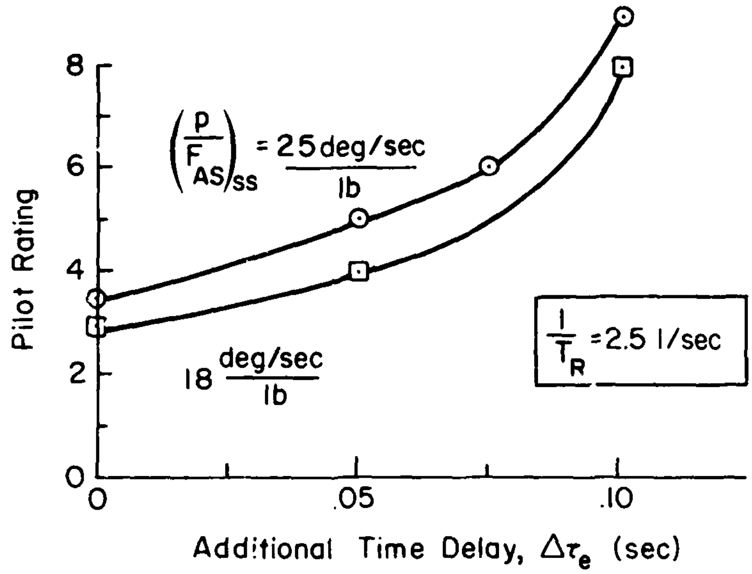
Appropriate values of τ_{e_p} will require equivalent system matching, as discussed above.

G. SUPPORTING DATA

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 3.2.1.1). This result is seen to extend to the lateral axis based upon the LATHOS study of Reference 258 (Figure 1). 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. Hence there is some evidence, though not enough to support a change in the requirement, that effective time delay may be more critical in the lateral than in the longitudinal axis.

H. LESSONS LEARNED

None available.



Note: $\Delta\tau_e$ represents the additional time delay above the basic NT-33 control system value of approximately .05 sec

Figure 1 (3.5.1.1.5). Effect of Time Delay, LATHOS Data (Reference 258)

3.5.2 Pilot-Induced Roll Oscillations

A. REASON FOR REQUIREMENT

This is simply a statement to expressly forbid neutral or unstable closed-loop oscillations in roll. The need for such a requirement is self-evident.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.3

C. STATEMENT OF REQUIREMENT

3.5.2 Pilot-Induced Roll Oscillations. There shall be no tendency for sustained or uncontrollable roll oscillations resulting from efforts of the pilot to control the airplane.

D. RATIONALE BEHIND REQUIREMENT

There is an obvious need for some requirement to prevent the development of pilot-induced oscillations (PIOs). 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 lateral PIO.

E. GUIDANCE FOR APPLICATION

This requirement should 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.

F. DEMONSTRATION OF COMPLIANCE

The existence of a 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, or terrain following may reveal PIO proneness.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

See 3.2.2.2 for discussion of applicable considerations and data, in that case directed at longitudinal PIOs in general. The occurrence of lateral PIOs has been less frequent, but there are one or two known cases and causes. For example, the M2-F2 lifting body (Reference 61) encountered several divergent PIOs during flight testing. The primary cause was found to be the coupled roll subsidence/spiral mode (see Lessons Learned for Para. 3.5.1.1.3).

A second cause of observed lateral PIO tendencies is the $\omega_\phi/\omega_d > 1$ effect noted and explained in Figure 2b (3.5.1.1.4) and also in Reference 225. Another less prevalent cause is associated with control-surface rate saturation. In this case the pilot tries to apply lateral control at a rate greater than the maximum surface rate, thereby getting out of phase. The quantitative aspects of such rate-limiting are given in the appendix of Reference 225 and involve gain and phase decrements that are functions of the ratio of commanded to saturation rate.

PIOs on recent aircraft have been related to roll responses which are both too low (F-18) and too high (YF-16). These cases are discussed in Paras. 3.5.9.1 and 3.5.10.3.

3.5.3 Residual Roll Oscillations

A. REASON FOR REQUIREMENT

This requirement is intended to prevent limit cycles in the control system or structural oscillations which might interfere with mission performance.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.1.1

C. STATEMENT OF REQUIREMENT

3.5.3 Residual Roll Oscillations. Any sustained residual oscillations in calm air shall not interfere with the pilot's ability to perform the tasks required in service use of the airplane.

D. RATIONALE BEHIND REQUIREMENT

The roll-axis requirements of 3.5.1 and 3.5.2 should prevent open-loop (aerodynamically induced) and closed-loop (pilot-induced) oscillations. This requirements sets limits on oscillations from other sources. Its intent is to recognize thresholds below which the pilot would be insensitive to roll oscillations.

E. GUIDANCE FOR APPLICATION

None required.

F. DEMONSTRATION OF COMPLIANCE

Flight testing for other roll-axis requirements should reveal any problems with residual oscillations.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.5.4 Linearity of Roll Response to Roll Controller

A. REASON FOR REQUIREMENT

Nonlinear responses to control inputs can result in poor handling qualities. This requirement is intended to disallow levels of nonlinear response that interfere with precision control of bank angle.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.4.4

C. STATEMENT OF REQUIREMENT

3.5.4 Linearity of Roll Response to Roll Controller. There shall be no objectionable nonlinearities in the variation of rolling response with roll control deflection or force. Sensitivity or sluggishness in response to small control deflections or force shall be avoided.

D. RATIONALE BEHIND REQUIREMENT

The requirement is directed at precision of control. Objectionable nonlinearities can be those due to excessive friction, detents, nonlinear force gradients, nonlinear $C_n(\delta_{as})$, or $C_\ell(\delta_{as})$, spoiler lag, etc.

E. GUIDANCE FOR APPLICATION

It has not been possible to specify values for tolerable levels of nonlinearities, so reliance must be placed on qualitative pilot evaluations.

F. DEMONSTRATION OF COMPLIANCE

Tests using 1/4, 1/2, 3/4, and full aileron are commonly used to demonstrate compliance. Such tests can also be used to help determine k for use in 3.6.2.1.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

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 necessary. This is discussed in Paragraph 3.5.10.3. As the treatise there would suggest, "linearity" is not the best word to use, since "no objectionable nonlinearities" implies that a linear roll rate response to stick input (p/F_{as} or p/δ_{as}) is desirable. Figure 2 (3.5.10.3) should be consulted for additional guidance.

3.5.5 Lateral Acceleration at Pilot Station

A. REASON FOR REQUIREMENT

In cases where the pilot location is well forward of the center of gravity or well above the roll axis, coordinated turns ($\beta \doteq 0$) are accompanied by large lateral accelerations at the cockpit. This requirements is included to limit such accelerations to acceptable levels.

B. RELATED MIL-F-8785C REQUIREMENT

None.

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.5 Lateral Acceleration at Pilot Station. The ratio of maximum lateral acceleration at the pilot station to maximum roll rate shall not exceed __________ for the first 2-1/2 seconds following a step roll control input.

Recommended values:

Level	${n_y}_{pilot_{max}}/p_{max}$ (g/deg/sec)
1	0.012
2	0.035
3	0.058

D. RATIONALE BEHIND REQUIREMENT

Concern over lateral acceleration is primarily due to ride qualities although in some cases aircraft control can be affected due to arm/bobweight effects.

A criterion based on the ratio ${n_y}_{pilot_{max}}/p_{max}$ includes in it the recognition that pilot acceptance of high accelerations is a function of aircraft rolling performance. Such a criterion was proposed by Chalk in

Reference 104 for large aircraft, and the recommended values are based on flight results with the Total In-Flight Simulator (TIFS) (see "Supporting Data"). Some refinement of the limits may be necessary for Class I, II, and IV aircraft.

E. GUIDANCE FOR APPLICATION

Due to the tentative nature of this requirement, it should be applied primarily as a guideline until more data can be obtained.

F. DEMONSTRATION OF COMPLIANCE

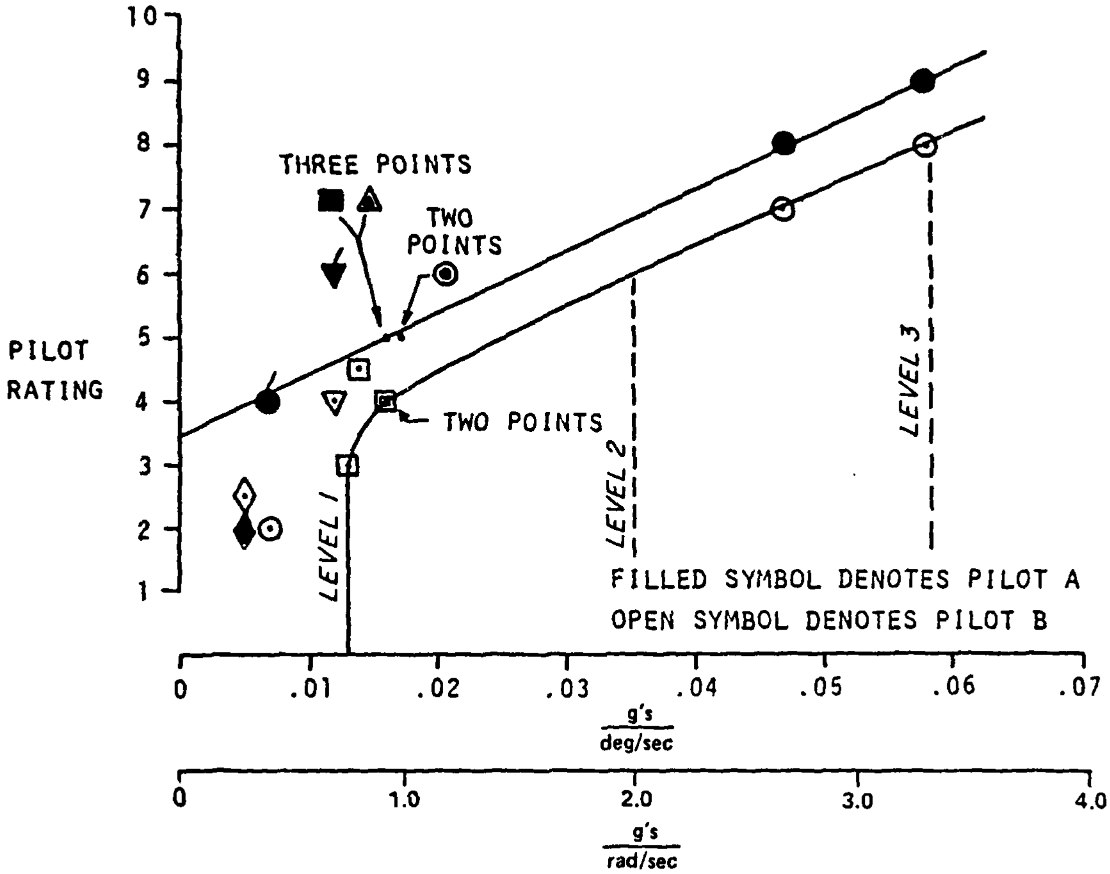
Both large and small step roll control inputs should be used, and $n_{y_{pilot_{max}}}$ and p_{max} measured within the first 2.5 sec of application of the control.

G. SUPPORTING DATA

The criterion was derived in Reference 104 as a proposed flying qualities requirement for Supersonic Cruise Research (SCR) aircraft. Figure 1 shows TIFS data compared with Cooper-Harper pilot ratings. Correlation is quite good, though more data should be gathered, especially for other Classes of aircraft.

H. LESSONS LEARNED

Pilots of some large aircraft, and of fighters at very high angles of attack, have complained of excessive lateral accelerations at the cockpit. When dealing with aircraft where the cockpit location is either well forward of the center of gravity or well above the roll axis, designing the rudder augmentor to coordinate the turns ($\beta = 0$) can produce unacceptable lateral accelerations at the pilot stations.



$$\left.\frac{n_{y_{pilot_{max}}}}{p_{max}}\right| \begin{array}{l}\text{Step Input} \\ t \leq 2.5 \text{ sec}\end{array}$$

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 1 (3.5.5). Lateral Acceleration Criterion Versus Pilot Rating from Reference 104

3.5.6 Roll Response to Yaw Controller

A. REASON FOR REQUIREMENT

This requirement is included to insure that the bank angle and roll control in straight, steady sideslips is conventional, in accordance with pilot experience.

B. RELATED MIL-F-8785C REQUIREMENTS

3.3.6, 3.3.6.2, 3.3.6.3, 3.3.6.3.1, 3.3.6.3.2

C. STATEMENT OF REQUIREMENT

3.5.6 Roll response to yaw controller. The following requirements are expressed in terms of characteristics in yaw-control-induced steady, zero-yaw-rate sideslips with the airplane 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 pounds of yaw-control-pedal force, or

c) Maximum roll control or surface deflection,

except that for single-propeller-driven airplanes 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 10 degree change in sideslip from the wings-level straight flight condition. At these sideslip angles the following shall apply:

a) A decrease in right bank angle shall not accompany an increase in right sideslip, and a decrease in left bank angle shall not accompany an increase in left sideslip. Zero roll control force or deflection is acceptable, whereas

b) A right roll-control deflection and/or force shall not accompany left sideslips, and a left roll-control deflection and/or force shall not accompany right 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 50 percent of roll-control power available to the pilot, and no more than 10 pounds of roll-control force are required in a direction opposite to that specified herein. In addition, 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 75 percent of roll-control power available to the pilot, and no more than 10 pounds of roll-stick force or 20 pounds of roll-wheel force, are required for sideslip angles that might be experienced in service employment.

D. RATIONALE BEHIND REQUIREMENT

While there is some evidence that pilots do not object to zero bank in straight sideslips, opposite bank seems to be disconcerting. Therefore, (a) requires only that bank angle does not change adversely with sideslip.

Requirement (b) is simply the "stick-free" version of (a); i.e., it is in terms of roll-control force to hold zero bank angle in sideslips. The relaxation of (b) 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 to take into account the fact that control surface position can be determined by both the pilot and the stability augmentation system. The pilot must be able both to maneuver and to cope with disturbances, so a control margin is provided.

Finally, (b) specifies allowable control power necessary for the sideslips and allowable roll-control forces as a function of type of controller. Since this relates directly to aircraft usage, that is, the size of sideslip that "might be experienced in service employment," and since this is a very strong function of aircraft type, the requirement is tied to normal operational usage. A margin of control power must be left for the pilot to cope with disturbances.

G. GUIDANCE FOR APPLICATION

This requirement is intended to preclude negative effective dihedral. It also places limits on roll control forces and displacements when the dihedral is large. 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.

F. DEMONSTRATION OF COMPLIANCE

A series of steady sideslips, as specified, should be performed over the operating envelope of the airplane.

G. SUPPORTING DATA

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 Reference 44 [see Figure 4 (3.5.1.1.1)] indicate that zero or low L_β is undesirable. Reference 44 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.

H. LESSONS LEARNED

None.

3.5.7 Roll Axis Control for Takeoff and Landing in Crosswinds

A. REASON FOR THIS REQUIREMENT

This paragraph assures good roll-axis flying qualities in crosswind takeoffs and landings and specifies the limiting crosswinds to be applied in various other roll control power and force requirements.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.7

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.7 Roll Axis Control for Takeoff and Landing in Crosswinds. It shall be possible to take off and land with normal pilot skill and technique in 90 deg crosswinds from either side of velocities up to ____ kt.

Recommended values:

TABLE 1 (3.5.7)

RECOMMENDED MINIMUM CROSSWIND VELOCITY REQUIREMENTS

LEVEL	CLASS	CROSSWIND
1 and 2	I	20 kt
	II, III, and IV	30 kt
	Water-based airplanes	20 kt
3	All	One-half the values for Levels 1 & 2

D. RATIONALE BEHIND REQUIREMENT

This requirement was taken directly from MIL-F-8785C. An attempt to specify pilot workload in terms of the Level definitions in Para. 3.9.1 was found to be too complex and hence was deleted from the draft report.

E. GUIDANCE FOR APPLICATION

The crosswind specified herein will affect not only this paragraph, but also the roll power (3.5.9.3) and force (3.5.10.6.3) requirements, all of which should be reviewed at the same time. In addition, the identical requirement for the yaw axis (3.6.3) should be considered, where it will be seen that the same crosswind velocities are specified.

F. DEMONSTRATION OF COMPLIANCE

Since flight testing in steady, 90 degree crosswinds of the specified velocity may be unacceptably hazardous for Level 2 and 3 operations, 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.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.5.8 Roll Axis Response to Other Inputs

3.5.8.1 Roll axis response to asymmetric thrust

A. REASON FOR REQUIREMENT

The need for this subjective requirement, for those airplanes for which it is applicable, is obvious.

B. RELATED MIL-F-8785B REQUIREMENT

3.3.9.3

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.8.1 Roll axis response to asymmetric thrust. The airplane motions following sudden asymmetric loss of thrust shall be such that dangerous conditions can be avoided by pilot corrective action. A realistic time delay of at least ___ second shall be considered.

Recommended time delay: 1 second.

D. RATIONALE BEHIND REQUIREMENT

This requirement is written qualitatively due to lack of real test data for specifying a quantitative limit. This is considered to be the best approach at this time in light of the many variables that would have to be considered in specifying a strictly quantitative requirement.

E. GUIDANCE FOR APPLICATION

Of primary importance in applying this requirement is choice of the most critical flight conditions. From a safety-of-flight standpoint, the most sensitive condition should be at V_{MC}, the minimum control speed. The following excerpt from FAR Part 25 (Reference 118, 25.149) serves as a reasonable guideline for designing for V_{MC}:

> [(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 5 degrees.
> [(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....]

In specifying "realistic" time delay, the user should consider the following:

> This time delay should 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.

This was how the time delay was defined in MIL-F-8785C. Depending upon expected initial pilot alertness and tightness of control, the magnitude, timing and unambiguity of pilot cues, and the type and variety of pilot action required, one second might be quite unrealistically short.

F. DEMONSTRATION OF COMPLIANCE

Simulated engine failure conditions must be performed in flight, covering at least the critical conditions specified by either the procuring activity or the contractor, and covering the range of service speed and altitude.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.5.8.2 Roll axis response to failures

A. REASON FOR REQUIREMENT

Adequate protection for failure transients and severity of failed conditions must be provided in the roll axis.

B. RELATED MIL-F-8785C REQUIREMENTS

3.4.8, 3.4.9, 3.5.5.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.8.2 Roll axis response to failures.

a) Closed-Loop: The aircraft motions following sudden aircraft system or component failures shall be such that dangerous conditions can be avoided by pilot corrective action. A time delay of at least ____ sec between the failure and initiation of pilot corrective action shall be incorporated when determining compliance. No single failure of any component or system shall result in Level 3 flying qualities; Special Failures States (1.6.3) are excepted. The crew member concerned shall be provided with immediate and easily interpreted indications whenever failures occur that require or limit any flight crew action or decision.

b) Open-Loop: With controls free, the aircraft motions due to partial or complete failure of the augmentation system shall not exceed the following limits: __________________, for at least ___ seconds following the failure.

Recommended values:

a) Minimum time delay: 1 second.

b) Transient motions (within first 2 seconds following failure):

Levels 1 and 2 (after failure): ±0.5 g incremental lateral acceleration at the pilot's station and ±10 deg per second roll rate, except that neither stall angle of attack nor structural limits shall be exceeded. In addition, for Category A, ±2 deg bank angle.

Level 3 (after failure): No dangerous attitude or structural limit is reached, and no dangerous alteration of the flight path results from which recovery is impossible.

D. RATIONALE BEHIND REQUIREMENT

The rationale behind and need for this qualitative requirement should be self-evident. Similar requirements are found in the pitch and yaw axes (3.2.7.2 and 3.6.4.2, respectively).

E. GUIDANCE FOR APPLICATION

Guidance for Application for Para. 3.5.8.1 should be consulted for information on defining a "realistic" time delay.

A pilot, especially if he is not alert for failures, may not detect a failure or adapt immediately. In some cases his consequent inability to adapt can result in a pilot/airframe closed-loop instability, even if the airplane itself remains stable (Reference 119). Allowance for this phenomenon should be made in deciding the suitability of any required pilot corrective action. The required failure indications depend on operational rules. Consistent maintenance and checkout capability and rules should be established. The flight control system specification should also be consulted, as should flight safety, maintenance, and reliability requirements.

F. DEMONSTRATION OF COMPLIANCE

Guidelines for demonstrating compliance with this requirement are difficult to specify; any failures specified under 3.1.6.2, "Generic failure analysis," should be evaluated in flight testing at the appropriate flight conditions.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.5.8.3 Roll axis response to configuration or control mode change

A. REASON FOR REQUIREMENT

Roll transients due to intentional mode switching must not be excessive.

B. RELATED MIL-F-8785C REQUIREMENTS

3.5.6, 3.5.6.1.

C. STATEMENT OF REQUIREMENTS AND RECOMMENDED VALUES

3.5.8.3 Roll axis response to configuration or control mode change. The transient motions and trim changes resulting from the intentional engagement or disengagement of any portion of the primary flight control system by the pilot shall be such that dangerous flying qualities never result. With controls free, the motion transients resulting from these situations shall not exceed the following limits for at least ____ seconds following the transfer: ____________________. These requirements apply only for Aircraft Normal States.

Recommended values for *transient motions* (within first 2 sec following transfer):

Within the Operational Flight Envelope: ±3 deg/sec roll

Within the Service Flight Envelope: ±5 deg/sec roll

D. RATIONALE BEHIND REQUIREMENT

Since the intent of a flight control system is to improve the aircraft response characteristics — whether measured by improved flying qualities or by increased mission effectiveness — any system which can be chosen by the pilot should not cause noticeable transient motions. There has been some speculation as to whether a 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.

G. GUIDANCE FOR APPLICATION

No specific guidance is offered except that tests should be conducted at the most critical flight conditions. Similar requirements for pitch (3.2.7.3) and yaw (3.6.4.3) should be considered concurrently.

F. DEMONSTRATION OF COMPLIANCE

Flight testing at the corners of the expected operational envelopes for any control systems must be performed (e.g., a SAS for power approach must be switched at the highest and lowest expected airspeeds, at low altitudes). Limited analytical and ground-based simulation may be used to supplement actual flight testing, especially in the early stages of development. But flight testing is ultimately required.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.5.8.4 Roll axis response to stores release

A. REASON FOR REQUIREMENT

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

B. RELATED MIL-F-8785C REQUIREMENT

3.4.6

C. STATEMENT OF REQUIREMENT

3.5.8.4 Roll axis response to stores release. The intentional release of any stores shall not result in objectionable flight characteristics for Levels 1 and 2. However, the intentional release of stores shall never result in dangerous or intolerable flight characteristics. This requirement applies for all flight conditions and store loadings at which normal or emergency store release is structurally permissible.

D. RATIONALE BEHIND THE REQUIREMENT

This paragraph is unchanged from MIL-F-8785C. It is a necessary catch-all requirement. Because of the variety of possibilities, it must be left qualitative. Similar pitch- and yaw-axis requirements have been specified (3.2.7.4, 3.6.4.4).

E. GUIDANCE FOR APPLICATION

Evaluation of this criterion should occur as a natural part of operational flight testing. It is not subtle and requires no special analysis or interpretation.

F. DEMONSTRATION OF COMPLIANCE

Operational flight test will be necessary for final demonstration.

G. SUPPORTING DATA

None available.

H. LESSONS LEARNED

None available.

3.5.8.5 Roll axis response to armament delivery

A. REASON FOR REQUIREMENT

This requirement is included to insure that armament delivery will not have an adverse effect on flying qualities that could impair mission effectiveness.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.7

C. STATEMENT OF REQUIREMENT

3.5.8.5 Roll axis response to armament delivery. Operation of moveable parts such as bomb bay doors, cargo doors, armament pods, refueling devices, and rescue equipment, or firing of weapons, release of bombs, or delivery or pickup of cargo shall not cause buffet, trim changes, or other characteristics which impair the tactical effectiveness of the airplane under any pertinent flight conditions. These requirements shall be met for Levels 1 and 2.

D. RATIONALE BEHIND THE REQUIREMENT

This paragraph has remained unchanged in MIL-F-8785C and in the MIL Standard.

E. GUIDANCE FOR APPLICATION

This requirement is similar to 3.5.8.4.

F. DEMONSTRATION OF COMPLIANCE

Operational flight test should be required.

G. SUPPORTING DATA

None available.

H. LESSONS LEARNED

None available.

3.5.9 Roll Axis Control Power

3.5.9.1 Roll axis control power — response to roll control inputs

A. REASON FOR REQUIREMENT

Roll power is specified in terms of bank angle change in a given time.

B. RELATED MIL-F-8785C REQUIREMENTS

3.3.4, 3.3.4.1, 3.3.4.1.1, 3.3.4.1.2, 3.3.4.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.9.1 Roll axis control power — response to roll control inputs. The response to full roll control input shall have the following characteristics: ________________________.

The following paragraphs indicate the recommended requirements.

1. Class I and II Airplanes

Roll performance in terms of a bank angle change in a given time, ϕ_t, is specified in Table 1 for Class I and Class II airplanes. 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 1 (3.5.9.1)

ROLL PERFORMANCE FOR CLASS I AND II AIRPLANES

Time to Achieve the Following Bank Angle Change (Seconds)

CLASS	LEVEL	CATEGORY A		CATEGORY B		CATEGORY C	
		60 deg	45 deg	60 deg	45 deg	30 deg	25 deg
I	1	1.3		1.7		1.3	
I	2	1.7		2.5		1.8	
I	3	2.6		3.4		2.6	
II-L	1		1.4		1.9	1.8	
II-L	2		1.9		2.8	2.5	
II-L	3		2.8		3.8	3.6	
II-C	1		1.4		1.9		1.0
II-C	2		1.9		2.8		1.5
II-C	3		2.8		3.8		2.0

2. Class III Airplanes

Roll performance in terms of ϕ_t for Class III airplanes is specified in Table 2 over the following ranges of airspeeds:

Speed Range Symbol	Airspeed Range: For Level 1	Airspeed Range: For Levels 2 and 3
L	$V_{o_{min}} \leq V < 1.8\ V_{min}$	$V_{min} \leq V < 1.8\ V_{min}$
M	$1.8\ V_{min}{}^{a} \leq V < 0.7\ V_{max}{}^{b}$	$1.8\ V_{min} \leq V < 0.7\ V_{max}$
H	$0.7\ V_{max}{}^{b} \leq V \leq V_{o_{max}}$	$0.7\ V_{max} \leq V \leq V_{max}$

[a] Or $V_{o_{min}}$, whichever is greater   [b] Or $V_{o_{max}}$, whichever is less

TABLE 2 (3.5.9.1)

CLASS III ROLL PERFORMANCE

Time to Achieve 30 deg Bank Angle Change (Seconds)

LEVEL	SPEED RANGE	CATEGORY A	CATEGORY B	CATEGORY C
1	L	1.8	2.3	2.5
	M	1.5	2.0	2.5
	H	2.0	2.3	2.5
2	L	2.4	3.9	4.0
	M	2.0	3.3	4.0
	H	2.5	3.9	4.0
3	ALL	3.0	5.0	6.0

3. Class IV Airplanes

Roll performance for Class IV airplanes is specified over the following ranges of airspeeds:

Speed Range Symbol	Equivalent Airspeed Range For Level 1	Equivalent Airspeed Range For Levels 2 and 3
VL	$V_{o_{min}} \leq V < V_{min} + 20$ KTS	$V_{min} \leq V \leq V_{min} + 20$ KTS
L	$V_{min} + 20$ KTS[a] $\leq V < 1.4\ V_{min}$	$V_{min} + 20$ KTS $\leq V < 1.4\ V_{min}$
M	$1.4\ V_{o_{min}} \leq V < 0.7\ V_{max}$ [b]	$1.4\ V_{min} \leq V < 0.7\ V_{max}$
H	$0.7\ V_{max}$ [b] $\leq V \leq V_{o_{max}}$	$0.7\ V_{max} \leq V \leq V_{max}$

[a] Or $V_{o_{min}}$, whichever is greater [b] Or $V_{o_{max}}$, whichever is less

Roll performance in terms of ϕ_t for Class IV airplanes is specified in Table 3. Roll performance for Class IV airplanes in Flight Phase CO is specified in Table 4 in terms of ϕ_t for 360 deg rolls initiated at 1 g, and in Table 5 for rolls initiated at load factors between 0.8 $n_o(-)$ and 0.8 $n_o(+)$. These requirements take precedence over Table 3. The roll performance requirements for Class IV airplanes in Flight Phase GA with large complements of external stores may be relaxed from those specified in Table 3, subject to approval by the procuring activity. For any external loading specified in the contract, however, the roll performance shall be not less than that in Table 6, where the roll performance is specified in terms of ϕ_t for rolls initiated at load factors between 0.8 $n_o(-)$ and 0.8 $n_o(+)$. For any asymmetric loading specified in the contract, roll control power shall be sufficient to hold the wings level at the maximum load factors specified in 3.2.8.2 with adequate control margin.

TABLE 3 (3.5.9.1)

ROLL PERFORMANCE FOR CLASS IV AIRPLANES

Time to Achieve the Following Bank Angle Change (Seconds)

LEVEL	SPEED RANGE	CATEGORY A			CATEGORY B	CATEGORY C
		30 deg	50 deg	90 deg	90 deg	30 deg
1	VL	1.1			2.0	1.1
	L	1.1			1.7	1.1
	M			1.3	1.7	1.1
	H		1.1		1.7	1.1
2	VL	1.6			2.8	1.3
	L	1.5			2.5	1.3
	M			1.7	2.5	1.3
	H		1.3		2.5	1.3
3	VL	2.6			3.7	2.0
	L	2.0			3.4	2.0
	M			2.6	3.4	2.0
	H		2.6		3.4	2.0

TABLE 4 (3.5.9.1)

FLIGHT PHASE CO ROLL PERFORMANCE IN 360 DEG ROLLS

Time to Achieve the Following Bank Angle Change (Seconds)

LEVEL	SPEED RANGE	30 deg	90 deg	180 deg	360 deg
1	VL	1.0			
	L		1.4	2.3	4.1
	M		1.0	1.6	2.8
	H		1.4	2.3	4.1
2	VL	1.6			
	L	1.3			
	M		1.3	2.0	3.4
	H		1.7	2.6	4.4
3	VL	2.5			
	L	2.0			
	M		1.7	3.0	
	H		2.1		

TABLE 5 (3.5.9.1)

FLIGHT PHASE CO ROLL PERFORMANCE

Time to Achieve the Following Bank Angle Change (Seconds)

LEVEL	SPEED RANGE	30 deg	50 deg	90 deg	180 deg
1	VL	1.0			
	L		1.1		
	M			1.1	2.2
	H		1.0		
2	VL	1.6			
	L	1.3			
	M			1.4	2.8
	H		1.4		
3	VL	2.5			
	L	2.0			
	M			1.7	3.4
	H		1.7		

TABLE 6 (3.5.9.1)

FLIGHT PHASE GA ROLL PERFORMANCE

Time to Achieve the Following Bank Angle Change (Seconds)

LEVEL	SPEED RANGE	30 deg	50 deg	90 deg	180 deg
1	VL	1.5			
	L		1.7		
	M			1.7	3.0
	H		1.5		
2	VL	2.8			
	L	2.2			
	M			2.4	4.2
	H		2.4		
3	VL	4.4			
	L	3.8			
	M			3.4	6.0
	H		3.4		

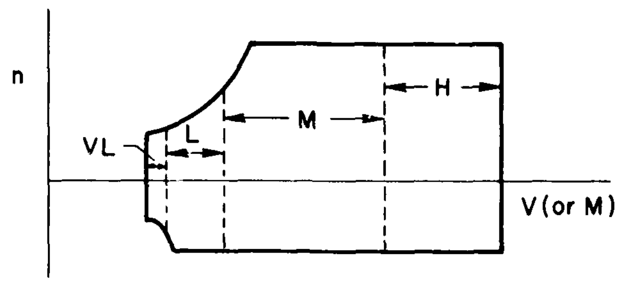
D. RATIONALE BEHIND REQUIREMENT

The tables, definitions, and wording of this requirement are collations of the various roll control effectiveness sections of MIL-F-8785C.

1. Class IV Requirements

For Class IV airplanes the Operational and Service Flight Envelopes have been divided into speed ranges with different requirements for the different speeds. This change reflects lessons learned that the roll requirements were too stringent at the extremes of the envelope. In general, the revision retains the MIL-F-8785B roll requirements in the speed range "M", with a relaxation in the other speed ranges for the Category A Flight Phases.

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 (Reference 163). 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 sketched. The problem then becomes one of defining the required speeds in a general way.



For the final version of MIL-F-8785C, which has been retained in the Standard, we returned to the definitions proposed in Reference 223. We believe that these represent the requirement for superior roll performance at combat flight conditions. The procuring activity should retain that philosophy in developing a system specification. 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 speeds for that task, with a relaxation permitted for speeds at which less maneuverability is normally required. A task requirement would then take precedence over the requirements in this section. In line with these speed ranges, the bank angle changes are compatible with the speed at which the roll performance will be demonstrated.

Relaxations in roll performance at low speed are concessions to the difficulty of doing better without adding excessive structural weight, actuator size, etc. We do this reluctantly, and some misgivings remain. The results of a recent air combat simulation (Reference 224) show the single outstanding factor influencing convergence and kill was high roll performance. 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.

2. Class III Requirements

Class III roll requirements have also been redefined 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: 30 deg in 3.3 sec, instead of 3.0

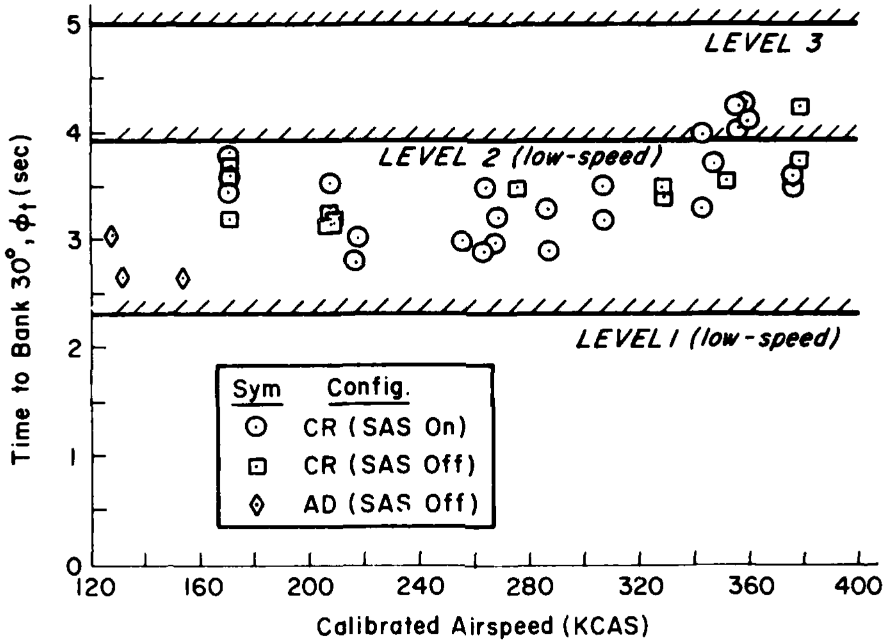
Category B, Level 3: 30 deg in 5.0 sec, instead of 4.0

Category C, Level 2: 30 deg in 4.0 sec, instead of 3.2

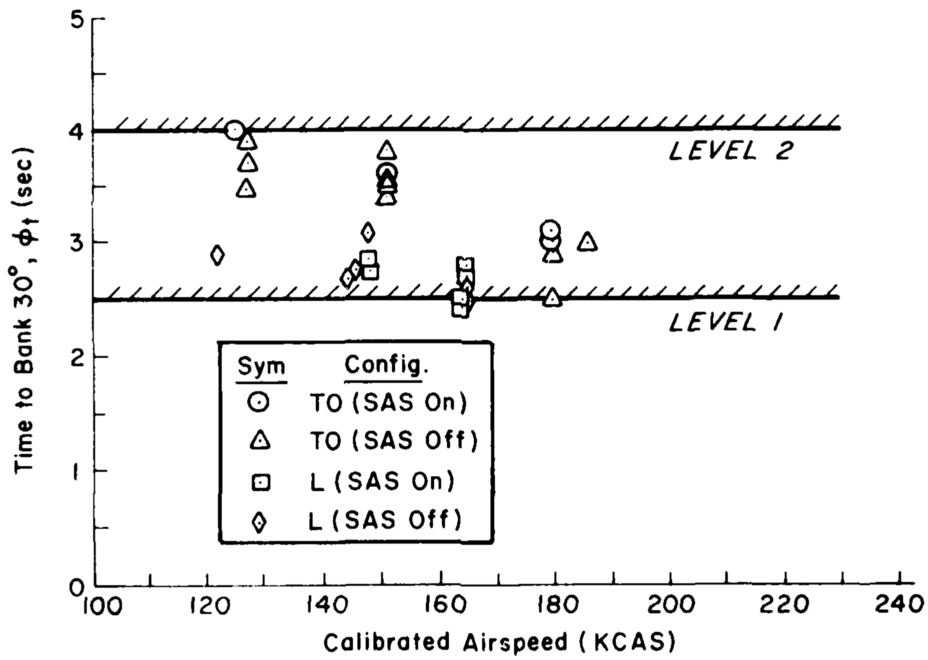
Category C, Level 3: 30 deg in 6.0 sec, instead of 4.0

Reference 59 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 were primarily for Class II and III airplanes.

Roll performance of the C-5A is shown in Figure 1 (see "Lessons Learned"). As can be seen, 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, also, the airplane was considered acceptable. Reference 36, on the other hand, 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.



a) Category B Flight Phases



b) Category C Flight Phases

Figure 1 (3.5.9.1). C-5A Flight Test Data
(From Reference 127)

E. GUIDANCE FOR APPLICATION

Of major importance in designing and producing an acceptable airplane 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's, ... 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. Rolling about any axis other than the flight path will generate sideslip, thus exciting dutch roll motion or even departure from controlled flight at high angle of attack. Other studies have shown that a major contributor to departure is the $p\alpha$ term in the side-force equation:

$$\Sigma Y = mV_o(\dot{\beta} + r - p\alpha)$$

and $p\alpha_o$ is of course zero in stability axes.

However, 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:

$$A_{y_p} = V_o\dot{\beta} + x_p\dot{r} + h_p\dot{p}$$

$$v_p = V_o\beta + x_p r + h_p p$$

Also, rolling about the flight path at high angle of attack creates a flywheel effect producing an incremental pitching moment of $p^2 I_{xz}$.

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 body and stability axes is greatest. In order to achieve the needed roll performance it may be necessary to accept some uncomfortable lateral accelerations. However, these accelerations should not exceed the limits established in Para. 3.5.5.

F. DEMONSTRATION OF COMPLIANCE

For rolls from banked flight, the initial condition shall be coordinated, that is, zero lateral acceleration. The requirements apply to roll commands to the right and to the left, initiated both from steady bank angles and from wings-level flight except as otherwise stated. Inputs shall be abrupt, with time measured from the initiation of control force application. The pitch control shall be fixed throughout the maneuver. Yaw control pedals shall remain free for Class IV airplanes for Level 1, and for all carrier-based airplanes in Category C Flight Phases for Levels 1 and 2; 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 airplane class and mission.

G. SUPPORTING DATA

Little in the way of supporting data has been generated since publication of the BIUG for MIL-F-8785B (Reference 11) in 1969. The changes made to 8785C (Reference 4) — specification of speed ranges for bank angle requirements -- are discussed in "Rationale Behind Requirement." For this Handbook and this important requirement, the focus will be on flight test experience with modern airplanes — i.e., on lessons learned.

H. LESSONS LEARNED

Considerable flight test data and pilot commentary are available on a large number of modern airplanes. 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, $\Delta\beta/k$) were Level 1, so qualitative evaluations may be somewhat biased by these other effects. Also, the operational speed ranges for Class III and IV airplanes 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 MIL-F-8785 -- bank angle change in one second.

'. Class III Airplanes

Reference 126 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 reduce but forces are still a bother). These comments are considered 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^{o}) 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 (Reference 127), comparison with the performance requirements is poor, as Figure 1 shows. Similar data for the C-141A, YC-141B, and L-1011 are given in Figure 2, from Reference 14. From Reference 14, "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 (Reference 127),

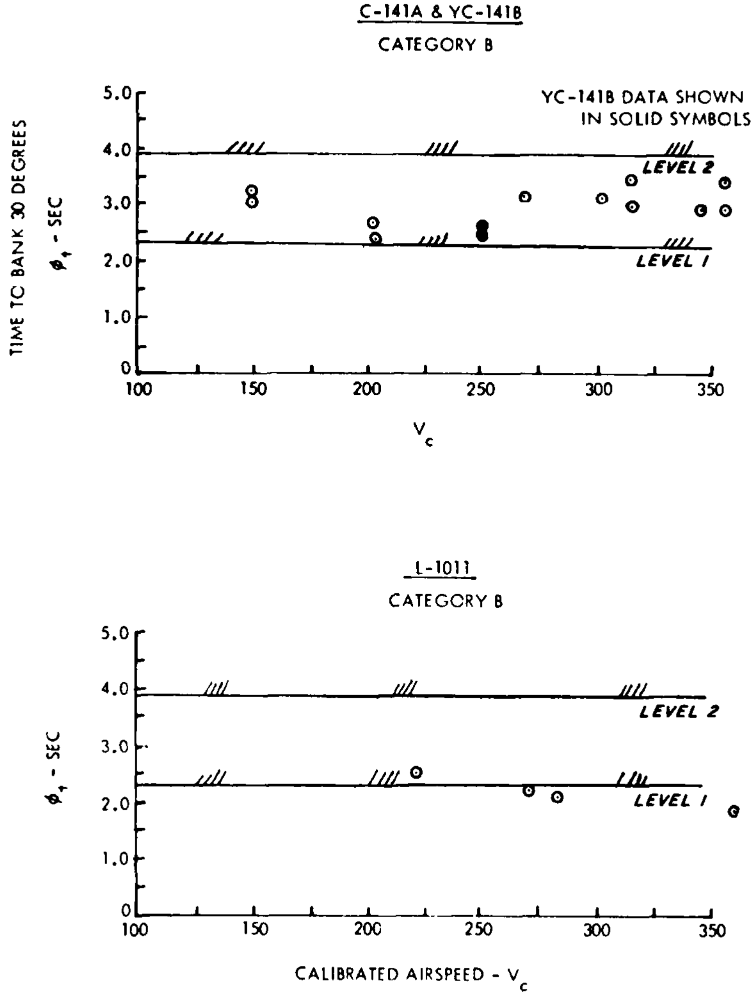


Figure 2 (3.5.9.1). Roll Performance for Class III Airplanes (From Reference 14)

> ...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 maneuver ability. The offset landing maneuver consists of approaching the runway with a 200 foot lateral misalignment on a 3 degree glideslope. At an altitude of 200 feet, the airplane is aligned with the runway centerline prior to touchdown.

Acceptability of low roll performance may be a function of the amount of lateral acceleration felt by the pilot, i.e., as the acceleration due to rapid roll control inputs increases, pilot acceptance of the roll response decreases. For the C-5A, from Reference 127,

> In order to meet the Level 1 requirements, the lateral control system would have to be improved to attain 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. Increasing the initial roll response of 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.

The flight program of Reference 128 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 30 deg bank angle. Cooper-Harper ratings for three pilots are compared in Figure 3 with times to bank 30 deg. Only those cases for which T_R is Level 1 are shown. These data support the Levels 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 one second due to high roll sensitivity.

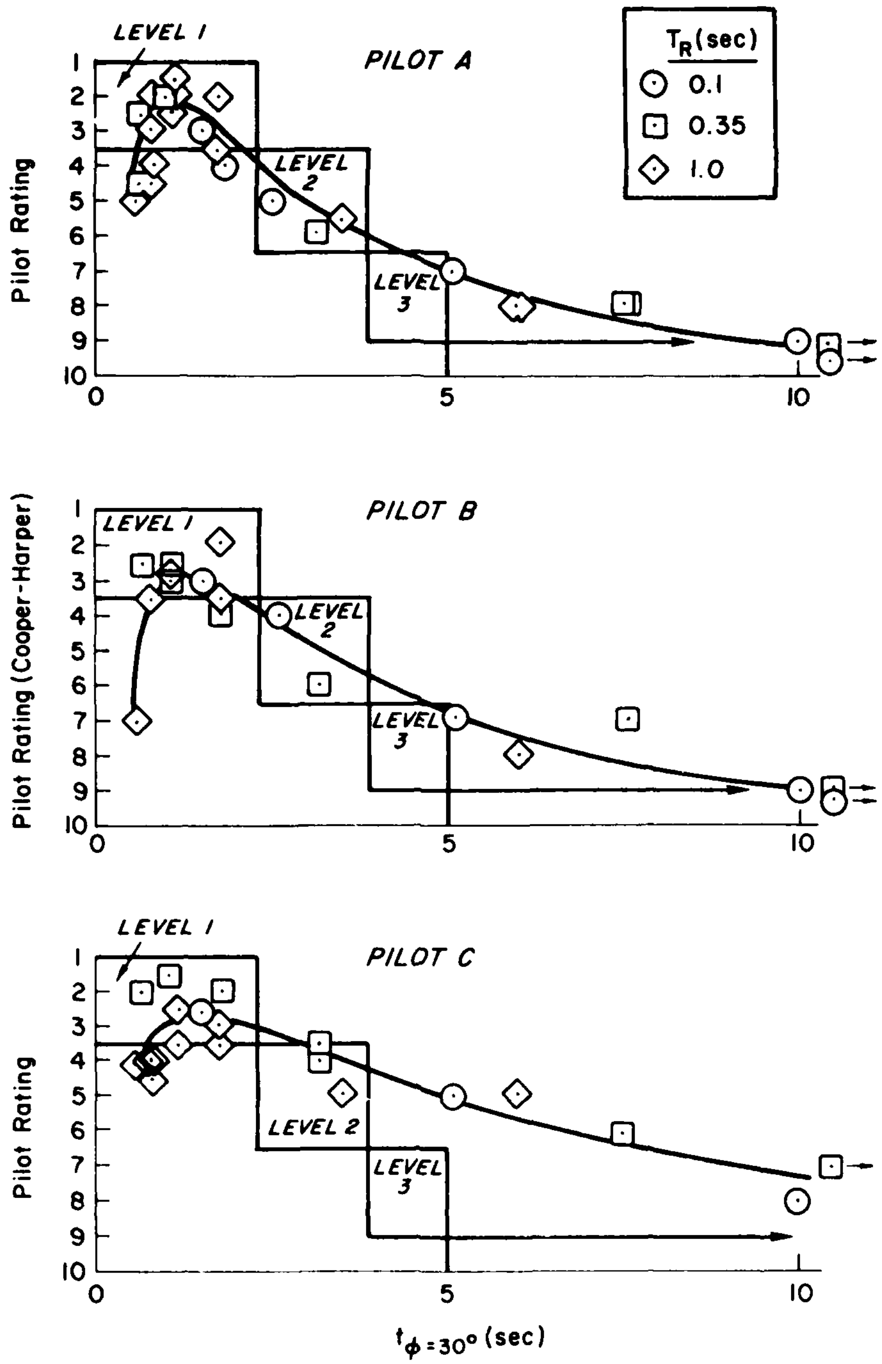


Figure 3 (3.5.9.1). Comparison of Pilot Ratings for Class III Aircraft in Category B Flight with Requirements of Table 2 (Reference 128 Data)

It must be pointed out that since the Jetstar is considerably smaller, and of different design than the C-5A, the test pilots of Reference 128 would not have been subjected to the large lateral accelerations discussed above. It seems clear, however, that there is some need for a method of relaxing the roll requirements when the Level 1 limits result in objectionable accelerations at the pilot's station.

Pilot ratings for a CV-990 (Reference 129) in Category B and C flight support the Level 1 boundaries of Table 2, as shown in Figure 4.

2. Class IV Airplanes

Data for the F-4 (Reference 130) are given in Figures 5, 6, and 7. 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 5 compares pilot ratings and ϕ_t with the Category B

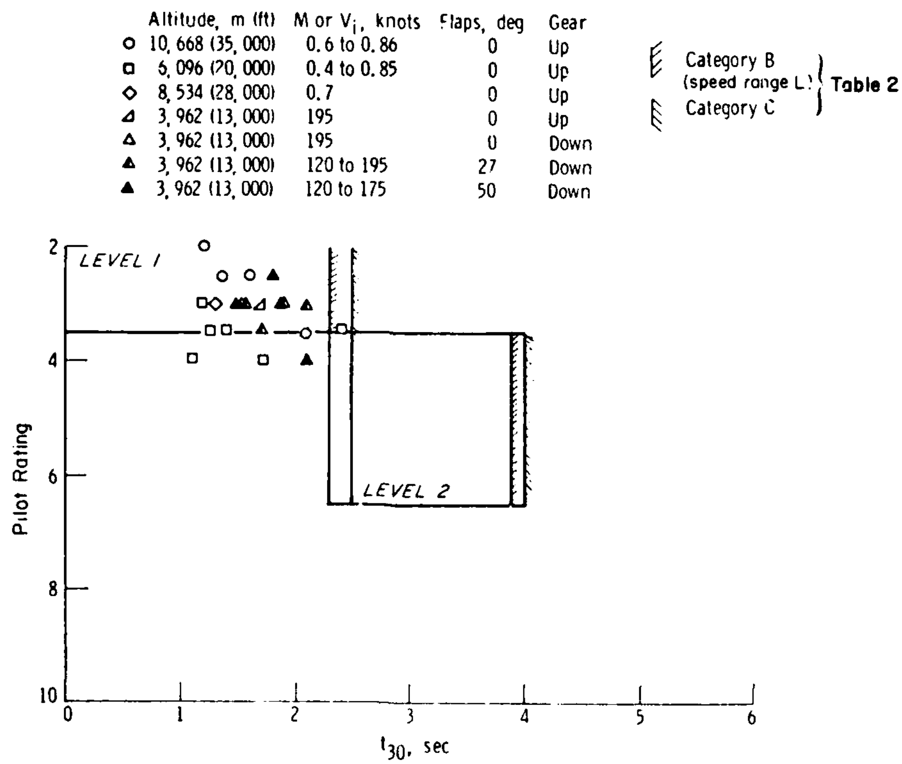


Figure 4 (3.5.9.1). Time to Bank 30° for CV-990
(Reference 129)

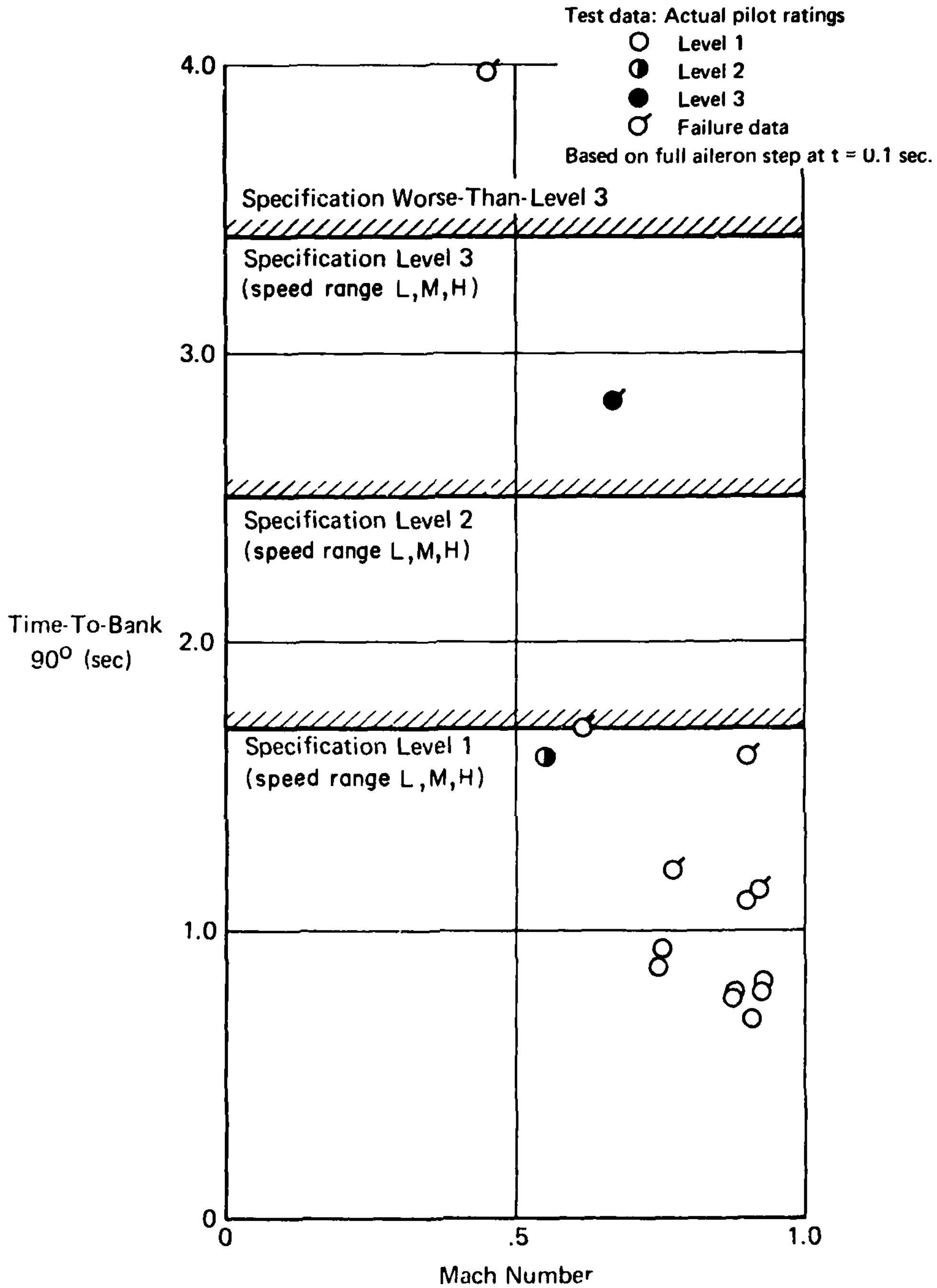


Figure 5. (3.5.9.1). F-4 Roll Control Effectiveness, Time-To-Bank 90°, CR Configuration (From Reference 130)

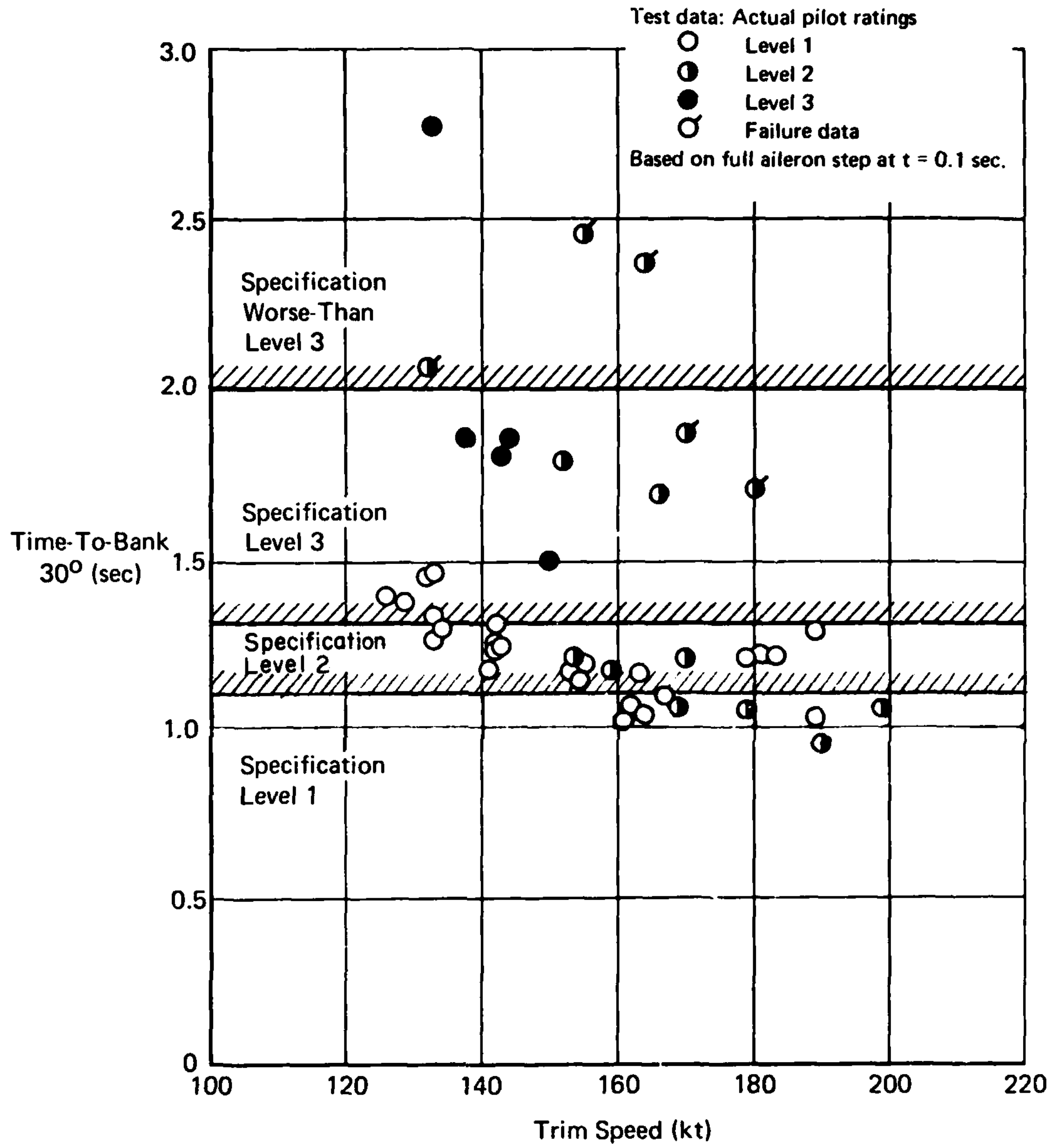
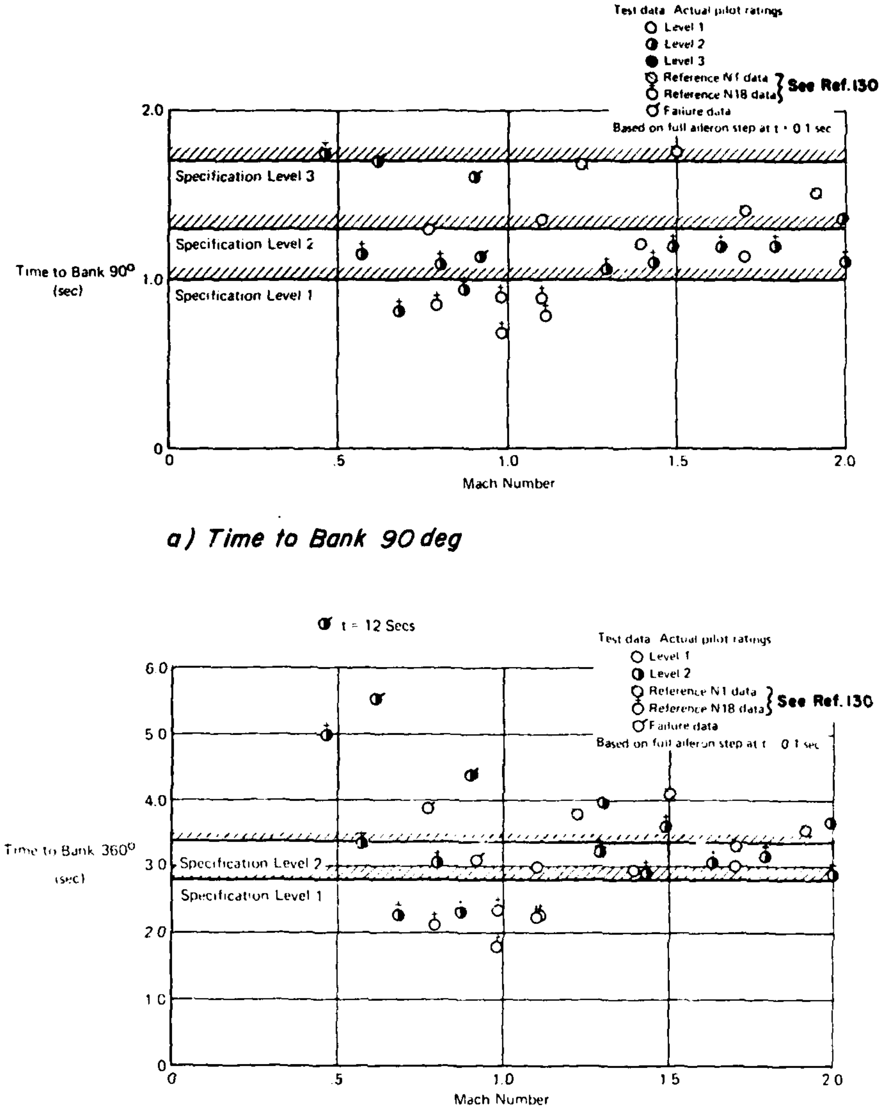


Figure 6 (3.5.9.1). F-4 Roll Control Effectiveness; Time-To-Bank 30°, PA Configuration (From Reference 130)



a) Time to Bank 90 deg

b) Time to Bank 360 deg

Figure 7 (3.5.9.1). F-4 Roll Control Effectiveness; CO Configuration. Limits Shown for Speed Range M, Table 5 (From Reference 130) requirements of Table 3 (for all speed ranges except VL). While the data are sparse, it at least appears that the Level 1 requirement of 90 deg in 1.7 sec is not too stringent. For Category C flight (Figure 6), it is suggested that the limits may be too severe. Reference 130 recommends that "For Class IV-L and -C aircraft, the Level 1 minimum time to bank to 30° should be relaxed to 1.3 seconds and the lower Level 3 boundary should be relaxed to 30° in 2.8 seconds." Of course, the impact of the Figure 6 data is mitigated by recognizing the possible inaccuracies involved in computing ϕ_t, without actual test data.

In Figure 7 the boundaries drawn for times to bank 90 deg and 360 deg are for the most stringent requirements of Table 4, i.e., for Speed Range M. The data tend to support the Level 1 limits, but suggest that the Level 2 limits could be relaxed considerably.

F-5E data (Figure 8, from Reference 6) for Flight Phase CO at elevated load factors do not agree well with the Level 1 limits of Table 5. The F-5E meets the requirement only in the High Speed Range.

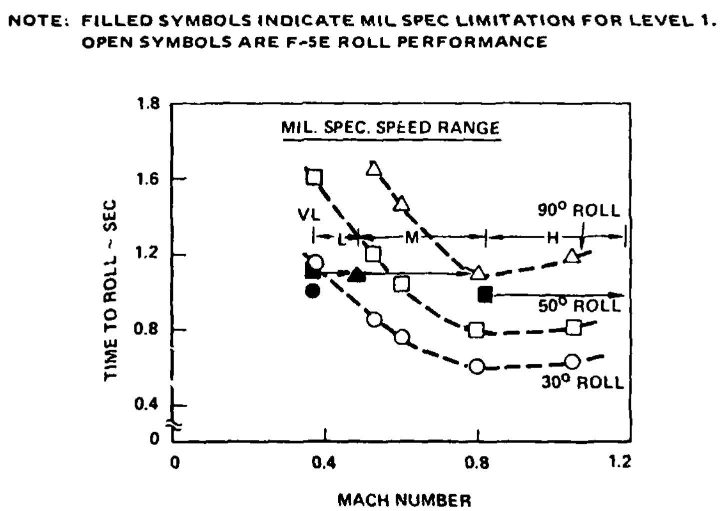


Figure 8 (3.5.9.1). F-5 E Roll Performance at 0.8 n_L, Configuration CO (from Reference 6)

Reference 6 describes the roll performance as "very satisfactory in operational use," and according to Reference 131, "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."

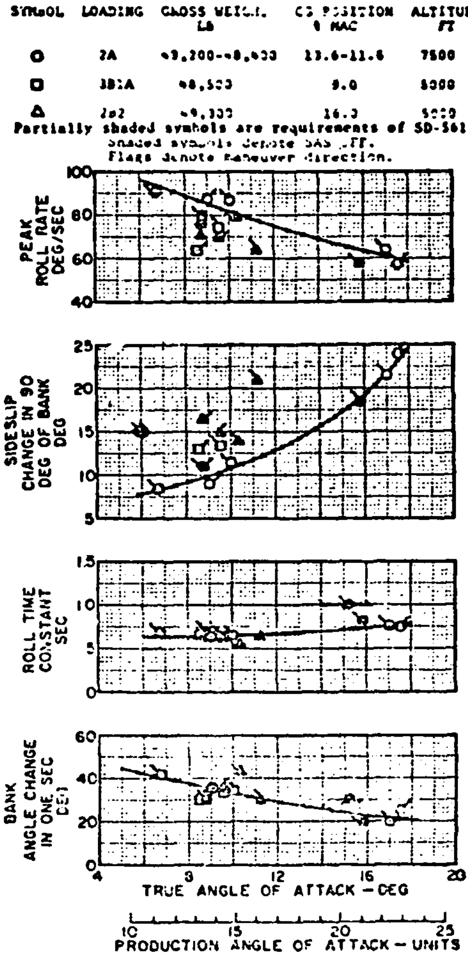
Flight test data for the F-14A in power approach are shown in Figure 9, from Reference 132. Roll performance in terms of bank angle change in one second suggests that the F-14A would meet the Table 3, Category C requirement of 30 deg in 1.1 sec at low angles of attack, and fail to meet it at high angles of attack. Reference 132 states that "Lateral control effectiveness in the PA configurations was adequate to perform the bank angle changes required during approach."

The F-15C (Reference 133) meets the Category A Level 1 requirement of 90 deg in 1.3 sec (Speed Range M), as Figure 10 shows. Unfortunately, we do not have any pilot ratings or comments relating specifically to F-15 roll performance.

Reference 134 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 in Figures 11 and 12. 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 50 deg/sec were encountered at 5000 ft altitude. From Reference 134,

> 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 $\bar{q}$ flight conditions will preclude the pilot's ability to effectively perform the air-to-air and air-to-ground missions.

The difference in times to bank for left versus right rolls (see Figures 11 and 12) 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



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 9 (3.5.9.1). F-14A Rolling Performance in Configuration PA; DLC on (From Reference 132)

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*

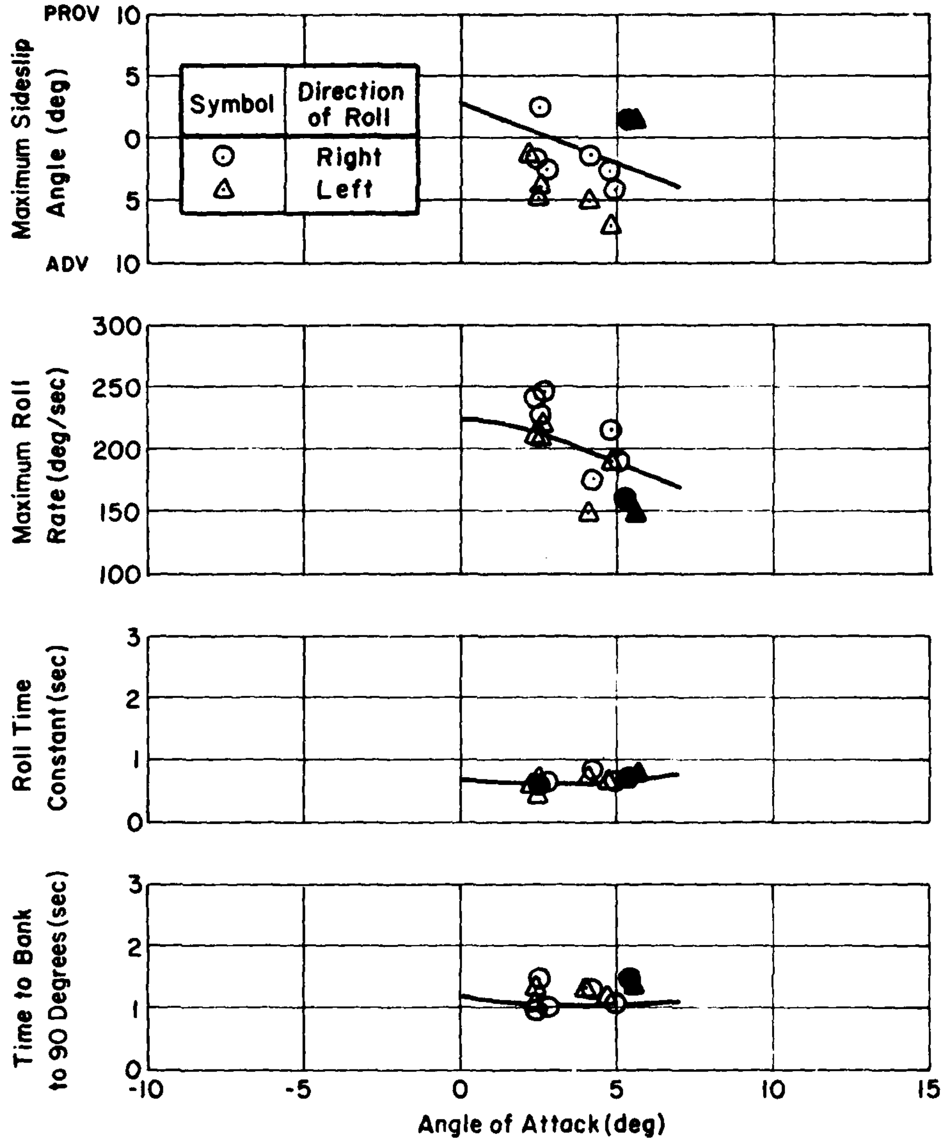


Figure 10 (3.5.9.1). F-15C Aileron Roll Characteristics
(From Reference 133)

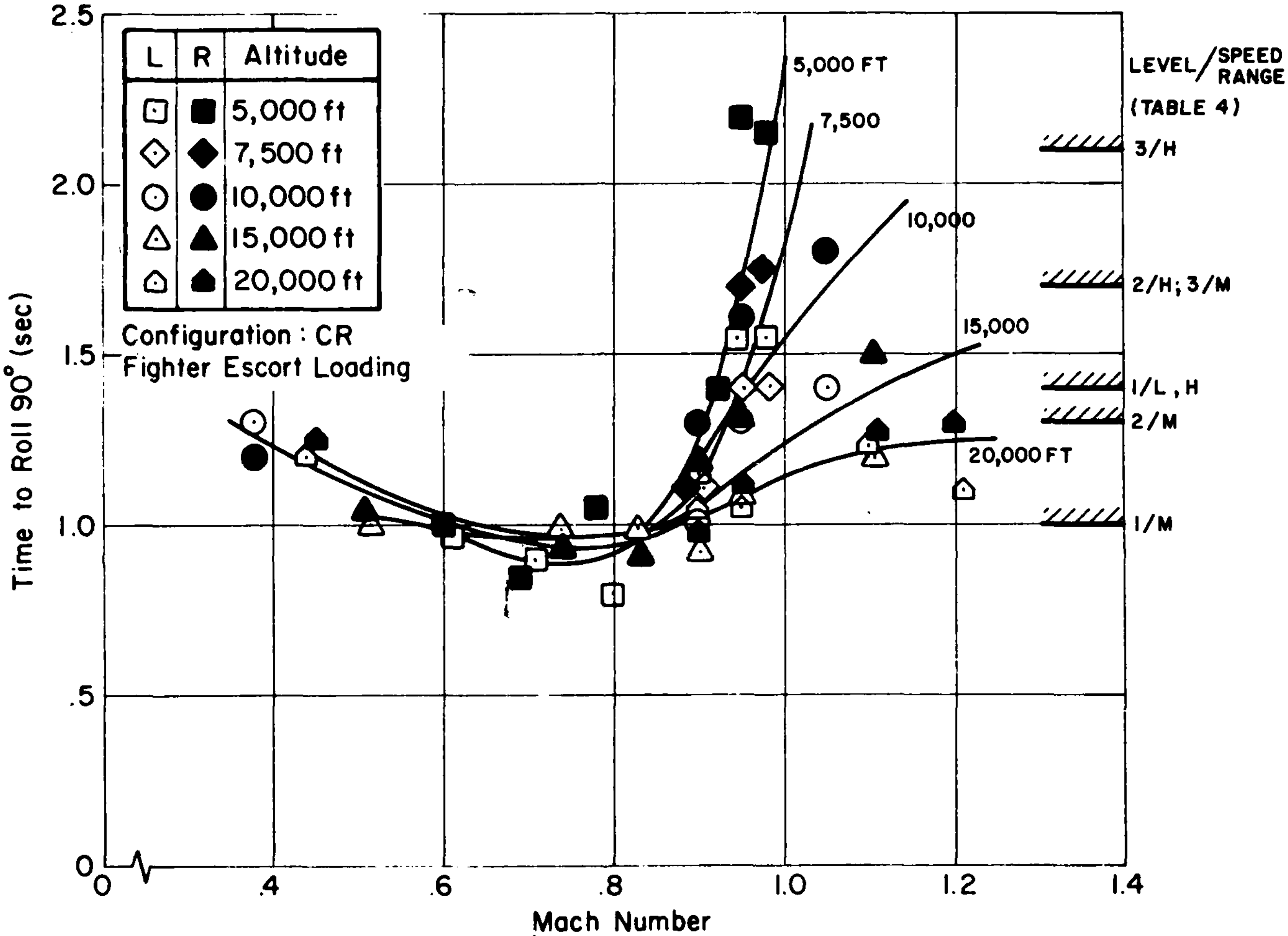


Figure 11 (3.5.9.1). Time to Roll 90° Versus Mach for F/A-18A (Reference 134)

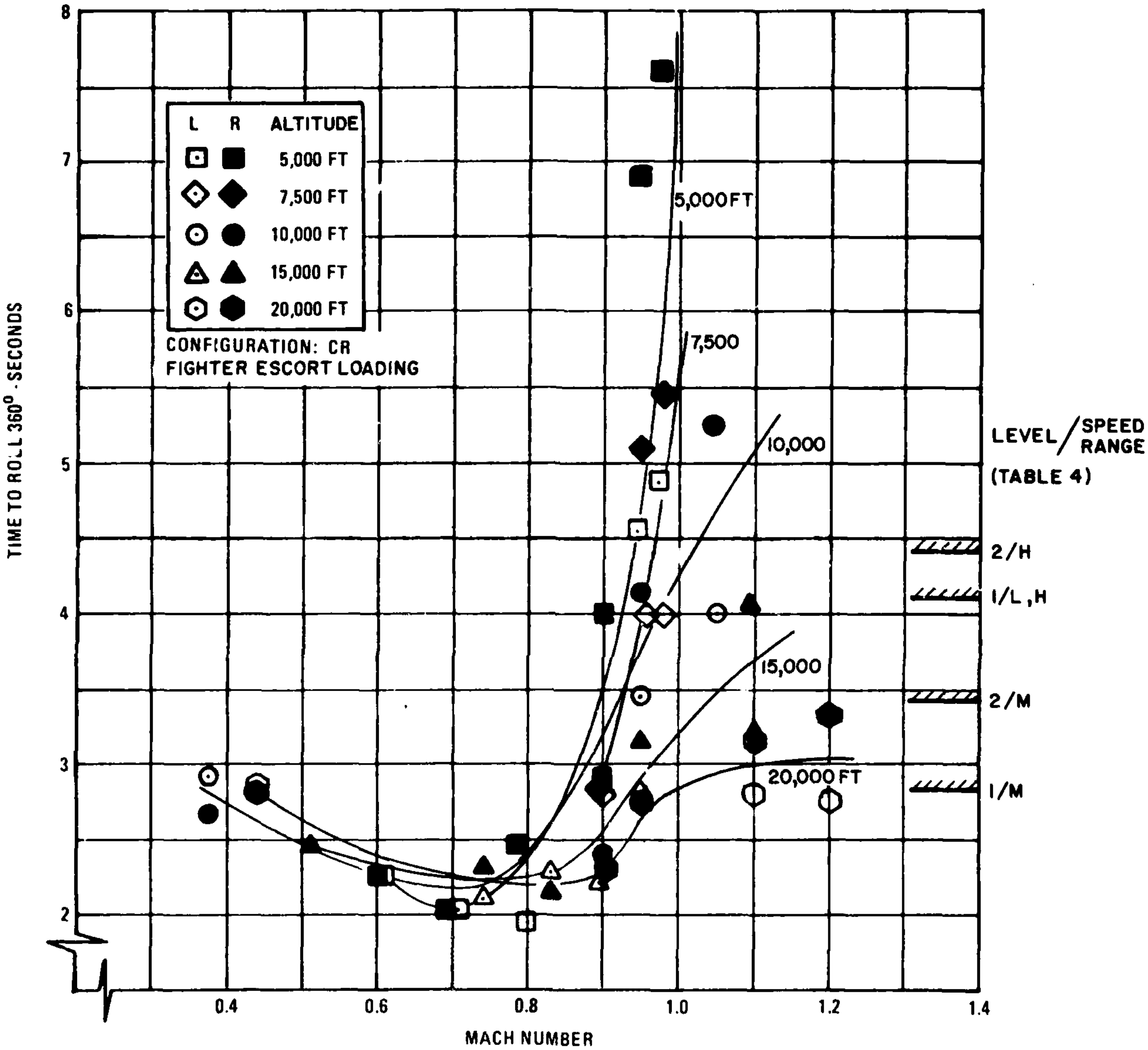


Figure 12 (3.5.9.1). Time to Roll 360° Versus Mach for F/A-18A (From Reference 134) significantly reduced the amount of control deflection change available to command a right roll as opposed to a left roll." While Figure 11 shows Level 1 roll performance at low airspeeds, fine, precise control was found to be sluggish: "the pilot was unable to perform the fine tracking task at 200 KCAS/2 g [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 (3.5.1.1.1) or τ_e (3.5.1.1.5).

F/A-18A roll performance in PA (landing and takeoff) configurations is seen to be marginally Level 1 (Figure 13) below 180 kt, i.e., the responses are very close to the required 30 deg in 1.1 sec.

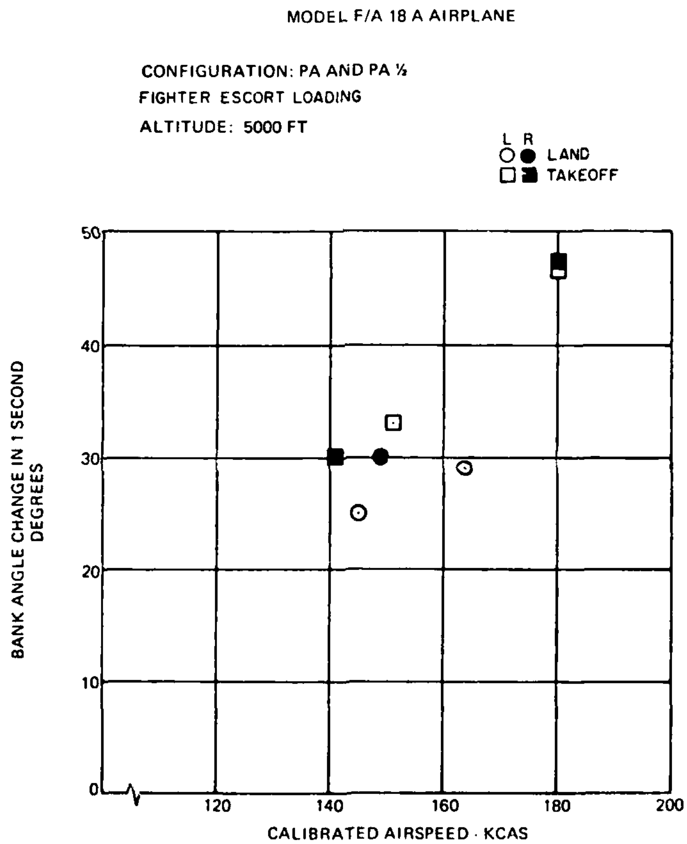


Figure 13 (3.5.9.1). Roll Performance Characteristics in Configuration PA (from Reference 134)

3.5.9.2 Roll axis control power in steady sideslips

A. REASON FOR REQUIREMENT

This requirement assures adequate roll power for operation in sideslips, with some additional control margin to correct for turbulence, etc.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.6.3.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUE

3.5.9.2 Roll axis 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 ____ percent of roll control power available to the pilot is required for sideslips which might be encountered in service deployment.

Recommended value: *no more than 75 percent of roll power should be* required.

D. RATIONALE BEHIND REQUIREMENT

Some limit on positive effective dihedral is necessary; this requirement places such a limit and assures sufficient roll power to counteract the dihedral effect.

E. GUIDANCE FOR APPLICATION

No discussion necessary.

F. DEMONSTRATION OF COMPLIANCE

Analysis should reveal any large dihedral effect. If it is anticipated to be a limiting factor on roll control power, flight testing should be performed. However, if dihedral effect is shown through analysis to be minimal, analytical compliance should be sufficient, though qualitative flight testing should be performed for final verification in any case.

G. SUPPORTING DATA.

None.

H. LESSONS LEARNED

None.

3.5.9.3 Roll axis control power in crosswinds

A. REASON FOR REQUIREMENT

Roll control power available beyond that required for flight in steady sideslip or steady crosswinds must be adequate for maneuvering and countering atmospheric disturbances.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.7.1, 3.3.7.2, 3.3.7.3, 3.3.9

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.9.3 Roll axis control power in crosswinds.

a) It shall be possible to taxi at any angle to a ____ kt wind.

b) Roll control power, in conjunction with other normal means of control, shall be adequate to maintain a straight path during the takeoff run, or landing rollout, in crosswinds up to those specified in 3.5.7.

c) Roll control power shall be adequate to maintain wings level with up to _____ deg of sideslip in the power approach. For Level 1 this shall require not more than _____ percent of the control power available to the pilot.

d) Following sudden asymmetric loss of thrust from any factor, the airplane shall be safely controllable in roll in the crosswinds of 3.5.7 from the unfavorable direction.

Recommended values:

Wind speeds for taxi:

Class I airplanes:	35 kt
Class II, III, and IV airplanes:	45 kt

For sideslip angles of 10 deg in the power approach, not more than 75 percent of available control power should be required.

D. RATIONALE BEHIND REQUIREMENT

The conditions under which it must be possible to taxi have been specified since there is generally no point in being able to take off or land in a given crosswind if the aircraft cannot be taxied. The wind speeds specified are a compromise between what is desired and what is reasonable to require.

Additionally, ability to counter rolling moments in sideslips or in crosswinds is essential for safe takeoffs and landings.

E. GUIDANCE FOR APPLICATION

Paragraphs 3.5.10.6.3 and 3.5.10.6.4 cover similar requirements in terms of roll control forces, and 3.6.5.1 covers yaw control power in these conditions. All should be considered as a group with consistent requirements throughout.

F. DEMONSTRATION OF COMPLIANCE

For requirement (a), ground taxi in winds that are at least close in magnitude to those required should be performed. Choice of wind speed conditions should account for variability and gustiness.

For discussion of requirements (b) and (c), the reader is referred to 3.5.10.6.3.

For requirement (d), as for all requirements on asymmetric thrust (and on crosswinds), the most critical operation is takeoff and landing. Crosswind landings performed to demonstrate compliance with 3.5.7 must include simulated asymmetric thrust, with the crosswind blowing in the most unfavorable direction [i.e., from the direction of the "good" engine(s)]. Operation with an engine failure will in most aircraft be Level 2, which implies a pilot rating of 6-1/2 in a 10 kt crosswind. It is clear that the subjective nature of this requirement ("safely controllable") allows considerable leeway in its application. Asymmetric loss of thrust may be caused by many factors including engine failure, inlet unstart, propeller failure or propeller-drive failure.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.5.9.4 Roll axis control power for engine failure

A. REASON FOR REQUIREMENT

This requirement is included to establish roll control power requirements to handle an engine failure during takeoff.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.9.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.9.4 Roll axis control power for engine failure. During the takeoff run it shall be possible to maintain roll control of the aircraft, following a sudden loss of thrust from the most critical propulsive source. This requirement shall apply from a minimum speed of $V_{min}(TO)$ to a maximum speed of $V_{max}(TO)$.

The roll control required shall not exceed ____ percent of the available roll control power. This assumes takeoff thrust is maintained on the operative engines with trim at normal setting for symmetric thrust. The aircraft may be banked up to 5 deg away from the inoperative engine.

Recommended values: Roll power required should be not more than 75 percent of available.

D. RATIONALE BEHIND REQUIREMENT

A control power margin is desired for any necessary maneuvering and for countering atmospheric disturbances.

E. GUIDANCE FOR APPLICATION

The requirements of 3.5.10.6.5, 3.6.4.1, and 3.6.6.2.8 should be applied in conjunction with this paragraph.

F. DEMONSTRATION OF COMPLIANCE

See discussion under 3.6.4.1.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.5.9.5 Roll axis control power in dives and pullouts

A. REASON FOR REQUIREMENT

Roll control power must be adequate to perform any dive maneuvers specified.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.8

C. STATEMENT OF REQUIREMENT

3.5.9.5 Roll axis control power in dives and pullouts. Roll control power shall be adequate to maintain wings level without retrimming, throughout the dives and pullouts of 3.2.9.7.3.

D. RATIONALE BEHIND REQUIREMENT

Since Paragraph 3.2.9.7.3 defines the pitch force limits for dives and pullouts, this requirement simply assures that roll power is not the limiting factor in performing the dives required to meet mission objectives.

E. GUIDANCE FOR APPLICATION

The qualitative nature of this requirement makes it straightforward to apply.

F. DEMONSTRATION OF COMPLIANCE

See discussion under 3.2.9.7.3.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.5.9.6 Roll axis control power for stores release

A. REASON FOR REQUIREMENT

This requirement is included to insure that intentional release of stores does not result in limitations in roll control power.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.6, 3.3.4.1.2

C. STATEMENT OF REQUIREMENT

3.5.9.6 Roll axis control power for stores release. Roll control power shall be adequate to regain wings level, without retrimming, following intentional release of any stores, to the maximum load factors specified in 3.2.8.2 with adequate control margin.

D. RATIONALE BEHIND REQUIREMENT

This is effectively a new requirement, though it is based on the general requirement of 8785C (Reference 4) that stores release "shall not result in objectionable flight characteristics." This has been translated as requiring sufficient roll power to regain wings-level conditions following any stores release.

E. GUIDANCE FOR APPLICATION

This requirement is closely related to the roll and yaw axis responses to stores release, 3.5.8.4 and 3.6.4.4, and to the similar yaw axis requirement, 3.6.5.4. All these paragraphs should be considered in combination for application.

F. DEMONSTRATION OF COMPLIANCE

Flight testing must be conducted with the stores configurations specified in 3.1.3, over the applicable range of flight conditions.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.5.9.7 Roll axis control power for two engines inoperative

A. REASON FOR REQUIREMENT

This paragraph states the need for roll control power in event of failure of more than one engine on multi-engine airplanes.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.9.5

C. STATEMENT OF REQUIREMENT

3.5.9.7 Roll axis control power for two engines inoperative. At the one-engine-out speed for maximum range with any engine initially failed, upon failure of the most critical remaining engine the roll control power shall be adequate to stop the transient motion and thereafter to maintain straight flight from that speed to the speed for maximum range with both engines failed. In addition, it shall be possible to effect a safe recovery at any service speed above $V_{o_{min}}$ (CL) following sudden simultaneous failure of the two critical engines.

D. RATIONALE BEHIND REQUIREMENT

The rationale for this requirement is self evident; it is taken essentially intact from MIL-F-8785C.

E. GUIDANCE FOR APPLICATION

The specialized nature of the requirement, as well as the general, qualitative terms in which it is worded, makes it straightforward to apply.

F. DEMONSTRATION OF REQUIREMENT

Flight testing at altitudes covering the operational envelope will be at conditions stated by the requirements, i.e., airspeeds will be either speed for maximum range with one engine out or representative speeds above $V_{o_{min}}$ (CL).

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.5.9.8 Roll axis control power for other conditions

A. REASON FOR REQUIREMENT

This catch-all specification is intended to assure adequate roll control power in any situation not already covered in the Standard.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.10

C. STATEMENT OF REQUIREMENT

3.5.9.8 Roll axis control power for other conditions. Control authority, rate and hinge moment capability shall be sufficient to assure safety throughout the combined range of all attainable angles of attack (both positive and negative) and sideslip. This requirement applies to the prevention of loss of control and to recovery from any situation for all maneuvering, including pertinent effects of factors such as regions of control-surface-fixed instability, inertial coupling, fuel slosh, the influence of symmetric and asymmetric stores, stall/post-stall/spin characteristics, atmospheric disturbances and Aircraft Failure States (maneuvering flight appropriate to the Failure State is to be included). Consideration shall be taken of the degrees 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.

D. RATIONALE BEHIND REQUIREMENT

The other paragraphs under 3.5.9 cover all normal, anticipated situations for roll control power. This paragraph is added to cover any unusual or unspecified conditions that might be encountered in flight. For further discussion on this requirement, see its equivalent in the pitch axis, 3.2.8.5.

E. GUIDANCE FOR APPLICATION

Due to its broad generality the requirement should be applied for all phases of analysis, simulation, and flight test. Excessive stability, as well as excessive instability, of the basic airframe is of concern with respect to available control authority and rate.

F. DEMONSTRATION OF COMPLIANCE

See 3.2.8.5.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

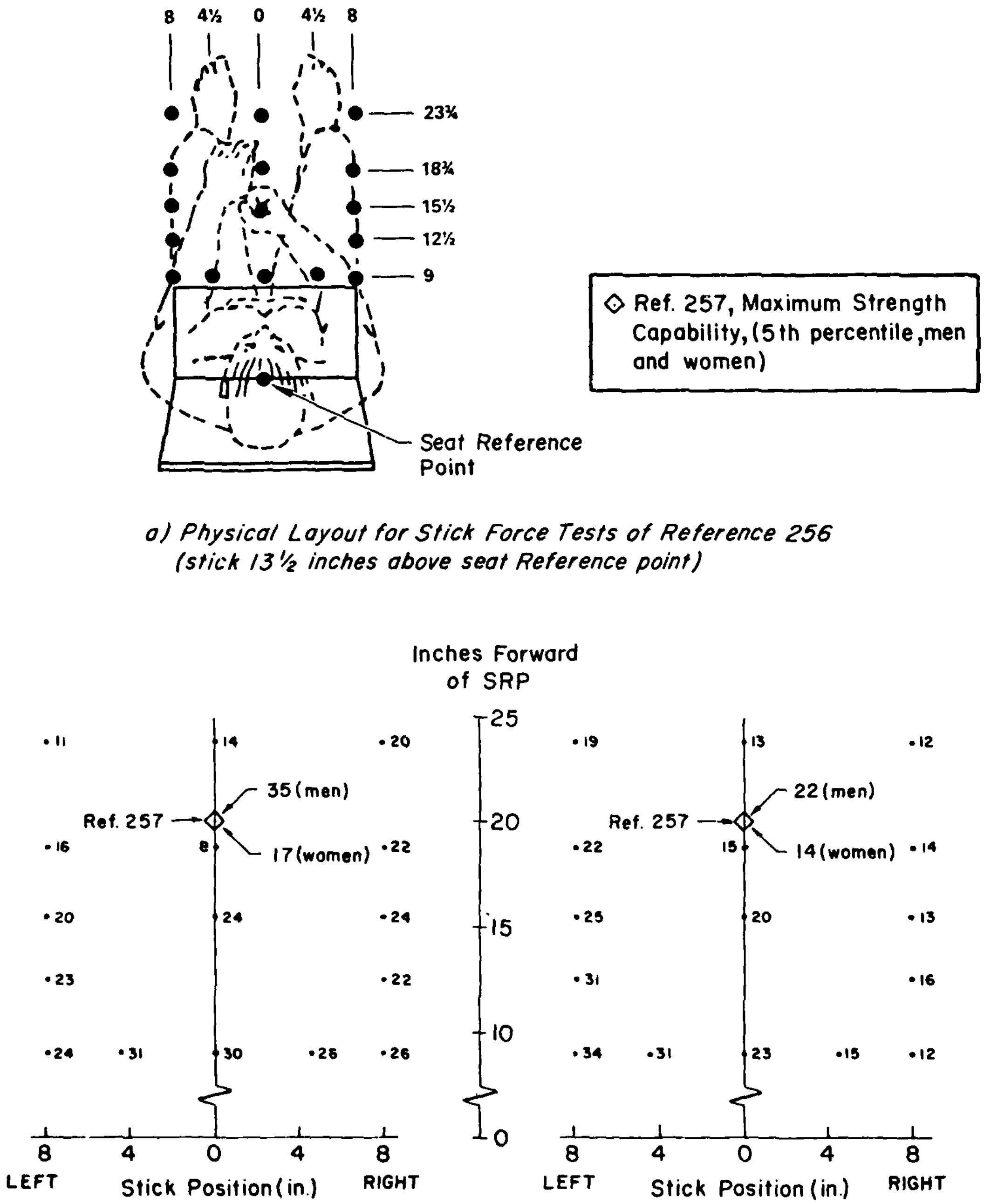
3.5.10 Roll Axis Control Forces and Displacements

DISCUSSION

The requirements of this section cover broad areas for forces and displacements of roll controllers. Many of the paragraphs (most specifically, the force limits of 3.5.10.6) set absolute upper limits on allowable control forces in various maneuvers. As with the pitch axis forces of 3.2.9, 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 is scarce, there is some available dealing with maximum forces for single-application tasks (References 256, 257). 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 1 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 2 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 Reference 256. Single test points from a more recent study by McDaniel (Reference 257) are shown on Figure 1 as a comparison between male and female strength characteristics for operating an aircraft control stick. Figures 1b and 1c 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 Reference 257) shown in Figures 1b and 1c 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., in Figure 1b the Reference 256 data show about 8 lb at the same location where the Reference 257 tests show 35 lb). The reason for this difference is not known.

The 5th, 50th, and 95th percentiles of maximum forces exerted on an aircraft control stick by 61 men and 61 women (from Reference 257) are summarized in Table 1.

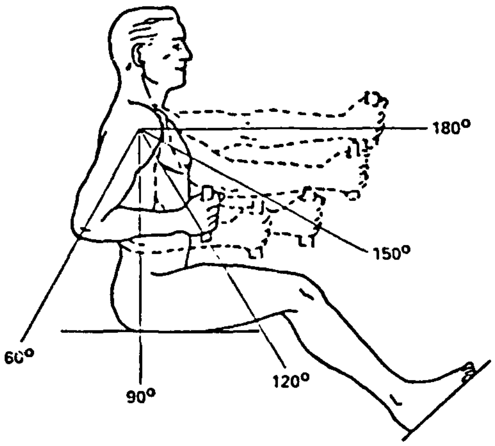


a) Physical Layout for Stick Force Tests of Reference 256 (stick 13½ inches above seat Reference point)

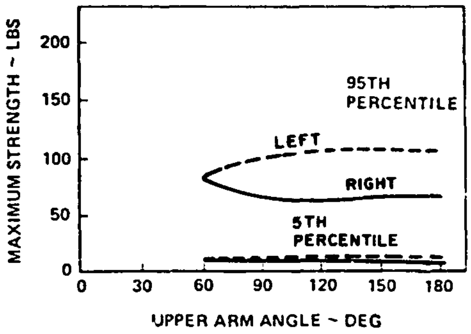
b) Maximum Force to the Left (lb)

c) Maximum Force to the Right (lb)

Figure 1 (3.5.10). 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 (Reference 256)



a) Physical Layout Fortest



b) Maximum Strength vs. Upper Arm Angle

Figure 2 (3.5.10). Effect of Upper Arm Angle on Maximum Applied Force to the Left and to the Right for the 5th and 95th Percentile Male (from Reference 256)

TABLE 1 (3.5.10)

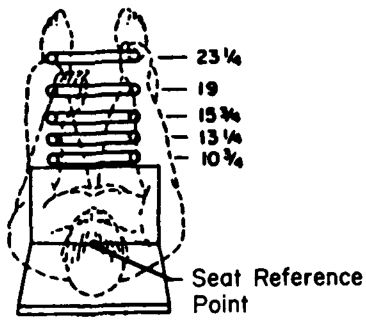
MAXIMUM FORCES EXERTED ON AIRCRAFT CONTROL STICK (LB) BY 61 MEN AND 61 WOMEN (REF. 257)

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

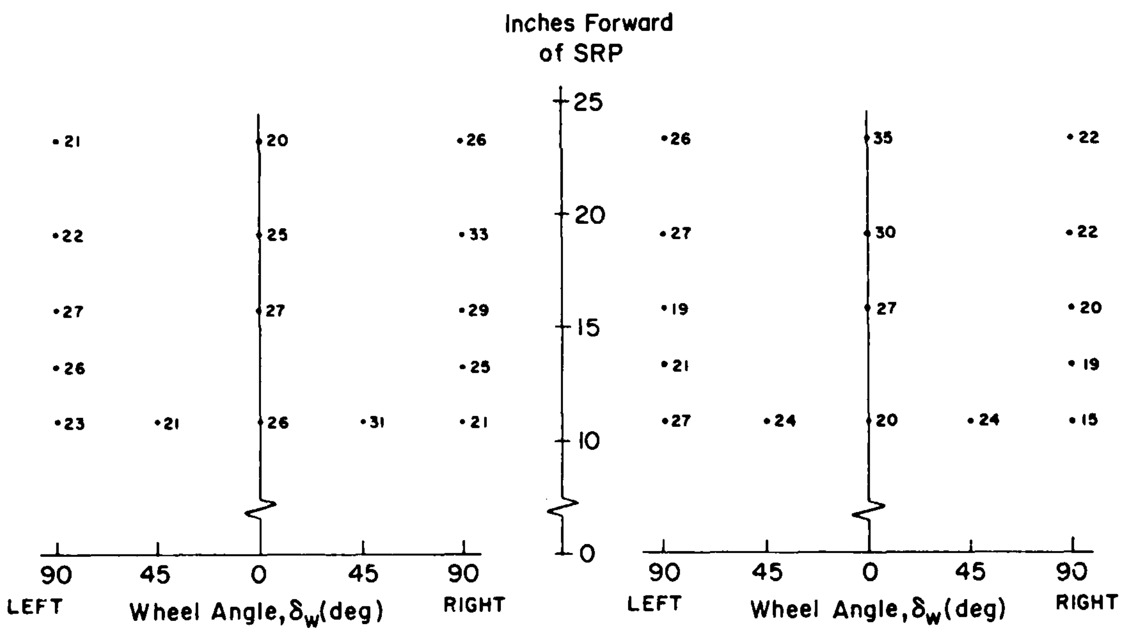
Figure 3 illustrates the effect of arm position and wheel angle on maximum applied force to the left and to the right for the 5th percentile male.

Reference 256 discusses one- vs. two-handed operation of controls and some general principles of control design; for example:

> ...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 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. When two hands are used 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 should be much lower.



a) Physical Layout for Wheel Force Tests of Reference 256 (wheel 18 inches above seat reference point)



b) Maximum Force to the Left (lb)

c) Maximum Force to the Right (lb)

Figure 3 (3.5.10). Effect of Arm Position and Wheel Angle on Maximum Applied Force to the Left and to the Right for the 5th Percentile Male (Reference 256)

3.5.10.1 Wheel control displacements

A. REASON FOR REQUIREMENT

This requirement is intended to place limits on the allowable control wheel angular displacement used to attain maximum rolling performance in Paragraph 3.5.9.

B. RELATED MIL-F-8785C REQUIREMENT AND RECOMMENDED VALUES

3.3.4.5

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.10.1 Wheel control displacements. For airplanes with wheel controllers, the wheel throw necessary to meet the roll performance requirements specified in 3.5.9 shall not exceed ____ degrees in either direction.

Recommended wheel displacement: 60 deg. For completely mechanical systems the requirement may be relaxed to 110 deg.

D. RATIONALE BEHIND REQUIREMENT

An upper limit on allowable wheel throw assures that unreasonable demands will not be made on the pilot. If the throw is too small, of course, the airplane can be overly sensitive to small inputs; however, from a comfort (and safety) standpoint, maximum throw is more crucial. The small throw of 60 deg is attainable in normal, one-handed operation without undue physical effort.

A wheel throw of 110 deg for completely mechanical systems has been specified in deference to the design problem posed by such systems.

E. GUIDANCE FOR APPLICATION

None required.

F. DEMONSTRATION OF COMPLIANCE

The tests required for compliance with applicable portions of 3.5.9 should include measurement of wheel control throw.

G. SUPPORTING DATA

As data from Reference 116 indicate (Figure 1), 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.)

Reference 117 makes recommendations concerning the amount of wheel throw for one-handed operation, and, although the comments pertain to VTOL vehicles, the recommendation may well be of general applicability. The Reference 117 recommendation is that for one-handed operation the wheel throw should not exceed 60 deg in each direction.

H. LESSONS LEARNED

None.

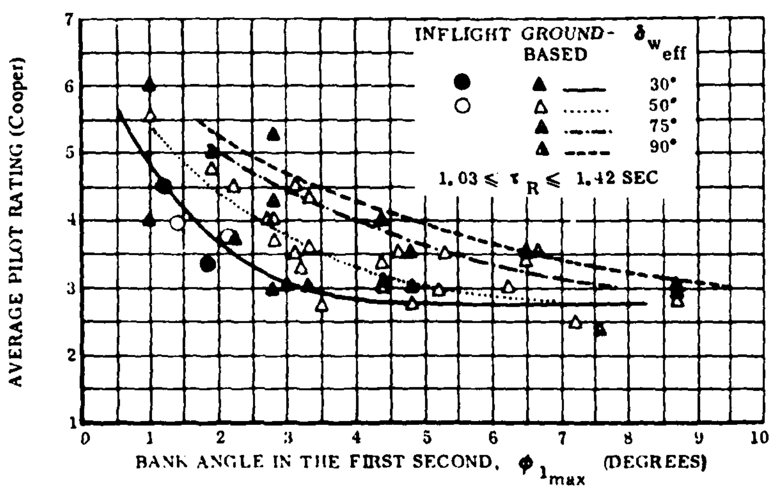


Figure 1 (3.5.10.1). Variation of Pilot Rating with Bank Angle in the First Second for Four Values of Effective Angle (from Reference 116)

3.5.10.2 Roll axis control forces to achieve required roll rates

A. REASON FOR REQUIREMENT

This paragraph limits the forces required to obtain the specified roll performance.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.4.3

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.10.2 Roll axis control forces to achieve required roll rates. The roll control force required to obtain the rolling performance specified in 3.5.9.1 shall be neither greater than ____________ nor less than __________.

TABLE 1 (3.5.10.2)

RECOMMENDED MAXIMUM ROLL CONTROL FORCE

LEVEL	CLASS	FLIGHT PHASE CATEGORY	MAXIMUM FORCE (Pounds)		
			CENTERSTICK	WHEEL	SIDESTICK*
1	I, II-C, IV	A, B C	20 20	40 20	
	II-L, III	A, B C	25 25	50 25	
2	I, II-C, IV	A, B C	30 20	60 20	
	II-L, III	A, B C	30 30	60 30	
3	All	All	35	70	

*No forces are recommended for sidestick controllers at this time. However, forces should not be so large or so small as to be objectionable to the pilot.

Recommended minimum roll control force for all controllers is the sum of the breakout force plus:

Level 1: One-fourth the values in Table 1

Level 2: One-eighth the values in Table 1

Level 3: Zero

D. RATIONALE BEHIND REQUIREMENT

In combination with the roll control power requirements of 3.5.9.1, this paragraph specifies control force gradients for good flying qualities. The maximum and minimum forces are unchanged from MIL-F-8785C.

E. GUIDANCE FOR APPLICATION

For airplanes with centerstick controllers, 3.5.10.3 should be applied along with this requirement.

F. DEMONSTRATION OF COMPLIANCE

In performing the maneuvers required by 3.5.9.1, the minimum and maximum forces necessary to meet the 3.5.9.1 requirements must be found.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.5.10.3 Roll axis control sensitivity

A. REASON FOR REQUIREMENT

The roll response to roll control force inputs, expressed in terms of roll sensitivity, is specified for stick-controlled Class IV airplanes.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.4.1.3

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.10.3 Roll axis control sensitivity. The roll control force gradient for stick-controlled Class IV airplanes shall have the following characteristics: ________________. In case of conflict between the requirements of 3.5.10.3 and 3.5.10.2, the requirements of 3.5.10.3 shall govern.

Recommended values are given in Table 1.

TABLE 1 (3.5.10.3)

RECOMMENDED MAXIMUM ROLL CONTROL SENSITIVITY

LEVEL	FLIGHT PHASE CATEGORY	MAXIMUM SENSITIVITY (deg in 1 sec)/lb
1	A	15.
	C	7.5
2	A	25.
	C	12.5

D. RATIONALE BEHIND REQUIREMENT

The roll power requirements of 3.5.9.1, in combination with the roll forces of 3.5.10.2, effectively specify roll control sensitivity except for aircraft with stick shaping networks such as discussed in "Guidance for Application." This paragraph is intended to place an absolute, firm upper limit on gradients for Class IV airplanes to prevent excessive sensitivity, which can easily result in a lateral PIO.

E. GUIDANCE FOR APPLICATION

Acceptable roll performance and force characteristics should result in easily meeting the conditions of this requirement. Effects of non-linear force deflection characteristics should 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 produces unacceptably high local sensitivity for small inputs. Special considerations are required for roll command augmentation systems to circumvent this problem. These take the form of nonlinear stick shaping, stick filters, and minimizing the lateral augmentor gains. There has been insufficient analysis of the data to write a quantitative criterion at this time. However, the following paragraphs present considerable guidance for roll CAS systems.

1. Roll Command Augmentation Systems

The elements of a command augmentation system (CAS)* are shown in Figure 1. A roll rate CAS utilizes an effective feedforward so that pilot control inputs are compared directly to actual roll response. Such CASs, as they are used today, can be limited in authority with parallel direct links (e.g., the F-14, F-15, F-18, and B-1), or full-authority with high command gains (e.g., the F-16). The latter are the more interesting from a handling qualities standpoint, though examples of both types will be reviewed here.

*In the past CAS has also been referred to as control, rather than command, augmentation systems. These terms are identical.

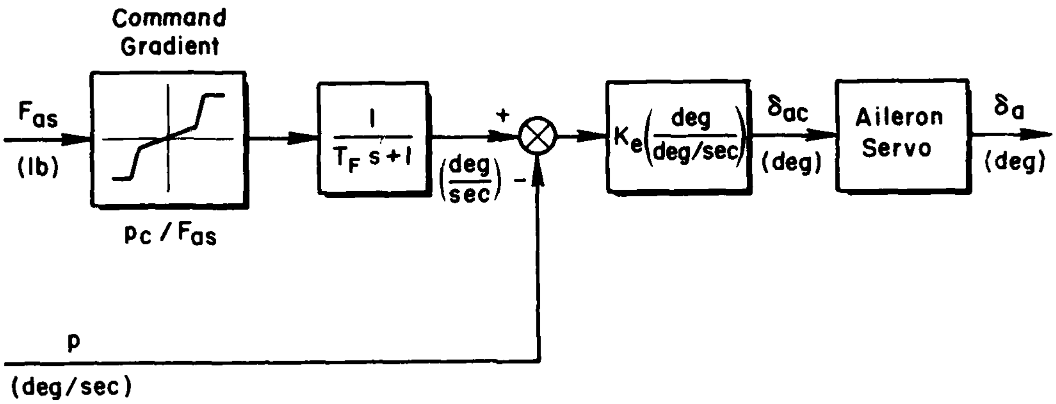


Figure 1 (3.5.10.3) Block Diagram Representation of Full-Authority Roll Rate Command Augmentation Systems

Flight test and operational experiences with Class IV airplanes with high-authority roll CAS have been very promising (References 123-125, 263-265): responses to command inputs are sharp and rapid; precision controllability is excellent; hands-off operation is improved. However, some distinct problems have arisen as well: oversensitivity to small control inputs; overcontrol with large inputs; pilot-induced oscillations; and the phenomenon known as "roll ratcheting." The causes of and cures for these shortcomings will be discussed.

a. Gradient Shaping

Experience with roll rate CAS has shown that a key element for acceptable handling qualities is the gradient between commanded roll rate and stick force, p_c/F_{as} (see Figure 1). High-gain, high-authority systems have had problems with extreme sensitivity for small inputs and inadequate roll performance with large inputs. The 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 in Figure 2. Experience with a limited number of CAS systems has shown that command networks which fall

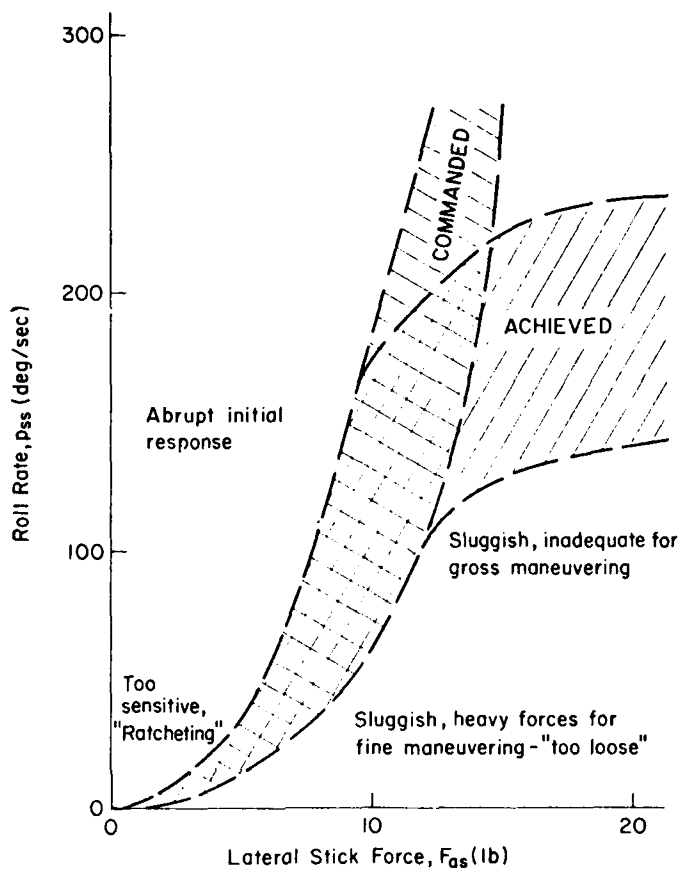


Figure 2 (3.5.10.3). Range of Acceptable Nonlinear Roll Command Shaping Networks Based on Flight Tests (Class IV Aircraft, Flight Phase Category A, Right Roll)

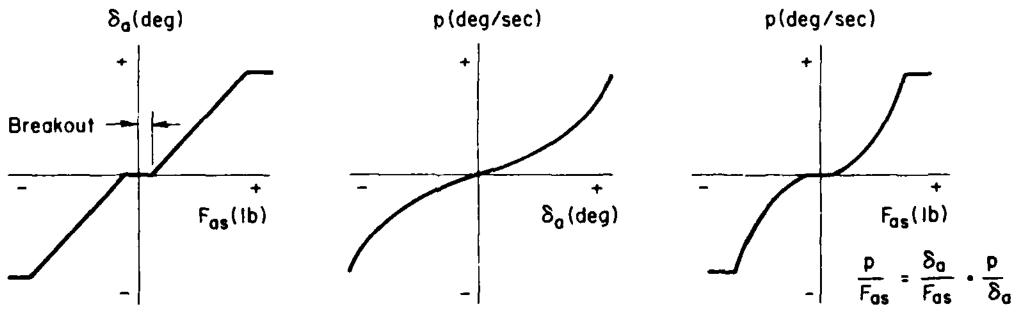
within the range shown in Figure 2 will have acceptable response properties, as long as the requirements and recommendations in Para. 3.5.1.1.1 are satisfied. Figure 2 also reflects the range of actual maximum roll rates achieved for these command networks. In general, the commanded roll rates were obtained for small inputs (on the order of 1/2 stick or less), but larger force inputs generally did not produce the commanded rates. This is a result of intentional design requirements, i.e., if the aircraft can achieve larger roll rates than commanded with full control input, it may "hunt" the commanded rate by overshooting initially.

Such an overshoot and possible oscillations about the commanded roll rate could be uncomfortable to the pilot.

In almost all high-performance augmented airplanes some amount of yaw damping is provided. This has the effect of enhancing roll performance by minimizing undesirable yawing motions. While yaw dampers are important to the aircraft to be discussed here, their effects on handling qualities and performance will be considered to be separate from the roll CAS systems under consideration.

b. Roll Responses for Conventional Aircraft

It is interesting to consider the reasons for the parabolic command gradient shaping shown in Figure 2. 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). This is sketched in Figure 3a. 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. This can be viewed as a nonlinear deflection/response characteristic, sketched in Figure 3b. The result of these force/deflection and deflection/response characteristics is a parabolic force/response curve, shown in Figure 3c. As an example, Figure 4 shows



a) Force/Deflection Gradient *b) Deflection/Response Characteristics* *c) Resultant p/F_{as} Characteristics*

Figure 3 (3.5.10.3). Roll Rate Response for Conventional Aircraft the p_{max}/F_{as} curves for three aircraft, taken from flight or wind tunnel/flight test results. Two of the three lie within the p_c/F_{as} gradient range identified in Figure 2 as acceptable for CAS systems. Parabolic p_c/F_{as} networks, therefore, artifically supply to the pilot what aircraft without CAS have naturally.

c. Roll CAS Gradients

Evolution of the F-16 CAS shaping network is a valuable lesson. Figure 5 illustrates the history of the CAS design (in which, it must be remembered, the unique characteristics of the near-isometric sidestick controller undoubtedly played a significant role). Simulations of the YF-16 (Reference 124) prior to first flight produced a very steep p_c/F_{as}

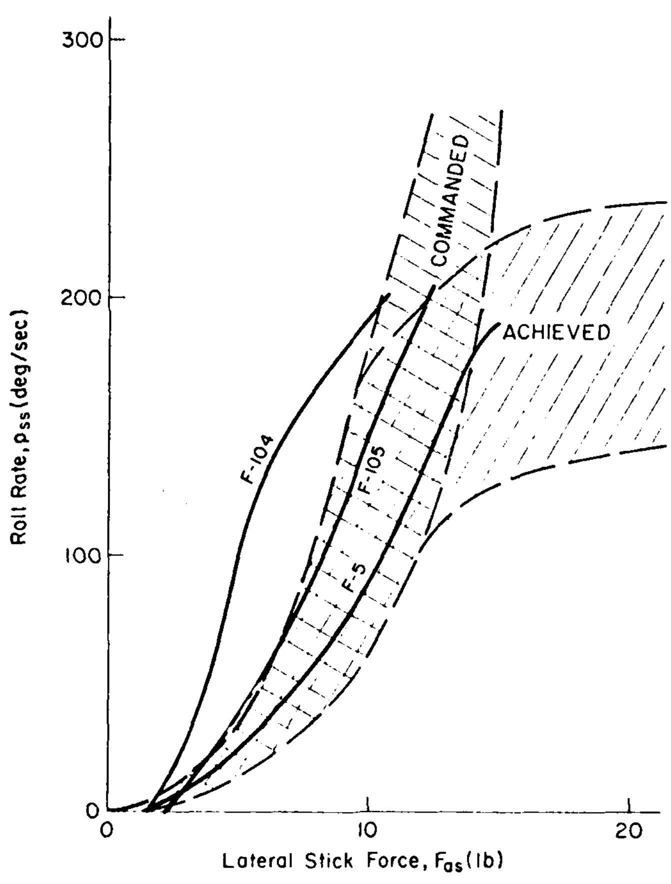


Figure 4 (3.5.10.3). Comparison of p_{max}/F_{as} for Several Conventional Class IV Aircraft with CAS Curves of Figure 2

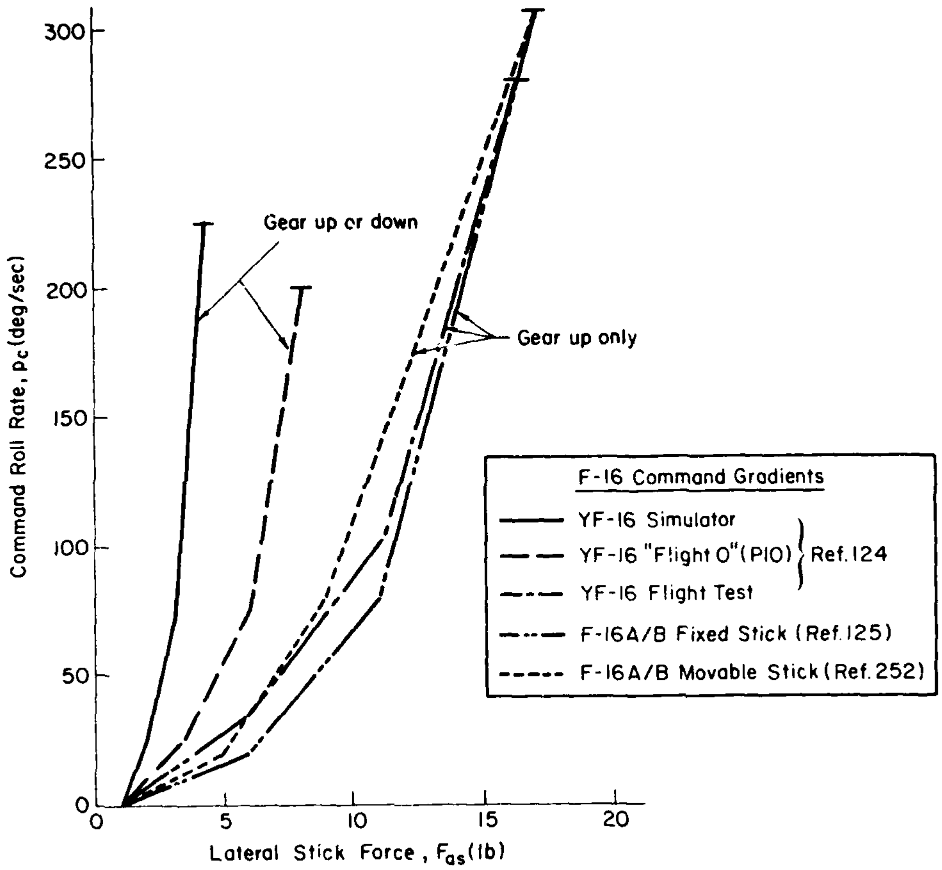


Figure 5 (3.5.10.3). Evolution of the F-16 CAS Shaping Network

gradient with maximum stick forces of 4 lb. 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 pilot-induced oscillation (PIO) was encountered on the prototype YF-16 after the aircraft inadvertently became airborne (Figure 6). The pilot, committed to fly, was then able to "back off" on his control gain, and the PIO stopped,

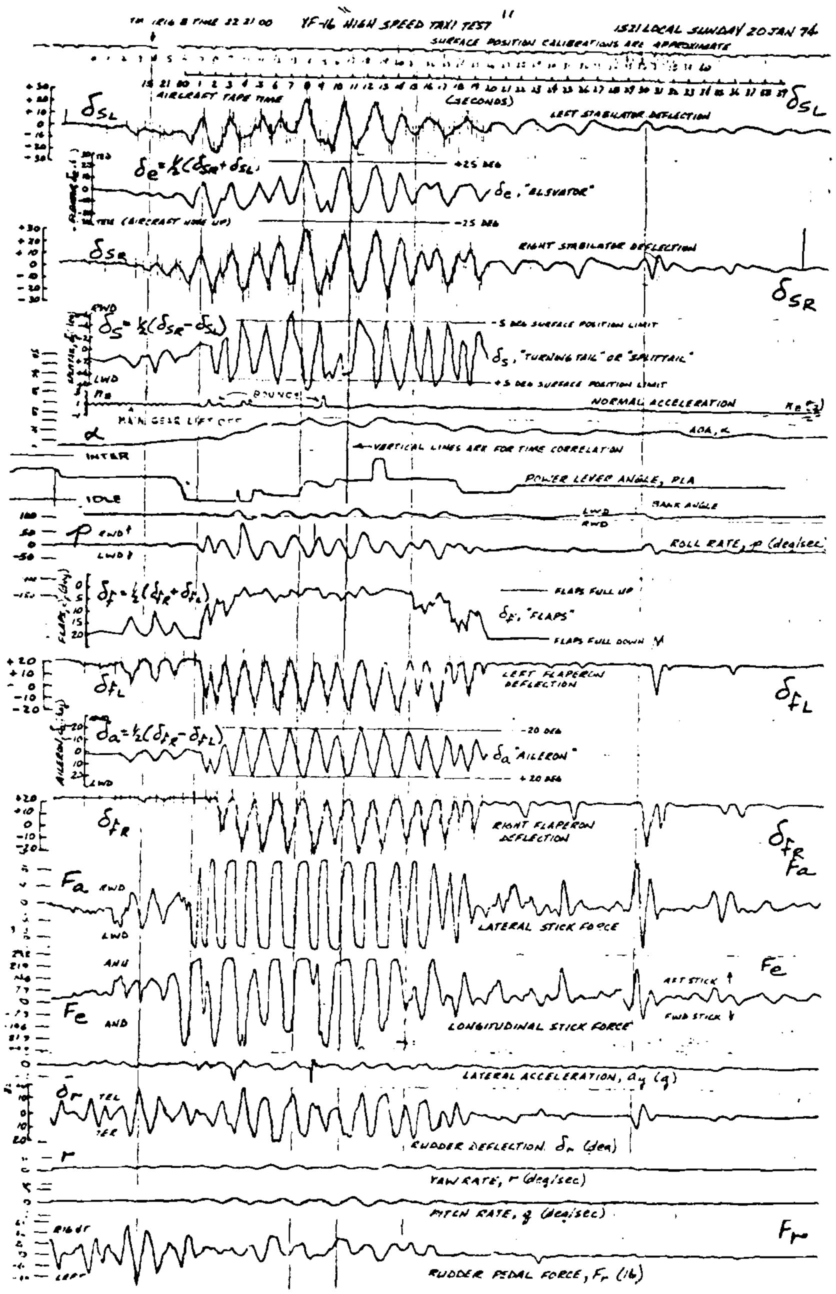


Figure 6 (3.5.10.3). YF-16 PIO Due to Excessive Lateral Stick Sensitivity after which a normal landing was made. This PIO was directly traceable to excessive stick sensitivity around zero, and after the flight (dubbed "Flight 0") the stick sensitivity was reduced further and the PIO tendency disappeared. The final F-16A/B (fixed-stick) (Reference 125) network was reduced even more (see Figure 5). With the latest F-16 variable roll prefilter (Reference 252), it has been possible to increase the CAS gradient somewhat.

The roll performance of the YF-16, shown in Table 2 and Figure 7, is comparable with present generation USAF fighter aircraft and is seen to be very good in comparison to the roll control power requirements of Tables 3 (3.5.9.1) and 4 (3.5.9.1). The final YF-16 roll command gradient of Reference 124 (Figure 5) 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."

TABLE 2 (3.5.10.3)

YF-16 ROLL PERFORMANCE CHARACTERISTICS
[From Reference 124]

$\bar{q}$ (psf)	V_c (KCAS)	M	H_c (ft)	α_s (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

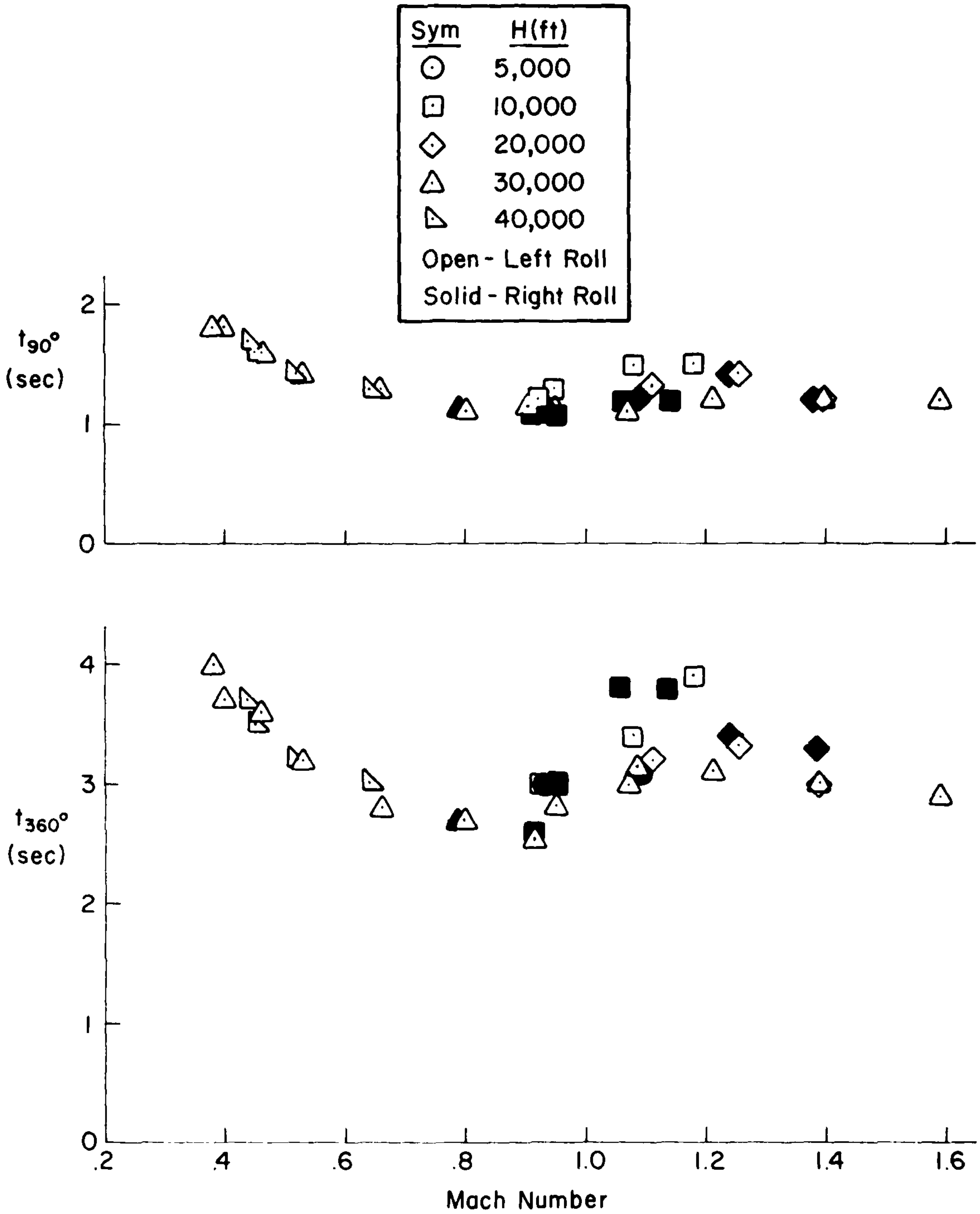


Figure 7 (3.5.10.3). YF-16 Rolling Performance Cruise Configuration; Reference 124

F-16 roll performance in 360 deg rolls for the CR (cruise) configuration (Figure 8) compares quite well with the Table 4 (3.5.9.1) requirements. The F-16 was Level 2-3 for most of the low-speed range in Power Approach (Figure 9). According to Reference 125, "the pilots were pleased with the F-16A/B CR configuration roll performance. PA configuration roll performance was acceptable." Hence, the nonlinear stick shaping was reasonably successful in achieving acceptable large amplitude rolling performance without excessively compromising the small amplitude precision tracking characteristics.

d. Roll Ratcheting

As mentioned earlier, concerns from the piloting point of view for roll CAS systems have been described variously as oversensitivity to small inputs, overcontrol or sluggishness for large inputs, and "roll ratcheting." All of these can create pilot-induced oscillations (PIOs), and, as we shall see, ratcheting is the hardest to identify, isolate, and correct. The other problems can be solved by relatively simple means.

Roll ratcheting has been reported as occurring on most CAS-equipped aircraft, including the F-4 SFCS (Reference 123), YF-16 (Reference 124), F-16 (Reference 125), and A-7D DIGITAC (Reference 265). It was also experienced during the Calspan Lateral High-Order System (LATHOS) program of Reference 258. All of these aircraft will be discussed in detail in the following paragraphs.

An example of roll ratcheting encountered on the DIGITAC (Reference 265) is shown in Figure 10. The ratchet was encountered during a series of bank-to-bank maneuvers. The oscillations exhibit limit cycles at a frequency of about 18 rad/sec. The roll CAS is presented in Figure 11; ratcheting was experienced with p_c/F_{as} Curve 1 in Figure 11. The fact that roll ratcheting occurred for this case is evidence that stick shaping is not a cure for this problem. The reason is that the stick sensitivity is reduced only around zero, allowing ratcheting to occur when the lateral stick is non-zero, such as in Figure 10. Figure 11 and Table 3 document several of the CAS networks flown on the DIGITAC in

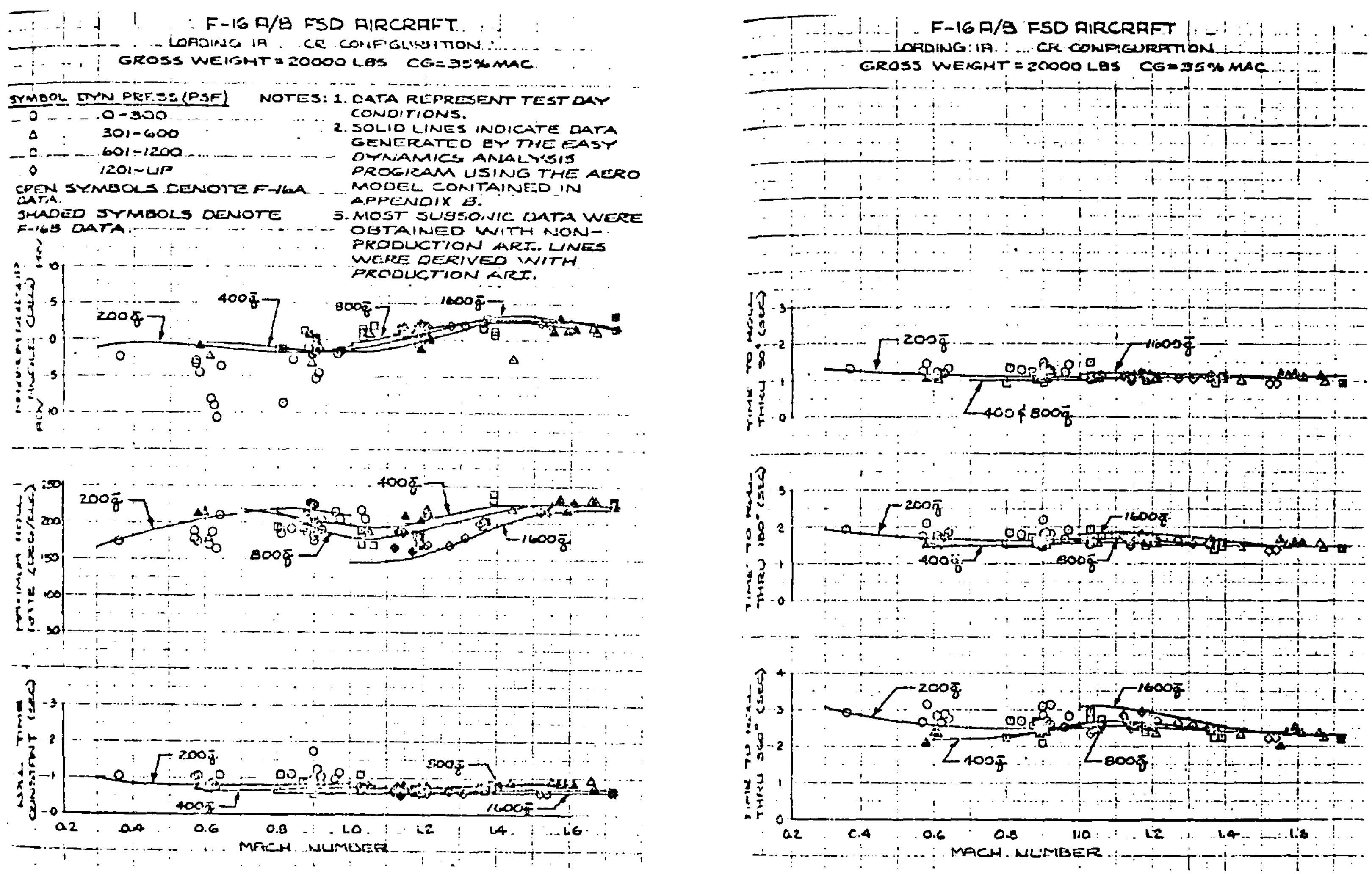


Figure 8 (3.5.10.3). Roll Performance Summary
(From Reference 125)

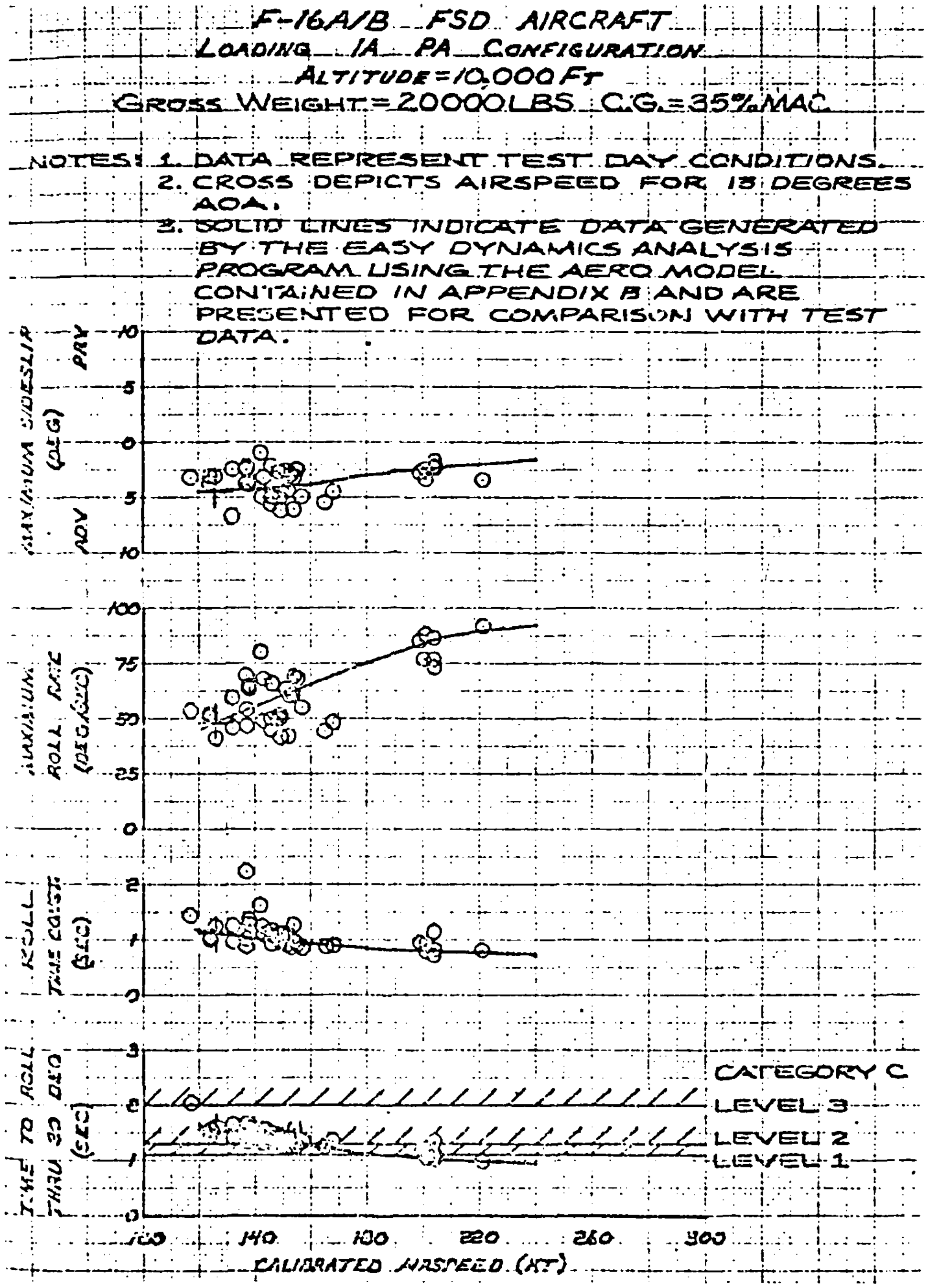


Figure 9 (3.5.10.3). Roll Performance Summary
(From Reference 125)

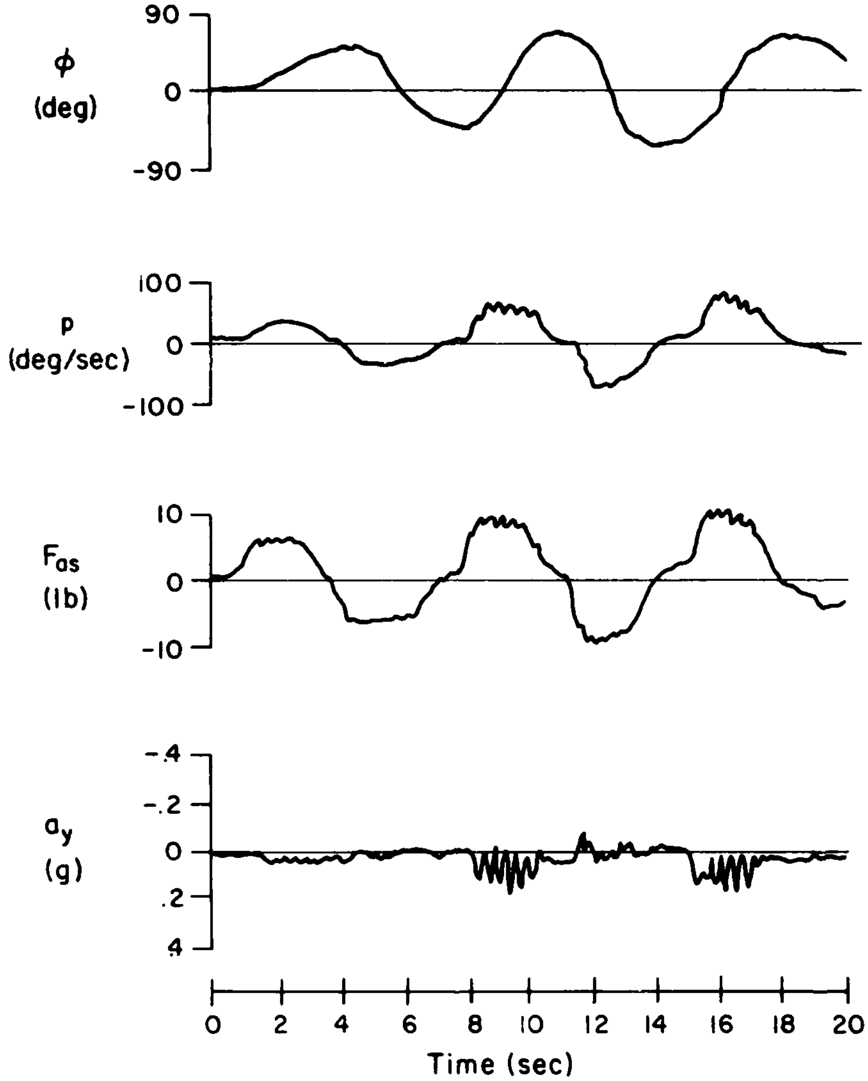


Figure 10 (3.5.10.3). Roll Ratchet During Banking Maneuvers (DIGITAC, Reference 265) h = 20,000 ft, M = 0.75

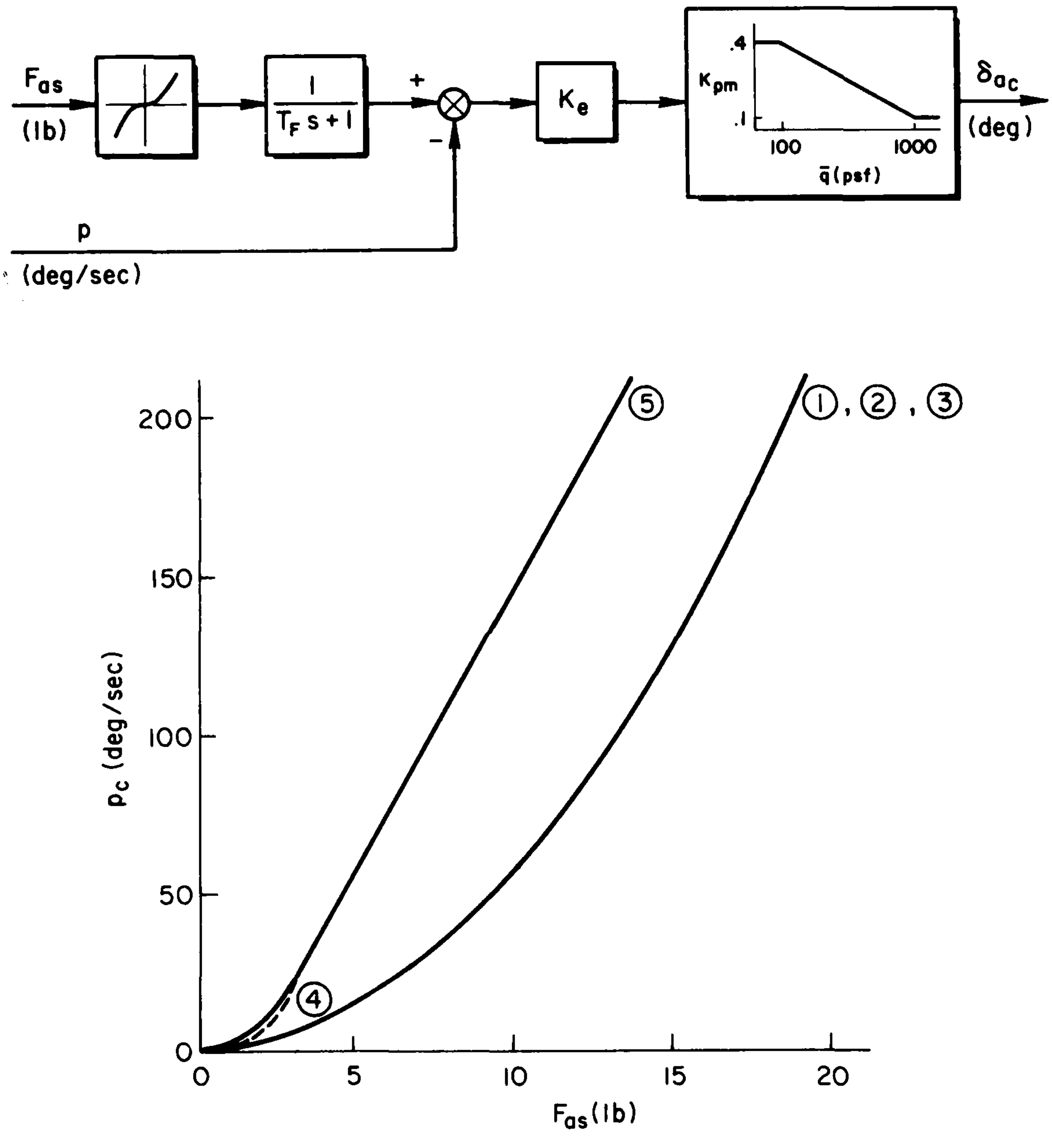


Figure 11 (3.5.10.3). Evolution of Roll CAS Network for YA-7D DIGITAC (Reference 265). See Table 3 for Values of T_F, K_e

TABLE 3 (3.5.10.3)

DESCRIPTIONS OF YA-7D DIGITAC 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 10)
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 11.

developing an optimum CAS. This is an excellent review of all the elements of a CAS, since several gradients, prefilter lags, and error gains were evaluated.

The CAS which produced the Figure 10 ratcheting had a prefilter lag at 10 rad/sec and an error gain K_e = 1.0 (Table 3). As Table 3 reflects, CAS System 2 involved only a reduction by one-half in K_e and eliminated the roll sensitivity. However, with the sensitivity reduced, the pilots were then aware that the steady-state roll response was much too low. With K_e = 1.0 and the prefilter lag reduced from 10 to 3 rad/sec (CAS 3), the roll sensitivity was reduced, although not enough. It was clear from CASs 1-3 that: a) the roll response for large inputs was too low; b) a reduction 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 11), including a 0.75 lb breakout, and lower T_F and K_e

(Table 3). 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) was found to be best for all-around response.

Another example of roll ratcheting, experienced on the YF-16 (Reference 124), is given in Figure 12a. 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. As the second roll on Figure 12 shows, the pilot was later able to perform a roll without encountering ratcheting. According to Reference 124, "Full-authority rolls did not involve the oscillation."

The roll ratcheting experienced on the YF-4E SFCS (Reference 123) was of a somewhat different character than that of the YF-16 or DIGITAC, as it occurred primarily during fine maneuvering rather than during large-input rolls. A representative time history is not available, but Reference 123 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 in Reference 123 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 of roll ratcheting we will examine 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, Reference 258) represents an excellent data base for detailed discussion on many of the handling quality concerns for modern aircraft.

While the LATHOS tests were not intended to investigate the handling qualities of command augmentation systems per se, mechanization of the lateral control effectiveness was such that it may be considered a CAS. That is, the NT-33 variable-stability system was devised to command a

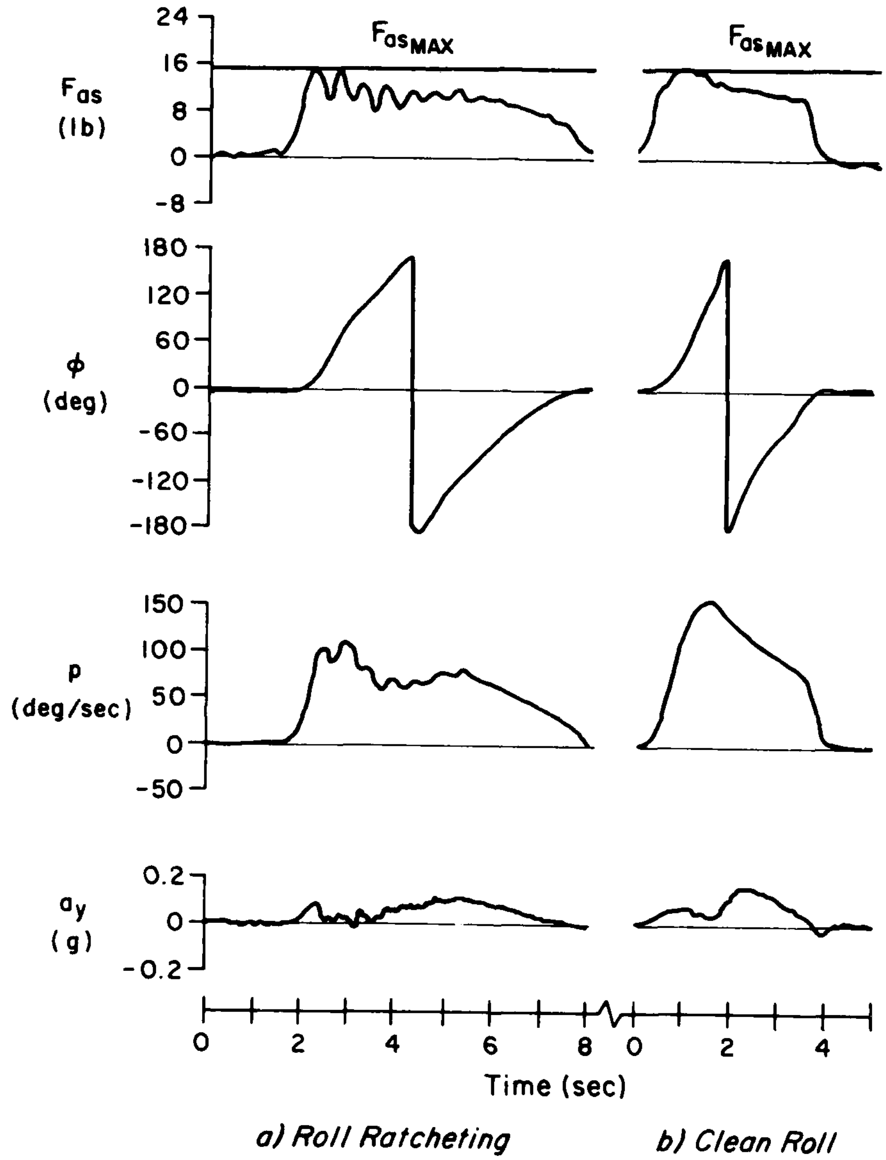


Figure 12 (3.5.10.3). Steady Rolls on YF-16 (Reference 124). The Roll in (b) was Performed 32 seconds after (a) and was Satisfactory. h = 10,000 ft, M = 0.80

certain ratio of steady-state roll rate to stick force (p_{ss}/F_{as}) -- a pseudo-CAS network. Additionally, the roll rate response was devised with a neutral spiral ($T_s \doteq \infty$) and $\omega_\phi/\omega_d \doteq 1$. Thus the lateral-directional response was effectively first order in combination with a variable prefilter and a time delay to account for actuator lags, i.e.,

$$\frac{p}{F_{as}} \doteq \frac{T_F L'_{F_{as}} e^{-.028s}}{(s + 1/T_R)(s + 1/T_F)}$$

For the configurations with ratcheting, $T_R = 0.15$ sec. Figure 13 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. In some instances the pilots complained about the "buzz." It is not known whether this "buzz" was common to all the $T_R = 0.15$ sec cases, or whether it influenced pilot ratings for these cases.) The ratcheting is best seen in the $\dot{p}$ and F_{as} traces, at a frequency of about 16 rad/sec.

Figure 14 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 2. No breakout or friction forces were mechanized. Several values of prefilter lag, T_F, were used with Configurations 5-2 and 5-3. Figure 15 shows the influence of prefilter time constant on pilot ratings.

For Configuration 5-2 ($p_{ss}/F_{as} = 10$), the roll response for small inputs lies well above the acceptable range in Figure 14, 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, 7). These ratings and comments were for $T_F = 0.025$ sec. However, increasing T_F did little to improve the ratings (see Figure 15) because of the inadequate response. Pilot comments reflect this: "Gross acquisition sluggish...Sensitivity low...Took a lot of force."

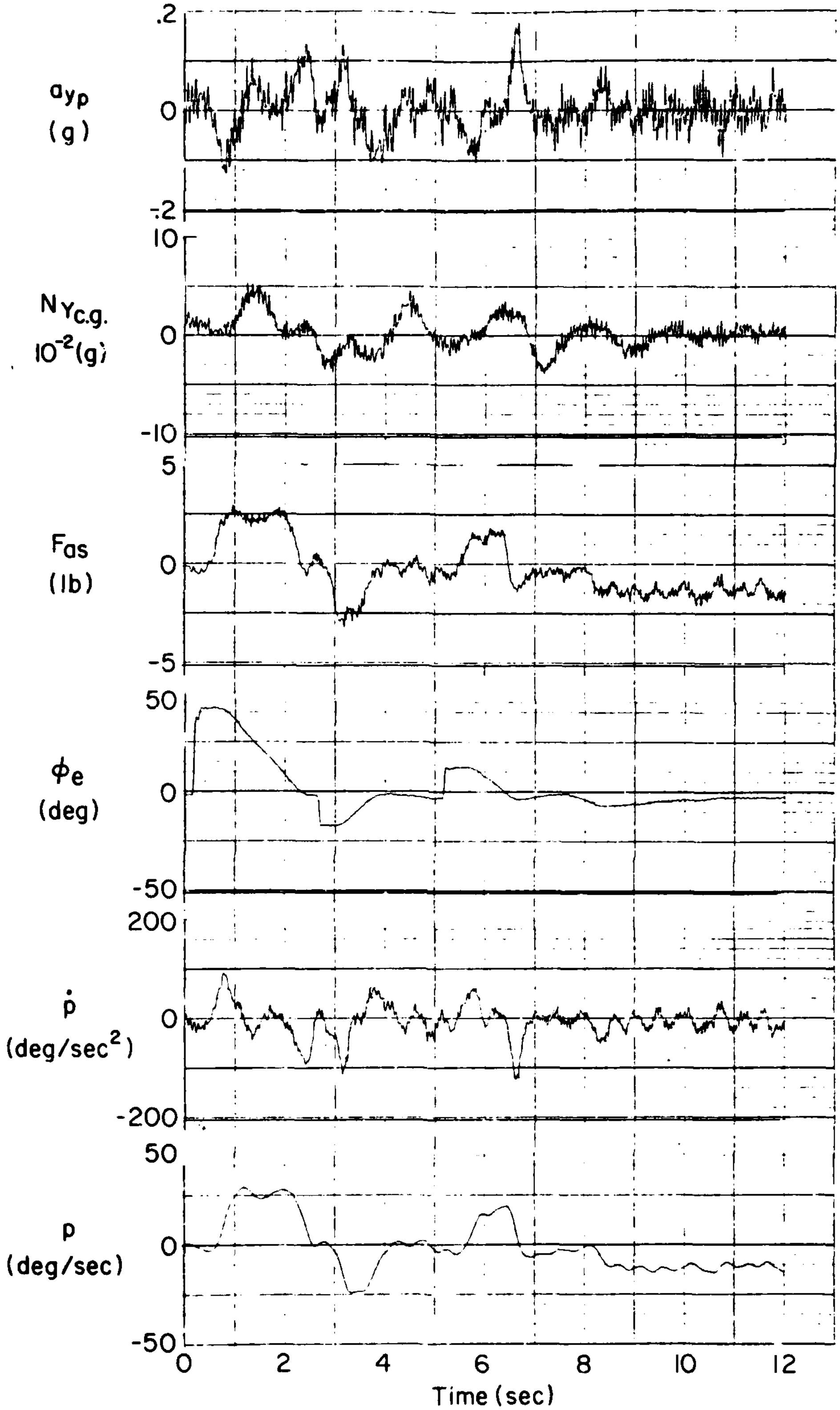


Figure 13 (3.5.10.3). Roll Ratcheting Experienced on LATHCS (Reference 258). Configuration 5-2

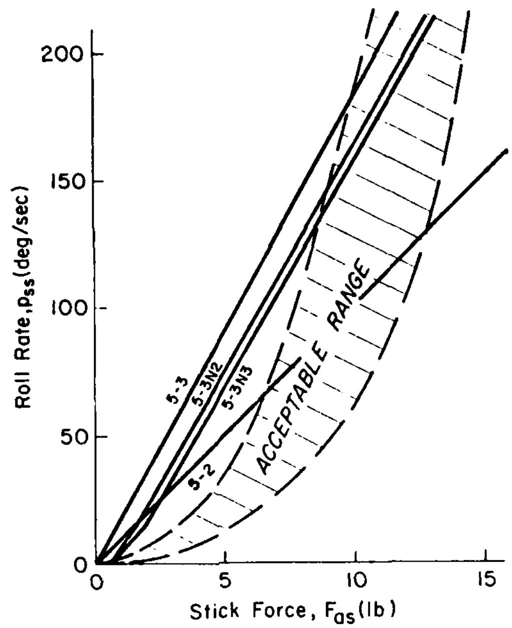


Figure 14 (3.5.10.3). Roll Gradients for LATHOS Configurations 5-2 and 5-3 (T_R = 0.15 sec) Compared with Acceptable Range from Figure 2

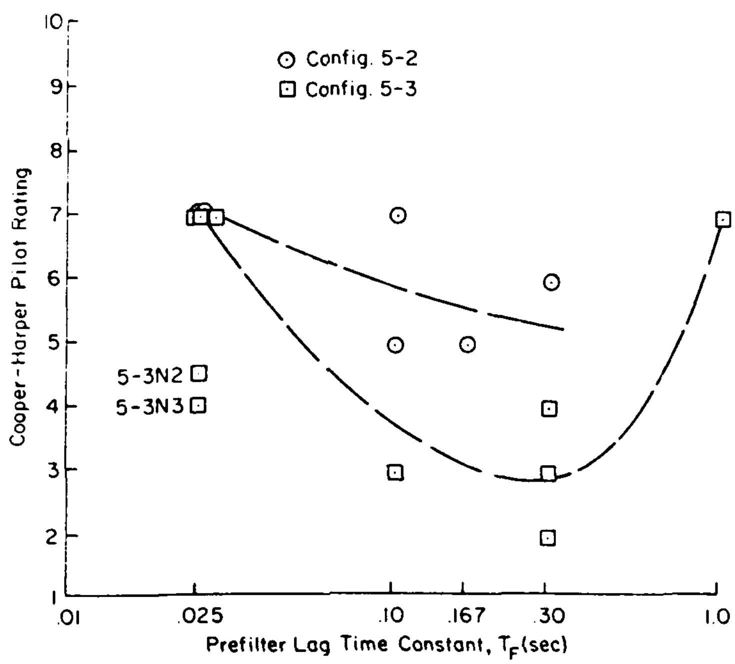


Figure 15 (3.5.10.3). Influence of Prefilter Lag on Pilot Ratings (Reference 258). T_R = 0.15 sec

For Configuration 5-3 (p_{ss}/F_{as} = 18, Figure 14) 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 15) 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 3.33-10 rad/sec (T_F = 0.10 and 0.3, Figure 15) 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 response to aggressive inputs."

Finally, two nonlinear gradients (5-3N2 and 5-3N3, Figure 14) 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 Section 3.2.1.1, stick prefilters are a major contributor to overall effective time delay, τ_e. For example, Figure 9a (3.2.1.1) shows that a 3 rad/sec prefilter contributes about 0.1 second 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 should not be looked on as a final solution.

e. <u>Guidance for Acceptable Sensitivity</u>

The following guidelines are offered to obtain adequate roll control power for large control inputs without incurring excessive abruptness and/or roll ratcheting for small inputs, as well as excessive time delay and hence lateral PIO tendencies.

- Utilize nonlinear lateral stick shaping in the region specified in Figure 2.

- Avoid excessively large values of $1/T_R$ by minimizing the gain on the roll rate feedback. Figure 14 (3.5.1.1.1) suggests $1/T_R < 4$.

- Stick filters will eliminate roll ratcheting. However, the break frequency should be carefully evaluated in terms of time delay [see Figure 1 (3.5.1.1.5)].

1. Program of Reference 46 (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 resulting "optimum" sensitivities are presented in Figure 16. (The spring rate was F_{as}/δ_{as} = 3.81 lb/in.) Only "optimum" sensitivities are shown because most configurations were Level 2 or worse in dutch roll or roll mode characteristics. Many of the points shown in Figure 16 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.

2. Program of Reference 38 (Flight Phase Category A)

In this program, which utilized a rolling simulator and several fighter aircraft, a parametric variation of $L_{\delta_a}\delta_{a_{max}}$ and T_R was made to determine lateral control requirements for fighter aircraft performing fighter missions. As such, the results of this program should be directly comparable to the results of the program of Reference 46. The Reference 38 data are plotted along with the Reference 46 data in Figure 16. 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 lb/in. It can be seen from Figure 16 that the data points of constant pilot rating lie approximately along lines of constant ϕ_1 (bank angle in one second). This suggests that, at least for Class IV aircraft performing fighter missions, roll response sensitivity can be best expressed in terms of ϕ_1/F_{as} (bank angle in 1 second per pound). It can further be seen that the data points from Reference 38 for Level 1 roll response sensitivity lie along a curve of $\phi_1/F_{as} \approx 15$ deg/lb; for

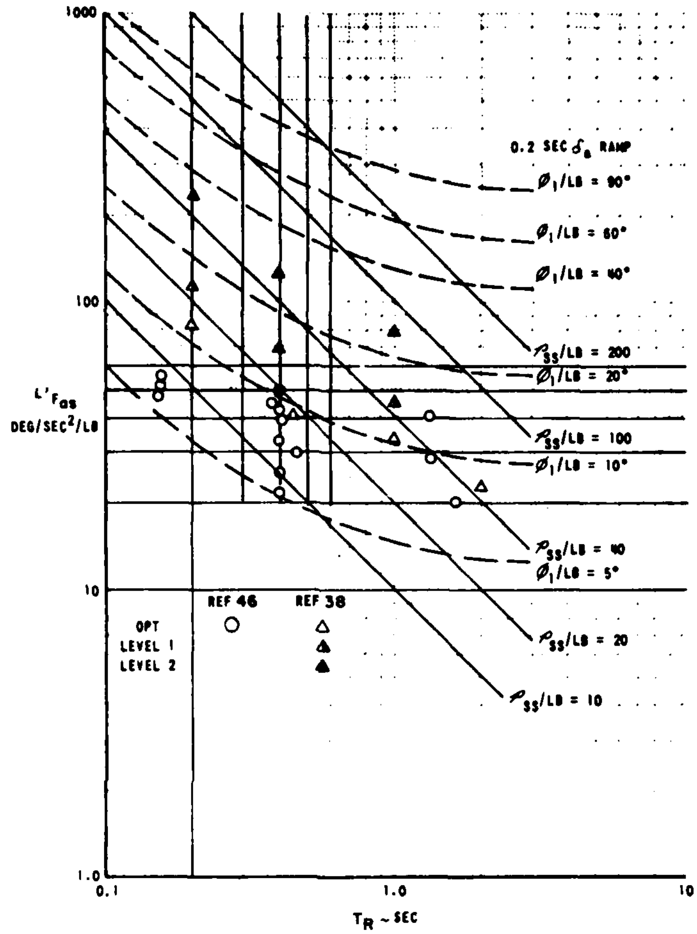


Figure 16 (3.5.10.3). Flight Phase Category A - Force Sensitivity

Level 2 flying qualities, $\phi_1/F_{as} \approx 25$ deg/lb. Both sets of data indicate that for optimum roll response sensitivity ϕ_1/F_{as} should be between 10 and 20 deg/lb. A possible exception is indicated by the low-T_R data of Reference 46, where somewhat lower optimum roll response sensitivities were selected by the pilots.

Figure 17 shows actual data from the tests of Reference 38. The pilot ratings from the moving-base simulation (Figure 17b) clearly support the gradient limits of Table 1. Differences between the fixed and rolling simulator results are presumably due to the additional accelerations sensed by the pilots in the rolling simulator. These results

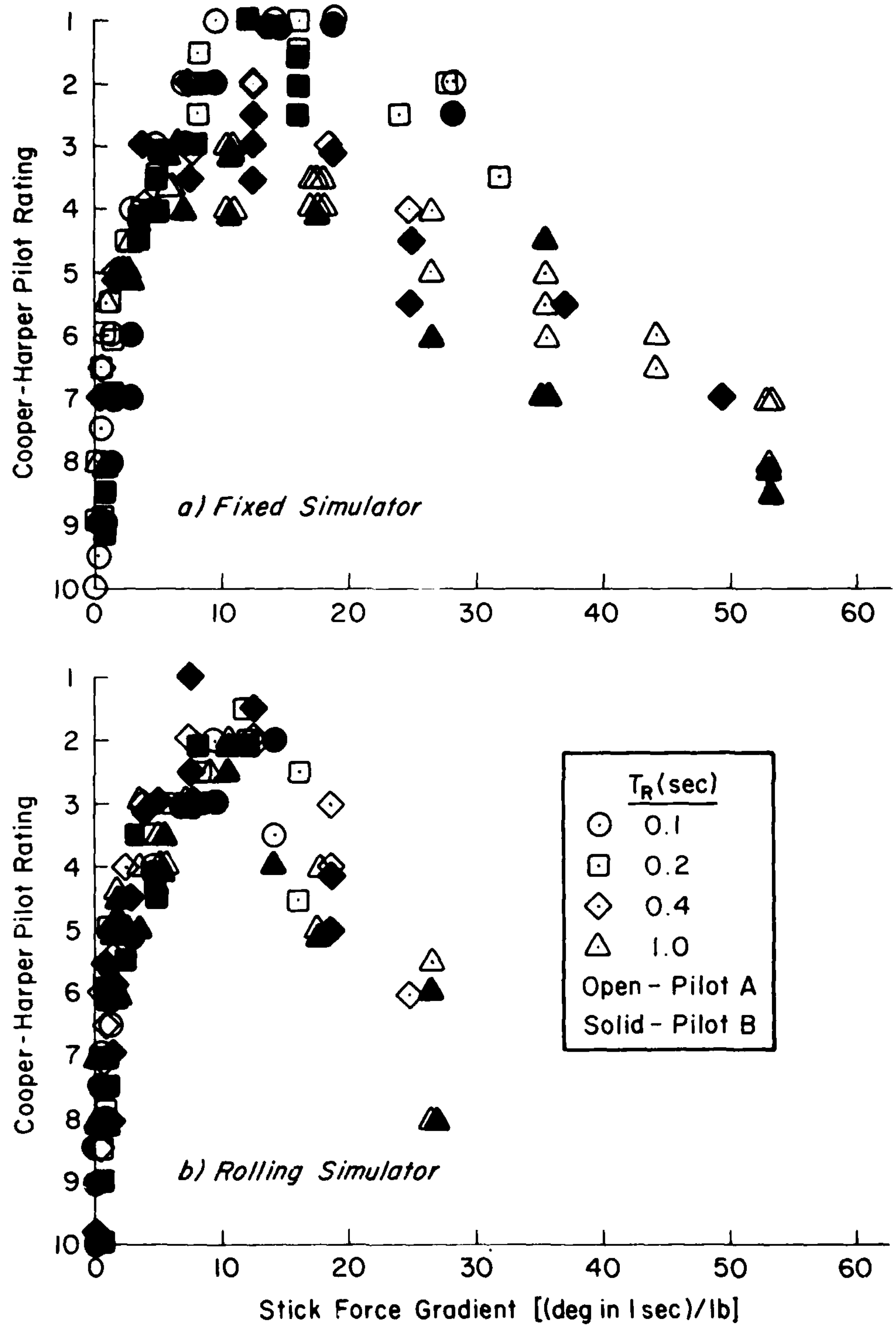


Figure 17 (3.5.10.3). Pilot Ratings from Reference 38
(Category A Flight Phase)

indicate that compliance with this requirement should involve a moving-base simulator as a minimum.

3. Program of Reference 44 (Flight Phase Category C)

In order to compare the fighter-airplane data for up-and-away flight with data for the landing approach, consider the in-flight data of Reference 44 shown in Figure 18. From comparison of Figures 16 and 18 it can be seen that the optimum roll response sensitivity, maximum satisfactory roll response sensitivity (Level 1), and maximum acceptable roll response sensitivity (Level 2), in terms of rolling accelerations per force for the landing approach, are about half that for respective levels of flying qualities for Flight Phase Category A. This is reflected in Table 1.

H. LESSONS LEARNED

Some flight test data are available for several current aircraft. These are compared with the Table 1 boundaries below.

The F-5 roll gradients in both Category A and C flight fall well within the Level 1 limits, as Figure 19 (from Reference 131) shows. Similarly, gradients for the F-18A in Category A (Figure 20) and C (Figure 21) flight phases are within the Level 1 limits.

As discussed in "Guidance for Application," airplanes 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-18 has a parabolic stick shaping network and the data shown in Figures 20 and 21 represent sensitivities for small stick deflections such as used for precision tracking.

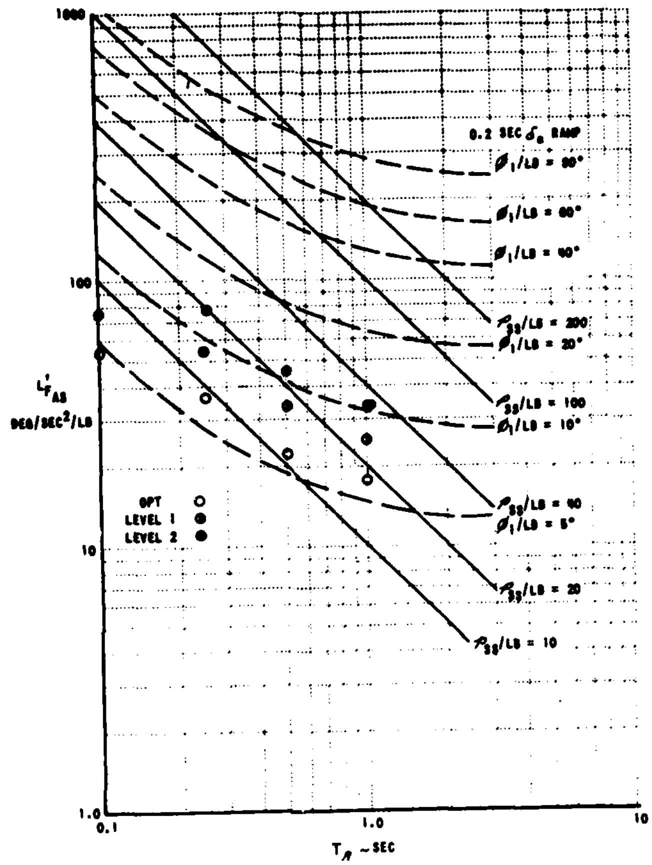


Figure 18 (3.5.10.3). Flight Phase Category C-Force Sensitivity (From Reference 44)

Configuration	Altitude (ft)	Symbol
o———o	10,000 20,000	⊙ △

Note: Flagged symbol indicates Cat. C

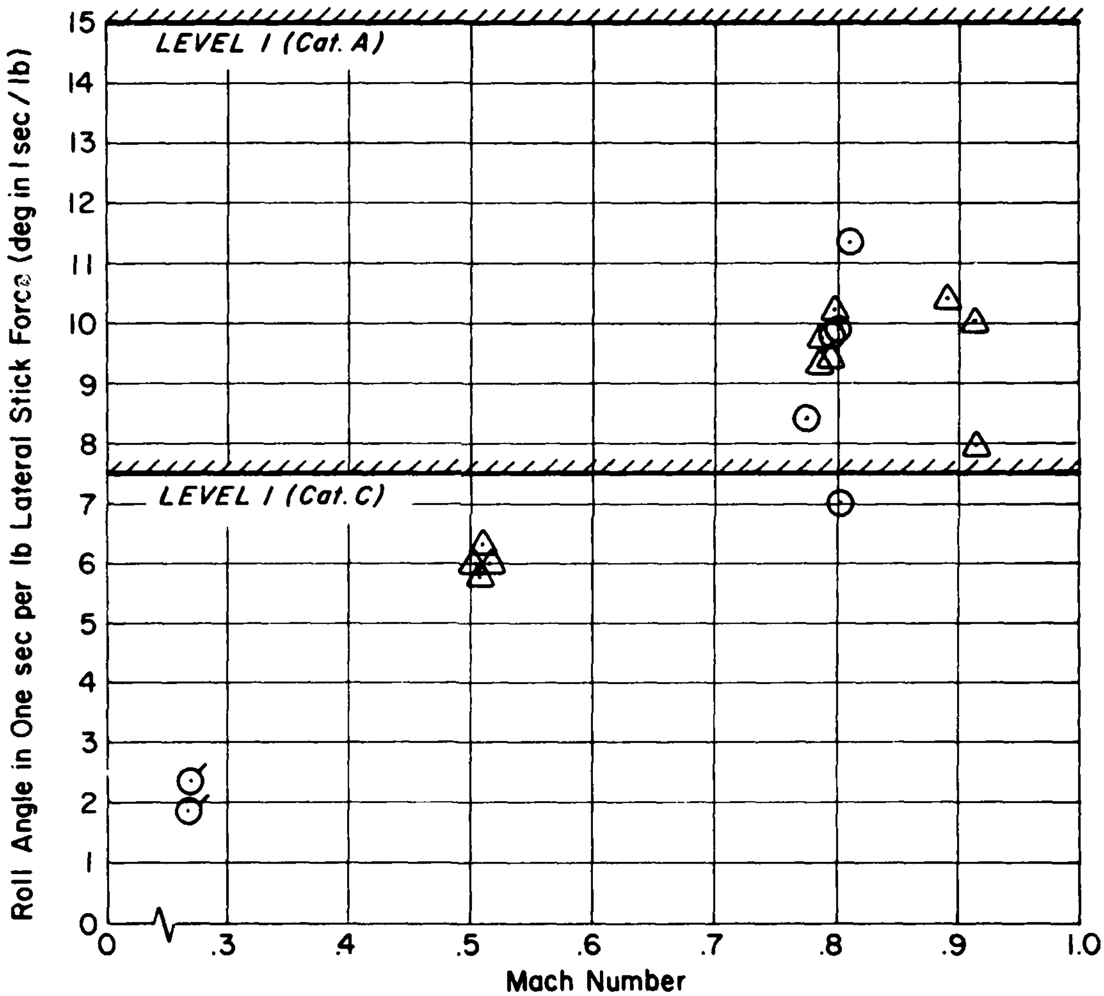


Figure 19 (3.5.10.3). Roll Response Per Pound for F-5
(From Reference 131)

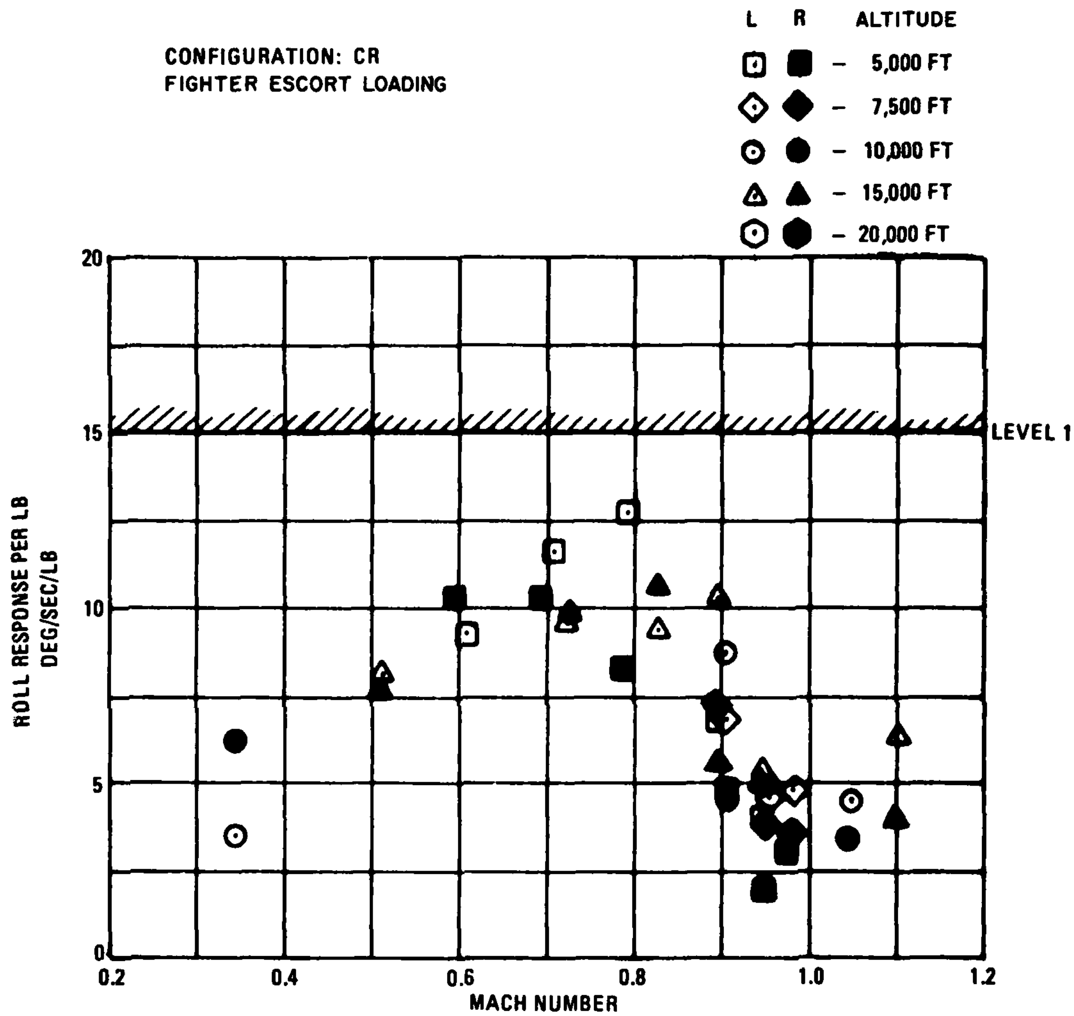


Figure 20 (3.5.10.3). Roll Response Per Pound Versus Mach for F-18A (Flight Test Data from Reference 134)

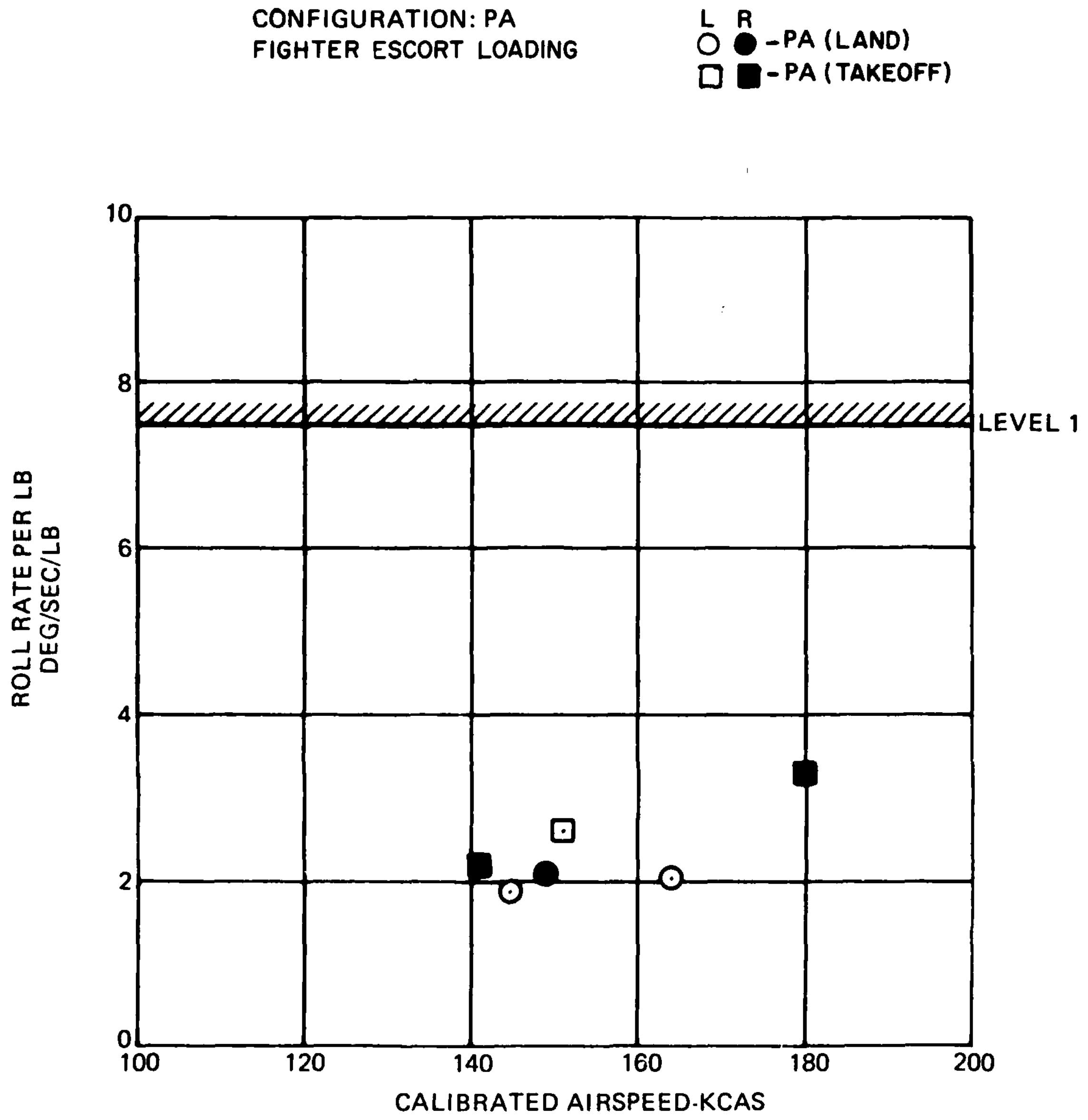


Figure 21 (3.5.10.3). Roll Sensitivity Characteristics in Configuration PA for F-18A (Flight Test Data from Reference 134)

3.5.10.4 Roll axis control forces — control centering and breakout forces

A. REASON FOR REQUIREMENT

This requirement is included to insure that the centering and breakout characteristics of the roll controller are acceptable to the pilot.

B. RELATED MIL-F-8785C REQUIREMENT

3.5.2.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.10.4 Roll axis control forces — control centering and breakout forces. Lateral controls should exhibit positive centering in flight at any normal trim conditions.

The combined effects of centering, breakout force, damping, and force gradient shall not produce objectionable flight characteristics.

Breakout forces, including friction, preload, etc., shall be within the following limits: ______________________.

Recommended breakout force limits are given in Table 1.

TABLE 1 (3.5.10.4)

RECOMMENDED ALLOWABLE BREAKOUT FORCES (lb)

LEVEL	CONTROL	CLASSES I, II-C, IV		CLASSES II-L, III	
		MINIMUM	MAXIMUM	MINIMUM	MAXIMUM
1 and 2	Sidestick	1/2	1		
	Centerstick	1/2	2	1/2	4
	Wheel	1/2	3	1/2	6
3	Sidestick	1/2	4		
	Centerstick	1/2	4	1/2	8
	Wheel	1/2	6	1/2	12

D. RATIONALE BEHIND REQUIREMENT

A discernible neutral point (or trim or equilibrium point) should always be provided in manual pitch, roll, or yaw controllers. That is, if the pilot chooses to release a control it should return to a neutral or trim state. If no cues are provided the pilot will be forced to manually search for such a trim condition. This can lead to poor maneuvering control, or, in the extreme, to pilot-induced oscillations.

The sidestick breakout forces in Table 1 are based upon recommendations of Reference 23.

E. GUIDANCE FOR APPLICATION

While this paragraph does not require absolute centering, the tendency for positive centering should always be detectable. 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.

F. DEMONSTRATION OF COMPLIANCE

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.

G. SUPPORTING DATA

None available.

H. LESSONS LEARNED

No information available at this time.

3.5.10.5 Roll axis control forces — free play

A. REASON FOR REQUIREMENT

While some amount of free play may be desirable to prevent oversensitivity to unintended control motions, the free play should not create an objectionably large dead zone.

B. RELATED MIL-F-8785C REQUIREMENT

3.5.2.2

C. STATEMENT OF REQUIREMENT

3.5.10.5 Roll axis control forces — free play. The free play in the lateral controller shall not result in objectionable flight characteristics, especially for small amplitude inputs. Free play should be within the following boundaries: ____________________.

D. RATIONALE BEHIND REQUIREMENT

The requirement prevents design of large dead zones in the roll controller. In normal operations, and especially in high-demand times such as turbulence penetration or air combat, free play can contribute to overcontrol and rapid pilot fatigue.

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.

E. GUIDANCE FOR APPLICATION

This requirement is not intended to eliminate all free play. Free play is often designed into a controller.

F. DEMONSTRATION OF COMPLIANCE

Measurement of free play in flight should be made over the operational load factor and airspeed ranges at the minimum and maximum operational altitudes.

G. SUPPORTING DATA

None available.

H. LESSONS LEARNED

None available.

3.5.10.6 <u>Roll axis control force limits</u>

3.5.10.6.1 Roll axis control force limits — steady turns

A. REASON FOR REQUIREMENT

This requirement is included to limit the pilot effort required to perform coordinated turns.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.2.6

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.10.6.1 <u>Roll axis control force limits — steady turns</u>. It shall be possible to maintain steady turns with the airplane trimmed for wings-level straight flight in either direction with the yaw controls free at the following combinations of bank angle and roll controller force characteristics: __________________________.

The <u>recommended values</u> constitute Levels 1 and 2.

Airplane Class	Bank Angle (deg)
I and II	45
III	30
IV	60

Maximum roll control forces:

Centerstick controller:	5 lb
Wheel controller:	10 lb

D. RATIONALE BEHIND REQUIREMENT

This requirement, in combination with Paragraph 3.6.6.2.2, limits the allowable control forces in steady turns. The objective of the requirement is to insure that only modest roll control forces are required when rudder pedals are not used. The steepness of the turn is a function of airplane Class, to correspond with normal operational use.

G. GUIDANCE FOR APPLICATION

This requirement applies to Levels 1 and 2 only, since it is expected that Level 3 operations would not involve steady, large-bank-angle turns.

F. DEMONSTRATION OF COMPLIANCE

Flight testing at the specified bank angle, at maximum operational altitude and minimum operational velocity, generally presents the most potential for large yawing moments and large effective dihedral.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.5.10.6.2 Roll axis control force limits — dives and pullouts

A. REASON FOR REQUIREMENT

This requirement is included to limit the forces necessary to maintain roll attitude during dives and pullouts.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.8

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.10.6.2 Roll axis control force limits — dives and pullouts. Roll control forces shall not exceed ___ lb in dives and pullouts to the maximum speeds specified in the Service Flight Envelope.

Recommended values:

Propulsion Type	Maximum Roll Control Force (lb)
Propeller	20
Other	10

D. RATIONALE BEHIND REQUIREMENT

As with similar requirements in the yaw axis, this paragraph distinguishes between propeller-driven and all other airplanes because of the normal crossflow effects due to turning propellers.

E. GUIDANCE FOR APPLICATION

The applicability of this requirement is dependent upon specification of dives by the procuring activity.

F. DEMONSTRATION OF COMPLIANCE

If dives are specified as normal operations, flight testing must cover the flight conditions for such dives. Otherwise, some amount of dive and pullout testing should be performed.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.5.10.6.3 Roll axis control force limits — crosswinds

A. REASON FOR THIS REQUIREMENT

This requirement is included to assure that roll control forces in crosswind takeoffs and landings are not unreasonably large.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.7

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.10.6.3 Roll axis control force limits — crosswinds. It shall be possible to take off and land in the crosswinds specified in 3.5.9.3 without exceeding the following roll control forces: ________________.

It is recommended that, as a maximum, roll control forces should be no greater than those specified by 3.5.10.2.

D. RATIONALE BEHIND REQUIREMENT

As it is written, this is simply a method of insuring that crosswind operations do not require more roll control force than normal rolling maneuvers, i.e., that the limiting factor on roll forces will never be wings-level flight in crosswinds.

E. GUIDANCE FOR APPLICATION

None required.

F. DEMONSTRATION OF COMPLIANCE

From a safety aspect, actual takeoffs and landings in crosswinds may be impractical for Level 2 operation. 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. At the Air Force Flight Test Center, for one place, there should be little difficulty in finding appropriate crosswinds for a flight-test buildup.

G. **SUPPORTING DATA**

None.

H. **LESSONS LEARNED**

None.

3.5.10.6.4 Roll axis control force limits — steady sideslips

A. REASON FOR REQUIREMENT

This requirement is included to insure that the amount of roll control force necessary to achieve a reasonable steady sideslip condition is never tiring for the pilot.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.7.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.10.6.4 Roll axis control force limits — steady sideslips. In final approach the roll control forces shall not exceed ___ lb when in a straight, steady sideslip of ___ deg.

Maximum recommended control forces:

Level 1:	10 lb
Levels 2 and 3:	20 lb

Sideslip specified should be same as for Paragraph 3.6.5.1.

D. RATIONALE BEHIND REQUIREMENT

This requirement augments the yaw control power requirement of Paragraph 3.6.5.1 to assure that coordinating roll forces in sideslips are reasonable.

E. GUIDANCE FOR APPLICATION

None required.

F. DEMONSTRATION OF COMPLIANCE

See discussion under Paragraph 3.6.5.1.

G. **SUPPORTING DATA**

None.

H. **LESSONS LEARNED**

None.

3.5.10.6.5 Roll axis control force limits — engine failures after takeoff

A. REASON FOR REQUIREMENT

This paragraph is intended to prevent excessive roll control forces to counter the effects of engine failure after takeoff.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.9.2, 3.3.9.4

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.10.6.5 Roll axis control force limits — engine failures after takeoff. Following a thrust loss from the most critical factor after takeoff the roll control forces shall not exceed ____ lb, with takeoff thrust maintained on the operative engines and trim at the normal settings for takeoff with symmetric thrust. Automatic devices that normally operate in the event of a thrust failure may be used, and the airplane may be banked up to 5 degrees away from the inoperative engine.

Recommended values: The roll control force limits should be those specified by Paragraph 3.5.10.2. With yaw controls free, forces should be the Level 2 upper limits specified in 3.5.10.2 for Levels 1 and 2, and the Level 3 upper limits for Level 3.

D. RATIONALE BEHIND REQUIREMENT

As with other roll force limits, this requirement is intended to assure that compensation for engine failure is not the limiting control force requirement.

E. GUIDANCE FOR APPLICATION

This requirement is a direct extension of the yaw control force requirement of Paragraph 3.6.4.1.

F. DEMONSTRATION OF COMPLIANCE

Roll control forces should be measured while demonstrating compliance with the applicable portion of Paragraph 3.6.4.1.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.5.10.6.6 Roll axis control force limits — configuration or control mode change

A. REASON FOR REQUIREMENT

Intentional engagement or disengagement of any portion of the flight control system should never result in unusual or unreasonable demands on the pilot to retain control.

B. RELATED MIL-F-8785C REQUIREMENT

3.5.6.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.5.10.6.6 Roll axis control force limits — configuration or control mode change. The control force changes resulting from the intentional engagement or disengagement of any portion of the primary flight control system by the pilot shall not exceed the following limits: ___________________.

It is recommended that for at least 5 seconds following the mode change the change in roll force not exceed 10 pounds.

D. RATIONALE BEHIND REQUIREMENT

Trim transients following intentional pilot actions with the flight control system should obviously be small.

Since this requirement deals with intentional modification of the flight control system, it is implied that no failures have occurred. Failures are covered explicitly by 3.5.10.6.5.

E. GUIDANCE FOR APPLICATION

Proper application of this requirement may be performed by careful design of the airplane augmentation systems. Mode switching should assure that the new mode chosen does not have any large transients in initialization.

F. DEMONSTRATION OF COMPLIANCE

This requirement is effectively a subset of 3.5.8.3. Simulation, analysis, or flight test demonstrations for that paragraph should include force response tests.

G. SUPPORTING DATA

At this time no supporting data are available.

H. LESSONS LEARNED

None.

3.6 HANDLING QUALITY REQUIREMENTS FOR YAW AXIS

3.6.1 Yaw Axis Response to Yaw Controller

3.6.1.1 Yaw axis lower-order equivalent system requirements

3.6.1.1.1 Dynamic response

A. REASON FOR REQUIREMENT

This requirement assures that any lateral-directional oscillatory (dutch roll) response to yaw controller is sufficiently stable and well damped.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.1.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.1.1.1 Dynamic response. The equivalent parameters describing the response of sideslip to a yaw control input shall have the following characteristics: ______________________. The requirements shall be met in trimmed and in maneuvering flight with cockpit controls fixed and with them free, in oscillations of any magnitude that might be experienced in operational use. If the oscillation is nonlinear with amplitude, the requirement shall apply to each cycle of the oscillation. In calm air residual oscillations may be tolerated only if the amplitude is sufficiently small that the motions are not objectionable and do not impair mission performance.

Recommended minimum dutch roll frequency and damping are given in Table 1. The parameters should be found by matching the higher-order sideslip response to yaw control input to the following lower-order form, over the frequency range from 0.1 rad/sec to 10 rad/sec:

$$\frac{\beta}{F_{rp}} = \frac{K_\beta e^{-\tau_{e_\beta} s}}{[s^2 + 2\zeta_d \omega_d s + \omega_d^2]}$$

The algorithms described in Appendix A should be used for the fitting process. No limits are set on τ_{e_β} at this time.

When $\omega_d^2|\phi/\beta|_d$ is greater than 20 $(\text{rad/sec})^2$, the minimum $\zeta_d\omega_d$ should be increased above the $\zeta_d\omega_d$ minimums listed in Table 1 by:

$$\text{Level 1:} \quad \Delta\zeta_d\omega_d = 0.014(\omega_d^2|\phi/\beta|_d - 20)$$

$$\text{Level 2:} \quad \Delta\zeta_d\omega_d = 0.009(\omega_d^2|\phi/\beta|_d - 20)$$

$$\text{Level 3:} \quad \Delta\zeta_d\omega_d = 0.005(\omega_d^2|\phi/\beta|_d - 20)$$

with ω_d in rad/sec.

TABLE 1 (3.6.1.1.1)

RECOMMENDED MINIMUM DUTCH ROLL FREQUENCY AND DAMPING

LEVEL	FLIGHT PHASE CATEGORY	CLASS	Min ζ_d*	Min $\zeta_d\omega_d$* (rad/sec)	Min ω_d (rad/sec)
	A (CO and GA)	IV	0.4	0.4	1.0
	A	I, IV II, III	0.19 0.19	0.35 0.35	1.0 0.4
1	B	All	0.08	0.15	0.4
	C	I, II-C, IV	0.08	0.15	1.0
		II-L, 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.

D. RATIONALE BEHIND REQUIREMENT

Allowable dutch roll oscillatory characteristics are specified in terms of minimum values of ζ_d, ω_d, and $\zeta_d\omega_d$; the latter is also a function of $|\phi/\beta|_d$ when $|\phi/\beta|_d$ is very large. From examination of supporting data it was 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, e.g., 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 $|\phi/\beta|_d$.

Limits on τ_{e_β} have not been specified. It is expected that τ_{e_β} is not as critical as delays in the pitch and roll axes, since the pilot does not normally perform high-gain precision tracking of sideslip with the yaw control. Time delay has been shown to be especially important only when aggressive closed-loop tracking is inherent to the flying task.

E. GUIDANCE FOR APPLICATION

While not specifically stated in the requirement, it is intended that equivalent values of ζ_d and ω_d be used for augmented airplanes. 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. For most airplanes an appropriate lower-order equivalent system for sideslip response to a rudder input is simply:

$$\frac{\beta}{F_{rp}} = \frac{K_\beta e^{-\tau_{e_\beta} s}}{(s^2 + 2\zeta_d\omega_d s + \omega_d^2)}$$

Hence, for cases where τ_{e_β} is small, simple measurements from a time response of sideslip to rudder kick will frequently be sufficient. Complications arise when $|\phi/\beta|_d$ is large and a significant portion of the dutch roll response occurs in roll. This is currently covered by the empirically developed formulas for $\Delta\zeta_d\omega_d$. The expressions indicate that an incremental increase in the required total damping ($\Delta\zeta_d\omega_d$) is necessary when $\omega_d^2|\phi/\beta|_d > 20$.

F. DEMONSTRATION OF COMPLIANCE

Compliance must be demonstrated through flight testing at the minimum and maximum specified operational altitudes, over the range of service speeds, with the airplane configured for maximum yawing moment of inertia. In most cases a simple rudder pulse or doublet is sufficient to excite the dutch roll mode. Classical time response measures should be sufficient to extract ζ_d and ω_d from the sideslip time histories.

G. SUPPORTING DATA

Because of the fundamental nature of this requirement there is a reasonably large data base wherein ζ_d, ω_d, and $|\phi/\beta|_d$ have been varied in a systematic manner. However, a review of this data base reveals that the minimum ζ_d for Class IV Category A (CO and GA) ($\zeta_d \geq 0.4$) is not supported. This is a result of the fact that stability augmentation has allowed the realization of much larger values of ζ_d than were previously possible. Given the option to fly with larger values of ζ_d, pilots have found significant improvements in tracking performance, both air-to-air (CO) and air-to-ground (GA) — hence the recommendation from the AFFTC to set ζ_d at 0.4 for these tasks. (Unfortunately, we have not received documentation, in the form of pilot ratings, from AFFTC.)

It should also be recognized that recent years have seen a large increase in the emphasis on aggressive pilot behavior in flight test experience. This would also have the effect of increasing the minimum levels of ζ_d which were quite low. It is expected that future experiments will show a need for increasing $\zeta_{d_{min}}$ for other aircraft Classes and Flight Phase Categories.

The supporting data that currently exist will be reviewed for each Flight Phase Category in the following paragraphs.

1. Categories A and B

Since most of the available flight test reports involve either open-loop bank angle maneuvers or landing approach tasks, there is very little data for substantiating Category A requirements. As a result, much of the data presented here may be more applicable to Category B Flight Phases. The data will be compared with the limits in Table 1 for both Flight Phases.

Reference 71 contains some of the pilot ratings that were used to formulate the Category B limit on ζ_d in MIL-F-8785B. As shown in Figure 1, the correlation is not very strong. The tasks were essentially open-loop: abrupt 45 to 60 deg bank-angle coordinated turn

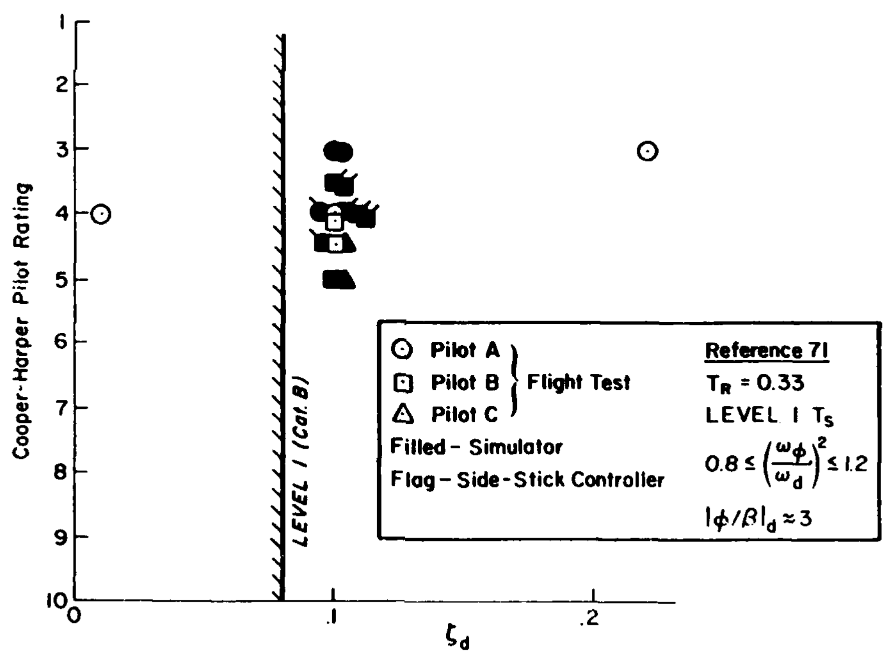


Figure 1 (3.6.1.1.1). Effect of ζ_d on Pilot Ratings for In-Flight and Fixed-Base Simulations of Reference 71; ω_d = 1.78 - 1.90 rad/sec (Category B Flight Condition) entries; abrupt aileron reversals with coordinated rudder; and rudder-fixed and -free 360 deg bank angle rolls. All that can really be concluded is that for $\omega_d \approx 1.9$ rad/sec, a ζ_d of 0.10 is on the boundary between Level 1 and Level 2. The data are included here only because they were used in the MIL-F-8785B BIUG (Reference 11) to support the dutch roll requirements.

Equally ambiguous data, obtained from Reference 72, are presented in Figure 2 (reproduced from Reference 11). The flight test program of Reference 72, performed in an F4U-5 airplane, included both Category A and B type tasks: release from a steady sideslip; entry into and recovery from a 45 deg banked turn and a standard rate turn (in simulated instrument flight); and "tracking of any available target in approximately level flight." Therefore, Figure 2 includes the Level 1 boundaries for both Category A and B Flight Phases from Table 1. It is clear that the ratings given support the Category B boundary quite well, but do not show support for the Category A boundary. But, again, this is probably more a consequence of the test maneuvers than of the boundaries.

Fixed-base simulator data from Reference 73 (Figure 3, reproduced from Reference 11) are again more supportive of the Category B boundaries. Tasks included entry to and exit from a standard rate turn; abrupt directional kicks and releases; ±60 and ±90 deg rolls; and abrupt rolls at elevated load factors (3-4 g). (Not surprisingly, the latter maneuvers added little to the pilots' evaluations, since the tests were conducted in a fixed-base simulator.)

Figure 4 shows data from the fixed-base simulations of Reference 74. Based upon the reentry mission simulated, and upon the specific maneuvers performed, the data should be considered applicable to the Category B Flight Phases. As stated in Reference 74, "The overall mission was described as the re-entry, descent and landing of a re-entry vehicle. In particular, each pilot was told that this mission did not require high maneuverability but did require fairly precise control of attitude." Tasks included straight flight, turning fight with shallow and steeply banked turns of up to 60 deg bank angle, and tracking of roll and sideslip random inputs and minimizing pitch disturbances.

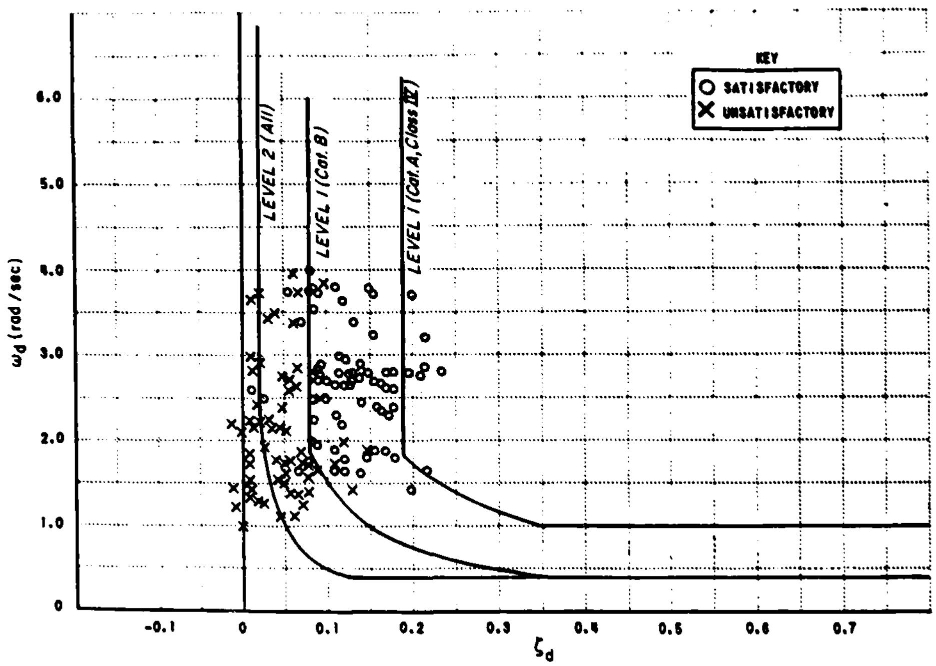


Figure 2 (3.6.1.1.1). Dutch Roll Data (From Reference 72)

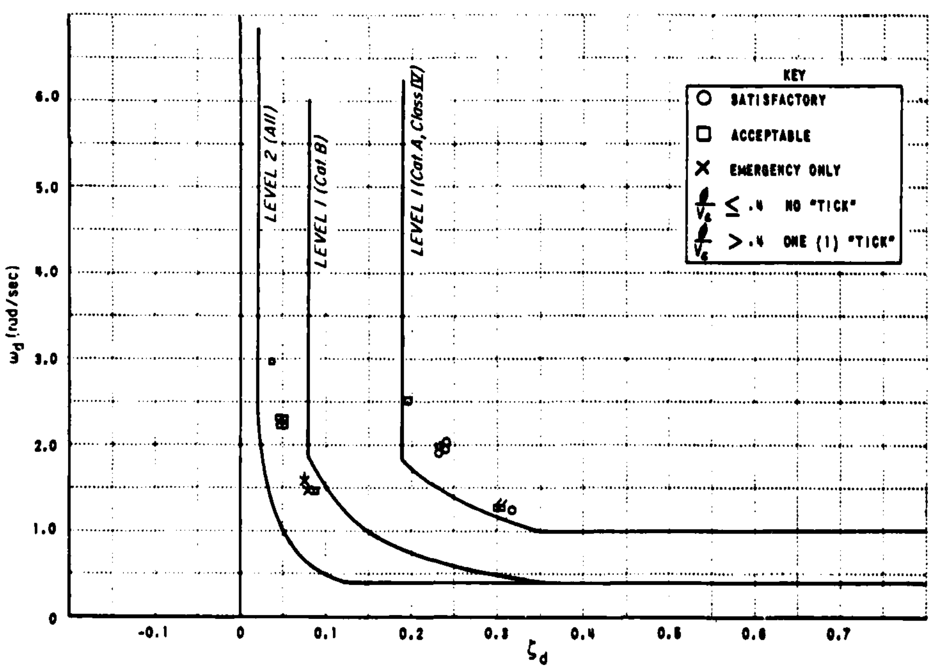


Figure 3 (3.6.1.1.1). Dutch Roll Data (From Reference 73)

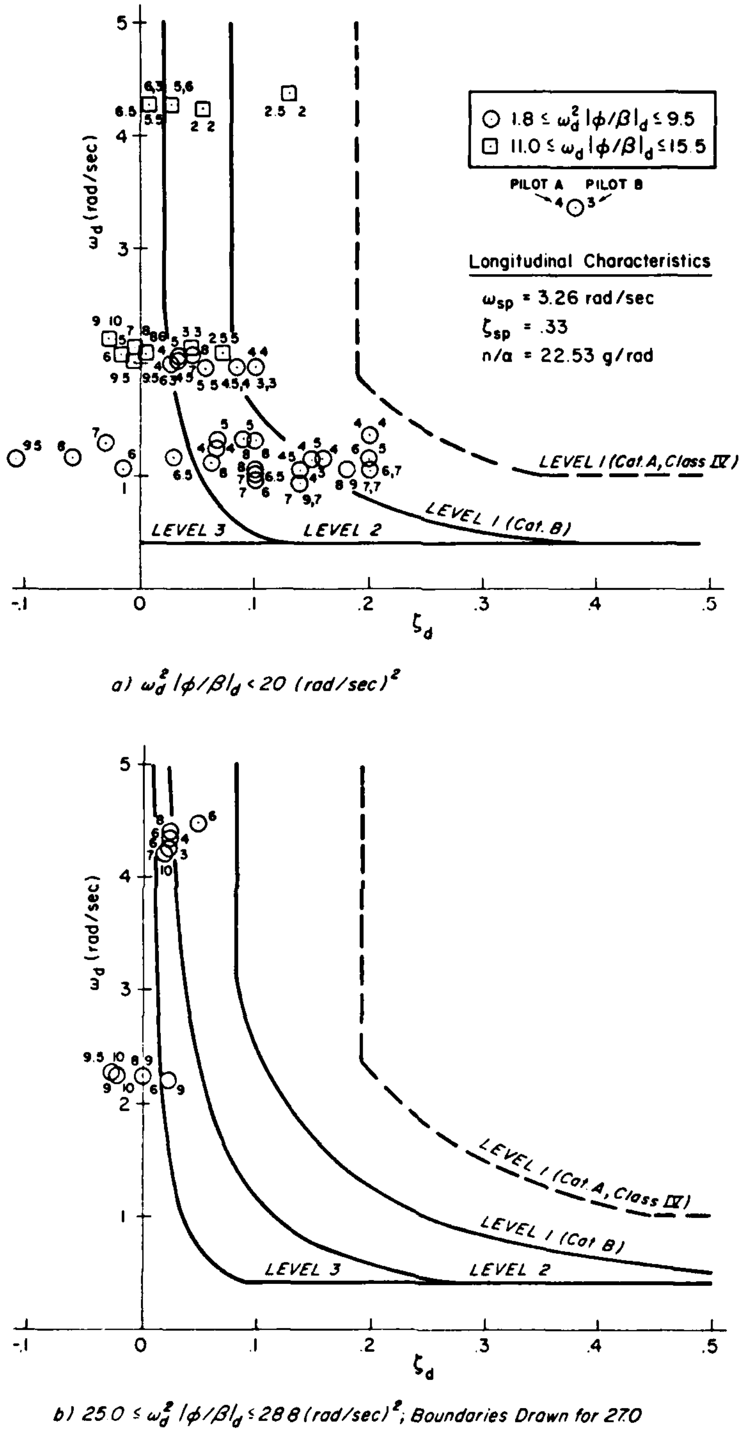
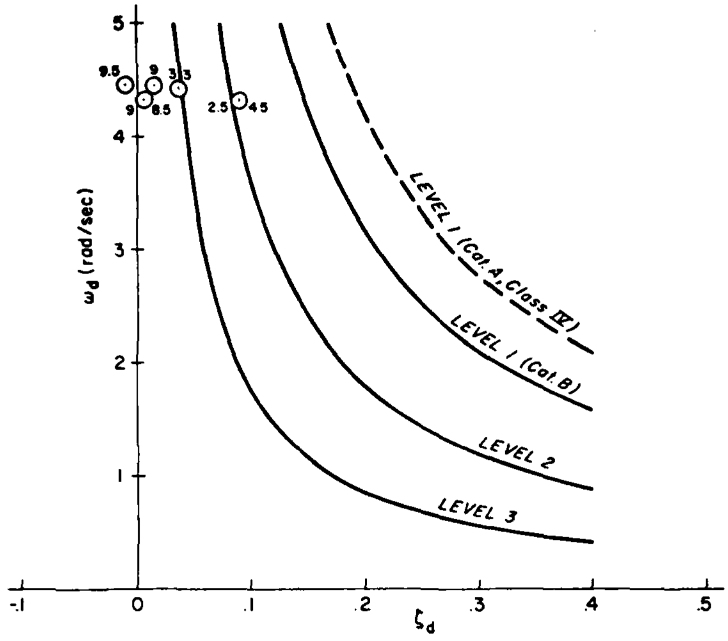
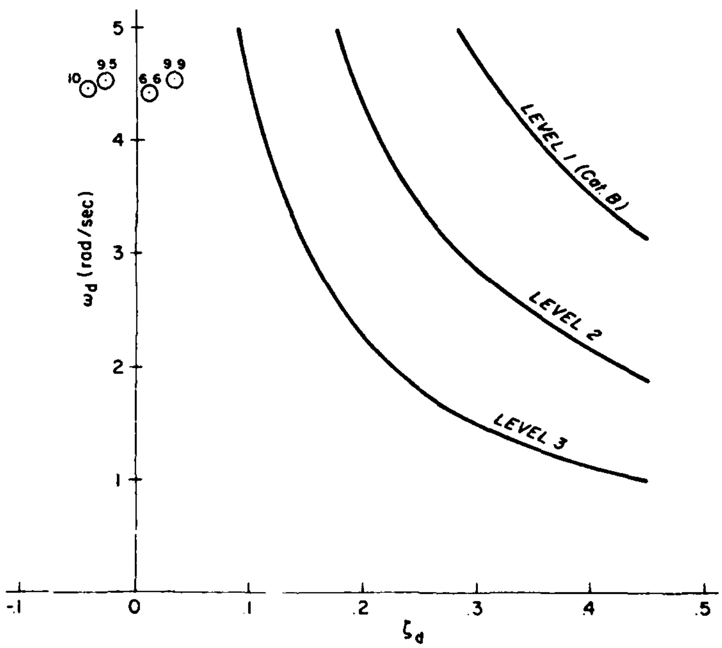


Figure 4 (3.6.1.1.1). Dutch Roll Data from Fixed-Base Simulation of Re-Entry Task (Reference 74)
$T_R \approx 0.40$ sec, $T_S \approx \infty$



c) $53.5 \le \omega_d^2\,|\phi/\beta|_d \le 55.5\ (rad/sec)^2$; Boundaries Drawn for 54.5



d) $97.9 \le \omega_d^2\,|\phi/\beta|_d \le 116.5 (rad/sec)^2$; Boundaries Drawn for 110.0

Figure 4 (3.6.1.1.1). (Concluded)

It is not clear from Reference 74 if the pitch disturbances occurred simultaneously with the lateral-directional random inputs, but it is possible that this could have affected pilot ratings.

The data of Figure 4a fit the Table 1 criteria quite well if the Level 1 limits are taken for Category A, and Class IV aircraft.

Reference 74 provides data for evaluating the additional damping requirements of Table 1. A $\Delta\zeta_d\omega_d$ is specified when the product $\omega_d^2|\phi/\beta|_d$ is greater than 20 $(rad/sec)^2$. The effects of this on the boundaries can be seen by comparing Figures 4a through 4d for increasing values of $\omega_d^2|\phi/\beta|_d$. The data of Figures 4b, 4c and 4d correlate well with the boundaries drawn. It should be noted, however, that the high $\omega_d^2|\phi/\beta|_d$ data correlate just as well with the basic boundaries of Figure 4a. More data would be desirable to validate this requirement.

Data from Reference 48 for fixed-base and in-flight simulations using the USAF/Calspan T-33 are presented in Figures 5 and 6. Only those configurations for which the pitch, roll, and spiral characteristics were Level 1, and for which $(\omega_\phi/\omega_d)^2$ was between about 0.8 and 0.12 are plotted. [Multiple ratings in Figures 5 and 6 are in some cases due to differing values of $(\omega_\phi/\omega_d)^2$; in general, the best ratings shown for any data point are for $(\omega_\phi/\omega_d)^2 \doteq 1.0$.] The tasks of Reference 48 are clearly Category B, i.e., a reentry vehicle flown in straight flight and turning flight with shallow (±30 deg bank angle) and medium (±60 deg) banked turns, and rolling turns of up to 180 deg bank. Additionally, the maneuvers were performed while a random noise signal was fed to the elevator, aileron and rudder actuators. These noise effects caused pilot rating degradations of 0 to 1-1/2 rating points. The data of Figure 5 (for low values of the parameter $\omega_d^2|\phi/\beta|_d$) support the Category B Level 1 damping boundary. Data for large values of $\omega_d^2|\phi/\beta|_d$ are shown in somewhat different form (Figure 6). Only one configuration [the in-flight simulation with $\zeta_d\omega_d$ = 0.17 rad/sec, $\omega_d^2|\phi/\beta|_d$ = 29.4 $(rad/sec)^2$] shows support for increasing the damping requirements as $\omega_d^2|\phi/\beta|_d$ increases. All other data fit the <u>basic</u> requirement ($\zeta_d\omega_d >$ 0.15 rad/sec for Level 1).

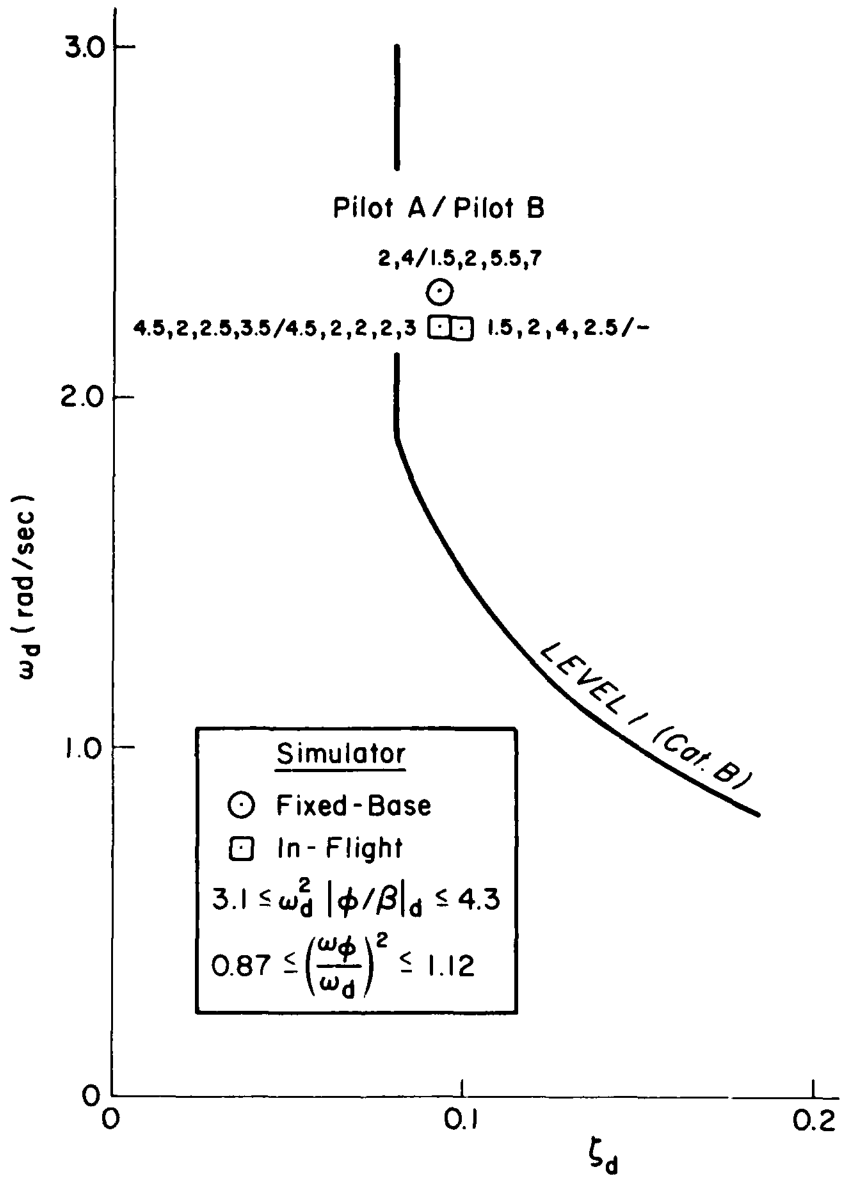


Figure 5 (3.6.1.1.1). Dutch Roll Data (From Reference 48; Low $\omega_d^2|\phi/\beta|_d$)

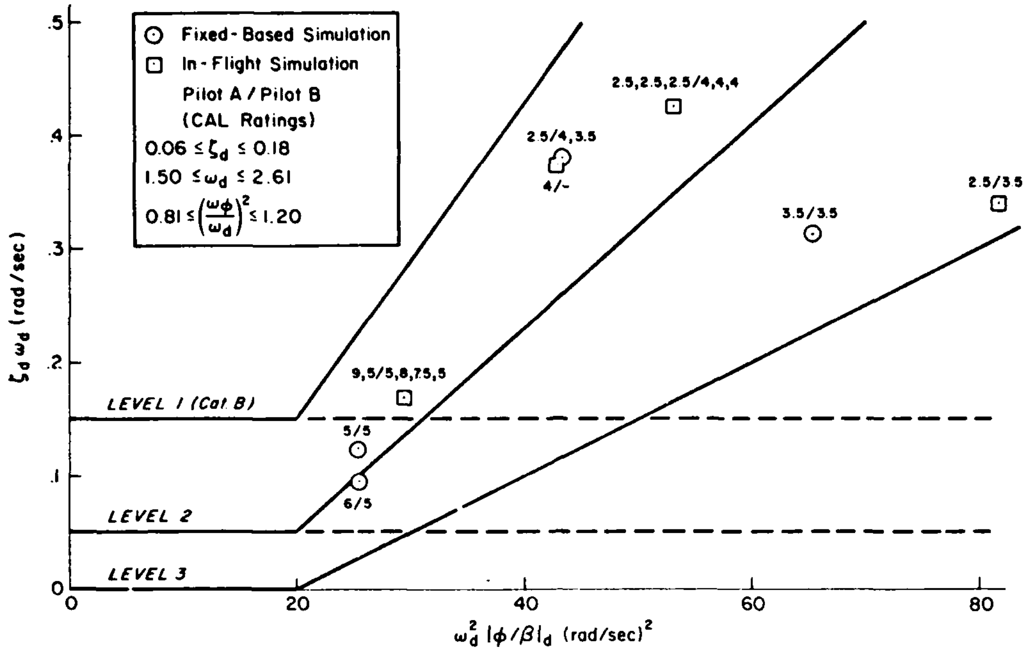


Figure 6 (3.6.1.1.1). Dutch Roll Data (From Reference 48; High $\omega_d^2|\phi/\beta|_d$)

Flight-test data of Reference 39, again simulating entry vehicles (and using the same maneuvers as for Reference 48), are presented in Figures 7 and 8. Figure 7 includes all applicable data from Reference 39 [i.e., those data for which ζ_{sp}, ω_{sp}, T_s, and T_R are Level 1, and $(\omega_\phi/\omega_d)^2 \approx 1$]. The low pilot ratings for cases with low $\omega_d^2|\phi/\beta|_d$ may be due to variations in other parameters (e.g., roll control effectiveness) rather than to dutch roll characteristics. Most of the data in Figure 7 fit the boundaries (drawn for low $\omega_d^2|\phi/\beta|_d$) quite well. The high $\omega_d^2|\phi/\beta|_d$ data are reproduced in Figure 8 (excluding the cases that are Level 2/3 based on the boundaries of Figure 7). The points that lie at $\omega_d^2|\phi/\beta|_d > 79$ (rad/sec)^2 indicate support for the Table 1 $\Delta\zeta_d\omega_d$ requirements. However, two points at $\zeta_d\omega_d \approx 0.4$–$0.5$ and $\omega_d^2|\phi/\beta|_d \approx 53$–$54$ (PR = 3) suggest that the boundaries may need refinement.

Applicable data points from the in-flight simulation of Reference 46 are shown in Figure 9. The maneuvers included straight and turning

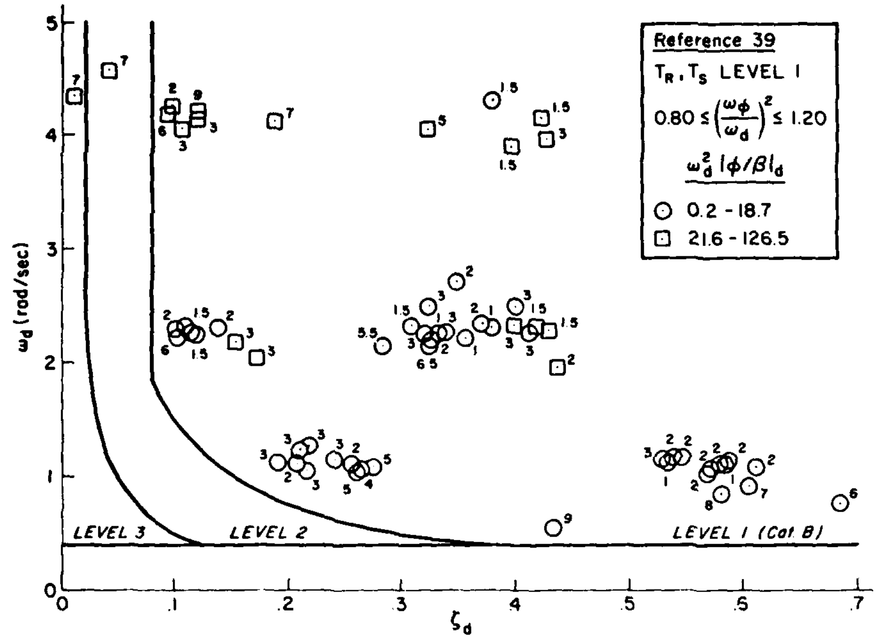


Figure 7 (3.6.1.1.1). Dutch Roll Data (From Reference 39)

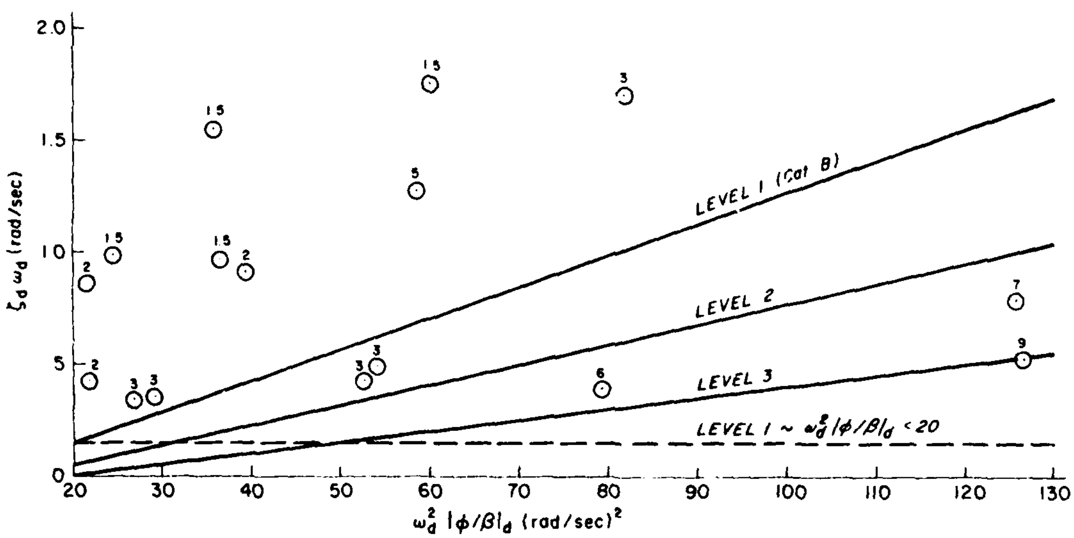


Figure 8 (3.6.1.1.1). Data for High $\omega_d^2|\phi/\beta|_d$ (From Reference 39)

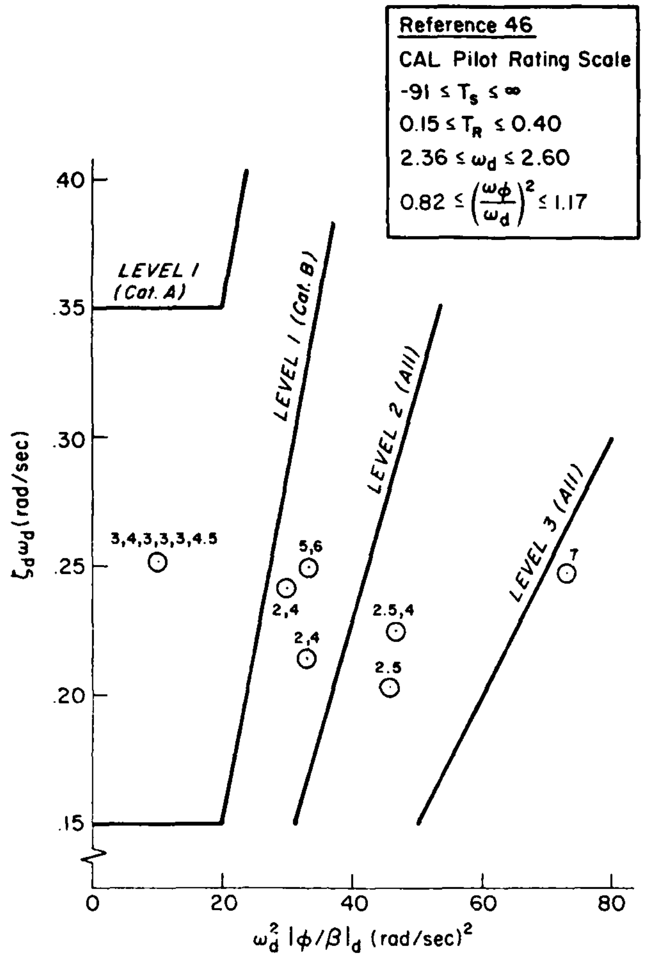


Figure 9 (3.6.1.1.1). Dutch Roll Data From In-Flight Simulation of Reference 46

flight, as well as a bank angle command tracking task and flight with artificial disturbances. Many of the points that lie in the Level 2-to-3 region on Figure 9 do not support the boundaries.

Summarizing the Category A/B data presented, there appears to be a definite trend of increased rating with increasing $\omega_d^2|\phi/\beta|_d$. However, the $\zeta_d\omega_d$ = 0.15 points show a scarcity of good data for Category A Flight Phases, especially at low values of ω_d. While the Level 2 and 3 boundaries are reasonably well supported by pilot ratings, some contradictions exist concerning the value of the supplemental $\Delta\zeta_d\omega_d$ requirement for $\omega_d^2|\phi/\beta|_d > 20$ (rad/sec)^2. Some in-flight and fixed-base simulation data strongly support it, while other data strongly refute it. (The origins of this requirement will be discussed shortly, after analysis of available Category C data.)

2. Category C

The most complete set of data available for Category C Flight Phases comes from Reference 76. The flight test program utilized a variable-stability F-86E with seven evaluation pilots. Details of the task are unknown; though the test flight conditions were 10,000 ft altitude at 170 KEAS, Reference 76 states that "Ratings were given for the landing-approach condition only." Ratings were based on controls-fixed characteristics, and handling qualities in smooth and simulated rough air. The rough air "corresponded to pilot A's impression of moderate to heavy turbulence." Aileron yaw ($N_{\delta_{as}}$) was optimized by the pilots for each condition. Data are presented in Reference 76 in terms of oscillation period and time and cycles to half (or double) amplitude. These have been converted to equivalent dutch roll damping ratio and frequency for presentation in Figures 10 and 11. The spiral and roll modes, however, are not known, and may have influenced the values of $1/T_{1/2}$ and $1/C_{1/2}$ reported in Reference 76. The subscript "equiv" is added to ζ_d and ω_d in Figures 10 and 11 to indicate that the equivalent value may include spiral and roll mode effects.

The low ω_d data of Reference 76 are plotted in Figure 10 and are seen to correlate with the Table 1 boundaries quite well. The few data points with large $|\phi/\beta|_d$ generally show a degradation in pilot rating. However, the high-ω_d data (Figure 11), for which the parameter $\omega_d^2|\phi/\beta|_d$ is large, do not show this degradation. In fact, these data strongly support the basic damping requirements of Table 1.

Power approach tests conducted with the Princeton variable stabilty Navion (Reference 68) also show limited support for the Category C requirements, Figure 12.

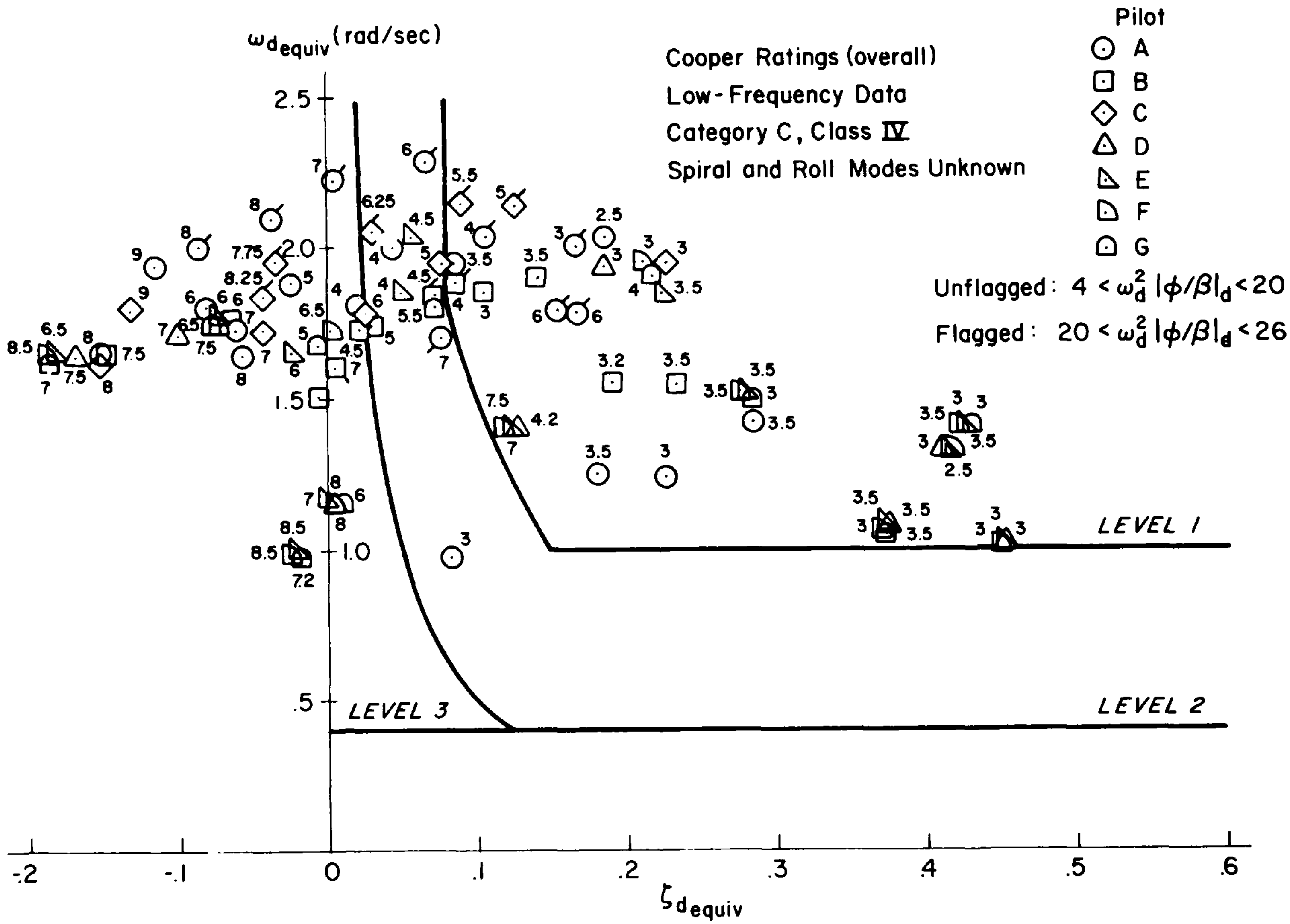


Figure 10 (3.6.1.1.1). Dutch Roll Data from Flight Tests of Reference 76 (F-86E; Low-Frequency Data)

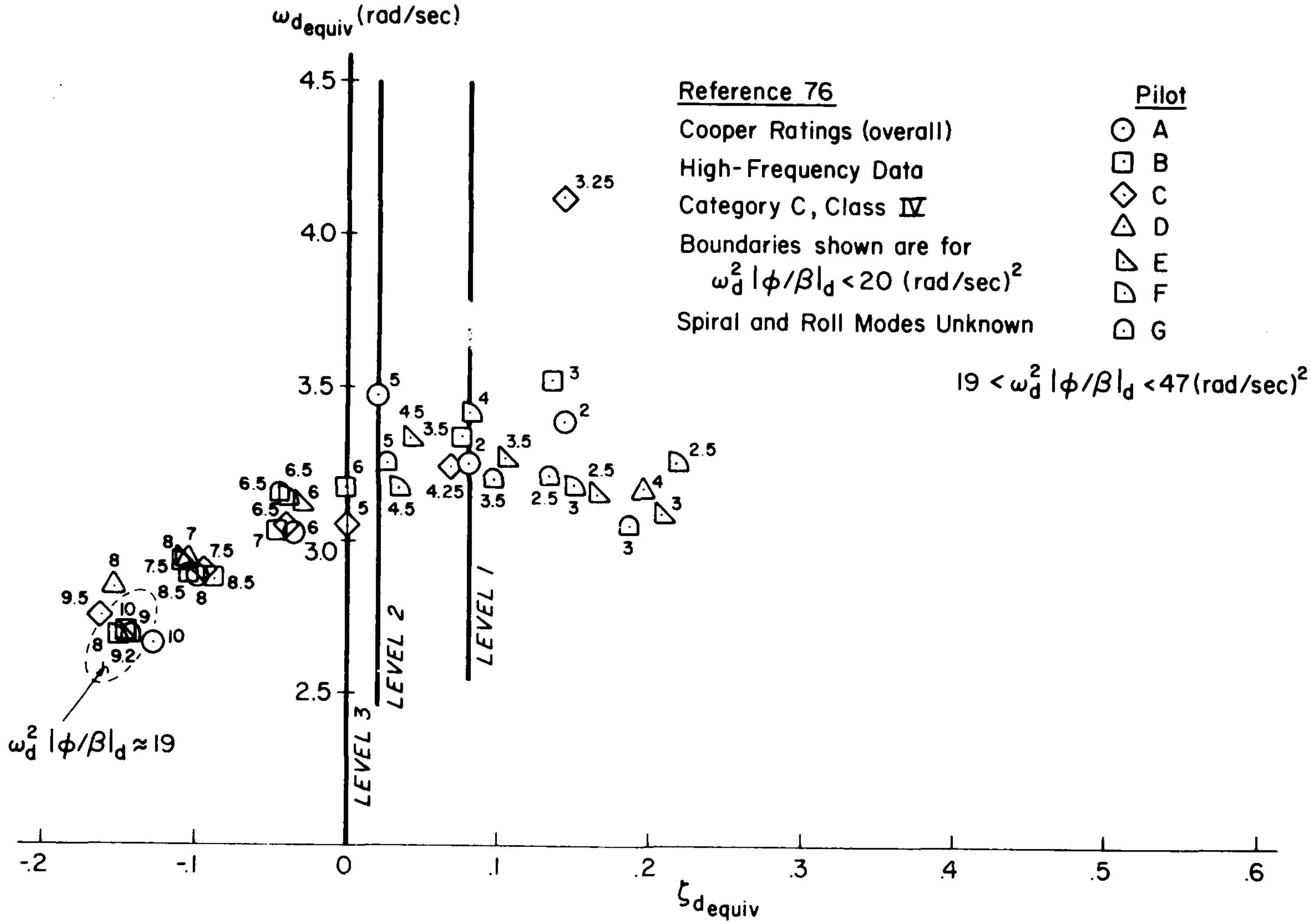


Figure 11 (3.6.1.1.1). Dutch Roll Data from Flight Tests of Reference 76 (F-86E; High-Frequency Data)

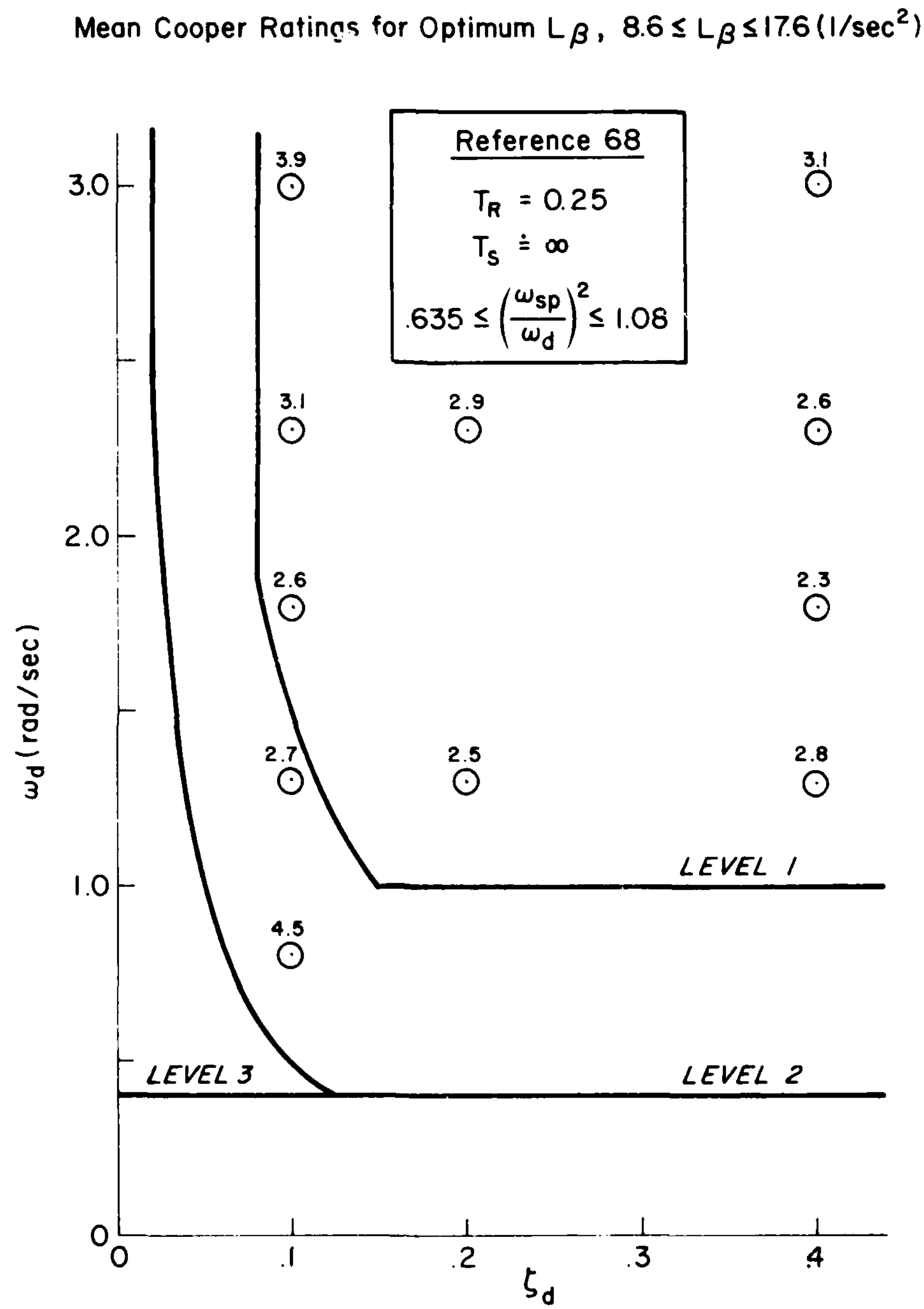


Figure 12 (3.6.1.1.1). Dutch Roll Data from In-Flight Simulation of Reference 68 (Navion; Pilot Ratings Shown for Optimum Values of L_β)

In the flight program of Reference 69, the USAF/Calspan T-33 was flown as a medium-weight Class II airplane. Cooper-Harper ratings for $N_{\delta_{as}} \equiv 0$ (Figure 13) support the boundaries for ζ_d, though the Level 2 limit could possibly be relaxed; but there are no data for $\omega_d < 1.0$ rad/sec, and the few low-frequency points suggest that, for Level 1, $\omega_{d_{min}} = 1.0$ rad/sec might be more appropriate. But there is too little data to justify an increase in $\omega_{d_{min}}$ from 0.4 to 1.0 rad/sec for Class II-L airplanes. These data show a dramatic effect of turbulence on pilot rating (see * in Figure 13). This could be evidence of the fact that turbulence is a dominant factor in setting limits on ζ_d. Future experiments should concentrate on this area. Additionally, any compliance demonstration should include moderate turbulence.

The Category C frequency and damping ratio boundaries for Class III airplanes are supported by moving-base simuator data from Reference 77 (Figure 14). Evaluation tasks consisted of turn entries and recoveries, roll reversals, sideslips, and dutch roll oscillations; instrument approaches; and instrument approaches with lateral offsets. Similar tests were flown on a variable-stability B-367-80 transport (including landing). While detailed data are not reported in Reference 77 for the flight tests, pilot ratings for a similar range of ζ_d, ω_d, and $(\omega_\phi/\omega_d)^2$ show general agreement with the simulator data (Figure 15). A major difference is the apparent insensitivity to $(\omega_\phi/\omega_d)^2$ in flight tests.

From the preceding review of available Category C data, it is clear that there is a lack of good, solid data: that few tests include touchdown and landing as a task; but that, as for Categories A and B, there is some mild support for the boundaries as they exist. If the Table 1 limits are to be refined or developed to be consistent and valid, much more testing is necessary.

3. Effect of $\omega_d^2|\phi/\beta|_d$

The criterion first proposed in Reference 62, in which the value of $\zeta_d\omega_d$ required to maintain a given pilot rating is made a function of $\omega_d^2|\phi/\beta|_d$ (Figure 16), has been retained in the lateral-directional oscillatory requirements of Table 1. It can be seen that $\omega_d^2|\phi/\beta|_d$ is

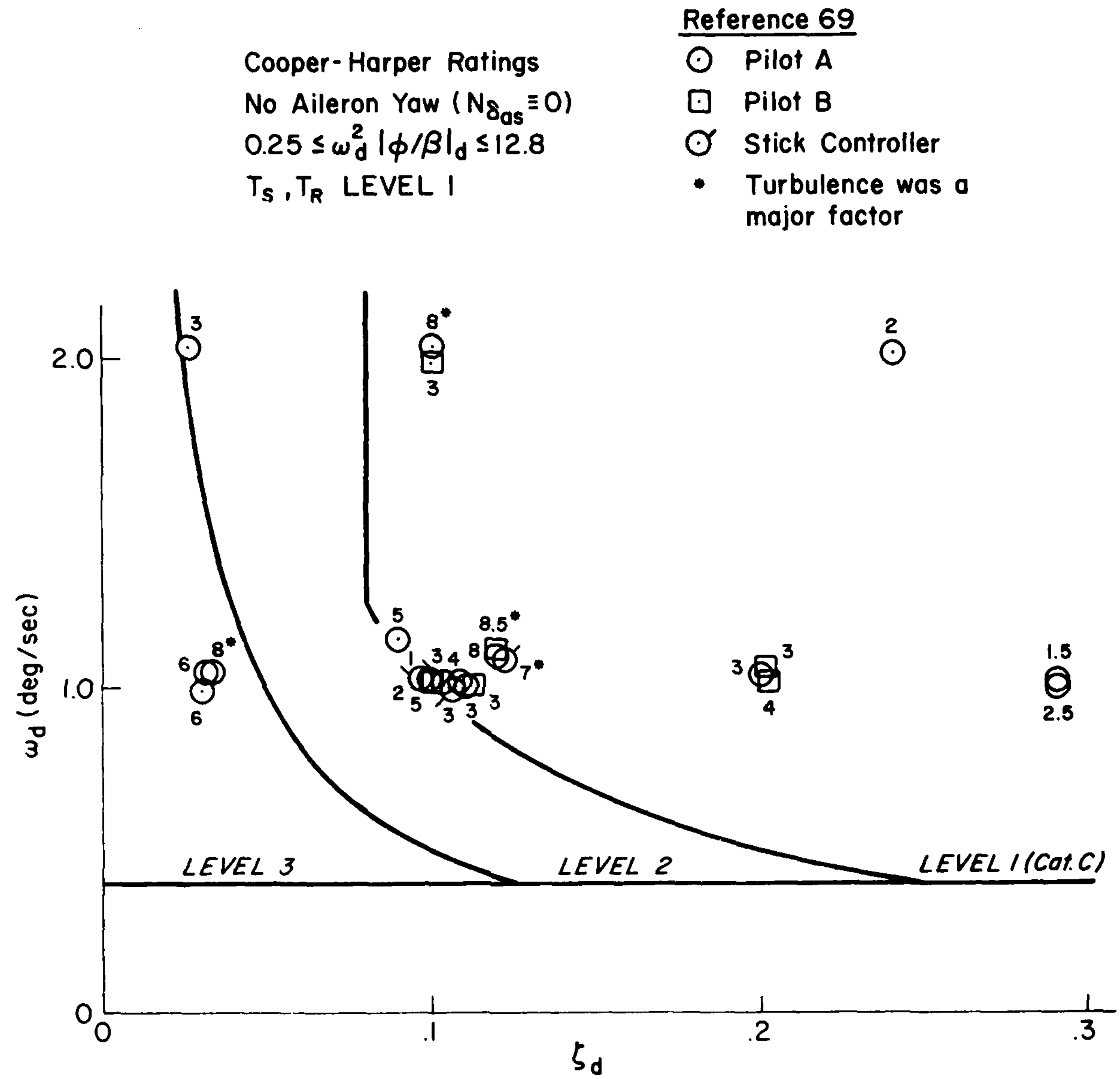


Figure 13 (3.6.1.1.1). Dutch Roll Data from In-Flight Simulation of Class II-L Airplanes in Landing Approach (T-33; Reference 69)

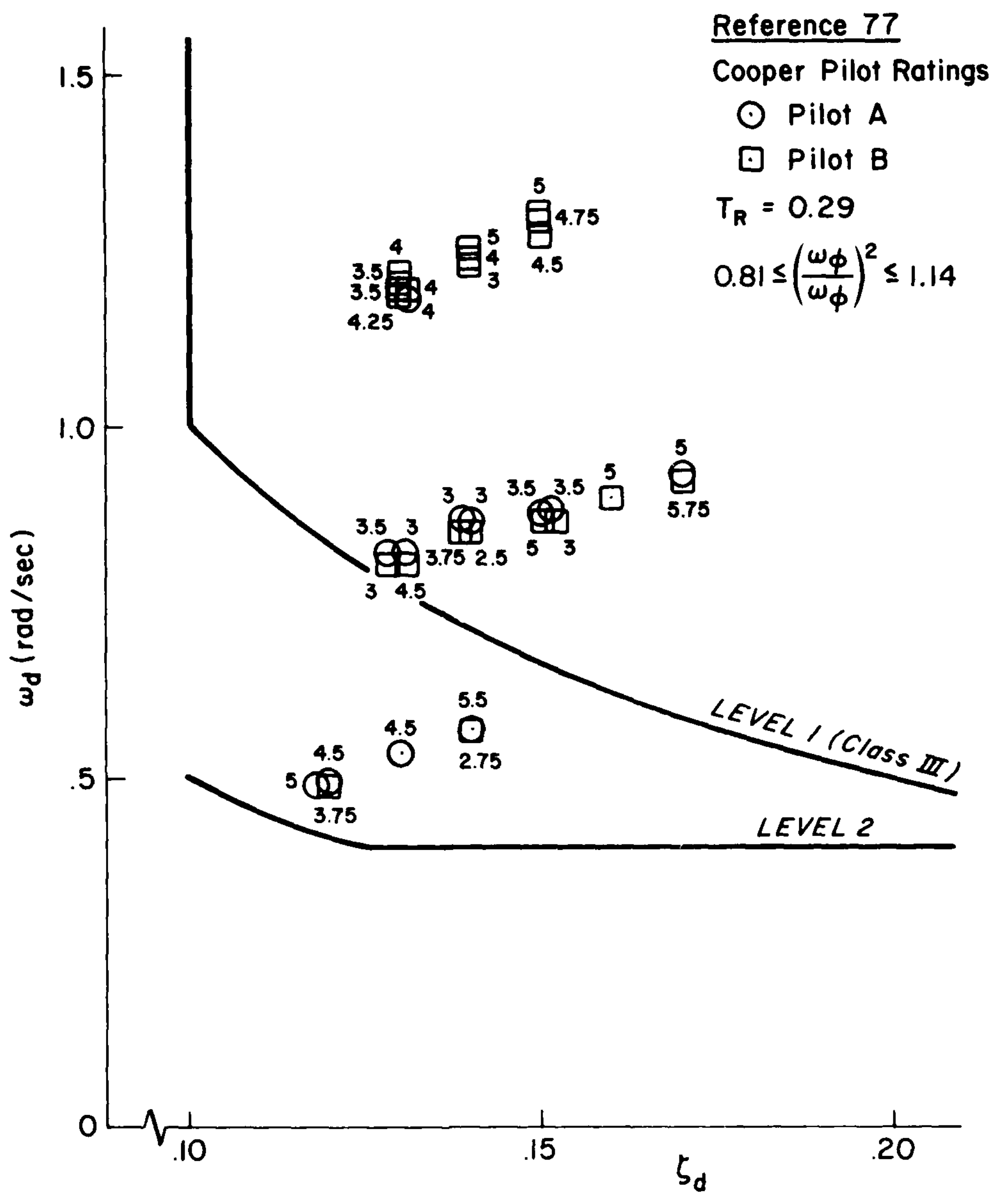


Figure 14 (3.6.1.1.1). Dutch Roll Data From Moving-Base Simulator Tests of Reference 77 (Supersonic Transport)

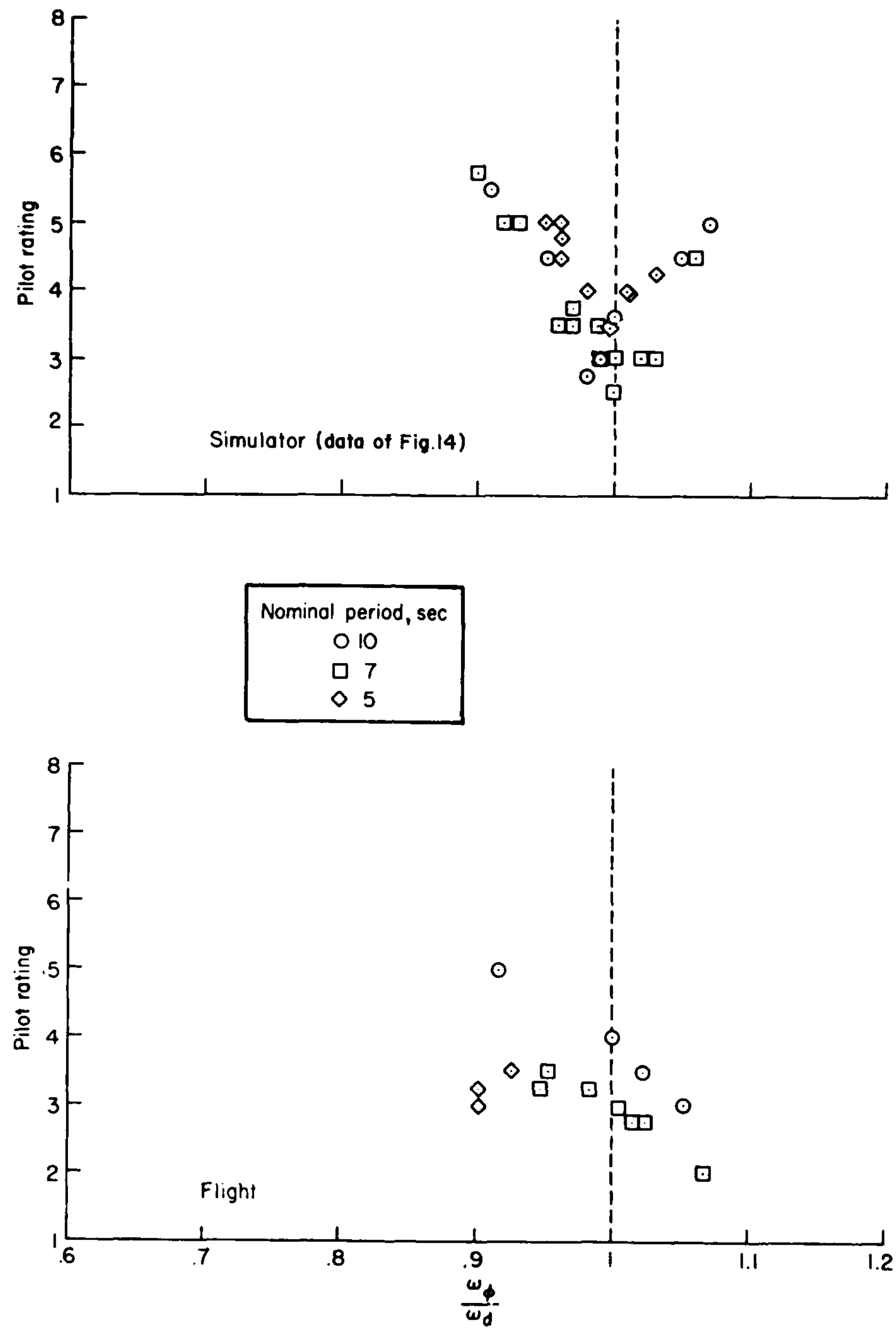


Figure 15 (3.6.1.1.1). Variation of Pilot Rating with ω_ϕ/ω_d for Moving-Base Simulator and Flight Data $\zeta_d \approx 0.15$ (Reference 77)

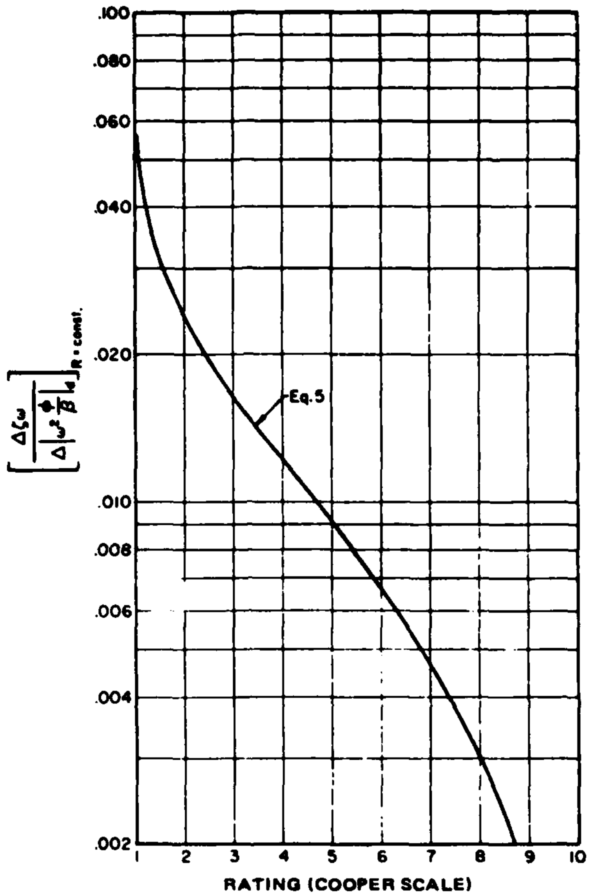


Figure 16 (3.6.1.1.1). $\Delta\zeta\omega/\Delta\left(\omega_d^2|\phi/\beta|_d\right)$ Required to Maintain a Given Basic Rating (From Reference 69), Based Upon Data of Figures 17 and 18 analogous to $|\ddot{\phi}/\beta|_d$. The data upon which the curve of Figure 16 is based are from Reference 78 and are presented in Figures 17 and 18.

Due in part to the lack of agreement between more recent test data and the $\omega_d^2|\phi/\beta|_d$ requirements, the basis for these requirements will be reviewed here. The $\Delta\zeta_d\omega_d$ requirements of Table 1 were determined from rudder kicks, and the dutch roll requirement specifies a yaw disturbance input; but the bulk of existing test data presented in support of the Table 1 requirements comes from closed-loop tasks. Most of these data do not show very strong support for the $\Delta\zeta_d\omega_d$ requirement (e.g., Figures 6, 8, 9, 10, and 11). In fact, for the more than 100 data points shown in Figures 4 through 11 with $\omega_d^2|\phi/\beta|_d > 20$, correlation with either the basic $\zeta_d\omega_d$ or additional $\Delta\zeta_d\omega_d$ requirement is almost identical — about 60 percent. The big difference is that of the pilot ratings that do not correlate, 50 percent are better than the basic requirement and 80 percent better than the $\Delta\zeta_d\omega_d$ requirement. (That is, either criterion will generally predict flying qualities at the same confidence level, but the $\Delta\zeta_d\omega_d$ requirement is — obviously — the more conservative.)

It has long been recognized that pilot acceptance of dutch roll oscillations is influenced by $|\phi/\beta|_d$, and it is for this reason that the $\Delta\zeta_d\omega_d$ requirements have been retained. However, the inconsistency of the supporting data should serve as a reminder that this is far from the best method of dealing with aircraft with large $|\phi/\beta|_d$ ratios. But so far nothing better has been suggested.

H. LESSONS LEARNED

Figure 19 illustrates the range of dutch roll damping and frequency found on some existing airplanes (Reference 79). Airplanes include the B-52, B-58, B-70, C-130, C-141, 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.

Characteristics for the Lockheed C-5A, C-141A, YC-141B, and L-1011 (Reference 14) are plotted in Figure 20 for the Category B Flight Phase

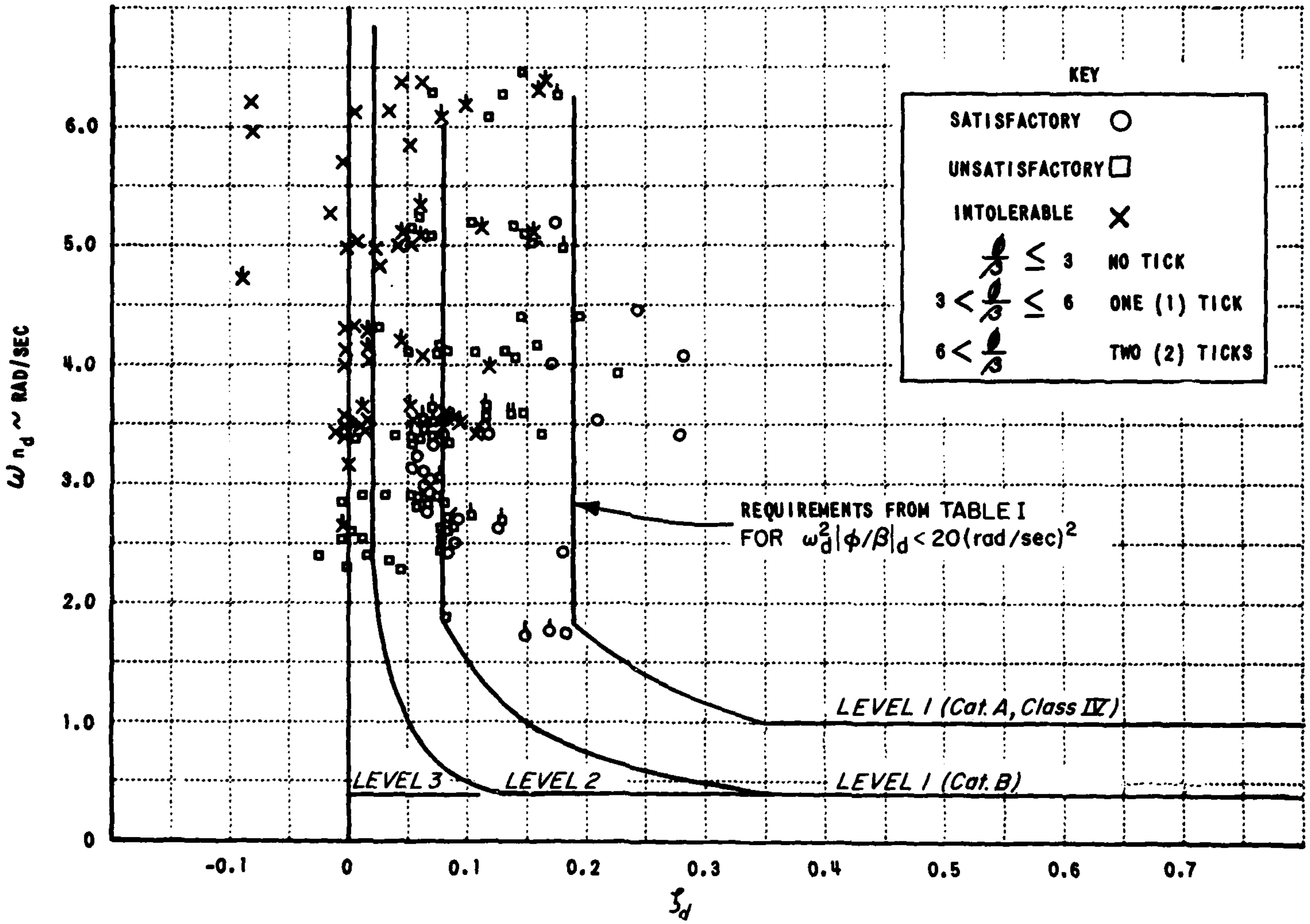
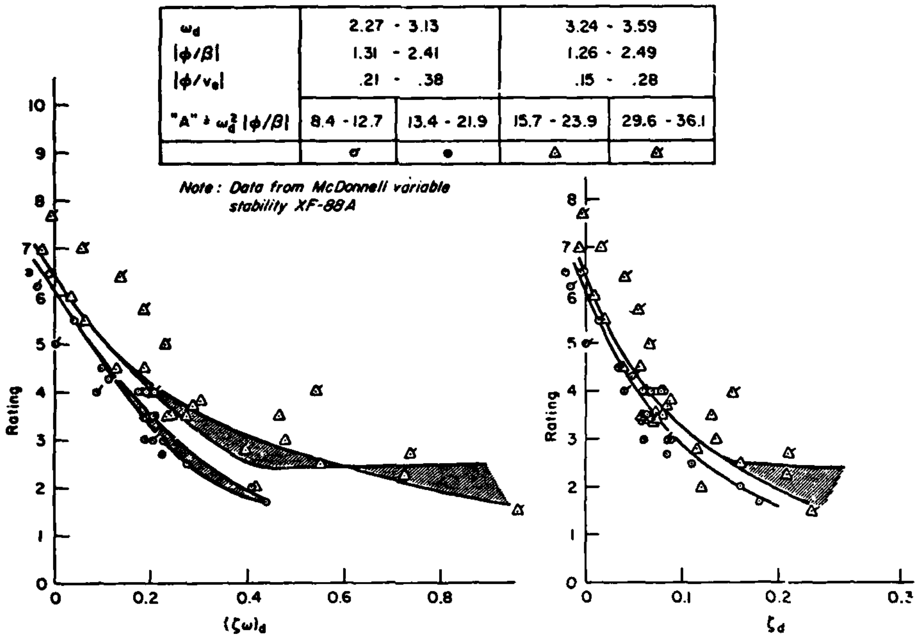
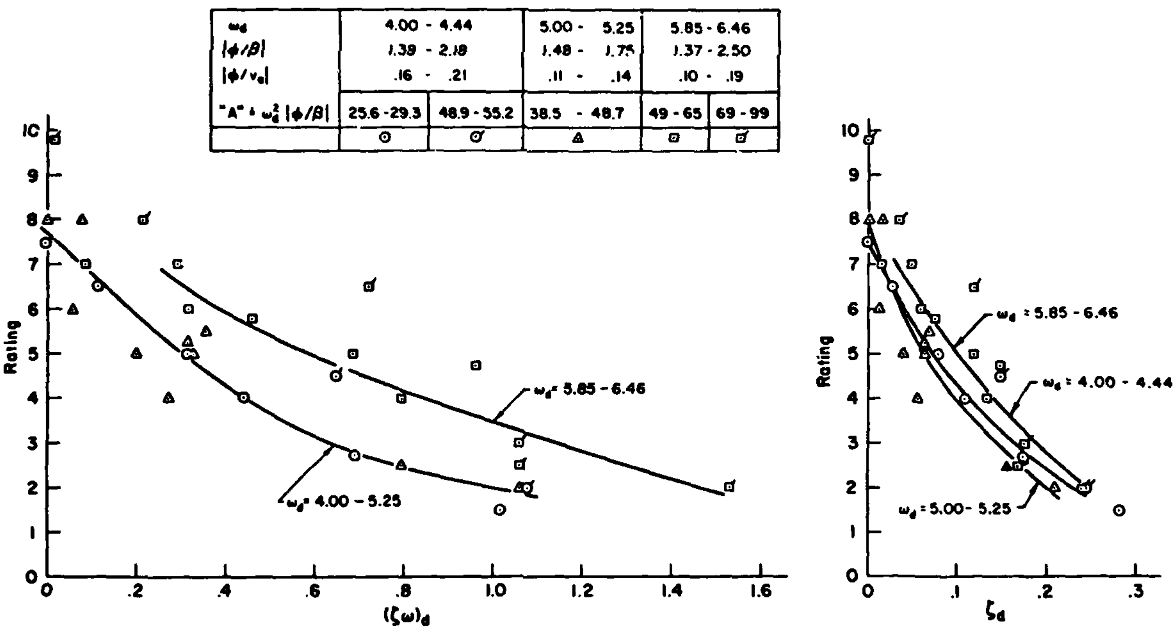


Figure 17 (3.6.1.1.1). Dutch Roll Data From Controls-Fixed Rudder Kicks of Reference 78 Compared to Limits of Table 1
(From Reference 11)



a) "Low Frequency"



b) "High Frequency"

Figure 18 (3.6.1.1.1). Data Upon Which Relation of Figure 16 Are Based (Controls-Fixed Rudder Kicks, Reference 78; Figure Reproduced From Reference 62)

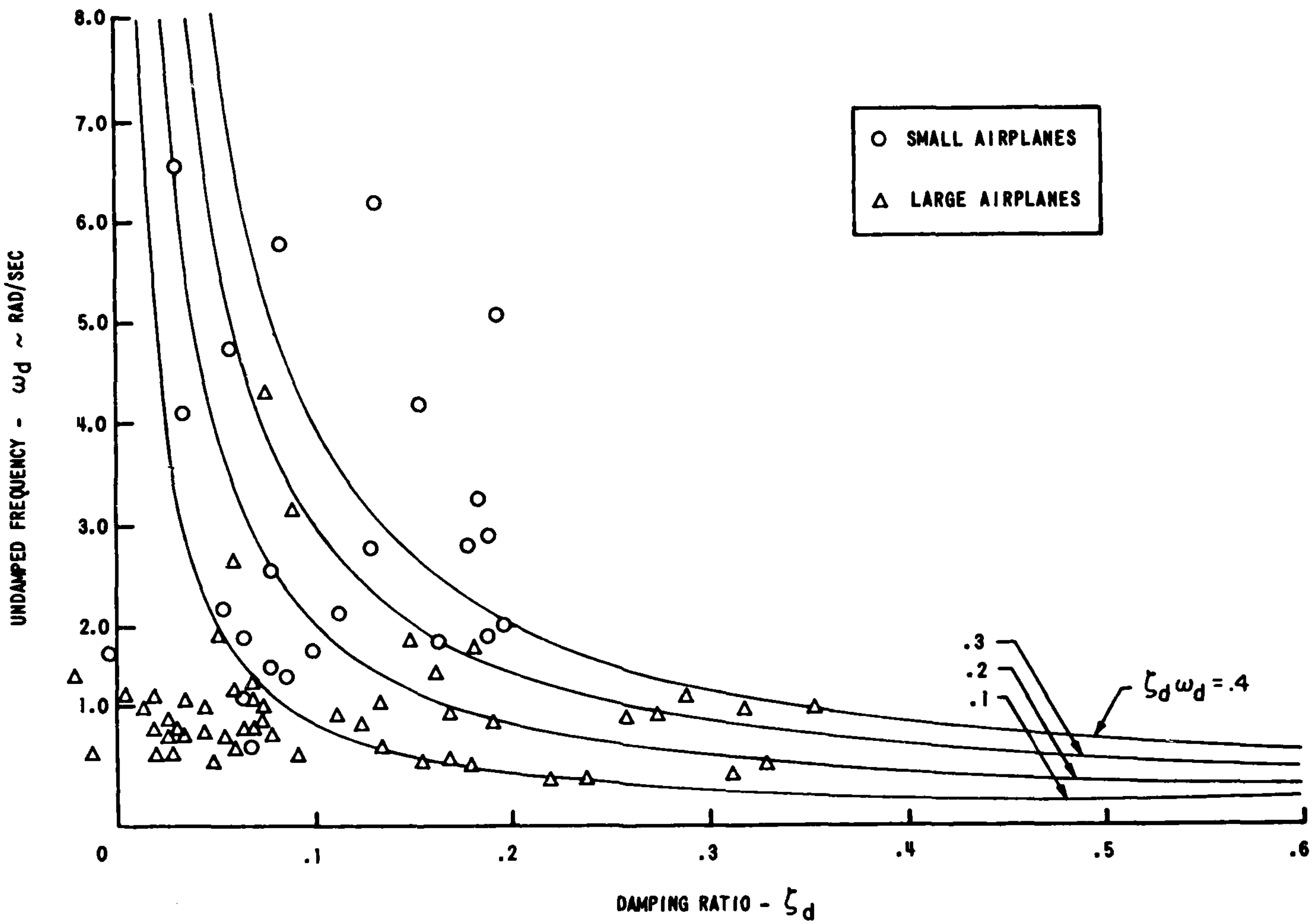


Figure 19 (3.6.1.1.1). Dutch Roll Data on Existing Airplanes
(From Reference 79)

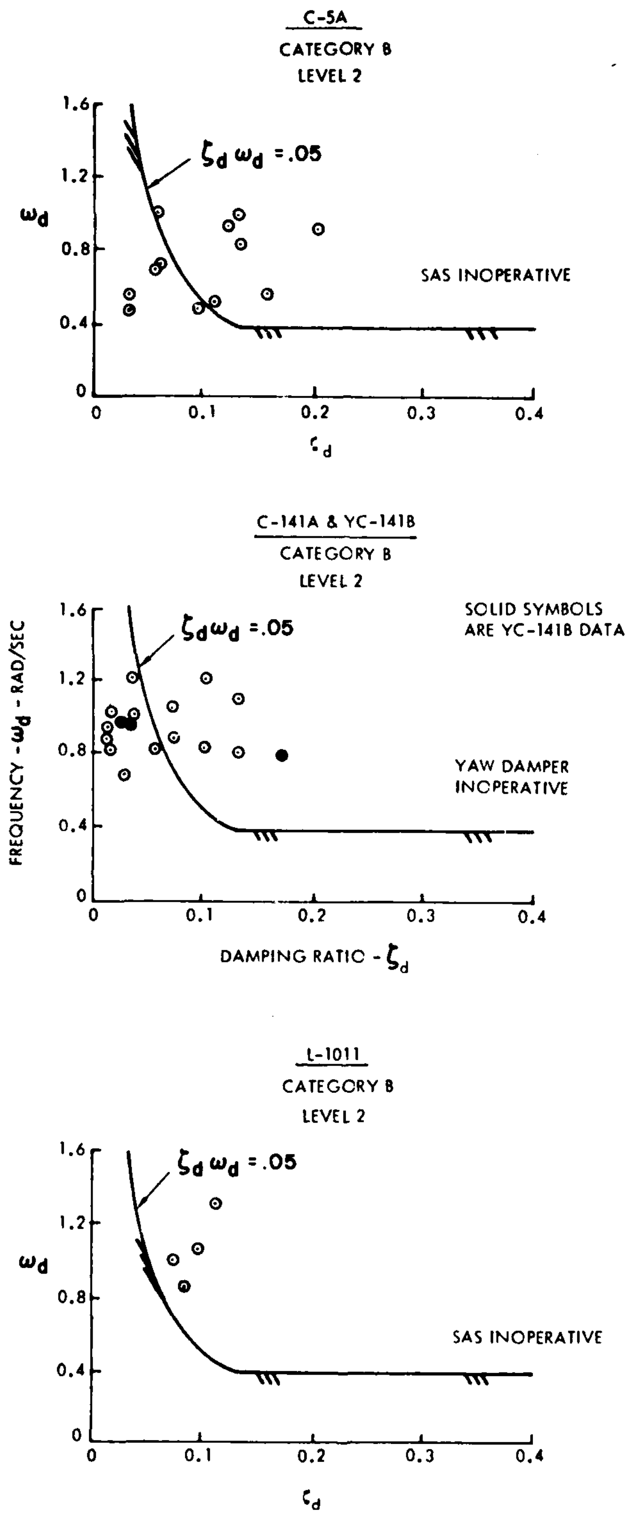


Figure 20 (3.6.1.1.1). Lateral Directional Damping for Some Class III Airplanes (From Reference 14)

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. Reference 14 discusses these results for the C-141A and C-5A:

> An evaluation of the C-141A dutch-roll recovery techniques with the yaw damper inoperative was conducted by the Air Force Flight Test Center in February 1977. Results of the tests ["C-141A Dutch Roll Recovery," AFFTC Technical Letter Report, by Picha and Klein] show Harper-Cooper 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 ±45 degrees. 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 Harper-Cooper 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 show the opposite trend: minimum allowable values of ζ_d have been increased for some current airplanes. 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.

3.6.1.1.2 Steady-state response

A. REASON FOR REQUIREMENT

This requirement is intended to provide static directional stability for reasonable sideslip angles.

B. RELATED MIL-F-8785C REQUIREMENTS

3.3.6, 3.3.6.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.1.1.2 Steady-state response. The long-term response to yaw-control-pedal deflections shall have the following characteristics: ____________________.

This requirement applies to yaw-control-induced steady, zero-yaw-rate sideslips with the airplane 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 pounds of yaw-control-pedal force, or

c) Maximum roll control or surface deflection,

except that for single-propeller-driven airplanes 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 10 deg change in sideslip from the wings-level straight flight condition.

Right yaw-control-pedal force shall produce left sideslips and left yaw-control-pedal force shall produce right sideslips. For Levels 1 and 2 the following requirements shall apply. The variation of sideslip angle with yaw-control-pedal force shall be essentially linear for sideslip angles between ____ degrees and ____ degrees. For larger sideslip angles, an increase in yaw-control-pedal force shall always be required for an increase in sideslip.

Recommended sideslip angle range is +15 deg to -15 deg.

D. RATIONALE BEHIND REQUIREMENT

This requirement regulates against the classical rudder lock problem wherein a lightening in pedal force occurs at large sideslip angles (see Reference 2). Controls-free static directional stability for small sideslip angles is required in Paragraph 3.6.1.1.1 by specifying ω_d greater than zero. However, there is no provision in that paragraph for large-β stability.

E. GUIDANCE FOR APPLICATION

The recommended requirements are very straightforward in interpretation and application. It is sensible to insure static stability over the range of sideslips that might be attained in normal operation and that the yaw controls operate in a "normal" manner. A requirement for positive rudder deflection gradients with β both provides a cue to the pilot of increasing sideslip and prevents conditions of rudder lock (see, e.g., Reference 2).

F. DEMONSTRATION OF COMPLIANCE

Flight testing at trimmed steady sideslips must be performed with the airplane configured for most aft center of gravity. Test conditions should cover the service speed range at the minimum, intermediate, and maximum operational altitudes.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

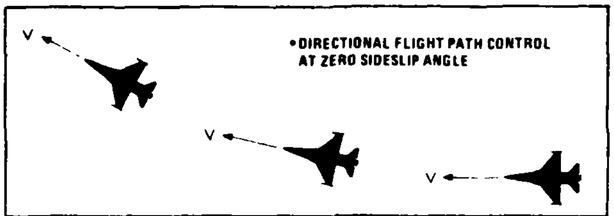
None.

3.6.1.2 <u>Yaw axis bandwidth requirements</u>

3.6.1.2.1 Bandwidth requirements for w..ugs-level turn mode

A. REASON FOR REQUIREMENT

This requirement is included to specify the response characteristics of aircraft utilizing direct force control (DFC) in the wings-level turn mode. This mode allows changes in heading to occur at zero bank angle and is sometimes referred to as the "A_y" mode (see, for example, Reference 229). A sketch of the airplane motions and a summary of the useful features of this mode are given below (from Reference 229):



• SINGLE VS MULTI AXIS FLIGHT PATH CORRECTIONS	• QUICKER LINE-UP ON GROUND TARGETS
• QUICK & PRECISE 10° TO 20° HEADING CHANGES	• SIMPLIFIED TRACKING OF MOVING GROUND TARGETS

B. RELATED MIL-F-8785C REQUIREMENT

3.4.11

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.1.2.1 <u>Bandwidth requirements for wings-level turn mode</u>.

a) <u>Dynamic response to direct force control (DFC) input</u>. The bandwidth of the open-loop response of heading or lateral flight path angle to the DFC control input shall be greater than ________ for Flight Phase ____. Turns shall occur at approximately zero sideslip angle and zero bank angle when using the DFC controller.

b) <u>Steady-state response to direct force control input</u>. Maximum DFC control inputs shall produce at least ___.

c) Direct force control forces and deflections. Use of the primary DFC control shall not require use of another control manipulator to meet the above dynamic response requirement. The controller characteristics shall meet the following requirements: ____________.

d) Pilot acceleration. Abrupt, large DFC inputs shall not produce pilot head or arm motions which interfere with task performance. Pilot restraints shall not obstruct his normal field of view nor interfere with manipulation of any cockpit control required for task performance.

Recommended values (Part a): The recommended values of required bandwidth depend on the piloting task associated with certain missions and mission phases as shown in Table 1.

TABLE 1 (3.6.1.2.1)

RECOMMENDED BANDWIDTH LIMITS FOR WINGS-LEVEL TURN MODE

TASK	REQUIRED BANDWIDTH (rad/sec)	
	LEVEL 1	LEVEL 2
Tracking (Cat. A) Air-to-air gunnery Strafing Dive bombing	1.25	0.60
Path Deviation (Cat. C) Formation Air-to-air refueling Approach	0.30	0.12

The parameters subject to the bandwidth limitation in Table 1 are given in Table 2.

TABLE 2 (3.6.1.2.1)

AIRCRAFT PARAMETERS SUBJECT TO BANDWIDTH LIMITATION
FOR WINGS-LEVEL TURN MODE

TASK	CONTROL VARIABLE
Air-to-air tracking	Heading angle if sideslip is not an important factor for weapon release Lateral path angle if sideslip must be small for weapon release
Air-to-air tracking Pointing tasks Strafing Flight path tasks Dive bombing	 Heading angle Lateral path angle, or lateral velocity

Recommended values (Part b):

Air-to-air: ±2.5 g lateral

Air-to-ground: ±1.0 g lateral

Cat. C tasks: ±0.5 g lateral

Recommended values (Part c): When the rudder pedals are to be used as the direct force controller, the requirements of 3.6.6 may be used as a guide. If a special-purpose controller such as a thumb switch or lever is to be used, acceptable characteristics should be determined in flight test.

D. RATIONALE BEHIND REQUIREMENT

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 achieve improved closed-loop tracking performance. The criterion originated from an old and well-accepted idea — namely, that a measure of the handling qualities of an airplane is its response characteristics when operated in a closed-loop compensatory tracking task.

The "bandwidth" (ω_{BW}) is a measure of the maximum frequency at which such closed-loop tracking can take place without threatening stability. It follows that airplanes capable of operating at a large value of bandwidth will have superior performance. An implicit characteristic of the requirement is that inter-axis coupling or contamination, regardless of type or source, affects the pilot opinion only insofar as it affects the bandwidth. This is a highly desirable feature because the very large varieties of coupling that can occur would make it virtually impossible to classify and set limits on each type.

The bandwidth criterion was also used to set limits on pitch attitude dynamics in Paragraph 3.2.1.2.

For additional background on the use of bandwidth as a criterion for direct force control (DFC) modes the reader is referred to References 115, 230, and 231.

E. GUIDANCE FOR APPLICATION

1. Definition of Bandwidth

The bandwidth frequency, ω_{BW}, to be used in Table 1 is defined from the open-loop frequency response plot of heading or lateral flight path angle to cockpit direct force control input (i.e., ψ/F_{DFC} or λ/F_{DFC}). Specifically, it is the frequency at which either the phase margin is 45 deg or the gain margin is 6 dB, whichever is lower (Figure 1). In order to apply this definition, first determine the frequency for neutral stability from the phase portion of the Bode plot (ω_{180}). The next step is to note the frequency at which the phase margin is 45 deg. This is the bandwidth frequency defined by phase, $\omega_{BW_{phase}}$. Finally, note the amplitude corresponding to ω_{180} and subtract 6 dB. The frequency at which this value occurs on the amplitude curve is $\omega_{BW_{gain}}$. 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

Bandwidth is the lesser of two frequencies $\omega_{BW_{phase}}$ and $\omega_{BW_{gain}}$

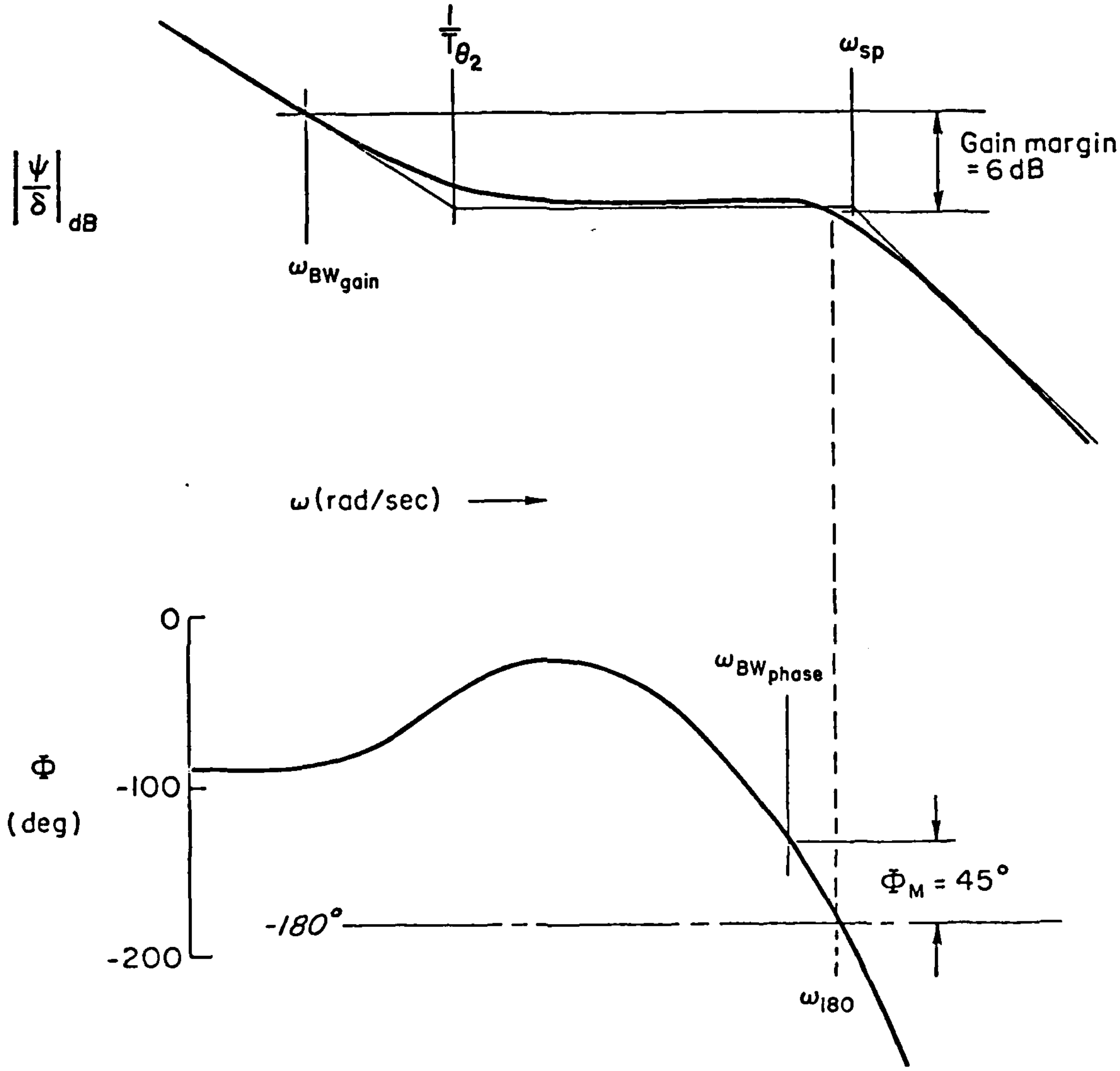


Figure 1 (3.6.1.2.1). Definition of Bandwidth Frequency deal of phase margin, ϕ_M, but increasing the gain slightly causes ϕ_M to decrease rapidly. Such systems are characterized by frequency response amplitude plots that are flat, combined with phase plots that roll off rapidly, such as shown in Figure 1.

2. Control Sensitivity

Some guidance regarding DFC control sensitivity may be found in the Reference 115 flight tests of the wings-level turn mode. The in-flight simulator was set up so that DFC control 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 found that 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 3 for high favorable yaw coupling and Figure 4 for very high favorable roll coupling with Figure 2 for low coupling. It is interesting to note that the nominal value of control sensitivity used for the latter case (0.008 g/lb) was found to be unacceptably high for the favorable coupling cases. The scatter in the data shown in Figure 4 is primarily due to pilot MP. In order to help explain why MP's ratings are higher than those of the other pilots, his comments have been annotated near the appropriate data points in Figure 4. 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

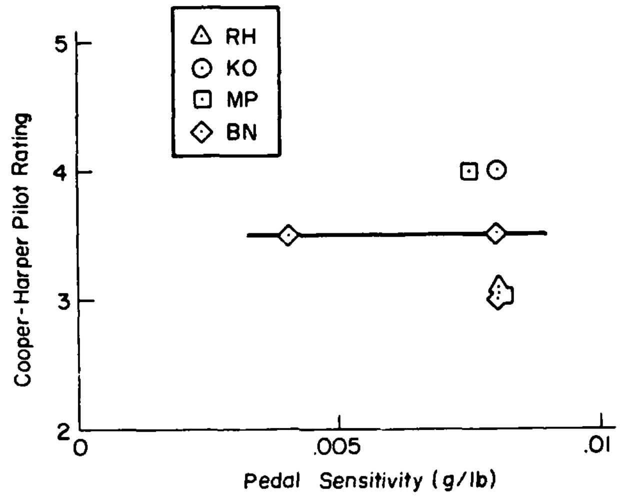


Figure 2 (3.6.1.2.1). Effect of DFC Manipulator Sensitivity
Configuration WLT1 (Very Low Coupling)
(From Reference 115)

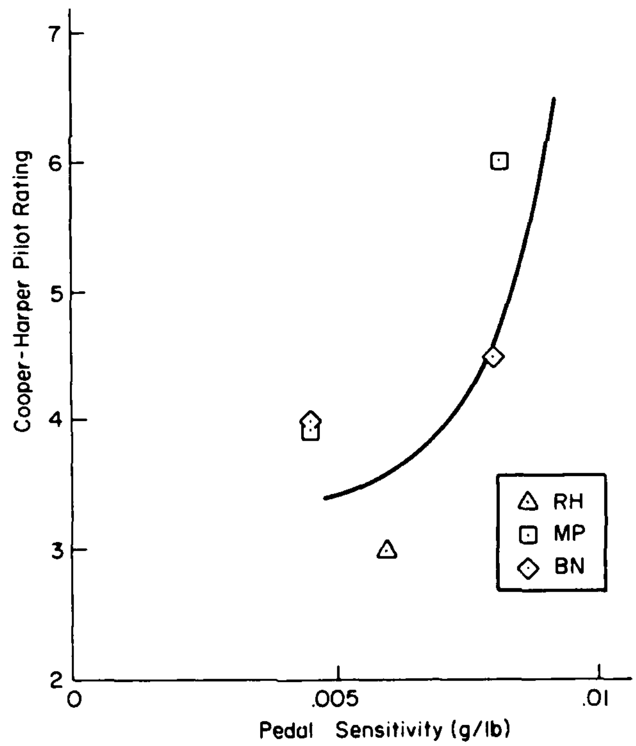


Figure 3 (3.6.1.2.1). Effect of DFC Manipulator Sensitivity
Configuration WLT5 (High Favorable Yaw Coupling)
(From Reference 115)

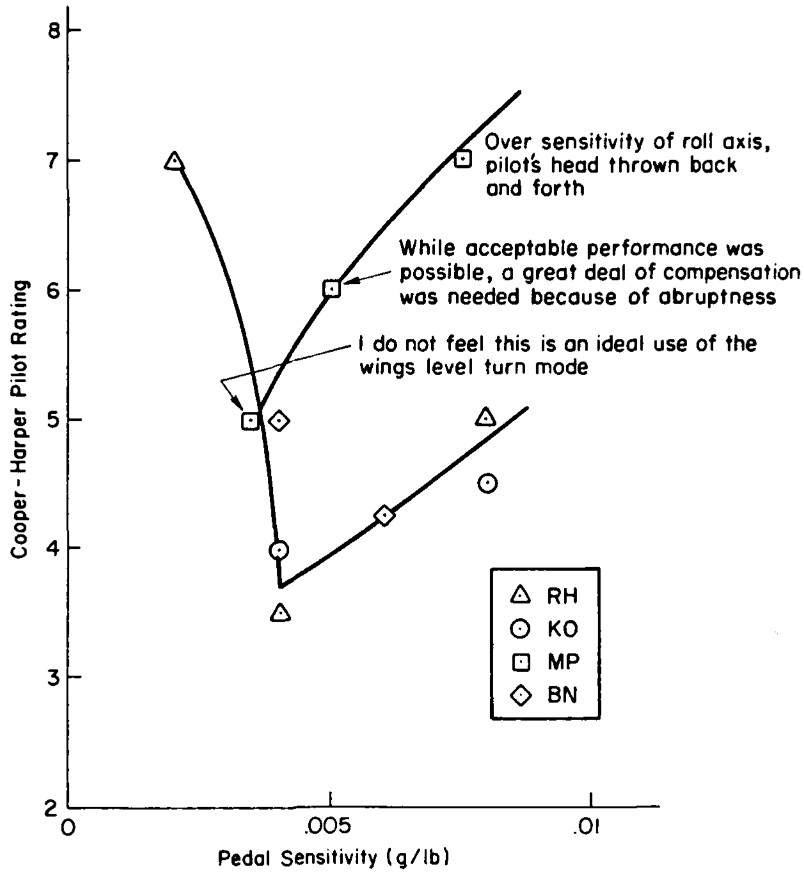


Figure 4 (3.6.1.2.1). Effect of DFC Manipulator Sensitivity Configuration WLT12 (Very High Favorable Roll Coupling) (from Reference 115)

for configurations where the pilot was farther from the roll axis (than in the Navion) and therefore subject to more roll-induced lateral acceleration.

F. DEMONSTRATION OF COMPLIANCE

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 "Demonstration of Compliance" for Para. 3.2.1.1. 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 in Figure 1.

Part (d) 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.

G. SUPPORTING DATA

The variable-stability flight test experiment of Reference 115 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 Reference 115 experiment are plotted versus heading bandwidth in Figure 5 for the air-to-air tracking task using the wings-level turn mode. The open symbols in Figure 5 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 closed symbols in Figure 5 indicate that the heading

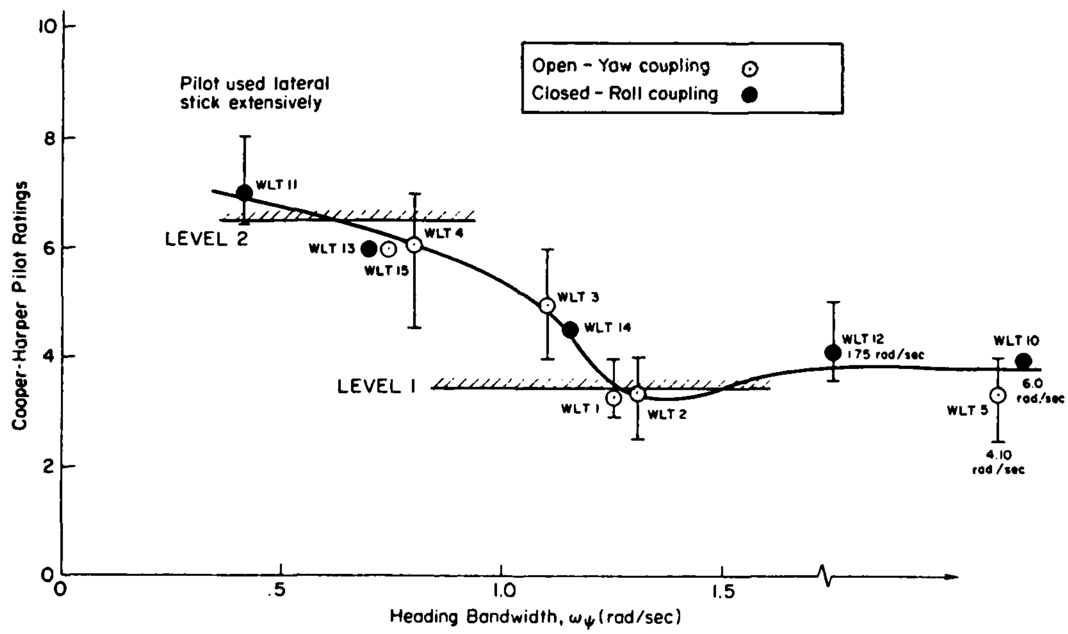


Figure 5 (3.6.1.2.1). Correlation of Pilot Ratings with Heading Bandwidth; Wings-Level Turn Mode; Air-to-Air Tracking Task bandwidth was varied via changes in roll coupling, i.e., the DFC control to aileron gain. To the pilot, favorable yaw coupling appears as a tendency for the nose to abruptly move in the direction of the commanded turn, whereas unfavorable yaw coupling appears as 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 improving the basic response characteristics (provided roll is not too large). Finally, adverse roll coupling appears to the pilot as 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 5:

- The pilot rating 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 5, 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 5 indicates that 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 variable-stability aircraft was the Princeton University Navion, which has an operational speed of 105 kt. 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 (Reference 240) indicates 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 the air-to-air tasks in Reference 115 represent the upper limit of pilot tolerance to lateral accelerations.

The rating data in Figure 5 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 Reference 115 indicates that the WLT1 configuration had very acceptable flying qualities and that the desired performance in tracking was "easily" attained (but apparently involved more than "minimal compensation"). Hence, the inability to attain average pilot ratings better than 3 is felt to be attributable not to the configuration but rather to the difficulty of the task involved. Pilot ratings of 2 for the wings-level turn mode were obtained in Reference 232. 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 Reference 115.

The use of secondary controls was allowed in Reference 115. 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 where 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 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 Reference 115 experiments, the pilots utilized the centerstick any time it appeared as if the target bank angle was excessively large to the point where 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 control were unsuccessful.

The bandwidth requirements stated for the path deviation task (Cat. C in Table 1) are based on the lateral translation mode results given in Reference 115, as well as heading control results obtained for conventional aircraft in previous programs. An example of such results is shown in Figure 6, taken from Reference 146. Figure 6 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 6) as 0.12 rad/sec.

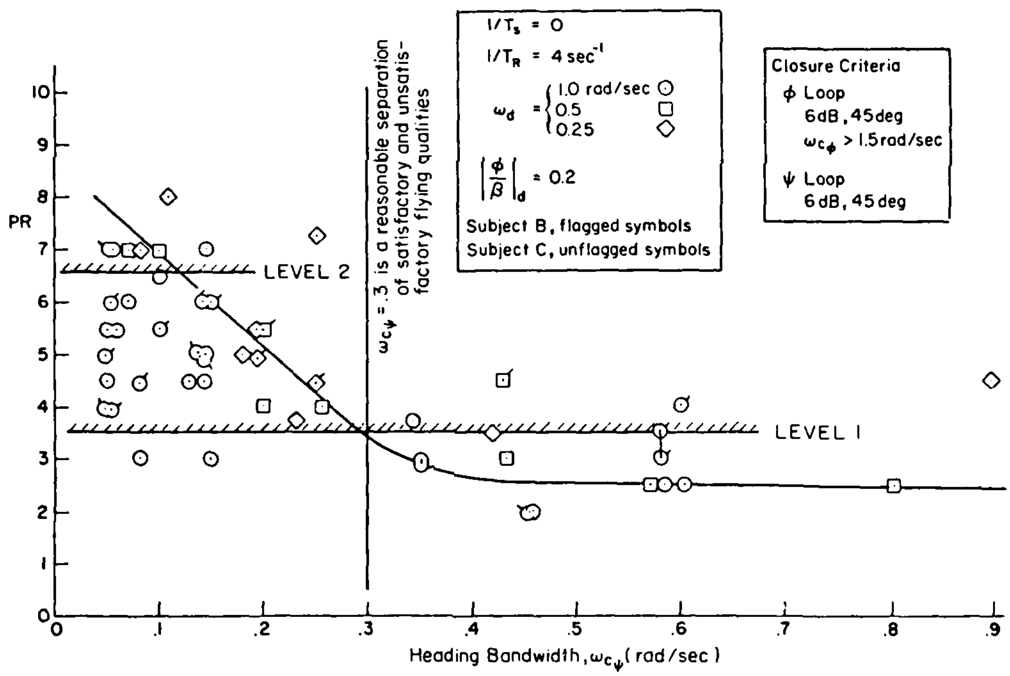


Figure 6 (3.6.1.2.1). Correlation of Pilot Ratings with Heading Bandwidth for Conventional Aircraft; ILS Approach Task

Part (b) of the requirement relates to the turn rate or lateral acceleration necessary to accomplish the desired task. Reference 230 indicated that ±1 lateral g was sufficient for air-to-ground and Reference 115 indicated that ±2.5 lateral g would be required for air combat maneuvering. Obviously, some form of lateral pilot restraint will be required. As mentioned previously, recent results from the AFAMRL centrifuge (Reference 240) have indicated that a restrained pilot can track up to about ±2.5 lateral g.

H. LESSONS LEARNED

The F-16 CCV utilized a wings-level turn mode (Reference 229) 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 providing adequate lateral pilot 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 Reference 231).

3.6.2 Yaw Axis Response to Roll Controller

3.6.2.1 Coordination in turn entry and exit

3.6.2.1.1 Coordination in turn entry and exit — requirement 1

A. REASON FOR REQUIREMENT

This requirement is intended to insure that the yaw control needed to coordinate turns is not objectionable to the pilot. It uses the value of sideslip angle that accompanies roll control inputs as the measure of acceptance. An alternate requirement is presented that places limits directly on the rudder pedal required for turn coordination (3.6.2.1.2).

B. RELATED MIL-F-8785C REQUIREMENTS

3.3.2.4, 3.3.2.4.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.2.1.1 Coordination in turn entry and exit -- requirement 1. The sideslip excursions to step roll control inputs with yaw control free shall meet the following criterion: ___________________.

Recommended values:

a. The amount of sideslip following a yaw-control-free (small input) step roll control command should be within the limits as shown in Figure 1 for Levels 1 and 2. This requirement should apply for step roll control commands up to the magnitude that causes a 60 degree bank angle change with T_d or 2 seconds, whichever is longer.

b. Following a yaw-control-free (large input) step roll control command, the ratio of the sideslip increment, $\Delta\beta$, to the parameter k (4.2.6) should be less than the values specified herein. The roll command should be held fixed until the bank angle has changed at least 90 degrees.

LEVEL	FLIGHT PHASE CATEGORY	ADVERSE SIDESLIP (RIGHT ROLL COMMAND CAUSES RIGHT SIDESLIP)	PROVERSE SIDESLIP (LEFT ROLL COMMAND CAUSES RIGHT SIDESLIP)
1	A	6 degrees	2 degrees
	B and C	10 degrees	3 degrees
2	All	15 degrees	4 degrees

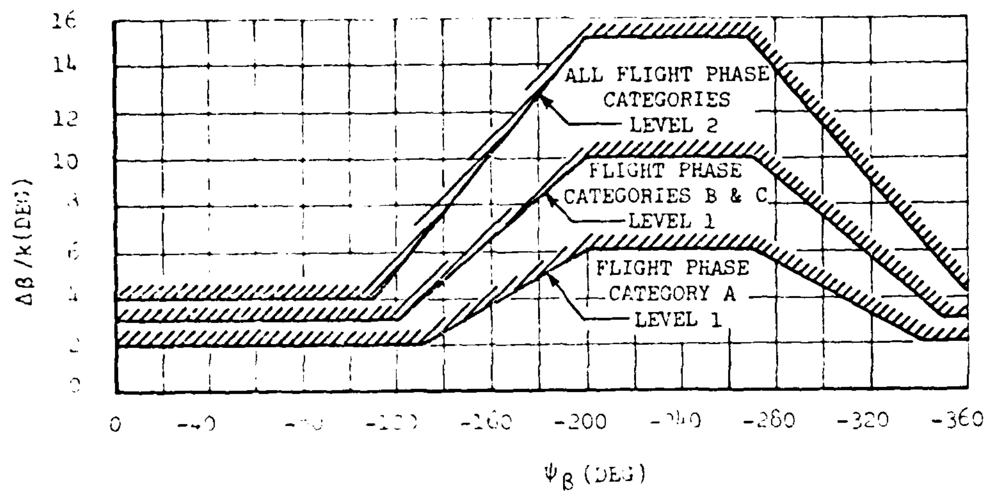


Figure 1 (3.6.2.1.1). Sideslip Excursion Limitations

D. RATIONALE BEHIND REQUIREMENT

This requirement was first introduced in MIL-F-8785B. The following discussion is reprinted from the BIUG for that document (Reference 11).

The primary source of data from which the sideslip requirement evolved is the low $|\phi/\beta|_d$ ($\approx$ 1.5) configurations of Reference 46 (Figure 2). The pilot comments associated with these configurations indicated that the pilots' difficulties were almost exclusively associated with sideslip, rather than with bank angle tracking as was the case for larger $|\phi/\beta|_d$ ($\approx$ 6) configurations.

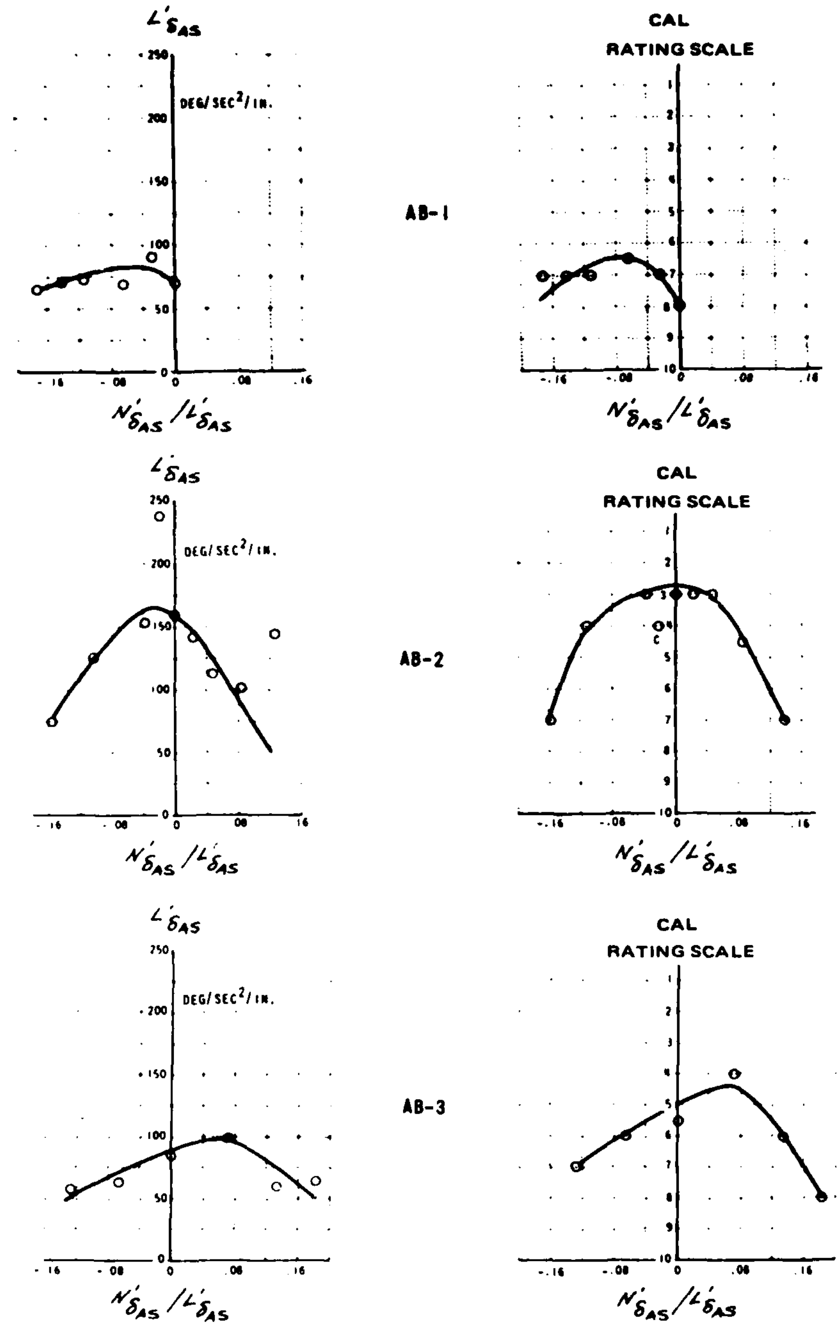


Figure 2 (3.6.2.1.1). Pilot Ratings and Optimum Aileron Sensitivity (Low $|\Phi/\beta|_d$, Medium T_R) (Reference 46)

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, the pilot can tolerate quite a bit of sideslip. On the other hand, when the phasing is such that β is primarily proverse, 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. 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 rudder-pedals-free 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 occurring in sideslip. The latter was a better measure of the amount of coordination required. Thus, the data from this program were plotted in Figure 3 as the maximum change in sideslip occurring during a rudder-pedals-fixed rolling maneuver, $\Delta\beta$, versus the phase angle of the dutch roll component of the sideslip, ψ_β. (Note that the dutch roll damping ratio is Level 2 for the data of Figure 3.)

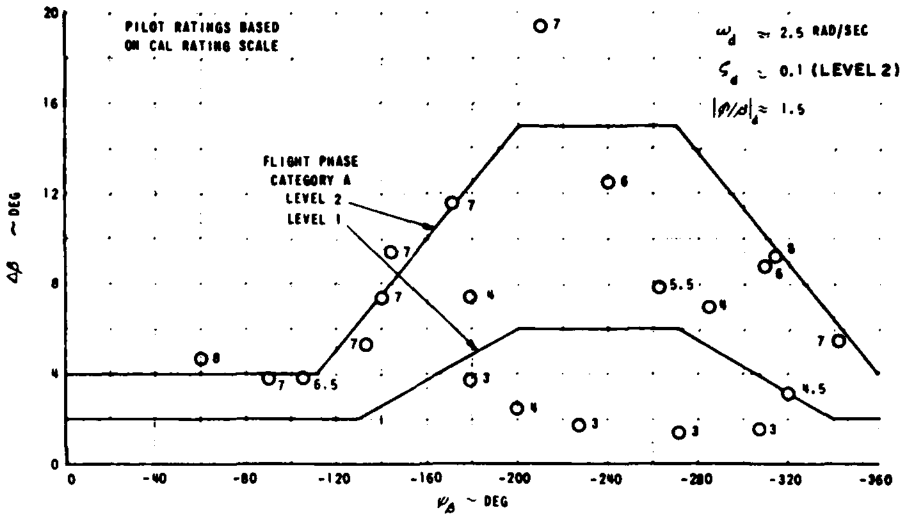
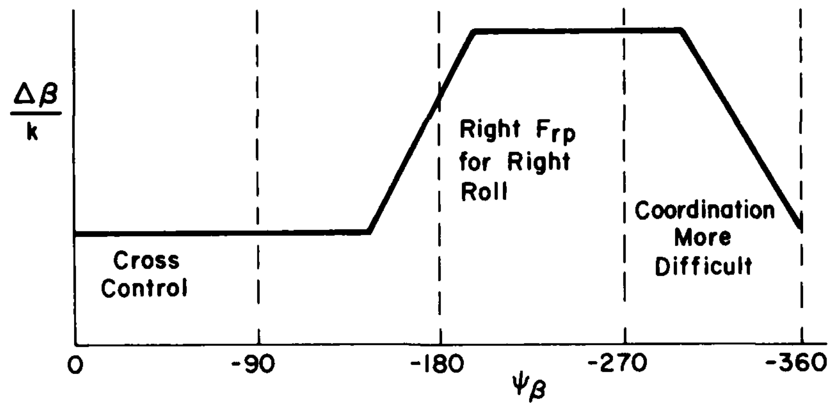


Figure 3 (3.6.2.1.1). Flight Phase Category A Data from Reference 46

The phase angle, ψ_β, is a measure of the sense of the initial sideslip response, whether adverse or proverse, while $\Delta\beta$ is a measure of the amplitude of the sideslip generated. Both the sense and the amplitude affect the coordination problem.

It was observed from examination of the low $|\phi/\beta|_d$ data plotted on Figure 3 that the break points in curves of iso-pilot ratings occurred at almost exactly the same values of ψ_β as for the moderate $|\phi/\beta|_d$ configurations (see the discussion of Paragraph 3.5.1.1.4), even though the degradation of flying qualities was due to sideslip problems 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 ($\Delta\beta$ 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 shown in the sketch on the following page.



As with the p_{osc}/p_{av} requirement, it can be seen from this sketch that the specified value of $\Delta\beta$ varies significantly with ψ_β. This difference is almost totally due to the differences in 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 $\Delta\beta$ with ψ_β is to be expected. For -180 deg $\leq \psi_\beta \leq -260$ deg, coordination may be effected, that is, right rudder pedal for right rolls. Thus, even if large sideslip excursions occur in rudder-pedal-free rolls (the condition under which the $\Delta\beta$ tests are conducted), when coordinating in the normal manner sideslip oscillations can be readily minimized. As ψ_β varies from -270 deg to -360 deg, coordination becomes increasingly difficult, and in the range -360 deg $<$ $\psi_\beta < 90$ deg 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 deg $< \psi_\beta < -90$ deg, oscillations in sideslip either go unchecked or are amplified by the pilot's efforts to coordinate with rudder pedals.

The parameter "k" relates the amount of allowable sideslips to the roll performance requirements. 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.

E. GUIDANCE FOR APPLICATION

While $\Delta\beta$ is relatively straightforward, the parameter k is a function of Flight Phase, airplane Class, speed range, and actual roll performance. In addition, questions arise about the influence of the yaw controller, since the roll performance requirements of 3.5.9.1 allow use of yaw controls in some instances. Therefore, $\Delta\beta$ is a measure of yaw-control-free response, and k is a measure of roll response with combined roll and yaw controls.

Because the required ϕ_t values of 3.5.9.1 are different for Levels 1, 2, and 3, use of k ($\phi t_{command}/\phi t_{requirement}$) 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 $\Delta\beta/k$. The supporting data for this requirement do not show a need for such special treatment: the proper value of $\phi t_{requirement}$ to use is that specified in 3.5.9.1, and the proper value of $\phi t_{command}$ will be the value obtained by performing the tests required by 3.5.9.1. For example, if a prototype Class IV airplane in Flight Phase CO and Speed Range M is required to have Level 1 roll performance, $\phi t_{requirement}$ = 90 deg in 1 sec, and this is the value used for k, no matter what $\phi t_{command}$ is. Clearly, since $k = \phi t_{command}/\phi t_{requirement}$, a poor roll performance (i.e., large $\phi t_{command}$) will <u>increase</u> $\Delta\beta/k$.

As a result, the "correct" way to compute k is with the commanded and required ϕ_t values obtained from 3.5.9.1. The resulting $\Delta\beta/k$, whether Level 1 or not, is then compared to the requirements of 3.6.2.1.1. As shown in "Supporting Data" and "Lessons Learned," 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 airplanes 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 3.5.1, "Roll Response to Roll Controller."). In general, the available data suggest that $\Delta\beta/k$ is not as useful as p_{osc}/p_{av} when $|\phi/\beta|_d$ > 3.5–5.0 (see "Supporting Data").

Finally, some of the shortcomings discussed for p_{osc}/p_{av} in 3.5.1 are relevant here as well: for example, if the spiral mode is divergent, the $\Delta\beta$ response will be heavily influenced by T_s. Similarly, for very low dutch roll frequencies, the step roll control inputs must be very small (see Reference 59).

F. DEMONSTRATION OF COMPLIANCE

The flight testing to obtain $\Delta\beta$ and $\phi t_{command}$ should cover the range of operational altitudes and service speeds. At the least, testing should be performed at the flight conditions corresponding to those of 3.5.1.

G. SUPPORTING DATA

As was discussed in "Rationale Behind Requirement," the data of Reference 46 were the basis for developing the sideslip excursion requirements of MIL-F-8785B. Figure 3 shows the data as presented in Reference 11; in Figure 4, these data are compared directly to the $\Delta\beta/k$ versus ψ_β requirements of Figure 1.*

The configurations of Reference 143 that meet Level 1 dutch roll mode and p_{osc}/p_{av} requirements are illustrated in Figure 5. The Cooper-Harper pilot ratings generally agree with the Level 1 and 2 boundaries. Note that the high $|\phi/\beta|_d$ ($\approx$ 5.0) data correlate well with the boundaries. However, it is generally true that larger values of $|\phi/\beta|_d$ result in small $\Delta\beta/k$, and any roll-yaw coupling problems would more likely show up only on the p_{osc}/p_{av} requirements of 3.5.1.

---

*The values of ψ_β in Figures 3 and 4 do not agree for all data points. The ψ_β and $\Delta\beta/k$ of Figure 4 were taken from Reference 59, as was much of the data used in the following figures. No attempt has been made to account for the differences in ψ_β, or to decide which is the "correct" set of data. The more recent data (Figure 4) result in a slightly poorer correlation with the boundaries than the earlier data (Figure 3) which were used to define the boundaries in the first place.

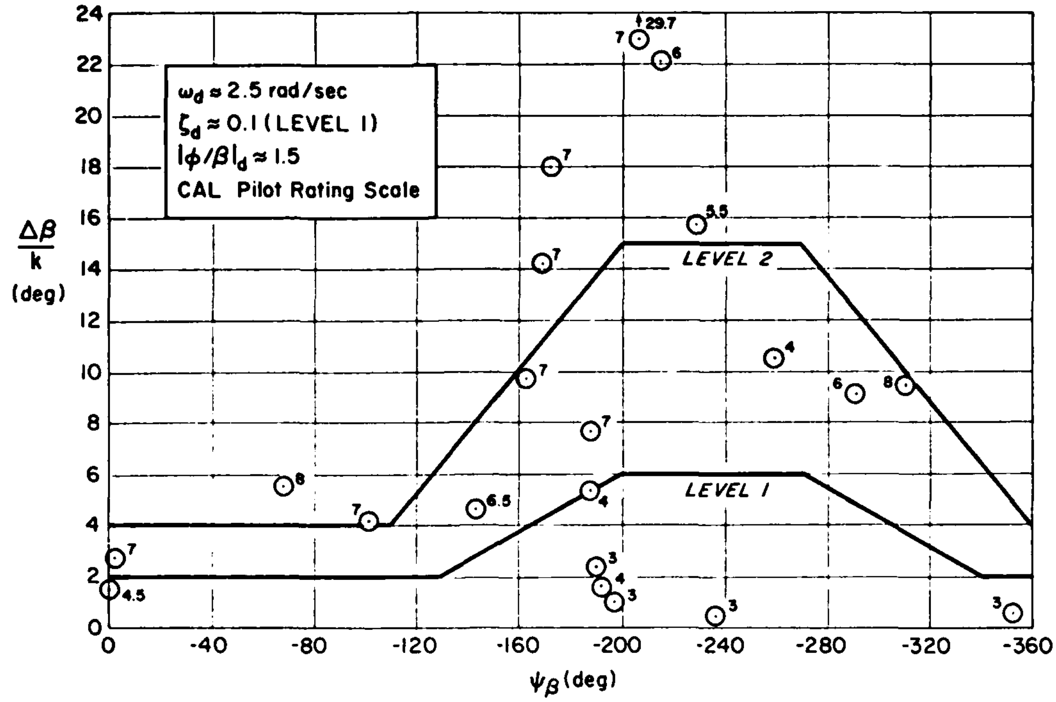


Figure 4 (3.6.2.1.1). Flight Phase Category A Data from Reference 46
($\Delta\beta/k$, ψ_β from Reference 59)

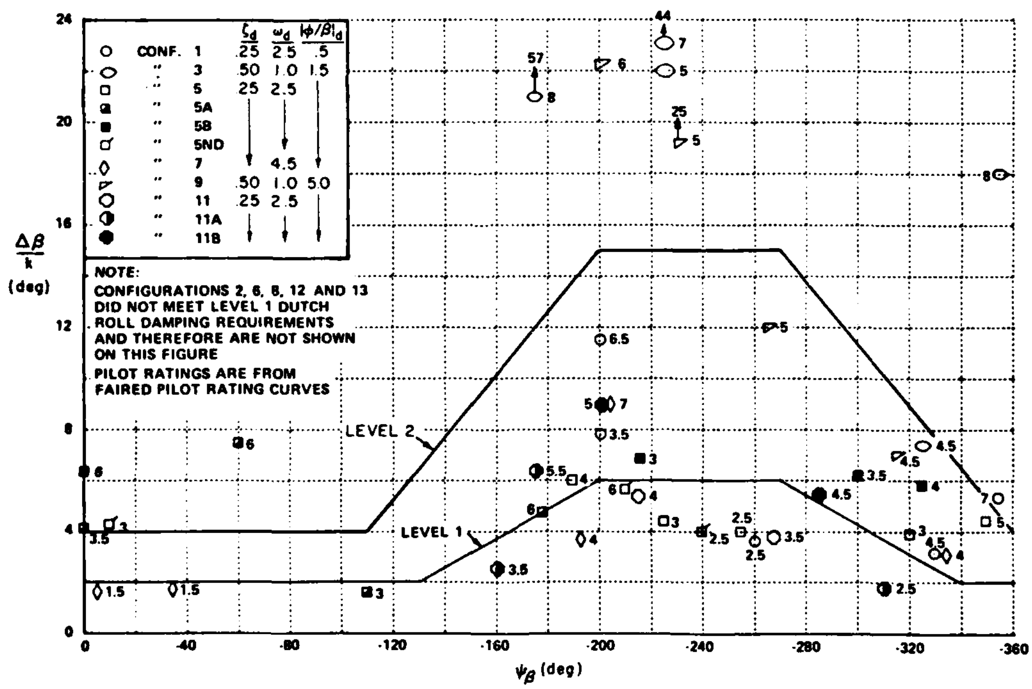


Figure 5 (3.6.2.1.1). $\Delta\beta_{max}/k$ Versus ψ_β for Evaluation Points That Meet Level 1 p_{osc}/p_{av} Criteria, Category A Data (from Reference 143)

Figure 6 compares relevant Category B data of Reference 39 with the boundaries of Figure 1. Again, only those data for which T_s, T_R, ζ_d, ω_d, and p_{osc}/p_{av} are all Level 1 are shown. The low-$|\phi/\beta|_d$ of Figure 6a were used in the MIL-F-8785B BIUG (Reference 11) to develop the Category B boundaries. Figure 6b shows high-$|\phi/\beta|_d$ data; clearly, when $|\phi/\beta|_d$ is large, sideslip excursions are not a problem.

Category C data from Reference 44 (also utilized in the 8785B BIUG to define the Category C limits) are given in Figure 7. The few data points above the Level 1 boundary do not show very good correlation.

Figure 8 shows data from Reference 69, and again correlation is poor: configurations in the Level 2 to 3 regions received Cooper-Harper ratings of 2, 2.5, and 3.

In summary, the supporting data for $\Delta\beta/k$ versus ψ_β appear to show a need for such a criterion for Category A Flight Phases; however, for Categories B and C, other lateral-directional requirements serve well to define acceptable flying qualities. For $|\phi/\beta|_d$ above some nominal value ($\approx$ 5.0), $\Delta\beta/k$ adds little to the specification of flying qualities, and p_{osc}/p_{av} is the important parameter.

As with the roll rate oscillation requirements of 3.5.1.1.4, the sideslip requirements of Figure 1 are applicable for small inputs only. In order to be able to test for large control inputs, an additional but more lenient requirement has been specified. In this way the more comprehensive requirement of Figure 1 on sideslip limitations can be incorporated without losing the ability to flight test for compliance with large control inputs.

H. LESSONS LEARNED

Comparisons of operational aircraft with the sideslip excursion requirements can be obtained from four AFFDL-sponsored validation reports, References 126, 127, 130, and 131, as summarized in the following paragraphs.

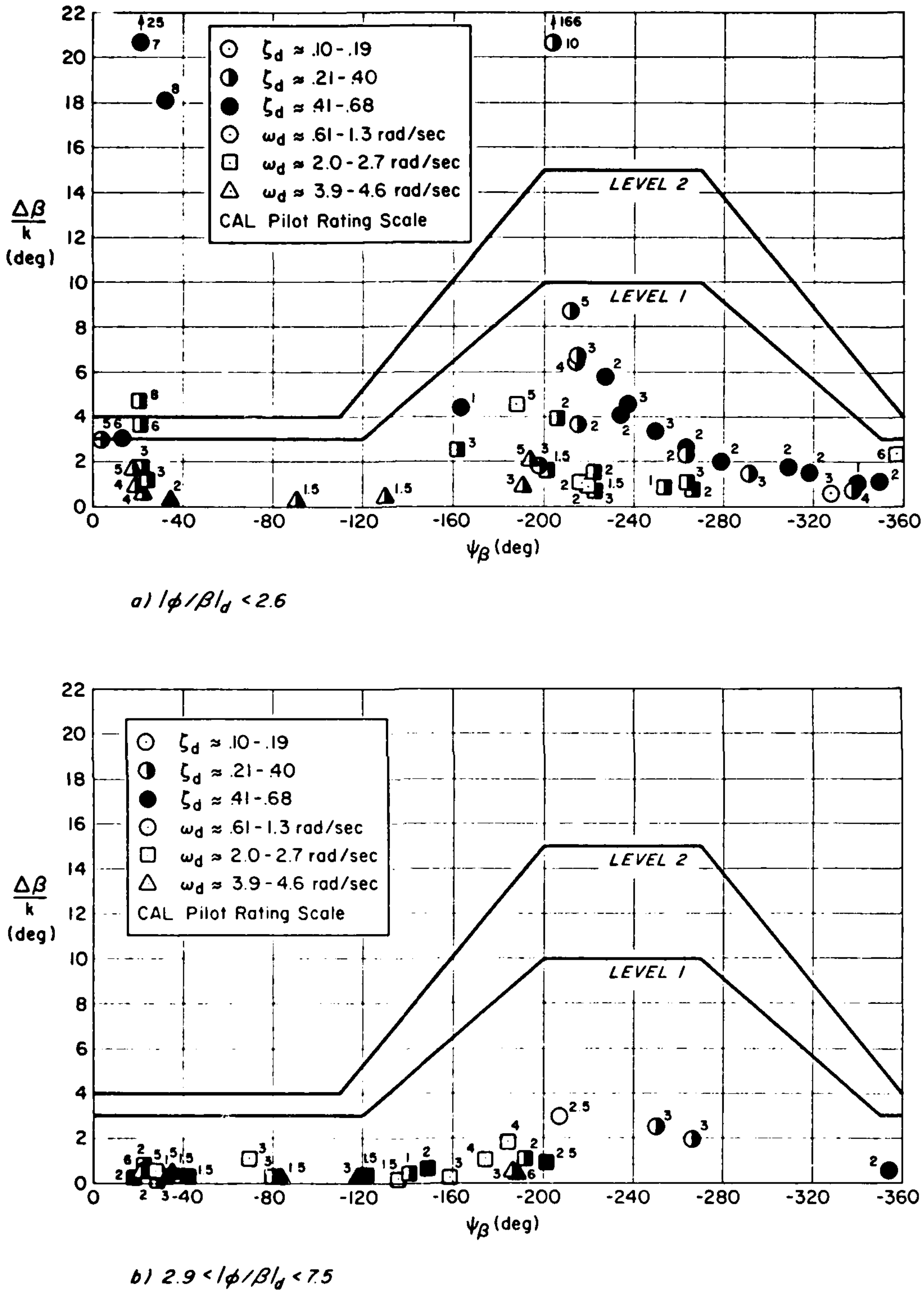


Figure 6 (3.6.2.1.1). Category B Data of Reference 39

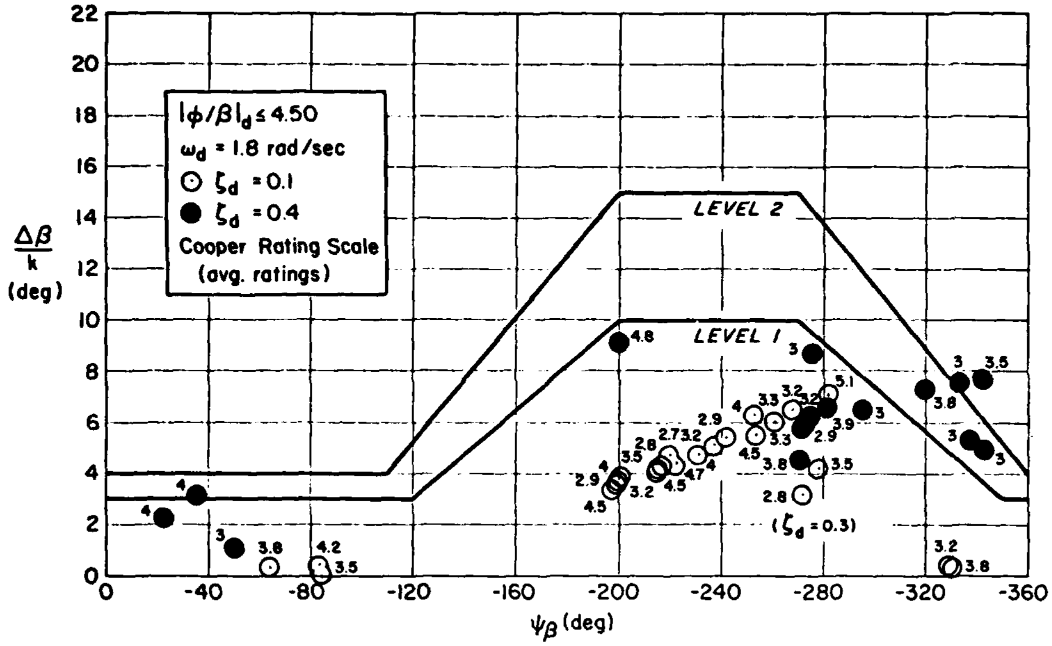
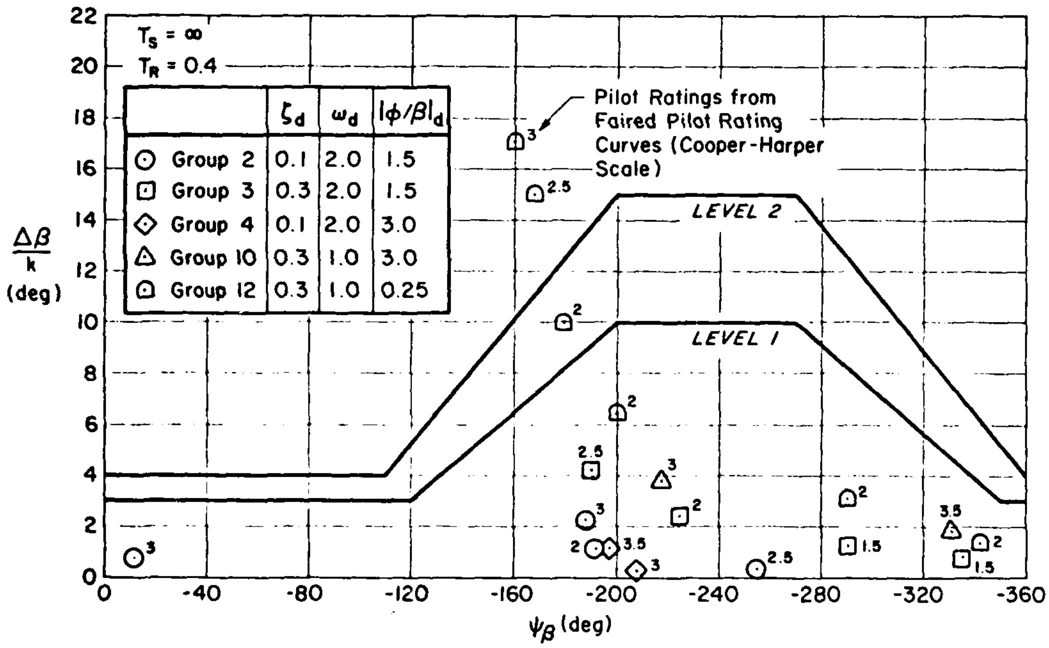


Figure 7 (3.6.2.1.1). Category C Configurations of Reference 44 ($\Delta\beta/k$, ψ_β from Reference 59)



	ζ_d	ω_d	$\lvert\phi/\beta\rvert_d$
Group 2	0.1	2.0	1.5
Group 3	0.3	2.0	1.5
Group 4	0.1	2.0	3.0
Group 10	0.3	1.0	3.0
Group 12	0.3	1.0	0.25

Figure 8 (3.6.2.1.1). $\Delta\beta_{max}/k$ Versus ψ_β for Configurations That Meet Level 1 p_{osc}/p_{av}, ζ_d, and $\zeta_d\omega_d$ Criteria (from Reference 69)

1. P-3B (Class III)

Reference 126 discusses the characteristics of the P-3B and correlation with the large-input sideslip excursion requirements:

> In PA, the adverse yaw observed ranged between about 12 degrees at 120 KEAS to a minimum of 3 degrees at about 170 KEAS [k = 1.0]. The pilot commentary indicated Level 1 flying qualities were associated with these results....The results are felt to substantiate the Level 1 requirement of the specification for Category C Flight Phases.
>
> The [Category A] results....revealed that the adverse yaw characteristics of the aircraft failed to meet the Level 1 requirements of the current specification between 130 KEAS ($\Delta\beta/k = 25^o$) and 190 KEAS ($\Delta\beta/k = 6^o$). In addition, the aircraft failed to meet the current Level 1 requirement in Flight Phase AS between 140 KEAS ($\Delta\beta/k = 13^o$) and about 180 KEAS ($\Delta\beta/k = 6^o$) while meeting the Level 1 requirement up to 360 KEAS ($\Delta\beta/k = 2^o$)....The comments which were received indicate that pilots would normally go ahead and coordinate the turn with pedal and would not be annoyed at the pedal coordination requirement.

2. C-5A (Class III)

Figure 9 shows C-5A flight test results compared with the large-input requirements. According to Reference 127,

> The sideslip excursions are not considered undesirable. Hence, the uniform applicability of the requirements to all classes of aircraft is questioned....The requirement to hold the aileron command fixed until the bank angle has changed at least 90 degrees is unnecessary for Class III aircraft. The aileron command should be held long enough to establish the parameters, $\phi_{t_{command}}$ and $\Delta\beta$....

Additional data for the C-141A, YC-141B, and C-5A are shown in Figure 10, from Reference 14. As described there,

> Pilot rating data obtained during the YC-141B flight test program show a value of 2 (Harper-Cooper Rating Scale) with augmentation operative and 4 with the augmentation inoperative. These data indicate that the handling characteristics correspond to Level 1 conditions even though the data fall outside Level 1 requirements at the lower airspeeds.

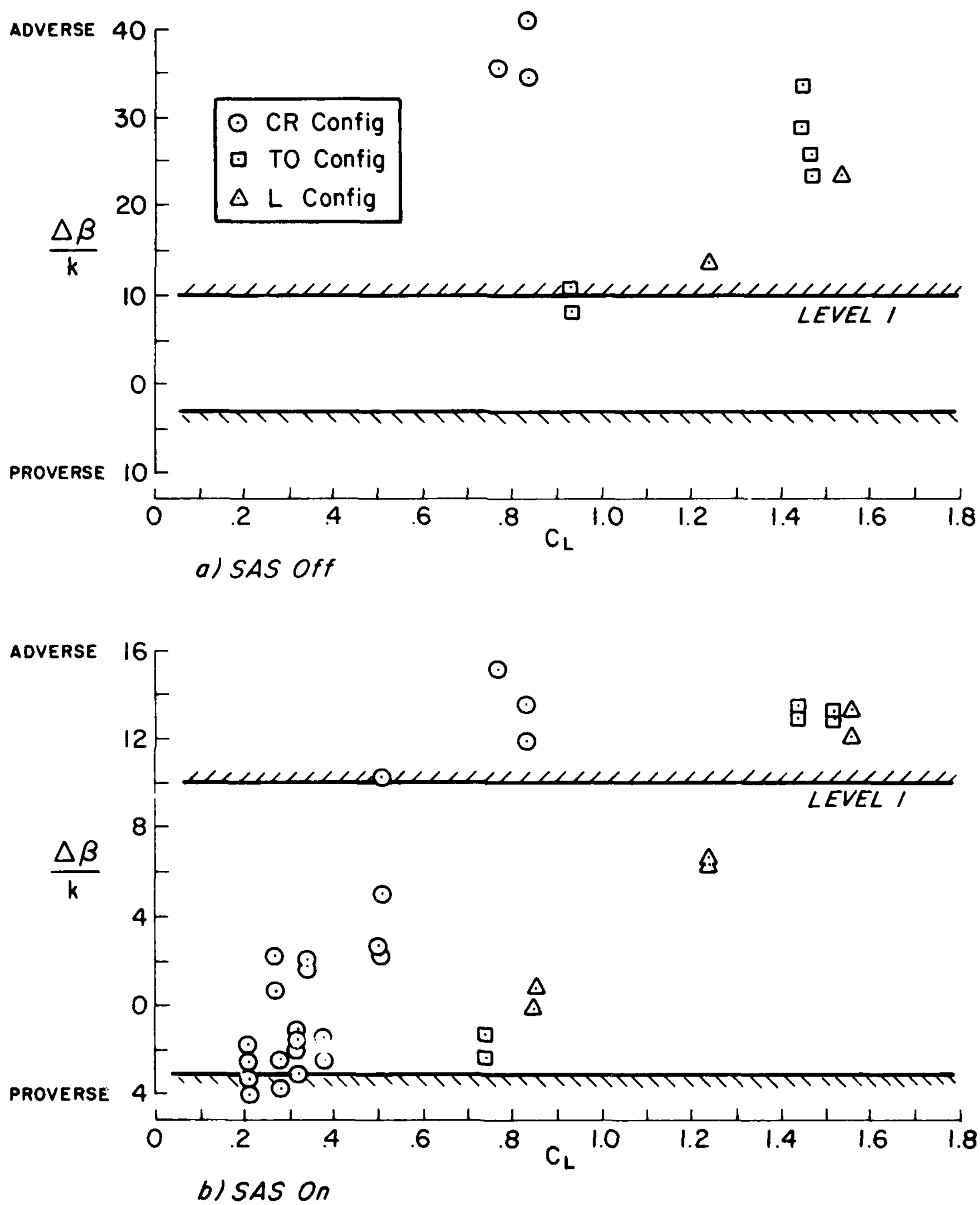


Figure 9 (3.6.2.1.1). C-5A Flight Test Data
(from Reference 127)

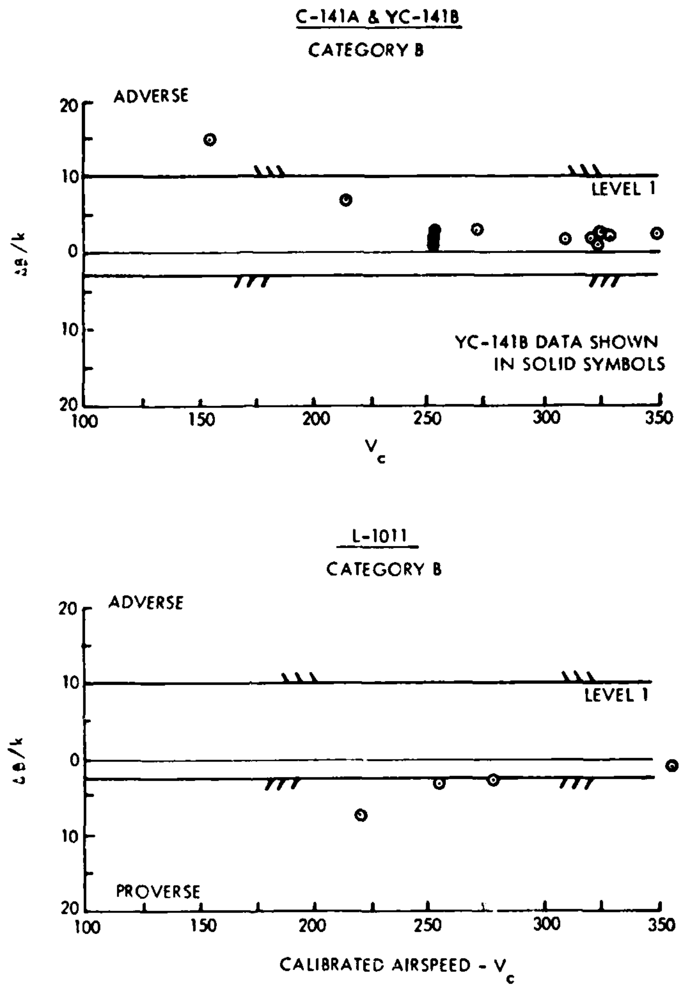


Figure 10 (3.6.2.1.1). Sideslip Excursion Data for Class III Aircraft in Category B Flight Phases (from Reference 14)

It should be noted that the L-1011 nearly complies with roll performance requirements, but the sideslip excursions created as a result, as shown herein, exceed allowable limits [Figure 10]. The L-1011 sideslip excursions have not prompted any objectionable comments from flight test or airline pilots.

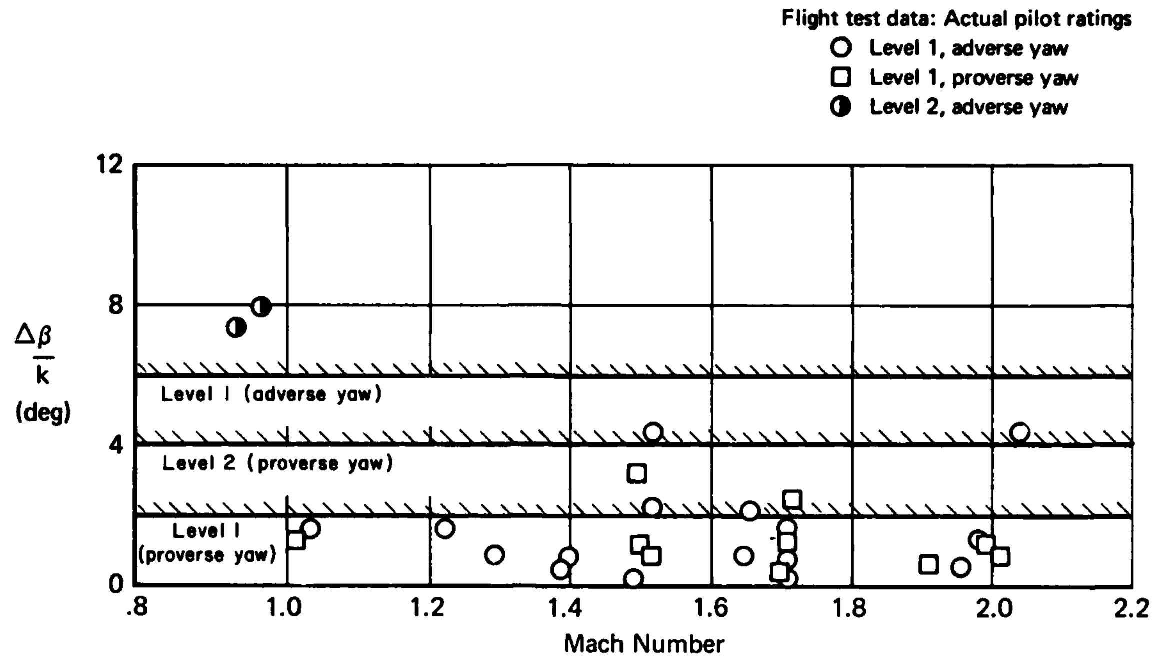
3. F-4 (Class IV)

Flight test data for the F-4H-1 are shown in Figure 11, from Reference 130:

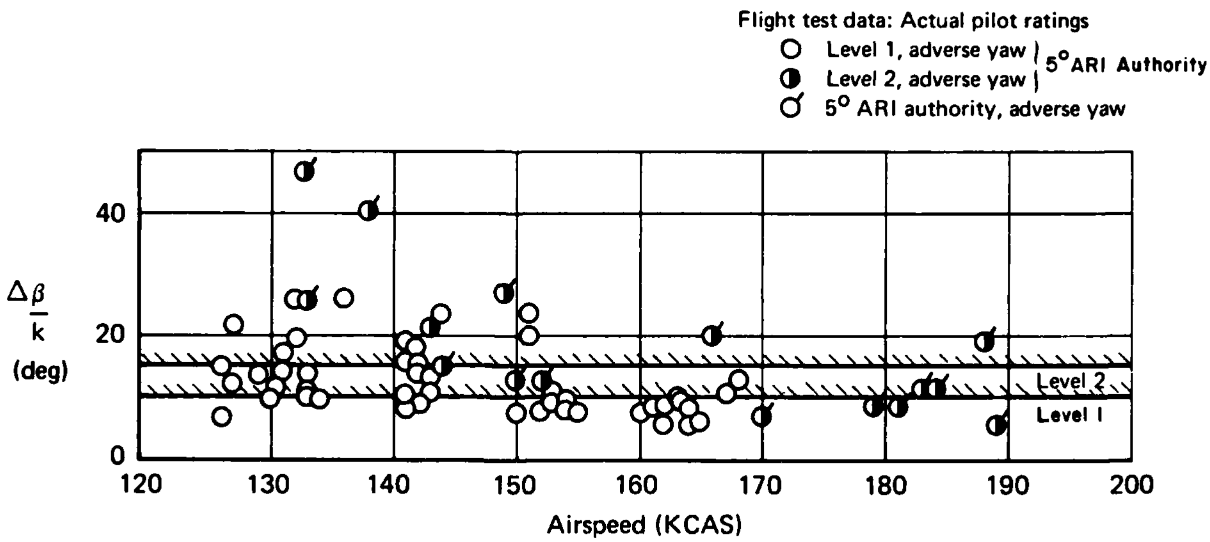
> The Category A data [Figure 11a]...provide good validation of the Level 1 adverse and proverse boundaries with the exception of two test points in which the roll command has induced proverse sideslip. These two points were rated Level 1 but fall outside the Level 1 proverse boundary. Available data do not permit evaluation of the Level 2 boundaries.
>
> The PA data — Category C — [with 5 deg ARI authority, Figure 11b] did not correlate as well. These data were given a blanket rating of Level 2, however a significant number of test points met the specification Level 1 requirements. Each of these had relatively high roll performance resulting in a higher k and a correspondingly higher allowable β. Adverse sideslip was in the low range compared to the other data.
>
> The data of [Figure 11b] -- in which the PA configuration lateral-directional characteristics were modified by increasing ARI rudder authority to $\pm 15^{\circ}$ -- provide inconclusive results. From the pilot comment, an estimated Level 1 was given to all the data. However, approximately half of the data are Level 2 according to the requirement.
>
> When the PA configuration data...are combined as shown in Figure [11b], there is some indication that the Level 1 adverse boundary may be a function of airspeed.

4. F-5A (Class IV)

Test data for the F-5A are compared to the small-input requirements in Figure 12a, and the large-input requirements in Figure 12b, from Reference 131. The F-5A is seen to comply with the limits, though no pilot rating information is given.



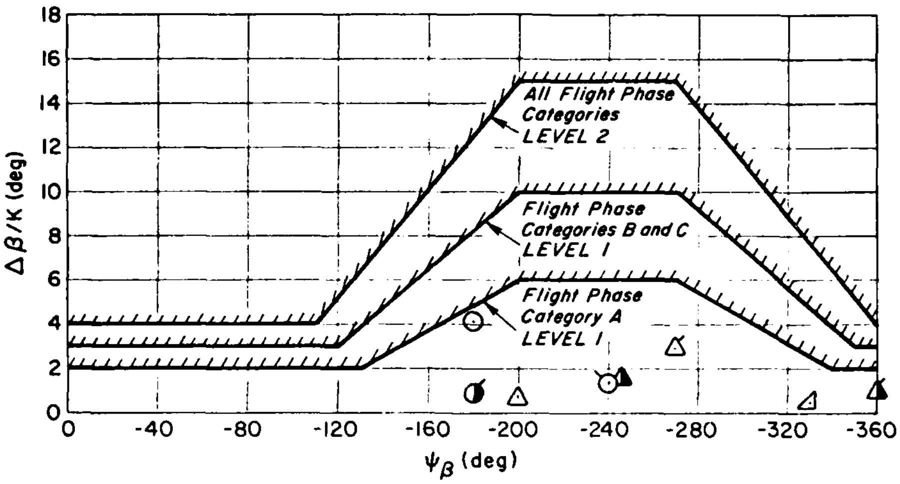
a) Category A Flight Phase



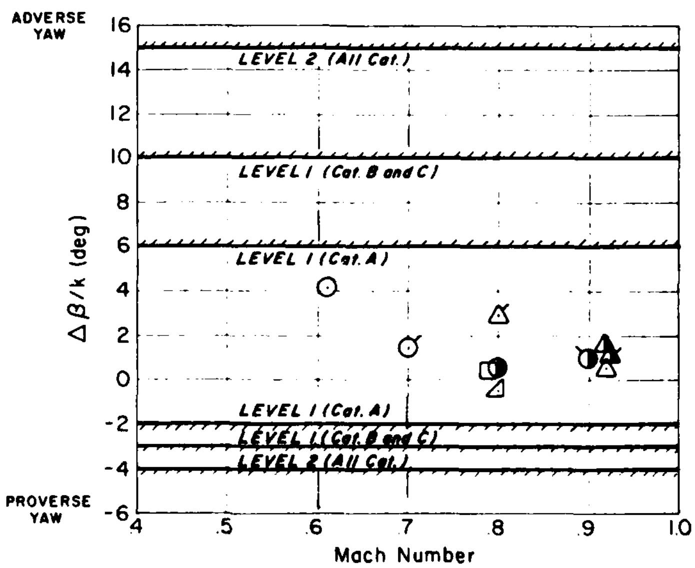
b) Category C Flight Phase

Figure 11 (3.6.2.1.1). Sideslip Excursions for F-4H-1 Airplane (from Reference 130)

Configuration	Altitude (ft)	MN	Entry N_z (g)	Sym
	35,000	.61	1.0	⊙
	10,000	.70	3.0	⊙
	10,000	.80	3.5	◐
	20,000	.90	3.0	◐
	10,000	.92	4.0	△
	20,000	.80	4.0	△
	20,000	.92	4.0	▲
	35,000	.925	-0.2	▲
	9,900	.79	2.7	⊡
	10,000	.80	1.0	◿



a) Small Inputs



b) Large Inputs

Figure 12 (3.6.2.1.1). F-5A Flight Test Data

5. Possible Revision

Calspan proposed a revision to the requirement in Reference 59. The new handling quality parameters proposed there are:

$$\frac{1}{\omega_d} \frac{V_T}{g} \frac{|\Delta\beta|_{t<1.2T_d}}{\hat{\phi}_1} \quad \text{versus} \quad \psi_{\beta_{impulse}}$$

The above noted trend with airspeed is seen to be included in the requirement.

Data comparisons with the Reference 59 parameters show insufficient justification for adopting them. For example, using the data of Reference 46 (Category A), correlations are almost identical for the Calspan revision and for $\Delta\beta/k$ vs. ψ_β (48 percent vs. 46 percent). Likewise, the Reference 69 data (Category C) show no real improvement (70 percent vs. 66 percent). Some parameters similar to those above may be in order, but further work is necessary.

3.6.2.1.2 Coordination in turn entry and exit — requirement 2

A. REASON FOR REQUIREMENT

This requirement is offered as an alternative to $\Delta\beta/k$ versus ψ_β. It is stated directly in terms of the required magnitude and shaping of rudder pedal inputs to coordinate turns. Hence it should be used when analytical guidance is a primary factor. Straightforward application requires that the aircraft sideslip response to aileron and rudder transfer functions be available.

B. RELATED MIL-F-8785C REQUIREMENT

None.

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.2.1.2 Coordination in turn entry and exit — requirement 2. The yaw control characteristics required to maintain zero sideslip for roll control inputs shall meet the following criterion: ______________.

Recommended values: The yaw control crossfeed required to maintain zero sideslip should be within the limits shown in Figure 1 for Levels 1 and 2. In addition, for values of the ratio of yawing to rolling accelerations due to roll controller, $|N'_{\delta_{as}}/L'_{\delta_{as}}|$, less than 0.07, the crossfeed parameter δ'_{rp} must be within the limits of Table 1.

TABLE 1 (3.6.2.1.2)

LIMITS ON $\delta'_{rp}(3)$ FOR $|N'_{\delta_{as}}/L'_{\delta_{as}}| < 0.07$

LEVEL	ADVERSE YAW	PROVERSE YAW
1	-0.39	0.11
2	-1.15	0.78

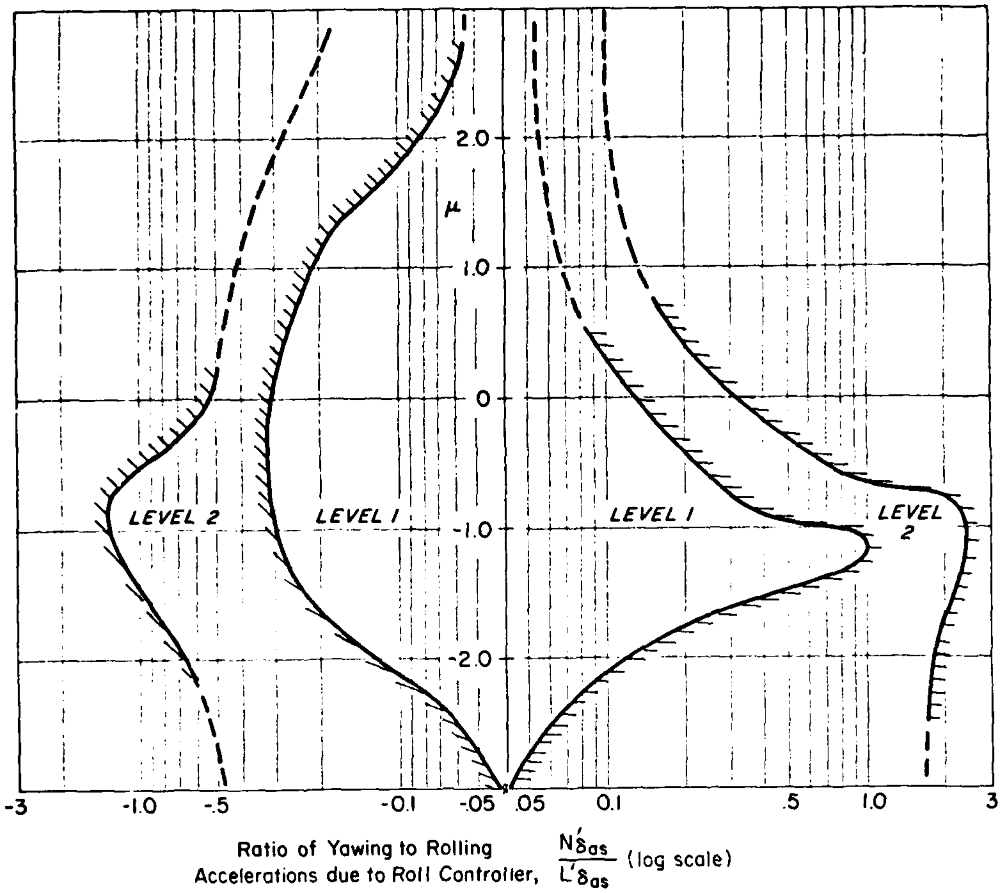


Figure 1 (3.6.2.1.2). Crossfeed Parameter Boundaries

D. RATIONALE BEHIND REQUIREMENT

The ability to make precise changes in aircraft heading is a key factor in pilot evaluation of lateral-directional handling qualities. Assuming other good qualities (e.g., adequate roll response, yaw frequency/damping, etc.), deficiencies in heading control, which can nevertheless exist, 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 ($\Delta\beta/k$, or References 59, 144, 145) are based on aileron-only parameters, and the effects of rudder control are only indirectly apparent as they may have influenced individual pilot ratings. The fact that these criteria are not satisfactory is shown in Reference 146, 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 airplane 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 2 contains a set of requirements intended to serve as a checklist for good roll control.

1. Analysis and Basic Concept

In general, coordinated flight implies minimum yaw coupling due to roll entries and exits which can be quantified in many ways, e.g.: 1) zero sideslip angle ($\beta = 0$); 2) zero lateral acceleration at the c.g.; 3) turn rate consistent with bank angle and speed ($r = g\phi/U_0$); and 4) zero lateral acceleration at the cockpit (ball in the middle).

Conditions 1 through 3 are equivalent when the side forces due to lateral stick, $Y_{\delta_{as}}$, and rudder pedal, δ_{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 mainly associated with ride qualities and not with heading control itself (Reference 146). Based on these considerations it appears that sideslip angle is an appropriate indicator of turn coordination.* Accordingly, the following

*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 nose of the aircraft tracks bank angle. In fact, when turn coordination is critical, such 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. Therefore, we feel that β and not a_y is the appropriate parameter. For most aircraft the difference between a_y and β is very small.

TABLE 2 (3.6.2.1.2). GROUND RULES FOR APPLICATION OF RATING DATA TO HEADING CONTROL CRITERIA

1) T_R meets requirements of Para. 3.5.1.1.1

2) ω_d meets requirements of Para. 3.6.1.1

3) ζ_d meets requirements of Para. 3.6.1.1

4) $|\phi/\beta|_d \leq 1.5$ when turbulence is a factor and $|N'_{\delta_{as}}/L'_{\delta_{as}}| > 0.03$

5) Meets L'_β vs. ω_d boundaries when $|N'_{\delta_{as}}/L'_{\delta_{as}}| < 0.03$ (Figure 2)

6) Meets Level 2 p_{osc}/p_{av} (Para. 3.5.1.1.4)

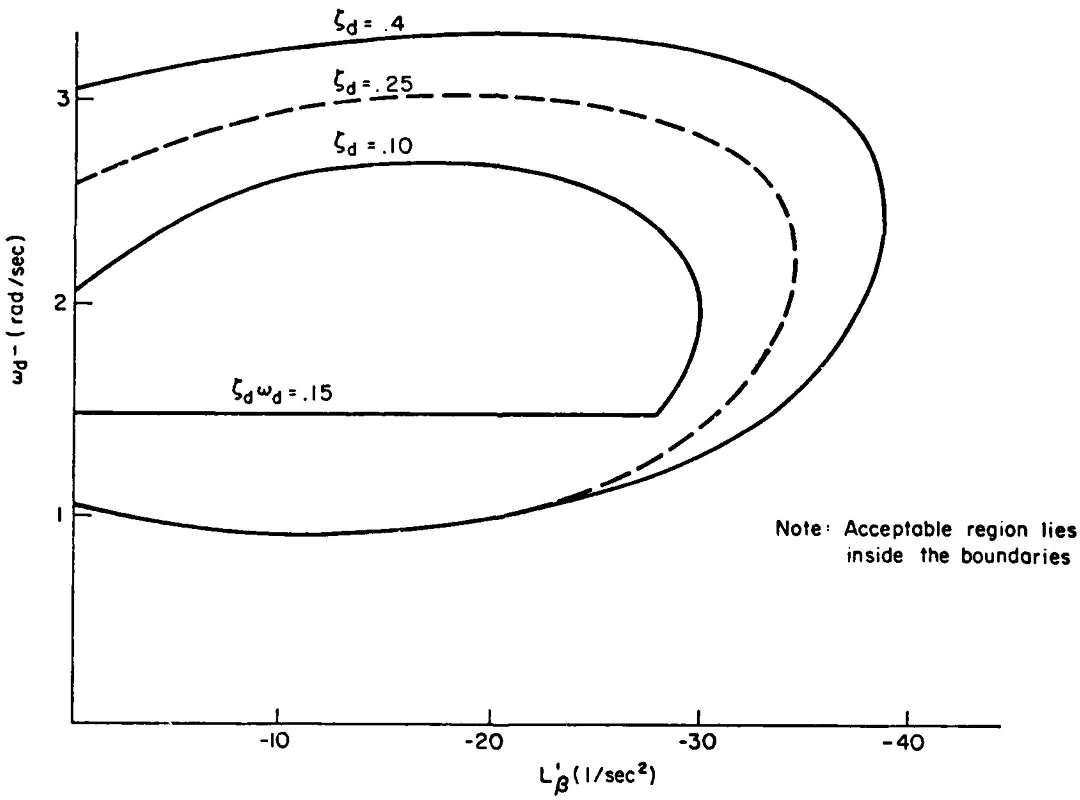


Figure 2 (3.6.2.1.2). Pilot Rating Boundaries for Acceptable Roll Control in Turbulence with $N'_{\delta_{as}}/L'_{\delta_{as}} < 0.03$ (from Reference 68) formulation undertakes to identify the parameters that govern the aileron-rudder shaping required to maintain coordinated flight as defined by zero sideslip angle ($\beta = 0$).

With an aileron-rudder crossfeed, Y_{CF}, the rudder, by definition, is given by

$$\delta_{rp} \equiv Y_{CF}\delta_{as} \tag{1}$$

where δ_{rp} is the rudder pedal deflection at the cockpit; and δ_{as} is the lateral stick (or wheel) deflection at the pilot's grip. For the assumed ideal (zero sideslip) coordination

$$\beta = \frac{N^{\beta}_{\delta_{as}}}{\Delta} + Y_{CF}\frac{N^{\beta}_{\delta_{rp}}}{\Delta}\,\delta_{as} \equiv 0 \tag{2}$$

See Reference 66 for the definitions of $N^{\beta}_{\delta_{as}}$ and $N^{\beta}_{\delta_{rp}}$. Therefore, the ideal crossfeed is:

$$Y_{CF} \equiv \frac{\delta_{rp}}{\delta_{as}} = -\frac{N^{\beta}_{\delta_{as}}}{N^{\beta}_{\delta_{rp}}} \tag{3}$$

For augmented airplanes these numerators are high order and cannot be generalized. However, as was shown in Reference 87, 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 3.2.1.1) has shown that a HOS which cannot be fit to a LOS form is generally unsatisfactory to the human pilot.

Based on the approximate factors for conventional airplanes obtained from Reference 66, the appropriate LOS form for Y_{CF} is:

$$Y_{CF} = \frac{N'_{\delta_{as}}[s + A_{as}(g/U_o)][s + (1/T_{\beta_{as}})]}{Y^*_{\delta_{rp}}[s + A_{rp}(g/U_o)][s + (1/T_{\beta_{rp}})][s - (N'_{\delta_{rp}}/Y^*_{\delta_{rp}})]} \tag{4}$$

where

$$A_i = \frac{L'_r - (L'_{\delta_i}/N'_{\delta_i})N'_r}{L'_p - (L'_{\delta_i}/N'_{\delta_i})[N'_p - (g/U_o)]}$$

$$1/T_{\beta_i} = -L'_p + (L'_{\delta_i}/N'_{\delta_i})[N'_p - (g/U_o)]$$

and i = as (aileron stick) or rp (rudder pedal). For the frequency range of interest, i.e., excluding both low and high frequencies $[A_i(g/U_o) << s << N'_{\delta_{rp}}/Y^*_{\delta_{rp}}]$,

$$Y_{CF} \doteq - \frac{N'_{\delta_{as}}[s + 1/T_{\beta_{as}})]}{N'_{\delta_{rp}}[s + 1/T_{\beta_{rp}})]} \tag{5}$$

To provide a meaningful reference for the control cross-coupling term, $N'_{\delta_{as}}$, in Equation 5, it is expressed as the ratio of yawing to rolling acceleration, $N'_{\delta_{as}}/L'_{\delta_{as}}$. Also, since the rudder sensitivity can be separately optimized and does not usually represent a basic airframe limitation, it is appropriate to remove it from consideration. Accordingly, the *resulting LOS representation of the crossfeed*, Y'_{CF}, is given as*:

$$Y'_{CF} \equiv Y_{CF}\frac{N'_{\delta_{rp}}}{L'_{\delta_{as}}} = -\frac{N'_{\delta_{rp}}}{L'_{\delta_{as}}}\frac{N^{\beta}_{\delta_{as}}}{N^{\beta}_{\delta_{rp}}} \doteq -\frac{N'_{\delta_{as}}[s + 1/T_{\beta_{as}})]}{L'_{\delta_{as}}[s + (1/T_{\beta_{rp}})]} \tag{6}$$

Equation 6 indicates that the aileron-to-rudder shaping required to maintain coordinated flight ($\beta = 0$) is directly related to the separation between the aileron (wheel or stick) and rudder (pedal) sideslip zeros.

As a basis for direct correlation with pilot opinion, a "rudder shaping parameter," μ, is arbitrarily defined as the separation between $1/T_{\beta_{rp}}$ and $1/T_{\beta_{as}}$ and normalized by $1/T_{\beta_{rp}}$, i.e.,

*All derivatives are in the stability axis system.

$$\mu = \frac{(1/T_{\beta_{as}}) - (1/T_{\beta_{rp}})}{1/T_{\beta_{rp}}}$$

which simplifies to

$$\mu = (T_{\beta_{rp}}/T_{\beta_{as}}) - 1 \tag{7}$$

The frequency response characteristics of Y_{CF}', Equation 6, as a function of the sign of μ are shown in Figure 3 in terms of literal expressions for the Bode asymptotes. These asymptotes indicate that the <u>magnitude</u> of the coordinating rudder is a function of $N_{\delta_{as}}'/L_{\delta_{as}}'$* at all frequencies and that the <u>shaping</u> of the rudder response is determined by μ. These parameters are summarized in terms of their analytical and pilot-centered functions in Table 3.

The parameters $N_{\delta_{as}}'/L_{\delta_{as}}'$ and μ are a natural choice for correlation of heading control pilot rating data since they completely define the aileron-to-rudder crossfeed necessary for turn coordination. Such an ideal crossfeed is difficult to isolate with simple flight test procedures, but is nevertheless considered a viable correlation concept because the Military Standard permits analysis methods to demonstrate specification compliance.

TABLE 3 (3.6.2.1.2)

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.
$N_{\delta_{as}}'/L_{\delta_{as}}'$	Defines magnitude of Y_{CF}'	Determines magnitude or rudder required and/or high-frequency yawing induced by aileron inputs.

*Note that $N_{\delta_{as}}'/L_{\delta_{as}}'$ is simply $\dot{r}_o/\dot{p}_o$ for a δ_{as} input.

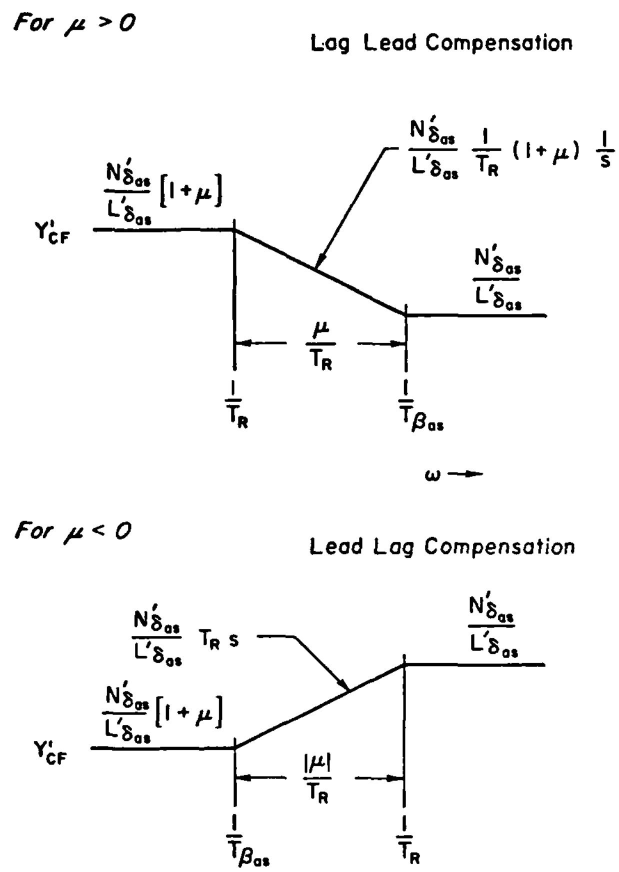


Figure 3 (3.6.2.1.2). Asymptotes of Aileron-Rudder Crossfeed

Since the rudder sequencing with aileron inputs is the key issue, it was decided to use a LOS form in the time domain. The rudder shaping was derived by using a unity high-frequency gain for the Equation 6 form, and the ideal definition of μ (Equation 7), the rudder time history required to coordinate a unit step wheel or stick input, is:

$$\delta_{rp}(t) = 1 + \mu\left(1 - e^{-t/T_{\beta_{rp}}}\right) \tag{8}$$

Note that $\delta_{rp}(t)$ refers to the rudder pedal motion (thereby including effects of rudder gearing and accounting for the SAS). Solving Equation 8 for the rudder shaping parameter, μ:

$$\mu = \frac{\delta_{rp}(t) - 1}{1 - e^{-t/T_{\beta_{rp}}}} \tag{9}$$

The value of t is properly set by the lower limit on the frequency range of interest for piloted heading control. The simulation experiments of Reference 147 indicated that a minimum heading crossover of about 1/3 rad/sec was necessary for desirable handling qualities. Therefore, a corresponding time of 3 sec was selected as being most pertinent to a pilot-centered characterization of crossfeed properties. Recognizing further (Equation 4) that $T_{\beta_{rp}} \doteq -1/L_p'$ is approximately equal to 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 (3.5.1.1.1) sets the following limits on the exponential in Eq. 9.

$$\begin{array}{llll} T_R < 1.0\ ^{*} & & e^{-3/T_R} < 0.049 \\ < 1.4\ ^{**} & & \phantom{e^{-3/T_R}} < 0.117 \end{array}$$

---

*For small, light, or highly maneuverable airplanes.

**For medium to heavy weight, low to medium maneuverability airplanes.

Accordingly, Equation 9 reduces within a maximum error of 5-10 percent, depending on airplane class, to

$$\mu \doteq \delta_{rp}(3) - 1 \qquad (10)$$

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, the normalized rudder parameter, $\delta_r(3)$, may be calculated from Y_{CF} (Equation 3) 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 washed out yaw rate feedback and a first-order lagged aileron/rudder crossfeed results in seventh-order β numerators of unaugmented airplanes. 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 should 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 Reference 148 configurations used in the pilot rating correlations is given as:

$$\frac{\delta_{rp}}{\delta_{as}} = \frac{.19(s - .102)(s - .922)(s + 605.2)}{(s - .057)(s + 5.6)(s + 109.9)} \qquad (12)$$

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:

$$\frac{\delta_{rp}}{\delta_{as}} = \frac{(s - .102)(s - .922)}{(s - .057)(s + 5.6)} \tag{13}$$

The rudder time responses to a unit wheel input for Equations 12 and 13 are plotted in Figure 4. 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 1/2 sec later). The value of μ corresponding to this response is $\delta_r(3) - 1 = -1.17$.

Figure 5 presents typical coordinating ($\beta = 0$) rudder time histories for step aileron inputs on a grid of μ versus $N'_{\delta_{as}}/L'_{\delta_{as}}$. Moving vertically on this grid changes the shape (μ) of the crossfeed, Y_{CF}, keeping the initial value (high-frequency gain) constant. Moving horizontally produces a change in the crossfeed gain $\left(N'_{\delta_{as}}/L'_{\delta_{as}}\right)$ at all frequencies

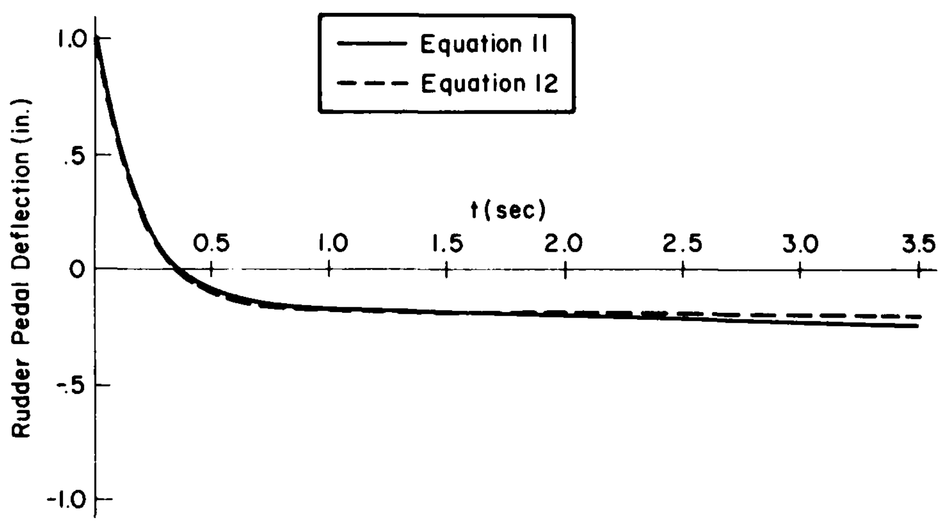


Figure 4 (3.6.2.1.2). Effect of Removing High-Frequency Roots from β Numerators

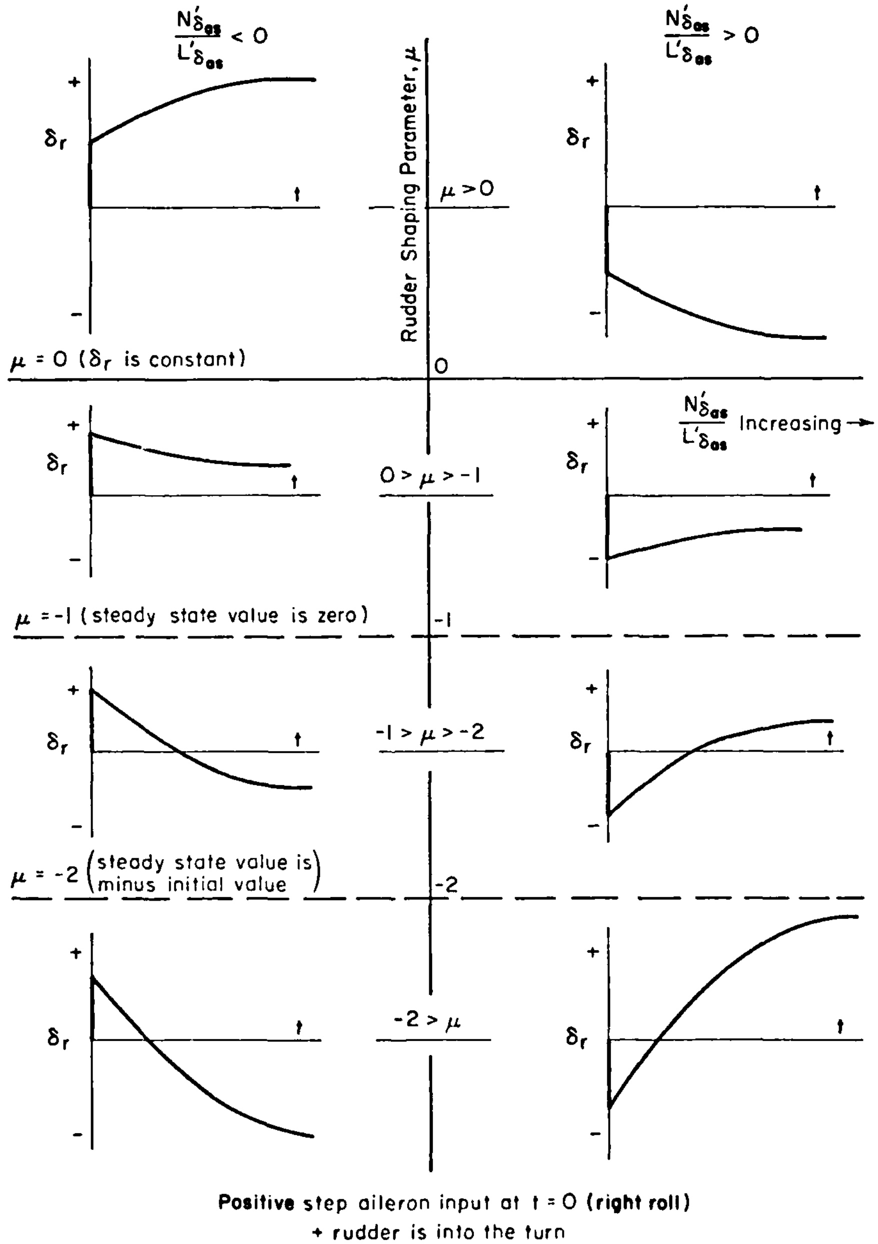


Figure 5 (3.6.2.1.2). Typical Rudder Time Histories for Zero Sideslip without changing the shape. Note that this is consistent with Table 2 and Figure 3, where it is shown that μ dictates the required aileron-to-rudder shaping and $N'_{\delta_{as}}$ defines the magnitude of the gain for all times (and frequencies). The basic shapes of the time histories in Figure 4 are indicative of the fundamental assumption that the rudder time history can be fit by the Equation 6 form. The basic implication of this form is that, after an initial input proportional to the aileron stick step, the rudder response is essentially monotonic in the frequency range of interest.

A physical interpretation relating the cross-coupling derivatives $N'_{\delta_{as}}$ and N'_p with the rudder shaping parameter, μ, is given in Table 4.

TABLE 4 (3.6.2.1.2)

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.
$\mu = 0$	$N'_p = g/U_o$, 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 in 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 complex rudder reversals for coordination. If rudder not used, the nose will appear to oscillate during turn entry and exit.

2. Physical Interpretation

The iso-opinion lines in Figure 1 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:

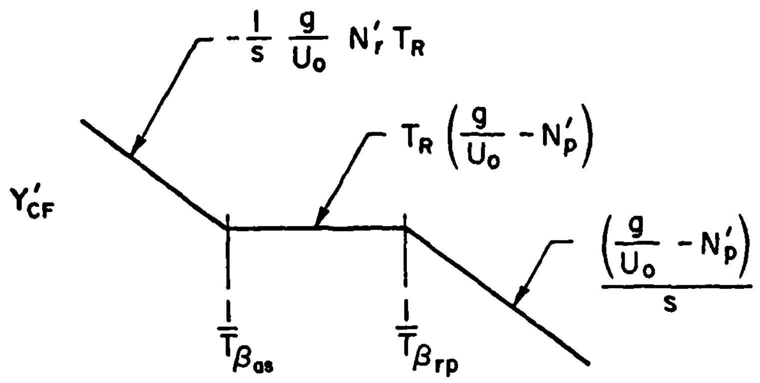
1) Moderately high proverse (positive) $N'_{\delta_{as}}$ is acceptable in the region where $\mu \doteq -1$. Physically, this corresponds to a sudden initial heading response in the direction of turn followed by decreasing rudder requirements. (Required steady-state rudder is zero when $\mu = -1$, see Figure 4). 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 greater than -1. Physically this corresponds to an increase in the requirement for low-frequency cross-control rudder activity (see Figure 4), which is highly objectionable.

3) The pilot ratings are less sensitive to the required rudder shaping when $N'_{\delta_{as}}$ is negative (adverse yaw). Recall that adverse yaw is consistent with conventional piloting technique (rudder with the turn to augment roll into the turn).

3. $N'_{\delta_{as}}/L'_{\delta_{as}}$ Near Zero

Control cross-coupling effects are obviously not a factor when $|N'_{\delta_{as}}/L'_{\delta_{as}}|$ 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). For $N'_{\delta_{as}}/L'_{\delta_{as}}$ identically zero, the required aileron-rudder crossfeed takes the Bode asymptote form shown in Figure 6 for unaugmented conventional airplanes. The rudder magnitude required to coordinate mid-frequency and high-frequency aileron (wheel) inputs is seen to be dependent on the roll cross-coupling, $g/U_o - N'_p$, whereas low-frequency rudder requirements are dependent on N'_r. The required rudder shaping has the characteristics of



$$\frac{1}{T_{\beta_{as}}} \doteq \frac{-\frac{g}{U_o}N_r'}{\left(\frac{g}{U_o}-N_p'\right)}$$

Figure 6 (3.6.2.1.2). Required Crossfeed for $N'_{\delta_{as}} = 0$

a rate system (ramp δ_{rp} to step δ_{as} input) 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 required rudder crossfeed.

From Figure 6 it is seen that $g/U_o - N_p'$ provides a good measure of such magnitude; and, in fact, correlation of pilot rating data (for $|N'_{\delta_{as}}/L'_{\delta_{as}}| < 0.03$) with $g/U_o - N_p'$ is quite good. However, difficulties associated with estimating an effective $g/U_o - N_p'$ for augmented airframes presents 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 6) can be important. A more general approach is to compute a time history based on a unit step aileron 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):

$$\delta'_{rp} \equiv Y'_{CF}\delta_{as} = \frac{N'_{\delta_{rp}}}{N'_{\delta_{as}}}\delta_{rp} \qquad (14)$$

Utilizing the same response time considerations as in the computation of μ, $\delta'_{rp}(3)$ is suggested as the correlating parameter when $|N'_{\delta_{as}}/L'_{\delta_{as}}|$ is small or when the denominator of Y_{CF} is of higher order than the numerator. The question of what specifically constitutes a "small" value of $N'_{\delta_{as}}/L'_{\delta_{as}}$ has proven to be somewhat difficult to quantify. Reasonably good correlations were found by plotting the $N'_{\delta_{as}}/L'_{\delta_{as}} < 0.03$ pilot ratings versus $\delta'_{rp}(3)$ as shown in Figure 7. More recent experience in utilizing the μ parameter has revealed that $N'_{\delta_{as}}/L'_{\delta_{as}} < 0.07$ results in better correlations. When $N'_{\delta_{as}}/L'_{\delta_{as}}$ is between 0.03 and 0.07, both Figure 1 and Table 1 should 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. This result is multiplied by $N'_{\delta_{rp}}/L'_{\delta_{as}}$ to give $\delta'_{rp}(3)$.

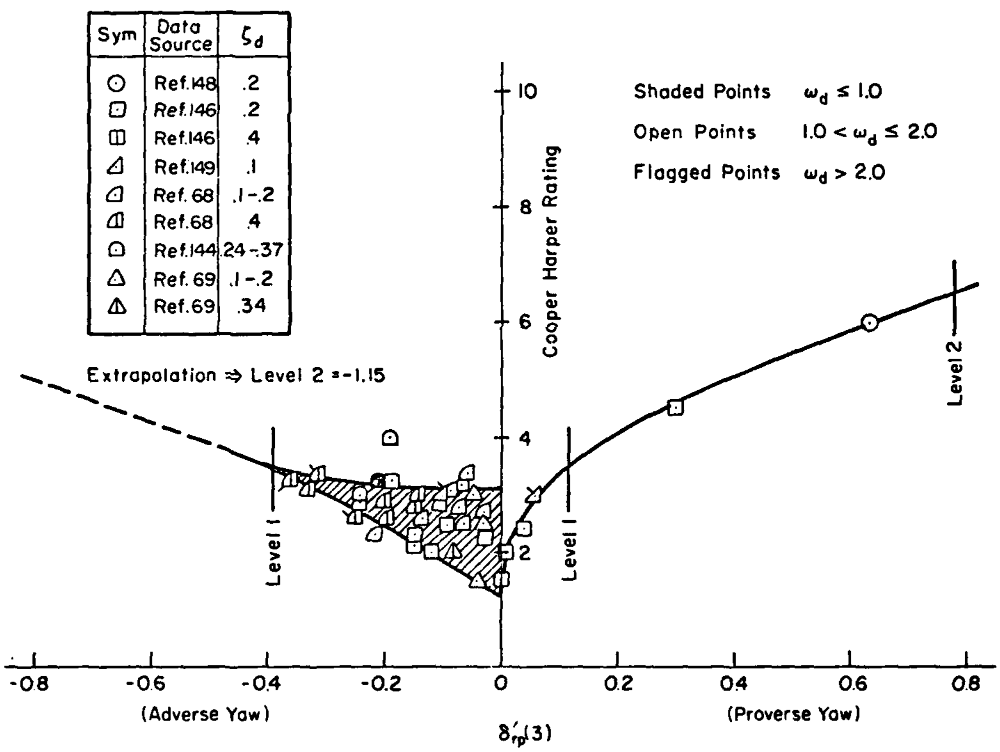


Figure 7 (3.6.2.1.2). Pilot Rating Correlations When $|N'_{\delta_{as}}/L'_{\delta_{as}}|$ Is Small

4. Complex Rudder Shaping

As discussed earlier, reasonable fits of the HOS to the LOS form implicit in Equation 10 presume that the aileron-to-rudder shaping is at least monotonic in the region of piloted control. This assumes that if the required rudder coordination to a step aileron input is non-monotonic in the region of control, pilot opinion will be poor. Since we have been unable to find any configurations that have such a non-monotonic shape, and that have been tested for pilot opinion of heading control, it is not possible to quantify the mismatch effects at this time.

E. GUIDANCE FOR APPLICATION

Clearly, the character of the requirement is such that application can only be through analysis, necessitating a good analytical model of the airplane, and especially of the β numerators of the augmented aircraft. With these, application of the criterion is straightforward.

The rules for application of this criterion are summarized as follows:

1) If $|N'_{\delta_{as}}/L'_{\delta_{as}}| < 0.03$, 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 airplanes the transfer functions 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 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 13).

4) Calculate $\delta_{rp}(3)$ from the time response of Y_{CF} (as modified by Step 2) to a unit step input, i.e., $\delta_{rp}(3) = {}^{-1}\{(1/s)Y_{CF}(s)\}$ evaluated at $t = 3$ sec.

5) Calculate μ as: $\mu = \delta_{rp}(3) - 1$ and plot on Figure 1.

6) If $|N'_{\delta_{as}}/L'_{\delta_{as}}| \leq 0.07$, calculate the normalized rudder required, $\delta'_{rp}(3)$, as follows:

- Calculate the magnitude of the time response of Y_{CF} (from Step 2) to a unit step input at t = 3 sec.
- Multiply the result by $N'_{\delta_{rp}}/L'_{\delta_{as}}$, i.e.,

$$\delta'_{rp}(3) = Y_{CF}(3)\frac{N'_{\delta_{rp}}}{L'_{\delta_{as}}}$$

Compare $\delta'_{rp}(3)$ with Table 1.

7) If $0.03 \leq |N'_{\delta_{as}}/L'_{\delta_{as}}| \leq 0.07$, utilize the most conservative result from Steps 5 and 6.

8) If the configuration does not meet the requirements, see Figure 4 to determine expected piloting problems.

F. DEMONSTRATION OF COMPLIANCE

An analytical model of the aircraft and all relevant control augmentation systems must be supplied by the contractor. Transfer functions are needed of sideslip responses to roll and yaw control inputs. The procuring activity may specify flight conditions for compliance demonstration.

G. SUPPORTING DATA

A summary of the data sources considered is given in Table 5. Each of the data points found to be applicable to heading control (i.e., met the groundrules) is plotted and faired on a logarithmic grid of $N'_{\delta_{as}}/L'_{\delta_{as}}$ versus μ in Figure 8. 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 5 correlate quite nicely with the criterion (Figure 8). If anything, the criterion is conservative in that 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

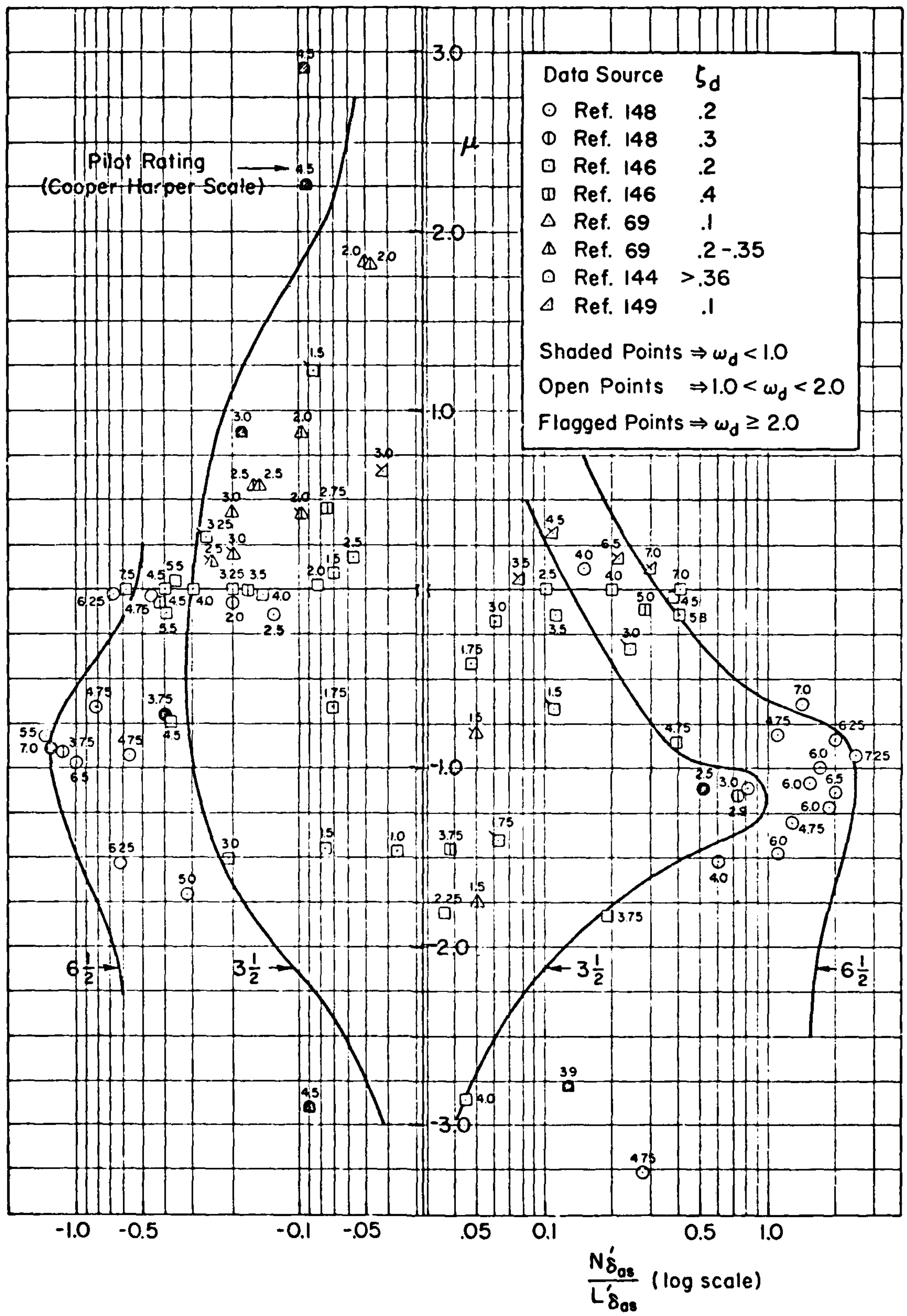


Figure 8 (3.6.2.1.2). Pilot Rating Correlation with Crossfeed Parameters

TABLE 5 (3.6.2.1.2). SUMMARY OF CURRENT DATA

TYPE OF AIRCRAFT SIMULATED	DESCRIPTION OF SIMULATOR	REFERENCE	TOTAL NUMBER OF DATA POINTS	NUMBER OF POINTS MEETING GROUNDRULES
Executive jet and military Class II	Variable stability T-33	69	84	16
STOL	Variable stability helicopter	145	109	30
General aviation (light aircraft)	Variable stability Navion	149	26	6
Jet fighter-carrier approach	Variable stability Navion	68	36	22
Space Shuttle vehicle	6 DOF moving-base with Redifon display (NASA Ames FSAA)	146	52	52
STOL	3 DOF moving-base (NASA Ames S-16)	144	8	7

configurations. This result indicates that good heading control characteristics are dependent on a fundamental aspect of piloting 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.

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 from simulation or flight test.

A review of the data sources for Figure 8 indicates that the requirement as devised is applicable primarily to low-speed flight, and especially to Category C (approach and landing) Flight Phases. 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 1 covers the worst cases.

H. LESSONS LEARNED

Figure 9 (from Reference 227) compares the rudder shaping parameter for several aircraft with the Figure 1 requirements. Available pilot rating data for the F-111 with and without adverse yaw compensator (AYC) support the boundaries. The pilot ratings are from Reference 228, 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."

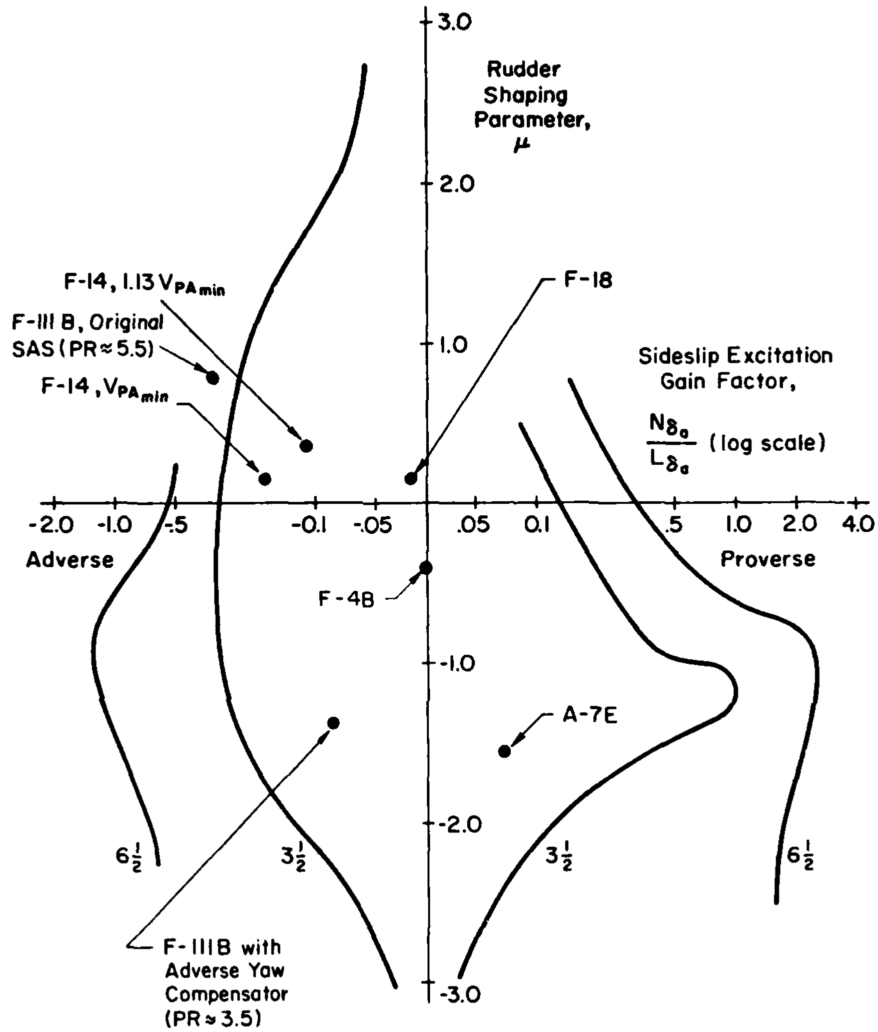


Figure 9 (3.6.2.1.2). Required Aileron-Rudder Sequencing for Several Operational Aircraft, SAS/CAS On (from Reference 227)

3.6.2.2 Pilot-induced yaw oscillations

A. REASON FOR REQUIREMENT

This is simply a statement to expressly forbid neutral or unstable closed-loop oscillations in roll. The need for such a requirement is self-evident.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.3

C. STATEMENT OF REQUIREMENT

3.6.2.2 Pilot-induced yaw oscillations. There shall be no tendency for sustained or uncontrollable yaw oscillations resulting from efforts of the pilot to control the aircraft.

D. RATIONALE BEHIND REQUIREMENT

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 lateral PIO. This requirement is identical to the roll-axis requirement of 3.5.2.

E. GUIDANCE FOR APPLICATION

This requirement should 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.

F. DEMONSTRATION OF COMPLIANCE

The existence of a 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, or in-flight refueling (receiver) may reveal PIO proneness.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

The pitch-axis PIO requirement, 3.2.2.2, contains some discussion on the cases and causes of PIOs.

3.6.2.3 Residual yaw oscillations

A. REASON FOR REQUIREMENT

This requirement is intended to prevent limit cycles in the control system or structural oscillations which might interfere with mission performance.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.1.1

C. STATEMENT OF REQUIREMENT

3.6.2.3 Residual yaw oscillations. Any sustained residual oscillations in calm air shall not interfere with the pilot's ability to perform the tasks required in service use of the aircraft.

D. RATIONALE BEHIND REQUIREMENT

The yaw axis requirements of 3.6.1 and 3.6.2.2 should prevent open-loop (aerodynamically induced) and closed-loop (pilot-induced) oscillations. This requirement sets limits on oscillations from other sources.

E. GUIDANCE FOR APPLICATION

None required.

F. DEMONSTRATION OF COMPLIANCE

Flight testing for other yaw axis requirements should reveal any problems with residual oscillations.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.3 Yaw Axis Control for Takeoff and Landing in Crosswinds

A. REASON FOR REQUIREMENT

This paragraph assures good yaw-axis flying qualities in crosswind takeoffs and landings and specifies the limiting crosswinds to be applied in various other yaw control power and force requirements.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.7

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.3 Yaw Axis Control for Takeoff and Landing in Crosswinds. It shall be possible to take off and land with normal pilot skill and technique in 90 deg crosswinds from either side of velocities up to __________.

Recommended values are given in Table 1.

TABLE 1 (3.6.3.1)

RECOMMENDED MINIMUM CROSSWIND VELOCITY REQUIREMENTS

LEVEL	CLASS	CROSSWIND
1 and 2	I	20 kt
	II, III, and IV	30 kt
	Water-based airplanes	20 kt
3	All	One half the values for Levels 1 and 2

D. RATIONALE BEHIND REQUIREMENT

This requirement was taken directly from MIL-F-8785C. An attempt to specify pilot workload in terms of the Level definitions in Para. 3.9.1 was found to be too complex and was therefore deleted from the draft report.

E. GUIDANCE FOR APPLICATION

The crosswind specified herein will affect not only this paragraph, but also the yaw control power (3.6.5.2) and force (3.6.6.2.4) requirements, all of which should be reviewed at the same time. In addition, the identical requirement for the roll axis (3.5.7) should be considered, where it will be seen that the same crosswind velocities are recommended.

F. DEMONSTRATION OF COMPLIANCE

Since flight testing in steady, 90 degree crosswinds of the specified velocity may be unacceptably hazardous for Level 2 and 3 operations, 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.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.4 <u>Yaw Axis Response to Other Inputs</u>

3.6.4.1 <u>Yaw axis response to asymmetric thrust</u>

A. REASON FOR REQUIREMENT

The transient and steady-state effects of asymmetric thrust are limited to compensatable amounts.

B. RELATED MIL-F-8785C REQUIREMENTS

3.3.9, 3.3.9.1, 3.3.9.2, 3.3.9.3, 3.3.9.4

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.4.1 <u>Yaw axis response to asymmetric thrust</u>. It shall be possible for the pilot to maintain directional control of the aircraft following a loss of thrust from the most critical propulsive source.

a) <u>Takeoff</u>: During takeoff it shall be possible to maintain a straight path without deviations of more than ___ ft. 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 airplane 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 or 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 that normally operate in the event of a thrust failure may be used in either case.

b) <u>After takeoff</u>: After takeoff it shall be possible without a change in selected configuration to achieve straight flight following sudden asymmetric loss of thrust from the most critical factor at speeds from V_{min} (TO) to V_{max} (TO), and thereafter to maintain straight flight throughout the climbout. Automatic devices that normally operate in the event of a thrust failure may be used, and the airplane may be banked up to 5 degrees away from the inoperative engine.

c) Takeoff and landing in crosswinds: The aircraft shall be safely controllable in the crosswinds of 3.6.3 from the unfavorable direction.

d) In-flight: The airplane motions following sudden asymmetric loss of thrust shall be such that dangerous conditions can be avoided by pilot corrective action. A realistic time delay of at least ____ second shall be incorporated. In addition, the static directional stability shall be such that at all speeds above ____, with asymmetric loss of thrust from the most critical factor while the other engine(s) develop normal rated thrust, the airplane with yaw control pedals free may be balanced directionally in steady straight flight. The trim settings shall be those required for wings-level straight flight prior to the failure.

Recommended values:

Maximum path deviation during takeoff:	30 ft
Minimum time delay:	1 second
Minimum speed, yaw controls free:	1.4 V_{min}

D. RATIONALE BEHIND REQUIREMENT

This requirement consists of portions of several paragraphs from MIL-F-8785C. It assures directional control by the pilot under adverse conditions (i.e., crosswinds), and insures a match between upsetting yawing moments due to asymmetric thrust and restoring moments from static directional stability. 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. Similarly, the requirement insures that following thrust loss after takeoff the pilot can safely continue climb-out. The intent is that V_{min} (TO) normally should be set by other considerations and adequate control provided down to that speed.

E. GUIDANCE FOR APPLICATION

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 airplane caused by thrown turbine blades: for example, hydraulic lines should be routed (or enough armor used) so that thrown engine, fan, or propeller parts cannot sever all hydraulic systems needed for flight control.

F. DEMONSTRATION OF COMPLIANCE

Since each statement of this requirement specifies the applicable regime (except the general requirement for pilot control and static stability, which extends over the operational altitude range), testing must cover the applicable conditions. For operations after takeoff, the testing may be performed at some safe (but still low) altitude.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.4.2 Yaw axis response to failures

A. REASON FOR REQUIREMENT

This requirement limits the severity of failures on controllability in the yaw axis.

B. RELATED MIL-F-8785C REQUIREMENTS

3.4.8, 3.4.9, 3.5.5.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.4.2. Yaw axis response to failures. The yawing motions following sudden airplane system or component failures shall be such that dangerous conditions can be avoided by pilot corrective action. A realistic time delay between the failure and initiation of pilot corrective action shall be incorporated when determining compliance. No single failure of any component or system shall result in dangerous or intolerable flying qualities; Special Failure States (1.6.3) are excepted. The crew member concerned shall be provided with immediate and easily interpreted indications whenever failures occur that require or limit any flight crew action or decision. With controls free, the yawing motions due to failures shall not exceed ________________.

Recommended values: It is recommended that the yaw excursions not exceed the following limits for at least 2 seconds following the failure:

Levels 1 and 2 (after failure)	± 0.5 g incremental lateral acceleration at the pilot's station, except that structural limits shall not be exceeded. In addition, for Category A, lateral excursions of 5 ft.
Level 3 (after failure)	No dangerous attitude or structural limit is reached, and no dangerous alteration of the flight path results from which recovery is impossible.

D. RATIONALE BEHIND REQUIREMENT

This paragraph is primarily qualitative in nature because of the number of unknowns that might affect a failed condition. The severity of 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 by requiring Level 3 or better flying qualities following any single failure).

E. GUIDANCE FOR APPLICATION

Clearly, application of this requirement is contingent on the specification of those failures considered to be most critical.

The time delay should 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 that represents the time required for the pilot to diagnose the situation and initiate corrective action.

F. DEMONSTRATION OF COMPLIANCE

Flight testing the failures specified in 1.6.3 will involve those conditions most critical for such failures. Therefore, it is expected that simulation may be required to demonstrate compliance. Adequate motion cues should be available to simulate the acceleration environment with one-to-one fidelity for at least 2 seconds following the failure.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.4.3 Yaw axis response to configuration or control mode change

A. REASON FOR REQUIREMENT

Pitch transients due to intentional mode switching must not be excessive.

B. RELATED MIL-F-8785C REQUIREMENTS

3.5.6, 3.5.6.1.

C. STATEMENT OF REQUIREMENTS AND RECOMMENDED VALUES

3.6.4.3 Yaw axis response to configuration or control mode change. The transient motions and trim changes resulting from the intentional engagement or disengagement of any portion of the primary flight control system by the pilot shall be such that dangerous flying qualities never result. With controls free, the transients resulting from these situations shall not exceed the following limits for at least _____ seconds following the transfer: ______________________. These requirements apply only for Aircraft Normal States, within the Service Flight Envelope.

Recommended Transient motions (within first 2 sec following transfer): The lesser of ±5 degrees sideslip and the structural limit.

D. RATIONALE BEHIND REQUIREMENT

Since the intent of a flight control system is to improve the aircraft response characteristics — whether measured by improved flying qualities or by increased mission effectiveness — any system which can be chosen by the pilot should not cause noticeable transient motions.

E. GUIDANCE FOR APPLICATION

No specific guidance is offered except that tests should be conducted at the most critical flight conditions.

F. DEMONSTRATION OF COMPLIANCE

Flight testing at the corners of the expected operational envelopes for any control systems must be performed (e.g., a SAS for power approach must be switched at the highest and lowest expected airspeeds, at low altitudes). Limited analytical and ground-based simulation may be used to supplement actual flight testing, especially in the early stages of development. But flight testing is ultimately required.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.4.4 Yaw axis response to stores release

A. REASON FOR REQUIREMENT

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

B. RELATED MIL-F-8785C REQUIREMENT

3.4.6

C. STATEMENT OF REQUIREMENT

3.6.4.4 Yaw axis response to stores release. The intentional release of any stores shall not result in objectionable flight characteristics for Levels 1 and 2. However, the intentional release of stores shall never result in dangerous or intolerable flight characteristics. This requirement applies for all flight conditions and store loadings at which normal or emergency store release is structurally permissible.

D. RATIONALE BEHIND THE REQUIREMENT

This paragraph is unchanged from MIL-F-8785C. It is a necessary catch-all requirement. Because of the variety of possibilities, it must be left qualitative. Similar pitch- and roll-axis requirements have been specified (3.2.7.4, 3.5.8.4).

E. GUIDANCE FOR APPLICATION

Evaluation of this criterion should occur as a natural part of operational flight testing. It is not subtle and requires no special analysis or interpretation.

F. DEMONSTRATION OF COMPLIANCE

Operational flight test will be necessary for final demonstration.

G. **SUPPORTING DATA**

None available.

H. **LESSONS LEARNED**

None available.

3.6.4.5 Yaw axis response to armament delivery

A. REASON FOR REQUIREMENT

This requirement is included to insure that armament delivery will not have an adverse effect on flying qualities that could impair mission effectiveness.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.7

C. STATEMENT OF REQUIREMENT

3.6.4.5 Yaw axis response to armament delivery. Operation of movable parts such as bomb bay doors, cargo doors, armament pods, refueling devices, and rescue equipment, or firing of weapons, release of bombs, or delivery or pickup of cargo shall not cause buffet, trim changes, or other characteristics which impair the tactical effectiveness of the aircraft under any pertinent flight conditions. These requirements shall be met for Levels 1 and 2.

D. RATIONALE BEHIND THE REQUIREMENT

This paragraph has remained unchanged in MIL-F-8785C and in the MIL Standard.

E. GUIDANCE FOR APPLICATION

This requirement is similar to 3.6.4.4.

F. DEMONSTRATION OF COMPLIANCE

Operational flight test should be required.

G. SUPPORTING DATA

None available.

H. LESSONS LEARNED

None available.

3.6.5 Yaw Axis Control Power

A. REASON FOR THIS REQUIREMENT

This qualitative requirement is based on the fundamental necessity to establish equilibrium in the yaw axis in the presence of disturbances. Specific requirements are given in Paragraphs 3.6.5.1 through 3.6.5.3.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.5

C. STATEMENT OF REQUIREMENT

3.6.5 Yaw Axis Control Power. Directional stability and control characteristics shall enable the pilot to balance yawing moments and control yaw and sideslip.

D. RATIONALE BEHIND REQUIREMENT

The yaw controller must always be sufficiently powerful to overcome any anticipated yawing moment. This requirement allows for directional stability (i.e., C_{n_β}) to augment the control power (i.e., $C_{n_{\delta_{rp}}}$).

E. GUIDANCE FOR APPLICATION

The requirement should be applied during all phases of development.

F. DEMONSTRATION OF COMPLIANCE

Flight testing will be required, emphasizing steady sideslips at all conditions.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.5.1 Yaw axis control power for takeoff, landing, and taxi

A. REASON FOR REQUIREMENT

This paragraph defines yaw control power for operations on or near the ground, especially in crosswinds.

B. RELATED MIL-F-8785C REQUIREMENTS

3.3.7.1, 3.3.7.2, 3.3.7.2.1, 3.3.7.2.2, 3.3.7.3

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.5.1 Yaw axis control power for takeoff, landing, and taxi.

a) It shall be possible to taxi on a dry surface at any angle to a ___ kt wind.

b) In taxi on wet, snow-packed, or icy runways, directional control shall be maintained by use of aerodynamic controls alone at all airspeeds above ____ kt. For very slippery runways, the requirement need not apply for crosswind components at which the force tending to blow the airplane off the runway exceeds the opposing tire-runway frictional force with the tires supporting all of the airplane's weight.

c) In the takeoff run, landing rollout, and taxi, yaw control power shall be adequate to maintain a straight path on the ground or other landing surface. This applies to calm air and in crosswinds up to the values specified in 3.6.3, on wet runways for all aircraft, and on snow-packed and icy runways for aircraft intended to operate under such conditions.

d) Yaw axis control power shall be adequate to develop ___ deg of sideslip in the power approach.

e) All carrier-based airplanes shall be capable of maintaining a straight path on the ground without the use of wheel brakes, at airspeeds of 30 knots and above, during takeoffs and landings in a 90-degree crosswind of at least 0.1 $V_{S(L)}$.

Recommended values:

Wind speeds for taxi:	
Class I airplanes:	35 kt
Class II, III, and IV airplanes:	45 kt
Minimum sideslip in power approach:	10 deg
Minimum controllable taxi speeds:	
Class IV	50 kt
Others	30 kt

D. RATIONALE BEHIND REQUIREMENT

Since the major definitions of yaw control power in operations near the ground are concerned with sideslip generation or compensation for crosswinds, this paragraph incorporates the relevant MIL-F-8785C requirements into one paragraph.

E. GUIDANCE FOR APPLICATION

Application of this requirement as stated is straightforward; however, some caution should be exercised in applying the crosswinds specified in 3.6.3, since actual crosswinds normally include unsteady gusts, and designing to just meet this requirement might not leave a margin of safety in real crosswind operations.

F. DEMONSTRATION OF COMPLIANCE

As with all requirements on control power, it is desirable to show compliance through actual testing. However, as with all crosswind requirements, this can be difficult. Operations in some level of crosswinds that falls within the maximum specified in 3.6.3 should provide some indication of trends. Careful analysis to extrapolate these results to the limits of this paragraph may be accepted in lieu of further testing.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.5.2 Yaw axis control power for two engines inoperative

A. REASON FOR REQUIREMENT

This paragraph states the need for yaw control power in event of failure of more than one engine on multi-engine airplanes.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.9.5

C. STATEMENT OF REQUIREMENT

3.6.5.2 Yaw axis control power for two engines inoperative. At the one-engine-out speed for maximum range with any engine initially failed, upon failure of the most critical remaining engine the yaw control power shall be adequate to stop the transient motion and thereafter to maintain straight flight from that speed to the speed for maximum range with both engines failed. In addition, it shall be possible to effect a safe recovery at any service speed above $V_{o_{min}}$ (CL) following sudden simultaneous failure of the two critical engines.

D. RATIONALE BEHIND REQUIREMENT

The rationale for this requirement is self evident; it is taken essentially intact from MIL-F-8785C.

E. GUIDANCE FOR APPLICATION

The specialized nature of the requirement, as well as the general, qualitative terms in which it is worded, makes it straightforward to apply.

F. DEMONSTRATION OF REQUIREMENT

Flight testing at altitudes covering the operational envelope will be at conditions stated by the requirements, i.e., airspeeds will be either speed for maximum range with one engine out or representative speeds above $V_{o_{min}}$ (CL).

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.5.3 Yaw axis control power with asymmetric loading

A. REASON FOR REQUIREMENT

This requirement assures adequate yaw control power to compensate for any specified condition of asymmetric loading.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.5.1.1

C. STATEMENT OF REQUIREMENT

3.6.5.3 Yaw axis control power with asymmetric loading. When initially trimmed directionally with each asymmetric loading specified in Paragraph 3.1.1 at any speed in the Operational Flight Envelope, yaw control power shall be sufficient to maintain a straight flight path.

D. RATIONALE BEHIND REQUIREMENT

It is common sense that specified (i.e., anticipated) conditions of asymmetric loading should not cause control power problems.

E. GUIDANCE FOR APPLICATION

This requirement is intended to be applied in conjunction with 3.6.6.2.5, yaw axis control force limits with asymmetric loading.

F. DEMONSTRATION OF COMPLIANCE

See 3.6.6.2.5.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.5.4 Yaw axis control power for stores release

A. REASON FOR REQUIREMENT

This requirement is included to insure that intentional release of stores does not result in limitations in roll control power.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.6, 3.3.4.1.2

C. STATEMENT OF REQUIREMENT

3.6.5.4 Yaw axis control power for stores release. Yaw control power shall be adequate to regain straight flight, without retrimming, following intentional release of any stores to the maximum load factors specified in 3.2.8.2 with adequate control margin.

D. RATIONALE BEHIND REQUIREMENT

This is effectively a new requirement, though it is based on the general requirement of 8785C that stores release "shall not result in objectionable flight characteristics." This has been translated as requiring sufficient yaw power to regain straight flight with any stores loading.

E. GUIDANCE FOR APPLICATION

This requirement is closely related to the roll and yaw axis responses to stores release, 3.5.8.4 and 3.6.4.4, and to the similar roll axis requirement, 3.5.9.6. All these paragraphs should be considered in combination for application.

F. DEMONSTRATION OF COMPLIANCE

Flight testing of the stores configurations specified in 3.1.3, over the applicable range of flight conditions, must be conducted.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.5.5 Yaw axis control power for other conditions

A. REASON FOR REQUIREMENT

This catch-all specification is intended to assure adequate yaw control power in any situation not already covered in the Standard.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.10

C. STATEMENT OF REQUIREMENT

3.6.5.5 Yaw axis control power for other conditions. Control authority, rate and hinge moment capability shall be sufficient to assure safety throughout the combined range of all attainable angles of attack (both positive and negative) and sideslip. This requirement applies to the prevention of loss of control and to recovery from any situation for all maneuvering, including pertinent effects of factors such as regions of control-surface-fixed instability, inertial coupling, fuel slosh, the influence of symmetric and asymmetric stores, stall/post-stall/spin characteristics, atmospheric disturbances and Aircraft Failure States (maneuvering flight appropriate to the *Failure State* is to be included). Consideration shall be taken of the degrees 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.

D. RATIONALE BEHIND REQUIREMENT

The other paragraphs falling under 3.6.5 cover all normal, anticipated situations for yaw control power. This paragraph is added to cover any unusual or unspecified conditions that might be encountered in flight.

E. GUIDANCE FOR APPLICATION

Due to its broad generality the requirement should be applied for all phases of analysis, simulation, and flight test. Excessive stability, as well as excessive instability, of the basic airframe is of concern with respect to available control authority and rate.

F. DEMONSTRATION OF COMPLIANCE

See the similar pitch-axis requirement, 3.2.8.5.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.6 Yaw Axis Control Forces

A. REASON FOR REQUIREMENT

This requirement is included as a direct carryover from MIL-F-8785C, where it served as a qualitative criterion on rudder sensitivity. It also was offered as a "catchall" requirement to insure that balancing moments can be achieved in the yaw axis.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.5

C. STATEMENT OF REQUIREMENT

3.6.6 Yaw Axis Control Forces. Sensitivity to yaw control pedal forces shall be sufficiently high that directional control and force requirements can be met and 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.

D. RATIONALE BEHIND REQUIREMENT

A quantitative requirement for yaw control force/position gradients is not included in the Standard, since there are little data for defining such a requirement. Instead, this requirement qualitatively limits the range of acceptable force gradients. Subparagraphs in this section place limitations on maximum allowable gradients and forces.

E. GUIDANCE FOR APPLICATION

From the basic nature of this requirement it is clear that it should be easily met by proper control design in the early stages of development.

F. DEMONSTRATION OF COMPLIANCE

In general, flight testing for compliance with any of the yaw axis requirements should reveal any problems with meeting this requirement. Special attention should be given when testing for compliance with the maximum force requirements that follow, since these can be considered subsets of this general statement.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.6.1 Yaw axis control force linearity

A. REASON FOR REQUIREMENT

For reasonable sideslip angles, the forces required from the pilot should be of a normal sense, and linear with sideslip.

B. RELATED MIL-F-8785C REQUIREMENTS

3.3.6, 3.3.6.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.6.1 Yaw axis control force linearity. The following requirements are expressed in terms of characteristics in yaw-control-induced steady, zero-yaw-rate sideslips with the airplane 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 pounds of yaw-control-pedal force, or

c) Maximum roll control or surface deflection,

except that for single-propeller-driven airplanes 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 10-degree change in sideslip from the wings-level straight flight condition.

Right yaw-control-pedal force shall produce left sideslips and left yaw-control-pedal force shall produce right sideslips. For Levels 1 and 2 the following requirements shall apply. The variation of sideslip angle with yaw-control-pedal force shall be essentially linear for sideslip angles between ____ degrees and ____ degrees. Although a lightening of pedal force is acceptable for sideslip angles outside this range, the pedal force shall never reduce to zero.

Recommended sideslip angle range is +10 deg to -10 deg.

D. RATIONALE BEHIND REQUIREMENT

Paragraph 3.6.1.1.2, the steady-state requirement for yaw response to yaw controller, specifies yaw pedal deflection characteristics. This requirement augments that paragraph by similarly defining the force characteristics.

E. GUIDANCE FOR APPLICATION

The applicable sideslips and sideslip range for this requirement adequately cover conditions expected in operational flight. The requirement is straightforward in wording, and should be so in application as well.

F. DEMONSTRATION OF COMPLIANCE

Flight testing at trimmed steady sideslips should be performed with the airplane configured for most aft center of gravity. Test conditions should cover the service speed range at the minimum, intermediate, and maximum operational altitudes.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.6.2 Yaw axis control force limits

3.6.6.2.1 Yaw axis control force limits in rolling maneuvers

A. REASON FOR REQUIREMENT

This paragraph is aimed at insuring that full coordination can be achieved during rapid turn entries and exits as well as during steady rolls with reasonable rudder-pedal forces.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.2.5

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.6.2.1 Yaw axis control force limits in rolling maneuvers. In the maneuvers described in 3.5.9, directional-control effectiveness shall be adequate to maintain zero sideslip with pedal force not greater than ___ lb.

Recommended values:

Class	Flight Phase Category	Level	Maximum Pedal Force, lb
IV	A	1	50
	All	2-3	100
All others	All	All	100

D. RATIONALE BEHIND REQUIREMENT

Yaw control forces required for the rolling maneuvers of 3.5.9 should be at reasonably low levels.

E. GUIDANCE FOR APPLICATION

If the yaw control forces required to coordinate the rolls specified in 3.5.9 seem excessive to the pilot, this requirement can be utilized to define an upper limit.

F. DEMONSTRATION OF COMPLIANCE

Yaw pedal forces should be recorded during testing for compliance with 3.5.9. No specific maneuvers are required for this paragraph.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.6.2.2 Yaw axis control force limits in steady turns

A. REASON FOR REQUIREMENT

The objective of this requirement is to insure that only modest yaw pedal forces are required when performing coordinated turns.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.2.6

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.6.2.2 Yaw axis control force limits in steady turns. It shall be possible to maintain steady coordinated turns in either direction, using ___ deg of bank with a pedal force not exceeding ___ lb, with the airplane trimmed for wings-level straight flight. These requirements constitute Levels 1 and 2.

Recommended values:

Airplane Class	Bank Angle (deg)	Maximum Pedal Force (lb)
I, II	45	40
III	30	40
IV	60	40

D. RATIONALE BEHIND REQUIREMENT

The maximum allowable pedal forces should be relatively small for banked turns, where the application of such forces will be sustained for some time. The forces specified herein are upper limits and good design practice would dictate considerably lower forces.

E. GUIDANCE FOR APPLICATION

This requirement applies to Levels 1 and 2 only, since it is expected that Level 3 operations would not involve steady, large-bank-angle turns.

F. DEMONSTRATION OF COMPLIANCE

Flight testing at the specified bank angle, at the minimum operational velocity, presents the most potential for large yawing moments.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.6.2.3 Yaw axis control force limits during speed changes

A. REASON FOR REQUIREMENT

This paragraph is included to insure that speed effects on yawing moment do not require unreasonably large yaw pedal forces to maintain trim.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.5.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.6.2.3 Yaw axis control force limits during 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 percent of the trim speed or ±100 kt equivalent airspeed, whichever is less (except where limited by boundaries of the Service Flight Envelope) with yaw-control-pedal forces not greater than ___ lb without retrimming.

Recommended values:

Propulsion Type	Level	Maximum Pedal Force (lb)
Propeller	1 and 2	100
	3	180
All others	1 and 2	40
	3	180

D. RATIONALE BEHIND REQUIREMENT

The maximum allowable forces are divided by type of propulsion. The large forces associated with sidewash and asymmetric blade loading result in a relaxation of the requirement for propellers.

E. GUIDANCE FOR APPLICATION

None required.

F. DEMONSTRATION OF COMPLIANCE

Acceleration/deceleration runs must be performed to cover the speed range specified by this requirement. Testing should be accomplished in all operational configurations.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.6.2.4 Yaw axis control force limits in crosswinds

A. REASON FOR REQUIREMENT

This requirement is included to provide limits on the yaw pedal forces required for crosswind takeoffs and landings.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.7

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.6.2.4 Yaw axis control force limits in crosswinds. It shall be possible to take off and land in the crosswinds specified in 3.6.3 without exceeding the following yaw control forces: ________________.

Recommended values:

Level	Maximum Pedal Force (lb)
1	100
2 and 3	180

D. RATIONALE BEHIND REQUIREMENT

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.

E. GUIDANCE FOR APPLICATION

Paragraph 3.6.3, "Yaw axis control for takeoff and landing in crosswinds," specifies the crosswind component to be applied for this requirement. Obviously the emphasis in this requirement is to limit the effort required of the pilot. The recommended limits are taken directly from MIL-F-8785C. However, they are felt to be quite high for continuous maneuvering, and in fact may be excessive for female pilots. It is recommended that the procuring activity consider lower limits on new aircraft. Reference 257 shows that of 61 men and 61 women tested in a short-duration (4 second) force test, almost all could exert 180 lb to rudder pedals. However, continuous maneuvering would likely be difficult for male or female pilots.

F. DEMONSTRATION OF COMPLIANCE

Compliance should be demonstrated by performing takeoffs and landings in the actual crosswinds stated in 3.6.3. The use of simulation is not recommended because of the difficulty in developing an accurate model of the landing gear dynamics.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.6.2.5 Yaw axis control force limits with asymmetric loading

A. REASON FOR REQUIREMENT

This requirement is included to insure that conditions of asymmetric loading do not result in excessive yaw pedal force demands on the pilot.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.5.1.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.6.2.5 Yaw axis control force limits with asymmetric ading. When initially trimmed directionally with each asymmetric loadir eci- fied in Paragraph 3.1.1 at any speed in the Operational Flight lope, it shall be possible to maintain a straight flight path throu the Operational Flight Envelope with yaw-control-pedal forces not cer than ___ lb without retrimming.

Recommended values:

Level	Maximum Pedal Force (lb)
1 and 2	100
3	180

D. RATIONALE BEHIND REQUIREMENT

Asymmetric external loading, whether a result of normal or unusual delivery or dropping of loads, should not place unrealistically large demands on the pilot.

E. GUIDANCE FOR APPLICATION

If conditions of asymmetric loading are expected to be regularly encountered, the procuring activity may want to reduce the yaw control force limits. The recommended limits have been taken from MIL-F-8785C and 180 lb seems unreasonably high (see Para. 3.6.6.2.4).

F. DEMONSTRATION OF COMPLIANCE

Flight testing with the aircraft configured as specified in 3.1.1 should be conducted throughout the Operational Envelope.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.6.2.6 Yaw axis control force limits in dives and pullouts

A. REASON FOR REQUIREMENT

This requirement is included to place limits on the yaw pedal force required to perform dives that will be required by the specified mission.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.8

C. STATEMENT OF REQUIREMENTS AND RECOMMENDED VALUES

3.6.6.2.6 Yaw axis control force limits in dives and pullouts. Throughout the dives and pullouts of 3.2.9.7.3, yaw-control-pedal forces shall not exceed ____ lb in dives and pullouts to the maximum speeds specified in the Service Flight Envelope.

Recommended values:

Propulsion Type	Maximum Pedal Force (lb)
Propeller	180
Other	50

D. RATIONALE BEHIND REQUIREMENT

As for Paragraph 3.6.6.2.3, "Yaw axis control force limits during speed changes," the maximum pedal forces are much larger for propeller-driven aircraft.

E. GUIDANCE FOR APPLICATION

It is evident that this requirement is most valuable only if dives are specified by the procuring activity, or expected in normal service. Otherwise, the more stringent requirements of Paragraph 3.6.6.2.3 should easily assure that this paragraph is met. As with earlier requirements, the recommended limits seem unreasonably high for female pilots (see Para. 3.6.6.2.4).

F. DEMONSTRATION OF COMPLIANCE

If dives are specified as normal operations, flight testing must cover the range of such dives. Otherwise, some amount of dive and pullout testing should be performed.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.6.2.7 Yaw axis control force limits for go-around

A. REASON FOR REQUIREMENT

The possibility of large, transient yaw pedal force requirements on initiation of go-arounds necessitates a limit.

B. RELATED MIL-F-8785C REQUIREMENT

3.3.5.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.6.2.7 Yaw axis control force limits for go-around. The response to thrust, configuration and airspeed change shall be such that the pilot can maintain straight flight during go-around initiated at speeds down to V_S (PA) with yaw-control-pedal forces not exceeding ____ lb when trimmed at $V_{o_{min}}$ (PA). The preceding requirements apply for Levels 1 and 2. The Level 3 requirement is to maintain straight flight in these conditions with yaw-control-pedal forces not exceeding ____ lb. Bank angles up to 5 deg are permitted for all Levels.

Recommended values:

Levels 1 and 2	Maximum Pedal Force (lb)
For propeller-driven Class IV, and all propeller-driven carrier-based aircraft	100
All others	40
Level 3	
All	180

D. RATIONALE BEHIND REQUIREMENT

Pedal forces tend to be very high during a 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.

E. GUIDANCE FOR APPLICATION

This very important requirement should be applied for all airplanes, at various weights and loadings, to assure that go-arounds will not overtax the pilot. The limits recommended for propeller-driven aircraft may be excessive for extended operations (see Para. 3.6.6.2.4 and Reference 257) and are retained here only because they appeared in MIL-F-8785C. They should be lowered if appropriate data can be made available.

F. DEMONSTRATION OF COMPLIANCE

Go-arounds may be simulated at a safe (but low) altitude, covering the speed range specified.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.6.2.8 Yaw axis control force limits for asymmetric thrust during takeoff

A. REASON FOR REQUIREMENT

This requirement is included to insure that the loss of one or more engines on a multi-engine aircraft on takeoff does not place unreasonably large force demands on the pilot.

B. RELATED MIL-F-8785C REQUIREMENTS

3.3.9.1, 3.3.9.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.6.2.8 Yaw axis control force limits for asymmetric thrust during takeoff.

a) During the takeoff ground run it shall be possible to achieve and maintain a straight path on the takeoff surface without a deviation of more than ____ ft from the path originally intended, with yaw-control forces not exceeding ____ lb.

b) For the continued takeoff it shall be possible, without a change in selected configuration, to achieve straight flight following sudden asymmetric loss of thrust from the most critical propulsive source at speeds from V_{min} (TO) to V_{max} (TO), and thereafter to maintain straight flight throughout the climbout without exceeding a maximum yaw control pedal force of ____ lb.

c) For the aborted takeoff the requirements above 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 that normally operate in the event of a thrust failure may be used in either case.

Recommended values:

Maximum path deviation: 30 ft

Maximum yaw pedal forces: 180 lb

D. RATIONALE BEHIND REQUIREMENT

This requirement is a compilation of two paragraphs of MIL-F-8785C, to cover ground run and continued or aborted takeoffs. The objective of the requirement is to insure that, following loss of thrust during the takeoff run, the pilot can either safely abort or safely continue the takeoff and climbout without loss of directional control.

E. GUIDANCE FOR APPLICATION

As with all the yaw control force limits, this one seems excessive (see Para. 3.6.6.2.4). This requirement seems particularly important because of the obvious safety implications. While no effort has been made to produce hard data, FAR Part 23 limits prolonged yaw control forces to 120 lb. This seems like a useful interim value until better data can be obtained.

F. DEMONSTRATION OF COMPLIANCE

With the airplane configured at its lightest weight, simulated engine-out takeoffs must be performed in the conditions specified by the requirement. Simulation is not recommended for the ground roll portion of this requirement because of the known problems with developing an accurate landing gear model.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.6.2.9 Yaw axis control force limits with failures

A. REASON FOR REQUIREMENT

This requirement is included to place limits on the maximum force to counter yaw trim change after a failure of any portion of the primary flight control system.

B. RELATED MIL-F-8785C REQUIREMENT

3.5.5.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.6.6.2.9 Yaw axis control force limits with failures. The change in yaw control force required to maintain constant heading following a failure shall not exceed ____ lb for at least 5 seconds following the failure.

Recommended value:

Maximum yaw control force: 50 lb

D. RATIONALE BEHIND REQUIREMENT

It is necessary to have limits on the control forces required to hold heading following flight control system failures. It seems reasonable to state a time limit during which this requirement applies. Two seconds generally should be time enough for the pilot to detect a significant transient and react, and it should be possible to retrim after 5 seconds.

E. GUIDANCE FOR APPLICATION

The requirement will apply for all failures specified in 3.1.6.

F. DEMONSTRATION OF COMPLIANCE

Flight testing must encompass the most critical areas for specified failures unless safety considerations prohibit such testing, in which case simulation should be sufficient to demonstrate compliance (in flight).

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.6.6.2.10 Yaw axis control force limits — configuration or control mode change

A. REASON FOR REQUIREMENT

This requirement is included to provide limits on the transient yaw pedal forces necessary following a transfer to alternate control modes.

B. RELATED MIL-F-8785C REQUIREMENT

3.5.6.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUE

3.6.6.2.10 <u>Yaw axis control force limits — configuration or control mode change</u>. The change in yaw control force required to maintain zero sideslip following intentional engagement or disengagement of any portion of the primary flight control system by the pilot shall not exceed the following limits: ________________. These requirements apply only for Aircraft Normal States.

It is recommended that for at least 5 seconds following the mode change the change in yaw control force not exceed 10 lb above basic controller breakout force.

D. RATIONALE BEHIND REQUIREMENT

The transfer to an alternate control mode should occur with a negligible transient. The recommended value of 10 lb above breakout is a result of a reduction from the 50 lb used in Paragraph 3.5.6.2 of MIL-F-8785C, which seems unreasonably large. While there are no hard data upon which to base this number, it is given as a recommended value only to show that a significant transient when transferring control modes is not acceptable.

E. GUIDANCE FOR APPLICATION

This requirement has been of minimal importance in the past; however, with development of direct force control augmentation (for example, Paragraph 3.6.1.2), the impact of this requirement will increase.

F. DEMONSTRATION OF COMPLIANCE

Demonstration will involve mode switching at representative altitudes throughout the Operational Flight Envelope, focusing on those areas of airspeed, altitude, and task that would produce the largest transients between the two modes involved.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

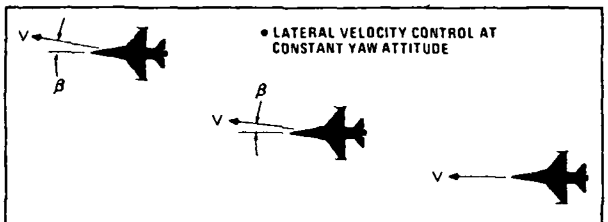
None.

3.7 HANDLING QUALITY REQUIREMENTS FOR LATERAL FLIGHT PATH AXIS

3.7.1 Bandwidth Requirement for Lateral Translation

A. REASON FOR REQUIREMENT

This requirement is included to specify the response characteristics of aircraft that utilize direct force control (DFC) in the lateral translation mode. This mode allows changes in lateral position to occur at zero bank angle and without any change in heading. It is sometimes referred to as the "β_2 mode." A sketch of the airplane motions, with a summary of useful features of this mode, is given below (taken from Reference 229).



• IDEAL FOR SMALL LATERAL POSITION CORRECTIONS	• USEFUL FOR AIR-TO-GROUND TRACKING CORRECTIONS	• NATURAL CONTROL FOR CROSSWIND DRIFT AND GROUND TARGET MOTION

B. RELATED MIL-F-8785C REQUIREMENT

3.4.11

C. STATEMENT OF REQUIREMENT

3.7.1 Bandwidth Requirement for Lateral Translation.

a) Dynamic response to direct force control input. The bandwidth of the open-loop response of lateral position to lateral translation control input shall be greater than _______ for Flight Phase ____. Lateral translations shall occur at essentially zero bank angle and zero change in heading.

b) Steady-state response to lateral translation control input. Maximum force control input shall produce at least ___ degrees of sideslip.

c) Lateral translation control forces and deflections. Use of the primary lateral translation control shall not require use of another control manipulator to meet Requirement a). The controller characteristics shall meet the following requirements: ______________.

d) Pilot accelerations. Abrupt, large control inputs shall not produce pilot head or arm motions which interfere with task performance. Pilot restraints shall not obstruct the crew's normal field of view nor interfere with manipulation of any cockpit control required for task performance.

Recommended values (Part a): There are no data of sufficient quality upon which to set a lower limit on lateral position bandwidth. Until the required data become available, the following qualitative requirement should be applied: Lateral translation response to control inputs shall be acceptable to the crew in performing the mission tasks.

Recommended values (Part b): It is recommended that it should be possible to generate at least 4 deg of sideslip at the flight conditions where lateral translation is to be utilized.

Recommended values (Part c): If conventional cockpit controls are to be used as the DFC controller, the requirements for these controls provides some guidance, i.e., Paragraph 3.6.6 for rudder pedals and 3.5.10 for stick.

D. RATIONALE BEHIND REQUIREMENT

The rationale for using a bandwidth criterion for DFC modes is given in the discussion in Paragraph 3.6.1.2.1.

E. GUIDANCE FOR APPLICATION

Procedures to be followed in applying the bandwidth criterion are given in "Guidance for Application" and "Demonstration of Compliance" for Paragraph 3.6.1.2.1.

F. DEMONSTRATION OF COMPLIANCE

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 compliance with this requirement should be based on flight test. In fact, the data generated during such tests should be utilized to upgrade the requirements.

G. SUPPORTING DATA

The only supporting data available are the F-16 CCV (Reference 229) and two configurations flight tested in Reference 115. The Reference 115 data are shown below in Table 1. 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 1 (3.7.1)

SUMMARY OF COOPER-HARPER PILOT RATINGS FOR LATERAL TRANSLATION MODE (REFERENCE 115)

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 1 is probably a measure of the degree to which each pilot objected to 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 Reference 115 (page 13) the lateral velocity response for a perfectly decoupled aircraft is

$$\frac{\dot{y}}{\delta_{DFC}} = \frac{Y_{\delta_{SF}}}{s - Y_V}$$

It follows that the basic response of the lateral translation (β_2) 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). 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 upon release of the CCV control. An example of this is quoted below from Reference 229 (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 β_2 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 more successful lateral translation mode could be developed by augmenting Y_v via feedback of sideslip to the direct side force controller. This of course has implications on the frequency response characteristics of the servo drive as well as the authority required for the direct side force control.

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 1).

H. LESSONS LEARNED

The above noted 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.

3.8 HANDLING QUALITY REQUIREMENTS FOR COMBINED AXES

3.8.1 Cross-Axis Coupling in Roll Maneuvers

A. REASON FOR REQUIREMENT

Cross-coupling of pitch and yaw motions -- both aerodynamic and inertial -- is common for modern airplanes. The ensuing motions can be violent in nature and can lead to prolonged loss of control.

B. RELATED MIL-F-8785 REQUIREMENT

3.4.3

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.8.1 Cross-Axis Coupling in Roll Maneuvers. In yaw-control-free, pitch-control-fixed, maximum-performance rolls through ____ deg, entered from straight flight or from turns, pushovers, or pullups ranging from 0 g to 0.8 n_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 combat-type maneuvers involving rolls through angles up to 360 degrees 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. These requirements define Level 1 and 2 operation. For Class II and III airplanes, these requirements apply in rolls through 120 degrees and rolls which are checked at a given bank angle.

Recommended values:

Airplane Class	Recommended Roll Angle (deg)
I and IV	360
II and III	120

D. RATIONALE BEHIND REQUIREMENT

The dynamics of the pitch and yaw coupling associated with rapid rolls are complex and nonlinear. In general, the dynamics involve interactions among the airplane inertia properties, aerodynamic properties, and the kinematics of the rolling motion. Because of the complexities involved, no attempt has been made to explain the mechanism of the various types of pitch-roll-yaw coupling. Instead, the reader is referred to References 167-171.

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 airplanes where very high roll rates are common or where reduced static stability is employed in the longitudinal axis, such as the F-16.

E. GUIDANCE FOR APPLICATION

It may be necessary to perform maneuvers with control combinations other than those specified to determine coupling problems, especially if the roll controller alone does not meet the control power requirements of 3.5.9.1. Such maneuvering is likely to occur during air combat evaluations with Class IV airplanes, as discussed in "Lessons Learned."

F. DEMONSTRATION OF COMPLIANCE

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.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

During prototype evaluation of the YF-16 Lightweight Fighter, coupled loss of control was encountered on two separate occasions. According to Reference 124, "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 full trailing-edge-down elevator."

Reference 124 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 stabilator saturation tests conducted 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 departures 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.

It should be noted that the so-called "stability axis" yaw damper (in which body centerline axis $r - p\alpha$ is fed back to the rudder) reduces sideslip in high-angle-of-attack maneuvering but increases roll rate. This results in increases in the pr and p^2 inertia coupling terms in the pitch axis, e.g.,

$$\dot{q} = \frac{I_z - I_x}{I_y} pr + \frac{I_{xz}}{I_y} (r^2 - p^2) + \frac{1}{I_y} M$$

and therefore aggravates pitch coupling. Use of such a yaw damper requires a compromise between allowable sideslip and inertia coupling.

3.8.2 Crosstalk Between Pitch and Roll Controllers

A. REASON FOR REQUIREMENT

Force and displacement requirements for pitch and roll controllers are separately specified elsewhere, but their operation in combination can cause problems if these characteristics are incompatible.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.4

C. STATEMENT OF REQUIREMENT

3.8.2 Crosstalk Between Pitch and Roll Controllers. The pitch- and roll-control force and displacement sensitivities and breakout forces shall be compatible so that intentional inputs to one control axis will not cause inadvertent inputs to the other.

D. RATIONALE BEHIND REQUIREMENT

Control harmony has several aspects. One problem is that the 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, for Class IV highly maneuverable aircraft it is often difficult to pull the centerstick straight back 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 these situations.

E. GUIDANCE FOR APPLICATION

No discussion necessary.

F. DEMONSTRATION OF COMPLIANCE

Compliance with this requirement will be shown through the course of normal simulation or flight testing, where pilot comments should reveal any potential deficiencies.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

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 ("cross-talk"), as described in Reference 124:

> ...*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 fifteen degrees.

Later operational experience with the F-16 indicates that crosstalk is indeed a noticeable problem. For example, left banks are commonly observed during the pitch rotation for takeoff and landing. The F-16 movable sidestick employs a 12 deg clockwise rotation that essentially eliminates crosstalk in takeoff and landing.

3.8.3 Control Harmony

A. REASON FOR REQUIREMENT

Normal maneuvering involving all three controllers can be taxing if any one controller requires especially large force inputs.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.4.1

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.8.3 Control Harmony. The following control force levels are considered to be limiting values compatible with the pilot's capability to apply simultaneous forces: ____________.

Recommended limits:

Control Type	Pitch	Roll	Yaw
Sidestick	20 lb	15 lb	
Centerstick	50 lb	25 lb	
Wheel	75 lb	40 lb (two-handed tasks) 25 lb (one-handed tasks)	
Pedal			175 lb

D. RATIONALE BEHIND REQUIREMENT

The cockpit control forces required to perform maneuvers which are normal for the airplane should have magnitudes which are related to the pilot's capability 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.

The 40 pounds 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 F-16 movable stick maximum forces (Reference 125) and results of the USAF Test Pilot School evaluations (Reference 23). The forces chosen are 75-90 percent of the maximum forces used.

E. GUIDANCE FOR APPLICATION

The key to applying this requirement is in definition of "normal" maneuvers. Such maneuvers should be specified by the missions defined for the aircraft.

F. DEMONSTRATION OF COMPLIANCE

In performing such maneuvers as dives, steady turns, and stalls, maximum forces should not exceed the combined forces specified herein.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

None.

3.8.4 Flight at High Angle of Attack

A. REASON FOR REQUIREMENT

Requirements on approach to stall, stalls, departures, and subsequent motions all apply for high-angle-of-attack (AOA) flight. This statement defines the purpose of the high-AOA requirements.

B. RELATED MIL-F-8785C REQUIREMENTS

3.4.2

C. STATEMENT OF REQUIREMENT

3.8.4 Flight at High Angle of Attack. The requirements of 3.8.4 through 3.8.4.3.2 concern stall warning, stalls, departures from controlled flight, post-stall gyrations, spins, recoveries, and related characteristics. They apply at speeds and angles of attack which in general are outside the Service Flight Envelope. They are intended to assure safety and the absence of mission limitations due to high-angle-of-attack characteristics.

D. RATIONALE BEHIND REQUIREMENT

There have been few changes to the MIL-F-8785C high-AOA requirements for the MIL Standard. As outlined in Reference 122, Interim Amendment 1 (USAF) completely revised the MIL-F-8785B requirements at high angle of attack, coordinated with the Air Force Flight Test Center's concurrent new stall/post-stall/spin demonstration requirements, MIL-S-83691. These changes were the result of reawakened interest in the area, occasioned by numerous aircraft losses. A large number of aircraft incidents have been attributed to loss of control at high angle of attack, and it was conjectured that many losses in Vietnam combat (with no evidence to determine a cause) might well be due to the same cause. Whereas previous requirements had concentrated on demonstration of acceptable stall and spin characteristics, the new requirements emphasize prevention of loss of control (departure) as well. All airplanes are covered with flight demonstration maneuvers and control abuse appropriate to the Class and mission. The requirements in this regime of nonlinearities remain largely qualitative. Amendment 2 changed many of Amendment 1's quantitative requirements related to test and evaluation techniques to qualitative statements, and MIL-F-8785C made no further changes in high-AOA requirements.

The stall and spin requirements that follow are related by their occurrence at high angles of attack. Therefore, this requirement is retained to serve as an overview of characteristics and problems with high-AOA flight. The discussions presented in "Lessons Learned" summarize recent insights and information on high-AOA flight, applicable in general to any of the stall/spin requirements.

E. GUIDANCE FOR APPLICATION

In applying the high-AOA requirements, it is worth considering that the term "high angle of attack" carries different connotations for different people. For example, 15 deg α may be considered high for a light single-engine Class I airplane, while pilots of highly maneuverable Class IV fighters might consider anything under 30 deg to be low angle of attack. Therefore, it is important to define just what one means.

Based upon the requirements of this section, high AOA is considered to be at and above the AOA for stall warning (3.8.4.2.1). This avoids need for or specification of a firm number for high AOA.

F. DEMONSTRATION OF COMPLIANCE

Not applicable.

G. SUPPORTING DATA

Not applicable.

H. LESSONS LEARNED

A recent survey of 33 aircraft manufacturers, research and test agencies, and operational commands and squadrons (Reference 177) provides considerable information on "mission phases or tasks involving high-AOA flight, past or present flying quality problems, stall/departure/spin encounter, future desires, etc." While information was sought on all Classes of airplane, most of the concern on high-AOA flight dealt with departure/spin resistance for Class IV, highly maneuverable airplanes. 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 Reference 177 survey is that, for Class IV airplanes,

> ...high-AOA maneuvering in combat, although spectacular or glamorous, is not a primary tactic. It is definitely a subordinate area but one which should not limit the 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 should not limit the aircraft usefulness, the major expressed concern involved departure/spin resistance, flight cues, and the role of the flight control system.

Most Class I airplanes are designed to meet FAA regulations (Reference 161) and adapted for military operations, so that high-AOA flight is looked at differently for their usage. Similarly, due to the very large inertias of Class III airplanes in all axes, high-AOA departure or large uncommanded motion is rarely encountered. The major concern for departures and spins (3.8.4.3) is therefore Class IV airplanes.

Table 1 (from Reference 177) summarizes pilot opinions on the high-AOA characteristics of several modern Class IV airplanes. In terms of design philosophy for high-AOA characteristics, Reference 177 concludes

TABLE 1 (3.8.4). DIGEST OF PILOT COMMENTS ON SPECIFIC AIRCRAFT HIGH-AOA FLYING CHARACTERISTICS (from Reference 177)

AIRCRAFT	OVERALL HIGH AOA F.Q.	DEPARTURE CHARACTERISTICS	CUES	OTHER
A-7	Departure hazard inappropriate to ground attack mission	Strong adverse yaw Severe nose slice Predictable-repeatable Easily recovered	Buffet	PCAS turned off at high α Wing stores increase stability
A-10	High AOA - usually defensive No adverse flying characteristics No worry about departure or spin High pitch rate capability — can pull through stall warning too fast	Very resistant to departure Very mild stall — little warning Mild buffet Some wing rock Mild yaw	α: Aural tone Peak performance -- steady Stall-beep V: Noise level Stick position	Ailerons remain effective in stall — like Cessna
F-4C,D,E	Acceptable to good for fighter Departure hazard for ground attack Good control effectiveness Must change control technique to rudder maneuvering	Strong adverse yaw Abrupt nose slice/roll Predictable-repeatable Recoverable (if sufficient altitude)	α: Buffet (poor, early, heavy) Stick position V: Stick force Dig-in Opt. Turn: Aircraft buzz	Force harmony problems at low dynamic pressure Can over-rotate or over-g Roll SAS turned off
F-4E (Leading edge slat)	Excellent Better separation between $C_{L_{max}}$ and departure α Less roll rate capability Use aileron and rudder to roll	Reduced adverse yaw Departure resistant Roll departure Somewhat unpredictable at very high α Recovers quickly	α: Buffet (good, steady increase) Aural tone Stick position V: Buffet increase Stick force Opt. Turn: Aircraft buzz	Roll SAS turned off
F-5E	Excellent Can point aircraft at very low speeds *Never worry about α* Loose aileron roll power -- *must use rudder maneuvering*	Departure resistant Rudder induced high yaw rate Difficult to recover	α: Buffet; stick position V: Flap horn Opt. Turn: Buffet	No roll rate CAS Full aft stick - max α Centerline stores degrade stability significantly

TABLE 1 (3.8.4). (Concluded)

AIRCRAFT	OVERALL HIGH AOA F.Q.	DEPARTURE CHARACTERISITICS	CUES	OTHER
F-14A	Good - "Honest" High control power Requires rudder maneuvering	Adverse aileron yaw Departure resistant Yaw/roll departure Severity is speed dependent	Generally poor Buffet Stick position Stick force	Main problem with asymmetric thrust PCAS turned off at high α
F-15A	Excellent High longitudinal control power Some worry about over-g SRI makes airplane consistant and repeatable Can override SRI	Departure resistant Nose slice Recover hands off Auto roll if inverted	α: Mild wing rock decreasing roll power nose drop at stall Opt. Turn: Light buffet	Constant Fs/g longitudinal CAS PCAS turned off at high α SRI provides all stick maneuvers $p\alpha + \delta_r$ causes inverted auto roll
F-16A	Excellent maneuvering Maneuver with abandon: no worry about g or departure Tendency to excessive use of high α because of poor cues Limiters "take over control", save poorly skilled pilot, restrict highly skilled pilot	Departure preventing system Can be tricked into Lat/Dir departure g-overshoot super stall Automatic anti-spin system Recovery sometimes difficult	None No stick cues No buffet No artificial cues	Constant F_s/n CAS SRI provides all stick maneuvering Maneuver limits on n, α, p Need limit changes with stores
F/FB-111	No warning of impending stall/departure Flying qualities excellent right up to departure Suddenly fall off cliff	Insidious Departure susceptible Nose slice/roll Unpredictable Non-recoverable at low altitude	No natural cues (stick, force, buffet) Artificial: Shaker Horn Lights	FCS provides uniform and cueless flying qualities Autotrim can produce inadvertent stall High α encounter often related to change in thrust

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 quality specification requirements should not dictate aircraft configuration, flight control system complexity, or even overly compromise primary mission performance."

It is as a result of these views, in combination, that the high-AOA requirements are essentially unchanged from MIL-F-8785C, except to make their wording more in the spirit of the MIL Standard.

3.8.4.1 Warning cues

A. REASON FOR REQUIREMENT

There is a constant need for clear, unambiguous cues to warn the pilot of potential stall, departure, and spin conditions in high-angle-of-attack flight.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.1.1

C. STATEMENT OF REQUIREMENT

3.8.4.1 Warning cues. Warning or indication of approach to stall, loss of aircraft control, and incipient spin shall be clear and unambiguous.

D. RATIONALE BEHIND REQUIREMENT

This statement is essentially identical to 3.1.7.1, "Warning and indication" for dangerous flight conditions. Its addition here is based upon three observations: 1) the requirement of 3.1.7.1 is intended for any dangerous flight condition, not specifically high-AOA flight; 2) there may be some instances (e.g., air combat maneuvering) for which high-AOA flight would not be considered "dangerous"; and 3) warning cues provided on many recent airplanes for high-AOA flight are considered inadequate (see "Guidance for Application" and "Lessons Learned").

E. GUIDANCE FOR APPLICATION

Providing a consistent, useful warning cue to the pilot continues to be a problem. A survey of pilots of Class IV airplanes (Reference 177) showed this is to be the case. Table 1 (3.8.4) lists the available high-angle-of-attack cues for various fighter airplanes. Reference 177 summarizes:

Lack of adequate high-AOA maneuvering/stall non-visual (e.g., tactile) cues ranked 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 stick 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 knot or g)
- Stick position
- Buffet level
- Uncommanded aircraft motion
- Artificial warning devices

It must be emphasized here that the intent of this requirement, like all the high-AOA requirements, is not to force an artificial limit on the airplane. The Reference 177 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.

F. DEMONSTRATION OF COMPLIANCE

Due to the complex nature of high-angle-of-attack flight, final compliance with this requirement will necessitate flight testing. If artificial warning cues are utilized, verification may include ground simulation.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

Pilot comments on the cues available in several fighter airplanes are summarized in Table 1 (3.8.4). Additional information on the F/A-18A at high angles of attack (Reference 245) shows that it has inadequate buffet and natural stick force cues, though an α feedback in the CAS provides a good artificial stick force cue. A warning tone is also employed.

Artificial warnings have proven to be inadequate on many airplanes. Reference 177 discusses the F/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 14 deg AOA, well below the departure AOA of 20-21 deg. 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.

3.8.4.2 Stalls

A. REASON FOR REQUIREMENT

This introductory statement specifies the conditions to be considered in applying the stall requirements of 3.8.4.2.1, 3.8.4.2.2, 3.8.4.2.3, and 3.8.4.2.4.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.2.1

C. STATEMENT OF REQUIREMENT

3.8.4.2 Stalls. The stall requirements apply for all Aircraft Normal States in straight unaccelerated flight and in turns and pullups with attainable normal accelerations up to n_L. Specifically, the Aircraft Normal States to be evaluated are: ____________________. Also, the requirements apply to Aircraft Failure States that affect stall characteristics.

D. RATIONALE BEHIND REQUIREMENT

The subparagraphs that fall under this statement contain qualitative and quantitative requirements pertaining to the stall. As Reference 122 concludes, stall classically corresponds to maximum lift coefficient, that is, $C_{L_\alpha} = 0$; but other accepted indicators of stall or maximum usable lift are uncommanded motion in pitch, roll or yaw, and intolerable buffeting. Consonant with deletion of specific rules for establishing the Permissible Flight Envelope, MIL-F-8785C deleted mention that V_S and α_S may be set by conditions other than aerodynamic flow separation. Although the contractor may set the low-speed bound of the Permissible Flight Envelope arbitrarily, there is no need to state that here. (Regardless of the boundary location, we would expect full stalls to be demonstrated if attainable.) Note that according to 3.8.4.1, the contractor must provide adequate warning or indication of approach to stalls.

A fixed-base simulator investigation of high-angle-of-attack characteristics (Reference 170) suggests that the definition of stall should be modified:

> [Previous specifications have defined] stall to be based on $C_{L_{max}}$; abrupt uncontrollable pitching, rolling, or yawing; or intolerable buffet. The results of our piloted simulation indicate that any abrupt aperiodic rolling or yawing motion which occurs without being preceded by noticeable "g-break" is considered [by the evaluation pilots] to be a departure, not a stall, and results in severely downgraded flying qualities. Thus, abrupt roll or yaw motion should be deleted as an allowable definition of stall and should not occur prior to stall. If such characteristics cannot be achieved with the airframe alone, then the flight control system should prevent reaching the AOA at which the abrupt rolling or yawing motion is obtained.

E. GUIDANCE FOR APPLICATION

The configuration, throttle settings, and trim settings of 4.2.2 may be specified for investigation.

F. DEMONSTRATION OF COMPLIANCE

Due to the extremely nonlinear nature of stalls and stall warnings, the procuring activity should consider simulation or flight testing in lieu of analysis for demonstrating compliance (see, e.g., Reference 170).

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

The definitions of stall (Paragraphs 4.2.2 and 4.2.5) include occurrence of "uncommanded pitching, rolling, or yawing." This also is a characteristic of (undesirable) departure. The difference between the two generally involves energy state. The higher the energy state, the more rapid the uncommanded motion. If uncommanded pitch, roll, or yaw defines both stall and departure, then some rate of motion boundary should be established to distinguish between the two. At present there is insufficient information to define such a boundary.

Results of the piloted simulation of Reference 170 indicate that "any abrupt aperiodic rolling or yawing motion which occurs without being preceded by noticeable 'g-break' is considered to be a departure, not a stall, and results in severely downgraded flying qualities. Thus, abrupt roll or yaw motion should be deleted as an allowable definition of stall and should not occur prior to stall." This could be especially critical in accelerated flight, where it is possible to pull rapidly through any stall warning or g-break, and into a depature.

3.8.4.2.1 Stall approach

A. REASON FOR REQUIREMENT

Approach to stall should always be clearly indicated to the pilot with sufficient margin (airspeed or angle of attack) to recover from the incipient stall.

B. RELATED MIL-F-8785C REQUIREMENTS

3.4.2.1.1, 3.4.2.1.1.1, 3.4.2.1.1.2

C. STATEMENT OF REQUIREMENTS AND RECOMMENDED VALUES

3.8.4.2.1 Stall approach.

a) The onset of warning of stall approach (3.8.4.1) shall occur within the following speed range for 1 g stalls: ____, and within the following range (or percentage) of lift for accelerated stalls: ____________, but not within the Operational Flight Envelope.

b) An increase in intensity of the warning with further increase in angle of attack shall be sufficiently marked to be noted by the pilot. The warning shall continue until the angle of attack is reduced to a value less than that for warning onset. Prior to the stall, uncommanded oscillations shall not result in flying qualities less than Level ___.

c) At all angles of attack 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.

Recommended warning ranges:

1 g Stalls:

Flight Phase	Minimum Speed for Onset	Maximum Speed for Onset
Approach	Higher of $1.05V_S$ or V_S + 5 knots	Higher of $1.10V_S$ or V_S + 10 knots
All other	Higher of $1.05V_S$ or V_S + 5 knots	Higher of $1.15V_S$ or V_S + 15 knots

Accelerated Stalls:

Flight Phase	Minimum Lift at Onset	Maximum Lift at Onset
Approach	82% C_L stall	90% C_L stall
All other	75% C_L stall	90% C_L stall

For Part b) of the requirement it is recommended that Level 2 flying qualities be required.

D. RATIONALE BEHIND REQUIREMENT

The MIL-F-8785C paragraphs combined in this requirement are largely unchanged. The largest modification has been in the wording of warning for stall approach. Where (a) refers to the new, all-encompassing warning requirement (3.8.4.1), MIL-F-8785C specified that "The stall approach shall be accompanied by an easily perceptible warning consisting of shaking of the cockpit controls, buffeting or shaking of the airplane, or a combination of both." Removal of this phrase is consistent with both the new warning cue requirement and the MIL Standard format of minimal wording in the Standard.

The stall approach requirement of MIL-F-8785 (3.4.2.1.1) explicitly forbade onset of stall warning within the Operational Flight Envelope. However, for Class III aircraft in particular, most stalls are encountered during high altitude cruise, in-flight refueling, or in the landing pattern where the Operational and Service Flight Envelope boundaries tend to coalesce. Thus stall warning onset within the Operational Flight Envelope may be desirable and should not be legislated against.

The warning range for accelerated stalls would, ideally, be mission dependent (e.g., air-to-ground versus air-to-air), considering the average altitude available for recovery, the rapidity of speed bleedoff for the vehicle/weapon configuration, and departure susceptibility/severity characteristics. There is insufficient data available to establish such mission-dependent criteria, so the requirements of MIL-F-8785C have been retained.

E. GUIDANCE FOR APPLICATION

A requirement limiting "uncommanded oscillations" is clearly quite arbitrary: one pilot may prefer no uncommanded motion associated with approach to stall, while another might consider such motion a "necessary evil" and find oscillations acceptable. The results of the piloted simulation of Reference 170 suggested that a noticeable "g-break" indicated stall while any aperiodic uncommanded motion (in any axis) of greater than 20 deg/sec signified departure.

F. DEMONSTRATION OF COMPLIANCE

Because of the nonlinear and unpredictable nature of high-angle-of-attack aerodynamics, demonstration of compliance should be accomplished in fight test.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

See Reference 170.

3.8.4.2.2 Stall characteristics

A. REASON FOR REQUIREMENT

In order for an airplane to be controllable in a developed stall, uncommanded angular excursions should be reasonably small and, in the case of pitch excursions, in a direction that will enhance controllability. If these conditions cannot be met, the aircraft is considered to have departed from controlled flight.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.2.1.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.8.4.2.2 Stall characteristics. The following apply for all stalls, including stalls entered abruptly:

a) In the unaccelerated stalls of 3.8.4.2.1, the aircraft shall not exhibit rolling, yawing, or downward pitching at the stall which cannot be controlled to stay within ___ deg.

b) It is desired that no pitchup tendencies occur in unaccelerated or accelerated stalls. However, in unaccelerated stalls, mild nose-up pitch may be acceptable if no pitch control force reversal occurs and if no dangerous, unrecoverable or objectionable flight conditions result. In accelerated stalls, mild nose-up tendency may be acceptable if the operational effectiveness of the airplane is not compromised and the airplane has adequate stall warning, pitch control effectiveness is such that it is possible to stop the pitchup promptly and reduce the angle of attack, and at no point during the stall, stall approach or recovery does any portion of the airplane exceed structural limit loads.

Recommended values: Recommended angular excursion limits are 20 deg for Classes I, II and III, or 30 deg for Class IV airplanes.

D. RATIONALE BEHIND REQUIREMENT

The 20 deg or 30 deg limits are on the amount of attitude change at stall. Any such change should be in a normal direction.

E. GUIDANCE FOR APPLICATION

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 (3.2.7.2, 3.5.8.2, 3.6.4.2) are recommended to be less than ±0.5 g laterally or longitudinally and ±10 deg/sec roll rate within 2 seconds. A similar constraint could be defined for unaccelerated stalls.

F. DEMONSTRATION OF COMPLIANCE

As with all the stall requirements, flight testing will be necessary.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

References 168 and 170 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 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, it probably should be; some sideslip is common, even unavoidable at high angles of attack. Airplanes rarely have a decent zero-sideslip reference.

This requirement legislates against pitchup tendencies, but not against pitch down. For large (Class III) airplanes, nose-down pitching is undesirable, according to Reference 177, 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 nonrecoverable 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 lost due to settling is less than that due to a diving recovery.

This is consistent with the training procedures used for civil transport aircraft. For example, Reference 246 describes the stall series used in 747 training and recurrent checks:

> ...a Vref (final approach) speed is computed for the landing weight and a "bug" positioned next to this number on the airspeed gauge. The first stall is made clean with wings level, the next in a 20-degree bank with 10 degrees of flaps, and the third straight ahead with the gear down and landing flaps (30 degrees). 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 five degrees 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 ensures a secondary buffet or shaker, or both, and substantial altitude loss.

3.8.4.2.3 Stall prevention and recovery

A. REASON FOR REQUIREMENT

Pitch control power at the start of the stall warning (see 3.8.4.1) must be sufficient to prevent the stall without excessive pilot effort; recovery after stalling should likewise be possible with moderate effort.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.2.1.3

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.8.4.2.3 Stall prevention and recovery.

a) It shall be possible to prevent the stall by moderate use of the pitch control alone at the onset of the stall warning.

b) 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 _______, and to regain level flight without excessive loss of altitude or buildup 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.

c) In the straight flight stalls of 3.8.4.2, with the aircraft trimmed at an airspeed not greater than 1.4 V_s, pitch control power shall be sufficient to recover from any attainable angle of attack.

Recommended value: It is recommended that the control force limits of 3.8.3, "Control Harmony," be applied.

D. RATIONALE BEHIND REQUIREMENT

Prevention of and recovery from the stall must always be simple for the pilot. MIL-F-8785C included the requirement that throttles remain 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) airplanes and heavy (Class III) airplanes: 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.

E. GUIDANCE FOR APPLICATION

None.

F. DEMONSTRATION OF COMPLIANCE

Part b) will almost assuredly require some flight testing; for Part a), simulation may be sufficient provided the aircraft aerodynamics are well modeled up to the stall.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

A potential criterion for specifying stall recovery requirements for Class III airplanes is "dwell time" (Reference 177). As mentioned in 3.8.4.2.2, "dwell time" is the time between occurrence of the stall and recovery of flying speed. This is in accordance with the 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.

Reference 247 shows that for three Class III airplanes (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 percent of the stalls. It therefore suggests that a pitch recovery criterion could be that the pitch control produce $\ddot{\theta} > 0.8$ rad/sec^2 at stall.

During stall testing of the F-16A/B with aft c.g. loadings, an upright deep stall was encountered, requiring a spin parachute for recovery. Figure 1 shows a time history of a deep stall. Analysis of

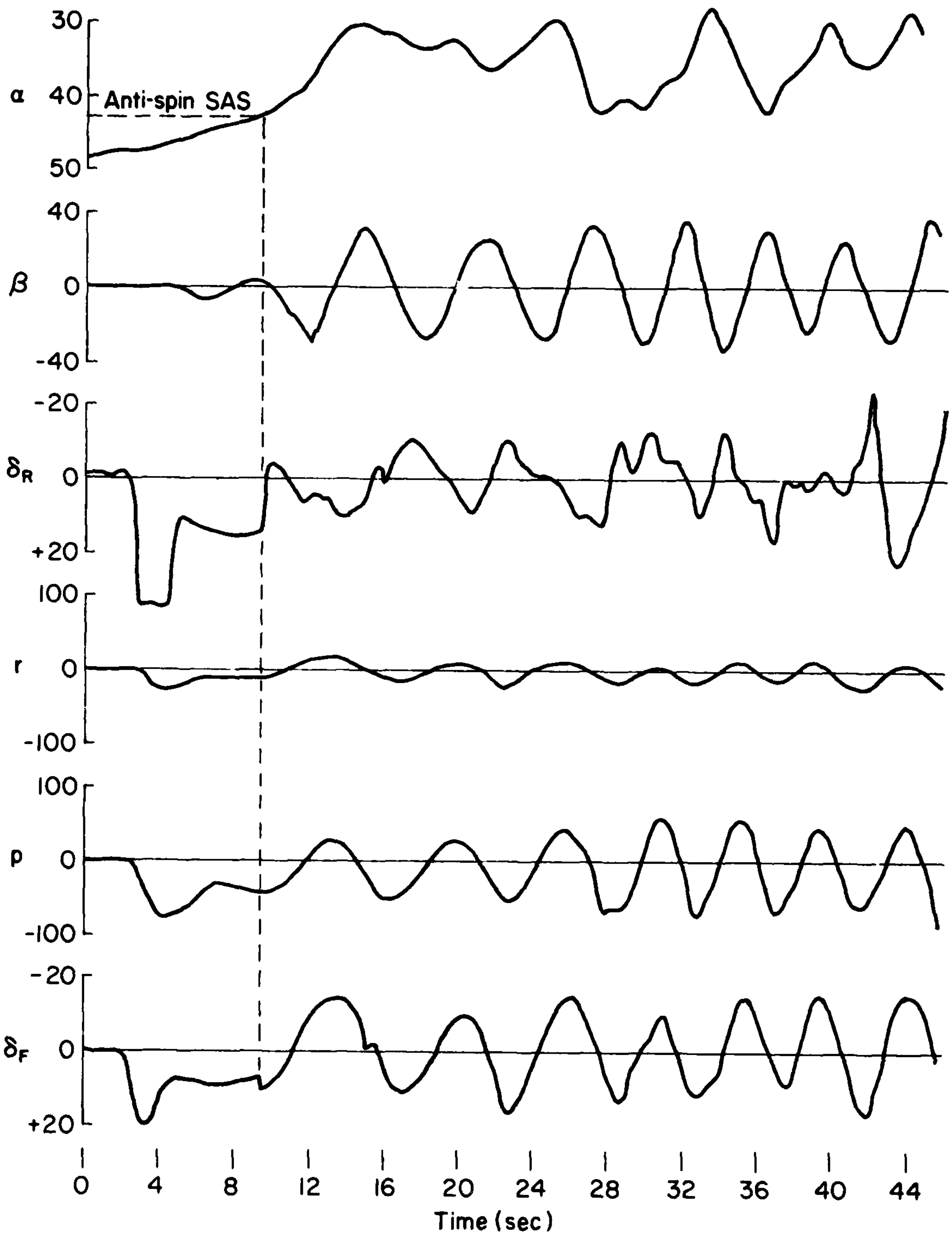


Figure 1 (3.8.4.2.3). Time History of Unrecoverable Deep Stall Encountered by F-16B (Reference 248) the F-16 flight control system suggests that the deep stall condition may have been aggravated by an anti-spin SAS (Figure 2) which is activated at $\alpha \geq 29$ deg, combined with a longitudinal stick gain to remove the pilot from the loop. Figure 1 shows the point at which the anti-spin SAS became active (t ≈ 10 sec). A lateral limit cycle oscillation developed, possibly caused by the anti-spin SAS, and cross-axis coupling caused the aircraft to pitch to still higher AOA and subsequent deep stall.

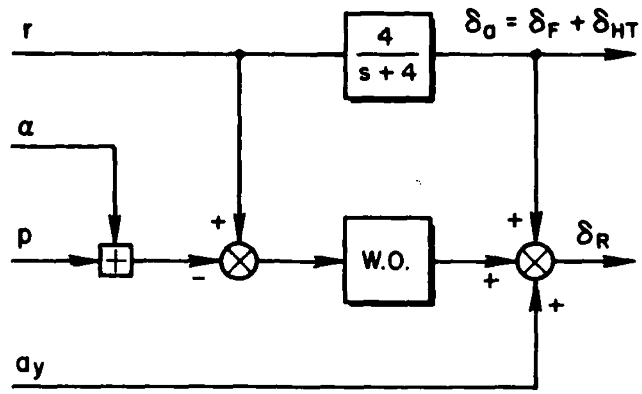


Figure 2 (3.8.4.2.3). Anti-Spin SAS for F-16B
(α > 29 deg)

Recovery from this deep stall [which might arguably be defined as a post-stall gyration (3.8.4.3.2)], without a spin chute, required a manual pitch override (MPO) in the longitudinal SAS (Reference 248):

> ...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 29 degrees, eliminated the AOA limiting and pitch integrator functions to allow trailing edge up (TEU) stabilator control.

An MPO switch is now included in production aircraft, but, according to Reference 248, 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 the 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 90 degrees bank angle) masked the pitching motion of the aircraft.

Such phasing between stick and airplane motion could be considered a violation of the wording of this requirement; i.e., this is not a "simple" use of the pitch control.

3.8.4.2.4 <u>One-engine-out stalls</u>

A. REASON FOR REQUIREMENT

Some multi-engine airplanes exhibit violent, unacceptable rolling or yawing tendencies in engine-out stalls.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.2.1.3.1

C. STATEMENT OF REQUIREMENT

3.8.4.2.4 <u>One-engine-out stalls</u>. On multi-engine aircraft it shall be possible to recover safely from stalls with the critical engine inoperative. Thrust on the remaining engines will be at: __________.

Thrust values or ranges should be provided by the procuring activity (see "Guidance for Application").

D. RATIONALE BEHIND REQUIREMENT

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.

E. GUIDANCE FOR APPLICATION

For best application of this requirement the procuring activity may choose to specify the Flight Phases and thrust settings for testing. MIL-F-8785C included the following table:

Flight Phase	Thrust
TO	Takeoff
CL	Normal climb
PA	Normal approach
WO	Waveoff

FAR Part 25 (Reference 118) requires that recovery be possible "with the remaining engines at up to 75 percent 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 (Reference 161) is more severe in that it has the additional requirement that the airplane not display any undue spinning tendency during the single engine stall demonstration.

Throttling back on the operating engines during stall recovery is allowable.

F. DEMONSTRATION OF COMPLIANCE

Flight testing will be necessary.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

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 condition due to a flat spin. This has occurred on contemporary fighter aircraft as well as on light twin engine aircraft, usually as a result of delayed recovery controls. Because of the inherent critical nature of this demonstration, it is recommended that auxiliary spin recovery devices be installed during the tests.

3.8.4.3 Departures and Spins

A. REASON FOR REQUIREMENT

The conditions for consideration of departure and recovery from post-stall gyrations and spins are delineated.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.2.2

C. STATEMENT OF REQUIREMENT

3.8.4.3 Departures and Spins. The post-stall gyration and spin requirements apply to all modes of motion that can be entered from upsets, decelerations, and extreme maneuvers appropriate to the Class and Flight Phase Category. The requirements hold for all Aircraft Normal States and for all states of stability and control augmentation systems, except approved Special Failure States. Store release shall not be allowed during loss of control, spin or gyration, recovery, or subsequent dive pullout. Automatic disengagement of augmentation systems, however, is permissible if it is necessary and does not prevent meeting any other requirements; re-engagement shall be possible in flight following recovery. Specific flight conditions to be evaluated are: ______________________.

D. RATIONALE BEHIND REQUIREMENT

Similar to the introductory requirement for stalls (3.8.4.2), the conditions to be considered are specified for departures and spins. Several lines of the MIL-F-8785C counterpart dealing with specific test conditions have been removed to encourage more mission- and Flight Phase-oriented requirements. If the user prefers, the more general MIL-F-8785C conditions may be specified (see "Guidance for Application").

E. GUIDANCE FOR APPLICATION

MIL-F-8785C included the following as guidelines for testing:

> Entries from inverted flight shall be included for Class I and Class IV airplanes. Entry angles of attack and sideslip up to maximum control capability and under dynamic flight conditions are to be included, except as limited by

> structural considerations. For all Class and Flight Phase Categories, thrust settings up to and including MAT shall be included, with and without one critical engine inoperative at entry.

Rather than tabulate explicit conditions to be investigated, the procuring activity may choose to simply require compliance with these guidelines. However, Reference 177 takes exception to the last sentence above:

> The one-engine-inoperative requirement should not be a universal requirement. It may be a legitimate requirement for large multi-engine aircraft but not for twin-engine fighters where asymmetric thrust moments can exceed available control moments. Inadvertent loss of one engine during departures in the F-14 invariably leads to a non-recoverable flat spin.

F. DEMONSTRATION OF COMPLIANCE

If spin testing is required for complying with 3.8.4.3.1 and 3.8.4.3.2, the contractor must follow the guidelines of this requirement. If actual demonstration is not required, some flight testing and some analysis may still be necessary (see 3.8.4.3.2 and discussion).

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

See discussions on the deep stall characteristics of the F-16 ("Lessons Learned," 3.8.4.2.3) concerning the augmentation system modifications used for recovery.

3.8.4.3.1 Departure from controlled flight

A. REASON FOR REQUIREMENT

Departure resistance is a prime concern for highly maneuverable aircraft operating in high-angle-of-attack flight. So far it has been difficult to arrive at an agreed-upon method of predicting departure susceptibility.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.2.2.1

C. STATEMENT OF REQUIREMENT

3.8.4.3.1 Departure from controlled flight. The aircraft shall be resistant to departure from controlled flight, post-stall gyrations and spins. Adequate warning of approach to departure (3.8.4.1) shall be provided. The airplane shall exhibit no uncommanded motion which cannot be arrested promptly by simple application of pilot control.

D. RATIONALE BEHIND REQUIREMENT

The definitions of departure susceptibility and resistance from MIL-S-83691A 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 and 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.
>
> Extemely resistant to departure: departure from controlled flight can only occur after an abrupt and inordinately sustained application of gross, abnormal, pro-departure controls.

MIL-F-8785C required the airplane to be <u>extremely</u> resistant; this has been reduced to resistant. In the words of Reference 177, "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 <u>resistant</u>." See also "Lessons Learned," 3.8.4. Easing this requirement also allows for those (admittedly rare) occasions when pilots of Class IV airplanes 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 3.8.4.1) reflects pilots' concerns. According to Reference 177,

> Warning is needed which is separate and distinct from stall warning. Margins (maximum and minimum) between warning onset and actual departure should be dependent upon pitch control power (how rapidly the aircraft can transit the warning region), departure severity, spin susceptibility, and aircraft mission.

Several quantitative requirements were suggested in Reference 170. While there is insufficient support for incorporating them as such, they may be very useful for guidance in early analysis. Detailed discussion of this reference is given in "Supporting Data." 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 pilot's theshold for uncommanded motion or ability to cope. As such it is a <u>pilot-centered criterion</u> which should be

> applicable 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 [Reference 175] 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 quality specification item.

The Lateral Control Divergence Parameter is defined, in stability axes, as

$$LCDP = C_{n_\beta} - \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}} C_{\ell_\beta}$$

Generally, LCDP should be greater than about -0.001. For the unaugmented airframe, according to Reference 170,

> 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 motion exceed 20 deg/sec was made in Reference 170, "based on a rough average of the simulation pilots' commentary as to <u>their</u> definitions of departure."

E. GUIDANCE FOR APPLICATION

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."

F. DEMONSTRATION OF COMPLIANCE

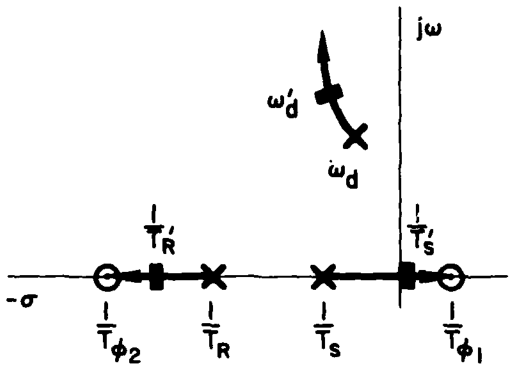
Flight testing is almost a necessity since it is difficult to define an accurate aerodynamic model for post-stall flight. If simulation is allowed, the procuring activity may prefer fixed-base over moving-base to avoid problems with confusing or unrealistic motions that might influence pilots' perceptions.

C. SUPPORTING DATA

Recent research contracts sponsored by the Air Force have generated information into the causes of departures and spins. Some of these are discussed in Reference 176; examples of resulting reports include References 168, 170, 175, and 177.

In the fixed-base piloted simulation of Reference 170, various maneuvers (bank-to-bank and windup turns, and pullups) were performed with and without a target aircraft. The simulated aircraft 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-83691A definitions, see "Rationale Behind Requirement") were different for the two pilots.

In closed-loop pilot/vehicle analysis it was found that flying techniques at high angles of attack were quite different for the two pilots. However, a correlating factor was found to be the value of one zero of the roll attitude numerator $N^{\phi}_{\delta_{stk}}$, where δ_{stk} simply indicates that lateral stick controls a combination of ailerons and rolling spoilers (of the six aerodynamic configurations evaluated, two included lateral augmentation with a stick-to-rudder interconnect as well). At high angles of attack and in asymmetric flight, Reference 170 shows that extreme adverse aileron yaw or lateral-longitudinal coupling can produce an unstable zero in the $N^{\phi}_{\delta_{stk}}$ numerator. (This zero may be first-order, or second-order if $\zeta_{\phi} < 1$), while the vehicle dynamics (i.e, ζ_p, ω_p, ζ_{sp}, ω_{sp}, $1/T_R$, $1/T_s$, ζ_d, and ω_d) may all be acceptable. Thus, the vehicle controls-free is stable, but pilot attempts to control roll attitude can produce a closed-loop instability, as shown in the sketch below:



Results of the Reference 170 simulation found strong correlation between the value of $1/T_{\phi_1}$ at departure and pilot evaluations of departure susceptibility, Figure 1.

For classical aircraft with no lateral-longitudinal coupling, the square of the $N^{\phi}_{\delta_{stk}}$ frequency term, ω_ϕ^2 [or $(1/T_{\phi_1})(1/T_{\phi_2})$] is approximated (Reference 66) by:

$$\omega_\phi^2 \doteq N_\beta' - L_\beta' \frac{N_{\delta_a}'}{L_{\delta_a}'}$$

$$\doteq \quad C_{n_\beta} - C_{\ell_\beta} \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}} \quad \frac{\bar{q}Sb}{I_z}$$

LCDP, which is defined by the bracketed expression above, is simply a non-dimensional approximation for ω_ϕ^2. In the Reference 170 evaluation, $1/T_{\phi_1}$ of -0.5 corresponded to LCDP of -0.001.

III. LESSONS LEARNED

None.

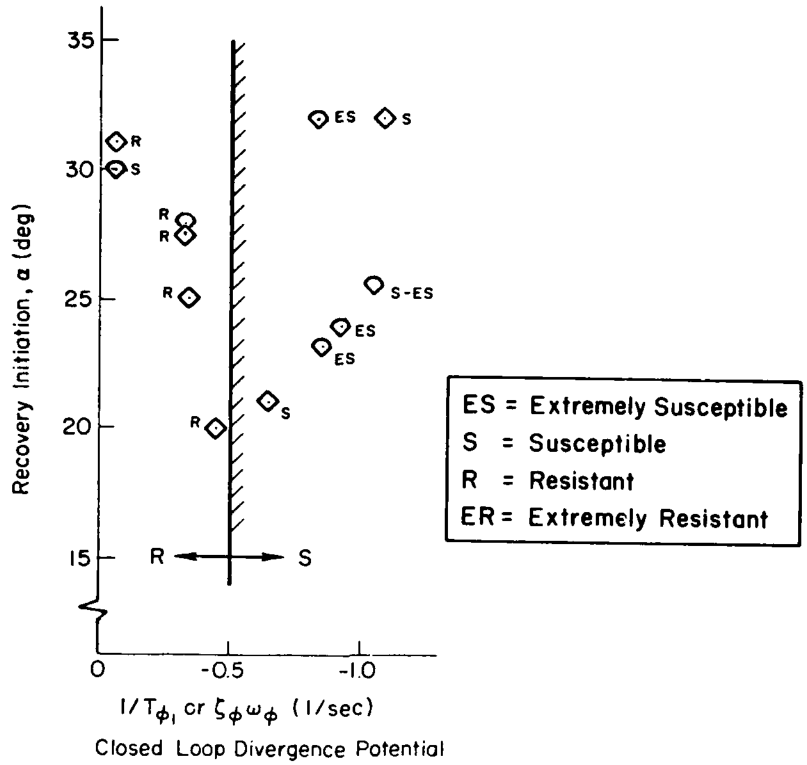


Figure 1 (3.8.4.3.1). Departure Susceptibility Rating Versus Lateral Closed-Loop Divergence Potential (from Reference 170)

3.8.4.3.2 Recovery from post-stall gyrations and spins

A. REASON FOR REQUIREMENT

Recovery from post-stall gyrations and spins must be possible and prompt, with simple control application.

B. RELATED MIL-F-8785C REQUIREMENT

3.4.2.2.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.8.4.3.2 Recovery from post-stall gyrations and spins. For aircraft that, according to MIL-A-8861, must be structurally designed for spinning:

a) The proper recovery technique(s) must be readily ascertained by the pilot, and simple and easy to apply under the motions encountered.

b) A single technique shall provide prompt recovery from all post-stall gyrations and incipient spins, without requiring the pilot to determine the direction of motion and without tendency to develop a spin. The same technique used to recover from post-stall gyrations and incipient spins, or at least a compatible one, is also desired for spin recovery. For all modes of spin that can occur, these recoveries shall be attainable within: ________________________.

c) Avoidance of a spin reversal or an adverse mode change shall not depend upon precise pilot control timing or deflection. It is desired that all aircraft be readily recoverable from all attainable attitudes and motions. The post-stall characteristics of those aircraft not required to comply with requirements of this paragraph shall be determined by analysis and model test.

d) Safe and consistent recovery and pullouts shall be accomplished without exceeding the following forces: ________________, and without exceeding structural limitations.

Recommended values: Recovery should be specified in terms of allowable altitude loss or number of turns, measured from the initiation of recovery action:

Class	Flight Phase	Turns for Recovery	Altitude Loss*
I	Category A, B	1-1/2	1000 ft
I	PA	1	800 ft
Other classes	PA	1	1000 ft
Other classes	Category A, B	2	5000 ft

*Not including dive pullout.

For conditions that require control actions in more than one axis it is recommended that the forces specified by Para. 3.8.3, Control Harmony, be applied.

D. RATIONALE BEHIND REQUIREMENT

Reference 122 described the evolution of the Mil-F-8785 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 airplanes to be spun, MIL-S-25015 required ready recovery from incipient and fully developed (5-turn) spins -- 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 on 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 for the start of recovery to be apparent within 3 seconds

> or one 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 real-world approaches to spin recovery. The specification of recovery in terms of altitude loss is a return to Amendment 1 of MIL-F-8785B, based upon what the pilot really is concerned about. For example, the piloted simulation of Reference 170 included an airplane model that would not spin, but showed a

> ...low-frequency wallowing that masks departure. At the same time, the wallowing does not generate sufficiently rapid motion to excite inertia cross-coupling and PSG. All pilots tended 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 recover by itself.

The 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."

Reference 177 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.

Ideally the altitude-loss requirement would also be a function of altitude above the ground, since a PSG at (say) 80,000 ft would not be as critical as one at 2000 ft above the ground.

E. GUIDANCE FOR APPLICATION

This requirement is intended for airplanes that must be designed to withstand the forces of post-stall gyrations and spins, based upon MIL-A-8861.

F. DEMONSTRATION OF COMPLIANCE

The procuring activity may choose to require flight testing for some airplanes (especially Classes I and IV). Analysis and spin-model testing may augment these flight tests or, for other aircraft, take their place.

G. SUPPORTING DATA

None.

H. LESSONS LEARNED

The F-4 series of airplanes serve as excellent examples of what is good and bad with this requirement. Reference 130 summarizes a wide body of experience in spin testing of the F-4. The airplane was predicted by model tests to have steep erect and inverted oscillatory modes as well as a flat spin mode. Reference 130 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 60° to 80° at the 1/4 turn position followed by an increase in yaw rate. After 1/2 turn in yaw the airplane would pitch up to near level and in some cases 10° to 20° ANU, depending upon the energy conditions at entry. The yaw rate was usually at a minimum when the pitch attitude (and angle of attack) was at a maximum. The airplane was concurrently oscillating ±60° 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 4-1/2 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 and developed spins was consistent and effective in all but flat spins. It also failed to meet the recovery requirement, since the pilot had to determine the direction of motion:

> 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 1/2 to 1-1/2 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 angle of attack was below stall the aircraft would roll up to 540° 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 airplane (Figure 1). The airplane 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 netralizing 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 Reference 130,

> 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 wake characteristics between the airplane and the spin chute. The only apparent effect of the control cycling was an increase in yaw rate to above 100°/sec.

These results serve to emphasize the importance of approaching spin testing with great care. More recent airplanes support the need to make this requirement very flexible.

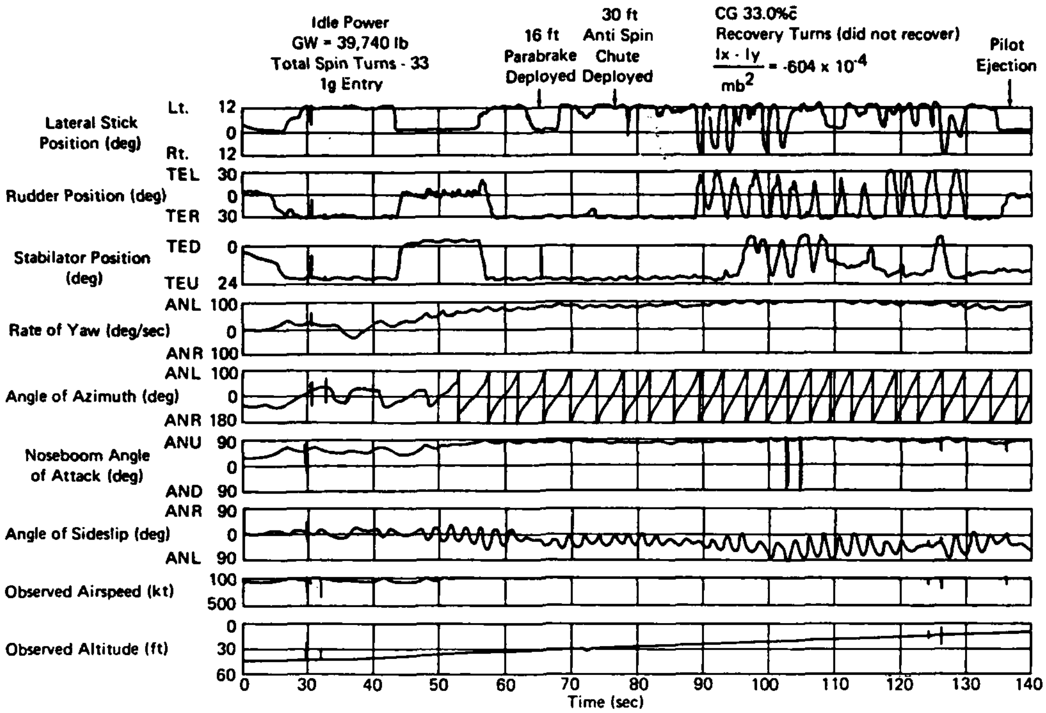


Figure 1 (3.8.4.3.2). Left Flat Spin, F-4B
(from Reference 130)

Recovery from the F-16 deep stall ("Lessons Learned," 3.8.4.2.3) required both a manual pitch SAS override switch and proper application of longitudinal stick to "rock" the airplane out of the stall -- an action which required the pilot to determine the direction of motion, albeit in pitch and not yaw.

High-angle-of-attack testing of the F-18 (Reference 245) 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 20° and 40°/second, an angle of attack between 50° and 60°, a steep nose-low attitude, and fairly smooth pitch and roll rates.
- An oscillatory intermediate mode with yaw rates between 50° and 80°/second and an angle of attack between 60° and 80°.

- A smooth flat mode at 90° to 140°/second yaw rate with an angle of attack between 80° and 85°.

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 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 mode" spin is not unlike 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.

3.9 HANDLING QUALITY REQUIREMENTS IN ATMOSPHERIC DISTURBANCES

3.9.1 Allowable Handling Qualities Degradations in Atmospheric Disturbance

A. REASON FOR REQUIREMENT

This requirement is included to provide a rational means for specifying the allowable degradation in handling qualities in the presence of increasing 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 Reference 222).

B. RELATED MIL-F-8785C REQUIREMENT

3.8

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.9.1 Allowable Handling Qualities Degradations in Atmospheric Disturbances. Level ___ flying qualities are required for atmospheric disturbance levels up to and including ______________________________ and wind shears of magnitude ______________________.

1. Recommendations for Flying Quality Levels

The required flying quality Levels are to be adjusted according to Table 1 or Table 2, depending on the definition selected in Paragraph 1.7 (i.e., pilot ratings or adjectival phrases).

2. Definitions of Moderate, Severe, and Extreme Turbulence

a. Flight Test

Wherever possible, the atmospheric disturbances that existed during the flight test should be measured and applied to Tables 1 and 2 according to the definitions in Table 3.

TABLE 1 (3.9.1). DEFINITION OF LEVELS WHEN LEVELS ARE DEFINED BY COOPER HARPER PILOT RATING SCALE IN PARAGRAPH 1.7

LEVEL	ATMOSPHERIC DISTURBANCES			
	LIGHT	MODERATE	SEVERE	EXTREME
1	3-1/2	5-1/2	7-1/2	Flying qualities such that control can be maintained long enough to fly out of the disturbance
2	6-1/2	7-1/2	Flying qualities such that control can be maintained long enough to fly out of the disturbance	Flying qualities such that pilot can regain control after being upset
3	9-1/2	Flying qualities such that control can be maintained long enough to fly out of the disturbance	Flying qualities such that pilot can regain control after being upset	No requirement

TABLE 2 (3.9.1). DEFINITION OF LEVELS WHEN LEVELS ARE DEFINED BY ADJECTIVAL PHRASES IN PARA 1.7

LEVEL	ATMOSPHERIC DISTURBANCES			
	LIGHT	MODERATE	SEVERE	EXTREME
1	Flying qualities clearly adequate for the mission Flight Phase	Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both exists	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed.	Flying qualities such that control can be maintained long enough to fly out of the disturbance
2	Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists.	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed.	Flying qualities such that control can be maintained long enough to fly out of the disturbance	Flying qualities such that pilot can regain control after being upset
3	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed.	Flying qualities such that control can be maintained long enough to fly out of the disturbance	Flying qualities such that pilot can regain control after being upset	No requirement

TABLE 3 (3.9.1). ATMOSPHERIC DISTURBANCE CRITERIA FOR IN-FLIGHT EVALUATIONS

INTENSITY	AIRCRAFT REACTION	REACTION INSIDE AIRCRAFT
LIGHT	Turbulence that momentarily causes slight, erratic changes in altitude and/or attitude (pitch, roll, yaw). Report as Light Turbulence;* or Turbulence that causes slight, rapid and somewhat rhythmic bumpiness without appreciable changes in altitude or attitude. Report as Light Chop.	Occupants may feel a slight strain against seat belts or shoulder straps. Unsecured objects may be displaced slightly.
MODERATE	Turbulence that is similar to Light Turbulence but of greater intensity. Changes in altitude and/or attitude occur but the aircraft remains in positive control at all times. It usually causes variations in indicated airspeed and flight path. Report as Moderate Turbulence;* or Turbulence that is similar to Light Chop but of greater intensity. It causes rapid bumps or jolts without appreciable changes in aircraft altitude or attitude. Report as Moderate Chop.	Occupants feel definite strains against seat belts or shoulder straps. Unsecured objects are dislodged. Food service and walking are difficult.
SEVERE	Turbulence that causes large, abrupt changes in altitude and/or attitude. It usually causes large variations in indicated airspeed. Aircraft may be momentarily out of control. Report as Severe Turbulence.*	Occupants are forced violently against seat belts or shoulder straps. Unsecured objects are tossed about.
EXTREME	Turbulence in which the aircraft is violently tossed about and is practically impossible to control. It may cause structural damage. Report as Extreme Turbulence.*	

*High level turbulence (normally above 15,000 feet ASL) not associated with cumuliform cloudiness, including thunderstorms, should be reported as CAT (clear air turbulence) preceded by the appropriate intensity, or light or moderate chop.

The definitions in Table 3 are suggested for establishing disturbance magnitudes encountered in flight test when quantitative measurements cannot be made.

b. Simulation

Light, moderate, and severe turbulence levels are defined quantitatively in terms of rms horizontal gust (σ_{u_g}) in Table 4. It is not necessary or even desirable to vary the gust magnitude as a function of position or altitude when complying via simulation. The recommended windshears are given as follows:

Decreasing Headwind:	$g\gamma_{max}$ not to exceed 3.4 ft/sec^2
Decreasing Tailwind:	$g\gamma_{min}$ not to exceed 1.7 ft/sec^2
Vector Shear:	9 deg/sec; V_w = 20 kt
Duration of All Shears:	At least 10 sec

where γ_{max} is the maximum power climb angle in the configuration used at wind shear initiation. γ_{min} is the flight path angle for flight idle in the configuration used at wind shear initiation.

TABLE 4 (3.9.1)

ATMOSPHERIC DISTURBANCE DEFINITIONS FOR SIMULATION AND FLIGHT TEST

MAGNITUDE	σ_{u_g} (ft/sec)
Light	0-3
Moderate	5
Severe	10
Extreme	24

Finally, the following steady crosswind components corresponding to "light," "moderate," and "severe" disturbances are recommended:

Qualitative Atmospheric Disturbance Level	Steady Crosswind (kt)
Light	0-10
Moderate	11-30
Severe	31-45

These crosswinds should exist at touchdown. When complying via piloted simulation, the wind 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.

D. RATIONALE BEHIND REQUIREMENT

The intent of this requirement is to insure that atmospheric turbulence is accounted for in a reasonable way. The adjustments in Level definitions are presented in terms of adjectival phrases and Cooper-Harper pilot ratings to be consistent with the definitions in Paragraph 1.7. The definitions that use pilot ratings allow a more fine-grained distinction. This has been utilized to define a more appropriate degradation in flying qualities with turbulence. For example, the Level 1 definition in Moderate turbulence is 5-1/2 (as opposed to 6-1/2). The rationale for this is summarized as follows:

- Adequate performance should be obtainable with considerable compensation; extensive compensation is felt to be excessive for flight in moderate turbulence (see Figure 1).
- During a several-year simulation effort to develop STOL airworthiness criteria for the FAA the evaluation pilots generally agreed that 5-1/2 represented adequate safety for normal operation (the standard σ_{u_g} used in that simulation was 4.5 ft/sec).

A Cooper-Harper rating of 7-1/2 was assigned to "severe" turbulence for Level 1 (see Table 1). This choice was based on the rationale that according to Table 3 control is momentarily lost in severe turbulence.

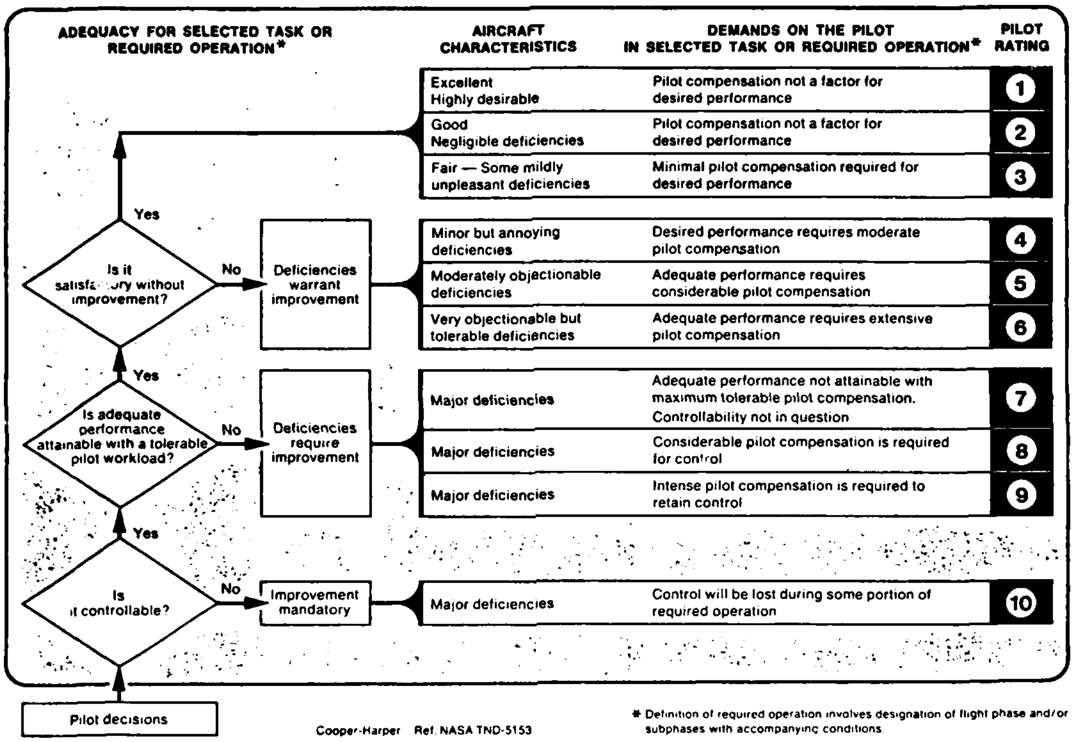


Figure 1 (3.9.1). Handling Qualities Rating Scale
Used to Define Handling Quality Levels

This seems consistent with a pilot rating of 7-1/2 (Figure 1), which is between "controllability not in question" and "considerable pilot compensation required for control." This latter distinction also seems appropriate for Level 2 flying qualities in "moderate" atmospheric dsiturbance (Table 1).

In some cases the expected motions due to turbulence are sufficiently extreme that pilot ratings are not appropriate. In these cases statements relating to survivability are used in both Tables 1 and 2.

Rationale for choosing the magnitudes of wind shear which are included in the definition of Moderate atmospheric disturbance is discussed in Paragraph 3.9.2.

The concept of accounting for degradations in flying qualities in a flying qualities specification was introduced in Reference 163 and discussed at some length in the flying quality workshops held in 1978 and 1979 at the USAF Flight Dynamics Laboratory (References 260 and 261). A primary objection has been that it is impractical to measure turbulence in flight test. An example where such measurements were successfully made is given by Reference 262. Since it is not always possible to make such measurements, we have elected to allow an alternate, more qualitative estimate according to the Table 3 descriptors. The weakness in such an approach lies in the obvious fact that the descriptors in Table 3 describe the aircraft response which in fact depends to a varying extent on the handling qualities that we are trying to determine. However, for all but direct force control augmentors, the descriptors of aircraft path response are a reasonable clue as to the approximate magnitude of atmospheric disturbances. It would seem, therefore, that there are a sufficient number of quantitative descriptors to allow test pilots to define the turbulence environment.

In simulation, the details of the disturbance environment are known exactly. The values of σ_{u_g} corresponding to "light," "medium," and "severe" turbulence are based on qualitative experience as reported in Reference 122. The plot in Figure 2 (taken from Reference 122) shows the relationship between the qualitative definitions and the probability of equaling or exceeding a given level of σ_{u_g}.

E. GUIDANCE FOR APPLICATION

Experience has shown that the atmospheric disturbance environment which we have labeled as "Moderate" is sufficient to force the pilot into the aggressive control activity that generally exposes handling deficiencies when they exist. Hence, it is recommended that the major effort be spent investigating the "Moderate" disturbance level. Similarly, during flight test there is no compelling reason to seek out mountain waves or thunderstorms to comply with the "Severe" and "Extreme" requirements stated in this requirement. Of course, if the mission specifically dictates flight in severe disturbances a significant portion of the time, these conditions should be duly accounted for in the specification compliance.

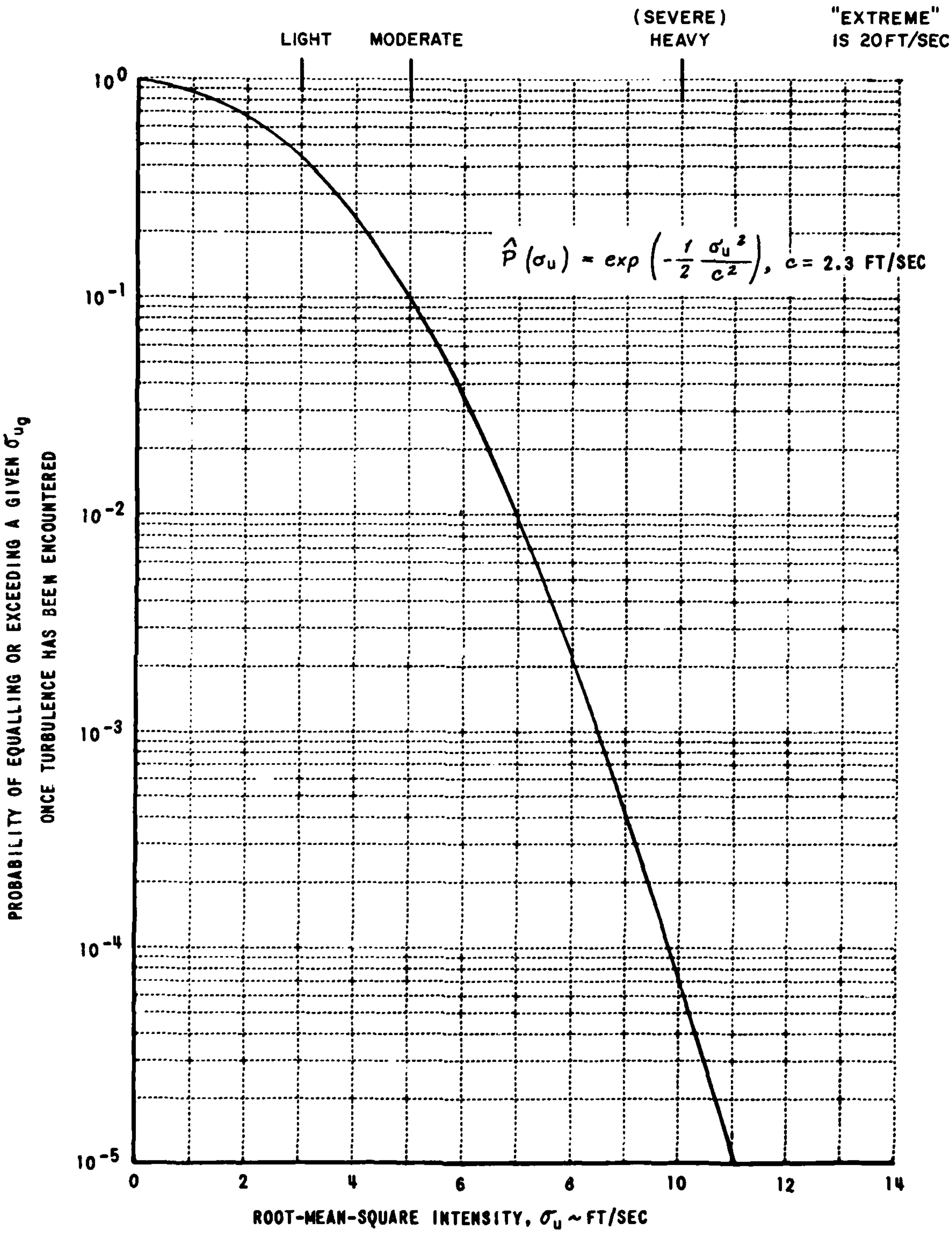


Figure 2 (3.9.1). RMS Intensity vs Exceedance Probability

There may be aerodynamic and flight control system nonlinearities that are affected by very large disturbances. Such effects should be investigated in manned simulation with the severe and extreme magnitudes of atmospheric disturbances specified in Table 4.

G. SUPPORTING DATA

Does not apply.

H. LESSONS LEARNED

No specific examples are felt to be necessary since it is well known that flying qualities are strongly affected by atmospheric disturbances.

3.9.2 Definition of Atmospheric Disturbance Model Form

A. REASON FOR REQUIREMENT

This requirement is included to provide a turbulence model to be utilized when, by agreement with the procuring activity, demonstration of compliance is to be performed via simulation.

B. RELATED MIL-F-8785C REQUIREMENT

3.7.1, 3.7.2, 3.7.3, 3.7.4

C. STATEMENT OF REQUIREMENT

3.9.2 Definition of Atmospheric Disturbance Model Form. When compliance via demonstration is to be carried out using piloted simulation, an atmospheric disturbance model appropriate to the piloting task shall be included. As a minimum, the atmospheric disturbance model shall consist of ________________.

1. Recommended Random Wind Component

The recommended standard random wind component consists of the basic Dryden spectral form for each of the translational and rotary components considered necessary. These spectral forms are:

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

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

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

$$\Phi_{p_g}(\Omega) = \sigma_{p_g}^2 \frac{2L_p}{\pi} \frac{1}{1 + (L_p\Omega)^2}$$

$$\Phi_{q_g}(\Omega) = \frac{\Omega^2}{1 + [(4b/\pi)\Omega]^2} \Phi_{w_g}(\Omega)$$

$$\Phi_{r_g}(\Omega) = \frac{\Omega^2}{1 + [(3b/\pi)\Omega]^2} \Phi_{v_g}(\Omega)$$

where

$$\sigma_x^2 = \int_o^\infty \Phi_x(\Omega)\ d\Omega$$

$$\Omega = \omega/V$$

$$b = \text{wing span}$$

The primary determinant of turbulence intensity is σ_{u_g} and the values to be used for evaluation of flying qualities are given in Table 4 (3.9.1). The relationships between translational intensities and scale lengths are:

$$\frac{\sigma_u^2}{L_u} = \frac{\sigma_v^2}{2L_v} = \frac{\sigma_w^2}{2L_w}$$

For the p-gust the intensity and scale length are associated with the w-gust by:

$$\sigma_{p_g} = \frac{1.9}{\sqrt{L_w b}} \sigma_{w_g}$$

and

$$L_p = \frac{\sqrt{L_w b}}{2.6}$$

Some lengths are set according to altitude by the following relationships:

$$L_u = L_v = L_w = h_1 \qquad \text{for } h > 1750 \text{ ft}$$

$$\left.\begin{aligned} L_u &= L_v = \sqrt[3]{h_1^2 h} \\ L_w &= h \end{aligned}\right\} \qquad \text{for } h_o < h < h_1$$

$$\left.\begin{aligned} L_u &= L_v = \sqrt[3]{h_1^2 h_o} \\ L_w &= h_o \end{aligned}\right\} \qquad \text{for } h < h_o \text{ ft}$$

where h is the center of gravity height above ground, $h_o = 10$ ft, and $h_1 = 1750$ ft.

A summary of the recommended digital filter implementation is given in Table 1.

2. Recommended Wind Shear Component

The recommended standard wind shear is represented by a constant time rate of change of wind speed and direction.

For $t < t_o$,

$$u_g = V_o \cos \psi_o \quad , \quad v_g = V_o \sin \psi_o$$

for $t > t_f$,

$$u_g = V_f \cos \psi_f \quad , \quad v_g = V_f \sin \psi_f$$

and, for the duration of the shear, $t_o < t < t_f$

$$u_g = V_o \cos \psi_o + \frac{t - t_o}{t_f - t_o} \left(V_f \cos \psi_f - V_o \cos \psi_o\right)$$

TABLE 1 (3.9.2). DIGITAL FILTER IMPLEMENTATION

COMPONENT	INTENSITY	BREAK FREQUENCY	DIGITAL FILTER FINITE DIFFERENCE EQUATION
u_g	$\sigma_u = \sigma$	$a_u = \frac{V}{L_u}$	$u_g = (1 - a_u T)u_g + \sqrt{2a_u T}\,\frac{\sigma_u}{\sigma_\eta}\,\eta_1$
v_g	$\sigma_v = \sigma$	$a_v = \frac{2V}{L_v}$	$v_g = (1 - a_v T)v_g + \sqrt{2a_v T}\,\frac{\sigma_v}{\sigma_\eta}\,\eta_2$
w_g	$\sigma_w = \sigma$	$a_w = \frac{2V}{L_w}$	$w_g = (1 - a_w T)w_g + \sqrt{2a_w T}\,\frac{\sigma_w}{\sigma_\eta}\,\eta_3$
$p_g{}^*$	$\sigma_p = \frac{1.9}{\sqrt{L_w b}}\,\sigma_w$	$a_p = \frac{2.6V}{\sqrt{L_w b}}$	$p_g = (1 - a_p T)p_g + \sqrt{2a_p T}\,\frac{\sigma_p}{\sigma_\eta}\,\eta_4$
$q_g{}^{**}$	$\sigma_q = \sqrt{\frac{\pi}{2L_w b}}\,\sigma_w$	$a_q = \frac{\pi V}{4b}$	$q_g = (1 - a_q T)q_g + \frac{\pi}{4b}\left(w_g - w_{g_{PAST}}\right)$
$r_g{}^{\ddagger}$	$\sigma_r = \sqrt{\frac{2\pi}{3L_v b}}\,\sigma_v$	$a_r = \frac{\pi V}{3b}$	$r_g = \left(1 - a_r T\right)r_g + \frac{\pi}{3b}\left(v_g - v_{g_{PAST}}\right)$

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

** q_g is significant only if $\sqrt{\frac{\pi b}{8L_w}}\,C_{m_q} > C_{m_\alpha}$

‡ r_g is significant only if $\sqrt{\frac{\pi b}{6L_v}}\,C_{n_r} > C_{n_\beta}$

and

$$v_g = V_o \sin \psi_o + \frac{t - t_o}{t_f - t_o} \left(V_f \sin \psi_f - V_o \sin \psi_o\right)$$

where

V_o = Initial wind velocity

ψ_o = Initial wind angle

V_f = Final wind velocity

ψ_f = Final wind angle

t_o = Time shear is initiated

t_f = Time shear is terminated

The maximum wind shear magnitude is set according to the incremental flight path change capability of the aircraft up to a limit, i.e.,

$$\frac{V_f - V_o}{t_f - t_o} = g\Delta\gamma < 3.4 \text{ ft/sec}^2$$

The shear duration should be at least 10 sec and the shear should terminate at an altitude of 50 ft for landing simulations. The mean wind at altitude should be set so the wind at touchdown is zero. $\Delta\gamma$ in the up direction is established at maximum climb power and in the down direction at flight idle. Generally speaking the wind shear magnitude should not be required to exceed 3.4 ft/sec^2. At least four critical wind shear cases are considered:

- Decreasing headwind
- Decreasing tailwind
- Decreasing crosswind
- Headwind to tailwind (constant wind speed)

The standard wind shear component is accompanied by one half the standard level of turbulence, i.e., $\sigma_{u_g} = 2.5$ ft/sec.

3. Recommended Mean Wind

The capability for simulating a mean wind from any direction should be included. The wind should be parallel to the earth's surface.

4. Discrete Gust Model

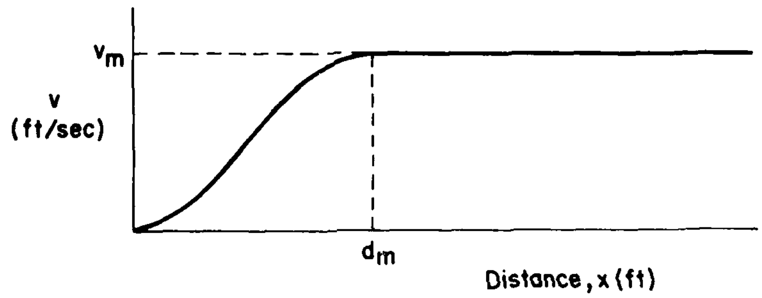
The discrete gust model may be used for any of the three gust-velocity components and, by derivation, any of the three angular components.

The discrete gust has the "1 - cosine" shape given by:

$$v = 0 \quad , \quad x < 0$$

$$v = \frac{v_m}{2}\left(1 - \cos\frac{\pi x}{d_m}\right) \quad , \quad 0 \leq x \leq d_m$$

$$v = v_m \quad , \quad x > d_m$$



The discrete gust above may be used singly or in multiples in order to assess airplane response to, or pilot control of, large disturbances. Step function or linear ramp gusts may also be used.

D. RATIONALE BEHIND REQUIREMENT

The philosophy being applied is based upon two fundamental premises: a) keep the modeling form simple; and b) 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 airplane flying qualities has been approached in a diverse number of 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. Airplane 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.

It is appropriate first to define what is meant by flying qualities, in order to keep the discussion in perspective. One accepted definition is "those airplane characteristics which govern the ease or precision with which the pilot can accomplish the mission." Further, flying qualities are often "measured" by subjective pilot opinion according to the Cooper-Harper rating scale wherein it is stated that flying qualities are tied to accomplishing a specific task. Due consideration of environmental conditions is, in turn, implied. An airplane can have characteristics that make the task of landing relatively easy in calm air. The same task becomes very demanding in strong disturbance, even though the airplane characteristics may not have changed.

For the purposes of the handling qualities Standard, an engineering model of atmospheric disturbances is required. This engineering model may be considered as 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.

The approach taken herein is to define a basic utility model that can be applied to most handling quality evaluations. This model is defined in detail under "Guidance for Application." For some applications, the procuring activity may want to designate a specialized model. For example, if a high-fidelity model is required to reproduce very high-frequency effects, the von Karman model would be appropriate. A table of alternative disturbance models is given in "Guidance for Application."

The basic utility model presented herein represents a major simplification from past flying qualities specifications. It consists of a random turbulence component and a constant gradient wind shear. The random component is modeled using the Dryden form because it is easy to mechanize and is valid over the frequency spectrum of interest for handling qualities evaluations. It is notable that most serious handling quality investigations have utilized the Dryden form. It was decided to recommend that the primary emphasis be placed on a single value of σ_{u_g} on the basis of experience with a large number of piloted simulator handling quality investigations. This experience showed that a σ_{u_g} of 4.5 to 5 ft/sec is large enough to expose handling deficiencies. A review of many handling quality investigations revealed that nearly all of the simulation runs involving atmospheric disturbance were made with a σ_{u_g} in the vicinity of 4.5 to 5 ft/sec.

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 wind shears occur at the critical point on only a few runs, resulting in discrepancies in pilot ratings and comments. A discrete wind shear model is defined to insure that all pilots will experience the critical input. When discrete shears are used, the σ_{u_g} is decreased to 2.5 ft/sec to avoid cases in which a large shear component of the random model adds to the discrete shear to give an unreasonably large total wind shear.

The wind shear model allows for a constant gradient of wind magnitude and direction. A great number of combinations of wind shear can be derived from such a model. However, only four limiting cases are recommended for compliance: decreasing headwind, tailwind, and sidewind shears, as well as a constant magnitude shear with changing azimuth.

Finally, the independent variable in the shear model is time. This is done to insure that the shear is independent of aircraft trajectory, the objective being run-to-run correspondence rather than verisimilitude. This is done with the knowledge that any shear occurring as a function of position or altitude in the real world can be reproduced as a function of time.

E. GUIDANCE FOR APPLICATION

The use of wind shear in handling quality evaluations is primarily to provide a disturbance that forces aggressive closed-loop pilot behavior. It is felt to be unnecessary to simulate vertical shears since the horizontal shear disturbs the aircraft both in airspeed and flight path. For specialized applications, such as the carrier landing burble, a vertical shear should be included.

For most cases, application of this requirement is straightforward. However, for some cases a more complex special-purpose model may be required -- for example, carrier landing.

The remainder of this section is oriented towards providing guidance for selecting alternate disturbance models. The discussion is taken from Reference 184.

1. Evaluating Atmospheric Disturbance Model Needs

Atmospheric modeling needs vary greatly with the specific application -- even for a single given aircraft and flight condition. Some analysis procedures require only a simple one-dimensional turbulence model (e.g., Dryden) and a single gust component. At the other extreme, elaborate simulations can involve a fully defined two-dimensional, non-stationary turbulence field along with a spatially or time-varying mean wind field (i.e., wind shear). It is be the role of the Handling Qualities Handbook to offer guidance in evaluating such needs and selecting appropriate disturbance model options among the variety of modeling choices and identifying the appropriate method of demonstrating compliance.

Some ways of viewing the modeling needs of a user include:

- How disturbance components enter the airframe force and moment equations.
- Inner/outer loop structure hierarchy for mission/aircraft centered features.
- The need for determinism versus randomness in the flying qualities application.

Consider briefly how each of these could be approached.

Table 2 illustrates how various atmospheric disturbance components might enter a set of linearized force and moment equations. Based on our knowledge of the various stability derivatives and respective gust component intensities, we can estimate the relative effect of various gust terms in order to judge:

- Axis cross coupling (e.g., longitudinal and lateral-directional forces and moments are likely to be fairly well decoupled).
- Translational motion (e.g., force equations are mainly affected by gust velocity components alone).
- Rotational motion (e.g., moment equations are affected by gust velocity, time derivative, and gradient components).

The loop structure hierarchy in mission/aircraft-centered features provides us with another way of judging atmospheric disturbance model needs. Figure 1 shows a spectral comparison of mission/aircraft-centered features against atmospheric disturbance features. Although the spectral boundaries of each feature are admittedly more ill-defined

TABLE 2 (3.9.2)

LINEARIZED GUST DERIVATIVE TERMS IN AIRFRAME DYNAMICS

Term / Equation	u_g	w_g	$\dot{u}_g$	$\dot{w}_g$	$(-q_g)$ $\frac{\partial u_g}{\partial x}$	$\frac{\partial w_g}{\partial x}$	v_g	$\dot{v}_g$	$(-r_{1_g})$ $\frac{\partial u_g}{\partial y}$	(r_{2_g}) $\frac{\partial v_g}{\partial x}$	$\frac{\partial v_g}{\partial y}$	(p_g) $\frac{\partial w_g}{\partial y}$
ΣX	$-X_u$	$-X_w$					$-X_v$					
ΣZ	$-Z_u$	$-Z_w$		small $\cancel{-Z_{\dot{w}}}$		small Z_q	small $\cancel{-Z_v}$					
ΣM	$-M_u$	$-M_w$		$-M_{\dot{w}}$		M_q	$-M_v$					
ΣY							$-Y_v$	small $\cancel{-Y_{\dot{v}}}$		small $\cancel{-Y_r}$		
ΣL							$-L_v$		L_r			$-L_p$
ΣN							$-N_v$	$-N_{\dot{v}}$		$-N_r$		$-N_p$

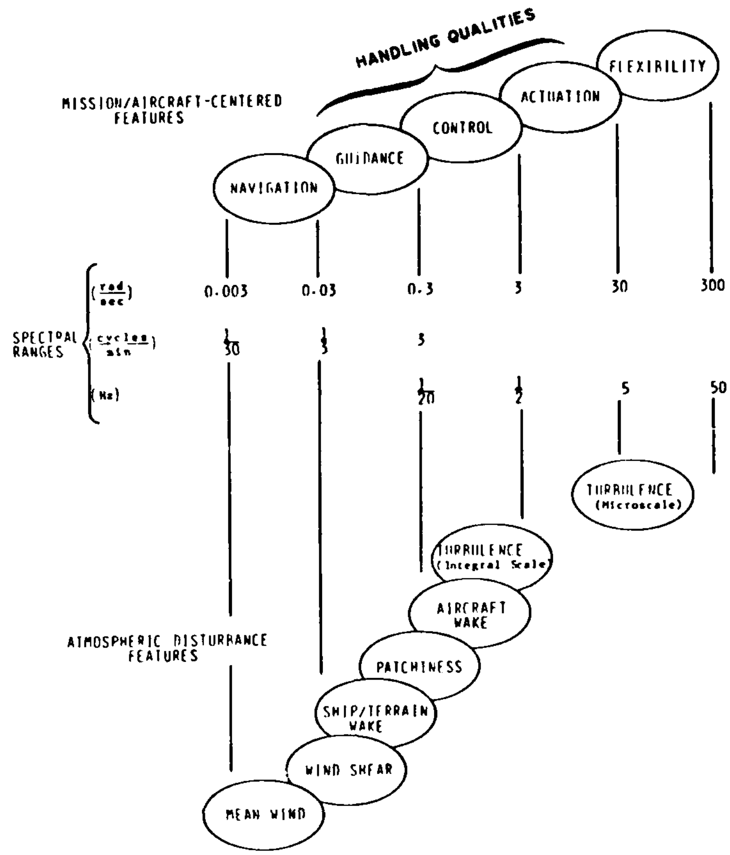


Figure 1 (3.9.2). Comparative Approximate Frequency Regimes of Mission/Aircraft-Centered and Atmospheric Disturbance Features

than shown, we can nevertheless illustrate a point. That is, any mission/aircraft features that are to be analyzed require the significant atmospheric disturbance features acting within the same spectral range. Conversely, atmospheric disturbance features much outside that spectral range are superfluous. Taking the argument to the extreme, navigation considerations are not likely to involve the microscale* or

*The microscale of turbulence is an indication of the distance of time separation over which gusts remain highly correlated, i.e., the inertial subrange (Reference 190). Von Karman turbulence involves a non-zero microscale; Dryden is zero.

even integral* scale range of turbulence. Likewise, short-term stability augmentation system or flexibility effects would not require inclusion of mean wind or wind shear features.

Continuing in a similar vein, the results obtained from exciting an airplane by atmospheric disturbances depend greatly upon how the airplane is being operated, i.e., what the pilot is doing. The gust response can vary dramatically between hands-off operation and tight pilot regulation of attitudes and flight path. Frequently the effects of wind shear are evaluated by measurement of the flight path excursion for a controls-fixed penetration of the shear. The phugoid is, of course, the dominant response mode in this case, and the result is a large-amplitude, undamped, roller-coaster-like flight path oscillation. But pilots do not characteristically operate hands-off in a wind shear environment. Rather, aircraft attitude is likely to be very well regulated by the pilot, hence the flight path and airspeed modes would be exponentially decaying according to heave and speed damping stability derivatives (Z_w and X_u, respectively, Reference 185). These two cases lead to vastly different conclusions regarding performance and identification of critical flying qualities parameters.

We need also to consider how determinism and randomness affect our choice of atmospheric disturbance models. Strict reliance upon a wholly random gust model for a small-sample, short-term task evaluation is both impractical and improper. As investigators and evaluators, we desire to control disturbances well enough so that critical conditions and events can be staged, especially in the case of manned 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. Therefore a compromise must be reached. This is an area that deserves to be addressed in a systematic way, but sometimes solutions must be based more upon experience than clear rationale.

*The integral scale of turbulence is equal to the area under the normalized autocorrelation function and much larger than the microscale. Correct measurement of the integral scale depends upon stationarity (Reference 191).

2. Atmospheric Disturbance Features

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 approximate Dryden form. The same kind of tradeoff between convenience and correctness is a dominant theme in several other respects as we shall discuss below.

a. Determinism Versus Randomness

Atmospheric disturbance models can be separated according to their degree of determinism or randomness. While characteristics such as mean wind and wind shear are normally handled on a deterministic basis, turbulence is usually modeled as a randomly occurring phenomenon. Nevertheless, wind velocity or wind shear can be just as well 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, as we shall discuss shortly.

b. Probability Distribution

The probability distribution of gusts describes their range of amplitudes and frequency of occurrence. These can be quantified in terms of probability density, cumulative probability distribution, or a varying 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.** But the usefulness of these model features depends upon whether the specific application can accommodate a characteristic such as patchiness on a probabilistic basis. If the scenario involves a limited duration time frame and a limited number of sample runs (e.g., as in a manned simulation), then a more deterministic treatment of patchiness might be required. For example, Reference 187 describes a direct modulation of turbulence intensity to obtain a patchy 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 the 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 transformation 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

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*Patchiness is frequently considered as corresponding to a proportionately higher rate of occurrence of very large magnitude gusts than found in a Gaussian distribution and is reflected by the higher-order even central moments (4th, 6th, etc.) (Reference 186).

**Intermittency is the counterpart to patchiness when applied to gust velocity differences over a given time or space interval (Reference 187).

to which the turbulence will affect the aircraft's motion, as described in Reference 188. A time- or spatial-domain correlation function can be useful when generating gusts from multiple point sources (Reference 189).

The two most common ways of describing gust correlation are the Dryden and von Karman power spectral density forms as in MIL-F-8785C. And, 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 of Gust Field

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. 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 a one-dimensional field (Reference 193) 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 (Reference 189). A third dimension can be intoduced in the form of an altitude-dependent wind shear (e.g., References 194 and 195).

*The Taylor hypothesis (Reference 192) assumes a gust field frozen in space such that time and space dependencies along the relative wind are directly related by the airspeed.

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 stationarity 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 (Reference 196). Thus the basic noise source can remain Gaussian.

3. Practical Implementation Considerations

The application of ɛ spheric disturbance models can involve a number of practical implementation problems -- many associated with digital computer programming. One role of this Handbook 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 3 illustrates some of the practical implementation matters.

4. A Survey of Existing Models

The objective in defining an atmospheric disturbance model 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 investigator.

TABLE 3 (3.9.2)

EXAMPLES OF PRACTICAL IMPLEMENTAL MATTERS

Implementation Item	Handbook Method	Comments
Digital implementation of continuous filter forms Example: first-order Dryden form (applicable u_g or p_g)	Spectral form: $\Phi_{uu} = \sigma_{u_g}^2 \frac{2L_u}{\pi} \frac{1}{1 + (L_u\Omega)^2}$ Discrete realization: $u_g = C_1 u_g + C_2 \eta$ where $C_1 = \begin{cases} \text{either } \exp(-aT) & (\text{z-transform}) \\ \text{or } 1-aT & (\text{Euler integration}) \\ \text{or } \frac{2-aT}{2+aT} & (\text{Tustin transform}) \end{cases}$ $a = \frac{V}{L_u}$ and $C_2 = \sqrt{1-C_1^2}\,(\frac{\sigma_{u_g}}{\sigma_\eta})$ where 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-gust importance	Criterion: p-gust is significant relative to v-gust if: $\sqrt{\frac{b}{L}} \cdot \lvert C_{\ell_p}\rvert > \lvert C_{\ell_\beta}\rvert$ or $\frac{2}{\sqrt{L_v b}}\lvert L_p\rvert > \lvert L_v\rvert$ where b is span and L_v is gust scale length	p-gust can be an important disturbance component in the roll axis, especially if effective dihedral is small.
Determination of p-gust intensity	Holley-Bryson model: $\sigma_{p_g} = \frac{2.15\,\sigma_{w_g}}{\sqrt{bL_w\,(1+b/L_w)}}$ MIL-F-8785C: $\sigma_{p_g} = \frac{0.95}{\sqrt[3]{b^2 L_w}}$ An approximate intensity averaged over several models: $\sigma_{p_g} \doteq \frac{1.9}{\sqrt{bL_w}}\,\sigma_{w_g}$	If the p-gust 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-gust model forms all have different ways of expressing model parameters.
Realization of von Karman-like spectra	Boeing higher order linear filter forms: u_g: $\frac{(s+4\frac{V}{L})}{(s+0.84\frac{V}{L})(s+6\frac{V}{L})}$ v_g, w_g: $\frac{(s+0.38\frac{V}{L})(s+7.7\frac{V}{L})}{(s+0.48\frac{V}{L})(s+1.22\frac{V}{L})(s+11.1\frac{V}{L})}$ STI reduced order variation based on Boeing forms: v_g, w_g: $\frac{(s+8\frac{V}{L})}{(s+1.7\frac{V}{L})(s+12\frac{V}{L})}$	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.

Table 4 lists some of the models that have been surveyed and that offer some potential in flying qualities applications. For each table entry a few summary remarks are given along with a list of basic references.

G. SUPPORTING DATA

Previous flying quality specifications have suggested atmospheric disturbance models designed to represent a high degree of fidelity with the real world. As we gain more experience with handling qualities evaluations via piloted simulation, it has become apparent that only a few key features are important for our objectives in order to separate good and bad handling qualities. In fact, the complexity that is required to emulate real-world atmospheric disturbance 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 we have recommended a simplified disturbance model to be used for the majority of cases. 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. Th main feature of the Dryden and von Karman spectral forms plotted in Figure 2 is that both spectral forms concentrate the power in a region near the breakpoint. In this region a slightly better match could be obtained by simply decreasing the Dryden σ_{u_g} by about 8 percent. If, on the other hand, the frequency range of interest were higher, then the two spectral forms would be matched by sliding them laterally, i.e., adjusting the effective scale lengths.

We can conclude from the above that if our spectral range of interest, e.g., the airplane frequency response in the interval between $1/T_{\theta_1}$ and $1/T_{\theta_2}$, is in fact centered about the gust filter break frequency, then there is no major distinction between the two spectral

TABLE 4 (3.9.2)

A SURVEY OF ATMOSPHERIC DISTURBANCE MODELS

Model	Key Features	Sources
Dryden turbulence	A convenient spectral form based on an exponential autocorrelation function for the axial component.	197
von Karman turbulence	A spectral form for which the autocorrelation function includes a finite microscale, thus the relative proportion of spectral power at high frequencies exceeds that of the Dryden.	191, 198, 199
Ornstein-Uhlenbeck turbulence	A spectral form with first-order longitudinal and transverse components.	196
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.	1, 193, 200
Versine gust	A discrete gust waveform.	4
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.	201
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.	189, 202, 203
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.	189
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.	186, 204
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. Includes University of Washington model features extended to approximate transverse turbulence velocities and gradients.	187
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.	205, 206, 207
The Netherlands National Aerospace Laboratory model of non-Gaussian turbulence	Similar to the Royal Aeronautical Estab lishment model but extended to include patchiness and gust gradient components and transverse velocities.	208, 209
University of Virginia turbulence model	Models patchiness by randomizing gust variance and integral scale length of basic Dryden turbulence.	210

TABLE 4 (3.9.2). (Concluded)

Model	Key Features	Sources
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.	196
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.	194, 211
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.	195, 214
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.	213, 214
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.	11
MIL-F-8785C atmospheric disturbance model	Same as 8785B with the addition of a logarithmic planetary boundary layer wind, a vector shear, and a Nave carrier airwake model.	4
ESDU atmospheric turbulence	Rather general but contains comprehensive descriptive data for turbulence intensity, spectra, and probability density	215, 216
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.	217, 218
Wasicko carrier airwake model	Includes mean wind profile, effect of ship motion, and turbulence.	219
Nave ship airwake model	Includes free air turbulence filters plus steady, periodic, and random components of airwake which are functions of time and space.	221
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.	220
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.	200

Notes:

- Area under curves for a given frequency band corresponds to spectral power in that frequency band
- Total area under either curve is .4343 if unit on abscissa is taken as one decade
- Spatial frequency, Ω related to temporal frequency, ω by airspeed, V or $\Omega = \omega/V$
- $\Phi(\Omega)$ is power spectral density
- L_u is scale length

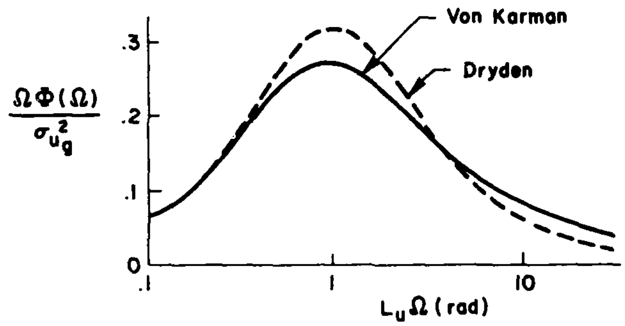


Figure 2 (3.9.2). Comparison of Horizontal Gust Power Spectral Density for Dryden and von Karman Forms Multiplied by Frequency

forms. Clearly, the Dryden form is the more attractive because of its convenience.

The variation of scale length with altitude has been taken directly from MIL-F-8785B. The scale length implemented in MIL-F-8785C for the low-altitude band resulted in a 700 ft altitude band where no scale length was defined. That is, the MIL-F-8785C low-altitude model only goes up to 1000 ft and the mid- to high-altitude model begins at 1750 ft. The choice was between lowering the mid- to high-altitude region to 1000 ft (and lowering L_u from 1750 to 1000 ft) or reverting to the 8785B model. Inasmuch as most researchers are still using the "B version" and the effect of scale length is not great (see Reference 188), it was decided to specify the earlier version (8785B).

Finally, the discrete wind shear magnitude prescribed in the model is based on aircraft performance. The basic rationale is that the handling qualities should be adequate to allow the pilot to operate up to the limit of performance (see, for example, Reference 195). In order to hold constant airspeed in a wind shear the aircraft inertial speed must be changed at the same rate as the changing wind, i.e., $\dot{V}_i = \dot{V}_w$. The maximum acceleration capability of an airplane is not always well known. However, it can be obtained from performance charts by noting that $mg(\gamma_{max} - \gamma_o)$ is the change in weight component along the flight path which corresponds to a thrust increase to the maximum availa[illegible], where γ_{max} is the maximum-power angle of climb at the reference spe[illegible] in the configuration being investigated (for example, gear down and lan[illegible] flaps for power approach); γ_o is the trim flight path angle. Then,

$$\dot{V}_{max} = g(\gamma_{max} - \gamma_o)$$

An upper limit on $\dot{V}_w$ of 3.4 ft/sec ($\Delta\gamma$ = 6 deg) has been specified to avoid requiring excessive shear for high-performance fighters where γ_{max} may be extreme.

H. LESSONS LEARNED

Experience has shown that aircraft with severe handling quality 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. One problem with using a random disturbances model is that the large low-frequency gusts (wind shears) occur in an unpredictable fashion. In the Reference 222 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 shear model is included to resolve this problem.

A random noise source (white noise) is required to mechanize the equations in Table 1. This source should be checked to insure adequate power at low frequencies. Experience with some simulations has shown that the noise source was deficient at low frequency. As mentioned above, it is the low-frequency component of the atmospheric disturbance that is important for handling qualities investigations.

Finally, when using the more complex models it seems nearly impossible to formulate a program 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.

3.9.3 Application of Disturbance Models in Analyses

A. REASON FOR REQUIREMENT

This requirement is included to provide guidance for incorporation of the disturbance models into a simulation program.

B. RELATED MIL-F-8785C REQUIREMENT

3.7.5

C. STATEMENT OF REQUIREMENT

3.9.3 Application of disturbance models in analyses. The gust and turbulence velocities shall be applied to the aircraft equations of motion through the aerodynamic terms only, and the direct effect on the aerodynamic sensors shall be included when such sensors are part of the airplane augmentation system. Application of the disturbance model depends on the range of frequencies of concern in the analyses of the airframe. When structural modes are significant, the exact distribution of turbulence velocities should be considered. For this purpose, it is acceptable to consider u_g and v_g as being one-dimensional, a function of both x and y, for the evaluation of aerodynamic forces and moments.

When structural modes are not significant, 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 airplane center of gravity. The angular velocities due to turbulence are equivalent in effect to airplane angular velocities. Approximations for these angular velocities are defined (precise only at very low frequencies) as follows:

$$-\dot{\alpha}_g = q_g = \frac{\partial w_g}{\partial x} \quad , \quad p_g = -\frac{\partial w_g}{\partial y} \quad , \quad r_g = -\frac{\partial v_g}{\partial x}$$

The spectra of the angular velocity disturbances due to turbulence are given in Paragraph 3.9.2.

For altitudes below 175 ft, the turbulence velocity components u_g, v_g, and w_g are to be taken along axes corresponding to u_g aligned along the relative mean wind vector and w_g vertical.

D. RATIONALE BEHIND REQUIREMENT

This requirement is included to insure proper implementation of the disturbance model. It is included simply as a reminder of a few key points and is not intended to be a comprehensive guide. It is believed that the level of competence of the majority of users is such that further guidance is not necessary.

E. GUIDANCE FOR APPLICATION

Does not apply.

F. DEMONSTRATION OF COMPLIANCE

None required.

G. SUPPORTING DATA

Does not apply.

H. LESSONS LEARNED

None required.

3.9.4 Requirements for Aircraft Failure States in Atmospheric Disturbances

A. REASON FOR REQUIREMENT

This requirement is intended to insure that certain failures when combined with operations in atmospheric disturbances do not result in unacceptable degradations in flying qualities.

B. RELATED MIL-F-8785C REQUIREMENT

3.8.3.2

C. STATEMENT OF REQUIREMENT AND RECOMMENDED VALUES

3.9.4 Requirements for Aircraft Failure States in Atmospheric Disturbances. When Aircraft Failure States exist (3.1.6), a degradation in flying qualities is permitted only if the probability of encountering a lower Level than specified in 3.9.1 is sufficiently small. At intervals established by the procuring activity, the contractor shall determine, based on the most accurate available data, the probability of occurrence of each Aircraft Failure State per flight and the effect of that Failure State on the flying qualities within the *Operational and Service Flight* Envelopes. These determinations shall be based on MIL-STD-756 except that:

a) All aircraft components and systems are assumed to be operating for a time period, per flight, equal to the longest operational mission time to be considered by the contractor in designing the aircraft, and

b) Each specific failure is assumed to be present at whichever point in the Flight Envelope being considered is most critical (in the flying qualities sense).

From these Failure State probabilities and effects, the contractor shall determine the overall probability, per flight, that one or more 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.

Table 1 specifies the recommended requirements as functions of the probability of encountering the degradation in flying qualities. An alternate requirement could be Figure 1 (3.1.6.1) at the discretion of the procuring activity.

TABLE 1 (3.9.4)

LEVELS FOR AIRCRAFT FAILURE STATES

ATMOSPHERIC DISTURBANCES	FAILURE STATE I*	FAILURE STATE II**
Light	Level 2	Level 3
Moderate	Level 2	Recoverable*** or better
Severe to Extreme	Recoverable or better	No requirement

*For flight in the Operational Flight Envelope:
Probability of encountering degraded levels of flying qualities due to failure(s) $< 2.5 \times 10^{-3}$/flight hr

**For flight in the Operational Flight Envelope:
Probability of encountering degraded levels of flying qualities due to failure(s) $< 2.5 \times 10^{-5}$/flight hr
for flight in the Service Flight Envelope:
Probability of encountering degraded levels of flying qualities due to failure(s) $< 2.5 \times 10^{-3}$/flight hr

***Recoverable is defined as: control can be maintained long enough to fly out of a disturbance.

3.9.4 Requirements for Aircraft Failure States in Atmospheric Disturbances [Alternate Requirement]. Failure States shall be evaluated in moderate levels of atmospheric disturbance.

a) A Level 2 aircraft shall not degrade below Level 3 in the presence of failures and moderate atmospheric disturbances.

b) A Level 3 aircraft shall have flying qualities in the presence of failures and moderate atmospheric disturbances such that control can be maintained long enough to fly out of the disturbance.

D. RATIONALE BEHIND REQUIREMENT

The requirements for Light disturances are the same as in Table 1 (3.1.6.1). In Moderate disturbances, very low probability failures ($P < 10^{-4}$) are only required to be recoverable. Under the assumption that Severe to Extreme disturbances are very seldom encountered, a failure occurring with $P < 10^{-2}$ is only required to be recoverable in that environment. The basic reasoning is that we do not want to make unreasonable demands on the manufacturers by placing excessive requirements on very low probability events (i.e., turbulence plus a critical failure).

There were a number of comments during the review cycle of this report that this requirement is based on numbers which may not be substantiated. An alternative requirement was therefore included which does not depend on the calculation of failure probabilities.

E. GUIDANCE FOR APPLICATION

An alternative to Table 1 (3.9.2) is given in Paragraph 3.1.6.1 as Figure 1 (3.1.6.1). The two are consistent if we associate Light, Moderate, and Severe disturbances with probabilities of approximately 10^{-1}, 10^{-3}, and 10^{-5}. The use of Figure 1 (3.1.6.1) allows a more fine-grained interpolation than Table 1 (3.9.2). However, such interpolation is not justified by any hard data.

F. DEMONSTRATION OF COMPLIANCE

Experience with MIL-F-8785B and -8785C has shown that it is sometimes not practical to calculate probabilities of failures. This is accounted for in Para. 3.1.6.2 (Generic failure analysis) and the alternate requirement in this section. Such an approach assumes that a given component or series of components will fail with a probability of one. When this approach is being utilized, the effects of turbulence should be accounted for when making determination of the flying qualities in the failed state. If Level 3 flying qualities are predicted for turbulence levels that are equal to or less than Moderate, a probability analysis should be required to insure that the likelihood of such failures is indeed remote.

G. SUPPORTING DATA

None available.

H. LESSONS LEARNED

None available.

SECTION 4

4. NOTES

4.1 INTENDED USE

This specification contains the handling qualities requirements for piloted aircraft and forms one of the bases for determination by the procuring activity of aircraft acceptability. The specification consists of requirements in terms of criteria for use in stability and control calculations, analysis of wind tunnel test results, simulator evaluations, flight testing, etc. The requirements should be met as far as possible by providing an inherently good basic airframe. Cost, performance, reliability, maintenance, etc. tradeoffs are necessary in determining the proper balance between basic airframe characteristics and augmented dynamic response characteristics. The contractor should advise the procuring activity of any significant design penalties which may result from meeting any particular requirement.

4.2 DEFINITIONS

Terms and symbols used throughout this specification are defined as follows.

4.2.1 General

S	Wing area
s	Laplace operator
q	Dynamic pressure
MSL	Mean Sea Level
T_2	Time to double amplitude; $T_2 = -.693/\zeta\omega$ for oscillations, $T_2 = .693T$ for first-order divergences
τ	Time delay
Aircraft Normal States	The nomenclature and format of Table 1 (1.6.1) shall be used in defining the Aircraft Normal States
Service ceiling	Altitude at a given airspeed at which the rate of climb is 100 ft/min at stated weight and engine thrust
Combat ceiling	Altitude at a given airspeed at which rate of climb is 500 ft/min at stated weight and engine thrust
Cruising ceiling	Altitude at a given airspeed at which rate of climb is 300 ft/min at NRT at stated weight
h_{max}	Maximum service altitude (defined in 1.5.2)
$h_{o_{max}}$	Maximum operational altitude (1.5.1)
$h_{o_{min}}$	Minimum operational altitude (1.5.1)
c.g.	Airplane center of gravity

4.2.2 Speeds

Equivalent airspeed, EAS	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
Calibrated airspeed, CAS	Airspeed-indicator reading corrected for position and instrument error but not for compressibility
Refusal speed	The maximum speed to which the aircraft can accelerate and then stop in the available runway length
M	Mach number
V	Airspeed along the flight path (where appropriate, V may be replaced by M in this specification)
V_S	Stall speed (equivalent airspeed), at 1g normal to the flight path, defined as the highest of:

a. Speed for steady straight flight at $C_{L_{max}}$, the first local maximum of the curve of lift coefficient (L/qS) vs. angle of attack which occurs as C_L is increased from zero

b. speed at which uncommanded pitching, rolling or yawing occurs (3.8.4.2)

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

Conditions for determining V_S.

The aircraft shall be initially trimmed at approximately 1.2 V_S with the following settings, after which the trim and throttle settings shall be held constant:

FLIGHT PHASE	THRUST SETTINGS*	TRIM SETTING
Climb (CL)	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
Wave-off/Go-around (WO)	Takeoff	For normal approach
Landing (L)	Idle	For normal approach
All other	TLF at 1.2 V_S	For straight flight

*Either on all engines or on remaining engines with critical engine inoperative, whichever yields the higher value of V_S.

In flight test, it is necessary to reduce speed very slowly (typically 1/2 knot per second or less) to minimize dynamic lift effects. The load factor will generally not be exactly 1g 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.

$V_S(X)$, $V_{min}(X)$, $V_{max}(X)$	short-hand notation for the speeds V_S, V_{min}, V_{max} for a given configuration, weight, center-of-gravity position, and external store combination associated with Flight Phase X. For example, the designation $V_{max}(TO)$ is used in 3.2.9.7.1 to emphasize that the speed intended (for the weight, center of gravity, and external store combination under consideration) is V_{max} for the configuration associated with the takeoff Flight Phase. This is necessary to avoid confusion, since the configuration and Flight Phase change from takeoff to climb during the maneuver.
V_{trim}	Trim speed
V_{end}	Speed for maximum endurance
$V_{L/D}$	Speed for maximum lift-to-drag ratio
$V_{R/C}$	Speed for maximum rate of climb
V_{range}	Speed for maximum range in zero wind conditions
V_{NRT}	High speed, level flight, normal rated thrust
V_{MRT}	High speed, level flight, military rated thrust
V_{MAT}	High speed, level flight, maximum augmented thrust
V_{max}	Maximum service speed (defined in 1.5.2)
V_{min}	Minimum service speed (defined in 1.5.2)
$V_{o_{max}}$	Maximum operational speed (1.5.1)
$V_{o_{min}}$	Minimum operational speed (1.5.1)
V_G	Gust penetration speed
V_{MC_A}	Minimum controllable airspeed (V_{MC})
V_{MC_G}	Minimum controllable ground speed

4.2.3 <u>Thrust and Power</u>

Thrust and power	For propeller-driven aircraft, the word "thrust" shall be replaced by the word "power" throughout the specification
TLF	Thrust for level flight
NRT	Normal rated thrust, which is the maximum thrust at which the engine can be operated continuously
MRT	Military rated thrust, which is the maximum thrust at which the engine can be operated for a specified period
MAT	Maximum augmented thrust: maximum thrust, augmented by all means available for the Flight Phase
Takeoff thrust	Maximum thrust available for takeoff

4.2.4 Control Parameters

Pitch, roll, yaw controls	The stick or wheel and pedals manipulated by the pilot to produce pitching, rolling and yawing moments respectively; the cockpit controls
Pitch control force, F_s	Component of applied force, exerted by the pilot the cockpit control, in or parallel to the plane of symmetry, acting at the center of the stick grip or wheel in a direction perpendicular to a line between the center of the stick grip and the stick or control column pivot
Roll control force, F_{as}	For a stick control, the component of control force exerted by the pilot in a plane perpendicular to the plane of symmetry, 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 control, 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
Yaw-control pedal force, F_{rp}	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 perpendicular to the pedals at the normal point of application of the pilot's instep on the respective yaw-control pedals
Direct normal force control	A device producing direct normal force for the primary purpose of controlling the flight path of the aircraft. Direct normal force control is the descriptive title given to the concept of directly modulating the normal force on an airplane by changing its lifting capabilities at a constant angle of attack and constant airspeed or by controlling the normal force component of such items as jet exhausts, propellers, and fans
Control power	Effectiveness of control surfaces in applying forces or moments to an aircraft. For example, 50 percent of available roll control power is 50 percent of the maximum rolling moment that is available to the pilot with allowable roll control force

4.2.5 Longitudinal Parameters

$1/T_{\theta_1}$	Low-frequency pitch attitude zero
$1/T_{\theta_2}$	High-frequency pitch attitude zero
ζ_{SP}	Damping ratio of the short-period oscillation
ω_{SP}	Undamped natural frequency of the short-period oscillation
ζ_p	Damping ratio of the phugoid oscillation
ω_p	Undamped natural frequency of the phugoid oscillation
ω_{BW}	Bandwidth frequency (3.2.1.2)
n	Normal acceleration or normal load factor, measured at the c.g.
n_z'	Normal acceleration measured at the instantaneous center of rotation for pitch control inputs
n_L	Symmetrical flight limit load factor for a given Aircraft Normal State, based on structural considerations
n_{max}, n_{min}	Maximum and minimum service load factors
$n(+)$, $n(-)$	For a given altitude, the upper and lower boundaries of n in the V-n diagrams depicting the Service Flight Envelope (1.5)
$n_{o_{max}}$, $n_{o_{min}}$	Maximum and minimum operational load factors
$n_o(+)$, $n_o(-)$	For given altitude, the upper and lower boundaries of n in the V-n diagrams depicting the Operational Flight Envelope (1.5)
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 airplane center of gravity

α_S	The stall angle of attack at constant speed for the configuration, weight, center of gravity position and external-store combination associated with a given Aircraft Normal State; defined as the lowest of the following: 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 (3.8.4.2) c. Angle of attack, for a given speed or Mach number, at which intolerable buffeting is encountered.
$C_{L_{stall}}$	Lift coefficient at α_S defined above.
n/α	The 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)
F_s/n	Gradient of steady-state pitch control force versus n at constant speed (3.2.9.1)
γ	Climb angle, positive for climbing flight $\gamma = \sin^{-1}$ (vertical speed/true airspeed)
θ	Pitch attitude, the angle between the x-axis and the horizontal
L	Aerodynamic lift plus thrust component normal to the flight path

4.2.6 Lateral-Directional Parameters

δ_{as}	Displacement of the roll control stick or wheel along its path
δ_{rp}	Displacement of the yaw control pedal
T_R	First-order spiral mode time constant, positive for stable mode
T_s	First-order roll mode time constant, positive for stable mode
ω_ϕ	Undamped natural frequency of numerator quadratic of ϕ/F_{as} transfer function
ζ_ϕ	Damping ratio of numerator quadratic of ϕ/δ_{as} transfer function
ω_d	Undamped natural frequency of the dutch roll oscillation
ζ_d	Damping ratio of the dutch roll oscillation
T_d	Damped period of the dutch roll, $T_d = \dfrac{2\pi}{\omega_d\sqrt{1 - \zeta_d^2}}$
ω_{RS}	Undamped natural frequency of a coupled roll-spiral oscillation
ζ_{RS}	Damping ratio of a coupled roll-spiral oscillation
ϕ	Bank angle measured in the y-z plane, between the y-axis and the horizontal
ϕ_t	Bank angle change in time t, in response to control deflection of the form given in 3.5.9
p	Roll rate about the x-axis
$\delta'_{rp}(3)$	Yaw-to-roll crossfeed parameter (3.6.2.1.2)

p_{osc}/p_{av}	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:

$$\zeta_d < 0.2: \quad \frac{p_{osc}}{p_{av}} = \frac{p_1 + p_3 - 2p_2}{p_1 + p_3 + 2p_2}$$

$$\zeta_d > 0.2: \quad \frac{p_{osc}}{p_{av}} = \frac{p_1 - p_2}{p_1 + p_2}$$

where p_1, p_2, and p_3 are roll rates at the first, second and third peaks, respectively. (Figures 1 and 2).

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 aileron control command:

$$\zeta_d < 0.2: \quad \frac{\phi_{osc}}{\phi_{av}} = \frac{\phi_1 + \phi_3 - 2\phi_2}{\phi_1 + \phi_3 + 2\phi_2}$$

$$\zeta_d > 0.2: \quad \frac{\phi_{osc}}{\phi_{av}} = \frac{\phi_1 - \phi_2}{\phi_1 + \phi_2}$$

where ϕ_1, ϕ_2, ϕ_3 are bank angles at the first, second and third peaks, respectively.

β	Sideslip angle at the center of gravity, angle between undisturbed flow and plane of symmetry; positive, or right sideslip corresponds to incident flow approaching from the right side of the plane of symmetry
$\Delta\beta$	Maximum change in sideslip occurring within 2 seconds or one half-period of the dutch roll, whichever is greater, for a step roll-control command (Figures 1 and 2).
k	Ratio of "command roll performance" to "applicable roll performance requirement" of 3.5.9.1, where:

a. "Applicable roll performance requirement", $(\phi_t)_{requirement}$, is determined from 3.5.9.1 for the Class and Flight Phase Category under consideration

b. "Commanded roll performance", $(\phi_t)_{command}$, is the bank angle attained in the stated time for a given step roll command with yaw control pedals employed as specified in 3.5.9.1

$$k = \frac{(\phi_t)_{command}}{(\phi_t)_{requirement}}$$

t_{n_β} — Time for the dutch roll oscillation in the sideslip response to reach the nth local maximum for a right step or pulse roll-control command, or the nth local minimum for a left command. In the event a step control input cannot be accomplished, the control shall 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

ψ_β — Phase angle expressed as a lag for a cosine representation of the dutch roll oscillation in sideslip, where

$$\psi_\beta = -\frac{360}{T_d} t_{n_\beta} + (n - 1)\ 360 \text{ (degrees)}$$

with n as in t_n above

p/β — 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 180 deg

$|\phi/\beta|_d$ — At any instant, the ratio of amplitudes of the bank-angle and sideslip-angle envelopes in the dutch roll mode

Examples showing measurement of roll-sideslip coupling parameters are shown on Figure 1 for right rolls and Figure 2 for left rolls. Since several oscillations of the dutch roll are required to measure these parameters, and since for proper identification large roll rates and bank angle 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

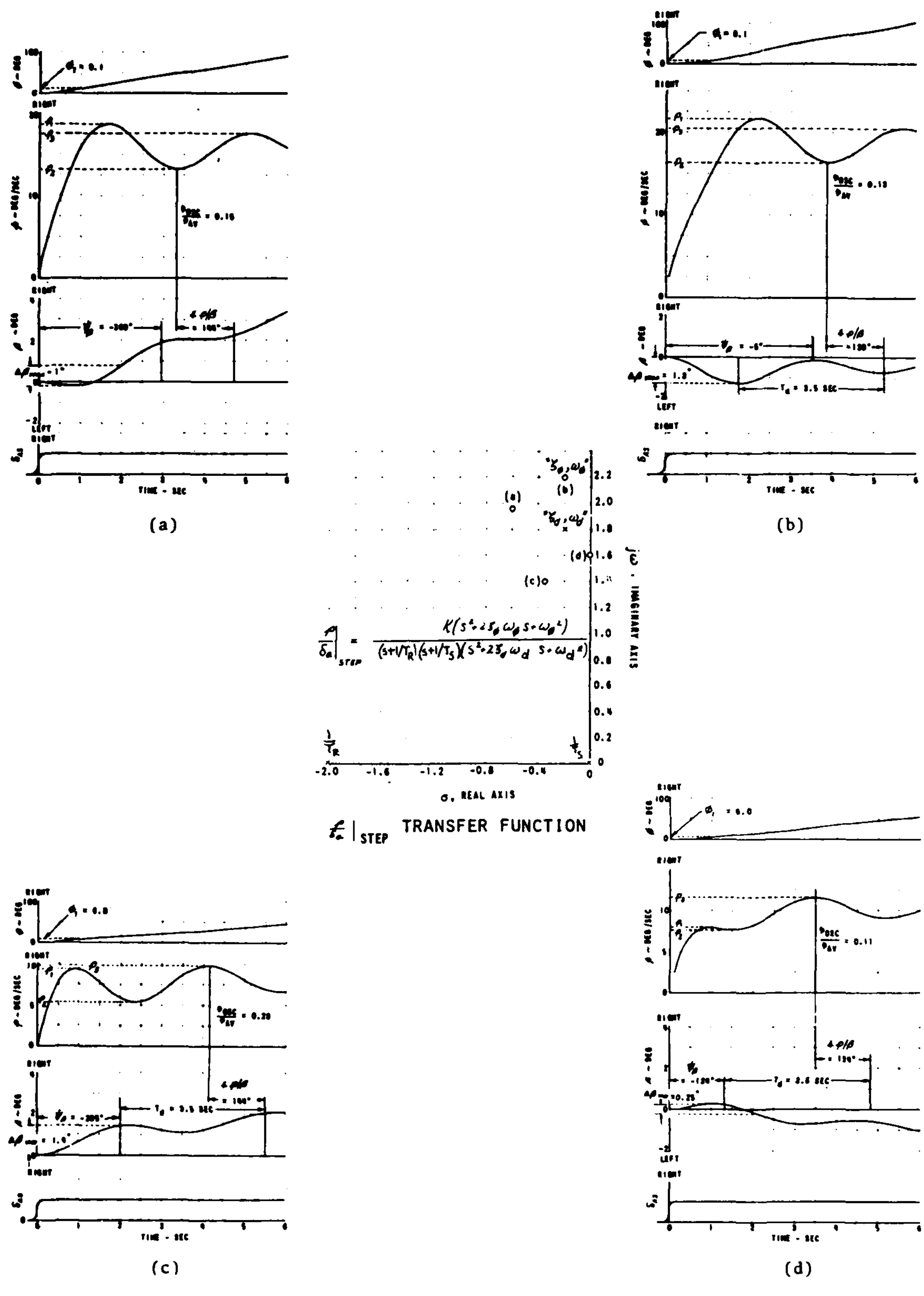


Figure 1 (4.2). Roll-Sideslip Coupling Parameters-Right Rolls

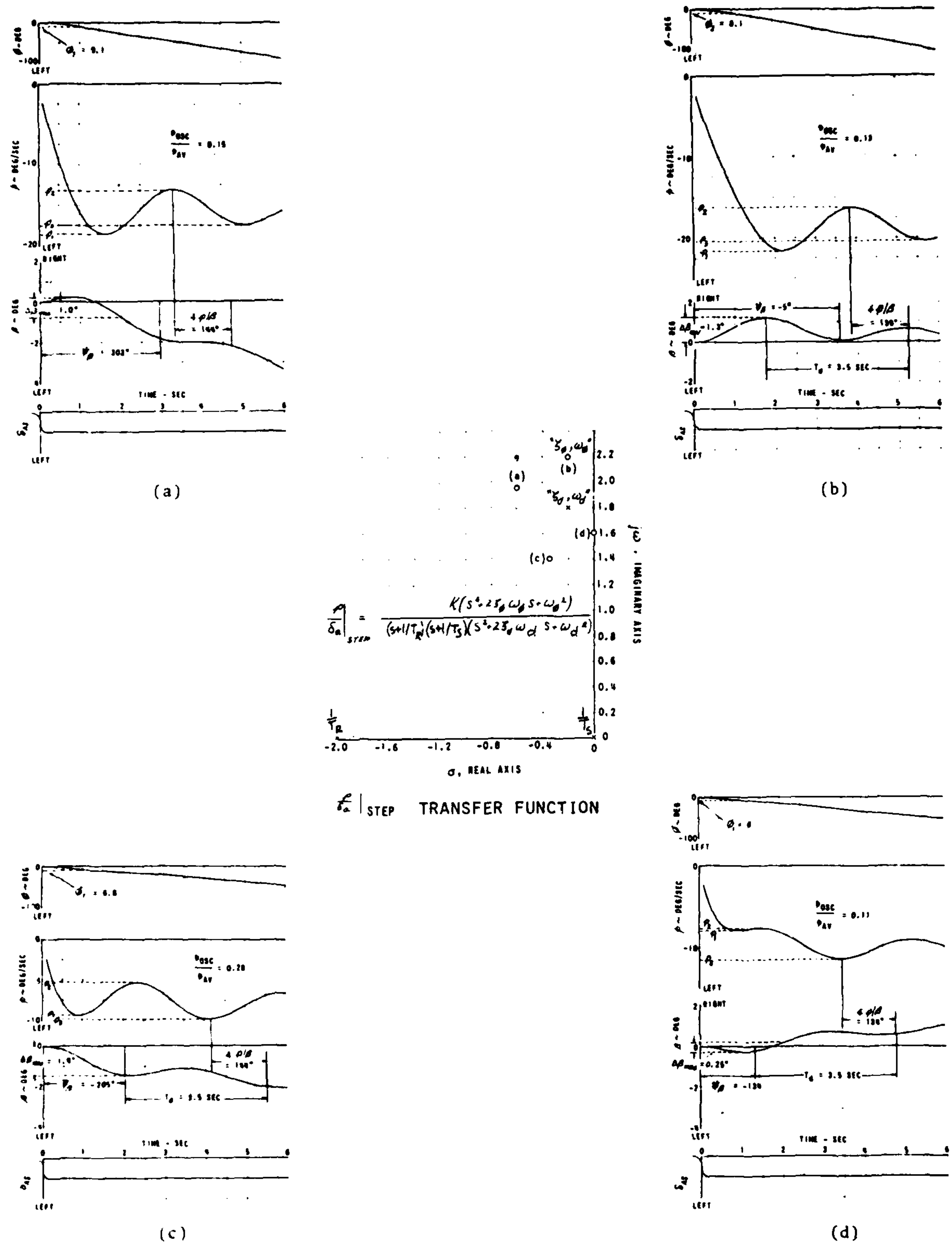
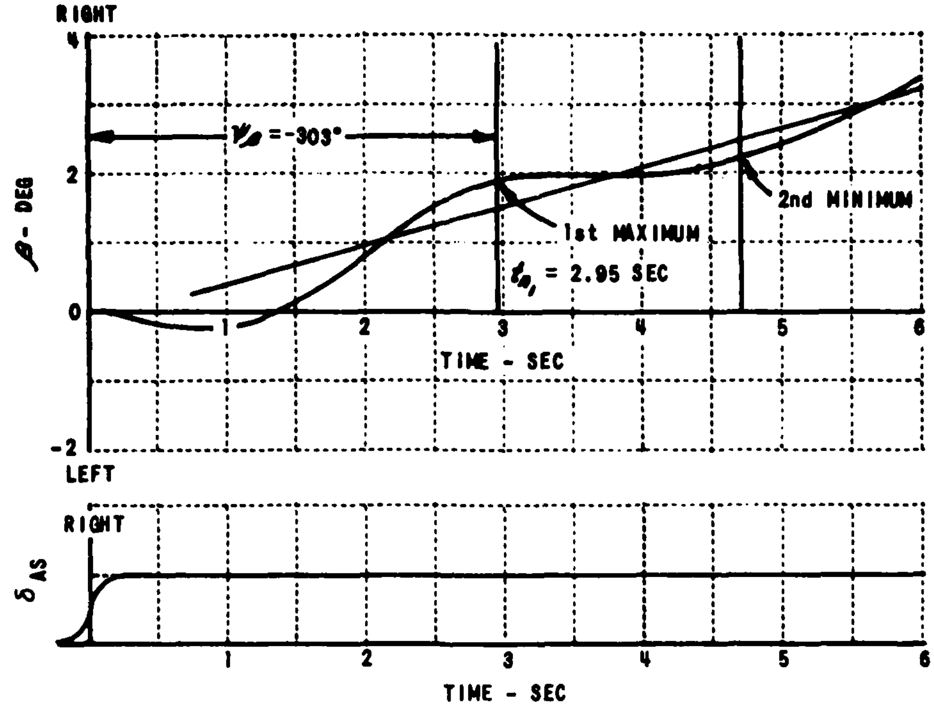


Figure 2 (4.2). Roll-Sideslip Coupling Parameters-Left Rolls select a peak far enough downstream that the position of the peak is not influenced by the roll mode. In practice, peaks occurring one or two roll mode time constants after the aileron input will be relatively undistorted. Care must also be 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 sketch on following page for Case (a) of Figures 1 and 2).



Since the first local maximum of the dutch roll component of the sideslip response occurs at t = 2.95 seconds,

$$\psi_\beta = \frac{-360}{T_d} t_{n_\beta} + (n - 1)\, 360 = \frac{-360}{3.5}(2.95) = -303 \text{ degrees}$$

Level 1 flying qualities of a Class IV aircraft in the approach are under examination; so the roll performance requirement from Table 1 (3.5.9.1) upon which the parameter "k" in the sideslip excursion requirement [Figure 1 (3.6.2.1.1)] is based, is ϕ_t = 30 degrees in 1 second with rudder pedals free (as in the rolls of 3.5.9.1). From the definitions, "k" for this condition is,

$$k = \frac{(\phi_1)_{command}}{(\phi_1)_{requirement}}$$

Therefore from Figures 1 and 2:

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

4.2.7 <u>Atmospheric Disturbances Parameters</u>

Ω	Spatial (reduced) frequency (radians per foot)
ω	Temporal frequency (radians per second), where $\omega = \Omega V$
t	Time (seconds)
u_g	Translational disturbance velocity along the x-axis, positive forward (feet per second)
v_g	Translational disturbance velocity along the y-axis, positive to pilot's right (feet per second)
w_g	Translational disturbance velocity along the z-axis, positive down (feet per second)
	Note: Random u_g, v_g, w_g have Gaussian (normal) distributions
σ, RMS	Root-mean-square disturbance intensity, where $\sigma^2 = \int_0^\infty \Phi(\Omega)d\Omega = \int_0^\infty \phi(\omega)d\omega$
σ_u	Root-mean-square intensity of u_g
σ_v	Root-mean-square intensity of v_g
σ_w	Root-mean-square intensity of w_g
L_u	Scale for u_g (feet)
L_v	Scale for v_g (feet)
L_w	Scale for w_g (feet)
$\Phi_{u_g}(\Omega)$	Spectrum for u_g, where $\Phi_{u_g}(\Omega) = V\phi_{u_g}(\omega)$
$\Phi_{v_g}(\Omega)$	Spectrum for v_g, where $\Phi_{v_g}(\Omega) = V\phi_{v_g}(\omega)$
$\Phi_{w_g}(\Omega)$	Spectrum for w_g, where $\Phi_{w_g}(\Omega) = V\phi_{w_g}(\omega)$
v_m	Generalized discrete gust intensity, positive along the positive axes, m = x, y, z (feet per second)
p_g	Rotary disturbance velocity about the x-axis
q_g	Rotary disturbance velocity about the y-axis

r_g	Rotary disturbance velocity about the z-axis
Φ_{p_g}	Spectrum for p_g
Φ_{q_g}	Spectrum for q_g
Φ_{r_g}	Spectrum for r_g

4.2.8 Terms Used in High Angle of Attack Requirements

Post-stall	The flight regime involving angles of attack greater than nominal stall angles of attack. The aircraft characteristics in the post-stall regime may consist of three more or less distinct consecutive types of aircraft motion following departure from controlled flight: post-stall gyration, incipient spin, and developed spin.
Post-stall gyration (PSG)	Uncontrolled motions about one or more airplane axis following departure from controlled flight. While this type of aircraft motion involves angles of attack higher than stall angle, lower angles may be encountered intermittently in the course of the motion.
Spin	That part of the post-stall aircraft motion which is characterized by a sustained yaw rotation. The spin may be erect 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.

4.3 GAIN SCHEDULING

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 should not be scheduled as a function of trim control settings since pilots do not always keep airplanes in trim.

4.4 ENGINE CONSIDERATIONS

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 (3.8.4); and the reduction at low rpm of engine-derived power for operating the flight control system.

4.5 EFFECTS OF AEROELASTICITY, CONTROL EQUIPMENT AND STRUCTURAL DYNAMICS

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

APPENDIX A

DETERMINING EQUIVALENT SYSTEMS

Many flight control mechanizations are complex, and their mathematical models are of high order. All the requirements for modal parameters require matching the high-order response with a low-order equivalent. The modal requirements then apply for all realizable input magnitudes at all operating points within the appropriate flight envelope. There are many procedures in the literature for extracting reduced-order realizations of dynamic models. These methods have high-order dynamic models as their input, and low-order equivalent models as their output. For uniformity this appendix defines the essential components of the procedure. However, the particular method used is left to the choice of the individual contractor. Methods which have been used by various investigators include:

- Matching frequency responses of high-order, linearized transfer functions
- Matching frequency responses extracted from flight time histories using a fast Fourier algorithm
- Matching frequency responses generated by stick cycling in flight
- Using a maximum likelihood technique to match flight time history data

The method shall adjust all the parameters in the equivalent system, with the exception of certain numerator parameters in single-response matching. This exception shall apply only if approved by the procuring activity. The method shall produce the minimum possible value of the weighted sum of the squares of the frequency response differences in magnitude and phase angle between the equivalent low-order system (LOS) and the input high-order system (HOS) at n discrete frequencies, i.e.:

$$\frac{20}{n} \sum_{\omega_1}^{\omega_n} [(\text{gain}_{HOS} - \text{gain}_{LOS})^2 + 0.02(\text{phase}_{HOS} - \text{phase}_{LOS})^2]$$

where

gain is in dB

phase is in degrees

ω denotes the input frequency

n is the number of discrete frequencies

When analytical data are used, the frequencies (at least ten per decade) shall be equi-spaced on the log scale. When experimental data are used, the low-order system shall be matched as closely as possible to the data at the experimental frequencies.

The weighting between gain and phase is such that 1 dB of gain mismatch and 7 degrees of phase mismatch have equal significance in the total mismatch function. In practice the results will be very insensitive to this weighting. For example, a value of 0.01745 (which appears to be a degrees-to-radians correction factor but actually is not) has been reported in the literature. This will not produce materially different results than 0.02. Experience with one matching program is reproduced in the following excerpt from a McDonnell paper in Journal R.Ae.S., February 1976:

> The nomimal 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 (radians) 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	ω_{sp}	ζ_{sp}	L_α	K_q
$(0.01745)^2$	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 mismatch function value to be compared with similarly defined mismatches in the literature.

When different responses are matched simultaneously (for example, roll rate to stick force and sideslip to rudder), each response shall have equal significance. However, note that the minimum value of the total mismatch function will usually occur with numerically unequal gain and phase mismatches and unequal mismatches for different responses.

When the modal parameters are common to the two responses, they shall be constrained to be identical. For example, the dutch roll mode shall have the same damping and frequency in the roll and sideslip responses. This requirement may be waived by the procuring activity for vehicles with flight control systems which utilize more than the conventional number of independent force and moment producers.

In the main body of the Handbook, mismatch envelopes are shown as a guide to determining whether a mismatch is allowable. The envelopes are defined as functions of the Laplace variable, s, as follows:

Upper Gain Envelope;

$$\frac{3.16s^2 + 31.61s + 22.79}{s^2 + 27.14s + 1.84}$$

Lower Gain Envelope;

$$\frac{0.095s^2 + 9.92s + 2.15}{s^2 + 11.6s + 4.95}$$

Upper Phase Envelope;

$$\frac{68.89s^2 + 1100.12s - 275.22}{s^2 + 39.94s + 9.99} e^{0.006s}$$

Lower Phase Envelope;

$$\frac{475.32s^2 + 184100s + 29460}{s^2 + 11.66s + 0.039} e^{-0.0072s}$$

These envelopes are to be used only after the matching process has been performed. Normally, mismatches will be far smaller than those the envelopes allow.

Components of Computer Program

The basic components of an equivalent systems computer program which is currently used by the U.S. Government are shown in the following simplified flow chart (Figure 1). The broken lines enclose three separate subroutines which are briefly described below.

Input — The Input section establishes the high-order response and the initial guesses for its low-order system. It accounts for elements that are held constant (e.g., the short-period pitch numerator root, $1/T_{\theta_2}$). If two systems are matched simultaneously, the objective vector would consist of two frequency responses and the search vector of two sets of transfer function coefficients. In addition the Input section also sets the frequency range, number of frequencies, and number of iterations.

Search — The Search section manipulates the search vector to make its frequency response approximate the objective vector. It is made up of four subsections: a search algorithm containing a minimization strategy, a cost function, a frequency response calculator, and a set of convergence criteria. The search algorithm is a general-purpose, multivariable optimization routine which will attempt to minimize any cost function by varying a search vector. A modified Rosenbrock search routine is used in the example program although a wide variety of possible methods exists. Figure 2 is a flow chart of the Rosenbrock routine used. A more detailed description is in Optimization — Theory and Practice by G.S.G. Beveridge and R. S. Schechter. The cost or mismatch function, described previously in this appendix, is a scalar sum of the squares of gain and phase differences between the low- and high-order frequency responses. The cost function subsection requires the frequency response of the current low-order system in order to calculate the mismatch. The convergence criteria determine whether an optimum match has been found. In the example program, convergence is considered

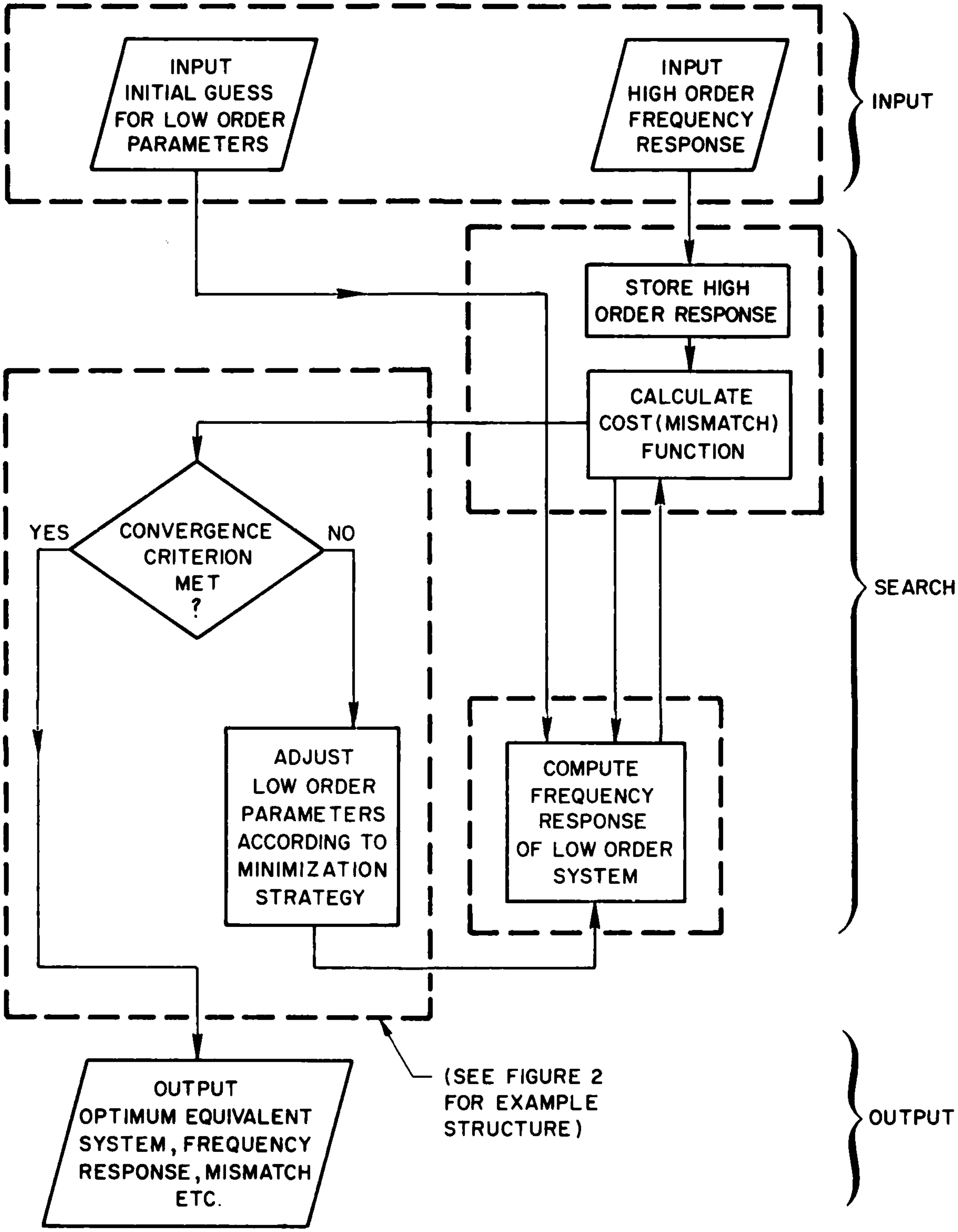


Figure 1 (Appendix A). Simplified Flow Chart for Equivalent System Computer Program

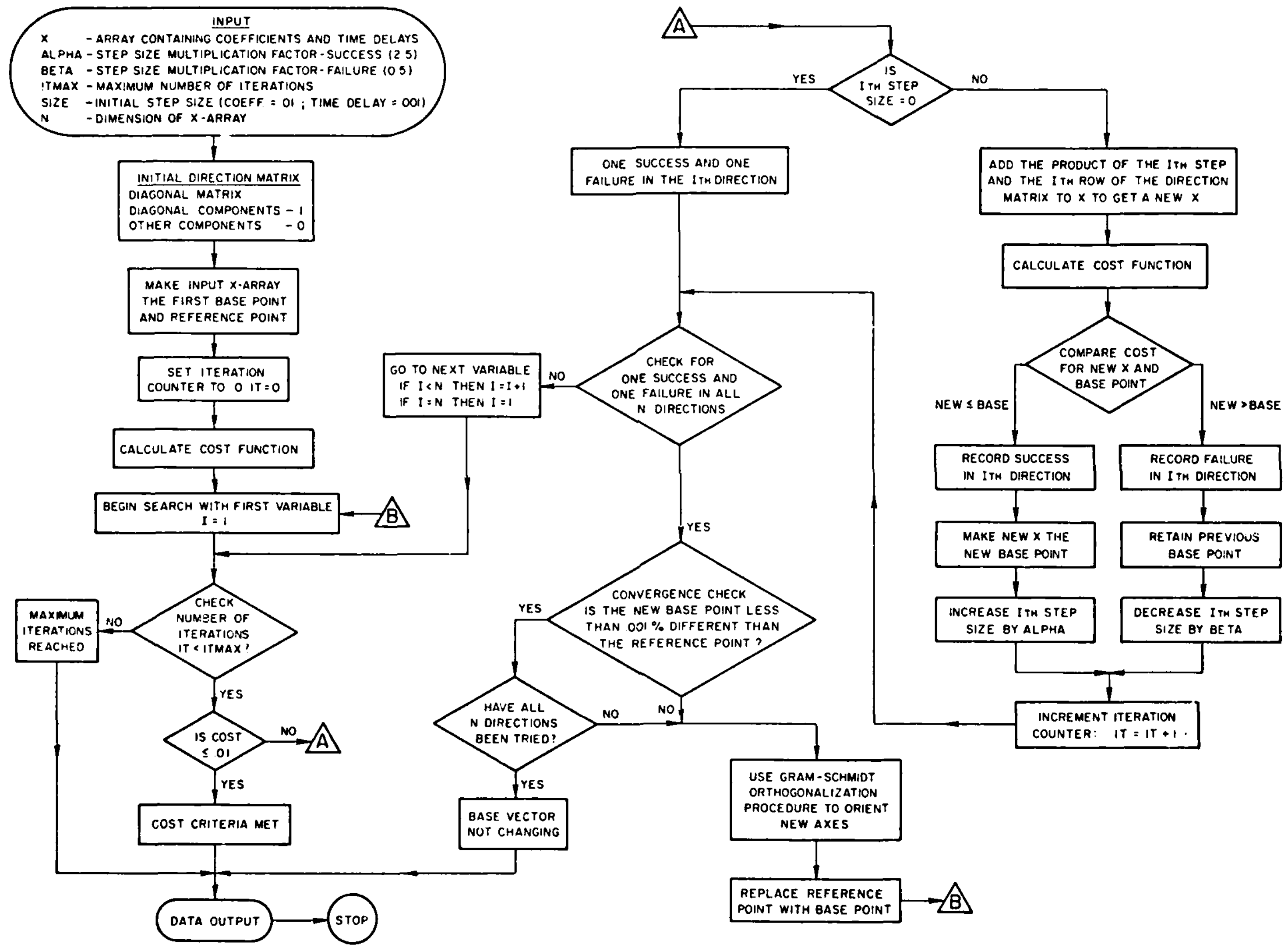


Figure 2 (Appendix A). Flow Chart for a Modified Rosenbrock Search Algorithm

optimum when the search vector changes by less than 0.001 percent between iterations.

<u>Output</u> — This section presents the results of the Search section to the user. The final optimum low-order system, the mismatch, and frequency responses of the high- and low-order systems are primary outputs.

The preceding example was intended to show how equivalent systems can be calculated, not how they must be. Although the example is based on an actual working program, the number of possible, equally good programs is limited only by the number and creativity of prospective users.

APPENDIX B

AN EXAMPLE MIL STANDARD

INTRODUCTION

It is recognized that the "cookbook" format of the MIL Standard represents a significant change from the generalized layout of MIL-F-8785C. In addition, users of the Standard and Handbook are confronted with a reorganization of the requirements into an axis-by-axis breakdown. Familiarity with both of these changes will come with time. In order to assist the user in interpreting and applying the MIL Standard, an example Standard has been assembled. In most cases, fill-ins for the blanks in the Standard were straightforward. Where additional guidance is needed, or where amplification on application of requirements is necessary, we have added discussions, set off in brackets ([]). Where alternative requirements exist, the MIL-F-8785C requirement has been used to clarify the connection between the old and new formats.

Some requirements from the Standard are not applicable to the example aircraft specified and are not included here. Inasmuch as the Standard sections are numbered to be consistent with the Handbook to facilitate cross-referencing the numbering is not consecutive. All tables and figures included herein are numbered sequentially to avoid any potential confusion (e.g., there is only one Table 1 in this appendix). Corresponding tables and figures can be found in the Handbook by consulting the sections in which they occur in this appendix.

The aircraft specified in this generic Standard is intended to be a single-seat lightweight (20,000-lb) highly maneuverable supersonic fighter/attack (Class IV) aircraft designed to operate from tactical airfields. It would be considered to represent an F-5- or F-16-type aircraft, though no such comparison is intended. This class of aircraft was chosen simply as an illustrative example, and should not be construed to reflect on the applicability of the Standard and Handbook for any actual aircraft, or aircraft of any other Class.

1. SCOPE AND OPERATIONAL OBJECTIVES

1.1 Scope. This specification contains the requirements for the flying and ground handling qualities of a U.S. military aircraft. It is intended to assure flying qualities for adequate mission performance and flight safety regardless of the design implementation or flight control system augmentation.

1.2 Application. The flying qualities of the aircraft proposed or contracted for shall be in accordance with this specification. The requirements are written in terms of the axis of vehicle motion and include all aspects of control for that axis, as well as vehicle responses to other inputs, e.g., turbulence, store release, etc. This approach therefore includes requirements for other (i.e., secondary) methods of control for a given axis (DLC, speed brakes, etc.). The requirements apply, as stated, to the combination of airframe and related subsystems. This includes stability augmentation and flight control systems (automatic and/or manual), when provided.

1.3 Aircraft Classification and Operational Missions. For the purpose of this Standard, the aircraft specified in this requirement is to accomplish the following missions: [In a complete procurement Standard, the full range of expected mission profiles would be summarized here. For illustrative purposes we shall choose a single mission which will exercise all elements of the Flight Phases (1.4) and Operational Flight Envelope (1.5.1). The speed, load factor, and weight ranges would be consistent with Table I (1.5.1) of the Handbook. The mission chosen consists of the following sequences: Takeoff, Climb, Cruise, Air-to-Air Combat, In-Flight Refueling (Receiver), Cruise, Descent, Approach, and Landing. Additional allowance is made for Go-Arounds if necessary]. The aircraft thus specified will be a Class IV-L aircraft.

1.4 Flight Phase Categories. To accomplish the mission requirements the following general Flight Phase categories are involved:

Flight Phase	Category
Takeoff (TO)	C
Climb (CL)	B
Cruise (CR)	B
Air-to-Air Combat (CO)	A
In-Flight Refueling (Receiver) (RR)	A
Cruise (CR)	B
Descent (D)	B
Approach (PA)	C
Go-Around (WO)	C
Landing (L), including tactical landings on a short, narrow, or bomb-cratered runway	C

1.5 Flight Envelopes

1.5.1 Operational Flight Envelopes. The Operational Flight Envelopes define the boundaries in terms of speed, altitude and load factor within which the aircraft must be capable of operating in order to accomplish the missions of Paragraph 1.3. The contractor shall use the representative conditions of Table 1 (1.5.1) of the Handbook for the applicable Flight Phases.

1.5.2 Service Flight Envelopes. For each Aircraft Normal State the contractor shall establish, subject to the approval of the procuring activity, Service Flight Envelopes showing combinations of speed, altitude, and normal acceleration derived from aircraft limits as distinguished from mission requirements. For each applicable Flight Phase and Aircraft Normal State, the boundaries of the Service Flight Envelopes can be coincident with or lie outside the corresponding Operational boundaries. The boundaries of the Service Flight Envelopes shall be based on considerations discussed in the Handbook.

1.5.3 Permissible Flight Envelopes. The contractor shall define Permissible Flight Envelopes which encompass all regions in which operation of the aircraft is both allowable and possible, and which the aircraft is capable of safely encountering. These Envelopes define boundaries in terms of speed, altitude, and load factor.

1.6 State of the Aircraft

1.6.1 Aircraft Normal States. The contractor shall define and tabulate all pertinent items to describe the Aircraft Normal States (no component or system failure) associated with each of the applicable Flight Phases. This tabulation shall be in the format of Table 1 and shall use the nomenclature specified in 4.2. Certain items, 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 will be treated as specific States, and which include the most critical values and the extremes encountered during the Flight Phase in question.

1.6.2 Aircraft Failure States. The contractor shall define and tabulate all Aircraft Failure States, which consist of Aircraft Normal States modified by one or more malfunctions in aircraft components or systems; for example, a discrepancy between a selected configuration and an actual configuration. Those malfunctions that result in center-of-gravity positions outside the center-of-gravity envelope defined in 3.1.1 shall be included. Each mode of failure shall be considered. Failures occurring in any Flight Phase shall be considered in all subsequent Flight Phases.

TABLE 1 (Appendix B)

AIRCRAFT 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	CL													
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													

1.6.3 Aircraft Special Failure States. 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. Special Failure States of this type need not be considered in complying with the requirements of Section 3 if justification for considering them as Special Failure States is submitted by the contractor and approved by the procuring activity.

1.7 Levels of Flying Qualities. The acceptability of the handling characteristics of an aircraft are quantified herein in terms of "Levels" that are defined as follows:

Level 1 Flying qualities clearly adequate for the mission Flight Phase. Aircraft is satisfactory without improvement.

Level 2 Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists. Aircraft deficiencies warrant improvement.

Level 3 Flying qualities such that the aircraft can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate, or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed. Aircraft deficiencies require improvement.

Where possible, the requirements of Section 3 are stated in terms of three limiting values of one or more flying quality parameters. Each value, or combination of values, represents a minimum condition necessary to meet one of the three "Levels" of acceptability.

In some cases sufficient simulation or flight test data do not exist to allow the specification of numerical values of a flying quality parameter. In such cases it is not possible to explicitly define the lower boundary of each Level. These cases are handled by stating the required "Level" of flying qualities for specified piloting tasks, which require compliance by demonstration in flight or via piloted simulation.

It is expected that flying qualities will degrade with increasing atmospheric disturbances and/or Aircraft Failure States. To account for this, the Levels will be adjusted as a function of turbulence magnitude and failures. These adjustments to the definition of flying quality Levels are to be used for those requirements where numerical values are not specifically stated. The adjusted Level definitions should not be construed as a recommendation to degrade flying qualities with increasing values of atmospheric disturbances.

The requirements for aircraft Levels as a function of flight envelopes and failure states are presented in Paragraph 3.1.5. The effect of atmospheric disturbances on Levels is given in Paragraphs 3.9.1 and 3.9.4.

2. APPLICABLE DOCUMENTS. The following specifications and standards, of the issue in effect on the date of invitation for bids or request for proposal, form a part of this specification to the extent specified herein. Copies of specifications and standards required by contractors in connection with specific procurement functions should be obtained from the procuring activity or as directed by the contracting officer.

Specifications

MIL-A-8861	Airplane Strength and Rigidity — Flight Loads
MIL-D-8708	Demonstration Requirements for Airplanes
MIL-F-9490	Flight Control Systems — Design, Installation and Test of, Piloted Aircraft, General Specification for
MIL-C-18244	Control and Stabilization Systems, Automatic, Piloted Aircraft, General Specification for
MIL-F-18372	Flight Control Systems, Design, Installation and Test of, Aircraft (General Specification for)
MIL-S-83691	Stall/Post-Stall/Spin Flight Test Demonstration Requirements for Airplanes
MIL-W-25140	Weight and Balance Control Data (for Airplanes and Rotorcraft)

Standard

MIL-STD-756	Reliability Prediction

Related Documents

Specifications

MIL-C-5011	Charts; Standard Aircraft Characteristics and Performance, Piloted Aircraft
MIL-M-7700	Manual, Flight
MIL-A-8860	Airplane Strength and Rigidity — General Specification for
MIL-A-8871	Airplane Strength and Rigidity Flight and Ground Operations Test
MIL-G-38478	General Requirements for Angle-of-Attack-Based Systems

Standard

MIL-STD-882 Systems Safety Program for Systems and Associated Subsystems and Equipment: Requirement for

Publications

AFSC Design Handbooks:
DH 1-0 General
DH 2-0 Aeronautical Systems

AFFDL Technical Report:
TR 69-72 Background Information and User Guide for MIL-F-8785B, Military Specification — Flying Qualities of Piloted Airplanes, August 1969

AFWAL Technical Report:
TR 81-3109 Background Information and User Guide for MIL-F-8785C, Military Specification — Flying Qualities of Piloted Airplanes, July 1982

3. REQUIREMENTS

3.1 GENERAL REQUIREMENTS

3.1.1 Loadings. The envelope of center of gravity and weight for each flight phase shall be specified by the contractor. In addition, the contractor shall specify the maximum c.g. excursion attainable through failure in systems or components for each flight phase.

3.1.2 Moments and Products of Inertia. The contractor shall define the moments and products of inertia of the aircraft associated with all loadings of 3.1.1. The requirements of this specification shall apply for all moments and products of inertia so defined.

3.1.3 External Stores. The external stores and store combinations to be considered are as follows: 0, 1, or 2 air-to-air wingtip missiles. The requirements of this Standard shall apply to these store conditions. The effects of external stores on the weight, moments of inertia, center of gravity position, and aerodynamic characteristics of the aircraft shall be considered for each mission Flight Phase. When the stores contain expendable loads, the requirements of this Standard apply throughout the range of store loadings.

3.1.4 Configurations. The requirements of this specification shall apply for all configurations required or encountered in the applicable Flight Phases of Section 1.4. 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 selected configurations to be examined must consist of those required for performance and mission accomplishment. Control positions which activate stability augmentation necessary to meet the requirements of this standard are considered to be always on unless otherwise specified.

3.1.5 Allowable Levels for Aircraft Normal States

3.1.5.1 Within Operational Flight Envelopes. The minimum required flying qualities for the Aircraft Normal State within the Operational Flight Envelope will be Level 1. To account for degradation in handling qualities due to atmospheric disturbances the requirements will be adjusted as a function of disturbance magnitude according to the requirements of Paragraph 3.9.1.

3.1.5.2 Within Service Flight Envelopes. The minimum required flying qualities for the Aircraft Normal State within the Service Flight Envelope but outside the Operational Flight Envelope will be Level 2.

3.1.5.3 Within Permissible Flight Envelopes. From all points in the Permissible Flight Envelopes and outside the Service Flight Envelope, it shall be possible readily and safely to return to the Service Flight Envelope without exceptional pilot skill or technique. The requirements on flight at high angle of attack, dive characteristics, dive recovery devices and dangerous flight conditions shall also apply.

3.1.5.4 For ground operation. Some requirements pertaining to taxiing involve operation outside the Operational, Service, and Permissible Flight Envelopes, as at V_s or on the ground. When requirements are stated at conditions such as these, the Levels shall be applied as if the conditions were in the Operational Flight Envelope.

3.1.6 Allowable Levels for Aircraft Failure States

3.1.6.1 Probability Calculation. When Aircraft Failure States exist (1.6.2), a degradation in flying qualities is permitted only if the probability of encountering a lower Level than specified in Para. 3.1.5 is sufficiently small. The contractor shall determine, based on the most accurate available data, the probability of occurrence of each Aircraft Failure State per flight hour within the Operational and Service Flight Envelopes. Each specific failure is assumed to be present at whichever point in the Flight Envelope being considered is most critical (in the flying qualities sense). From these Failure State probabilities and effects, the contractor shall determine the overall probability, per flight hour, that one or more 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 values shown in Table 2.

TABLE 2 (Appendix B)

ALLOWABLE LEVELS FOR AIRCRAFT FAILURE STATES

PROBABILITY OF ENCOUNTERING	WITHIN OPERATIONAL FLIGHT ENVELOPE	WITHIN SERVICE FLIGHT ENVELOPE
Level 2 after failure	$< 2.5 \times 10^{-3}$ per flight hr	
Level 3 after failure	$< 2.5 \times 10^{-5}$ per flight hr	$< 2.5 \times 10^{-3}$ per flight hr

3.1.7 Dangerous Flight Conditions. Dangerous conditions may exist where the aircraft should not be flown. When approaching these flight conditions, it shall be possible by clearly discernible means for the pilot to recognize the impending dangers and take preventive action.

3.1.7.1 Warning and indication. Warning and indication of approach to a dangerous condition shall be clear and unambiguous. For example, a pilot must be able to distinguish readily among stall warning (which requires pitching down or increasing speed), Mach buffet (which may indicate a need to decrease speed), and normal aircraft vibration (which indicates no need for pilot action).

3.1.7.2 Devices for indication, warning, prevention, recovery. It is intended that dangerous flight conditions be eliminated and the requirements of this specification met by appropriate aerodynamic design and mass distribution, rather than through incorporation of a special device or devices. As a minimum, these devices shall perform their function whenever needed but shall not limit flight within the Operational Flight Envelope. Neither normal nor inadvertent operation of such devices shall create a hazard to the aircraft. For Levels 1 and 2, nuisance operation shall not be possible. Functional failure of the devices shall be indicated to the pilot.

3.1.8 Interpretation of Subjective Requirements. In several instances throughout the specification subjective terms, such as objectionable flight characteristics, realistic time delay, normal pilot technique and excessive loss of altitude or buildup of speed, have been employed where insufficient information exists to establish absolute quantitative criteria. Final determination of compliance with requirements so worded will be made by the procuring activity.

3.1.9 Interpretation of Quantitative Requirements. The numerical requirements of this specification generally are stated in terms of a linear mathematical description of the aircraft. Certain factors, for example flight control system nonlinearities and higher-order characteristics or aerodynamic nonlinearities, can cause the aircraft response to differ significantly from that of the linear model. The contractor shall determine equivalent classical systems which have responses most closely matching those of the actual aircraft. Then those numerical requirements of Section 3 which are stated in terms of linear system parameters (such as frequency, damping ratio and modal phase angles) apply to the parameters of that equivalent system rather than to any particular modes of the actual higher-order system. The adequacy of the response match between equivalent and actual aircraft shall be agreed upon by the contractor and the procuring activity.

3.1.10 Quality Assurance

3.1.10.1 Compliance demonstration. Compliance with the quantitative requirements of Section 3 shall be demonstrated through analysis. In addition, compliance with many of the requirements will be demonstrated by simulation, flight test, or both. The methods for demonstrating compliance shall be established by agreement between the procuring activity and the contractor. Representative flight conditions, configurations, external store complements, loadings, etc., shall be determined for detailed investigations in order to restrict the number of design and test conditions. The selected design points must be sufficient to allow accurate extrapolation to the other conditions at which the requirements apply.

a) Analysis. The analytical methods, procedures, assumptions, etc., applied shall be made available to the procuring activity. In some instances (e.g., control power) compliance may be demonstrated partially or wholly by analysis when the analytical model is validated with flight test data and approved by the procuring activity. In other instances (e.g., control in turbulence) analysis will provide information on specific test conditions requiring simulation, flight test, or both.

b) Simulation. The danger, extent or difficulty of flight testing may dictate simulation rather than flight test to evaluate some conditions and events, such as the influence of Severe disturbances, events close to the ground (except 3.2.8.4 shall be demonstrated in flight), combined Failure States and disturbances, etc. In addition, by agreement with the procuring activity, piloted simulation shall be performed before first flight of a new aircraft design in order to demonstrate the suitability of the handling qualities, and also to demonstrate compliance with qualitative requirements in atmospheric disturbances. Where simulation is the ultimate method of demonstrating compliance for a requirement, the simulation model shall be validated with flight test data.

c) Flight test. The required flight tests will be defined by operational, technical, and safety considerations as decided jointly by the procuring activity, the test agency, the contractor, and other involved agencies using results from 3.1.10.1a and 3.1.10.1b. It is expected that flight test demonstration of the requirements in calm air and selected requirements in at least Moderate turbulence will be accomplished to insure that flying quality degradations are not excessive.

3.1.10.2 Design and test conditions. Table 3 specifies general guidelines, but the peculiarities of the specific aircraft design may require additional or alternate test conditions.

a) Terms specified in Table 3 such as "heaviest weight" and "greatest moment of inertia" mean the heaviest and greatest consistent with 3.1.1 and 3.1.2. When a critical center-of-gravity position is identified, the aircraft weight and associated moments of inertia shall correspond to the most adverse service loading in which that critical center-of-gravity position is obtained.

b) Terms specified in Table 3 such as "most forward c.g." and "most aft c.g." mean the most forward or most aft consistent with 3.1.1. When a critical weight or moment of inertia is identified, the center-of-gravity position shall correspond to the most adverse service loading in which that critical weight or moment of inertia is obtained.

c) For terminal Flight Phases, it will normally suffice to examine the selected Aircraft States at only one altitude below 10,000 feet (low altitude). For nonterminal Flight Phases, it will normally suffice to examine the selected Aircraft States at one altitude below 10,000 feet or at the lowest operational altitude (low altitude), the maximum operational altitude ($h_{o_{max}}$), and one intermediate altitude. When the maximum operational altitude is above 40,000 feet or when stability or control characteristics vary rapidly with altitude, more intermediate altitudes than specified in Table 3 shall be investigated. When the Service Flight Envelope extends far above or below the Operational Flight Envelope, the service-altitude extremes must be considered.

TABLE 3 (Appendix B). DESIGN AND TEST CONDITION GUIDELINES

REQUIREMENT NUMBER	TITLE	CRITICAL LOADING	LOAD FACTOR	ALTITUDE	SPEED	FLIGHT PHASE
Section 3.2	HANDLING QUALITY REQUIREMENTS FOR PITCH AXIS					
3.2.1	Response to Pitch Controller	Most forward c.g. † and most aft c.g. §	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	*,CR,RT,PA L,CT
3.2.2	Pilot-Induced Oscillations	--	Minimum permissible to maximum permissible			
3.2.3	Residual Oscillations		1.0		$V_{o_{min}}$ to $V_{o_{max}}$	*,PA
3.2.7.2	Response to failures		All	$h_{o_{min}}$ and $h_{o_{max}}$	V_{min} to V_{max}	--
3.2.7.3	Response to configuration or control mode change		1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$		
3.2.7.4	Response to stores release		$n_o(-)$ to $n_o(+)$		$V_{o_{min}}$ to $V_{o_{max}}$	CO,GA,WD, AD
3.2.7.5	Response to armament delivery					*,RT
3.2.7.6	Response to buffet					*
3.2.8.1	Control power in unaccelerated flight	Most forward c.g.	1.0		V_{min} to V_{max}	--
3.2.8.2	Control power in maneuvering flight	Most forward c.g. †	As required		$V_{o_{min}}$ to $V_{o_{max}}$	CO,GA,AR, TF,CR,PA
3.2.8.3	Control power for takeoff	Most forward c.g. (nose-wheel), most aft c.g. (tail-wheel aircraft)	1.0	Low	As required	TO
3.2.8.4	Control power for landing	Most forward c.g.	1.0	Low	$V_s(L)$ or geometric limit	L
3.2.8.5	Control power for other conditions	--	All	$h_{o_{min}}$, medium, $h_{o_{max}}$	All	--
3.2.9.1	Steady-state control force per g	Most forward c.g. † and most aft c.g. §	$n(-)$ to $n(+)$		V_{min} to V_{max}	*,RT,CR,PA L,CT
3.2.9.2	Transient control force per g	Most aft c.g. §	1.0			--
3.2.9.3	Control force variations during rapid speed charges	--	As required		$V_{o_{min}}$ to $V_{o_{max}}$ and transonic	CO,GA,DE
3.2.9.4.1	Control force vs. deflection -- steady-state gradient	Most forward c.g. †	$n_o(-)$ to $n_o(+)$		V_{min} to V_{max}	*,RT,CR,PA, L,CT
3.2.9.4.2	Transient control force vs. deflection	Most aft c.g. §	1.0			--
3.2.9.5	Control centering and breakout forces	--	$n_o(-)$ to $n_o(+)$	$h_{o_{min}}$ and $h_{o_{max}}$		
3.2.9.6	Free play					
3.2.9.7.1	Force limits -- takeoff	Most forward c.g. and most aft c.g.	As required	Low	0 to V_{max}(TO)	TO,CT

† Combined with heaviest weight.

§ Combined with lightest weight.

* All applicable Category A Flight Phases.

-- No general guidance can be provided.

"As required" -- flight conditions are specified in requirement or are determined by nature of test maneuver.

TABLE 3 (Continued)

REQUIREMENT NUMBER	TITLE	CRITICAL LOADING	LOAD FACTOR	ALTITUDE	SPEED	FLIGHT PHASE
3.2.9.7.2	Force limits -- landing	Most forward c.g.	1.0	Low	$V_S(L)$ or geometric limit	L
3.2.9.7.3	Force limits -- dives: SFE	Most forward c.g. † and most aft c.g. §	As required	2000 ft MSL to h_{max}	V_{min} to V_{max}	D,ED,CO, CR,GA
	PFE	↓	↓	As required	V_{MAT} to maximum permissible	↓
3.2.9.7.4	Force limits -- sideslips	--	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	CO,CR,PA,L
3.2.9.7.6	Force limits -- failures	--	All	$h_{o_{min}}$ and $h_{o_{max}}$	V_{min} to V_{max}	--
3.2.9.7.7	Force limits -- configuration or control mode charge	↓	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	↓	↓
3.2.9.8	Trim systems	Most forward c.g. and most aft c.g.	↓	↓		
3.2.9.8.1	Trim systems -- rate of operation	--		As required	As required	D,ED,CO, GA
3.2.9.8.2	Trim systems -- stalling of trim systems	Most forward c.g. †	As required	↓	Start of dive recovery to V_{max}	D,ED,CO, CR
3.2.9.8.3	Trim systems -- irreversibility	--	1.0	MSL to h_{max}	V_{min} to V_{max}	--
3.2.10.1	Control displacements -- takeoff	Most forward c.g. and most aft c.g.	As required	Low	0 to $V_{max}(TO)$	TO,CT
3.2.10.2	Control displacements -- maneuvering	↓	n(-) to n(+)	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	*,RT,CR,PA, L,CT
3.2.10.3	Control displacements -- gust regulation	--	$n_o(-)$ to $n_o(+)$	$h_{o_{min}}$ and $h_{o_{max}}$	↓	--
SECTION 3.3	HANDLING QUALITY REQUIREMENTS FOR VERTICAL FLIGHT PATH AXIS					
3.3.1.2.1	Response to attitude change -- steady-state response	--	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	$V_{o_{min}}$ and $V_{o_{min}}$ - 5 kt	PA
SECTION 3.4	HANDLING QUALITY REQUIREMENTS FOR LONGITUDINAL AXIS					
3.4.1	Response to Attitude Changes	Most aft c.g.	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	CO,RR,FF,CR LO,RT,All Category C
3.4.1.1	Relaxation in transonic flight	↓	↓	↓	Transonic	CO
SECTION 3.5	HANDLING QUALITY REQUIREMENTS FOR ROLL AXIS					
3.5.1	Roll Response to Roll Controller	--	1.0 and $n_o(+)$	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	*,CL,CR,LO, RT,DE,PA,L

† Combined with heaviest weight.

§ Combined with lightest weight.

* All applicable Category A Flight Phases.

-- No general guidance can be provided.

"As required" -- flight conditions are specified in requirement or are determined by nature of test maneuver.

TABLE 3 (Continued)

REQUIREMENT NUMBER	TITLE	CRITICAL LOADING	LOAD FACTOR	ALTITUDE	SPEED	FLIGHT PHASE
3.5.2	Pilot-Induced Oscillations	↓	Minimum permissible to maximum permissible	MSL to h_{max}	V_{min} to V_{max}	--
3.5.4	Linearity of Roll Response to Roll Controller	Greatest rolling moment of inertia	As required (not above $0.8n_L$)	$h_{o_{min}}$, medium, $h_{o_{max}}$	↓	CO,GA,TF,CL, CR,TO,CT
3.5.6	Roll Response to Yaw Controller	Lightest weight	1.0	↓	↓	CO,CR,PA,L
3.5.7	Roll Control for Takeoff and Landing in Crosswinds	--	↓	Low	As required	Taxi,TO,L
3.5.8.1	Response to asymmetric thrust	Lightest weight	1.0	All	V_{min} to V_{max}	CO,GA,TF,CL, CR,TO,CT
3.5.8.2	Response to failures	--	All	$h_{o_{min}}$ and $h_{o_{max}}$	↓	--
3.5.8.3	Response to configuration or control mode change	--	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	--
3.5.8.4	Response to stores release	↓	$n_o(-)$ to $n_o(+)$	↓	$V_{o_{min}}$ to $V_{o_{max}}$	CO,GA,WD,AD
3.5.8.5	Response to armament delivery	↓	↓	↓	↓	*,RT
3.5.9.1	Control power - response to roll control inputs	Greatest and smallest rolling moments of inertia	As required (not above $0.8\ n_L$)	$h_{o_{min}}$, medium, $h_{o_{max}}$	↓	As required
3.5.9.2	Control power - steady sideslips	Lightest weight	1.0	↓	↓	CO,CR,PA,L
3.5.9.3	Control power - crosswinds	--	As required	Low	As required	TO,L,PA
3.5.9.4	Control power - engine failures	Lightest weight	1.0	$h_{o_{min}}$	Down to $V_{min}(TO)$	TO,CT
3.5.9.5	Control power - dives and pullouts	--	As required	2000 ft MSL to h_{max}	V_{MAT} to V_{max}	D,ED
3.5.9.6	Control power - stores release	--	$n_o(-)$ to $n_o(+)$	$h_{o_{min}}$, medium, $h_{o_{max}}$	$V_{o_{min}}$ to $V_{o_{max}}$	CO,GA,WD,AD
3.5.9.7	Control power - two engines inoperative	Lightest weight	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{range} (1 and 2 engines out)	--
3.5.9.8	Control power for other conditions	--	All	↓	All	--
3.5.10.1	Wheel control displacements	Greatest rolling moment of inertia	As required (not above $0.8n_L$)	↓	V_{min} to V_{max}	CO,GA,AR,TF, CR,GA,L
3.5.10.2	Forces to achieve required roll rates	Greatest and smallest rolling moments of inertia	↓	↓	↓	↓
3.5.10.3	Sensitivity	Smallest rolling moment of inertia	↓	↓	↓	↓
3.5.10.4	Breakout and centering forces	--	$n_o(-)$ to $n_o(+)$	$h_{o_{min}}$ and $h_{o_{max}}$	V_{min} to V_{max}	--

† Combined with heaviest weight.

§ Combined with lightest weight.

* All applicable Category A Flight Phases.

-- No general guidance can be provided.

"As required" -- flight conditions are specified in requirement or are determined by nature of test maneuver.

TABLE 3 (Continued)

REQUIREMENT NUMBER	TITLE	CRITICAL LOADING	LOAD FACTOR	ALTITUDE	SPEED	FLIGHT PHASE
3.5.10.5	Free play	--	$n_o(-)$ to $n_o(+)$	$h_{o_{min}}$ and $h_{o_{max}}$	V_{min} to V_{max}	--
3.5.10.6.1	Force limits - steady turns	↓	As required	$h_{o_{min}}$, medium, $h_{o_{max}}$	$V_{o_{min}}$	CO,CR,LO,PA
3.5.10.6.2	Force limits - dives and pullouts	↓	↓	2000 ft MSL to h_{max}	V_{MAT} to V_{max}	D,ED
3.5.10.6.3	Force limits - cross-winds	↓	1.0	Low	As required	TO,L
3.5.10.6.4	Force limits - steady sideslips	↓	↓	↓	V_{min} to V_{max}	↓
3.5.10.6.5	Force limits - engine failures after takeoff	Lightest weight	↓	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min}(TO) to 1.4V_{min}	CR,TO,CT
<u>SECTION 3.6</u>	HANDLING QUALITY REQUIREMENTS FOR YAW AXIS					
3.6.1.1.1	Equivalent systems requirement - transient response	Greatest rolling moment of inertia	1.0 and $n_o(+)$	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	*,CR,RT,PA,L
3.6.1.1.2	Equivalent systems requirement - steady-state response	Lightest weight	1.0	↓	↓	CO,CR,PA,L
3.6.2.1	Yaw response to roll controller - coordination in turn entry and exit	Greatest yawing and rolling moments of inertia	↓	↓	↓	*,CR,PA,L
3.6.2.2	Pilot-induced oscillations	--	Minimum permissible to maximum permissible	MSL to h_{max}	↓	--
3.6.3	Yaw Control for Takeoff and Landing in Crosswinds	--	1.0	Low	As required	TO,L,Taxi
3.6.4.1	Response to asymmetric thrust	Lightest weight	1.0	$h_{o_{min}}$	0 to V_{max}(TO)	TO,CT
		↓	↓	All	V_{min} to V_{max}	CO,GA,TF,CR CL,TO,CT
		↓	↓	$h_{o_{min}}$, medium, $h_{o_{max}}$	1.4V_{min}	CR
3.6.4.2	Response to failures	--	All	↓	V_{min} to V_{max}	--
3.6.4.3	Response to configuration or control mode change	--	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	--
3.6.4.4	Response to stores release	↓	$n_o(-)$ to $n_o(+)$	↓	$V_{o_{min}}$ to $V_{o_{max}}$	CO,GA,WD,AD
3.6.4.5	Response to armament delivery	--	↓	↓	↓	*,RT
3.6.5.1	Control power - takeoff, landing, and taxi	--	As required	Low	0 to V_{max}(TO)	PA,TO,L,Taxi
3.6.5.2	Control power - two engines inoperative	Lightest weight	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{range} (1 and 2 engines out)	--

* All applicable Category A Flight Phases. -- No general guidance can be provided.

"As required" -- flight conditions are specified in requirement or are determined by nature of test maneuver.

TABLE 3 (Concluded)

REQUIREMENT NUMBER	TITLE	CRITICAL LOADING	LOAD FACTOR	ALTITUDE	SPEED	FLIGHT PHASE
3.6.5.3	Control power - asymmetric loading	--	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	CO,GA,CR,D, PA,L
3.6.6	Yaw Axis Control Forces	↓	$n_o(-)$ to $n_o(+)$	↓	↓	*,CR,PA,L
3.6.6.1	Force linearity	Lightest weight	1.0			CO,CR,PA,L
3.6.6.2.1	Force limits - rolling maneuvers	Greatest rolling moment of inertia	As required			CO,GA,AR,TF, CR,PA,L
3.6.6.2.2	Force limits - steady turns	--	↓		$V_{o_{min}}$	CO,CR,LO,PA
3.6.6.2.3	Force limits - speed changes	↓	1.0		V_{min} to V_{max}	CO,GA,CR,D, PA,L
3.6.6.2.4	Force limits - crosswinds	--	1.0	Low	As required	TO,L
3.6.6.2.5	Force limits - asymmetric loading	↓	↓	$h_{o_{min}}$, medium, $h_{o_{max}}$	$V_{o_{min}}$ to $V_{o_{max}}$	--
3.6.6.2.6	Force limits - dives and pullouts		As required	2000 ft MSL to h_{max}	V_{MAT} to V_{max}	D,ED
3.6.6.2.7	Force limits - go-arounds	Lightest weight	1.0	Low	V_{min}(PA) or landing speed	WO
3.6.6.2.8	Force limits-asymmetric thrust	↓	↓	$h_{o_{min}}$	0 to V_{max}(TO)	TO,CT
3.6.6.2.9	Force limits - failures	--	All	$h_{o_{min}}$ and $h_{o_{max}}$	V_{min} to V_{max}	--
3.6.6.2.10	Force limits - configuration or control mode changes	↓	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	↓	↓
SECTION 3.8	HANDLING QUALITY REQUIREMENTS FOR COMBINED AXES					
3.8.1	Cross-Axis Coupling in Roll Maneuvers	--	0 to $0.8n_L$	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	CO,GA,AR,TF
3.8.2	Crosstalk Between Pitch and Roll Controllers	↓	$n_o(-)$ to $n_o(+)$	↓	↓	--
3.8.3	Control Harmony	See MIL-S-83691 or MIL-D-8708, whichever is applicable for flight demonstration. More severe conditions generally will be investigated by analysis and model testing.				
3.8.4	Flight at High Angle of Attack	↓	↓	↓	↓	↓

* All applicable Category A Flight Phases. -- No general guidance can be provided.

"As required" flight conditions are specified in requirement or are determined by nature of test maneuver.

3.2 HANDLING QUALITY REQUIREMENTS FOR PITCH AXIS

3.2.1 Pitch Attitude Response to Pitch Controller

3.2.1.1 Pitch axis equivalent systems requirements. The equivalent parameters describing the responses of pitch rate and normal load factor (at the center of rotation) to a pitch control force input shall have the characteristics specified by Table 4.

1) Short term response

$$\frac{\dot{\theta}}{F_s} = \frac{K_\theta[s + (1/T_{\theta_2})]e^{-\tau_{e_\theta} s}}{s^2 + 2\zeta_{sp}\omega_{sp}s + \omega_{sp}^2}$$

$$\frac{n_z'}{F_s} = \frac{K_{n_z}e^{-\tau_{e_n} s}}{s^2 + 2\zeta_{sp}\omega_{sp}s + \omega_{sp}^2}$$

Notes:

a) The equivalent systems are to be obtained from simultaneous matching of the $\dot{\theta}/F_s$ and n_z'/F_s higher order system responses over a frequency range of approximately 0.1 to 10 rad/sec.

b) n_z' is normal acceleration as measured at the aircraft center of rotation.

2. Long term response

$$\frac{\dot{\theta}}{F_s} = \frac{K_\theta[s + (1/T_{\theta_1})]}{s^2 + 2\zeta_p\omega_p s + \omega_p^2}$$

Note: While a lower order equivalent system match could be used, the parameter ζ_p is the only one specified and it can be generally calculated directly from a time response.

Mismatch in terms of amplitude and phase angle for the short-term pitch rate response shall be compared with the envelopes of Figure 1 for Category A and C Flight Phases. For any cases that fall outside these envelopes the requirements of 3.2.1.2 shall apply.

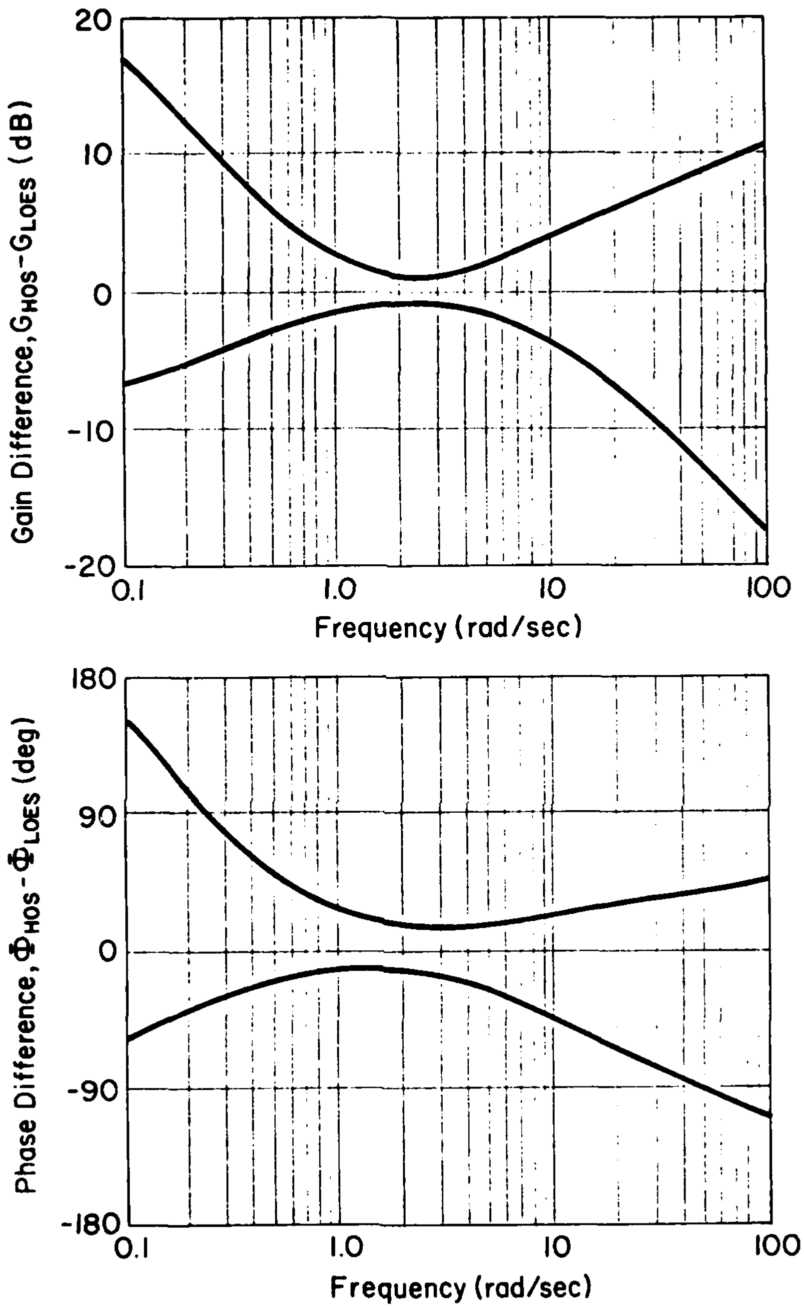


Figure 1 (Appendix B). Envelopes of Allowable Mismatch (Pitch Rate Response)

TABLE 4 (Appendix B)
LOCATIONS OF REQUIREMENTS FOR PITCH RESPONSE
TO PITCH CONTROLLER

PARAMETER	ω_{sp} vs. n/α
ζ_p	Table 5
ω_p	No Requirement
ζ_{sp}	Table 6
ω_{sp}	Figure 2
n/α	Figure 2
$1/T_{\theta_1}$	At least greater than Zero
$1/T_{\theta_2}$	--
τ_{e_θ}, τ_{e_n}	Table 7

TABLE 5 (Appendix B)
EQUIVALENT PHUGOID DAMPING RATIO LIMITS

Level 1	ζ_p at least 0.04
Level 2	ζ_p at least 0
Level 3	T_2 at least 55 sec ($T_2 = -0.693/\zeta_p\omega_p$) and $\lvert\zeta_p\rvert < 1.0$

TABLE 6 (Appendix B)
EQUIVALENT SHORT-PERIOD DAMPING RATIO LIMITS

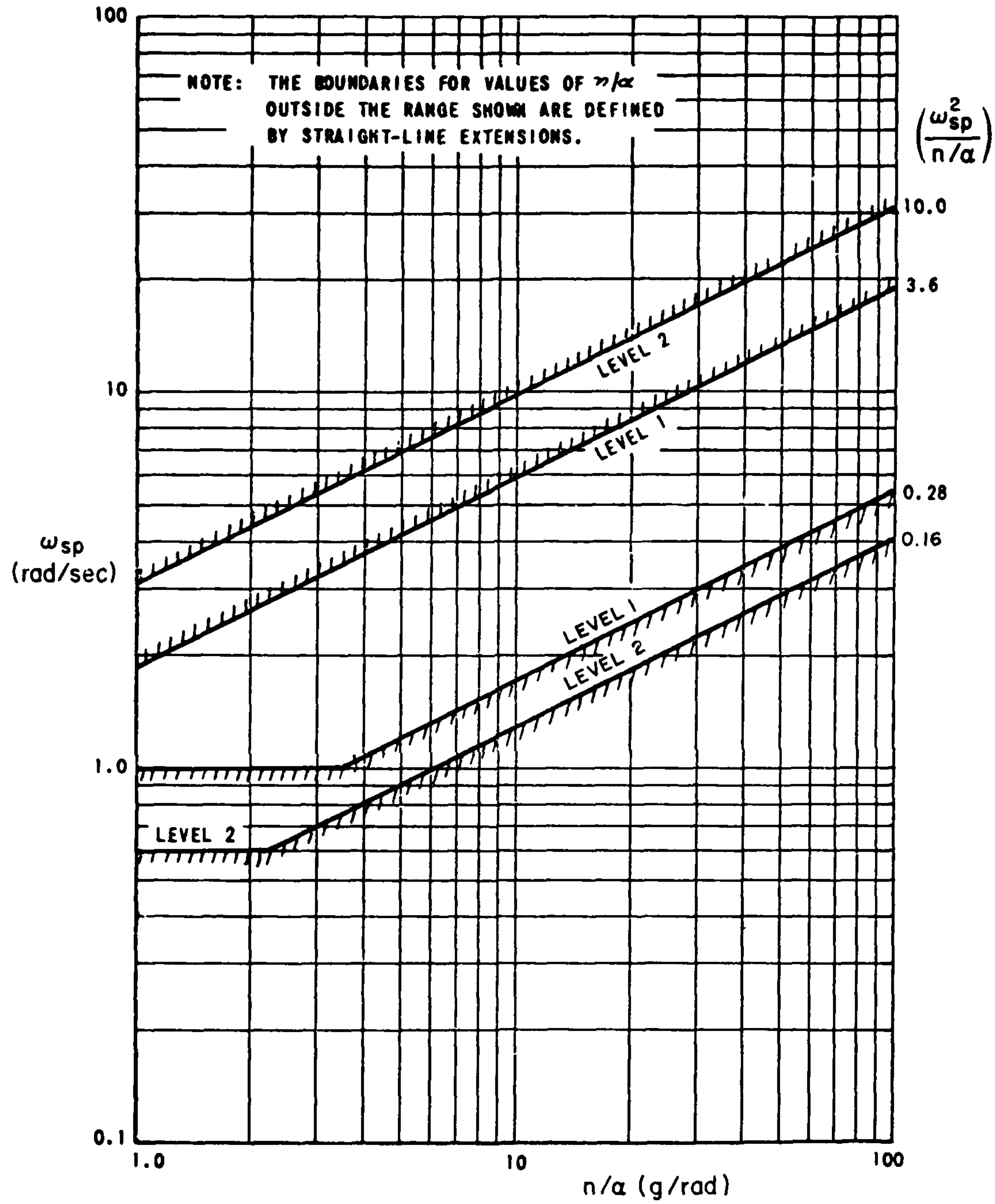
LEVEL	CATEGORY A AND C FLIGHT PHASES		CATEGORY B FLIGHT PHASES	
	MINIMUM	MAXIMUM	MINIMUM	MAXIMUM
1	0.35	1.30	0.30	2.00
2	0.25	2.00	0.20	2.00
3	$T_2 > 6$ sec*	—	$T_2 > 6$ sec*	--

*In the presence of one or more other Level 3 flying qualities, ζ_{sp} shall be at least 0.05 unless flight safety is otherwise demonstrated to the satisfaction of the procuring activity. T_2 applies to the value of an unstable first-order short-period root

TABLE 7 (Appendix B)
LIMITS ON AIRCRAFT RESPONSE DELAY, τ_e

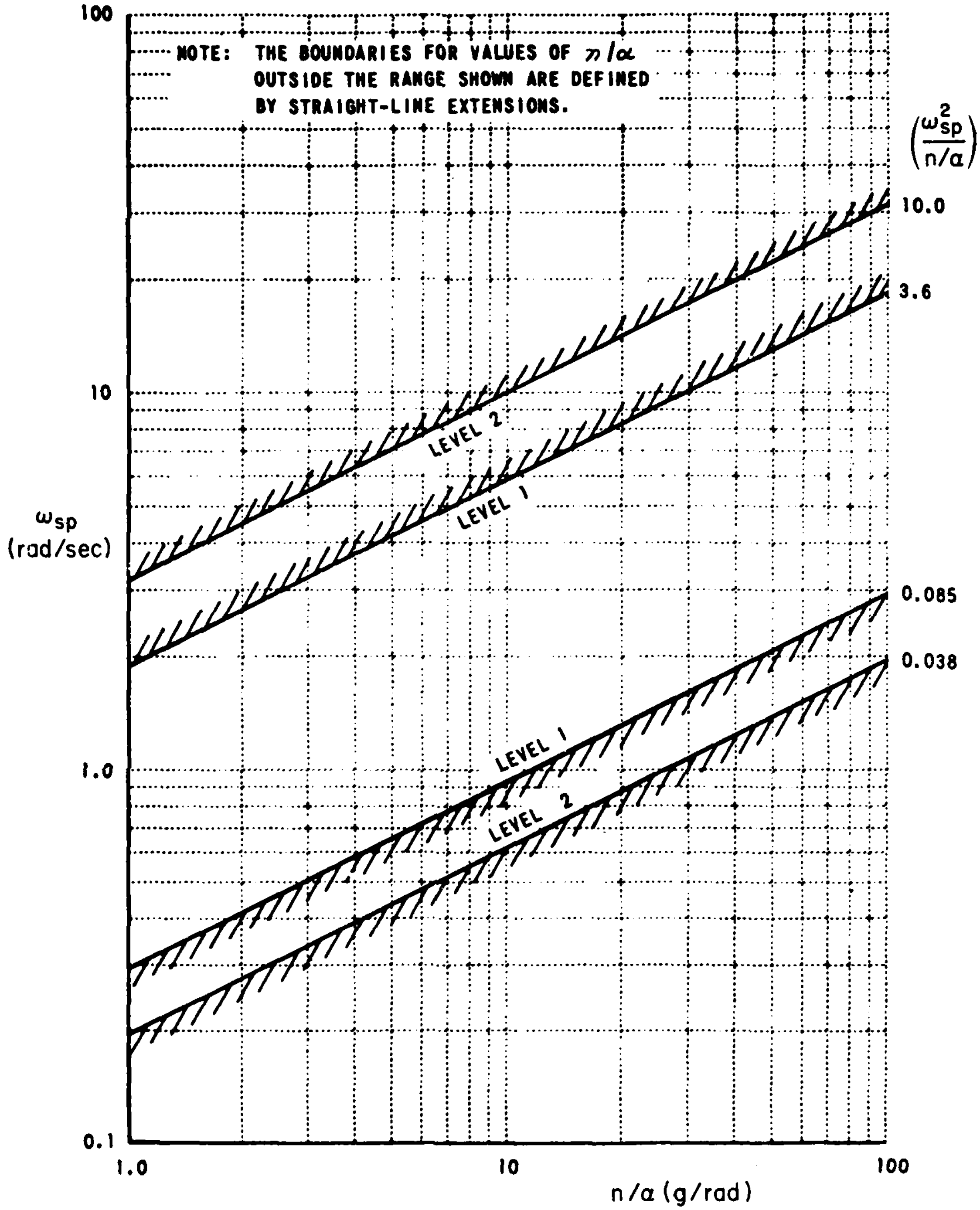
LEVEL	ALLOWABLE DELAY (sec)
1	0.10
2	0.20
3	0.25

τ_e is the greater of τ_{e_θ} or τ_{e_n}



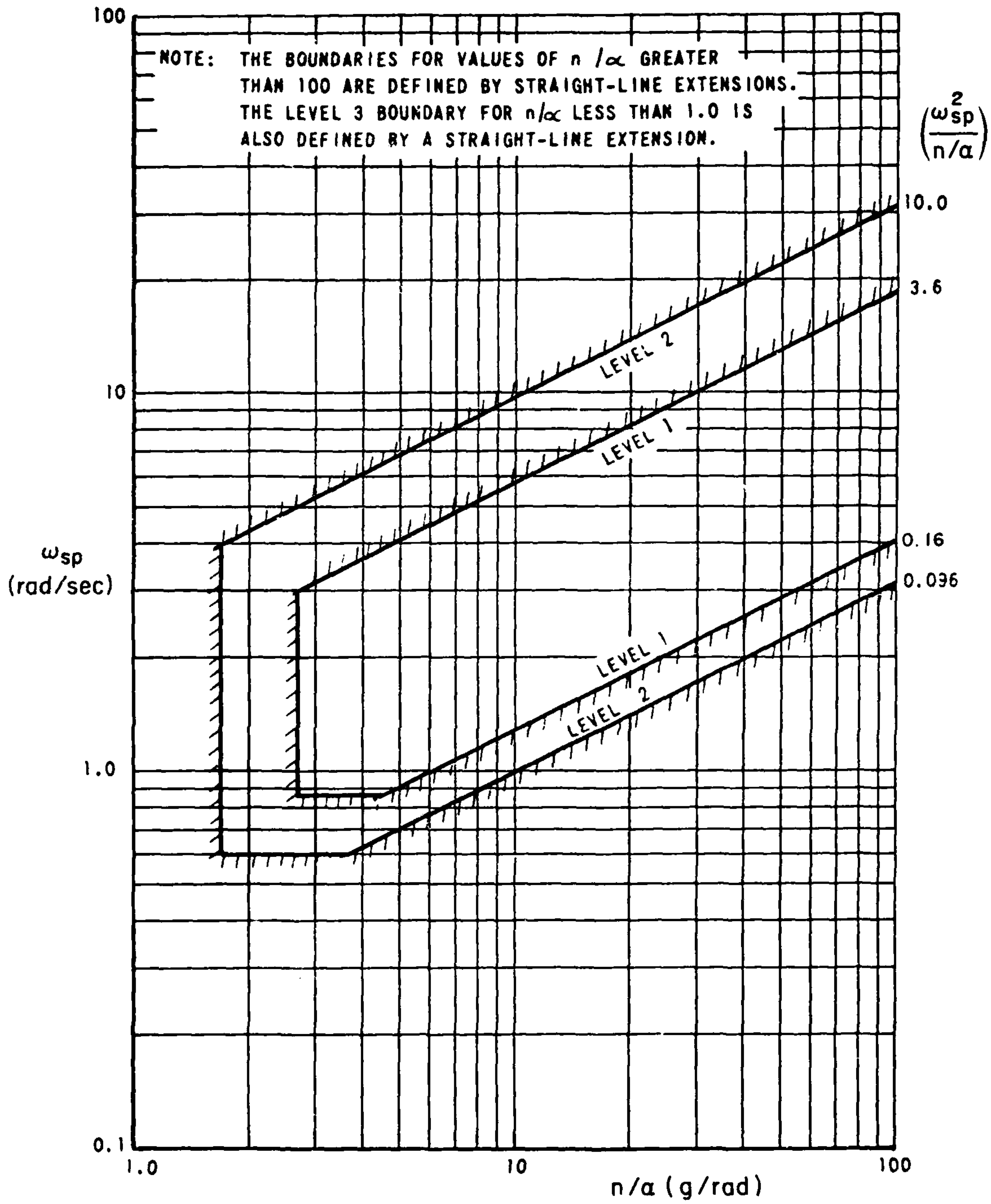
a) Category A Flight Phases

Figure 2 (Appendix B). Requirements for Short-Term Pitch Response to Pitch Controller (ω_{sp} vs. n/α)



b) Category B Flight Phases

Figure 2. Continued



c) Category C Flight Phases

Figure 2. Concluded

3.2.1.2 Pitch axis bandwidth requirements. For any equivalent system of 3.2.1.1 with mismatch outside the envelopes of Figure 1, the bandwidth of the open-loop pitch attitude response to pitch controller shall meet the requirements of Figure 3.

3.2.2 Pilot-induced pitch oscillations — qualitative requirement. There shall be no tendency for pilot-induced oscillations, that is, sustained or uncontrollable oscillations resulting from the efforts of the pilot to control the aircraft. The pitch attitude response dynamics of the airframe plus control system shall not change abruptly with the motion amplitudes of pitch, pitch rate or normal acceleration unless it can be shown that this will not result in a pilot-induced oscillation.

3.2.3 Residual Pitch Oscillations. Any sustained residual oscillations in calm air shall not interfere with the pilot's ability to perform the tasks required in service use of the aircraft. For Levels 1 and 2, oscillations in normal acceleration at the pilot's station greater than ±0.02 g will be considered excessive for any Flight Phase. These requirements shall apply with the pitch control fixed and with it free.

3.2.4 Vertical Acceleration at Pilot Station. Vertical acceleration at the pilot station due to pitch control inputs shall not be objectionable to the pilot.

3.2.5 Pitch Axis Response to Secondary Controllers. The pitch attitude response to a rapid change in secondary cockpit flight control (throttle, DLC, etc.) shall not be objectionable to the pilot.

3.2.6 [Reserved]

3.2.7 Pitch Axis Response to Other Inputs

3.2.7.1 Pitch axis response to auxiliary controls. The pitch response to any auxiliary control shall not be objectionable to the pilot.

3.2.7.2 Pitch axis response to failures

a) Closed-Loop: The pitch attitude motions following sudden aircraft system or component failures shall be such that dangerous conditions can be avoided by pilot corrective action. A time delay of at least 1 sec between the failure and initiation of pilot corrective action shall be incorporated when determining compliance. No single failure of any component or system shall result in Level 3 pitch-axis flying qualities; Special Failure States (1.6.3) are excepted. The crew member concerned shall be provided with immediate and easily interpreted indications whenever failures occur that require or limit any flight crew action or decision.

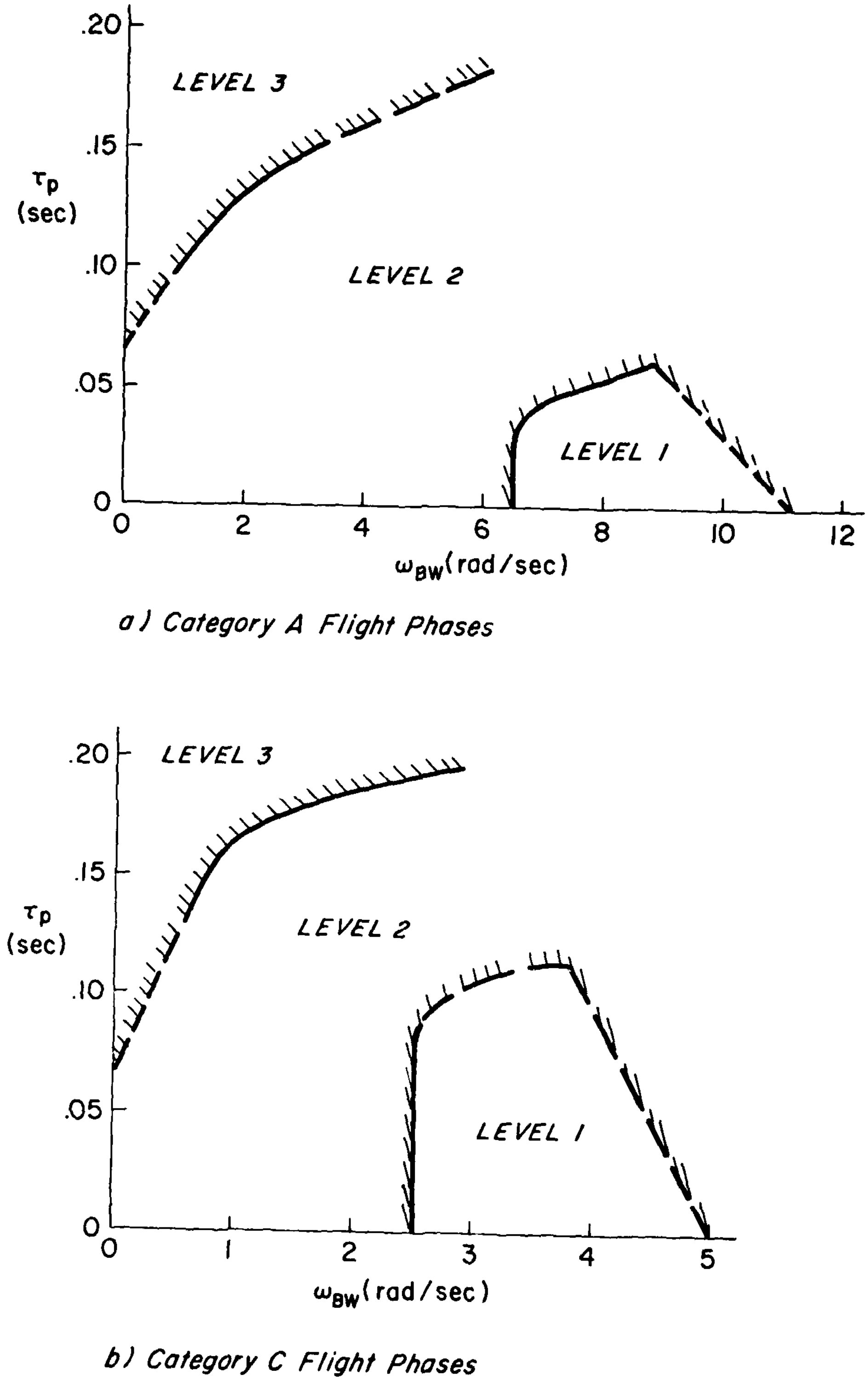


Figure 3 (Appendix B). Bandwidth Requirements

b) Open-Loop: With controls free, the aircraft motions due to partial or complete failure of the augmentation system shall not exceed the following limits for at least 2 seconds following the failure:

Levels 1 and 2 (after failure): ±0.5 g incremental normal acceleration at the pilot's station, except that neither stall angle of attack nor structural limits shall be exceeded. In addition, for Category A, vertical excursions of 5 feet.

Level 3 (after failure): No dangerous attitude or structural limit is reached, and no dangerous alteration of the flight path results from which recovery is impossible.

3.2.7.3 Pitch axis response to configuration or control mode change. The transient motions and trim changes resulting from the intentional engagement or disengagement of any portion of the primary flight control system by the pilot shall be such that dangerous flying qualities never result. With controls free, the motion transients resulting from these situations shall not exceed the following limits for at least 2 seconds following the transfer: ±0.05 g normal acceleration at the pilot's station. These requirements apply only for Aircraft Normal States (1.6.1).

3.2.7.4 Pitch axis response to stores release. The intentional release of any stores shall not result in objectionable flight characteristics for Levels 1 and 2. However, the intentional release of stores shall never result in dangerous or intolerable flight characteristics. This requirement applies for all flight conditions and store loadings at which normal or emergency store release is structurally permissible.

3.2.7.5 Pitch axis response to armament delivery. Operation of movable parts or firing of weapons shall not cause buffet, trim changes, or other characteristics which impair the tactical effectiveness of the aircraft under any pertinent flight conditions. These requirements shall be met for Levels 1 and 2.

3.2.7.6 Buffet. Within the boundaries of the Operational Flight Envelope, there shall be no objectionable buffet which might detract from the effectiveness of the aircraft in executing its intended missions.

3.2.8 Pitch Axis Control Power

3.2.8.1 Pitch axis control power in unaccelerated flight. In steady 1 g flight at all service altitudes, the attainment of all speeds between V_S and V_{max} shall not be limited by the effectiveness of the longitudinal control or controls.

3.2.8.2 Pitch axis control power in maneuvering flight. Within the Operational Flight Envelope, it shall be possible to develop, by use of the pitch control alone, the following range of load factors:

Levels 1 and 2: $n_o(-)$ to $n_o(+)$

Level 3: $n = 0.5$ g to the lower of:

a) $n_o(+)$

b) $n = 2.0$ for $n_o(+) \leq 3$ g

$0.5\ [n_o(+) + 1]$ for $n_o(+) > 3$ g

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 percent or ±50 kt equivalent airspeed (except where limited by the boundaries of the Operational Flight Envelope).

3.2.8.3 Pitch axis control power in takeoff. The effectiveness of the pitch control shall not restrict the takeoff performance of the aircraft. Satisfactory takeoffs shall not be dependent upon use of the trimmer control during takeoff or on complicated control manipulation by the pilot. It shall be possible to obtain and maintain the following attitude during the takeoff roll:

- The pitch attitude at 0.9 V_{min} that will result in a liftoff at V_{min}.

These requirements shall be met on hard surface runways and on debris-strewn or shell-damaged runways.

[This requirement regulates against tailwheel airplanes].

3.2.8.4 Pitch axis control power in landing. The pitch control shall be sufficiently effective in the landing Flight Phase in close proximity to the ground so that during landing flare and rollout with the aircraft trimmed for the minimum recommended approach speed not to exceed 1.3 $V_S(L)$:

- The geometry limited touchdown attitude can be achieved at touchdown.
- The guaranteed minimum landing speed [$V_{min}(L)$] can be achieved when flaring from shallow ($\gamma \approx -3^o$) and steep ($\gamma \approx -6^o$) approaches.
- The nosewheel can be gently lowered to the ground at speeds down to 0.9 $V_{min}(L)$.

3.2.8.5 Pitch axis control power for other conditions. Control authority, rate and hinge moment capability shall be sufficient to assure safety throughout the combined range of all attainable angles of attack (both positive and negative) and sideslip. This requirement applies to the prevention of loss of control and to recovery from any situation for all maneuvering, including pertinent effects of factors such as regions of control-surface-fixed instability, inertial coupling, fuel slosh, the influence of symmetric and asymmetric stores, stall/post-stall/spin characteristics, atmospheric disturbances and Aircraft Failure States (maneuvering flight appropriate to the Failure State is to be included). Consideration shall be taken of the degrees 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.

3.2.9 Pitch Axis Control Forces

3.2.9.1 Pitch axis control forces — steady-state control force per g.

a) Control Feel and Stability in Maneuvering Flight at Constant Speed. In steady turning flight and in pullups and pushovers at constant speed, for Levels 1 and 2 there shall be no tendency for the aircraft pitch attitude or angle of attack to diverge aperiodically with controls fixed or with controls free. For the above conditions, the incremental control force 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 those required to initiate the change. These requirements apply for all local gradients throughout the range of service load factors defined in 1.5.2.

b) Control Forces in Maneuvering Flight. 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 following load factor ranges:

CLASS	MIN.	MAX.
I, II & III	0.5	$0.5[n_o(+) + 1]$ or 3
IV	0	Whichever is less

Outside this range, a departure from linearity resulting in a local gradient which differs from the average gradient for the maneuver by more than 50 percent is considered excessive, except that larger increases in force gradient are permissible at load factors greater than 0.85 n_L. The local force gradients shall be within the limits of Table 8 for centerstick controllers. In addition, F_s/n should be near the Level 1 upper boundaries of these gradients for combinations of high frequency and low damping. The term gradient does not include that portion of the force versus n curve within the breakout force.

For side stick controllers, the contractor shall show that the control force gradients will produce suitable flying qualities.

TABLE 8 (Appendix B). PITCH MANEUVERING FORCE GRADIENT LIMITS - CENTER STICK CONTROLLERS

LEVEL	MAXIMUM GRADIENT $(F_s/n)_{max}$, lb/g	MINIMUM GRADIENT $(F_s/n)_{min}$, lb/g
1	$240/(n/\alpha)$ but not more than 28.0 nor less than $56/(n_L - 1)$*	The greater of $21/(n_L - 1)$ and 3.0
2	$360/(n/\alpha)$ 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

*For $n_L < 3$, $(F_s/n)_{max}$ is 28.0 for Level 1, 42.5 for Level 2.

3.2.9.2 Pitch axis control forces — transient control force per g. The buildup of control force during the maneuver entry must not lag the buildup of normal acceleration at the pilot's location. In addition, the frequency response of normal acceleration at the pilot station to pitch control force input shall be greater than the following for all frequencies greater than 1.0 rad/sec. Units are pounds per g.

Level 1	Level 2	Level 3
$\frac{14}{n_L - 1}$	$\frac{12}{n_L - 1}$	$\frac{8}{n_L - 1}$

3.2.9.3 Pitch axis control forces — 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.

3.2.9.4 Pitch axis control forces — control force vs. control deflection

3.2.9.4.1 Steady-state control force/deflection gradient. The average gradient of pitch-control force per unit of pitch-control deflection at constant speed shall be within the following range for Category A Flight Phases.

	MINIMUM	MAXIMUM
Centerstick controllers	5 lb/in.	No data
Sidestick controllers	1 lb/in.	2.5 lb/in.

For other Flight Phases, the gradient shall not be so large or so small as to be objectionable.

3.2.9.4.2 Transient control force vs. deflection The deflection of the pilot's control must not lead the control force throughout the frequency range of pilot control inputs. In addition, the peak pitch control forces developed during abrupt maneuvers shall not be objectionably light.

3.2.9.5 Pitch axis control forces — control centering and breakout forces. Longitudinal controls should 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. Breakout forces, including friction, preload, etc., shall be within the limits of Table 9. These values refer to the cockpit control force required to start movement of the control surface.

TABLE 9 (Appendix B). PITCH AXIS BREAKOUT FORCES (LB)

CONTROL	CLASSES I, II-C, IV		CLASSES II-L, III	
	MINIMUM	MAXIMUM	MINIMUM	MAXIMUM
Centerstick	1/2	3	1/2	5
Sidestick	1/2	1	1/2	1

Values for Levels 1 and 2 (Upper Limits Doubled for Level 3)

3.2.9.6 Pitch axis control forces — free play. The free play (and possible associated hysteresis) in the longitudinal controller shall not result in objectionable flight characteristics, especially for small amplitude inputs.

3.2.9.7 Pitch axis control force limits

3.2.9.7.1 Pitch axis control force limits — takeoff. With the trim setting optional but fixed, the pitch-control forces required during all types of takeoffs for which the aircraft is designed, including short-field takeoffs, shall be within the following limits: for centerstick controllers, 30 pounds pull to 10 pounds push; for sidestick controllers the force shall not be objectionable to the pilot. The term takeoff includes the ground run, rotation, and liftoff, and the ensuing acceleration to V_{max} (TO). Takeoff encompasses operation in both the presence and absence of ground effect. Takeoff power should 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).

3.2.9.7.2 Pitch axis control force limits — landing. The pitch control forces for landing shall be less than 35 pounds pull for centerstick controllers for the recommended approach speed and fixed trim settings. For sidestick controllers the forces shall not be objectionable to the pilot. This applies in both presence and absence of ground effect.

3.2.9.7.3 Pitch axis control force limits — dives

Service Flight Envelope. With the aircraft trimmed for level flight at speeds throughout the Service Flight Envelope, the control forces in dives to all attainable speeds within the Service Flight Envelope shall not exceed ___(a)___. In similar dives, but with use of trim following the dive entry, it shall be possible with normal piloting techniques to maintain the forces within the following limits: ___(b)___.

Permissible Flight Envelope. With the aircraft trimmed for level flight at V_{MAT} but with use of trim optional in the dive, it shall be possible to maintain the pitch control force within the following limits in dives to all attainable speeds within the Permissible Flight Envelope: (c) . The force required for recovery from these dives shall not exceed: (d) . Trim and deceleration devices, etc., may be used to assist in recovery if no unusual pilot technique is required.

Note: Letters in blanks correspond to values in Table 10.

TABLE 10 (Appendix B)
FORCE LIMITS FOR DIVES AND RECOVERY FROM DIVES

REQUIREMENT NUMBER	CONTROLLER	FORCE (lb)	
		PUSH	PULL
(a)	Centerstick Sidestick	50 *	10 *
(b)	Centerstick Sidestick	10 *	10 *
(c)	Centerstick Sidestick	50 *	35 *
(d)	Centerstick Sidestick	120 *	120 *

*Limits for sidestick controllers have not been established. However, the forces must be acceptable to the pilot.

3.2.9.7.4 Pitch axis control force limits — sideslips. With the aircraft trimmed for straight, level flight with zero sideslip, the pitch-control force required to maintain constant speed in steady sideslips with up to 50 pounds of pedal force in either direction, or in sideslips as specified in the Operational Flight Envelope, 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 10 pounds pull or 3 pounds push for centerstick controllers. For sidestick controllers the forces must be acceptable to the pilot. If a variation of pitch-control force with sideslip does exist, it is preferred that increasing pull force accompany increasing sideslip, and that the magnitude and direction of the force change be similar for right and left sideslips. For Level 3 there shall be no uncontrollable pitching motions associated with the sideslips discussed above.

3.2.9.7.5 [Reserved]

3.2.9.7.6 Pitch axis control force limits — failures. The change in longitudinal control force required to maintain trim pitch attitude following complete or partial failure of the augmentation system shall not exceed 20 pounds for at least 5 seconds following the failure.

3.2.9.7.7 Pitch axis control force limits — configuration or control mode change. The control force changes resulting from the intentional engagement or disengagement of any portion of the primary flight control system by the pilot shall not exceed 20 pounds for at least 5 seconds following the mode change.

3.2.9.8 Pitch axis trim systems. In straight flight, throughout the Operational Flight Envelope the trimming system shall be capable of reducing the steady-state control forces to zero for Level 1 or 2, and no greater than 20 pounds (push or pull) for Level 3. The failures to be considered in applying Level 2 and 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.

3.2.9.8.1 Pitch axis trim systems — rate of 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.

3.2.9.8.2 Pitch axis trim systems — stalling of trim systems. Stalling of a trim system due to aerodynamic loads during maneuvers shall not result in an unsafe condition. Specifically, the longitudinal trim system shall be capable of operating during the dive recoveries of 3.2.9.7.3 at any attainable permissible n, at any possible position of the trimming device.

3.2.9.8.3 Pitch axis trim systems — 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, provision shall be made to ensure the accurate return of the device to its initial trim position on removal of each interconnect or augmentation command.

3.2.10 Pitch Axis Control Displacements

3.2.10.1 Pitch axis control displacements — takeoff. With the trim setting optional but fixed, the pitch-control travel during all types of takeoffs for which the aircraft is designed shall not exceed 75 percent of the total travel, stop-to-stop. Here the term takeoff includes ground run, rotation and liftoff, and the ensuing acceleration to V_{max} (TO). 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).

3.2.10.2 Pitch axis control displacements — maneuvering. For all types of pitch controllers, the control motions in maneuvering flight shall not be so large or so small as to be objectionable. In steady turning flight and in pullups at constant speed, the incremental 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 those required to initiate the change.

3.2.10.3 Pitch axis control displacements — gust regulation. The ability of the aircraft to perform operational maneuvers required of it shall not be limited in the Moderate atmospheric disturbances defined in 3.9 by control displacement or control surface deflection rates. 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 when establishing compliance with this requirement.

3.3 HANDLING QUALITY REQUIREMENTS FOR VERTICAL FLIGHT PATH AXIS

3.3.1 Vertical Axis Response to Attitude Change

3.3.1.1 Vertical axis response to attitude change — transient response. The short-term flight path response to attitude changes shall be acceptable to the pilot.

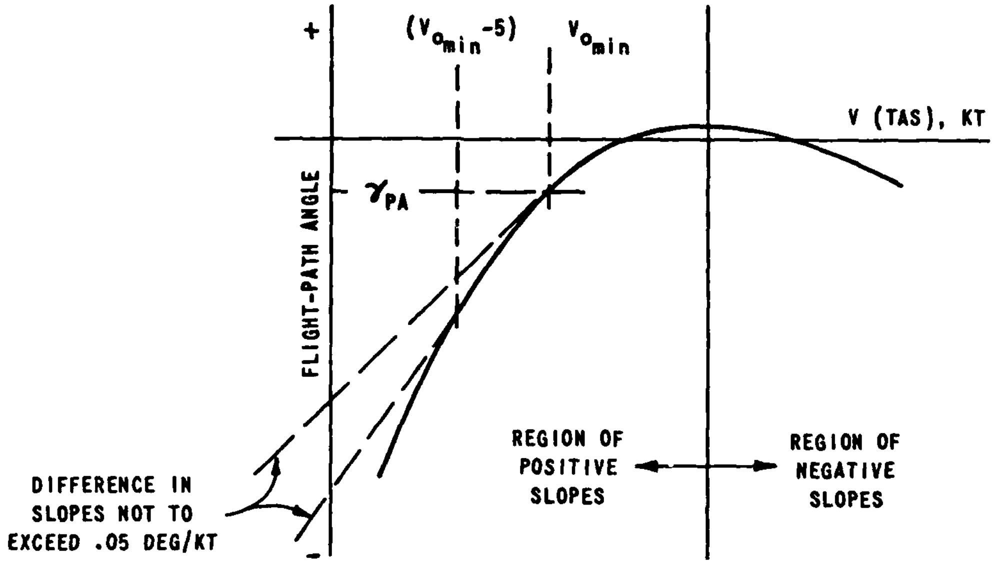
3.3.1.2 Vertical axis response to attitude change — steady-state response. The steady-state path response to attitude inputs shall be as follows. For the landing approach Flight Phase, the curve of flight-path angle versus true airspeed shall have a local slope at $V_{o_{min}}$ that is negative or less positive than:

a. Level 1: 0.06 degrees/knot

b. Level 2: 0.15 degrees/knot

c. Level 3: 0.24 degrees/knot

The thrust setting shall be that required for the normal approach glide path at $V_{o_{min}}$. The slope of the curve of flight-path angle versus airspeed at 5 knots slower than $V_{o_{min}}$ shall not be more than 0.05 degrees per knot more positive than the slope at $V_{o_{min}}$, as illustrated by the sketch on the following page.



[The specific wording of this requirement makes it clear that a STOL flight path control is disallowed — i.e., the $\partial\gamma/\partial V$ values are not consistent with an aircraft operating well on the backside of the power curve to the point where the primary flight path controller would be power].

3.3.3 Vertical Axis Response to Other Inputs

3.3.3.1 Vertical axis response to auxiliary controls, stores release, and armament. There shall be no objectionable transients in flight path response due to the use of other auxiliary controls, or stores or armament release.

3.3.3.2 Vertical axis response to failures. No single failure of any component or system shall result in objectionable flying qualities.

3.4 HANDLING QUALITY REQUIREMENTS FOR LONGITUDINAL (SPEED) AXIS

3.4.1 Speed Response to Attitude Changes

a. The correlation between airspeed and pitch attitude shall be as follows:

- Transient Response. For rapid attitude changes the short-term airspeed change shall be in the same direction as the final value.

- Steady State Response. For a fixed positive change in attitude from trim, airspeed shall not increase. This applies over a speed range of ±15 percent about trim or ±50 kt, whichever is less.

b. For Levels 1 and 2 there shall be no tendency for the airspeed to diverge aperiodically when the aircraft pitch attitude is disturbed from trim by any means. This requirement shall be considered satisfied if the gradient of pitch control force with airspeed is negative. Demonstration of positive phugoid damping in Paragraph 3.2.1 shall also be accepted as evidence of compliance.

c. For Level 3, the airspeed divergence characteristics must be within the following limits: the time for airspeed to double amplitude following a pitch attitude disturbance from trim shall not be less than 6 seconds. Additionally, airspeed divergences shall not be allowed in the presence of one or more other Level 3 flying qualities unless the flight safety of that combination of characteristics can be demonstrated.

3.4.1.1 Speed response to attitude changes — relaxation in transonic flight. The requirements of 3.4.1 may be relaxed in the transonic speed range as follows (provided any divergent aircraft motions with speed are gradual and not objectionable to the pilot):

a. Levels 1 and 2: For centerstick controllers, no local force gradient shall be more unstable than 3 pounds per 0.01 M nor shall the force change exceed 10 pounds in the unstable direction.

b. Level 3: For centerstick controllers, no local force gradient shall be more unstable than 6 pounds per 0.01 M nor shall the force ever exceed 20 pounds in the unstable direction.

For sidestick controllers any local gradient shall be acceptable to the pilot. This relaxation shall not apply to Level 1 for any Flight Phase which requires prolonged transonic operation.

3.5 HANDLING QUALITY REQUIREMENTS FOR ROLL AXIS

3.5.1 Roll Response to Roll Controller

3.5.1.1 Roll axis equivalent system requirements

3.5.1.1.1 Roll mode. The equivalent roll mode time constant, T_R, shall be no greater than the following:

FLIGHT PHASE CATEGORY	LEVEL		
	1	2	3
A & C	1.0	1.4	
B	1.4	3.0	1.0

The lower order equivalent roll rate transfer function is defined as follows:

$$\frac{p}{F_{as}} = \frac{K_p(0)[\zeta_\phi, \omega_\phi]e^{-\tau_{e_p} s}}{(1/T_s)(1/T_R)[\zeta_d, \omega_d]} \tag{1}$$

The equivalent system is to be obtained from matching of the higher order system response over a frequency range of approximately 0.1 to 10 rad/sec.

3.5.1.1.2 Spiral stability. The combined effects of spiral stability, flight-control-system characteristics and rolling moment change with speed shall be such that the bank angle response shall have the following characteristics following a disturbance in bank of up to 20 degrees. This requirement shall be met with the airplane trimmed for wings-level, zero-yaw-rate flight with the cockpit controls free.

MINIMUM TIME TO DOUBLE AMPLITUDE

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

The spiral mode may be determined from the equivalent system match of 3.5.1.1.1 or from time histories taken from flight test.

3.5.1.1.3 Coupled roll-spiral oscillation. A coupled roll-spiral mode will be permitted provided it has the following characteristics:

MAXIMUM VALUES FOR ROLL-SPIRAL DAMPING COEFFICIENT, $\zeta_{RS}\omega_{RS}$

LEVEL	CATEGORY A	CATEGORIES B AND C**
1	*	0.5
2	*	0.3
3	*	0.15

*The aircraft shall not exhibit a coupled roll-spiral mode in Category A Flight Phases.

**The aircraft shall not exhibit a coupled roll-spiral oscillation in Category C Flight Phases requiring rapid turning maneuvers such as short approaches.

3.5.1.1.4 Roll rate oscillations. The value of the parameter p_{osc}/p_{av} following a yaw-control-free step roll command shall be within the limits of Figure 4. This requirement applies for step roll commands up to the magnitude that causes a 60 degree bank angle change in $1.7T_d$ seconds.

3.5.1.1.5 Time delay. The value of the equivalent time delay, τ_{e_p}, determined in the equivalent system match of 3.5.1.1.1, shall be no greater than the following:

LEVEL	ALLOWABLE DELAY (sec)
1	0.10
2	0.20
3	0.25

3.5.2 Pilot-Induced Roll Oscillations. There shall be no tendency for sustained or uncontrollable roll oscillations resulting from efforts of the pilot to control the airplane.

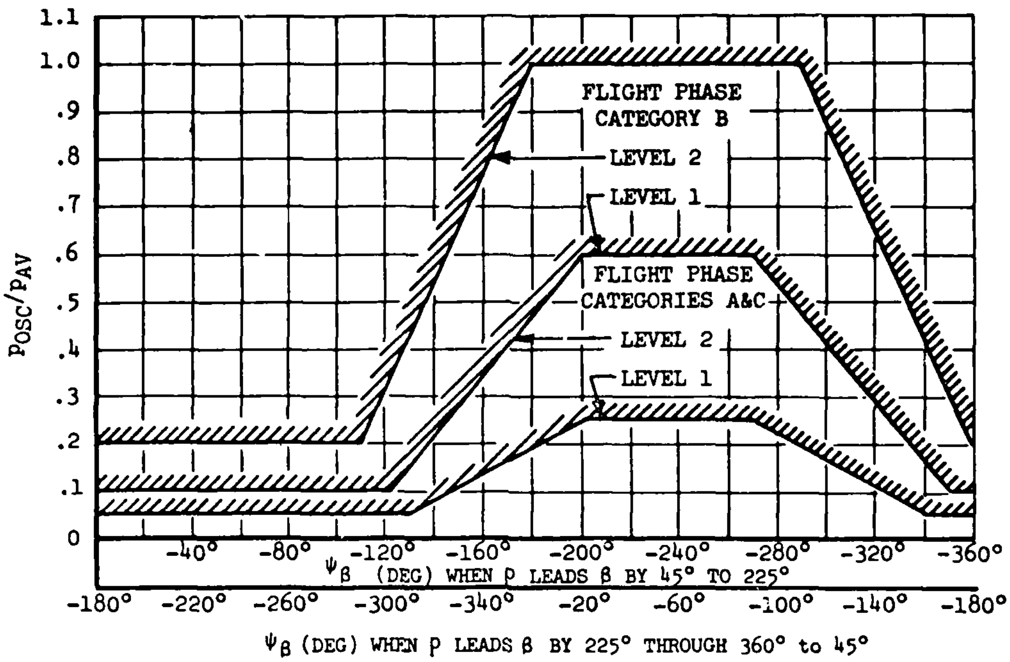


Figure 4 (Appendix B). Roll Rate Oscillation Limitations

3.5.3 Residual Roll Oscillations. Any sustained residual oscillations in calm air shall not interfere with the pilot's ability to perform the tasks required in service use of the airplane.

3.5.4 Linearity of Roll Response to Roll Controller. There shall be no objectionable nonlinearities in the variation of rolling response with roll control deflection or force. Sensitivity or sluggishness in response to small control deflections or force shall be avoided.

3.5.5 Lateral Acceleration at Pilot Station. The ratio of maximum lateral acceleration at the pilot station to maximum roll rate shall not exceed the following for the first 2-1/2 seconds following a step roll control input.

Level	$n_{y_{pilot_{max}}}/p_{max}$ (g/deg/sec)
1	0.012
2	0.035
3	0.058

3.5.6 Roll response to yaw controller. The following requirements are expressed in terms of characteristics in yaw-control-induced steady, zero-yaw-rate sideslips with the airplane 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 pounds of yaw-control-pedal force, or

c) Maximum roll control or surface deflection.

At these sideslip angles the following shall apply:

a) A decrease in right bank angle shall not accompany an increase in right sideslip, and a decrease in left bank angle shall not accompany an increase in left sideslip. Zero roll control force or deflection is acceptable, whereas

b) A right roll-control deflection and/or force shall not accompany left sideslips, and a left roll-control deflection and/or force shall not accompany right 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 50 percent of roll-control power available to the pilot, and no more than 10 pounds of roll-control force are required in a direction opposite to that specified herein. In addition, 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 75 percent of roll-control power available to the pilot, and no more than 10 pounds of roll-stick force are required for sideslip angles that might be experienced in service employment.

3.5.7 Roll axis control for takeoff and landing in crosswinds. It shall be possible to take off and land with normal pilot skill and technique in 90 deg crosswinds from either side of velocities up to 30 kt for Levels 1 and 2, and 15 kt for Level 3.

3.5.8 Roll Axis Response to Other Inputs

3.5.8.1 Roll axis response to asymmetric thrust. The airplane motions following sudden asymmetric loss of thrust shall be such that dangerous conditions can be avoided by pilot corrective action. A realistic time delay of at least 1 second shall be considered.

3.5.8.2 Roll axis response to failures.

a) Closed-Loop: The aircraft motions following sudden aircraft system or component failures shall be such that dangerous conditions can be avoided by pilot corrective action. A time delay of at least 1 sec between the failure and initiation of pilot corrective action shall be incorporated when determining compliance. No single failure of any component or system shall result in Level 3 flying qualities; Special Failures States (1.6.3) are excepted. The crew member concerned shall be provided with immediate and easily interpreted indications whenever failures occur that require or limit any flight crew action or decision.

b) Open-Loop: With controls free, the aircraft motions due to partial or complete failure of the augmentation system shall not exceed the following limits for at least 2 seconds following the failure:

- Levels 1 and 2 (after failure): ±0.5 g incremental lateral acceleration at the pilot's station and ±10 deg per second roll rate, except that neither stall angle of attack nor structural limits shall be exceeded. In addition, for Category A, ±2 deg bank angle.

- Level 3 (after failure): No dangerous attitude or structural limit is reached, and no dangerous alteration of the flight path results from which recovery is impossible.

3.5.8.3 Roll axis response to configuration or control mode change. The transient motions and trim changes resulting from the intentional engagement or disengagement of any portion of the primary flight control system by the pilot shall be such that dangerous flying qualities never result. With controls free, the motion transients resulting from these situations shall not exceed the following limits for at least 2 seconds following the transfer: ±3 deg/sec roll within the Operational Flight Envelope, or ±5 deg/sec roll within the Service Flight Envelope. These requirements apply only for Aircraft Normal States.

3.5.8.4 Roll axis response to stores release. The intentional release of any stores shall not result in objectionable flight characteristics for Levels 1 and 2. However, the intentional release of stores shall never result in dangerous or intolerable flight characteristics. This requirement applies for all flight conditions and store loadings at which normal or emergency store release is structurally permissible.

3.5.8.5 Roll axis response to armament delivery. Operation of moveable parts or firing of weapons shall not cause buffet, trim changes, or other characteristics which impair the tactical effectiveness of the airplane under any pertinent flight conditions. These requirements shall be met for Levels 1 and 2.

3.5.9 Roll Axis Control Power

3.5.9.1 Roll axis control power — response to roll control inputs. The response to full roll control input shall have the following characteristics. Over the equivalent airspeed ranges listed below, roll performance is specified in Table 11. In addition, roll performance for Flight Phase CO is specified in Table 12 for 360 deg rolls initiated at 1 g, and in Table 13 for rolls initiated at load factors between 0.8 $n_o(-)$ and 0.8 $n_o(+)$. These requirements take precedence over Table 11.

Speed Range Symbol	Equivalent Airspeed Range	
	For Level 1	For Levels 2 and 3
VL	$V_{o_{min}} < V < V_{min} + 20$ KTS	$V_{min} < V < V_{min} + 20$ KTS
L	$V_{min} + 20$ KTS[a] $< V < 1.4\ V_{min}$	$V_{min} + 20$ KTS $< V < 1.4\ V_{min}$
M	$1.4\ V_{o_{min}} < V < 0.7\ V_{max}$ [b]	$1.4\ V_{min} < V < 0.7\ V_{max}$
H	$0.7\ V_{max}$ [b] $< V < V_{o_{max}}$	$0.7\ V_{max} < V < V_{max}$

[a] Or $V_{o_{min}}$, whichever is greater [b] Or $V_{o_{max}}$, whichever is less

3.5.9.2 Roll axis 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 75 percent of roll control power available to the pilot is required for sideslips which might be encountered in service deployment.

TABLE 11 (Appendix B)
ROLL PERFORMANCE
Time to Achieve the Following Bank Angle Change (Seconds)

LEVEL	SPEED RANGE	CATEGORY A			CATEGORY B	CATEGORY C
		30 deg	50 deg	90 deg	90 deg	30 deg
1	VL	1.1			2.0	1.1
	L	1.1			1.7	1.1
	M			1.3	1.7	1.1
	H		1.1		1.7	1.1
2	VL	1.6			2.8	1.3
	L	1.5			2.5	1.3
	M			1.7	2.5	1.3
	H		1.3		2.5	1.3
3	VL	2.6			3.7	2.0
	L	2.0			3.4	2.0
	M			2.6	3.4	2.0
	H		2.6		3.4	2.0

TABLE 12 (Appendix B)
FLIGHT PHASE CO ROLL PERFORMANCE IN 360 DEG ROLLS
Time to Achieve the Following Bank Angle Change (Seconds)

LEVEL	SPEED RANGE	30 deg	90 deg	180 deg	360 deg
1	VL	1.0			
	L		1.4	2.3	4.1
	M		1.0	1.6	2.8
	H		1.4	2.3	4.1
2	VL	1.6			
	L	1.3			
	M		1.3	2.0	3.4
	H		1.7	2.6	4.4
3	VL	2.5			
	L	2.0			
	M		1.7	3.0	
	H		2.1		

TABLE 13 (Appendix B)
FLIGHT PHASE CO ROLL PERFORMANCE
Time to Achieve the Following Bank Angle Change (Seconds)

LEVEL	SPEED RANGE	30 deg	50 deg	90 deg	180 deg
1	VL	1.0			
	L		1.1		
	M			1.1	2.2
	H		1.0		
2	VL	1.6			
	L	1.3			
	M			1.4	2.8
	H		1.4		
3	VL	2.5			
	L	2.0			
	M			1.7	3.4
	H		1.7		

3.5.9.3 Roll axis control power in crosswinds.

a) It shall be possible to taxi at any angle to a 45 kt wind.

b) Roll control power, in conjunction with other normal means of control, shall be adequate to maintain a straight path during the takeoff run, or landing rollout, in crosswinds up to those specified in 3.5.7.

c) Roll control power shall be adequate to maintain wings level with up to 10 deg of sideslip in the power approach. For Level 1 this shall require not more than 75 percent of the control power available to the pilot.

d) Following sudden asymmetric loss of thrust from any factor, the airplane shall be safely controllable in roll in the crosswinds of 3.5.7 from the unfavorable direction.

3.5.9.4 Roll axis control power for engine failure. During the takeoff run it shall be possible to maintain roll control of the aircraft, following a sudden loss of thrust from the most critical propulsive source. This requirement shall apply from a minimum speed of $V_{min}(TO)$ to a maximum speed of $V_{max}(TO)$.

The roll control required shall not exceed 75 percent of the available roll control power. This assumes takeoff thrust is maintained on the operative engine for multi-engine aircraft, with trim at normal setting for symmetric thrust. The aircraft may be banked up to 5 deg away from the inoperative engine. [The wording of this paragraph implies that the aircraft may be either single- or twin-engine].

3.5.9.5 Roll axis control power in dives and pullouts. Roll control power shall be adequate to maintain wings level without retrimming, throughout the dives and pullouts of 3.2.9.7.3.

3.5.9.6 Roll axis control power for stores release. Roll control power shall be adequate to regain wings level, without retrimming, following intentional release of any stores, to the maximum load factors specified in 3.2.8.2 with adequate control margin.

3.5.9.8 Roll axis control power for other conditions. Control authority, rate and hinge moment capability shall be sufficient to assure safety throughout the combined range of all attainable angles of attack (both positive and negative) and sideslip. This requirement applies to the prevention of loss of control and to recovery from any situation for all maneuvering, including pertinent effects of factors such as regions of control-surface-fixed instability, inertial coupling, fuel slosh, the influence of symmetric and asymmetric stores, stall/post-stall/spin characteristics, atmospheric disturbances and Aircraft Failure States (maneuvering flight appropriate to the Failure State is to be included). Consideration shall be taken of the degrees 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.

3.5.10 Roll Axis Control Forces and Displacements

3.5.10.2 Roll axis control forces to achieve required roll rates. The roll control force required to obtain the rolling performance specified in 3.5.9.1 shall be neither less than the minimum nor greater than the maximum listed below for centerstick controllers. For sidestick controllers, forces shall not be so large or so small as to be objectionable to the pilot.

	FORCE (lbs)	
LEVEL	MINIMUM*	MAXIMUM
1	5	20
2	4	30
2 (Cat C)	2-1/2	20
3	0	35

*Above breakout force.

3.5.10.3 Roll axis control sensitivity. The roll control force gradient shall have the following characteristics. In case of conflict between the requirements of 3.5.10.3 and 3.5.10.2, the requirements of 3.5.10.3 shall govern.

LEVEL	FLIGHT PHASE CATEGORY	MAXIMUM SENSITIVITY (deg in 1 sec)/lb
1	A	15.
	C	7.5
2	A	25.
	C	12.5

3.5.10.4 Roll axis control forces — control centering and breakout forces. Lateral controls should exhibit positive centering in flight at any normal trim conditions.

The combined effects of centering, breakout force, damping, and force gradient shall not produce objectionable flight characteristics.

Breakout forces, including friction, preload, etc., shall be within the following limits: for Levels 1 and 2, 1/2 lb to 2 lb (1 lb for side-stick controllers); for Level 3, 1/2 lb to 4 lb.

3.5.10.5 Roll axis control forces — free play. The free play in the lateral controller shall not result in objectionable flight characteristics, especially for small amplitude inputs.

3.5.10.6 Roll axis control force limits

3.5.10.6.1 Roll axis control force limits — steady turns. It shall be possible to maintain steady turns with the airplane trimmed for wings-level straight flight in either direction with the yaw controls free at the following combinations of bank angle and roll controller force characteristics: 60 deg and 5 lb.

3.5.10.6.2 Roll axis control force limits — dives and pullouts. Roll control forces shall not exceed 10 lb in dives and pullouts to the maximum speeds specified in the Service Flight Envelope.

3.5.10.6.3 Roll axis control force limits — crosswinds. It shall be possible to take off and land in the crosswinds specified in 3.5.9.3 without exceeding the roll control forces specified by 3.5.10.2.

3.5.10.6.4 Roll axis control force limits — steady sideslips. In final approach the roll control forces shall not exceed 10 lb for Level 1 or 20 lb for Level 2 when in a straight, steady sideslip of 10 deg.

3.5.10.6.5 Roll axis control force limits — engine failures after takeoff. Following a thrust loss from the most critical factor after takeoff the roll control forces shall not exceed those specified by Paragraph 3.5.10.2. With yaw controls free, forces shall be the Level 2 upper limits specified in 3.5.10.2 for Levels 1 and 2, and the Level 3 upper limits for Level 3. These requirements apply with takeoff thrust maintained on the operative engine for multi-engine aircraft and trim at the normal settings for takeoff with symmetric thrust. Automatic devices that normally operate in the event of a thrust failure may be used, and the airplane may be banked up to 5 degrees away from the inoperative engine.

3.5.10.6.6 Roll axis control force limits — configuration or control mode change. The control force changes resulting from the intentional engagement or disengagement of any portion of the primary flight control system by the pilot shall not exceed 10 lb for at least 5 sec following the mode change.

3.6 HANDLING QUALITY REQUIREMENTS FOR YAW AXIS

3.6.1 Yaw Axis Response to Yaw Controller

3.6.1.1 Yaw axis equivalent system requirements

3.6.1.1.1 Dynamic response. The equivalent parameters describing the response of sideslip to a yaw control input shall have the characteristics specified in Table 14. The requirements shall be met in trimmed and in maneuvering flight with cockpit controls fixed and with them free, in oscillations of any magnitude that might be experienced in operational use. If the oscillation is nonlinear with amplitude, the requirement shall apply to each cycle of the oscillation. In calm air residual oscillations may be tolerated only if the amplitude is sufficiently small that the motions are not objectionable and do not impair mission performance.

The parameters shall be found by matching the higher-order sideslip response to yaw control input to the following lower-order form, over the frequency range from 0.1 rad/sec to 10 rad/sec:

$$\frac{\beta}{F_{rp}} = \frac{K_\beta e^{-\tau_{e_\beta} s}}{[s^2 + 2\zeta_d \omega_d s + \omega_d^2]}$$

TABLE 14 (Appendix B). MINIMUM DUTCH ROLL FREQUENCY AND DAMPING

LEVEL	FLIGHT PHASE CATEGORY	Min ζ_d*	Min $\zeta_d\omega_d$* (rad/sec)	Min ω_d (rad/sec)
1	A (CO)	0.4	0.4	1.0
	A	0.19	0.35	1.0
	B	0.08	0.15	0.4
	C	0.08	0.15	1.0
2	All	0.02	0.05	0.4
3	All	0	—	0.4

*The governing damping requirement is that yielding the larger value of ζ_d.

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 14 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.

3.6.1.1.2 Steady-state response. The long-term response to yaw-control-pedal deflections shall have the following characteristics:

Right yaw-control-pedal force shall produce left sideslips and left yaw-control-pedal force shall produce right sideslips. For Levels 1 and 2 the following requirements shall apply. The variation of sideslip angle with yaw-control-pedal force shall be essentially linear for sideslip angles between +15 degrees and -15 degrees. For larger sideslip angles, an increase in yaw-control-pedal force shall always be required for an increase in sideslip.

This requirement applies to yaw-control-induced steady, zero-yaw-rate sideslips with the airplane 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 pounds of yaw-control-pedal force, or

c) Maximum roll control or surface deflection.

3.6.2 Yaw Axis Response to Roll Controller

3.6.2.1 Coordination in turn entry and exit. The sideslip excursions to step roll control inputs with yaw control free shall meet the following critera:

a. The amount of sideslip following a yaw-control-free (small input) step roll control command shall be within the limits as shown in Figure 5 for Levels 1 and 2. This requirement shall apply for step roll control commands up to the magnitude that causes a 60 degree bank angle change with T_d or 2 seconds, whichever is longer.

b. Following a yaw-control-free (large input) step roll control command, the ratio of the sideslip increment, $\Delta\beta$, to the parameter k shall be less than the values specified below. The roll command shall be held fixed until the bank angle has changed at least 90 degrees.

LEVEL	FLIGHT PHASE CATEGORY	ADVERSE SIDESLIP (RIGHT ROLL COMMAND CAUSES RIGHT SIDESLIP)	PROVERSE SIDESLIP (LEFT ROLL COMMAND CAUSES RIGHT SIDESLIP)
1	A B and C	6 degrees 10 degrees	2 degrees 3 degrees
2	All	15 degrees	4 degrees

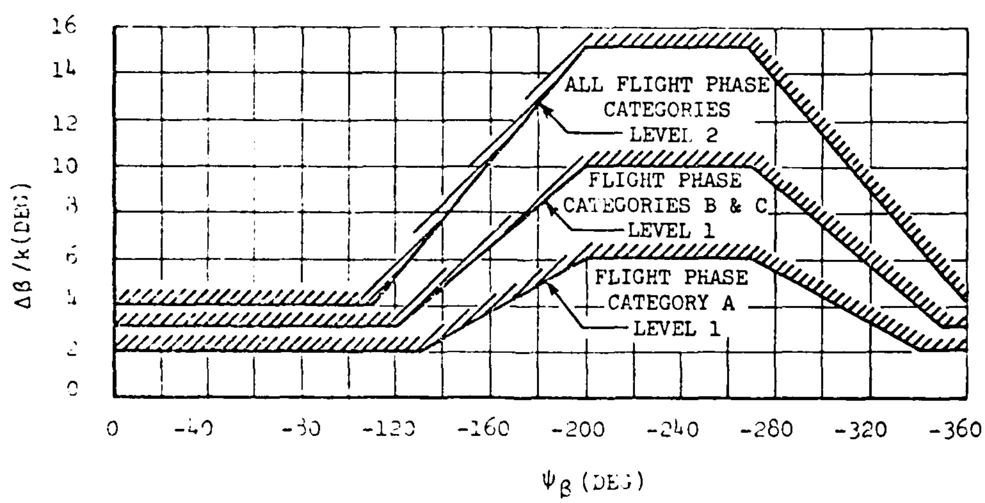


Figure 5 (Appendix B). Sideslip Excursion Limitations

3.6.2.2 Pilot-induced yaw oscillations. There shall be no tendency for sustained or uncontrollable yaw oscillations resulting from efforts of the pilot to control the aircraft.

3.6.2.3 Residual yaw oscillations. Any sustained residual oscillations in calm air shall not interfere with the pilot's ability to perform the tasks required in service use of the aircraft.

3.6.3.1 Yaw Axis Control for Takeoff and Landing in Crosswinds. It shall be possible to take off and land with normal pilot skill and technique in 90 deg crosswinds from either side of velocities up to 30 kt for Levels 1 and 2 or 15 kt for Level 3.

3.6.4 Yaw Axis Response to Other Inputs

3.6.4.1 Yaw axis response to asymmetric thrust. For multi-engine aircraft, it shall be possible for the pilot to maintain directional control of the aircraft following a loss of thrust from the most critical propulsive source.

a) Takeoff: During takeoff it shall be possible to maintain a straight path without deviations of more than 30 ft. 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 airplane is designed to operate) to the maximum takeoff speed, with takeoff thrust maintained on the operative engine, using only controls not dependent upon friction against the takeoff surface or 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 nose-wheel steering and differential braking may be used. Automatic devices that normally operate in the event of a thrust failure may be used in either case.

b) After takeoff: After takeoff it shall be possible without a change in selected configuration to achieve straight flight following sudden asymmetric loss of thrust from the most critical factor at speeds from V_{min} (TO) to V_{max} (TO), and thereafter to maintain straight flight throughout the climbout. Automatic devices that normally operate in the event of a thrust failure may be used, and the airplane may be banked up to 5 degrees away from the inoperative engine.

c) Takeoff and landing in crosswinds: The aircraft shall be safely controllable in the crosswinds of 3.6.3 from the unfavorable direction.

d) In-flight: The airplane motions following sudden asymmetric loss of thrust shall be such that dangerous conditions can be avoided by pilot corrective action. A realistic time delay of at least 1 second shall be incorporated. In addition, the static directional stability shall be such that at all speeds above 1.4 V_{min} with asymmetric loss of thrust from the most critical factor while the other engine develop normal rated thrust, the airplane with yaw control pedals free may be balanced directionally in steady straight flight. The trim settings shall be those required for wings-level straight flight prior to the failure.

[Note that this paragraph is intended only for multi-engine aircraft. It does not, however, prevent the contractor from designing the candidate aircraft with only a single engine.]

3.6.4.2 Yaw axis response to failures. The yawing motions following sudden airplane system or component failures shall be such that dangerous conditions can be avoided by pilot corrective action. A realistic time delay between the failure and initiation of pilot corrective action shall be incorporated when determining compliance. No single failure of any component or system shall result in dangerous or intolerable flying qualities; Special Failure States (1.6.3) are excepted. The crew member concerned shall be provided with immediate and easily interpreted indications whenever failures occur that require or limit any flight crew action or decision. With controls free, the yawing motions due to failures shall not exceed the following limits for at least 2 seconds following the failure:

- Levels 1 and 2 (after failure) — ±0.5 g incremental lateral acceleration at the pilot's station, except that structural limits shall not be exceeded. In addition, for Category A, lateral excursions of 5 ft.

- Level 3 (after failure) — No dangerous attitude or structural limit is reached, and no dangerous alteration of the flight path results from which recovery is impossible.

3.6.4.3 Yaw axis response to configuration or control mode change. The transient motions and trim changes resulting from the intentional engagement or disengagement of any portion of the primary flight control system by the pilot shall be such that dangerous flying qualities never result. With controls free, the transients resulting from these situations shall not exceed the following limits for at least 2 seconds following the transfer: the lesser of ±5 deg sideslip and the structural limit. These requirements apply only for Aircraft Normal States, within the Service Flight Envelope.

3.6.4.4 Yaw axis response to stores release. The intentional release of any stores shall not result in objectionable flight characteristics for Levels 1 and 2. However, the intentional release of stores shall never result in dangerous or intolerable flight characteristics. This requirement applies for all flight conditions and store loadings at which normal or emergency store release is structurally permissible.

3.6.4.5 Yaw axis response to armament delivery. Operation of movable parts or firing of weapons shall not cause buffet, trim changes, or other characteristics which impair the tactical effectiveness of the aircraft under any pertinent flight conditions. These requirements shall be met for Levels 1 and 2.

3.6.5 Yaw Axis Control Power. Directional stability and control characteristics shall enable the pilot to balance yawing moments and control yaw and sideslip.

3.6.5.1 Yaw axis control power for takeoff, landing, and taxi.

a) It shall be possible to taxi on a dry surface at any angle to a 45 kt wind.

b) In taxi on wet, snow-packed, or icy runways, directional control shall be maintained by use of aerodynamic controls alone at all airspeeds above 50 kt. For very slippery runways, the requirement need not apply for crosswind components at which the force tending to blow the airplane off the runway exceeds the opposing tire-runway frictional force with the tires supporting all of the airplane's weight.

c) In the takeoff run, landing rollout, and taxi, yaw control power shall be adequate to maintain a straight path on the ground or other landing surface. This applies to calm air and in crosswinds up to the values specified in 3.6.3, on wet runways for all aircraft, and on snow-packed and icy runways for aircraft intended to operate under such conditions.

d) Yaw axis control power shall be adequate to develop 10 deg of sideslip in the power approach.

3.6.5.3 Yaw axis control power with asymmetric loading. When initially trimmed directionally with each asymmetric loading specified in Paragraph 3.1.1 at any speed in the Operational Flight Envelope, yaw control power shall be sufficient to maintain a straight flight path.

3.6.5.4 Yaw axis control power for stores release. Yaw control power shall be adequate to regain straight flight, without retrimming, following intentional release of any stores to the maximum load factors specified in 3.2.8.2 with adequate control margin.

3.6.5.5 Yaw axis control power for other conditions. Control authority, rate and hinge moment capability shall be sufficient to assure safety throughout the combined range of all attainable angles of attack (both positive and negative) and sideslip. This requirement applies to the prevention of loss of control and to recovery from any situation for all maneuvering, including pertinent effects of factors such as regions of control-surface-fixed instability, inertial coupling, fuel slosh, the influence of symmetric and asymmetric stores, stall/post-stall/spin characteristics, atmospheric disturbances and Aircraft Failure States (maneuvering flight appropriate to the Failure State is to be included). Consideration shall be taken of the degrees 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.

3.6.6 Yaw Axis Control Forces. Sensitivity to yaw control pedal forces shall be sufficiently high that directional control and force requirements can be met and 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.

3.6.6.1 Yaw axis control force linearity. The following requirements are expressed in terms of characteristics in yaw-control-induced steady, zero-yaw-rate sideslips with the airplane 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 pounds of yaw-control-pedal force, or

c) Maximum roll control or surface deflection.

Right yaw-control-pedal force shall produce left sideslips and left yaw-control-pedal force shall produce right sideslips. For Levels 1 and 2 the following requirements shall apply. The variation of sideslip angle with yaw-control-pedal force shall be essentially linear for sideslip angles between +10 degrees and -10 degrees. Although a lightening of pedal force is acceptable for sideslip angles outside this range, the pedal force shall never reduce to zero.

3.6.6.2 Yaw axis control force limits

3.6.6.2.1 Yaw axis control force limits in rolling maneuvers. In the maneuvers described in 3.5.9, directional-control effectiveness shall be adequate to maintain zero sideslip with pedal force not greater than 50 lb in Flight Phase Category A, Level 1, and 100 lb for all other combinations of Flight Phase Category and Level.

3.6.6.2.2 Yaw axis control force limits in steady turns. It shall be possible to maintain steady coordinated turns in either direction, using 60 deg of bank with a pedal force not exceeding 40 lb, with the airplane trimmed for wings-level straight flight. These requirements constitute Levels 1 and 2.

3.6.6.2.3 Yaw axis control force limits during 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 percent of the trim speed or ±100 kt equivalent airspeed, whichever is less (except where limited by boundaries of the Service Flight Envelope) with yaw-control-pedal forces not greater than 40 lb for Levels 1 and 2, or 180 lb for Level 3, without retrimming.

3.6.6.2.4 Yaw axis control force limits in crosswinds. It shall be possible to take off and land in the crosswinds specified in 3.6.3 without exceeding the following yaw control forces: 100 lb for Level 1 and 180 lb for Levels 2 and 3.

3.6.6.2.5 Yaw axis control force limits with asymmetric loading. When initially trimmed directionally with each asymmetric loading specified in Paragraph 3.1.1 at any speed in the Operational Flight Envelope, it shall be possible to maintain a straight flight path throughout the Operational Flight Envelope with yaw-control-pedal forces not greater than 100 lb for Levels 1 and 2 or 180 lb for Level 3 without retrimming.

3.6.6.2.6 Yaw axis control force limits in dives and pullouts. Throughout the dives and pullouts of 3.2.9.7.3, yaw-control-pedal forces shall not exceed 50 lb in dives and pullouts to the maximum speeds specified in the Service Flight Envelope.

3.6.6.2.7 Yaw axis control force limits for go-around. The response to thrust, configuration and airspeed change shall be such that the pilot can maintain straight flight during go-around initiated at speeds down to V_S (PA) with yaw-control-pedal forces not exceeding 40 lb when trimmed at $V_{o_{min}}$ (PA). The preceding requirements apply for Levels 1 and 2. The Level 3 requirement is to maintain straight flight in these conditions with yaw-control-pedal forces not exceeding 180 lb. Bank angles up to 5 deg are permitted for all Levels.

3.6.6.2.8 Yaw axis control force limits for asymmetric thrust during takeoff. For multi-engine aircraft the following requirements apply.

a) During the takeoff ground run it shall be possible to achieve and maintain a straight path on the takeoff surface without a deviation of more than 30 ft from the path originally intended, with yaw-control forces not exceeding 180 lb.

b) For the continued takeoff it shall be possible, without a change in selected configuration, to achieve straight flight following sudden asymmetric loss of thrust from the most critical propulsive source at speeds from V_{min} (TO) to V_{max} (TO), and thereafter to maintain straight flight throughout the climbout without exceeding a maximum yaw control pedal force of 180 lb.

c) For the aborted takeoff the requirements above 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 that normally operate in the event of a thrust failure may be used in either case.

3.6.6.2.9 Yaw axis control force limits with failures. The change in yaw control force required to maintain constant heading following a failure shall not exceed 50 lb for at least 5 seconds following the failure.

3.6.6.2.10 Yaw axis control force limits — configuration or control mode change. The change in yaw control force required to maintain zero sideslip following intentional engagement or disengagement of any portion of the primary flight control system by the pilot shall not exceed 10 lb above basic controller breakout force for at least 5 seconds following the mode change. These requirements apply only for Aircraft Normal States.

3.8 HANDLING QUALITY REQUIREMENTS FOR COMBINED AXES

3.8.1 Cross-Axis Coupling in Roll Maneuvers. In yaw-control-free, pitch-control-fixed, maximum-performance rolls through 360 deg, entered from straight flight or from turns, pushovers, or pullups ranging from 0 g to 0.8 n_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 combat-type maneuvers involving rolls through angles up to 360 degrees 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. These requirements define Level 1 and 2 operation.

3.8.2 Crosstalk Between Pitch and Roll Controllers. The pitch- and roll-control force and displacement sensitivities and breakout forces shall be compatible so that intentional inputs to one control axis will not cause inadvertent inputs to the other.

3.8.3 Control Harmony. The following control force levels are considered to be limiting values compatible with the pilot's capability to apply simultaneous forces.

Control Type	Pitch	Roll	Yaw
Sidestick	20 lb	15 lb	
Centerstick	50 lb	25 lb	
Pedal			175 lb

3.8.4 Flight at High Angle of Attack. The requirements of 3.8.4 through 3.8.4.3.2 concern stall warning, stalls, departures from controlled flight, post-stall gyrations, spins, recoveries, and related characteristics. They apply at speeds and angles of attack which in general are outside the Service Flight Envelope. They are intended to assure safety and the absence of mission limitations due to high-angle-of-attack characteristics.

3.8.4.1 Warning cues. Warning or indication of approach to stall, loss of aircraft control, and incipient spin shall be clear and unambiguous.

3.8.4.2 Stalls. The stall requirements apply for all Aircraft Normal States in straight unaccelerated flight and in turns and pullups with attainable normal accelerations up to n_L. Specifically, the Aircraft Normal States to be evaluated are those associated with the configurations, throttle settings and trim settings of 4.2.2. Also, the requirements apply to Aircraft Failure States that affect stall characteristics.

3.8.4.2.1 Stall approach

a) The onset of warning of stall approach (3.8.4.1) shall occur within the following speed range for 1 g stalls, and within the following range (or percentage) of lift for accelerated stalls, but not within the Operational Flight Envelope.

1 g Stalls:

Flight Phase	Minimum Speed for Onset	Maximum Speed for Onset
Approach	Higher of $1.05V_S$ or $V_S + 5$ knots	Higher of $1.10V_S$ or $V_S + 10$ knots
All other	Higher of $1.05V_S$ or $V_S + 5$ knots	Higher of $1.15V_S$ or $V_S + 15$ knots

Accelerated Stalls:

Flight Phase	Minimum Lift at Onset	Maximum Lift at Onset
Approach	82% C_L stall	90% C_L stall
All other	75% C_L stall	90% C_L stall

b) An increase in intensity of the warning with further increase in angle of attack shall be sufficiently marked to be noted by the pilot. The warning shall continue until the angle of attack is reduced to a value less than that for warning onset. Prior to the stall, uncommanded oscillations shall not result in flying qualities less than Level 2.

c) At all angles of attack 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.

3.8.4.2.2 Stall characteristics. The following apply for all stalls, including stalls entered abruptly:

a) In the unaccelerated stalls of 3.8.4.2.1, the aircraft shall not exhibit rolling, yawing, or downward pitching at the stall which cannot be controlled to stay within 30 deg.

b) It is desired that no pitchup tendencies occur in unaccelerated or accelerated stalls. However, in unaccelerated stalls, mild nose-up pitch may be acceptable if no pitch control force reversal occurs and if no dangerous, unrecoverable or objectionable flight conditions result. In accelerated stalls, mild nose-up tendency may be acceptable if the operational effectiveness of the airplane is not compromised and the airplane has adequate stall warning, pitch control effectiveness is such that it is possible to stop the pitchup promptly and reduce the angle of attack, and at no point during the stall, stall approach or recovery does any portion of the airplane exceed structural limit loads.

3.8.4.2.3 Stall prevention and recovery.

a) It shall be possible to prevent the stall by moderate use of the pitch control alone at the onset of the stall warning.

b) 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 those specified in 3.8.3, and to regain level flight without excessive loss of altitude or buildup 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.

c) In the straight flight stalls of 3.8.4.2, with the aircraft trimmed at an airspeed not greater than 1.4 V_s, pitch control power shall be sufficient to recover from any attainable angle of attack.

3.8.4.2.4 One-engine-out stalls. On multi-engine aircraft it shall be possible to recover safely from stalls with the critical engine inoperative. Thrust on the remaining engine will be as specified below.

Flight Phase	Thrust
TO	Takeoff
CL	Normal climb
PA	Normal approach
WO	Go-Around

3.8.4.3 Departures and Spins. The post-stall gyration and spin requirements apply to all modes of motion that can be entered from upsets, decelerations, and extreme maneuvers appropriate to the Class and Flight Phase Category. The requirements hold for all Aircraft Normal States and for all states of stability and control augmentation systems, except approved Special Failure States. Store release shall not be allowed during loss of control, spin or gyration, recovery, or subsequent dive pullout. Automatic disengagement of augmentation systems, however, is permissible if it is necessary and does not prevent meeting any other requirements; re-engagement shall be possible in flight following recovery. Specific flight conditions to be evaluated shall include entries from inverted flight. Entry angles of attack and sideslip up to maximum control capability and under dynamic flight conditions are to be included, except as limited by structural considerations. For all Flight Phase Categories, thrust settings up to and including MAT shall be included, with and without one critical engine inoperative at entry.

3.8.4.3.1 Departure from controlled flight. The aircraft shall be resistant to departure from controlled flight, post-stall gyrations and spins. Adequate warning of approach to departure (3.8.4.1) shall be provided. The airplane shall exhibit no uncommanded motion which cannot be arrested promptly by simple application of pilot control.

3.8.4.3.2 Recovery from post-stall gyrations and spins. For aircraft that, according to MIL-A-8861, must be structurally designed for spinning:

a) The proper recovery technique(s) must be readily ascertained by the pilot, and simple and easy to apply under the motions encountered.

b) A single technique shall provide prompt recovery from all post-stall gyrations and incipient spins, without requiring the pilot to determine the direction of motion and without tendency to develop a spin. The same technique used to recover from post-stall gyrations and incipient spins, or at least a compatible one, is also desired for spin recovery. For all modes of spin that can occur, these recoveries shall be attainable within the following number of turns for recovery or altitude loss, whichever is less, measured from the initiation of recovery action:

Flight Phase	Turns for Recovery	Altitude Loss*
PA	1	1000 ft
Category A, B	2	5000 ft

*Not including dive pullout.

c) Avoidance of a spin reversal or an adverse mode change shall not depend upon precise pilot control timing or deflection. It is desired that all aircraft be readily recoverable from all attainable attitudes and motions. The post-stall characteristics of those aircraft not required to comply with requirements of this paragraph shall be determined by analysis and model test.

d) Safe and consistent recovery and pullouts shall be accomplished without exceeding the forces specified in 3.8.3, and without exceeding structural limitations.

3.9 HANDLING QUALITY REQUIREMENTS IN ATMOSPHERIC DISTURBANCES

3.9.1 Allowable Handling Qualities Degradations in Atmospheric Disturbances. Level 1 flying qualities as defined by Table 15 are required for atmospheric disturbance levels up to and including moderate, as defined in Table 16.

TABLE 15 (Appendix B). DEFINITION OF LEVELS

LEVEL	ATMOSPHERIC DISTURBANCES			
	LIGHT	MODERATE	SEVERE	EXTREME
1	Flying qualities clearly adequate for the mission Flight Phase	Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both exists	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed.	Flying qualities such that control can be maintained long enough to fly out of the disturbance
2	Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists.	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed.	Flying qualities such that control can be maintained long enough to fly out of the disturbance	Flying qualities such that pilot can regain control after being upset
3	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed.	Flying qualities such that control can be maintained long enough to fly out of the disturbance	Flying qualities such that pilot can regain control after being upset	No requirement

TABLE 16 (Appendix B)

ATMOSPHERIC DISTURBANCE DEFINITIONS FOR SIMULATION AND FLIGHT TEST

MAGNITUDE	σ_{u_g} (ft/sec)
Light	0-3
Moderate	5
Severe	10
Extreme	24

The windshears corresponding to moderate turbulence are given as follows:

Decreasing Headwind:	$g\gamma_{max}$ not to exceed 3.4 ft/sec^2
Decreasing Tailwind:	$g\gamma_{min}$ not to exceed 1.7 ft/sec^2
Vector Shear:	9 deg/sec; V_w = 20 kt
Duration of All Shears:	At least 10 sec

where γ_{max} is the maximum power climb angle in the configuration used at wind shear initiation. γ_{min} is the flight path angle for flight idle in the configuration existing at wind shear initiation.

The following steady crosswind components corresponding to "light," "moderate," and "severe" disturbances are specified:

Qualitative Atmospheric Disturbance Level	Steady Crosswind (kt)
Light	0-10
Moderate	11-30
Severe	31-45

These crosswinds shall exist on short approach and at touchdown. When complying via piloted simulation, the wind values may be invariant with time, position, or altitude.

3.9.2 Definition of Atmospheric Disturbance Model Form. When compliance via demonstration is to be carried out using piloted simulation, an atmospheric disturbance model appropriate to the piloting task shall be included. As a minimum, the atmospheric disturbance model shall consist of the elements outlined in the following paragraphs.

1. Random Wind Component

The standard random wind component consists of the basic Dryden spectral form for each of the translational and rotary components considered necessary. These spectral forms are:

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

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

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

$$\Phi_{p_g}(\Omega) = \sigma_{p_g}^2 \frac{2L_p}{\pi} \frac{1}{1 + (L_p\Omega)^2}$$

$$\Phi_{q_g}(\Omega) = \frac{\Omega^2}{1 + [(4b/\pi)\Omega]^2} \Phi_{w_g}(\Omega)$$

$$\Phi_{r_g}(\Omega) = \frac{\Omega^2}{1 + [(3b/\pi)\Omega]^2} \Phi_{v_g}(\Omega)$$

where

$$\sigma_x^2 = \int_0^\infty \Phi_x(\Omega)\, d\Omega$$

$$\Omega = \omega/V$$

$$b = \text{wing span}$$

The primary determinant of turbulence intensity is σ_{u_g} and the values to be used for evaluation of flying qualities are given in Table 17. The relationships between translational intensities and scale lengths are:

$$\frac{\sigma_u^2}{L_u} = \frac{\sigma_v^2}{2L_v} = \frac{\sigma_w^2}{2L_w}$$

For the p-gust the intensity and scale length are associated with the w-gust by:

$$\sigma_{p_g} = \frac{1.9}{\sqrt{L_w b}}\, \sigma_{w_g}$$

and

$$L_p = \frac{\sqrt{L_w b}}{2.6}$$

Scale lengths are set according to altitude by the following relationships:

$$L_u = L_v = L_w = h_1 \qquad \text{for } h > 1750 \text{ ft}$$

$$\left.\begin{aligned} L_u &= L_v = \sqrt[3]{h_1^2 h} \\ L_w &= h \end{aligned}\right\} \qquad \text{for } h_o < h < h_1$$

$$L_u = L_v = \sqrt[3]{h_1^2 h_o} \qquad \text{for } h < h_o \text{ ft}$$

$$L_w = h_o$$

where h is the center of gravity height above ground, h_o = 10 ft, and h_1 = 1750 ft.

2. Wind Shear Component

The standard wind shear is represented by a constant time rate of change of wind speed and direction.

For $t < t_o$,

$$u_g = V_o \cos \psi_o \quad , \quad v_g = V_o \sin \psi_o$$

for $t > t_f$,

$$u_g = V_f \cos \psi_f \quad , \quad v_g = V_f \sin \psi_f$$

and, for the duration of the shear, $t_o < t < t_f$

$$u_g = V_o \cos \psi_o + \frac{t - t_o}{t_f - t_o} \left(V_f \cos \psi_f - V_o \cos \psi_o\right)$$

and

$$v_g = V_o \sin \psi_o + \frac{t - t_o}{t_f - t_o} \left(V_f \sin \psi_f - V_o \sin \psi_o\right)$$

where

V_o = Initial wind velocity

ψ_o = Initial wind angle

V_f = Final wind velocity

ψ_f = Final wind angle

t_o = Time shear is initiated

t_f = Time shear is terminated

The maximum wind shear magnitude is set according to the incremental flight path change capability of the aircraft up to a limit, i.e.,

$$\frac{V_f - V_o}{t_f - t_o} = g\Delta\gamma < 3.4 \text{ ft/sec}^2$$

The shear duration shall be at least 10 sec and the shear shall terminate at an altitude of 50 ft for landing simulations. The mean wind at altitude shall be set so the wind at touchdown is zero. $\Delta\gamma$ in the up direction is established at maximum climb power and in the down direction at flight idle. The wind shear magnitude shall not exceed 3.4 ft/sec^2. At least four critical wind shear cases are considered:

- Decreasing headwind
- Decreasing tailwind
- Decreasing crosswind
- Headwind to tailwind (constant wind speed)

The standard wind shear component is accompanied by the light level of random turbulence, i.e., $\sigma_{u_g} = 3.0$ ft/sec.

3. Mean Wind

The capability for simulating a mean wind from any direction shall be included. The wind shall be parallel to the earth's surface.

4. Discrete Gust Model

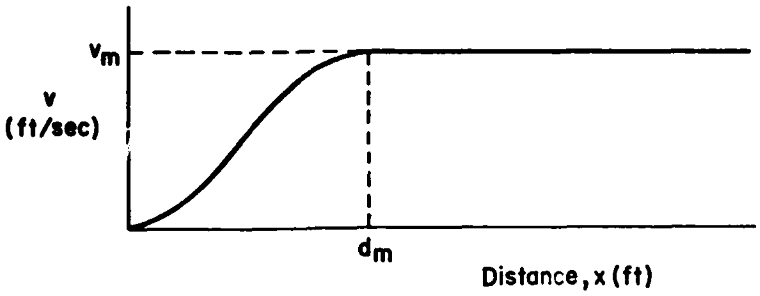
The discrete gust model may be used for any of the three gust-velocity components and, by derivation, any of the three angular components.

The discrete gust has the "1 - cosine" shape given by:

$$v = 0 \quad , \quad x < 0$$

$$v = \frac{v_m}{2}\left(1 - \cos\frac{\pi x}{d_m}\right) \quad , \quad 0 \leq x \leq d_m$$

$$v = v_m \quad , \quad x > d_m$$



The discrete gust above may be used singly or in multiples in order to assess airplane response to, or pilot control of, large disturbances. Step function or linear ramp gusts may also be used.

3.9.3 Application of disturbance models in analyses. The gust and turbulence velocities shall be applied to the airplane equations of motion through the aerodynamic terms only, and the direct effect on the aerodynamic sensors shall be included when such sensors are part of the airplane augmentation system. Application of the disturbance model depends on the range of frequencies of concern in the analyses of the airframe. When structural modes are significant, the exact distribution of turbulence velocities should be considered. For this purpose, it is acceptable to consider u_g and v_g as being one-dimensional, a function of both x and y, for the evaluation of aerodynamic forces and moments.

When structural modes are not significant, 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 airplane center of gravity. The angular velocities due to turbulence are equivalent in effect to airplane angular velocities. Approximations for these angular velocities are defined (precise only at very low frequencies) as follows:

$$-\dot{\alpha}_g = q_g = \frac{\partial w_g}{\partial x} \quad , \quad p_g = -\frac{\partial w_g}{\partial y} \quad , \quad r_g = -\frac{\partial v_g}{\partial x}$$

The spectra of the angular velocity disturbances due to turbulence are given in Paragraph 3.9.2.

For altitudes below 175 ft, the turbulence velocity components u_g, v_g, and w_g are to be taken along axes corresponding to u_g aligned along the relative mean wind vector and w_g vertical.

3.9.4 Requirements for Aircraft Failure States in Atmospheric Disturbances. When Aircraft Failure States exist (3.1.6), a degradation in flying qualities is permitted only if the probability of encountering Level 2 or 3 in turbulence is sufficiently small. At intervals established by the procuring activity, the contractor shall determine, based on the most accurate available data, the probability of occurrence of each Aircraft Failure State per flight and the effect of that Failure State on the flying qualities within the Operational and Service Flight Envelopes. These determinations shall be based on MIL-STD-756 except that:

a) All airplane components and systems are assumed to be operating for a time period, per flight, equal to the longest operational mission time to be considered by the contractor in designing the airplane, and

b) Each specific failure is assumed to be present at whichever point in the Flight Envelope being considered is most critical (in the flying qualities sense).

From these Failure State probabilities and effects, the contractor shall determine the overall probability, per flight, that one or more 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. Table 17 specifies the requirements as functions of the probability of encountering the degradation in flying qualities.

TABLE 17 (Appendix B)
LEVELS FOR AIRCRAFT FAILURE STATES

ATMOSPHERIC DISTURBANCES	FAILURE STATE I*	FAILURE STATE II**
Light	Level 2	Level 3
Moderate	Level 2	Recoverable*** or better
Severe to Extreme	Recoverable or better	No requirement

*For flight in the Operational Flight Envelope:
Probability of encountering degraded levels of flying qualities due to failure(s) $< 2.5 \times 10^{-3}$/flight hr

**For flight in the Operational Flight Envelope:
Probability of encountering degraded levels of flying qualities due to failure(s) $< 2.5 \times 10^{-5}$/flight hr
for flight in the Service Flight Envelope:
Probability of encountering degraded levels of flying qualities due to failure(s) $< 2.5 \times 10^{-3}$/flight hr

***Recoverable is defined as: control can be maintained long enough to fly out of a disturbance.

4. NOTES

The contractor shall refer to all applicable areas of Sections 4.1 through 4.5 of the MIL Handbook.

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