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BACKGROUND INFORMATION AND USER GUIDE FOR MIL-F-83300-MILITARY SPECIFICATION— FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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FOREWORD

This document was prepared for the United States Air Force by the Cornell Aeronautical Laboratory, Inc. (CAL) Buffalo, New York in partial fulfillment of Contracts AF33(615)-3736 and F33615-70-C-1322, under Project 698DC. Also prepared under this contract was a proposed new specification for V/STOL flying qualities: Military Specification - Flying Qualities of Piloted V/STOL Aircraft, dated July 1970, which has been submitted for adoption by the Air Force, Navy, and Army.

The contracted work was performed by CAL's Flight Research Department under the sponsorship of the Air Force Flight Dynamics Laboratory, Air Force Systems Command.

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This document is published to aid in interpreting the V/STOL Specification, MIL-F-83300. The Background Information and User Guide is not a contractual document.

The authors released this document for publication in February 1971. Revisions to this document will be issued as MIL-F-83300 is amended in the future. Comments, suggestions and requests for copies of the Background Information and User Guide should be addressed to AFFDL (FGC), Wright-Patterson AFB, Ohio 45433.

This technical report has been reviewed and is approved.

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ABSTRACT

This document is published in support of Military Specification MIL-F-83300, "Flying Qualities of Piloted V/STOL Aircraft".

The specification was compiled after an extensive literature review and many meetings and discussions with personnel from essentially all concerned civilian and governmental organizations. This report attempts to explain the concept and philosophy underlying the V/STOL Specification and to present some of the data and arguments upon which the requirements were based.

The document should also serve as a summary of the state of the V/STOL flying qualities art as determined from flight test, simulation, analysis, and theory.

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LIST OF SYMBOLS AND ABBREVIATIONS

SYMBOLS

$\bar{c}$	Mean chord of wing, feet
F_s	Pitch control force applied by pilot, pounds
g	Acceleration of gravity, feet/second^2
h	Height above ground level (AGL) or above mean sea level (MSL), feet
h_{max}	Maximum service altitude (defined in 3.1.8.4)
$h_{o_{max}}$	Maximum operational altitude (3.1.7)
$h_{o_{min}}$	Minimum operational altitude (3.1.7)
I_x, I_y, I_z	Moments of inertia about the x, y and z axes, respectively, slug-feet^2
I_{xz}	Product of inertia, slug-feet^2
j	$\sqrt{-1}$
L	Lift
L	Rolling moment about the x-axis, including thrust effects, positive for right wing down, foot-pounds
L_i	$= \frac{1}{I_x}\frac{\partial L}{\partial i}$, $i = v, p, r, \delta_{AS}, \delta_{RP}, \phi$, $\frac{\text{radians}}{\text{second}^2 - (\text{units of } i)}$
L'_i	$= \left[1 - \frac{I_{xz}^2}{I_x I_z}\right]^{-1}\left[\frac{1}{I_x}\frac{\partial L}{\partial i} - \frac{I_{xz}}{I_x I_z}\frac{\partial N}{\partial i}\right]$, $i = v, p, r, \beta, \delta_{AS}, \delta_{RP}, \phi$ $\frac{\text{radians}}{\text{second}^2 - (\text{units of } i)}$
m	Mass of aircraft, slugs
M	Pitching moment about the y-axis, including thrust effects, positive nose-up, foot-pounds
M_i	$= \frac{1}{I_y}\frac{\partial M}{\partial i}$, $i = u, w, \dot{w}, q, \alpha, \delta_{ES}, \delta_\lambda, \delta_{th}, \theta$ $\frac{\text{radians}}{\text{second}^2 - (\text{units of } i)}$

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SYMBOLS

n, n_z	Normal load factor
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 (defined in 3.1.8.5)
$n_{o_{max}}, n_{o_{min}}$	Maximum and minimum operational load factors (3.1.7)
$n(+), n(-)$	For a given altitude, the upper and lower boundaries of n in the V-n diagrams depicting the Operational Flight Envelope
N	Yawing moment about the z-axis, including thrust effects, positive nose right, foot-pounds
N_i	$= \frac{1}{I_z} \frac{\partial N}{\partial i}$, $i = v, p, r, \beta, \delta_{AS}, \delta_{RP}$, $\frac{\text{radians}}{\text{second}^2 - (\text{units of } i)}$
N_i'	$= \left[1 - \frac{I_{xz}^2}{I_x I_z}\right]^{-1} \left[\frac{1}{I_z}\frac{\partial N}{\partial i} - \frac{I_{xz}}{I_x I_z}\frac{\partial L}{\partial i}\right]$, $i = v, p, r, \beta, \delta_{AS}, \delta_{RP}$ $\frac{\text{radians}}{\text{second}^2 - (\text{units of } i)}$
n/α	The steady-state normal acceleration change per unit change in angle of attack for an incremental pitch control deflection at constant speed, g/radians
p	Roll rate about the x-axis, radians/second
PM_h	Phase margin for the altitude loop, degrees
q	Pitch rate about the y-axis, radians/second
q	Dynamic pressure, pounds/feet^2
r	Yaw rate about the z-axis, radians/second
R	Pilot rating; calculated from rating expression
s	Laplace transform variable, 1/second
S	Wing area, feet^2
t	Time, seconds
t_a	Control input ramp time, seconds

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SYMBOLS

t_{n_β}	Time for the Dutch roll oscillation in the sideslip response to reach the n^{th} local maximum for a right pulse roll command, or the n^{th} local minimum for a left command (See Figures 8 and 9 of paragraph 6.2.6)
T	Thrust, pounds
T_2	Time to double amplitude, $T_2 = -0.693/\zeta\omega_n$ for an oscillation, $T_2 = -0.693\,\tau$ for a first order divergence, seconds
T_d	Damped period of the Dutch roll, seconds, $T_d = \frac{2\pi}{\omega_{n_d}\sqrt{1-\zeta^2}}$
$1/T_{sp}$	Aperiodic root of the longitudinal hovering cubic characteristic equation, $seconds^{-1}$
$1/T_{sp_1}, 1/T_{sp_2}, 1/T_{sp_3}$	Real roots of the longitudinal hovering cubic characteristic equation, $seconds^{-1}$
$1/T_\theta$	First order zero of the hovering pitch attitude to pitch control transfer function, $seconds^{-1}$
T_{L_θ}	Pilot lead time constant in pitch, seconds
T_{L_x}	Pilot lead time constant in longitudinal displacement, seconds
U_o	Total velocity along the x reference axis
u	Incremental velocity along the x reference axis, feet/second
u_g	Random gust velocity along the x body axis, feet/second
v	Incremental velocity along the y reference axis, feet/second
v_g	Random gust velocity along the y body axis, feet/second
V	Airspeed, along the flight path
$V_{max}(x)$, $V_{min}(x)$	Shorthand notation for the speeds V_{max}, V_{min} for a given configuration, weight, center-of-gravity position, and external store combination associated with Flight Phase X
V_{end}	Speed for maximum endurance
V_{range}	Speed for maximum range in zero wind conditions
V_{MAT}	High speed, level flight, maximum augmented thrust
V_{max}	Maximum service speed (defined in 3.1.8.1)
V_{min}	Minimum service speed (3.1.8.2)

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SYMBOLS

$V_{o_{max}}$	Maximum operational speed (defined in 3.1.7)
$V_{o_{min}}$	Minimum operational speed (3.1.7)
V_{con}	The speed which establishes the upper limit of applicability of the requirements of this specification and the lower limit of applicability of the requirements of MIL-F-8785. No more precise definition of V_{con} will be attempted as it is assumed that V_{con} will be chosen by the contractor subject to approval by the procuring activity. Factors to be considered in the selection of V_{con} are discussed in 6.7 of Reference 1.
$V_{o_{max/min}}(X)$	The maximum/minimum operational speed associated with Flight Phase X
V_s	Stall Speed as defined in Reference 84
$V_s(X)$	Stall speed associated with Flight Phase X
V_e	Equivalent airspeed
V_T	True airspeed
V_{UN}	Unlock speed. That is the maximum speed at which the wings, ducts, etc. can be unlocked and conversion commenced, knots EAS
w	Incremental velocity along the z reference axis, feet/second
w_o	Initial trim velocity along the z reference axis, feet/second
w_g	Random gust velocity along the z body axis, feet/second
W	Weight, pounds
x	Body-fixed axis of the aircraft, along the projection of the undisturbed (trim or operating-point) velocity onto the plane of symmetry, with its origin at the c.g. Positive forward.
X	Force along the x-axis, aerodynamic plus thrust, pounds
X_i	$= \frac{1}{M}\frac{\partial X}{\partial i}$, $i = u, w, q, \delta_{es}, \delta_{th}, \delta_R, \theta$ $\frac{\text{feet}}{\text{second}^2 - (\text{units of } i)}$
y	Body-fixed axis of the aircraft perpendicular to the plane of symmetry, directed out the right wing, with its origin at the c.g.

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SYMBOLS

Y	Force along the y-axis, aerodynamic plus thrust, pounds
Y_i	$= \frac{1}{M}\frac{\partial Y}{\partial i}$, $i = u, v, p, r, \delta_{AS}, \delta_{RP}$ $\frac{\text{feet}}{\text{second}^2 \text{ - (units of } i \text{)}}$
$\mathcal{Z}$	Body-fixed axis of the aircraft, directed downward perpendicular to the x and y axes, with its origin at the c.g.
Z	Force along the z-axis, aerodynamic plus thrust, pounds
Z_i	$= \frac{1}{M}\frac{\partial Z}{\partial i}$, $i = u, w, q, \delta_{ES}, \delta_{th}, \delta_{\lambda}$ $\frac{\text{feet}}{\text{second}^2 \text{ - (units of } i \text{)}}$
α	Angle of attack, the angle between the fuselage reference line and the projection of the relative wind, measured in the plane of symmetry. Positive angle of attack corresponds to flow approaching from below the x-y plane.
β	Sideslip angle, the angle between the relative wind and the projection of the relative wind on the x-z plane of symmetry. Positive sideslip corresponds to flow approaching from the right side of the plane of symmetry.
$\Delta\beta$	The maximum change in sideslip following an abrupt roll control pulse command within time $t_{\Delta\beta}$; where $t_{\Delta\beta}$ is the lesser of 6 seconds or one-half the Dutch roll period, and is measured from a point halfway through the duration of the pulse command (Figures 8 and 9) ~ degrees
γ	Climb angle, $= \sin^{-1}\frac{\text{vertical speed}}{\text{true airspeed}}$, positive for climb
Δ	Used in combination with other parameters to denote an incremental change from the initial value
δ_{AS}	Displacement of the cockpit roll control along its path, positive control produces right rolling moments, inches
δ_{ES}	Cockpit pitch control deflection, positive control produces nose-up pitching moments, inches
δ_{RP}	Cockpit yaw control deflection, positive produces nose-right yawing moments, inches
δ_f	Flap deflection ~ degrees

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SYMBOLS

δ_{th}	Thrust magnitude control deflection
δ_{λ}	Thrust vector angle control position
ζ	Damping ratio
ζ_d	Damping ratio of the Dutch roll oscillation
ζ_p	Damping ratio of the phugoid oscillation
ζ_{sp}	Damping ratio of the longitudinal short-period oscillation
ζ_{ϕ}	Damping ratio of the numerator quadratic of the ϕ/δ_{AS} transfer function
θ	Pitch angle, angle between the fuselage reference line and the horizontal, positive nose-up, radians
λ	Thrust vector angle, wing tilt angle, etc.
λ	Aperiodic root of the longitudinal hovering cubic characteristic equation, seconds^{-1}
σ_x	Standard deviation of longitudinal displacement, feet
σ_q	Standard deviation of pitch rate, radians/second
$\sigma_{u_g}, \sigma_{v_g}$	Root-mean-square intensities of u_g, v_g respectively, feet/second
σ	Square root of the ratio of air density at a given altitude to sea level density
τ	First-order time constant, seconds
τ_{th}	First-order time constant in thrust control system
τ_R	Roll mode time constant, seconds
τ_s	Spiral mode time constant, seconds
ϕ	Bank angle measured in the y-z plane, between the y-axis and the horizontal, positive right wing down, radians
ϕ_1, ϕ_2, ϕ_3	Bank angles at the first, second and third peaks, respectively

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SYMBOLS

$\frac{\phi_{osc}}{\phi_{AV}}$ — A measure of the ratio of the oscillatory component of bank angle to the average component of bank angle following an impulse roll control command with yaw control free:

$$\zeta_d \leq 0.2: \quad \frac{\phi_{osc}}{\phi_{AV}} = \frac{\phi_1 + \phi_3 - 2\phi_2}{\phi_1 + \phi_3 + 2\phi_2}$$

$$\zeta_d > 0.2: \quad \frac{\phi_{osc}}{\phi_{AV}} = \frac{\phi_1 - \phi_2}{\phi_1 + \phi_2}$$

ϕ_{AV}

$$\zeta_d \leq 0.2: \quad \phi_{AV} = \frac{1}{4}(\phi_1 + \phi_3 + 2\phi_2)$$

$$\zeta_d > 0.2: \quad \phi_{AV} = \frac{1}{2}(\phi_1 + \phi_2)$$

$\ddot{\phi}_o$ — Maximum roll control power $L_{\delta_{AS}} \delta_{AS}$, radians/second^2

$\left|\frac{\phi}{\beta}\right|_d$ — At any instant, the ratio of amplitudes of the bank-angle and sideslip-angle envelopes in the Dutch-roll mode

ψ — Yaw angle, positive nose right, radians

ψ_β — Phase angle expressed as a lag for a cosine representation of the Dutch roll oscillation in sideslip, where

$$\psi_\beta = \frac{-360}{T_d} t_{\eta_\beta} + (n - 1)\ 360 \text{ (degrees)}$$

with n as in t_{η_β}

$\psi(1)$ — Yaw angle change within one second following a one inch step yaw control input, degrees

ω — Imaginary part of a complex dynamic root, radians/second

ω_c — Cross-over frequency, radians/second

ω_n — Undamped natural frequency of second order system, radians/second

ω_{n_d} — Undamped natural frequency of the Dutch roll oscillation, radians/second ($\omega_{n_d}^2 > 0$ is indicative of positive weathercock stability.)

ω_p — Undamped natural frequency of the phugoid oscillation, radians/second

ω_{sp} — Undamped natural frequency of the short-period oscillation, radians/second

ω_ϕ — Frequency of numerator quadratic of ϕ/δ_{AS} transfer function

$(\dot{})$ — dot denotes differentation with respect to time

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ABBREVIATIONS

c.g.	Center of gravity
EAS	Equivalent airspeed
FOP	Fixed operating point
IFR	Instrument flight rules
kt	Knots
mph	Miles per hour
MAT	Maximum augmented thrust: maximum thrust, augmented by all means available for the Flight Phase
MRT	Military rated thrust, which is the maximum thrust at which the engine can be operated for a specified period
MSL	Mean sea level
NRT	Normal rated thrust, which is the maximum thrust at which the engine can be operated continuously
PIO	Pilot-induced oscillation
SAS	Stability augmentation system
TAS	True airspeed
V/STOL	Vertical/short takeoff and landing
VFR	Visual flight rules

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Section I

INTRODUCTION

This document is published in support of Military Specification MIL-F-83300 "Flying Qualities of Piloted V/STOL Aircraft" (Reference 1), as part of a four year long program performed by Cornell Aeronautical Laboratory (CAL) for the Air Force Flight Dynamics Laboratory. The intent of this document is to explain the concept and philosophy underlying the V/STOL Specification and to present some of the data and arguments upon which the requirements were based.

The presented material was obtained or generated following an extensive literature review and after many meetings and discussions with personnel from essentially all concerned civilian and governmental organizations. A number of subcontracts were performed to obtain supplemental, experimental and analytical data. The results of these efforts have been published separately.

Section II outlines the historical development of the project and acknowledges the many organizations, both industrial and governmental, that contributed comments, criticisms and suggestions in the form of review comments.

The philosophy and structure of the specification is outlined in Section III. This attempts to give the user of the specification an appreciation for the manner in which the requirements have been grouped; especially in distinguishing between the fixed operating point requirements and the requirements for the actual transition maneuver.

Section IV presents a review of the entire V/STOL Specification, in order, paragraph by paragraph. The format used is to present the pertinent paragraph, or group of paragraphs from the Specification, and then to follow this with a discussion of the requirement. Attention is directed at explaining the intent of the requirement, a discussion of the theoretical background and experimental data on which the requirement is based, and a discussion of the possible limitations or inadequacies of the requirement. Where a similar requirement or design criteria existed before, the earlier version is mentioned to provide a basis for comparison.

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Section II

HISTORICAL DEVELOPMENT

The development of a flying qualities specification for V/STOL aircraft was one of the prime efforts of an Air Force advanced development program called the VTOL Integrated Flight Control System (VIFCS) program. As originally conceived, this program had four basic parts which can be briefly described as:

1. Flight control system design, integration, and test including definition of the total flight control system criteria to meet VTOL requirements, and integration and fabrication of a total flight control system for control technology demonstration and validation in a modified XV-4.

2. Analysis, design, development, and flight investigation of specific flight path display techniques suitable for all-weather operation and their integration with the pilot-control system combination.

3. Development of VTOL handling qualities design criteria.

4. Modification of a jet VTOL airplane (the XV-4) for use as a variable stability test vehicle (the XV-4B).

The Cornell Aeronautical Laboratory, Inc. (CAL) was awarded a contract for part three above in April 1966. Under the contract, CAL's overall responsibilities included:

1. experimental simulator investigations into the handling qualities of VTOL airplanes,

2. developing techniques for analyzing and evaluating VTOL handling qualities, and

3. utilizing experimental data and analysis to generate VTOL handling qualities requirements and design criteria.

The initial effort during the first year of the program involved a survey of the VTOL flying qualities literature. This involved reading many reports and papers and attempting to digest the relevant information, data, opinions, ideas, and methods presented by various authors representing different agencies and companies.

In order to supplement the literature surveys, a series of meetings was held with representatives of airframe companies engaged in design, development, and manufacture of VTOL aircraft. At these meetings, held during the weeks of 10 October 1966 and 24 October 1966, the attendees discussed:

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(1) views, feelings and opinions on the applicability of existing handling qualities documents to VTOL aircraft, and

(2) the format and content of a future VTOL handling qualities specification.

The following manufacturers were represented:

Bell Aerosystems
Bell Helicopter
Boeing
Canadair
General Dynamics - Ft. Worth
Grumman
Kaman
Ling-Temco-Vought
Lockheed-California
Lockheed-Georgia
McDonnell
Norair
North American Aviation - Columbus
North American Aviation - Los Angeles
Republic
Ryan
Sikorsky

In addition, the following government agencies and contractors were present:

Air Force Flight Dynamics Laboratory
Air Force Aeronautical Systems Division
Air Force Flight Test Center
Army Aviation Materiel Laboratories
Cornell Aeronautical Laboratory, Inc.
Federal Aviation Agency
National Aeronautics and Space Administration
Systems Technology, Inc.

By providing a broad view of the overall V/STOL flying qualities picture, the literature surveys and meetings established a basis for more intelligent planning and coordination of the subsequent program activities. Reference 2 summarizes some of the results of the first year efforts.

To promote the attainment of the flying program objectives, CAL was authorized to issue subcontracts. These subcontracts were planned and coordinated so that the work devoted to preparing a V/STOL flying qualities specification would benefit from the experimental and analytical capability of other organizations known to have a direct interest in V/STOL. It should be mentioned that although the specification work originated as part of a broad Air Force program that included the development of the variable stability XV-4B, the unfortunate loss of this aircraft eliminated the possibility

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of fulfilling all of the VIFCS program objectives within the original time-tables. Thus the subcontract efforts took on additional importance as a means of acquiring relevant data and information to use in formulating a flying qualities specification.

During the course of the program, four organizations participated as subcontractors: United Aircraft Research Laboratories (UARL), Systems Technology Inc. (STI), Northrop-Norair, and National Research Council of Canada (NRC). Each subcontractor was selected so that, as shown in the following listing, V/STOL flying qualities could be systematically investigated by using different techniques and approaches to acquire and analyze data.

UARL	- fixed-base simulation
STI	- pilot model analyses
Norair	- moving-base simulation
NRC	- flight simulation with VSS helicopter

Both UARL and STI were awarded two subcontracts. The first subcontracts were initiated in late 1966 with work performed throughout most of 1967. The second subcontracts, basically extensions of the first, were pursued throughout most of 1968. Both the Norair and NRC work was started early in 1968 and continued for one year.

CAL efforts during 1967 and 1968 were, in addition to administering the subcontracts and participating in the simulations, concentrated on formulating flying qualities requirements using the pertinent data in the literature and the data generated during the subcontracts as it became available. This work culminated in the publication in October 1968 of the first version of a proposed V/STOL flying qualities specification (Reference 3) along with an accompanying report containing related backup information and data (Reference 4). Both of these documents were submitted to industry for review. Review comments were returned by the following industry organizations:

Bell Aerosystems Company
Boeing-Seattle
Boeing-Vertol
Grumman
Ling-Temco-Vought
Lockheed - California
Lockheed - Georgia
McDonnell
North American Rockwell - Los Angeles
Ryan
Sikorsky

The two documents were also reviewed by, and comments received from, the following Government agencies:

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Air Force Flight Dynamics Laboratory
Air Force Aeronautical Systems Division
Air Force Flight Test Center
Army Aviation Materiel Laboratories
National Aeronautics and Space Administration
Naval Air Systems Command

CAL then proceeded with a thorough study of the review comments along with continued data analysis during much of 1969. A revised specification was prepared in September 1969 (Reference 5). In October 1969, Reference 5 was jointly reviewed by representatives of the Air Force, Army, Navy and CAL. This latter review took place in order to screen Reference 5 prior to submitting it to a second cycle of industry review. Some changes were recommended and these changes were incorporated into the pertinent requirement paragraphs and resulted in the publication of Reference 6.

A new document entitled Background Information and User Guide (BIUG) (Reference 7) was then prepared by CAL and in January 1970 these two documents (References 6 and 7) were distributed to industry and Government agencies for a second review cycle.

Detailed review comments were received from the following organizations:

Bell Aerospace
Boeing - Military Airplane Systems Division
Boeing - Vertol Division
Flight Systems
General Dynamics, Convair Division
General Electric, Aircraft Equipment Division
Grumman Aerospace
Lockheed - Georgia
LTV Aerospace -Vought Aeronautics Division
McDonnell Aircraft
North American Rockwell - Autonetics Division
North American Rockwell - Columbus Division
North American Rockwell - Los Angeles Division
Northrop - Aircraft Division
Princeton University
Teledyne Ryan Aeronautical
United Aircraft Research Laboratories
United Aircraft - Sikorsky Aircraft Division

In addition, letters giving an overall appraisal were received from:

Bell Helicopter Co.
Douglas Aircraft Co.
Kaman Aerospace Co.

The documents were also reviewed by, and written comments received from, the following U.S. Government and foreign agencies:

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Air Force Flight Dynamics Laboratory
Aerospace Research Pilots School
British Ministry of Technology (A and AEEE Boscombe Down)
Dornier Company
Messerschmitt - Boelkow - Blohm GMBH

On the basis of these comments, CAL prepared some suggested changes and in April 1970 distributed copies to potential attendees of an Air Force - Navy - Army review meeting. This review took place at the end of April 1970 and substantial agreement on a final version was obtained by the Air Force, Navy and Army representatives.

Resolution of final details continued until about 4 July 1970 when CAL published a new version (Reference 8). The Air Force made some minor additional changes and printed a version which was distributed for the third and final review coordination (Reference 9). Detailed review comments were received from the following organizations:

Boeing Military Airplane Systems Division
Boeing Vertol
General Dynamics - Convair Division
Grumman Aerospace
Kaman
Lear Siegler, Astronics Division
Lockheed - California
Lockheed - Georgia
LTV Aerospace - Vought Aeronautics Division
McDonnell Douglas - Douglas Aircraft
McDonnell Douglas - McDonnell Aircraft
North American Rockwell - Autonetics Division
North American Rockwell - Columbus Division
North American Rockwell - Los Angeles Division
Northrop - Aircraft Division
Sperry - Flight Systems Division
Systems Technology Incorporated
Teledyne Ryan Aeronautical
United Aircraft Research Laboratories
United Aircraft - Sikorsky Aircraft Division

These comments were reviewed and several changes made to the specification requirements. The final version was agreed to by the Air Force and Navy representatives on 11 December 1970, and submitted for adoption as MIL-F-83300. During development of the specification it was intended to cover all V/STOL aircraft, including helicopters, for the Air Force, Navy and Army. At the time of publication of this report, the specification has been adopted by the Air Force for all V/STOL's, by the Navy for all except helicopters, and the Army has not made any commitment.

While Reference 9 was being reviewed, CAL prepared the draft of a new Background Information and User Guide (BIUG) for the specification. The purpose of the BIUG is to document the substantiating data used

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in the specification and also provide notes and explanations which should help the user of the specification. The draft, which formed the basis of this report, was submitted to the Air Force on 15 September 1970. The review comments were given to CAL on 11 December 1970. These comments and the most recent changes made to the specification were incorporated, and the final version was submitted for publication 1 February 1971.

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Section III

SPECIFICATION STRUCTURE AND PHILOSOPHY

The V/STOL Specification contains six main sections:

1. Scope and Classifications
2. Applicable Documents
3. Requirements
4. Quality Assurance Provisions
5. Preparation for Delivery
6. Notes.

As is usual with specifications, the Index is at the end.

The bulk of the material is contained in the Requirements section which is broken down into eight subsections:

3.1 General requirements
3.2 Hover and low speed
3.3 Forward flight
3.4 Transition
3.5 Characteristics of the flight control system
3.6 Takeoff, landing and ground handling
3.7 Atmospheric disturbances
3.8 Miscellaneous requirements.

As the title implies, Section 1 Scope and Classifications, defines the scope and application of the specification. It also defines the framework for classifying the aircraft, the mission Flight Phases and the Levels of flying qualities.

The Requirements section commences with a general statement. This provides a detailed explanation of the framework used to determine the conditions at which the requirements of the specification should be applied. The conditions of the aircraft which have to be considered are defined, and the framework for determining the corresponding flight conditions, primarily in terms of speed, altitude and load factor, is explained. In addition, a detailed explanation is given for applying the concept of Levels of flying qualities.

The stability and response requirements are written for two flight regimes:

Fixed Operating Point Flight
Accelerated Flight.

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Fixed Operating Point (FOP) Flight

This is the name that has been used for flight consisting of maneuvering about a constant trim condition. For this condition, the techniques of linearized constant coefficient analysis, which have been used for years on conventional aircraft, seem to apply. As a result, the conventional techniques of understanding and specifying flying qualities have been extrapolated into the lower speed range.

Quantitative requirements are placed on familiar concepts such as static stability, dynamic stability, control power, response to control inputs (sensitivity), and control lags.

The requirements have to cover all speeds from hover to V_{con}, where V_{con} is the speed at which the requirements of the conventional airplane specification, MIL-F-8785B, Reference 10, begin to apply. Within this speed range significant changes take place which make it necessary to change the flying qualities requirements. Examples of this will be apparent in the detailed discussion of requirements; the reasons can be summarized as follows:

- The characteristic modes of motion undergo substantial changes in form, as forward speed increases.
- The change from direct lift to aerodynamic lift, as forward speed increases, results in changes in important stability derivatives, and necessitates changes in pilot control technique.
- The parameters, and the specific values of those parameters, which adequately describe a level of handling qualities in hover, are inadequate or inappropriate to assure a similar level at high forward speeds.

It would be ideal if the requirements could be made a continuous function of some parameter such as speed. Unfortunately the detail of existing knowledge of V/STOL flying qualities has not allowed this, and so a two part arrangement has been chosen with the division at 35 knots. There is nothing profound about 35 knots - it is a compromise chosen on the basis of our present understanding and includes the following considerations:

- There is a substantial amount of published data resulting from experiments done in and around the hover condition. These experiments typically involved tasks in which the vehicle achieved translation velocities as high as 35 knots.
- Many aircraft begin to develop "significant" amounts of aerodynamic lift above 35 knots, at which time there often exists a basic change in the dynamics. For example, one usually finds that hover approximations, such as an effectively decoupled height (or $\dot{w}$) mode, begin to break down at about 35 knots.

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- Along with the changing nature of the dynamics there is usually a change in the piloting technique.

- Hovering over a spot at any angle to a 35 knot wind is a requirement of the proposed specification (and others). Consideration was given, by the Air Force, to increasing the wind speed in which hovering capability is required. However, it was found that the probability of encountering winds greater than 30-40 knots did not justify a change. Certainly winds higher than 35 knots can be encountered, but it was assumed that the margin of the Service Flight Envelope over the Operational Flight Envelope will provide Level 2 hovering capability at speeds greater than 35 knots. Further margins may have to be demanded in special cases.

- Since 35 knots is a satisfactory dividing speed from the point of view of both aircraft dynamics and operational considerations, it was convenient to group the requirements by speeds. If it had been decided that hovering capability was necessary up to some significantly higher speed such as 60 knots, then a more complex division of the dynamics and operational aspects would have been necessary and thereby created the need for a much more complicated specification structure.

With the step change at 35 knots, considerable care has been exercised to allow the requirements for speeds less than 35 knots to blend with the requirements for speeds greater than 35 knots, and to blend with the requirements of MIL-F-8785B at V_{con}.

Sections 3.2 and 3.3 then, provide requirements for Flight Phases which involve maneuvering about trim speeds in the range ± 35 knots and 35 knots to V_{con} respectively. For example, an STOL aircraft required to perform its landing approach at 60 knots would have to satisfy the requirements of 3.3 for that flight condition. A V/STOL which has to be able to perform tasks involving flight at 25 knots has to satisfy the requirements of 3.2 at that flight condition. An aircraft with an Operational Flight Phase which spans 35 knots has to satisfy 3.2 and 3.3 at the appropriate flight conditions.

Accelerated Flight

When considering the flying qualities of conventional aircraft it has been possible to virtually ignore the effects of acceleration (acceleration referring here to changing flight condition, particularly speed, rather than accelerations due to maneuvering) except for a few special conditions, such as passing through the transonic speed range. This happy circumstance is probably because the changes occur relatively slowly when compared to the frequencies of the rigid body modes. The acceleration capabilities of a V/STOL and the significant changes in dynamics and response which occur between zero speed and, say, 100 knots make it unlikely that we will be so lucky with V/STOL aircraft. It is desirable that a good understanding of the importance and extent of the transition problem be obtained as soon as

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possible.

Unfortunately the dynamics involved in a rapid transition are not yet well understood. For understanding, it is tempting to consider the dynamics of transition as though represented by a sequence of equilibrium or fixed operating points. Figures 1 and 2 show how the longitudinal dynamics of the X-22A change for such a sequence of points; Figure 1 showing the basic aircraft roots, and Figure 2 showing the roots for SAS on, or augmented.

Clearly the changes in dynamics are considerable, so bearing in mind that the changes can occur in as little as 18 seconds, the question is how to interpret these changes. There is, as yet, no general answer to this question; however, some comments can be made.

First, if one does wish to treat the accelerated flight condition as a series of "frozen points" it is necessary to evaluate the aerodynamic characteristics for the appropriate aircraft state. An aircraft such as the X-22A can encounter a wide range of speeds and power settings, at a given conversion angle, depending on the rate of conversion and whether accelerating or decelerating (Reference 11). Such changes can have a very marked influence on the nature of the aerodynamic force and moment characteristics and hence on the "frozen dynamics".

Second, it is simple to show (Reference 11) that representing a time-varying system as a sequence of time-invariant systems can give misleading information about the nature of the dynamics. However, it is not a simple matter to put a quantitative measure on such effects or devise alternative techniques which can be used to understand the dynamics of a rapid transition. A notable attempt has been made in Reference 12 to develop such a technique, and some interesting trends are shown for simple variations of the derivatives (e.g., linearly proportional to speed). However the problem is by no means solved.

Now consider how this complicated dynamic situation has been accommodated in the specification.

Transition can be thought of as two basic parts:

1. Control of the speed and altitude as though the aircraft was a point mass.

2. Control of perturbations in speed, altitude and attitude from the desired values.

A knowledge of the first part can be obtained by controlling the aircraft with a very tight feedback loop. From this can be obtained a pseudo-trim for that particular transition. When flying, the pilot has to provide this pseudo-trim and also control the perturbations from the desired transition.

The difficulty of this task will be strongly dependent on how quickly the aircraft diverges from the desired nominal value. This will be a function

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of the rate at which the out-of-trim moments increase, and the basic dynamics of the aircraft. The difficulty of the task will also be influenced by how much effort the pilot has to exert to keep the aircraft within the transition corridor, i.e., within the permissible space of speed, conversion angle, power settings, and angle of attack. These are the factors to which attention has been directed in the requirements. Because of the current lack of knowledge concerning the dynamics in transition, these have not been prescribed. If an aircraft is designed to comply with the FOP requirements, it seems reasonable to assume that the resulting transition dynamics will also be acceptable or at least can be made to meet the qualitative requirements without excessive redesign. Some V/STOL aircraft (e.g., X-22A, XC-142) have flight phases which require FOP operation at most speeds below V_{con} and as a result will have to comply with the FOP requirements. Other aircraft, such as the P.1127, may have no flight phase which requires FOP flight at speeds between about 35 knots and V_{con}. Applying the FOP requirements between 35 knots and V_{con} might be unduly conservative for these aircraft and so the flight-phase flight-envelope structure has been arranged so that the FOP requirements are not imposed unless the mission requires such operation. This statement has to be slightly qualified because any aborted transition is in fact required to be safe. Of course, the manufacturer may still use the FOP requirements as a design guide, but research needs to be performed to determine whether or not the resulting transition characteristics are adequate, ultraconservative or deficient. If they are deficient, of course, even the vehicle which has to perform FOP flight will have unsatisfactory transition characteristics. This is a subject which needs systematic research.

A final point in this general discussion of transition; what is meant by "transition"?

Reference 14 defines transition as "the act of going from the powered lift regime to the aerodynamic flight regime and vice versa".

For the purposes of applying section 3.4 of the V/STOL Specification, the following definition is preferred: "transition is the act of changing from one fixed operating point to another".

This latter definition reduces, in the limit, to the trivial case of maneuvering about trim. However, since at the present time it is not possible to specify what level of acceleration is significant, it is not possible to be more definitive. Certainly consideration should not be restricted to complete conversions/reconversions, as implied by the first definition. Also, it is desired that aircraft which do not change configuration when they accelerate (e.g., helicopters) should also be required to satisfy the transition requirements. This intention is accommodated more clearly by the preferred definition since it does not involve any extraneous concepts such as having to define the limits of the powered lift regime.

The remaining requirements in the specification have been collected into four groups:

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- Characteristics of the flight control system. This section places requirements on mechanical characteristics such as control force breakout and gradients, and trim characteristics.

- Takeoff, landing and ground handling requirements. Apart from convenience, this subgrouping does emphasize the fact that landing and takeoff are distinct flight phases, and that military aircraft can have missions other than takeoff and landing in the speed range below V_{con}.

- Winds and turbulence. Some requirements are written in terms of a steady wind speed, in which case compliance with the requirement should be demonstrated in flight, in that wind condition. Other requirements are written with reference to operation in all potential atmospheric environments. In the future it is hoped to include a suitable turbulence model in the specification. For the present time the turbulence model and intensity to be considered will be chosen by the procuring activity, and compliance will be demonstrated by suitable analysis.

- Miscellaneous requirements. Miscellaneous requirements that are equally valid at all speeds are collected together in the miscellaneous sections. Topics covered include: Warning and Prevention of Approach to Dangerous Flight Conditions, Pilot-Induced Oscillations, Cross-Coupling Effects, Transients Following Failures, and Control Following Thrust Loss.

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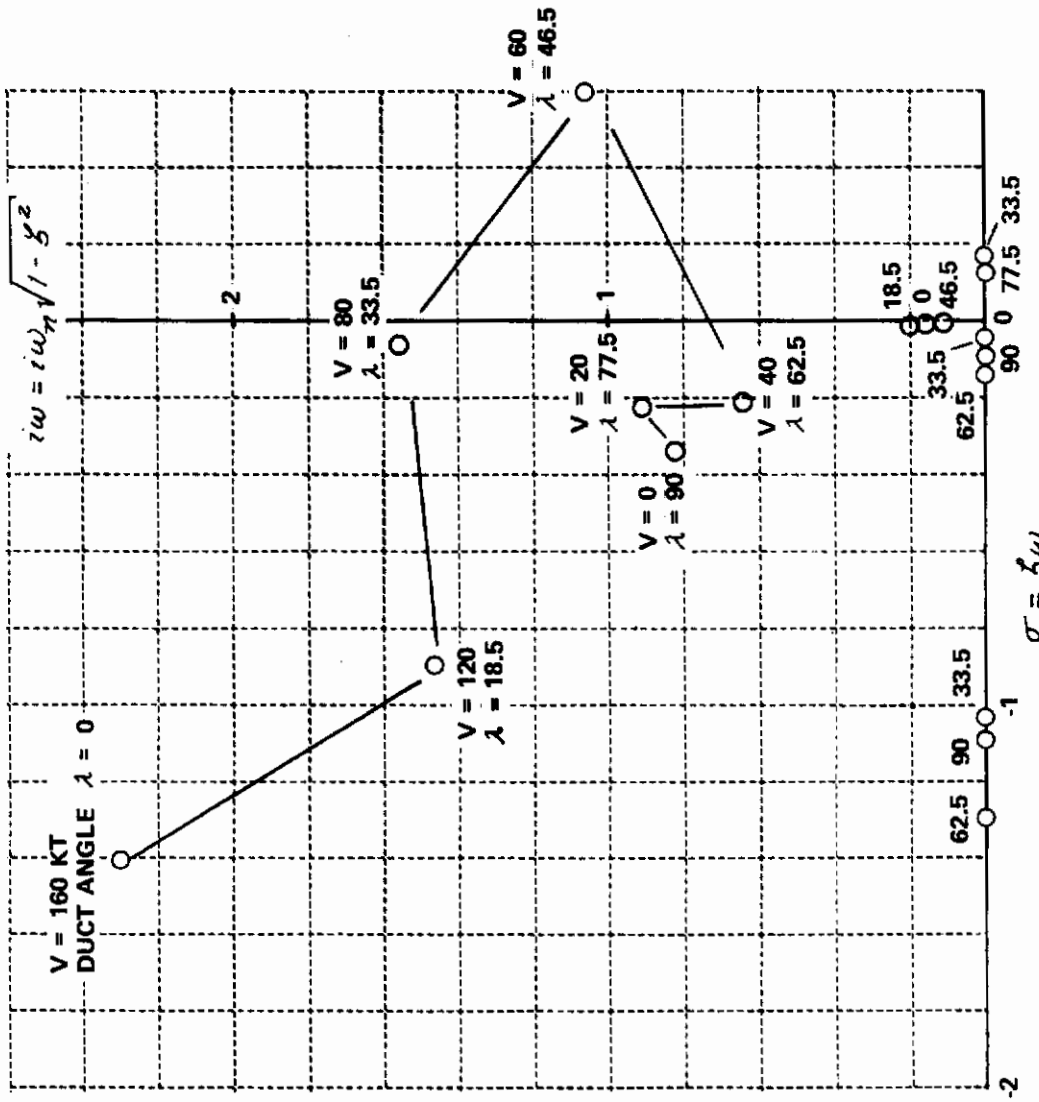

Figure 1 LOCUS OF X-22A FIXED OPERATING POINT ROOTS BETWEEN V = 0 AND V = 100 KT (BASIC AIRCRAFT) (DATA FROM REFERENCE 13)

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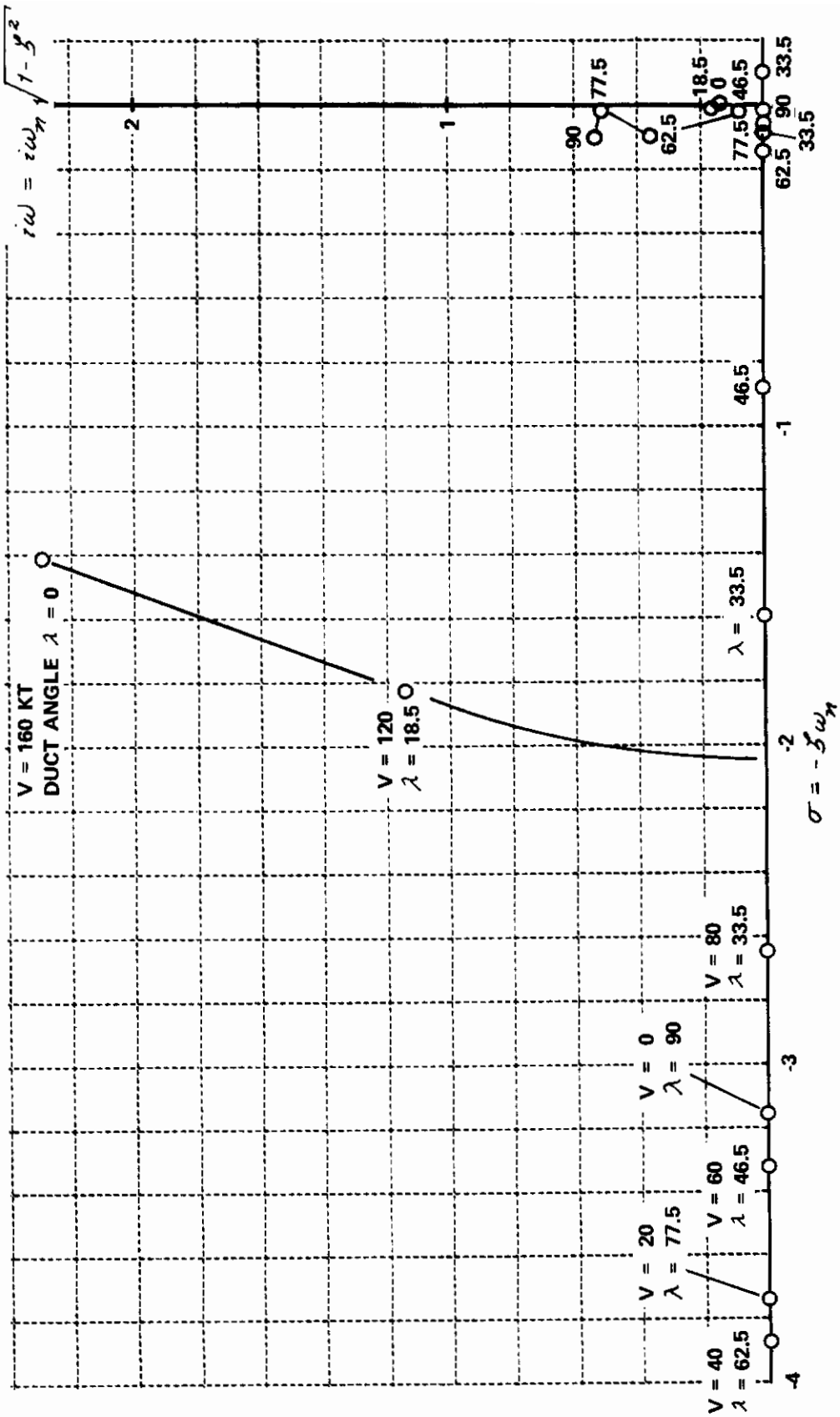

Figure 2 LOCUS OF X-22A FIXED OPERATING POINT ROOTS BETWEEN V = 0 AND V = 160 KT (AUGMENTED) (DATA FROM REFERENCE 13)

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Section IV

STATEMENT AND DISCUSSION OF REQUIREMENTS

1. SCOPE AND CLASSIFICATIONS

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

1. The kind of airplane (Class)
2. The job to be done (Flight Phase)
3. How good the flying qualities must be to do the required job (Level).

The following table shows how these considerations are associated and illustrates that use of this framework would permit stating 36 different values for a given flying qualities parameter, even after combining the Flight Phases into three categories.

This detailed breakdown makes the structure parallel the conventional airplane specification, MIL-F-8785B (Reference 10), and helps the phasing into MIL-F-8785B at $V = V_{con}$.

It is unlikely that the full potential of such a fine breakdown of the requirements will ever be required. At the present time only a few requirements make use of more than the Levels breakdown. However, the concept of Flight Phase Category is useful and has been used in clarifying the intent of a number of requirements.

Framework for Stating Flying Qualities Requirements

Class	Flight Phase Category	Level		
		1	2	3
I	A			
	B			
	C			
II	A			
	B			
	C			
III	A			
	B			
	C			
IV	A			
	B			
	C			

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1.1 SCOPE

REQUIREMENT

1.1 Scope. This specification contains the requirements for the flying qualities of U. S. military piloted vertical and short takeoff and landing (V/STOL) aircraft operating at speeds less than V_{con}.

DISCUSSION

The statement 1.1 above is much like that of MIL-F-8785B (Reference 10), with two important exceptions. First, the word aircraft appears in place of airplane; second, the application is limited to V/STOL aircraft operating below the speed V_{con}. The word airplane has had the connotation of aerodynamic lift rather than direct powered lift. For example, the FAA defines the two words as follows:

> "Airplane" means an engine-driven fixed-wing aircraft heavier than air, that is supported in flight by the dynamic reaction of the air against its wings.
>
> "Aircraft" means a device that is used or intended to be used for flight in the air.

It is clear that the definition of "airplane" is too restrictive for V/STOL vehicles since they may not have fixed wings (perhaps no wings at all), and may rely on direct lift from their power plants rather than from wing lift. The word "aircraft" is not restrictive in this sense, and so has been used throughout the specification. It is also apparent that 1.1 and 1.2 do not specify the aircraft's means of support in the atmosphere, i.e., no distinction is made here between rotor lift, tilt-wing lift, direct thrust lift, deflected slipstream lift, etc. Thus it is envisioned that any piloted vehicle, flying at speeds less than V_{con}, will be required to meet the provisions of this specification, while realizing that it is within the prerogative of the procuring activity to impose additional requirements which it feels are necessary to handle the peculiar characteristics of the particular vehicle in question. The underlying philosophy inherent in this concept is that the parameters, and the values of those parameters, which adequately describe a certain level of handling qualities for a given task, are not a function of the means of aircraft support in the atmosphere. To use a simple example, it is believed that the characteristics of the longitudinal dynamic modes of motion necessary to insure a pilot rating of 3.5 or better for the precision hover task are not different for aircraft whose weight, for example, is being supported by a rotating rotor than they are for a vehicle whose weight is being supported by pure jet thrust.

This is not to say that the parameters are insensitive to the means of control. There may very well be differences, in some respects, between an aircraft which tilts to translate and one which has a thrust rotation capability independent of fuselage attitude. It does say, however, that the means of support and the means of control are not necessarily dependent,

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and that the specification should not exclude vehicles which may look like helicopters but have independent thrust vectoring, and neither should it exclude vehicles which may look like airplanes but have to tilt to translate. It is therefore further intended for this document to replace MIL-H-8501A (Reference 15). (It had been hoped to include helicopters universally, thereby completely superseding MIL-H-8501A. However, at this time, only the Air Force has done so, as indicated in 6.6 of Reference 1).

The second, and most important, exception to MIL-F-8785B in the wording of 1.1 above, is the inclusion of "operating at speeds less than the conversion speed (V_{con})."

Conversion Speed, V_{con}

In establishing a definition of V_{con}, it is tempting to base the definition on configuration-related factors such as wing, duct, or rotor tilt angle, lift engines ON or OFF, thrust diverter valve position, etc. Consideration could also be given to what portion of the vertical force required to sustain flight is derived from aerodynamic reactions on fixed surfaces and what portion is derived from engine power through jet, fan, propeller or rotor reaction forces.

For flying qualities specification purposes, however, it is thought that other considerations should form the basis for selecting V_{con}.

Surely it is immaterial that the aircraft has not yet turned off its lift engines or that the wing is not "down and locked", if at these conditions the aircraft can be flown and controlled fundamentally the same way as a conventional airplane. Conversely, if the aircraft can be flown at speeds normally associated with conventional airplanes and is required to perform tasks for which conventional aircraft characteristics are desired, then these speeds should be above V_{con} even if the configuration is not fully converted.

Definition of V_{con} on the basis of configuration change would be a moot point for V/STOL aircraft which do not change configuration. Helicopters are a perfect example of this. Presumably since they never convert, they never reach V_{con} and would therefore never have to satisfy MIL-F-8785B. This is spurious reasoning; it is contended that towards the upper end of their speed range most helicopters are capable of being controlled in a similar way to an airplane (i.e., use of collective control is de-emphasized in flight path control to providing a function similar to a throttle in a conventional airplane rather than being used more as a direct lift control). At such flight conditions it is suggested that most helicopters would be improved in their handling characteristics if they had to meet the requirements of MIL-F-8785B.

The question "when is a short takeoff and landing aircraft a STOL?" has not been given a general unambiguous answer. For the purpose of applying this specification it can be given the answer "when flying at speeds below V_{con}". But for this to have any meaning, V_{con} must be defined in

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general unambiguous terms. Certainly definitions based on configuration related factors will not suffice.

For a flying qualities specification, flying qualities considerations should be the bases on which V_{con} is defined. There are known to be basic changes in control technique as an aircraft is flown at slower and slower speeds: Longitudinally, the pitch control becomes less and less effective at controlling the flight path, and eventually a speed is reached where it is necessary to provide the pilot with some form of direct lift control. Laterally, there is a speed at which the pilot no longer needs to turn in a fully coordinated manner, but is quite happy to make lateral displacements in flight path by slipping or skidding. There are probably other fundamental changes which are equally important, but they will all probably be at speeds less than that at which flight path can be controlled adequately by the pitch control. For this reason, the suggested definitions for V_{con} are based on the ability to control flight path with the pitch control. Thus the philosophy is that when the pilot can control flight path using only the elevator, he will be flying the aircraft like a conventional airplane and should meet the conventional airplane requirements (MIL-F-8785B). When the flight path cannot be controlled adequately using only the pitch control, some form of direct lift control, or minimum vertical thrust control, or autostabilization of airspeed may be required. These alternative control techniques are called for (in a qualitative statement, 3.3.5.2) in the V/STOL specification, Reference 1. Three measures of the ability to control flight path using the pitch control could be related to V_{con} as follows.

1. At V_{con}, an incremental 0.20 g should be available through pitch control. This, of course, can be related to the stall speed for the aircraft state and power setting under consideration.

$$\frac{L}{W} = \left[\frac{V_{con}}{V_s}\right]^2 = 1.20, \text{ thus } V_{con} \doteq 1.10\, V_s.$$

2. The angle of attack and attitude change required to develop the lift increment of 0.20 should not exceed some reasonable limit such as 5 degrees. This would relate to a minimum n_z/α.

$$\frac{\Delta n_z}{\Delta \alpha} = \frac{.20 \times 57.3}{5} \doteq 2.3 \text{ g/radian}$$

3. The steady-state relation between flight path angle and airspeed, when the pitch control is used to constrain airspeed at constant power, has been shown to be a criterion for establishing the speed at which it is necessary to provide the pilot with direct lift control or some minimum vertical thrust response or perhaps autostabilization of airspeed. For Level 1, MIL-F-8785B (Reference 10) requires flight path stability to be such that $d\gamma/dV$ is negative or less positive than 0.06 deg/kt. Thus V_{con} could be defined as the speed at which $d\gamma/dV$ becomes less than 0.06 deg/kt.

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To provide an indication of the sort of speeds which would be involved by choosing the definitions $d\gamma/dV = 0.06$ or $n_z/\alpha = 2.3$, these parameters have been plotted versus speed for four helicopters and four (other) VTOL aircraft [Figures 1 (1.1) through 4 (1.1)] .

The erratic variation of $d\gamma/dV$ for the four VTOL aircraft puts the accuracy of the data in a questionable light. However, the fact that the configuration is different at each speed may be responsible for the scatter. For the helicopters, $d\gamma/dV = 0.06$ would result in V_{con} of approximately 80 knots for the UH-1D and CH-46, and 60 knots for the H-19. There is insufficient data for the H-34A.

Using $n_z/\alpha = 2.3$ gives a V_{con} of 50-60 knots for the helicopters, about 100 knots for the X-19 and XV-5A, 80 knots for the X-22A and 55 knots for the XC-142A.

In Figure 5 (1.1), lines indicating 1.1 V_s and $d\gamma/dV = 0.06$ deg/kt have been drawn on a landing performance plot of the Breguet 941. This curve suggests that the 1.1 V_s would be much too strongly dependent on power setting to be a meaningful definition. The $d\gamma/dV = 0.06$ does, however, define a reasonably constant speed of about 57 knots (for the particular configuration $\delta_f = 98°$).

The speeds defined by $n_z/\alpha = 2.3$ g/rad and $d\gamma/dV = 0.06$ do seem reasonable speeds to expect these various V/STOL's to behave like airplanes. It is perhaps worth pointing out that (if V_{con} is defined as discussed above), though V_{con} would undoubtedly be different at each thrust tilt angle (duct angle in the X-22A) in aircraft such as the X-22A, there is no conceptual problem in applying the specification. One merely satisfies the requirements in MIL-F-8785B or the V/STOL specification, depending on whether the flight condition of interest is above or below V_{con}. Where there is a conflict, the more demanding requirement should apply.

The choice of definition for V_{con} has been the subject of a considerable amount of discussion but is not yet resolved. The adopted version of the specification (Reference 1) provides the following note in paragraph 6.2.2:

> "V_{con} - the speed which establishes the upper limit of applicability of the requirements of this specification and the lower limt of applicability of the requirements of MIL-F-8785. No more precise definition of V_{con} will be attempted as it is assumed that V_{con} will be chosen by the contractor subject to approval by the procuring activity. Factors to be considered in the selection of V_{con} are discussed in the Background Information and User Guide (BIUG); see 6.7."

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Thus the specification references the above discussion and leaves final choice to agreement between the contractor and the procuring activity.

The impact of V_{con} on the definition of flight envelopes is discussed in this document under section 3.1.9.

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$$\frac{d\gamma}{dV} = \frac{1}{g}\left[X_u - \left(X_w - \frac{g}{V}\right)\left(\frac{Z_u}{Z_w}\right)\right]$$

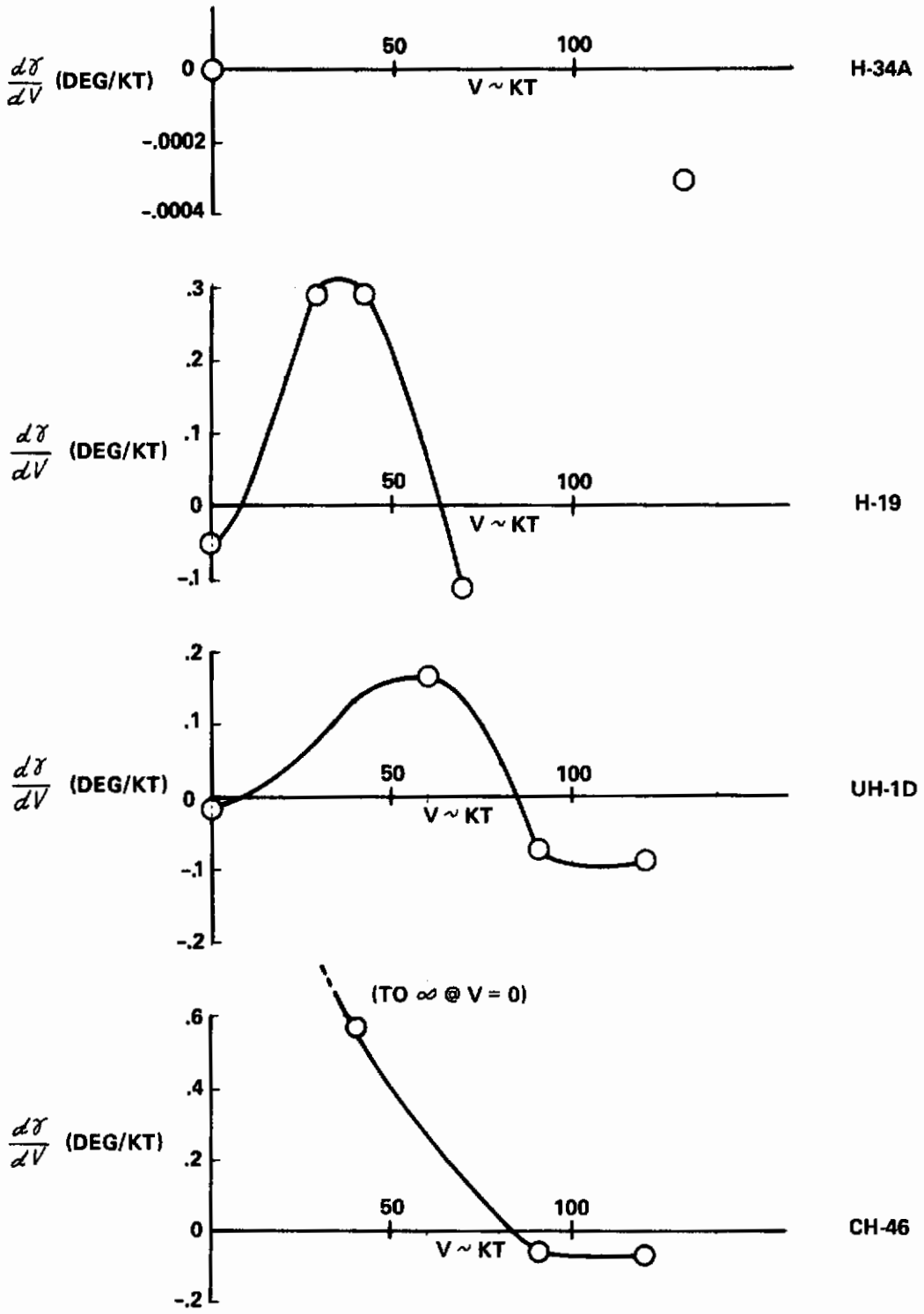

Figure 1 (1.1) VARIATION OF $\frac{d\gamma}{dV}$ FOR FOUR HELICOPTERS (DATA SOURCE REF. 16)

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$$\frac{d\gamma}{dV} = \frac{1}{g}\left[X_u - \left(X_w - \frac{g}{V}\right)\left(\frac{Z_u}{Z_w}\right)\right]$$

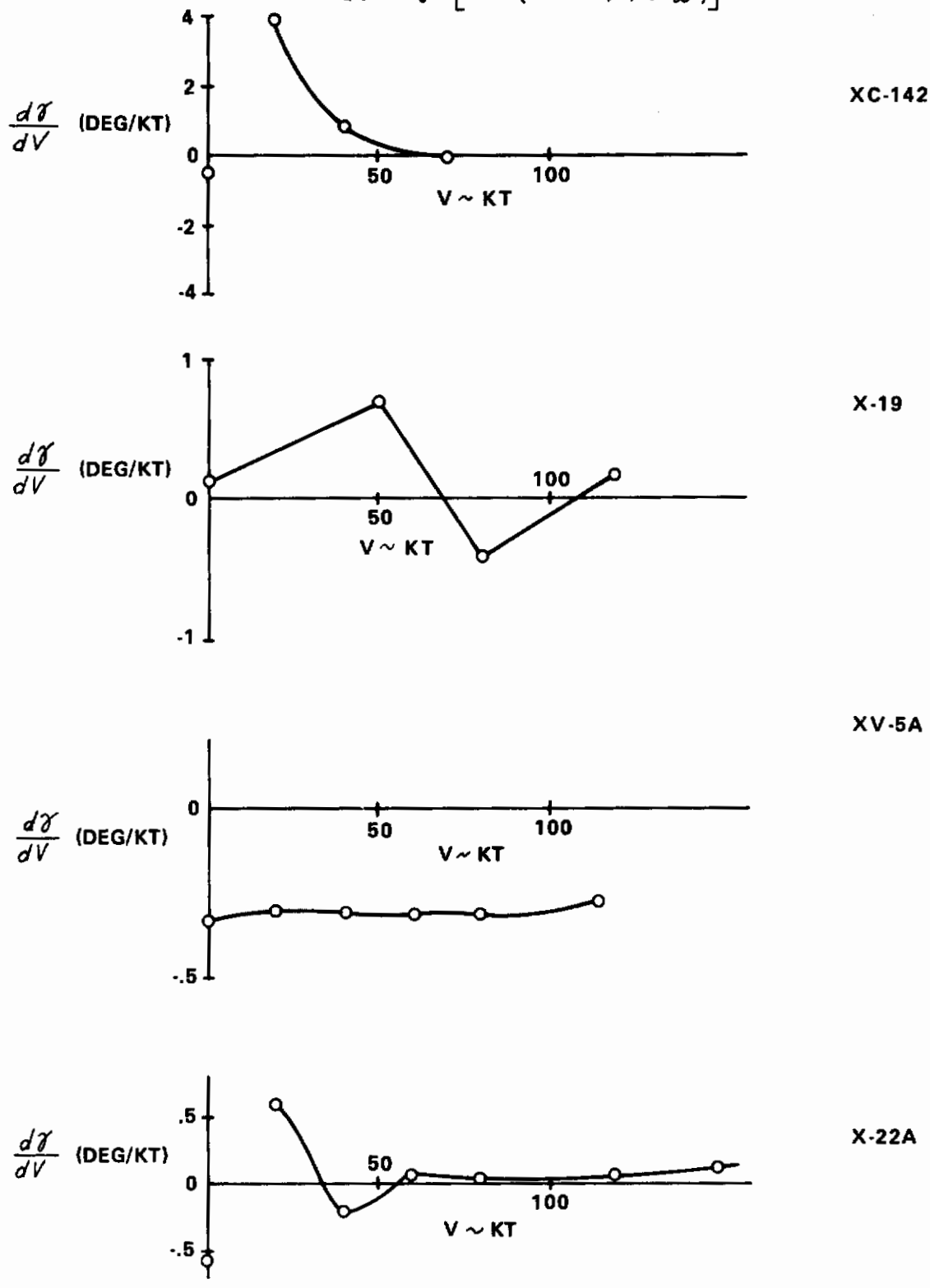

Figure 2 (1.1) VARIATION OF $\frac{d\gamma}{dV}$ FOR FOUR VTOL AIRCRAFT (DATA SOURCE REF. 13)

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$$\frac{n_z}{\alpha} = \frac{V}{g}\left(\frac{Z_{\delta_e} M_w - Z_w M_{\delta_e}}{M_{\delta_e} - \frac{1}{V} Z_{\delta_e} M_q}\right)$$

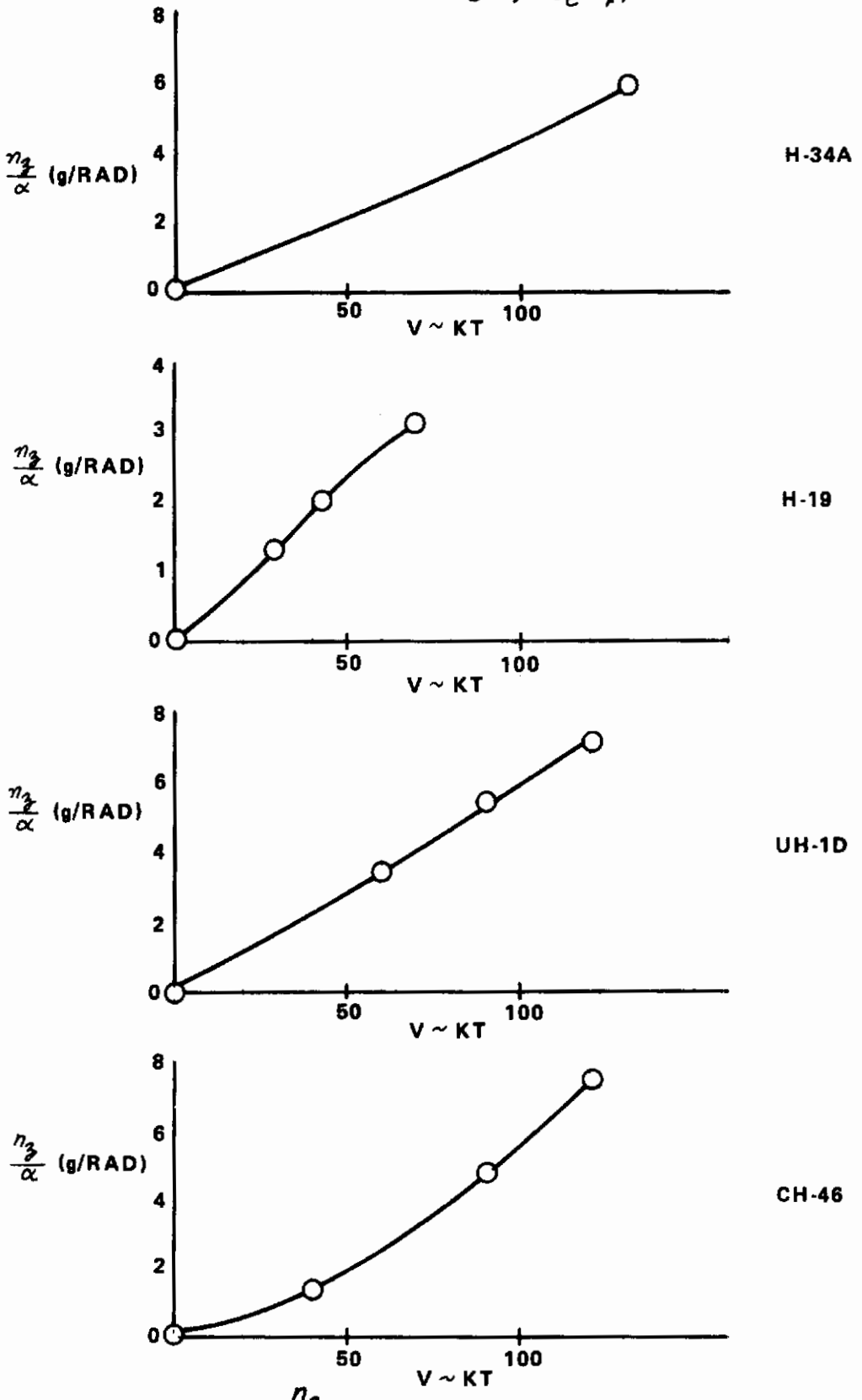

Figure 3 (1.1) VARIATION OF $\frac{n_z}{\alpha}$ FOR FOUR HELICOPTERS (DATA SOURCE REF. 16)

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$$\frac{n_z}{\alpha} = \frac{V}{g}\left(\frac{Z_{\delta e} M_w - M_{\delta e} Z_w}{M_{\delta e} - \frac{1}{V} Z_{\delta e} M_q}\right)$$

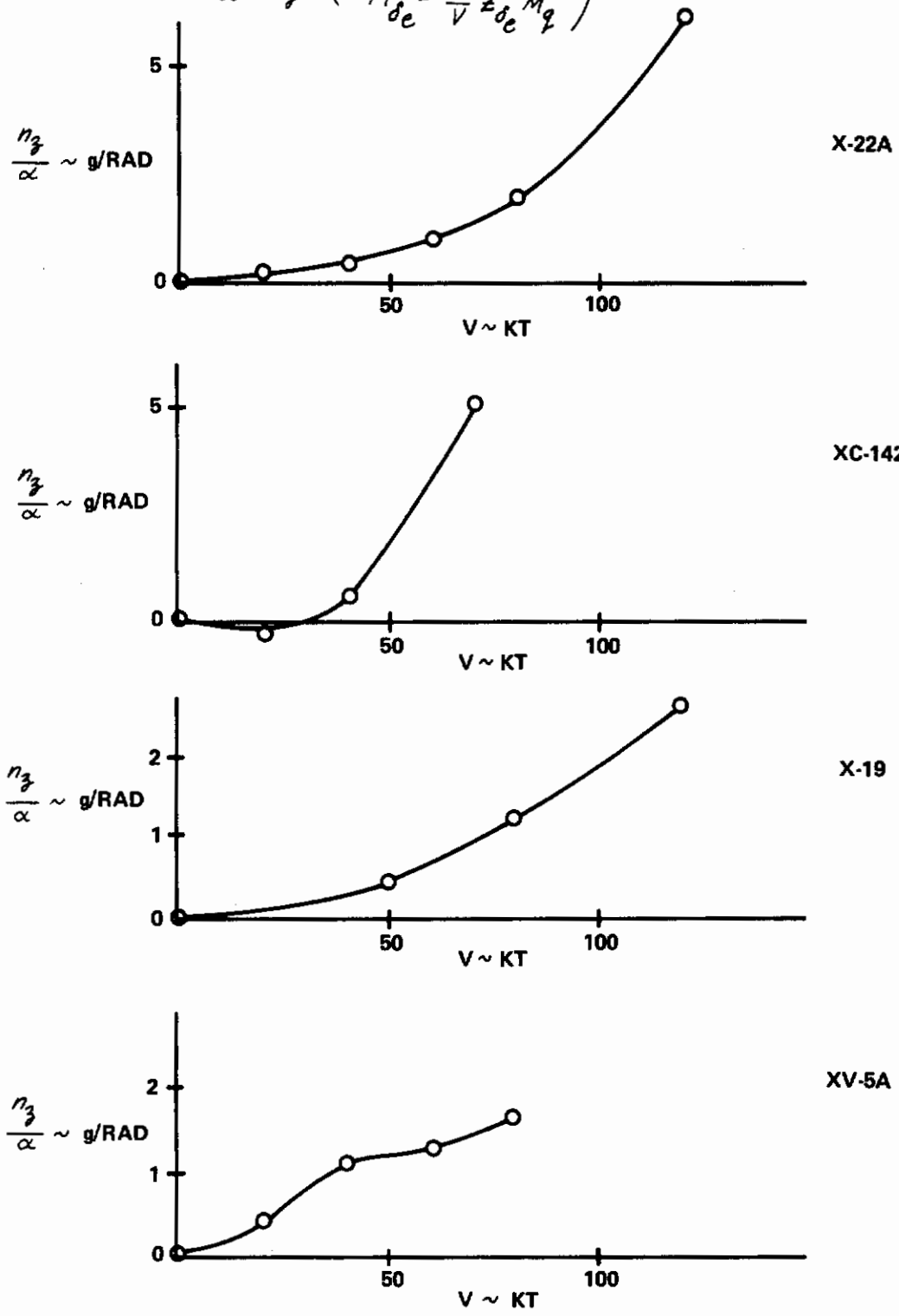

Figure 4 (1.1) VARIATION OF $\frac{n_z}{\alpha}$ FOR FOUR VTOL AIRCRAFT (DATA SOURCE REFERENCE 13)

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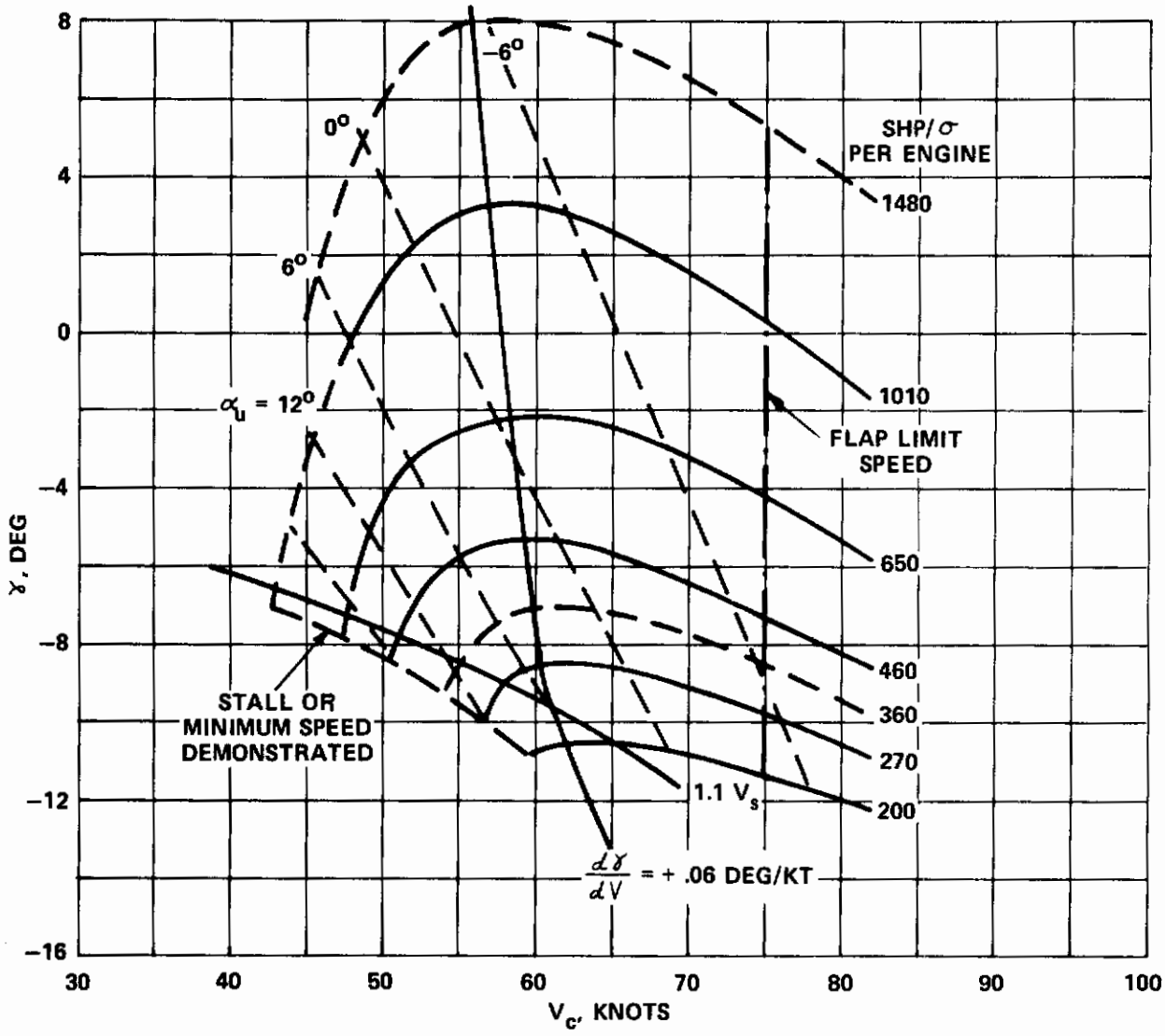

(d) APPROACH AND LANDING CONFIGURATION WITHOUT TRANSPARENCY, 1963 TESTS; $\delta_f = 98^o$.

(DATA SOURCE REF. 17)

Figure 5 (1.1) V_{con} FOR BREGUET 941 AS DEFINED BY 1.1 V_s AND $\frac{d\gamma}{dV}$ = +.06 DEG/KT

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1.2 APPLICATION

REQUIREMENT

1.2 Application. The requirements of this specification shall be applied to assure that no limitations on flight safety or on the capability to perform intended missions will result from deficiencies in flying qualities. The flying qualities for all V/STOL aircraft proposed or contracted for shall be in accordance with the provisions of this specification unless specific deviations are authorized by the procuring activity. Guidance on application of these requirements can be found in the Background Information and User Guide (BIUG) referenced in 6.7. Additional or alternate special requirements may be specified by the procuring activity. For example, if the form of a requirement should not fit a particular vehicle configuration or control mechanization, the procuring activity may at its discretion agree to a modified requirement that will maintain an equivalent degree of acceptability. The requirements of MIL-F-8785B shall apply for operation at speeds in excess of V_{con}.

DISCUSSION

An important clause in this paragraph is: "Additional or special requirements may be specified by the procuring activity. For example, if the form of a requirement should not fit a particular vehicle configuration or control mechanization, the procuring activity may at its discretion agree to a modified requirement that will maintain an equivalent degree of acceptability."

The purpose of this statement is to emphasize the fact that the writers of the specification do not want to inhibit innovation. It is impossible to cover all the possibilities for future V/STOL's; indeed there is insufficient data to cover all the known possibilities. Some requirements, such as the control force gradient limits and control sensitivity limits, are written for conventional stick (or wheel) and rudder pedal controls. They would clearly be inappropriate for a side-arm controller.

In situations such as this the onus will be on the contractor to demonstrate the superiority of his system to the contracting agency.

A general note of explanation of the intended use for specification is given in the "Notes":

> 6.1 Intended use. This specification contains the flying qualities requirements for military piloted V/STOL aircraft operating at speeds up to V_{con} and shall form one of the bases for determination, by the procuring activity, of aircraft acceptability. The specification shall serve as design requirements and as criteria for use in stability and control calculations, analysis of wind-tunnel test results, flying qualities simulation tests, and flight testing and evaluation. To the

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extent possible, this specification should be met by providing an inherently good basic airframe. Where that is not feasible, or where inordinate penalties would result, a mechanism is provided herein to assure that the flight safety, flying qualities and reliability aspects of stability augmentation and other forms of system complication will be considered fully.

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1.2.1 GROUND EFFECT

REQUIREMENT

1.2.1 Ground effect. Requirements are not written specifically for operations in or out of ground effect. The height above ground where compliance must be demonstrated is dictated by the requirements for the particular Flight Phase (1.4) of the operational mission under consideration.

DISCUSSION

It is recognized that when near the ground, V/STOL aircraft can encounter a variety of effects such as an increase or decrease in lift, turbulence, reingestion of exhaust gases, etc. These are generally grouped under the term "ground effect".

The philosophy taken in this specification is that the operational requirements dictate the altitude at which the various Flight Phases have to be performed. If the particular design encounters ground effect within the Operational Envelope, then these effects should not degrade the flying qualities below the required Level. This of course implies that certain types of configurations which may be particularly prone to adverse ground effects may not be acceptable for low level missions.

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1.2.2 OPERATION UNDER INSTRUMENT FLIGHT CONDITIONS

REQUIREMENT

1.2.2 Operation under instrument flight conditions. It is assumed that IFR capability is inherent in all military aircraft operational missions, and therefore the detailed requirements are intended to reflect this assumption. Exceptions to this general assumption are noted in specific requirements.

DISCUSSION

There is general agreement that IFR capability should be provided for all phases of military V/STOL aircraft operation. In developing requirements, the problem is that there is a strong interaction between the level of sophistication with which information is displayed and the corresponding minimum acceptable flying qualities characteristics. It is impossible to predict future display capabilities, but even if future capabilities could be defined, it is currently considered outside the scope of a flying qualities specification to specify display requirements.

There have been some experiments designed to investigate requirements in IFR as compared to VFR, but most of the available data have been generated for VFR tasks. In general, the requirements, especially those of Levels 2 and 3, have been determined with at least qualitative consideration of the instrument flight environment with present types of IFR displays. Hopefully as IFR displays improve, the aircraft will become as easy or easier to fly IFR than VFR.

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1.3 CLASSIFICATION OF AIRCRAFT

REQUIREMENT

1.3 Classification of aircraft. For the purpose of this specification, an aircraft shall be 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

- Utility
- Search and rescue
- 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
- Heavy search and rescue
- Trainer for Class III.

Class IV — High-maneuverability aircraft such as

- Fighter/interceptor
- Attack
- Tactical reconnaissance
- Observation
- Combat search and rescue
- Trainer for Class IV.

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

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DISCUSSION

One intuitively expects sports cars, small sedans, large luxury limousines, and buses to have different handling and riding qualities. But are these differences there by design or by the limits of practical design and economics? It is unlikely that the driver of a small sedan would complain if his car handled like a sports car, neither would the bus driver or passengers complain if the bus felt like a large luxury limousine. The point is that it would be prohibitively expensive to make a bus handle like a luxury limousine and people have grown to expect buses to be rougher, noisier, and slower.

It is suggested that there is an analogy between motor vehicles and aircraft. Pilots expect to see the characteristics of the various types of aircraft that they grew up with repeated in future versions; these characteristics were probably the best that could be achieved, NOT necessarily the most desirable. The correlation of a characteristic such as control power with weight is therefore a reflection of what is practical to obtain rather than the most desirable. Of course, the limits of practicability are severe restraints. It has to be recognized that as aircraft become larger, practical design considerations may dictate compromises between 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. Trends established from such data do indicate that satisfactory aircraft can be built, even though they are a compromise. It also indicates the extent of the compromise.

There are a number of factors which can influence the design compromise, for example:

- the intended operational use.
- the way in which the particular configuration responds to outside disturbances such as turbulence.
- the location of the pilot relative to the center of gravity.

How best to handle these factors 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 some relatively arbitrary Class definitions.

In the case of V/STOL aircraft, an almost unbelievable number of divisions have been suggested. Reference 18 suggests nineteen classes, while Reference 19 more conservatively proposes fifteen. Class systems have been suggested on the basis of:

1. Method of thrust vectoring (vehicle tilting, thrust deflection, dual propulsion, etc.)
2. Type of thrust producer (rotor, fan jet, etc.)
3. Vehicle weight
4. Maximum load factor.

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With this large number of possibilities and no data on which to base a detailed breakdown, it was decided to adopt the structure of MIL-F-8785B (Reference 10). This document has a double breakdown:

Aircraft Classes - primarily based on weight but modified by maneuverability requirements.

Flight Phase Categories - primarily grouped by tasks which are "terminal" or "nonterminal" but again modified by maneuverability requirements.

It is considered that since the breakdown is sufficiently versatile to provide the desired discrimination between conventional aircraft, for which there is a considerable body of data, it should be satisfactory for V/STOL aircraft too. This of course will not be confirmed until the body of background data is built up as experience is gained with various types. For the present time it suffices that the Class breakdown is part of the structure of the specification and it is used in a few instances.

Adopting the same breakdown as is used in MIL-F-8785B does of course provide the additional advantage of keeping both specifications very similar. Users familiar with one will be able to easily begin using the other. It may be noted, however, that it is quite possible for an aircraft to change Class as it goes from the V/STOL Specification to the MIL-F-8785B. The largest U.S. helicopter (Sikorsky S-64E) grosses about 38,000 lb. The Soviet MIL-M1-10K has a gross weight of about 96,000 lb. These would surely have to be classified large, heavy (Class III) V/STOL's. However, they are diminutive when compared to the C-141A (316,000 lb) or the C-5A (764,000 lb), so on a size basis would be medium (Class II aircraft for MIL-F-8785B). From the point of view of applying the specification, this change poses no problem; one merely reads requirements for Class III in the V/STOL Specification and Class II from MIL-F-8785B. Whether or not one would actually want to make the change is a different matter; consider the following examples.

For the helicopters cited above, the maximum speed is about 110 to 120 mph and the bulk of their mission will be at speeds much less than this. In this case it may not even be worth satisfying the MIL-F-8785B requirements at all (i.e., make $V_{con} > 120$ mph). If one decided to make $V_{con} < 120$ mph it would certainly be reasonable to expect these helicopters to be relatively ponderous and to put them in Class III in MIL-F-8785B even though they are only a tenth and twentieth respectively, of the weight of a C-5A.

If we were considering a 100,000 lb (DC-9 size) jet lift transport V/STOL instead of a helicopter, it would still be in Class III in the V/STOL Specification but would almost certainly be required to meet the Class II requirements of MIL-F-8785B at speeds above V_{con} where it is doing the same jobs as a DC-9.

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Perhaps a general way of indicating the intent is that the Class should correspond to the dominant role in the speed range of interest.

Users familiar with MIL-F-8785B (Reference 10) will notice that the V/STOL Specification has not incorporated the distinction between Land- (L) and Carrier- (C) based aircraft. It was decided that with the present lack of operational experience, few if any general distinctions could be made between the flying qualities needed by land- or carrier-based V/STOL aircraft. As specific aircraft are procured, the operational requirements specified by the procuring activity will take care of differences that can be defined.

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1.4 FLIGHT PHASE CATEGORIES

REQUIREMENT

1.4 Flight Phase Categories. The Flight Phases have been combined into three Categories which are referred to in the requirement statements. These Flight Phases shall be considered in the context of total missions so that there will be no gap between successive Phases of any flight. 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 shall apply to all three Categories. Flight Phases descriptive of most military aircraft missions are:

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)
j. Precision hover (PH)

Category B - Those nonterminal Flight Phases that are normally accomplished using gradual maneuvers and without precision tracking, although accurate flight-path control may be required. Included in this Category are:

a. Climb (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)
i. Hover (H)
j. Nonterminal transition (NT)

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Terminal Flight Phases:

Category C - Terminal Flight Phases that are normally accomplished using gradual maneuvers and usually require accurate flight-path control. Included in this Category are:

a. Vertical takeoff (VT)
b. Short takeoff (ST)
c. Approach (PA)
d. Wave-off/go-around (WO)
e. Vertical landing (VL)
f. Short landing (SL)
g. Terminal transition (TT)

When necessary, recategorization or addition of Flight Phases or delineation of requirements for special situations will be accomplished by the procuring activity.

DISCUSSION

Much of the discussion on Classes of aircraft (1.3) is pertinent to understanding the context in which Flight Phase Categories have been introduced into the V/STOL Specification.

The introduction of Flight Phase Categories brings the basic structure into line with the conventional airplane specification MIL-F-8785B (Reference 10). The concept is useful in generalizing the statement of certain requirements even though, with the presently available information, few requirements make any distinction for Flight Phase.

The Flight Phase definitions and groupings into categories are similar to those in MIL-F-8785B (Reference 10) except that additional Flight Phases, special to V/STOL aircraft, have been added. These are, of course, the hover and transition Flight Phases.

Experience with conventional 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). In many instances, therefore, the flying qualities specification should state requirements as a function of mission Flight Phase.

A mission will generally consist of a series of Flight Phases. A Flight Phase may be thought of as further subdivided into a series or group of tasks. For example, the Power Approach Flight Phase (PA) may involve tracking the glide slope, longitudinal trim changes and heading changes. In ground and flight simulator experiments, the pilot's overall rating of the acceptability of an aircraft's handling qualities for a particular Flight Phase is assigned on the basis of his evaluation of the effort and ability required to perform the related tasks. A discussion of the relationship of tasks to Flight Phases is given in Reference 19. The concept of a mission being made

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up from a series of Flight Phases led naturally to a statement of MIL-F-8785B flying qualities requirements in terms of the Flight Phases of 1.4. It is expected that as the data base is expanded, a similar use will be made of the Flight Phase Category breakdown in the V/STOL Specification.

Not all of the listed 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 have to delete some Phases and add others. Responsibility for choosing applicable Flight Phases should be defined contractually. The procuring activity should prepare the initial listing of Flight Phases. The contractor should be made contractually responsible for assuring that this listing is inclusive and exhaustive (for each operational mission for which the aircraft is designed), and for suggesting necessary additions so that the intent of the Flight Phase concept (i.e., there will be no gap between successive Phases of every flight, and the act of changing from Phase to Phase during a flight will be smooth) will be accomplished. It is the procuring activity's responsibility either to agree with the contractor's suggestions, to recategorize the Flight Phases, or to add Flight Phases to the listing supplied by the contractor.

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1.5 LEVELS OF FLYING QUALITIES

REQUIREMENT

1.5 Levels of flying qualities. 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.

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.

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.

DISCUSSION

The phrase "where possible" was used in 1.5 because, after considerable literature searching, the data available were inadequate to permit a rational statement of three values for every requirement. As more data become available, further separation of requirements into levels of acceptability should be achievable.

Amplification on Level usage is given in 6.5.2: "Level definitions. To determine the degradation in flying qualities parameters for a given Airplane Failure State the following definitions are provided:

a. Level 1 is better than or equal to the Level 1 boundary, or number, given in section 3.

b. Level 2 is worse than Level 1, but no worse than the Level 2 boundary, or number.

c. Level 3 is worse than Level 2, but no worse than the Level 3 boundary, or number.

When a given boundary, or number, is identified as Level 1 and Level 2, this means that flying qualities outside the boundary conditions shown, or worse than the number given, are at best Level 3 flying qualities...."

Application of Levels is discussed in Section 3.1.10.

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Qualitative Requirements

According to Section 4.2, the Level definitions of 1.5 are to be used directly in determining compliance with qualitative requirements.

There is a direct association between the three Levels of acceptability and the pilot rating scale recently developed by Cooper and Harper. The definitions of the three Levels in 1.5 were originally developed from an interim version of this scale, published in September 1966 (Reference 20). Since that time, the rating scale has been refined and republished after review by interested individuals and organizations in the U.S., Britain, and France (Reference 21). The refinements in the latest version deal mainly with details of language, however, and not the basic structure of the scale, i.e., the basic decision process related to mission Flight Phase accomplishment remains unchanged from that of Reference 20. The revised Cooper-Harper rating scale from Reference 21 is reproduced in Figure 1 (1.5). An alternative approach to rating scale development and rating correlations with pilot, vehicle, and system parameters is presented in Reference 22.

Although a direct association is intended between the Levels of Reference 1 and the Cooper-Harper rating scale in Figure 1 (1.5) the association with previous rating scales is not as direct. Since the majority of the experimental flying qualities data available at this time was produced using either the original Cooper scale or one of several scales employed by Cornell Aeronautical Laboratory, it was necessary to examine the context and the results of each experiment in detail before making associations between Levels and a particular pilot rating scale.

In general, however, the following association has been used between Levels and the major rating scales:

Level	Original Cooper Scale Table 6(3.2.2.1)	Standard CAL Scale	Interim Cooper-Harper Scale (Ref. 20) Table 7(3.2.2.1)	Final Cooper-Harper Scale (Ref. 21) Table 1(1.5)
1	1-3.5	1-3.5	1-3.5	1-3.5
2	3.5-5.5	3.5-6.5	3.5-6.5	3.5-6.5
3	5.5-7	6.5-9+	6.5-9+	6.5-9+

If a rating scale other than those shown above is used, the association between Levels is explained, along with the particular set of data under consideration.

Because the base configurations for parametric studies were different for different experiments, it was sometimes helpful to give consideration to the rate of change of pilot rating with a given parameter in establishing the association between parameter values and Levels. In addition, the selection of parameter values to use for Level 3 was sometimes tempered with philosophy and not strictly based on experimentally defined controllability limits.

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The evaluations and analyses on which many of the requirements are based were conducted with "good" values of all parameters except the ones that were varied. But Cooper and Harper have noted that the combined rating degradation caused by two or more poor flying qualities parameters can be significantly worse than the degradation caused by any one of the parameters. Such degradation is not always found, but it is a worrisome problem. For example, if Level 3 is barely met in several respects, the aircraft may be unflyable. Some Level 3 requirements have been stiffened arbitrarily, partly to account for this possibility. There is usually too little data to treat the problem more accurately, though some trends have been established in the development of the lateral-directional requirements (paragraph 3.3.8).

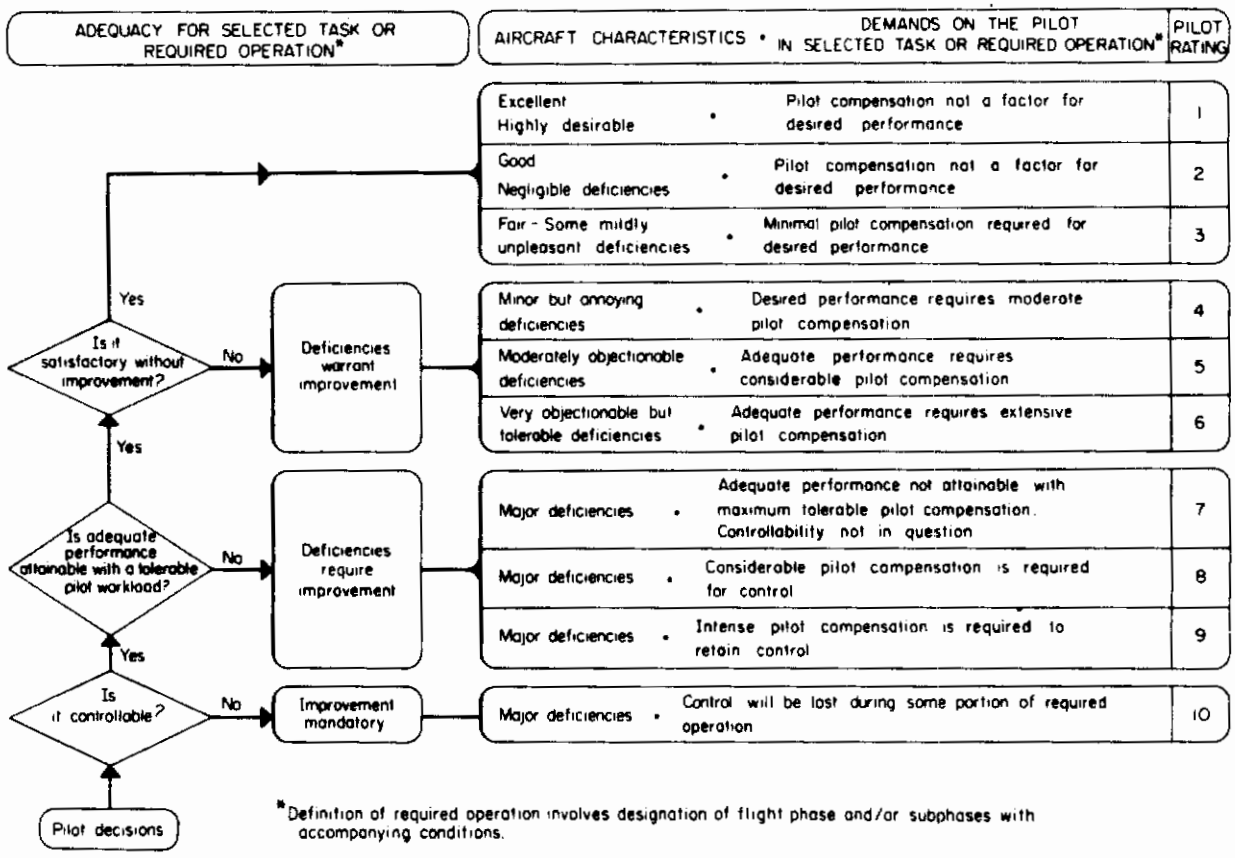

TABLE 1(1.5)
COOPER-HARPER RATING SCALE (FROM REFERENCE 21)

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2. APPLICABLE DOCUMENTS

2.1 The following documents, 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:

SPECIFICATIONS

Military

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-W-25140	Weight and Balance Control Data (for Airplanes and Rotorcraft)
MIL-F-8785	Flying Qualities of Piloted Airplanes

STANDARDS

MIL-STD-756	Reliability Prediction

(Copies of documents required by suppliers in connection with specific procurement functions should be obtained from the procuring activity or as directed by the contracting officer.)

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3. REQUIREMENTS

3.1 GENERAL REQUIREMENTS

REQUIREMENT

3.1 General requirements

DISCUSSION

The form of Section 3.1 closely parallels the conventional airplane specification MIL-F-8785B (Reference 10).

To avoid obfuscation in applying the concepts in this section, the user must keep in mind the fact that the definitions of Flight Envelopes apply to fixed operating point flight conditions, not the transition maneuver itself. For example, a tilt wing V/STOL may have the capability of flying partially converted (e.g. with wing tilt angle of 30°). In this configuration the concepts of flight envelopes are identical to those of a conventional airplane, though of course, the flight speed may be lower.

In general, Section 3.1 specifies the conditions under which the requirements of the specification apply. The main factors are determined by the operational missions for which the aircraft is to be designed. The aircraft described by its Aircraft State (weight, center-of-gravity position, external store complement, configuration and thrust setting together with the operational status of the components and systems), must meet the specified requirements under various conditions of speed, altitude and load factor.

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3.1.1 OPERATIONAL MISSIONS

REQUIREMENT

3.1.1 Operational missions. The procuring activity will specify the operational missions to be considered by the contractor in designing the aircraft to meet the flying qualities requirements of this specification. These missions will include the entire spectrum of intended operational usage.

DISCUSSION

The word "mission" unfortunately 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 categorizing the aircraft as fighter, bomber, reconnaissance, etc., or as in "accomplishing the mission" of bombing, strafing, etc. In 3.1.1 the object is to introduce to the designer in general terms the function of the aircraft he is to design. It should be sufficient for the procuring activity to refer to those paragraphs of the Systems Specification and Air Vehicle Specification which contain the overall performance requirements, the operational requirements, employment and deployment requirement (generally Sections 3.1 and 3.2 of those documents). The operational missions should be based on the above considerations as well as the mission profiles to be used for performance guarantees.

The operational missions considered should not be based on just the design mission profiles. But these profiles may be a starting point for determining variations that might normally be expected in service use while performing missions of the same character. Thus the procuring activity should examine 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" are intended to include training missions.

The intended use of an aircraft must be known before the required Operational Flight Envelopes can be defined and the design of the aircraft to meet the requirements of this specification undertaken. Should the using command decide to use an aircraft for an operational mission other than for which it was designed, the responsibility must be assumed by the using command since the aircraft designer can only be held responsible for the requirements specified in the contract covering procurement of the aircraft. 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 Operational Flight Envelopes for which the aircraft is to be designed. Once the intended uses

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or operational missions are defined, a Flight Phase analysis of each mission must be conducted and the Operational Flight Envelopes defined. The configurations and loading states required to perform the various Flight Phases throughout the corresponding Operational Flight Envelopes can then be defined and the Service and Permissible Flight Envelopes determined.

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3.1.2 LOADINGS

3.1.3 MOMENTS OF INERTIA

3.1.4 EXTERNAL STORES

REQUIREMENTS

3.1.2 Loadings. The contractor shall define the envelopes of center of gravity and corresponding weights that will exist for each Flight Phase. These envelopes shall include the most forward and aft center-of-gravity positions as defined in MIL-W-25140. In addition, the contractor shall determine the maximum center-of-gravity excursions attainable through failures in systems or components, such as fuel sequencing, hung stores, etc., for each Flight Phase to be considered in the Failure States of 3.1.6.2. Within these envelopes, plus a growth margin to be specified by the procuring activity, and for the excursions cited above, this specification shall apply.

3.1.3 Moments of inertia. The contractor shall define the moments of inertia associated with all loadings of 3.1.2. The requirements of this specification shall apply for all moments of inertia so defined.

3.1.4 External stores. The requirements of this specification shall apply for all combinations of external stores required by the operational missions. 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 specification apply throughout the range of store loadings. The external stores and store combinations to be considered for flying qualities design will be specified by the procuring activity. In establishing external store combinations to be investigated, consideration shall be given to asymmetric as well as to symmetric combinations, and to various methods of attachment to the airframe (e.g., single-point sling, multi-point sling, rigid, etc.).

DISCUSSION

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.

The requirements apply under all loading conditions associated with an aircraft's operational missions. Since there is an infinite number of possible internal and external loadings, each requirement generally is only examined at the critical loading with respect to the requirement. Only permissible center-of-gravity positions need be considered for Aircraft Normal States. But fuel sequencing and transfer failures or malperformance that get 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 (3.1.6.2.1) must be examined.

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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 structural 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.

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.

The designer should attempt to assure that there are no restrictions on store loading, 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.

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3.1.5 CONFIGURATIONS

REQUIREMENT

3.1.5 Configurations. The requirements of this specification shall apply for all configurations required or encountered in the applicable Flight Phases of 1.4. A (crew-) selected configuration is defined by the positions and adjustments of the various selectors and controls available (except for the pitch, roll, yaw, thrust magnitude, and trim controls), for example, flap setting, wing-angle setting, duct-rotation setting, nozzle setting, stability-augmentation-system (SAS)-selector setting, etc. The selected configurations to be examined must consist of those required for performance and mission accomplishment. Additional configurations to be investigated may be defined by the procuring activity.

DISCUSSION

The settings of such controls as thrust vector angle, (wing tilt, nozzle angle, etc.), flaps, speed brakes, landing gear are related uniquely to each aircraft design. The specification requires that the configurations to be examined shall be those required for performance and mission accomplishment. The position of roll, pitch, yaw controls, trim controls and the thrust magnitude control 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.

Where a distinction is required, the requirements are stated for Flight Phases, rather than for aircraft configurations, since the flying qualities should be a function of the job to be done rather than of the configuration of the aircraft. However, the designer must define the configuration or configurations which his aircraft will have during each Flight Phase.

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3.1.6 STATE OF THE AIRCRAFT

3.1.6.1 AIRCRAFT NORMAL STATES

3.1.6.2 AIRCRAFT FAILURE STATES

3.1.6.2.1 AIRCRAFT SPECIAL FAILURE STATES

REQUIREMENTS

3.1.6 State of the aircraft. The State of the aircraft is defined by the selected configuration together with the functional status of each of the aircraft components or systems, thrust magnitude, 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. The position of the thrust magnitude control shall not be considered an element of the Aircraft State when the thrust magnitude is specified in a requirement.

3.1.6.1 Aircraft Normal States. The contractor shall define and tabulate all pertinent items to describe the Aircraft Normal (no component or system failure) State(s) associated with each of the applicable Flight Phases. This tabulation shall be in the format and shall use the nomenclature shown in 6.2. Certain items, such as weight, moments of inertia, center-of-gravity position, thrust magnitude and thrust angle control settings, 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.

3.1.6.2 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 3.1.2 shall be included. Each mode of failure shall be considered. Failures occurring in any Flight Phase shall be considered in all subsequent Flight Phases.

3.1.6.2.1 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 the Failure States as Special is submitted by the contractor and approved by the procuring activity.

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 XIV. A

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TABLE XIV AIRCRAFT NORMAL STATES (FROM REFERENCE 1)

Flight Phase		Weight	C.G.	External Stores	Thrust Magnitude	Thrust Angle	High Lift Devices	Wing Sweep	Wing Incidence	Landing Gear	Speed Brakes	Bomb bay or Cargo Doors	Stability Augmentation	Other
Vertical Take-off	VT													
Short Take-off	ST													
Hover	H													
Precision Hover	PH													
Climb	CL													
Cruise	CR													
Loiter	LO													
Descent	D													
Emergency Descent	ED													
Emergency Deceleration	DE													
Non-terminal Transition	NT													
Terminal Transition	TT													
Approach	PA													
Wave-off/Go-Around	WO													
Vertical Landing	VL													
Short Landing	SL													
Air-to-air Combat	CO													
Ground Attack	GA													
Weapon Delivery/Launch	WD													
Aerial Delivery	AD													
Aerial Recovery	AR													
Reconnaissance	RC													
In-flight Refuel (Receiver)	RR													
In-flight Refuel (Tanker)	RT													
Terrain Following	TF													
Antisubmarine Search	AS													
Close Formation Flying	FF													

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particular design may have other variable features such as air inlets; if the position of any such feature can affect flying qualities independently of the items in Table XIV, 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 3.1.2 - 3.1.4, center-of-gravity positions that can be attained only with prohibited, failed, or malfunctioning fuel sequencing need not be considered for Aircraft Normal States.

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. Events of "unlikely" origin from recent flight experience are listed as illustrations:

- Failure of one bracket that held lines from both hydraulic systems led to loss of integrity of both systems.
- An extinguishable fire that burned through lines from all hydraulic systems, that were routed through the same compartment.
- Spilled coffee on the pilots' console that shorted out all electrical systems; lightning strikes might do this, too.
- A loose nut (too thick a washer was used, so the self-locking threads were not engaged) which shorted all three stability augmentation channels of a triply redundant system.
- Undetected impurities in a batch of potting compound used in packaging stability augmentation system components; all affected channels shorted out at the high temperatures of supersonic flight, after passing ground checkout.
- Complicated ground checkout equipment and lengthy procedures that were impractical to use very frequently on the flight line, resulting in long flight times between flight control system electronics checks.

The insidious nature of possible troubles emphasizes the need for caution in design application.

In discussing redundant systems, it is axiomatic that the whole system must be redundant. However, a recent design used multiple-redundant SAS, but required environmental control for the electronic components; the environmental control system was not redundant. Thus the complex multiple-redundant SAS could have been put out of action by any failure of the air conditioning equipment.

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When considering the necessity of redundancy, attention should not be focused on the control system to the exclusion of all else. For example, it may be necessary to duplicate certain essential instrumentation. The SV-5 had an extremely narrow angle-of-attack corridor during re-entry, but had only one angle-of-attack sensing vane and display. In such a case, where the information is so essential, redundancy may be warranted.

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.

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

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.

Note that the approval of Aircraft Special Failure States is at the discretion of the procuring activity. In conjunction with certain requirements that must be met regardless of component or equipment status, granting or refusing approval can be used as desired to require a level of stability for the basic airframe, to rule out fly-by-wire control systems, to demand consideration of vulnerability, or even to rule out a type of configuration. For example, a propeller pitch control failure on a V/STOL which uses differential thrust from left and right hand side propellers to provide roll control in hover, will result in loss of control; clearly no requirements can then be met, and the configuration is excluded, unless the pitch control failure is allowed as a special failure. The procuring activity should state the considerations to be imposed, as completely as possible at the outset; but it is evident that many decisions must be made subjectively and many will be influenced by the specific design.

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3.1.7 OPERATIONAL FLIGHT ENVELOPES

3.1.8 SERVICE FLIGHT ENVELOPES

3.1.9 PERMISSIBLE FLIGHT ENVELOPES

GENERAL DISCUSSION

The emphasis on Flight Envelopes is an attempt to restrict application of the requirements to regions in which compliance is essential. Thus, it is hoped to avoid the performance, cost and complexity penalties that might be associated with overdesign to provide excellent flying qualities at all flight conditions. Just as important, the Flight Envelopes should ensure that flying qualities will be acceptable wherever the aircraft is operated. In general, the boundaries of these envelopes should not be set by ability to meet the flying qualities requirements. Other factors will normally determine the boundaries unless specific deviations are granted. The rationale for each type of Envelope is presented later in the discussion of each paragraph; but here it is in order to discuss procedures in constructing and using the Envelopes, particularly as they should be interpreted for V/STOL aircraft as against conventional aircraft.

To start with, the procuring activity must set down the capability it wants for primary and alternate missions, including maneuverability over the speed-altitude range. These are the minimum requirements on the Operational Flight Envelopes. At this stage the Flight Phases will be known. In response to these and other requirements, a contractor will design an aircraft. For that design the contractor can relate the Flight Phases to Airplane Normal States, then:

- 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 conditions related to any pertinent performance guarantees.

What does all this mean for a V/STOL?

Consider a V/STOL which has the operation requirements indicated on Figure 1 (3.1).

At a particular altitude, a typical V/STOL aircraft will be able to perform the maneuvering requirements corresponding to a given speed and altitude at a range of configurations (wing tilt angle, duct angle, nozzle setting, etc.). Thus an additional dimension which depends on the configuration is introduced into the Flight Envelope. For an aircraft with a single configuration variable λ, the operational boundaries of Figure 1 (3.1) would become as shown on Figure 2 (3.1).

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Thus at each λ there would be a range of speeds over which the aircraft can be safely flown at the altitude being considered. The extremes of this range define the maximum and minimum service speeds for that configuration $[V_{max}(\lambda), V_{min}(\lambda)]$.

Also at each λ there is a range of speeds over which the operational requirements of a particular Flight Phase can be satisfied at this altitude. The extremes of this range define the maximum and minimum operational speeds for that particular configuration; they are NOT necessarily $V_{o\,max}$ and $V_{o\,min}$ for the particular Flight Phase.

Conversely, at a given speed there is a range of configurations at which the operational requirements of the Flight Phase can be satisfied.

The requirements of the specification apply at all points within the three-dimensional volume (speed, altitude and normal load factor, and possibly additional parameters such as rate of descent, flight path angle or side velocity) of the Flight Envelope, and also within the range of configurations. Hence, in effect, the requirements apply to a four-dimensional volume (or more if there is more than one independent configuration variable, e. g., wing tilt angle and flap angle would be two variables unless uniquely related). In picking the conditions within this four-dimensional space at which to determine compliance, consideration should be given to the critical flight conditions and how the aircraft will be flight tested.

Each Flight Phase will involve a range of loadings. Generally it will be convenient to represent this variation by superimposing boundaries for the discrete loadings of Table XIV, or possibly by bands denoting extremes. If different external store complements affect the Envelope boundaries significantly, it may be necessary to construct several sets of Envelopes for each Flight Phase, each set representing a family of stores. Hopefully 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 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 (paragraph 1.5). These Level boundaries should aid the using commands in tactical employment, even long after the procurement contract has been closed out.

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Separate Flight Envelopes are not normally allowed 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 thrust tilt angle failure that necessitates a partially converted landing) 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.

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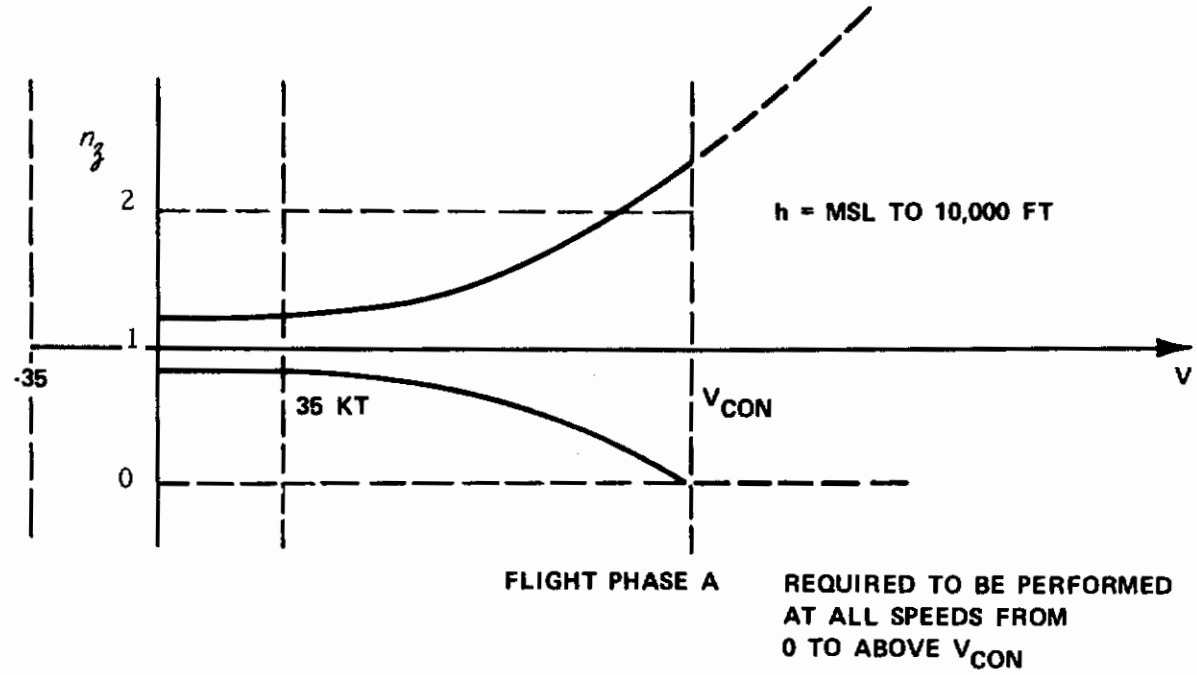

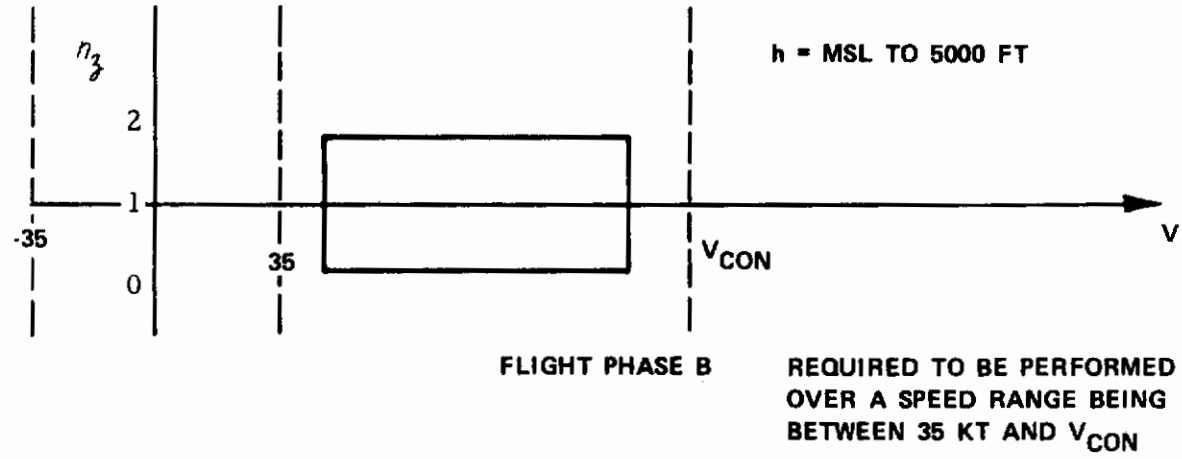

Figure 1 (3.1) OPERATIONAL FLIGHT ENVELOPES FOR FLIGHT PHASES A AND B

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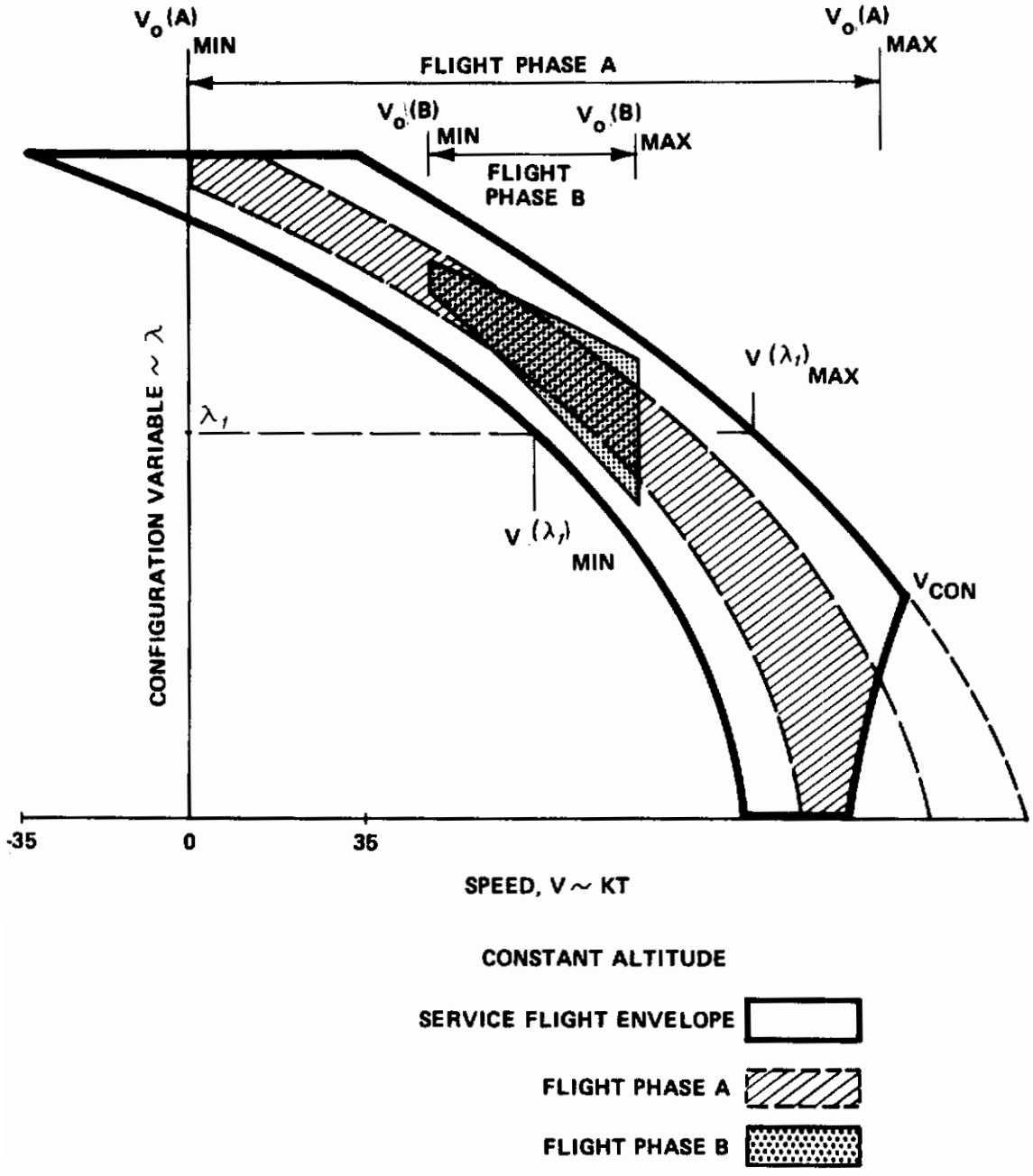

Figure 2 (3.1) OPERATIONAL FLIGHT ENVELOPES FOR FLIGHT PHASES A AND B, AND SERVICE FLIGHT ENVELOPE

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3.1.7 OPERATIONAL FLIGHT ENVELOPES

REQUIREMENT

3.1.7 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 3.1.1. Additional envelopes in terms of parameters such as rate of descent, flight-path angle, and side velocity may also be specified. Envelopes for each applicable Flight Phase shall be established with the guidance and approval of the procuring activity. In the absence of specific guidance, the contractor shall use the representative conditions of table I for the applicable Flight Phases.

DISCUSSION

Operational Flight Envelopes are regions in speed-altitude-load factor space (additional parameters such as rate of descent, flight path angle and side velocity may also be specified) where it is necessary for an aircraft, in the configurations 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 ensure that the aircraft can be controlled without undue concentration. The Operational Flight Envelopes are intended to permit the design task to be more closely defined. As a result, the cost and complexity of the aircraft and possibly the cost and time required for flight testing should be appreciably, but logically, reduced. The required size of the Operational Flight Envelopes for a particular aircraft should, to the extent possible, be given in the detail specification for the aircraft, but some boundaries will only be delineated during design of the weapon system. 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 by the using command.

(b) If design trade-offs indicate that significant penalties (in terms of performance, cost, system complexity, or reliability) are required to provide Level 1 flying qualities in the large Envelope of (a) 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.

Guidance for establishing Operational Flight Envelopes for various Flight Phases is contained in Table I of Reference 1. Almost all of the Flight Phases listed might conceivably make good use of extremely low speeds. Because experience is lacking, however, the parameter ranges

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TABLE I
OPERATIONAL FLIGHT ENVELOPE (FROM REFERENCE 1)

FLIGHT PHASE CATEGORY	FLIGHT PHASE	AIRSPEED		ALTITUDE		LOAD FACTOR	
		$V_{o_{min}}$	$V_{o_{max}}$	$h_{o_{min}}$	$h_{o_{max}}$	$n_{o_{min}}$	$n_{o_{max}}$
	AIR-TO-AIR COMBAT (CO)	X	V_{con}	MSL	COMBAT CEILING	-1.0	n_L
	GROUND ATTACK (GA)	0	V_{con}	MSL	MEDIUM	-1.0	n_L
	WEAPON DELIVERY LAUNCH (WD)	V_{range}**	V_{con}	MSL	COMBAT CEILING	0.5	*
A	AERIAL RECOVERY (AR)	0	V_{con}	MSL	COMBAT CEILING	0.5	n_L
	RECONNAISSANCE (RC)	35 KT	V_{con}	MSL	COMBAT CEILING	*	*
	IN-FLIGHT REFUELING (RECEIVER) (RR)	35 KT	V_{con}	MSL	COMBAT CEILING	0.5	2.0
	PRECISION HOVERING (PH)	X	35 KT	MSL	COMBAT CEILING	1.0	1.0
	TERRAIN FOLLOWING (TF)	V_{range}**	V_{con}	MSL	10,000 FT	0	3.5
	ANTISUBMARINE SEARCH (AS)	35 KT	V_{con}	MSL	MEDIUM	0	2.0
	CLOSE FORMATION FLYING (FF)	35 KT	V_{con}	MSL	COMBAT CEILING	-1.0	n_L
	CLIMB (CL)	0	V_{con}	MSL	CRUISING CEILING	0.5	2.0
	CRUISE (CR)	V_{range}**	V_{con}	MSL	CRUISING CEILING	0.5	2.0
	HOVER (H)	X	35 KT	MSL	CRUISING CEILING	1.0	1.0
	LOITER (LO)	.85 V_{end}**	1.3 V_{end}**	MSL	CRUISING CEILING	0.5	2.0
B	IN-FLIGHT REFUELING (TANKER) (RT)	35 KT	V_{con}	MSL	CRUISING CEILING	0.5	2.0
	DESCENT (D)	0	V_{con}	MSL	CRUISING CEILING	0.5	2.0
	NONTERMINAL TRANSITION (NT)	X	V_{con}	MSL	SERVICE CEILING'	0.5	2.0
	EMERGENCY DESCENT (ED)	0	V_{con}	MSL	CRUISING CEILING	0.5	2.5
	EMERGENCY DECELERATION (DE)	0	V_{con}	MSL	CRUISING CEILING	0.5	2.0
	AERIAL DELIVERY (AD)	X	V_{con}	MSL	10,000 FT	0	2.0
C	VERTICAL TAKEOFF (VT)	X	45 KT	MSL	10,000 FT	1.0	2.0
	SHORT TAKEOFF (ST)	**	V_{con}	MSL	10,000 FT	0.5	n_L
	APPROACH (PA)	MINIMUM NORMAL APPROACH SPEED	V_{con}	MSL	10,000 FT	0.5	2.0
	WAVE-OFF/ GO-AROUND (WO)	MINIMUM NORMAL APPROACH SPEED	V_{con}	MSL	10,000 FT	0.5	2.0
	VERTICAL LANDING (VL)	X	45 KT	MSL	10,000 FT	0.5	2.0
	SHORT LANDING (SL)	MINIMUM NORMAL LANDING SPEED**	V_{con}	MSL	10,000 FT	0.5	2.0
	TERMINAL TRANSITION (TT)	X	V_{con}	MSL	SERVICE CEILING**	0.5	2.0

' APPROPRIATE TO THE OPERATIONAL MISSION.

X - AIRSPEED IS 35 KT IN ANY DIRECTION

** BASED ON SELECTED CONFIGURATION.

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shown are quite approximate; for each application, careful attention should be paid to refining these boundaries. Information on the intended use of the aircraft (required operational capability) should facilitate significantly more precise definitions of the various limits to be made for a specific application. Figure 6 of Reference 1 illustrates a possible Operational Flight Envelope for non-terminal transition (NT) based on the limits in Table I. Side velocities resulting from the capability of translating at 35 knots in any direction are indicated on the $V \sim n$ diagram.

For aircraft requiring a particular descent capability, additional envelopes of $V \sim \gamma$ or $V \sim \dot{h}$ should be presented. Such envelopes may in any event be requested by the procuring activity. The procuring activity should also ensure that the Operational Flight Envelopes encompass the flight conditions at which all appropriate performance guarantees will be demonstrated.

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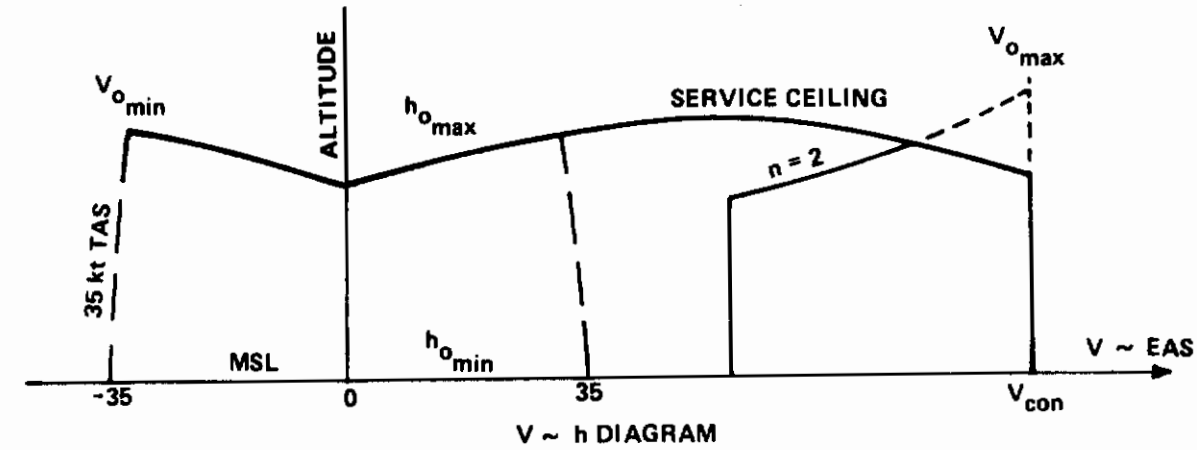
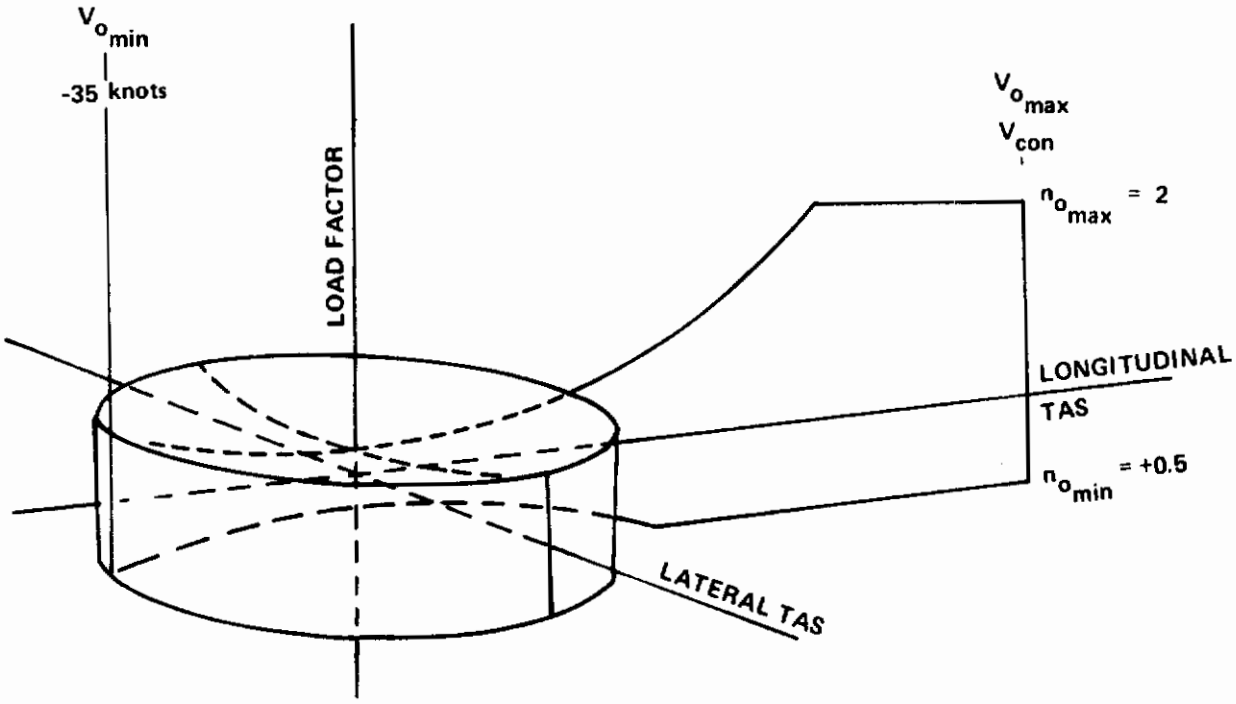


V ~ h DIAGRAM



V ~ n DIAGRAM (CONSTANT ALTITUDE)

Figure 6 TYPICAL OPERATIONAL FLIGHT ENVELOPE FOR FLIGHT PHASE CATEGORY B, NON-TERMINAL TRANSITION (NT), BASED ON LIMITS OF TABLE 1 (FROM REFERENCE 1)

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3.1.8 SERVICE FLIGHT ENVELOPES

REQUIREMENT

3.1.8 Service Flight Envelopes. For each Aircraft Normal State (but with thrust varying as required), the contractor shall establish, subject to the approval of the procuring activity, Service Flight Envelopes showing combinations of speed, altitude, and load factor derived from aircraft limits as distinguished from mission requirements. Additional envelopes in terms of parameters such as rate of descent, flight-path angle, and side velocity may also be specified. A certain set or range of Aircraft Normal States generally will be employed in the conduct of a Flight Phase. The Service Flight Envelopes for these States, taken together, shall at least cover the Operational Flight Envelope for the pertinent Flight Phase. The speed, altitude, and load factor boundaries of the Service Flight Envelopes shall be based on considerations discussed in 3.1.8.1, 3.1.8.2, 3.1.8.3, 3.1.8.4, and 3.1.8.5.

3.1.8.1 Maximum service speed. The maximum service speed, V_{max}, for each altitude below the service ceiling for the configuration under consideration is the lowest of:

a. The maximum permissible speed

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

c. The speed limited by an extreme nose-down pitch attitude

d. The maximum airspeed, in descents, from which recovery can be made without penetrating a safe margin from loss of control, intolerable buffet, or other dangerous behavior, and without exceeding structural limits.

3.1.8.2 Minimum service speed. The minimum service speed, V_{min}, for each altitude below the service ceiling for the configuration under consideration, in fore and aft flight, is the highest algebraically of:

a. 35 knots rearward (-35 knots)

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

c. A speed limited by reduced forward field of view or extreme nose-up pitch attitude

d. Other speed at MAT which is a safe margin above the value where pitch, roll, or yaw control available is insufficient to maintain 1-g level flight.

3.1.8.3 Service side velocity. The service side-velocity boundary for the configuration under consideration is defined by the maximum side velocity associated with each speed between V_{max} and V_{min} (as defined by 3.1.8.1 and 3.1.8.2) from which recovery to straight and level flight can be made without penetrating a safe margin from loss of control or other dangerous behavior.

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3.1.8.4 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 MAT.

3.1.8.5 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 at which the pitch control is in full aircraft-nose-up [nose-down] position with the thrust magnitude control in a position to maximize [minimize] the load factor

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

DISCUSSION

For each Aircraft Normal State (with thrust varying) there is an associated Service Flight Envelope which defines the boundaries of speed, altitude, load factor, and any additional limits as required, within which the aircraft is capable of safely operating. The limits of safe operation are defined by the maximum and minimum service speed, the maximum and minimum service load factors, the maximum service altitude and the service side velocity, Sections 3.1.8.1 through 3.1.8.5 of Reference 1.

A range of Aircraft Normal States will usually be required to perform a specific Flight Phase. The corresponding Service Flight Envelopes taken together shall at least cover the volume of the Operational Flight Envelope corresponding to the Flight Phase. Figure 7 of Reference 1 provides a sketch to illustrate this concept for $V \sim h$ and $V \sim n$ diagrams.

The volume formed by the Service Flight Envelopes encompasses the Operational Flight Envelope and denotes the extent of flight conditions that can be encountered without fear of exceeding aircraft limitations (safe margins should be determined analytically and experimentally). Requirements for the conditions outside the Operational Flight Envelope are less severe, but still stringent enough that the pilot can accomplish the associated mission Flight Phase. Mission effectiveness or pilot workload, or both, however, may suffer somewhat even with no failures.

This Envelope is intended to ensure that any deterioration of handling qualities will be gradual as flight proceeds out from 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.

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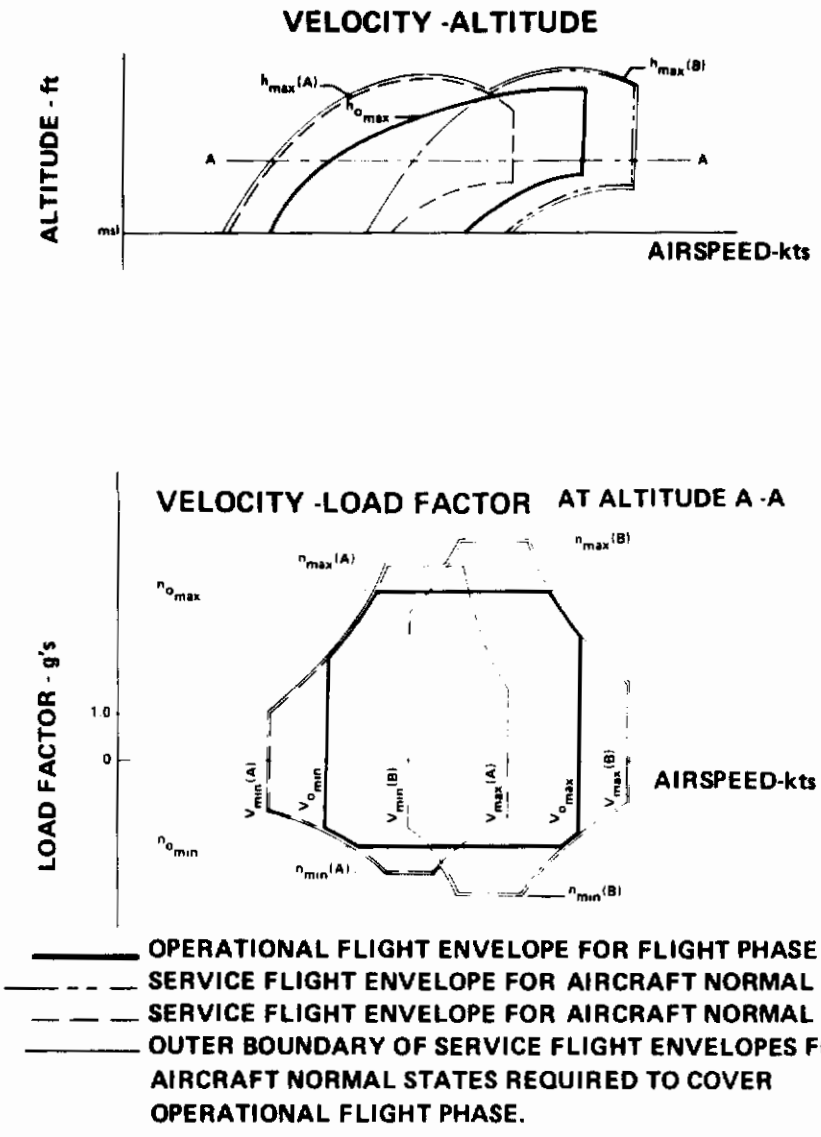

Figure 7 TYPICAL RELATIONSHIP BETWEEN OPERATIONAL ENVELOPE AND SERVICE FLIGHT ENVELOPE FOR A GIVEN FLIGHT PHASE REQUIRING TWO NORMAL STATES (FROM REFERENCE 1)

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3.1.9 PERMISSIBLE FLIGHT ENVELOPES

REQUIREMENT

3.1.9 Permissible Flight Envelopes. The Permissible Flight Envelopes encompass all regions in which operation of the aircraft is both allowable and possible. These are the boundaries of flight conditions outside the Service Flight Envelope which the aircraft is capable of safely encountering. Transient load factors, power settings, and emergency thrust settings may be representative of such conditions. The Permissible Flight Envelopes define the boundaries of these areas in terms of velocity, altitude, and load factor. Additional envelopes, in terms of parameters such as rate of descent, flight-path angle, and side velocity may also be specified.

3.1.9.1 Maximum permissible speed. The maximum permissible speed for each permissible altitude for the configuration under consideration shall be the lowest of:

a. The limit speed based on structural considerations

b. The limit speed based on engine considerations

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

d. The maximum airspeed, in descents, from which recovery can be made without encountering loss of control, intolerable buffet or structural vibration, and without exceeding structural limits.

3.1.9.2 Minimum permissible speed. The minimum permissible speed for each permissible altitude for the configuration under consideration, in fore and aft flight, shall be the highest algebraically of:

a. 35 knots rearward (-35 knots)

b. The speed, at MAT, below which pitch, roll, or yaw control available is insufficient to maintain 1-g level flight

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

DISCUSSION

It would be unreasonable to demand Level 2 flying qualities right up to stall, dive, and similar limits. Therefore the Service Flight Envelopes, where Level 2 is demanded for normal operation, may be cut short of these limits.

At each Aircraft Normal State (but with allowable thrust range) the maximum and minimum permissible speed can and must be defined for the pilot's information. Even in the case where a single variable (such as wing tilt angle, nozzle angle, duct angle, etc.) defines the transition configuration,

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the relation between Normal State and speed can be complex. It is therefore desirable that some way of indicating approach to the permissible boundary be provided.

It is important to realize that the concept of Flight Envelopes as used in the specification applies to fixed configurations [$\dot{\lambda} = 0$, see Figure 1(3.1.9)]. Unless this extra dimension is remembered, it apparently becomes possible to have flight outside the Permissible Flight Envelope [Figure 1(3.1.9)]. There is no basic reason why it should not be possible to encounter a combination of λ and V, while in rapid acceleration or deceleration, that is outside the Permissible Flight Envelope for fixed configuration. Presumably, if the Envelope was as indicated on Figure 1(3.1.9), stopping configuration change at A [defined by λ(A), V(A)] would result in equilibrium being achieved at B [defined by λ(B), V(B), where λ(A) = λ(B)]. Point B should be within the Permissible Flight Envelope. If B is not within the Permissible Flight Envelope then, by definition, the transition trajectory on which A lies is not permissible. The relation of Flight Envelopes to transition trajectories is discussed in more detail in the dicussion of requirements in Section 3.4.

Conversion Speed, V_{con}

The possible choices for defining the conversion speed, V_{con}, are discussed at length earlier in this document (Section 1.1, Scope). The current version of the requirements (Reference 1) leaves the final choice to agreement between the contractor and the procuring activity.

If a definition of V_{con} is chosen that is related to flying qualities, such as by the speed corresponding to a value of $d\dot{\gamma}/dV$, then V_{con} would in general be different at each configuration. This has been indicated in the hypothetical case sketched in Figure 1(3.1.9). At first sight this may be a little startling, especially if V_{con} for the fully converted configuration ($\dot{\lambda} = 0$) is less than the maximum speed at which conversion can be commenced (V_{UN}) (UN for unlocked). There is, of course, no real conflict since the concept of V_{con}, and all the requirements except those related to the transition maneuver itself (3.4) when $\dot{\lambda} \neq 0$, only apply to fixed configuration conditions. The transition requirements ($\dot{\lambda} \neq 0$) apply for any transition for all speeds up to V_{UN}.

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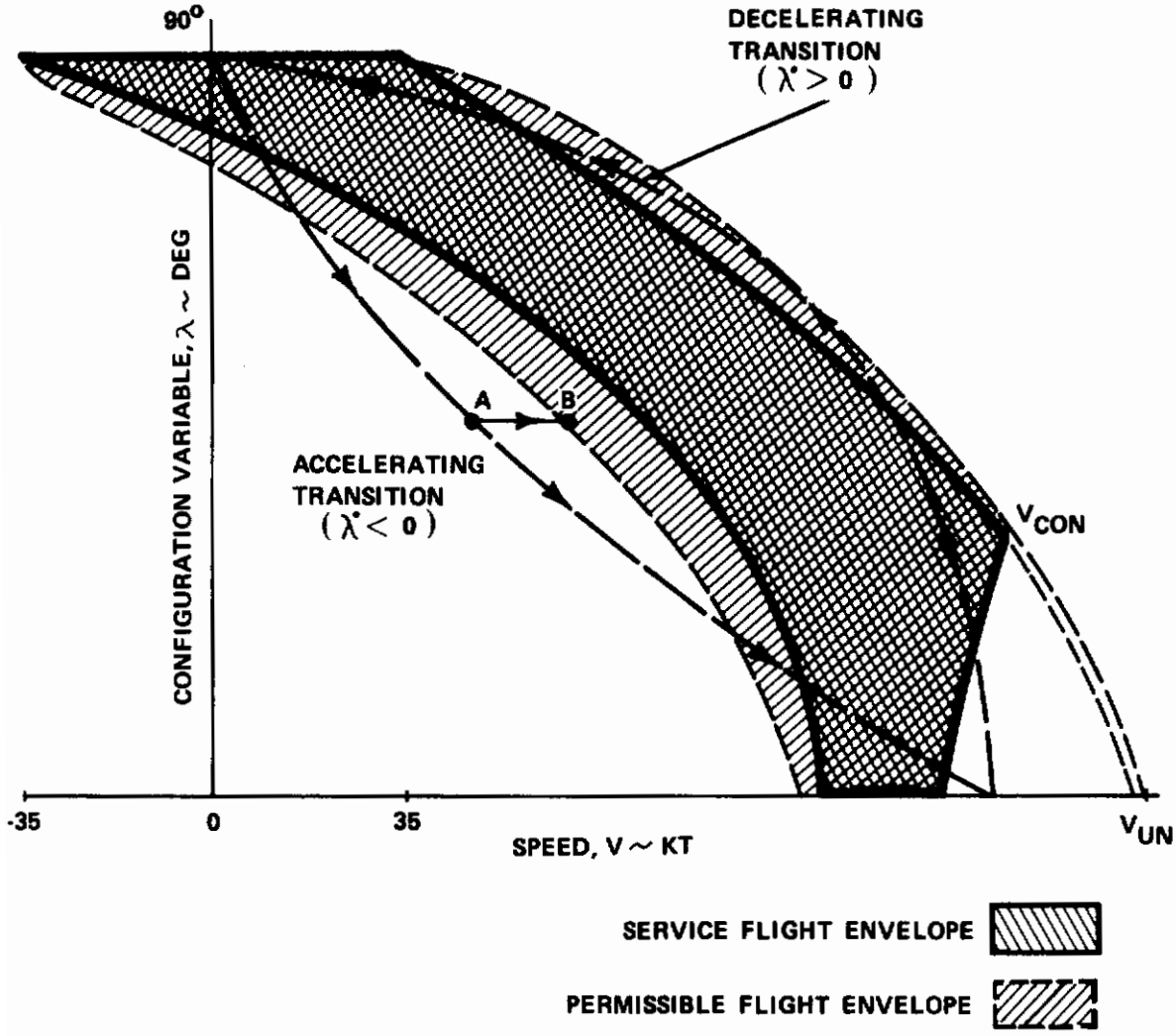

Figure 1 (3.1.9) PERMISSIBLE FLIGHT ENVELOPE

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3.1.10 APPLICATION OF LEVELS

REQUIREMENT

3.1.10 Applications of Levels. Levels of flying qualities as indicated in 1.5 are employed in this specification in realization of the possibility that the aircraft may be required to operate under abnormal conditions. Such abnormalities that may occur as a result of either flight outside the Operational Flight Envelope, the failure of aircraft components, or both, are permitted to comply with a degraded Level of flying qualities as specified in 3.1.10.1 through 3.1.10.3.3.

3.1.10.1 Requirements for Aircraft Normal States. The minimum required flying qualities for Aircraft Normal States (3.1.6.1) are as shown in table II.

TABLE II. Levels for Aircraft Normal States

Within Operational Flight Envelope	Within Service Flight Envelope
Level 1	Level 2

3.1.10.2 Requirements for Aircraft Failure States. When Aircraft Failure States exist (3.1.6.2), a degradation in flying qualities is permitted only if the probability of encountering a lower Level than specified in 3.1.10.1 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 and the effect of that Failure State on the flying qualities within the Operational and Service Flight Envelopes. These analyses shall be updated at intervals specified by the procuring activity. 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. These probabilities shall be less than the values shown in table III.

TABLE III. Levels for Aircraft Failure States

Probability of Encountering	Within Operational Flight Envelope	Within Service Flight Envelope
Level 2 after failure	$< 10^{-2}$ per flight	
Level 3 after failure	$< 10^{-4}$ per flight	$< 10^{-2}$ per flight

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In no case shall a Failure State (except an approved Special Failure State) degrade any flying quality outside the Level 3 limit.

3.1.10.2.1 Specific failures. The requirements on the effects of specific types of failures, e.g., propulsion (3.8.9) or flight control system (3.5.5), shall be met on the basis that the specific type of failure has occurred, regardless of its probability of occurrence.

3.1.10.3 Exceptions

3.1.10.3.1 Ground operation. Some requirements pertaining to takeoff, landing, and taxiing involve operation outside the Operational, Service and Permissible Flight Envelopes. When requirements are stated for these conditions, the Levels shall be applied as if the conditions were in the Operational Flight Envelope.

3.1.10.3.2 When Levels are not specified. Within the Operational and Service Flight Envelopes, all requirements that are not identified with specific Levels shall be met under all conditions of component and system failure except approved Aircraft Special Failure States (3.1.6.2.1).

3.1.10.3.3 Flight outside the Service Flight Envelope. From all points in the Permissible Flight Envelope, it shall be possible readily and safely to return to the Service Flight Envelope without exceptional pilot skill or technique, regardless of component or system failures. The requirements of 3.8.1 and 3.8.2 shall also apply.

3.1.10.3.4 Operation in critical height-velocity conditions. Some propulsion system failures in the critical height-velocity regime may be catastrophic. Although the aircraft would not meet the requirements of 3.1.10.3.3 for flight within the Permissible Flight Envelope following such propulsion system failures, there may be cases where the critical height-velocity conditions will fall within the Operational Flight Envelope for all other Aircraft States because flight under these conditions is essential to accomplishment of the operational missions of 3.1.1. Special requirements may be specified by the procuring activity for conditions encountered during and after the propulsion system failure. The size of the critical height-velocity regime is subject to the approval of the procuring activity.

DISCUSSION

The concept of Levels has been applied exactly as developed for MIL-F-8785B (Reference 10).

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 Level concept is directed at the achievement of adequate flying qualities without imposing undue requirements that could lead to unwarranted system complexity or decreased flight safety. Without actually requiring a good basic airframe,

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the general specification provides:

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

In short, a systems approach to the requirement specification is used. The following paragraphs discuss this concept in some detail.

The Level approach is straightforward in concept. The requirements specified for normal operation (no system failures) provide desirable flying qualities. Equipment failures, however, either in the flight control system or other subsystems, can cause a degradation in flying qualities. The emphasis in the specification is on the effects of failures, rather than the failures themselves. 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 flight must not exceed 10^{-2}, and the probability of encountering Level 3 on any portion of the flight must not exceed 10^{-4}. Somewhat reduced requirements are 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 V/STOL Specification do not apply. There is a qualitative requirement in 3.1.10.3.3 which refers to the requirements that do still apply.

Numerical Probabilities

The numerical values can, of course, be changed by the procuring agency to 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 design goals. However, the values given are reasonable, based on experience with contemporary (non-V/STOL) aircraft. To illustrate this, the following table presents actual control system failure information for several piloted aircraft:

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Source	System	Mean Time Between Malfunctions (MTBM)
Ref 23	F-101B	86 hours
Ref 23	F-104	300 hours
Ref 23	F-105D (Flight Control plus Electronics)	14 hours
Ref 23	E-1B	185 hours
Ref 24	B-58	20 hours

Unfortunately the flying qualities effects of the reported failures are not given along with the above data. Reference 25 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	.0093
F-104	.0027
F-105B	.057
E-1B	.0043
B-58	.040

which indicates that all systems, with the exception 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 26) and Anglo-French (Reference 27) supersonic transport design.

A similar comparison can be made between accident loss rates and the requirement for P (Level 3) $< 10^{-4}$. It should be emphasized that Level 3 as defined in paragraph 1.5 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 at least "safety related". Reference 28 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.

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Aircraft	1967 Loss Rate (losses/100,000 hours)	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 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 specified value is reasonable.

As a final note, Reference 29 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.

Implementation

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-STD-882 (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 airplane configurations, and even the mission Flight Phases, to be considered. All failures must be examined which could have occurred previously, as well as all failures which might occur during the Flight Phase being analyzed. For example, failure of the wings to tilt up during descent would require consideration of a wings-down 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 wing to tilt down after takeoff might mean a landing with the wing at the takeoff setting, with certain unexpended external stores; but cruise would be impossible. If the mission might be either continued or aborted, both contingencies need to be examined.

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

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.

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 Envelope, Level 3 in the Operational Envelope, and Level 3 in the Service Envelope, and

c) recommend 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.

Re-Evaluation: As the system design progresses, the initial evaluation is revised at intervals. This process continues throughout the design phase. The results of the analyses of vehicle flying qualities/flight safety may be used directly to:

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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.

Proof of compliance is, for the most part, analytical in nature as far as probabilities of failure are concerned. However, some equipment failure rate data 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.

Specific Failures: There are some specific requirements pertaining to failure of the engines and the flight control system (e.g., 3.8.10). For these requirements the specific 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.

Special Failures: Note that certain Special Failure States (3.1.6.2.1) 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.

Operation in Critical Height-Velocity Conditions: It is realized that certain types of V/STOL will occasionally be required to operate within their critical height-velocity regions. The apparent conflict is recognized by Section 3.1.10.3.4 (Reference 1) which requires the procuring activity to give special consideration to the critical height-velocity boundaries. The time to set the allowable size of this region is early in the design stage, while configuration changes are still feasible.

In summary, the Level concept was evolved in recognition of the obvious fact that flying qualities, flight safety, and system reliability are all very much related in the development of current piloted aircraft. This interrelationship is being exploited to improve aircraft in terms of overall

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effectiveness. The net result can be system improvement with a minimum expenditure of effort. Examples, using a similar approach, are presented in References 30, 31, 32, and 33.

Additional insight into the application of Levels is given in paragraph 6.5 of Reference 1 as follows:

"6.5 Application of Levels. Part of the intent of 3.1.10 is to ensure that the probability of encountering significantly degraded flying qualities because of component or subsystem failures is small. For example, the probability of encountering very degraded flying qualities (Level 3) must be less than specified values per flight.

6.5.1 Theoretical compliance. To determine theoretical compliance with the requirements of 3.1.10.2, the following steps must be performed:

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

b. Define the longest flight duration to be encountered during operational missions (3.1.1).

c. Determine the probability of encountering various Aircraft Failure States, per flight, based on the above flight duration (3.1.10.2).

d. Determine the degree of flying qualities degradation associated with each Aircraft Failure State in terms of Levels as defined in the specific requirements.

e. Determine the most critical Aircraft Failure States (assuming the failures are present at whichever point in the Flight Envelope being considered is most critical in a flying qualities sense), and compute the total probability of encountering Level 2 flying qualities in the Operational Flight Envelope due to equipment failures. Likewise, compute the probability of encountering Level 3 flying qualities in the Operational Flight Envelope, etc.

f. Compare the computed values above with the requirements in 3.1.10.2 and 3.1.10.3.

If the requirements are not met, the designer must consider alternate courses such as:

(a) Improve the aircraft flying qualities associated with the more probable Failure States, or

(b) Reduce the probability of encountering the more probable Failure States through equipment redesign, redundancy, etc.

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

6.5.2 Level definitions. To determine the degradation in flying qualities parameters for a given Aircraft Failure State the following definitions are provided:

a. Level 1 is better than or equal to the Level 1 boundary, or number, given in section 3.

b. Level 2 is worse than Level 1, but no worse than the Level 2 boundary, or number.

c. Level 3 is worse than Level 2, but no worse than the Level 3 boundary, or number.

When a given boundary, or number, is identified as Level 1 and Level 2, this means that flying qualities outside the boundary conditions shown, or worse than the number given, are at best Level 3 flying qualities. Also, since Level 1 and Level 2 requirements are the same, flying qualities must be within this common boundary, or number, in both the Operational and Service Flight Envelopes for Aircraft Normal States (3.1.10.1). Aircraft Failure States that do not degrade flying qualities beyond this common boundary are not considered in meeting the requirements of 3.1.10.2. Aircraft Failure States that represent degradations to Level 3 must, however, be included in the computation of the probability of encountering Level 3 degradations in both the Operational and Service Flight Envelopes. Again, degradation beyond the Level 3 boundary is not permitted regardless of component failures.

6.5.3 Computational assumptions. Assumptions a and b of 3.1.10.2 are somewhat conservative, but they simplify the required computations in 3.1.10.2 and provide a set of workable ground rules for theoretical predictions. The reasons for these assumptions are:

a. "...components and systems are....operating for a time period per flight equal to the longest operational mission time....." Since most component failure data are in terms of failures per flight hour, even though continuous operation may not be typical (e.g., yaw damper on during supersonic flight only), failure probabilities must be predicted on a per flight basis using a "typical" total flight time. The "longest operational mission time" as "typical" is a natural result. If acceptance cycles-to-failure reliability data are available (MIL-STD-756), these data may be used for prediction purposes based on maximum cycles per operational mission, subject to procuring

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activity approval. Also, finite wearout life components, such as engines at maximum takeoff thrust, may be considered as exceptions and failure calculations shall be based on maximum normal operating time per flight in these cases, again subject to procuring activity approval. In any event, compliance with the requirements of 3.1.10.2, as determined in accordance with section 4, is based on the probability of encounter per flight.

b. "...failure is assumed to be present at whichever point...is most critical...." This assumption is in keeping with the requirements of 3.1.6.2 regarding Flight Phases subsequent to the actual failure in question. In cases that are unrealistic from the operational standpoint, the specific Aircraft Failure States might fall in the Aircraft Special Failure State classification (3.1.6.2.1). "

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3.1.11 COCKPIT CONTROLS

REQUIREMENT

3.1.11 Cockpit controls. Requirements are written on the basis of conventional cockpit controls, e.g., stick or wheel plus rudder pedals, with either a conventional throttle or a helicopter-type collective control. The form of thrust angle control has not been assumed. Aircraft having cockpit controls other than conventional (e.g., side arm control) are excluded from the requirements which reflect the type of control (e.g., response to a 1-inch stick deflection), but not others (e.g., roll performance requirements). The procuring activity will impose alternate requirements for nonconventional cockpit controls.

DISCUSSION

Some unusual controllers, and methods of control, for VTOL's have been suggested in the literature. For example, side-arm controllers, side-force controllers, lift engine thrust control through the pitch control stick, etc. All such innovations cannot possibly be foreseen and covered in a general flying qualities specification and therefore only "conventional" controls, as described in 3.1.11, are considered. The requirements concerned with vehicle performance, for example, the roll requirements of 3.3.10, are believed to be applicable regardless of cockpit control arrangements.

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3.2 HOVER AND LOW SPEED

REQUIREMENT

3.2 Hover and low speed. The hover and low-speed requirements apply to those Flight Phases of the operational missions of the aircraft which include hovering at zero ground speed in steady winds from any direction up to the limits of the Service Flight Envelope, and maneuvering in any direction at speeds up to the limits of the Service Flight Envelope, except that the requirements specified under 3.3 apply for those conditions where the forward speed component is greater than 35 knots TAS. The requirements of 3.5, 3.6, 3.7, and 3.8 are also applicable in this speed regime.

DISCUSSION

This paragraph is a general introduction which delineates the flight conditions at which the subsequent requirements must be satisfied. The reason for satisfying requirements within the Service Flight Envelope rather than within the Operational Flight Envelope is to prevent the possibility of a rapid deterioration in flying qualities for those situations where flight conditions between the Operational and Service Flight Envelopes can be easily encountered. Flight Envelopes are discussed in Sections 3.1.7, 3.1.8, and 3.1.9.

The rationale for dividing the specification into the various sections is discussed at some length in Section III of this report, Specification Structure and Philosophy.

It should be noted that the requirements specified for the characteristics of the flight control system (3.5), takeoff and landing (3.6), atmospheric disturbances (3.7) and miscellaneous requirements are also applicable in this speed range.

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3.2.1 EQUILIBRIUM CHARACTERISTICS

REQUIREMENT

3.2.1 Equilibrium characteristics. Without attaining excessive attitudes, it shall be possible to hover over a spot in steady winds of up to 35 knots from any direction relative to the aircraft heading, except as limited by the boundaries of the Service Flight Envelope.

DISCUSSION

This requirement is self-explanatory. It is an established fact from both flight and simulator experiments that pilots do not like to hover with excessive pitch and roll attitudes. Not only does their visibility in one or more directions deteriorate, but the probability increases of having portions of the vehicle strike the ground in gusty conditions.

The word "excessive" is obviously not very precise. Even when discussing this attitude problem with experienced test pilots, it is difficult to determine with any precision what attitude limits should be, since opinions differ somewhat. There are apparently several factors such as the task, the experience with a particular airplane, visibility characteristics, etc., that combine to influence subjective opinion.

Although pilot opinion must be relied upon in demonstrating for compliance with 3.2.1, an attempt has been made to quantify attitude limits under particular conditions in the following two requirement paragraphs 3.2.1.1 and 3.2.1.2.

Consideration was given, by the Air Force, to increasing the wind speed in which hovering capability is required. However, it was found that the probability of encountering winds greater than 30-40 knots did not justify a change. Certainly winds higher than 35 knots can be encountered, but it was assumed that the margin of the Service Flight Envelope over the Operational Flight Envelope will provide Level 2 hovering capability at speeds greater than 35 knots. Further margins may have to be demanded in special cases.

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3.2.1.1 CHANGING TRIM

REQUIREMENT

3.2.1.1 Changing trim. The local slope of the equilibrium attitude-speed relationship shall not exceed 0.6 degrees per knot for speed perturbations of at least 10 knots in either direction about the trim speed. Thirty-five knots or the limits of the Service Flight Envelpe, ± 10 degrees roll attitude, or an attitude change of ± 10 degrees in pitch need not be exceeded. The configuration and trim may be different at each trim condition but they must remain fixed while determining the attitude-speed variations about the trim condition. The fuselage reference bank attitudes must not exceed ± 10 degrees at any trim speed. These requirements shall be satisfied at all forward trim speeds, backward trim speeds, and sideward trim speeds both to the left and to the right, up to the limits of the Service Flight Envelope or 35 knots, whichever is less in magnitude.

DISCUSSION

The basic intent of this requirement is to put a limit on the derivatives X_u and Y_v for aircraft that change pitch angle to translate and stems from a data base that reflects this means of translational control. Whether or not the requirement can be extended in whole or in part to aircraft which use independent X-force and Y-force controls to develop translational accelerations is a question that cannot be answered in detail at this time.

The requirement has been written in terms of steady-state variation of attitude with speed because this variation is determined by X_u and Y_v and is measurable. Experiments show that these force derivatives influence low-speed flying qualities in two distinct ways.

First, since these derivatives determine attitude variations with speed, relatively large attitude changes can occur during translational maneuvering when they have high values. In the Reference 34 moving-base simulation, pilots complained about these attitude changes. The maneuvers performed in this simulation included

(a) Tracing out a square while holding heading constant. This required forward, sideward and rearward flight.

(b) Performing a crosswind approach (10 knot wind) followed by a 90 degree turn into the wind over proposed touchdown spot and then hover.

For the vast majority of configurations having large magnitudes of X_u and Y_v (-0.2), the pilot comments indicated as objectionable the fact that the attitude required to compensate for the wind was too steep and that attitude changes required to initiate motion or maneuver were too great. In the Reference 34 experiment, there was no capability for rotating the thrust vector relative to the aircraft. If this capability were present, the pilots

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could have used thrust rotation as an auxiliary means for controlling trim attitudes. However, this would require additional control coordination efforts to perform the maneuvers with any degree of precision.

A second way that X_u and Y_v affect hover and low-speed flying qualities is through their influence on the aircraft's response to gusts. Since these derivatives determine the magnitude of X- and Y-forces applied by gusts to the airframe, precision hover becomes difficult when these derivatives take on high values because ground position perturbations become large. In particular, for a vehicle that does not have a direct force translational control system, a pilot's attitude control task becomes complicated because he has to control position through attitude while stabilizing attitude disturbances. In the Reference 34 moving-base simulation the rms gust level was 3.4 ft/sec and there were not too many complaints about gust effects on position control. However, in the fixed-base simulations of References 35 and 36, the rms gust level was 5.1 ft/sec, and values of X_u and Y_v of -0.2 and greater, i.e., more negative, were accompanied by definite complaints about the ability to maintain ground position.

Since the data discussed above indicated that X_u and Y_v magnitudes greater than -0.2 would be troublesome, paragraph 3.2.1.1 essentially imposes a limit on them in terms of an attitude-speed gradient. An attitude-speed gradient of 0.6 degrees per knot is equivalent to X_u (Y_v) of -0.2 based on calculations using the simple equilibrium X-force equation

$$0 = X_u u - g\theta$$

As a closing remark it is noted that the title "Changing trim" of paragraph 3.2.1.1 was chosen to emphasize that in demonstrating compliance with this requirement, the aircraft configuration at each selected trim speed can be different. For example, the pilot could adjust thrust vector angle so that the trim attitude is zero at every trim speed about which the local slope of the equilibrium attitude-speed relationship is to be determined. This is in contradistinction with the following requirement (3.2.1.2) which applies to only one configuration.

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3.2.1.2 FIXED TRIM

REQUIREMENT

3.2.1.2 Fixed trim. For aircraft required to perform rapid hovering turns in winds up to 35 knots or as otherwise specified by the procuring activity, the local slope of the equilibrium attitude-speed relationship shall not exceed 0.6 degrees per knot. The total fuselage reference pitch attitude change shall not exceed 20 degrees, and the bank attitude shall not exceed ± 10 degrees. These requirements apply at all steady forward speeds, backward speeds, sideward speeds both to the left and to the right, up to the limits of the Service Flight Envelope or 35 knots, whichever is less in magnitude, with configuration and trim fixed.

DISCUSSION

The basic intent of this requirement is similar to that of paragraph 3.2.1.1, namely, to place limits on the values of X_u and Y_v. However, there are important differences between these two requirements.

Paragraph 3.2.1.2 is intended to apply to those aircraft whose low speed operational maneuvering tactics include turning rapidly while hovering in winds. Since the performance of such tactics may be dictated to a large extent by external factors which a pilot cannot foresee, his full time attention will probably be devoted to the changing conditions around him. However, if the flight task workload is too high, he may not be able to assess these changing conditions to his advantage. It seemed desirable, therefore, to include a requirement aimed at minimizing the need for complex cockpit control trimming techniques that might otherwise interfere with mission effectiveness. Thus paragraph 3.2.1.2 recognizes the advantages of reducing workload for certain kinds of operational situations and contributes in this direction by stating that the configuration and trim must remain fixed while demonstrating compliance. Paragraph 3.2.1.1, on the other hand, is directed to those types of operational activities where rapid low speed maneuvering is not as vital. In these latter cases, a pilot may have more leeway in heading as well as more time to use available thrust vectoring capability to compensate for the pitch and roll attitude changes that would be caused by steady winds.

Note that for pitch, paragraph 3.2.1.2 places limits on the change in pitch attitute, i.e., the difference between the largest pitch attitude and the smallest pitch attitude, which occurs within the speed range stated in the requirement paragraph, cannot exceed 20°. For roll, however, this paragraph places limits on the absolute bank angle.

These limits are based on discussions with Air Force and Army pilots having experience in evaluating both helicopters and VTOL aircraft such as the P.1127 and X-22A. Some documented information relevant to

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the problem of attitude excursions appears in Reference 37 which reports flight test results for the X-22A.

High pitch and roll attitudes encountered during constant duct angle translations are cited as contributing to the difficulty of holding the required trim attitudes. Specifically, bank angles of about 10 degrees are described as "excessively large." In addition, as experience was gained in the aircraft, the pilots came to prefer using duct rotation for longitudinal translations, probably due to the more moderate pitch excursions required in this mode of control.

Rearward translations at constant duct angle were considered difficult because of the high nose-up fuselage attitudes required. Contributing to this difficulty were the lack of a rearward field of view and the loss of attitude cues looking forward.

With respect to diagonal translations at fixed duct angle, pilot comment indicates that the large pitch and roll attitudes required for 10 to 15 knot speed perturbations made this a difficult task. This difficulty may be attributable to cockpit visibility, since the pilot's field of view was partially obscured by the top of the cockpit and the instrumentation control switch panel.

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3.2.1.3 COCKPIT CONTROL GRADIENTS

REQUIREMENT

3.2.1.3 Cockpit control gradients. The following requirements shall be satisfied at all forward trim speeds, backward trim speeds, and sideward trim speeds both to the left and to the right, up to the limits of the Service Flight Envelope or 35 knots, whichever is less in magnitude. This requirement shall apply for speed perturbations of at least 10 knots in both directions about the trim speed except that the aircraft need not exceed 35 knots or the limits of the Service Flight Envelope. The configuration and trim may be different at each trim condition, but they must remain fixed while determining the control gradients.

Level 1: The variations of cockpit control force and control position with airspeed shall be smooth and the local gradients stable or zero for both the pitch and roll cockpit controls. The gradients shall be essentially linear with no objectionable changes in the slope of force or position with speed.

Level 2: For those Flight Phases of the operational missions of 3.1.1 for which IFR operation is required, the Level 2 requirement is the same as for Level 1. In all other cases, the Level 2 requirement is the same as Level 3.

Level 3: The Level 1 requirements shall apply for the local control force gradients. The local pitch and roll control position gradients may be unstable, provided the change in cockpit control position does not exceed one-half inch in the unstable direction over the speed range specified.

Stable pitch control gradients mean that incremental pull forces and aft displacement of the cockpit control are required to maintain slower or more rearward airspeeds and the opposite to maintain faster or more forward airspeeds.

Stable roll control gradients mean that incremental right force and right displacement of the cockpit control are required to maintain right translations or right sideslips and the opposite to maintain left translations or left sideslips.

The term gradient does not include that portion of the control force or control position versus airspeed curve within the preloaded breakout force or friction band.

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DISCUSSION

In an informal sense, this requirement is intended to prevent "unnatural" control feel characteristics when performing speed changing maneuvers at low speed flight conditions. An acceptable definition of "natural" or "unnatural" control feel would probably be difficult to obtain. However, the major problem is not definitions but a lack of experimental data based on experiments specifically designed to study how various ranges of force and position gradients with airspeed influence pilot opinion. Such experiments would probably be quite complicated to carry out because of the many factors involved. For one, if a ground simulator were used, it would need a complex control system. A simple spring loaded stick would limit an investigation because combinations of unstable force gradient and stable position gradient (and vice versa) could not be evaluated. Also, even if complex simulator control system hardware were available it would be very important to consider the task assigned to the pilot along with other detailed instructions. For example, the results could depend on whether or not he was allowed to use the trim system. All this is particularly important for requirements because it may turn out that a system has undesirable characteristics if used in one way but be perfectly acceptable if used in another way.

There are various opinions regarding the relative importance, as handling qualities parameters, of force gradients with speed and position gradients with speed. In many instances such opinions seem, no doubt, to be based on experiences with vehicles whose control systems and dynamics are a bit more complex than the representations used in many flying qualities simulation and analyses programs. Thus a knowledge of many details is needed before the experience being voiced by these opinions can be molded into a quantitative requirement that is a significant improvement over paragraph 3.2.1.3. Unfortunately, the needed details are hard to obtain.

As an illustration of the need to consider the details when drawing conclusions regarding the influence of control force and position gradients with speed, we present the following relatively straightforward example.

Assume that the vehicle is described by the following equations

$$\dot{u} = X_u u - g\theta$$

$$\dot{q} = M_u u + M_q q + M_{\delta_{ES}} \delta_{ES}$$

The u/δ_{ES} transfer function is given by

$$\frac{u(s)}{\delta_{ES}} = \frac{M_{\delta_{ES}}}{\Delta(s)}$$

where $\Delta(s)$ is the characteristic cubic. The form of $\Delta(s)$ depends on whether the roots are real or complex. Since $\Delta(s)$ is a cubic there are two possibilities:

$$\Delta(s) = \left(s + \frac{1}{T_{SP_1}}\right)\left(s^2 + 2\zeta\omega_n s + \omega_n^2\right)$$

$$\Delta(s) = \left(s + \frac{1}{T_{SP_1}}\right)\left(s + \frac{1}{T_{SP_2}}\right)\left(s + \frac{1}{T_{SP_3}}\right)$$

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Thus the steady-state values of the u/δ_{ES} transfer function are

$$\left.\frac{u}{\delta_{ES}}\right)_{SS} = \frac{M_{\delta_{ES}}}{\frac{\omega_n^2}{T_{SP}}}$$

or

$$\left.\frac{u}{\delta_{ES}}\right)_{SS} = \frac{M_{\delta_{ES}}}{\left(\frac{1}{T_{SP_1}}\right)\left(\frac{1}{T_{SP_2}}\right)\left(\frac{1}{T_{SP_3}}\right)}$$

If the vehicle has one real root and if $M_{\delta_{ES}}$ is positive, we see that $\left.\frac{u}{\delta_{ES}}\right)_{SS}$ is an indicator of the sign of the real root. Thus the speed vs position gradient is a measure of divergent tendencies. Requiring that this gradient be positive (stable) guarantees the absence of a divergence in this case.

Now suppose that the vehicle has three real roots, only two of which are unstable (positive). The sign of $\left.\frac{u}{\delta_{ES}}\right)_{SS}$ in this case is the same as if all the roots were stable. Mathematically it would follow that the speed-position gradient is not necessarily a measure of divergent tendencies. However, drawing such a conclusion with regard to the significance of the speed-position gradient is based on a key assumption - namely that the vehicle has three real roots. Now this assumption follows from the fundamental theorem of algebra which states that a third degree polynomial has at most 3 roots and that complex roots occur in pairs. But the fundamental theorem of algebra is based on assumption also. The one we wish to point out is that this theorem assumes that the coefficients of a polynomial can take on any values independently of one another. A look at the characteristic equation

$$s^3 - (X_u + M_q)s^2 + X_u M_q s + M_u g = 0$$

shows that the coefficients s^2 and s are dependent on one another. This dependency along with realistically possible values of the derivatives may, in fact, make the existence of two real roots a highly unlikely situation. If this is true, then it is misleading to state that the speed-control position gradient is not necessarily a measure of divergent tendencies.

The point of the above discussion is that a simplified general analysis can result in conclusions that are not very fruitful with regard to improving our understanding of the nature of control force and position gradients with speed as flying qualities parameters.

Since there are so very little data and analyses available in the literature, the requirements of 3.2.1.3 have been based on review comments received on earlier versions of the proposed specification. The gist of these review comments is that some instability should be allowed since most VTOL's will have unstable control gradients somewhere in the 15-70 knot speed range (assuming simple control systems). There has been some experience gained with the use of series actuators (i.e., actuators which move the cockpit control one way while the actual control surface moves in the opposite sense) to eliminate or minimize the control gradient instabilities, but apparently such control systems also create some undesirable problems.

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Discussions with pilots indicated that for IFR flight, both force and position gradients are preferred to be stable. For example, pilots mentioned one helicopter in which they preferred not to fly at the speed for best rate of climb so as to avoid a speed instability. Hence, for IFR flight, 3.2.1.3 imposes both stable stick force and position stability for Level 1 and Level 2. The unstable control position gradient allowed for in the Level 3 requirement of 3.2.1.3 is considered moderate and is based on the equivalent requirement of MIL-H-8501A (Reference 15). Since some aircraft may be used only for VFR, the Level 2 VFR requirement is relaxed to allow small unstable position gradients similar to Level 3.

A similar problem exists in the forward flight regime and the equivalent requirement 3.3.1 has been written to be similar to 3.2.1.3. The rationale is discussed further in that section.

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3.2.2 DYNAMIC RESPONSE REQUIREMENTS

3.2.2.1 PITCH [ROLL]

REQUIREMENT

3.2.2.1 Pitch [roll]. The following requirements shall apply to the dynamic responses of the aircraft with the cockpit controls free and with them fixed following an external disturbance or an abrupt pitch [roll] control input in either direction. The requirements apply for responses of any magnitude that might be experienced in operational use. If oscillations are nonlinear with amplitude, the oscillatory requirements shall apply to each cycle of the oscillation.

Level 1: All aperiodic responses (real roots of the longitudinal characteristic equation and the lateral-directional characteristic equation) shall be stable. Oscillatory modes of frequency greater than 0.5 radians per second shall be stable. Oscillatory modes with frequency less than or equal to 0.5 radians per second may be unstable provided the damping ratio is less unstable than -.10. Oscillatory modes of frequency greater than 1.1 radians per second shall have a damping ratio of at least 0.3.

Level 2: For those Flight Phases of the operational missions of 3.1.1 for which IFR operation is required, the Level 2 requirement is the same as for Level 1. In all other cases, for Level 2, divergent modes of aperiodic response shall not double amplitude in less than 12 seconds. Oscillatory modes may be unstable provided their frequency is less than or equal to 0.84 radians per second and their time to double amplitude is greater than 12 seconds. Oscillatory modes of frequency greater than 0.84 radians per second shall be stable.

Level 3: Divergent modes of aperiodic response shall not double amplitude in less than 5 seconds. Oscillatory modes may be unstable provided their frequency is less than or equal to 1.25 radians per second and their time to double amplitude is greater than 5 seconds. Oscillatory modes of frequency greater than 1.25 radians per second shall be stable.

DISCUSSION

A major problem faced during the course of developing this requirement has been that of achieving an acceptable compromise. Much work and effort have gone into acquiring and analyzing experimental data. Hence, there is a reasonable amount of confidence in the conclusions that have been drawn regarding what should be explicit in the requirement. At the same time, however, there has been a great deal of exchange of ideas during Government and industry review cycles regarding the practical or utilitarian

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aspects which should be taken into account. Most of these aspects are qualitative in nature and therefore present conflicts of various sorts with the philosophy that flying qualities requirements should strive to be quantitative and rest on an appropriate experimental and analytical foundation. Nevertheless, the motivation for concern for these aspects stems from solid past experience and cannot be ignored. Initial versions of a dynamic requirement reflected the attempt to formulate a quantitative requirement that was a distillation of the maximal amount of information that could be extracted from experimental and analytical findings. This final version of the requirement reflects a compromise with utilitarian considerations.

Data Base

The dynamic requirements are based to a large extent on data that were obtained from fixed-base and moving-base simulations (References 34, 35, and 36). The general objective of these experiments was to obtain the additional data needed to extend our knowledge of flying qualities for a broader set of vehicle dynamics than had heretofore been reported in the literature. More specifically, these experiments included the effects of X_u, M_{ug}, Y_v, and L_{vg}. Some early experiments did not treat these derivatives. Those few that did study them indicated the need for broader investigations.

Among some of the early relevant data which can be cited are References 38 and 39. The Reference 38 results showed that, with an optimum control sensitivity, the pilot could give satisfactory Cooper ratings even with zero damping in a single-degree-of-freedom situation. However, if the pilot were also given an additional degree of freedom, in this case yaw, satisfactory ratings could not be obtained with pitch damping $(-M_q)$ greater than about -0.5. The X-14 flight tests reported in Reference 39 give results that agreed more closely with the Reference 38 single-degree-of-freedom results than with the two-degree-of-freedom results. Figure 1(3.2.2.1) compares the significant longitudinal results of these two references.

An important thing to mention here is that the representation of the vehicle dynamics in Reference 38 was that of a simple first-order rate system defined completely by the value of a damping derivative. Likewise in Reference 39, the dynamics of the actual vehicle were assumed to be adequately described as being a first-order system.

The importance of the speed stability derivative, M_u, on hovering and low-speed flying qualities was best brought to attention by the Princeton flight tests (Reference 40). Speed stability can create unstable oscillatory modes and determines gust response characteristics. The Reference 40 study pointed out that, depending on the magnitude of M_u and the level of turbulence, the pitch damping required for satisfactory flying qualities was much higher than that indicated by previous experiments which did not account for these factors.

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In Reference 41, Miller and Clark of the United Aircraft Research Laboratories (UARL) presented results of further simulator studies on low-speed stability and control. Under the VIFCS program, CAL subcontracted with UARL to extend this work. References 35 and 36 contain the results of the UARL effort under the CAL subcontract.

The diversity of dynamic characteristics evaluated by UARL is indicated by the following range of parameters which were varied.

Longitudinal

M_{ug}	~ 0	to	2.0
X_u	~ 0	to	-0.3
M_q	~ -0.5	to	-10
M_θ	~ 0	to	-50

Lateral

L_{vg}	~ 0	to	-2.0
Y_v	~ 0	to	-0.3
L_p	~ -0.5	to	-10
L_ϕ	~ 0	to	-30

Control sensitivities were selected by the pilot for each configuration. This tended to minimize the influence of sensitivity on the evaluation of the dynamic characteristics.

The experiments reported in References 35 and 36 were performed on a fixed-base simulator. In order to obtain an expanded data base, CAL subcontracted with Norair to perform moving-base simulations (Reference 34). This study also covered a wide range of dynamics and more pilots were used. The pilots were allowed to select control sensitivities in this program also. In summary, the experiments reported in References 34, 35, and 36 provided a large amount of the data that were used to formulate the hover and low-speed dynamic requirements.

Basic Equations

The details involved in analyzing experimental data depend to a large extent on the mathematical model used to describe the vehicle. In what follows, a set of longitudinal equations for hovering is derived. These equations are indicative of the mathematical model that is currently prevalent in the VTOL flying qualities literature. The derivation is brief and intended primarily to emphasize that the final set of equations involves assumptions of sorts in their development and application. An in-depth review of these assumptions is not undertaken.

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The form of the resulting longitudinal equations for hover has the same form as the corresponding lateral-directional equations. Because of this similarity, the following discussion is applicable to the lateral-directional case, with the following changes in symbols:

$$u \rightarrow v \qquad w \rightarrow r \qquad \theta \rightarrow \phi$$
$$X_u \rightarrow Y_v \qquad M_w \rightarrow -L_v \qquad Z_u \rightarrow N_v$$
$$Z_w \rightarrow N_r \qquad Z_{\delta_{th}} \rightarrow N_{\delta_{RP}} \qquad M_q \rightarrow L_p$$
$$X_{\delta_{ES}} \rightarrow Y_{\delta_{AS}} \qquad M_{\delta_{ES}} \rightarrow L_{\delta_{AS}}$$

A general set of equations of motion linearized about some fixed operating point can be written in vector-matrix form as follows:

$$\dot{\bar{v}} = [A]\ \bar{v} + \bar{\Delta} \tag{1}$$

$$\bar{\Delta} = [B]\ \bar{\delta} + [C]\ \bar{v} \tag{2}$$

where

$\bar{v}$ = vector whose components are airframe motion variables

$[A]$ = matrix whose elements are stability derivatives (and g)

$\bar{\Delta}$ = vector whose components are forces and moments applied to airframe by the control system

$[B]$ = matrix whose elements are control derivatives

$\bar{\delta}$ = vector whose components are cockpit control deflections

$[C]$ = matrix whose elements are feedback gains (in the appropriate units) of a stability augmentation system.

Substituting (2) into (1), we obtain

$$\dot{\bar{v}} = [A]\ \bar{v} + [B]\ \bar{\delta} + [C]\ \bar{v}$$

or

$$\dot{\bar{v}} = [A + C]\ \bar{v} + [B]\ \bar{\delta} \tag{3}$$

Equation (3) shows that the elements of matrix [C] are incremental stability derivatives created by the stability augmentation system. In the current flying qualities literature the "natural" and "artificial" derivatives are not usually explicitly identified by using separate symbolism. That is, the elements of matrix [A+C] are the "effective" stability derivatives for the vehicle. The same procedure will be followed here and matrix [A+C] will be replaced by matrix [A*] so that Equation (3) becomes

$$\dot{\bar{v}} = [A^*]\ \bar{v} + [B]\ \bar{\delta} \tag{4}$$

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In an expanded form applicable to hover, Equation (4) becomes

$$\begin{bmatrix} \dot{w} \\ \dot{u} \\ \dot{q} \\ \dot{\theta} \end{bmatrix} = \begin{bmatrix} Z_w & Z_u & Z_q & Z_\theta \\ X_w & X_u & X_q & X_\theta \\ M_w & M_u & M_q & M_\theta \\ 0 & 0 & 1 & 0 \end{bmatrix} \begin{bmatrix} w \\ u \\ q \\ \theta \end{bmatrix} + \begin{bmatrix} Z_{\delta_{ES}} & Z_{\delta_{th}} & Z_{\delta_\lambda} \\ X_{\delta_{ES}} & X_{\delta_{th}} & X_{\delta_\lambda} \\ M_{\delta_{ES}} & M_{\delta_{th}} & M_{\delta_\lambda} \\ 0 & 0 & 0 \end{bmatrix} \begin{bmatrix} \delta_{ES} \\ \delta_{th} \\ \delta_\lambda \end{bmatrix} \tag{5}$$

The w equation is commonly used in single-degree-of-freedom height control studies. To make the w equation independent of the others, it is necessary to assume that the stability derivative matrix has Z_u, Z_q, and Z_θ equal to zero. By assuming further that $Z_{\delta_{ES}}$ and Z_{δ_λ} equal zero in hover, the w equation becomes a simple uncoupled first-order equation with a single input, δ_{th}. The above equations can now be written:

$$\begin{bmatrix} \dot{w} \\ \dot{u} \\ \dot{q} \\ \dot{\theta} \end{bmatrix} = \begin{bmatrix} Z_w & 0 & 0 & 0 \\ X_w & X_u & X_q & X_\theta \\ M_w & M_u & M_q & M_\theta \\ 0 & 0 & 1 & 0 \end{bmatrix} \begin{bmatrix} w \\ u \\ q \\ \theta \end{bmatrix} + \begin{bmatrix} 0 & Z_{\delta_{th}} & 0 \\ X_{\delta_{ES}} & X_{\delta_{th}} & X_{\delta_\lambda} \\ M_{\delta_{ES}} & M_{\delta_{th}} & M_{\delta_\lambda} \\ 0 & 0 & 0 \end{bmatrix} \begin{bmatrix} \delta_{ES} \\ \delta_{th} \\ \delta_\lambda \end{bmatrix} \tag{6}$$

The u and q equations are still aerodynamically coupled to the w equation through the derivatives X_w and M_w. These derivatives must be assumed zero to remove this coupling. The u and q equations are also "control-coupled" to the w equation through $X_{\delta_{th}}$ and $M_{\delta_{th}}$. Thus, assuming $X_{\delta_{th}}$ and $M_{\delta_{th}}$ equal to zero removes this control coupling.

Equation (6) can now be written as two independent sets:

$$\dot{w} = Z_w w + Z_{\delta_{th}} \delta_{th}$$

$$\begin{bmatrix} \dot{u} \\ \dot{q} \\ \dot{\theta} \end{bmatrix} = \begin{bmatrix} X_u & X_q & X_\theta \\ M_u & M_q & M_\theta \\ 0 & 1 & 0 \end{bmatrix} \begin{bmatrix} u \\ q \\ \theta \end{bmatrix} + \begin{bmatrix} X_{\delta_{ES}} & X_{\delta_\lambda} \\ M_{\delta_{ES}} & M_{\delta_\lambda} \\ 0 & 0 \end{bmatrix} \begin{bmatrix} \delta_{ES} \\ \delta_\lambda \end{bmatrix} \tag{7}$$

Eliminating the w equation from further consideration, assuming that $X_q = 0$ and $X_\theta = -g$, and also that the cockpit control for thrust vectoring (δ_λ) is not used, we are left with the following equations:

$$\begin{bmatrix} \dot{u} \\ \dot{q} \\ \dot{\theta} \end{bmatrix} = \begin{bmatrix} X_u & 0 & -g \\ M_u & M_q & M_\theta \\ 0 & 1 & 0 \end{bmatrix} \begin{bmatrix} u \\ q \\ \theta \end{bmatrix} + \begin{bmatrix} X_{\delta_{ES}} \\ M_{\delta_{ES}} \\ 0 \end{bmatrix} \delta_{ES} \tag{8}$$

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Equations (8) are considered to be the basic mathematical model of hovering and low speed VTOL dynamics in the discussion which follows.

The characteristic equation of the above set of equations is:

$$s^3 - (X_u + M_q)s^2 + (X_u M_q - M_\theta)s + X_u M_\theta + M_u g = 0 \tag{9}$$

In order to distinguish between some important stability features of vehicles whose augmentation systems provide various combinations of rate and attitude feedback, some root loci of the characteristic equation will be presented.

Root loci for rate systems ($M_\theta = 0$)

First consider that M_θ equals zero. The characteristic equation becomes:

$$s^3 - (X_u + M_q)s^2 + X_u M_q s + M_u g = 0 \tag{10}$$

To show the influence of $M_u g$ on the root, the equation can be written:

$$-1 = \frac{M_u g}{s(s - X_u)(s - M_q)} \tag{11}$$

Figure 2(3.2.2.1) is a sketch of the root loci as $M_u g$ varies. Figure 2a(3.2.2.1) shows that, as $M_u g$ becomes more positive, an unstable oscillatory pair develops. Figure 2b(3.2.2.1) shows that, as $M_u g$ becomes more negative, an unstable aperiodic mode develops.

The last form of the characteristic equation (Equation 11) is also useful in showing where, in the s-plane, oscillatory roots are likely to occur for a range of derivatives representative of that appearing in the flying qualities literature. Figure 3(3.2.2.1) shows the $M_u g$ loci for two pairs of M_q and X_u and identifies such a region for the majority of rate damped configurations considered during the data analysis.

To obtain the effects of M_q on the roots, the characteristic equation can be written

$$1 = \frac{M_q s(s - X_u)}{s^3 - X_u s^2 + M_u g} \tag{12}$$

Figure 4(3.2.2.1) shows the two possible forms of the M_q loci. In Figure 4(a)(3.2.2.1), the cubic $s^3 - X_u s^2 + M_u g$ has a pair of complex conjugate roots, while in Figure 4(b)(3.2.2.1) the roots are all real. As can be seen, making M_q more negative tends to stabilize both oscillatory and aperiodic modes. Since M_q and X_u are symmetrical in the characteristic equation, an

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interchange of the symbols X_u and M_q on Figure 4(3.2.2.1) will give the X_u locus.

Table 1(3.2.2.1) summarizes the previous discussion on rate systems.

Root loci for combined rate and attitude feedback

When M_θ is present, its influence on the roots can be studied by using the following form of the characteristic equation:

$$1 = \frac{M_\theta (s - X_u)}{s^3 - (X_u + M_q)s^2 + X_u M_q s + M_u g} \tag{13}$$

There are two forms of the M_θ locus as shown in Figure 5(3.2.2.1). The one that comes about in a particular numerical calculation depends on the relative positions of the poles and zero. In both cases shown, the dominant feature is the development of a high-frequency oscillatory mode that is more stable than the original oscillatory mode with $M_\theta = 0$. The real root moves towards the right on the real axis. The form of the characteristic equation allows us to interpret the behavior of the M_θ locus as being such that the total system damping remains constant. That is, M_θ does not appear in the coefficient of s^2, and hence cannot influence the sum of the real parts of the roots. Thus, when the damping factor, $\zeta\omega_n$, of the oscillatory mode increases, the damping factor of the aperiodic mode, $1/\tau$, must decrease. A similar interpretation holds if all the roots are real.

The influence of M_q when M_θ is present is similar to that when M_θ is absent. For example, Figure 6(3.2.2.1) is similar to Figure 4(a)(3.2.2.1). Although not shown, there is also an M_q locus similar to Figure 4(b)(3.2.2.1), but with $M_\theta \neq 0$.

The combined effects of M_θ and M_q can be determined from the following equation:

$$1 = \frac{M_q (s - X_u)\left(s + \frac{M_\theta}{M_q}\right)}{s^3 - X_u s^2 + M_u g} \tag{14}$$

If the ratio of M_θ to M_q remains constant as M_q varies, there are two possible loci as shown in Figure 7(3.2.2.1). The lower figure indicates that the situation is very similar to that of the M_q locus for rate systems. The upper figure shows that the oscillatory mode becomes very stable while the aperiodic mode moves toward the origin.

Table 2(3.2.2.1) summarizes the above discussion.

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Requirement Evolution

Root locus characteristics were guiding factors in analyzing data to formulate dynamic requirements. A particularly important feature of the loci is that which is seen by comparing the way the real root moves.

When M_β augmentation produces a high frequency oscillation, the real root moves toward the zero near the origin and takes on relatively small values. On the other hand, M_q augmentation which tends to reduce oscillatory mode frequencies and increase the damping ratio, drives the real root away from the origin to relatively large values.

If SAS systems could provide independent control over all the forces and moments that can be applied to a vehicle, then a proper selection of feedback control loops would allow for root locations of practically unlimited geometric configurations. However, since present systems are basically moment augmentors, the relative locations of real and oscillatory roots are not totally independent of each other. With such constraints it becomes important to determine how real and oscillatory roots influence the flying qualities characteristics experienced by the pilots. In general, this is not an easy question to answer especially within the context of writing specifications.

Pilot rating correlations using the available data did not result in precise partitioning of desired values of frequency, damping ratio, and time constants. Hence in formulating the initial version of dynamic requirements, consideration was given to both the pilot rating data and the results of closed-loop pilot model analyses. The results of these pilot-model analyses (References 42 and 43) indicated that configurations having satisfactory dynamics fell into the two general classifications that came to be called rate systems and attitude systems.

The first version of the requirements is outlined in Table 3(3.2.2.1). Note that these requirements allowed a vehicle to meet one or the other of two requirements depending on whether or not the vehicle had a stability augmentation system that was classified as a rate system or an attitude system. The use of this classification scheme seemed to provide a simple way to make specific statements about desired combinations of real and oscillatory mode parameters while at the same time remaining consistent with stability features indicated by root locus studies.

Review comments on the first version of dynamic requirements made it clear that the "rate system" and "attitude system" classification had some practical deficiencies which basically were the result of relying on the hovering cubic characteristic equation to be a reasonably faithful indicator of the kind of stability characteristics that might be encountered on real airplanes.

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As mentioned earlier, this cubic equation is based on an assumed mathematical model of low speed dynamics that consider the vertical motions to be uncoupled from the pitching and horizontal translational motions. This model is certainly valuable in analyzing the results of simulator experiments when the simulated configurations themselves are known to be described by essentially identical equations.

However, a specification must contend with numerous realistic possibilities that are not always adequately accounted for in basic mathematical models.

For example, the hovering cubic does not account for the situation where two pairs of oscillatory modes exist. Such an accounting can only occur by re-examining the assumption of an uncoupled height control mode. One of the industry review comments indicated that these assumptions were not necessarily valid and that it was realistic to expect two pairs of oscillatory roots.

The first version of dynamic requirements was not clear as to what happens when one oscillatory pair satisfied the attitude system boundary and the other oscillatory pair did not.

Another practical problem resulting from the use of the rate system and attitude system classification scheme in the specification stems from the fact that actual stability augmentation schemes may introduce additional dynamic modes so that the overall aircraft becomes a dynamic system with perhaps five or six characteristic roots rather than three or four. In such a case the classification scheme could easily lead to confusion in interpreting how the requirement should be applied.

The basic objective, then, in subsequent efforts aimed at revising the first set of dynamic requirements was to write a requirement paragraph that was less prone to ambiguities which could arise in practice but at the same time could still deal effectively with the combined influence of aperiodic and oscillatory modes on flying qualities. The concept of response matching was considered as a possible means of achieving the objective.

The idea of second-order response matching emerged from comparing the attitude time histories of configurations in both the rate- and attitude- system categories.

The attitude responses for these two classifications of vehicle dynamics are sketched in Figure 8(3.2.2.1). Also given there is the general form of the equation for $\theta(t)$ which results from obtaining the inverse transform of the following transfer function for a step elevator input.

$$\frac{\theta(s)}{\delta_{ES}(s)} = \frac{M_{\delta_{ES}}\left(s+\frac{1}{T_\theta}\right)}{\left(s+\frac{1}{T_{sp}}\right)\left(s^2+2\zeta_p\omega_p s+\omega_p{}^2\right)} \qquad \text{where} \quad \delta_{ES}(s)=\frac{\delta_{ES}}{s} \qquad (15)$$

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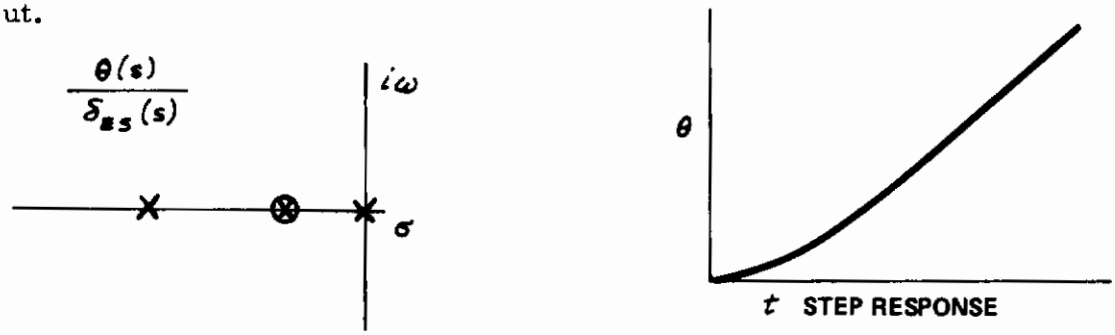
In an informal sense, we can generally say that the short-term or transient response of the rate system is a strong function of the real root value while that of the attitude system is a weak function of the real root value. To expand on this comment we will briefly consider two limiting cases.

Limiting case of a rate system - Suppose M_u and M_θ are both zero in the simplified equations of motion. Then the attitude transfer function will be of the form

$$\frac{\theta(s)}{\delta_{ES}(s)} = \frac{M_{\delta_{ES}}\left(s+\frac{1}{T_\theta}\right)}{s\left(s+\frac{1}{T_{SP}}\right)\left(s+\frac{1}{T_\theta}\right)} = \frac{M_{\delta_{ES}}}{s\left(s+\frac{1}{T_{SP}}\right)} \qquad (16)$$

where $1/T_{SP} = -M_q$. The pole-zero pattern for this transfer function is sketched below along with the corresponding time history, $\theta(t)$ for a step input.

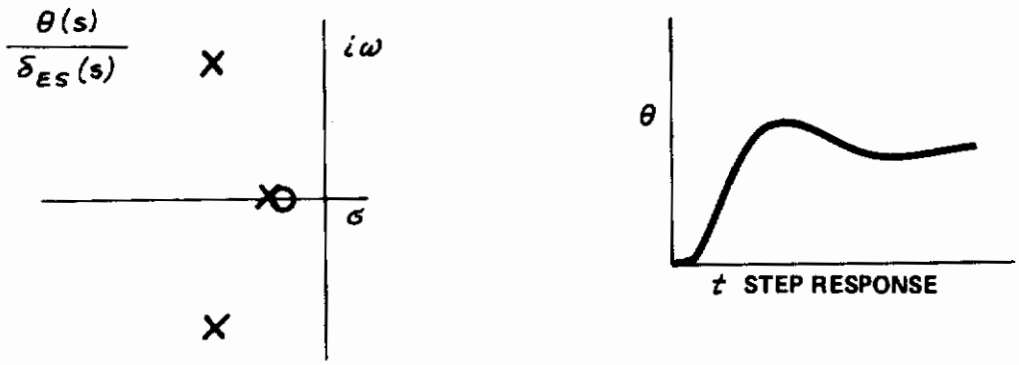


It is evident that the character of the response depends very much on the real root ($1/T_{SP}$) whose location depends on the amount of damping.

Limiting case of an attitude system - Suppose the derivative M_θ is very large. Then the attitude transfer function can be approximated quite accurately by

$$\frac{\theta(s)}{\delta_{ES}(s)} = \frac{M_{\delta_{ES}}\left(s+\frac{1}{T_\theta}\right)}{\left(s+\frac{1}{T_{SP}}\right)\left(s^2+2\zeta_p\omega_p s+\omega_p^2\right)} \approx \frac{M_{\delta_{ES}}}{\left(s^2+2\zeta_p\omega_p s+\omega_p^2\right)} \qquad (17)$$

The pole-zero pattern and time history for a step input are sketched below.



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In between these limiting cases are the intermediate situations where a good approximate transfer function cannot always be obtained by assuming pole-zero cancellation. That is, the actual transfer function must be used to determine the θ response accurately. However, depending on the values of the various coefficients ($1/T_\theta$, $1/T_{sp}$, ζ_p and ω_p), we can still have a response whose basic characteristics, during an appropriately selected time interval, are either of the rate-system type or of the attitude-system type.

The above considerations suggested the idea that the step input responses for both rate and attitude systems, are, "for all practical purposes", similar to second-order systems over some initial time interval. This led to attempts at matching attitude responses with an equivalent second-order transfer function of the general form

$$\frac{\theta_2(s)}{\delta_{ES}(s)} = \frac{K}{s^2 + a_0 s + b_0}, \quad \delta_{ES}(s) = \frac{1}{s} \tag{18}$$

The values of the coefficients a_0 and b_0 needed to achieve accurate time history matching are to be indicative of the amount of "damping" and "stiffness" possessed by the actual system.

Data correlations using the response matching technique indicated that the technique showed promise as an effective means for dealing with both "attitude" and "rate" type vehicle dynamics. (These data correlations will be discussed at a later date in a final project report, Reference 44.)

A set of dynamic requirements based on the response matching idea was included in a revised specification document which was published and sent through another review cycle. Table 4(3.2.2.1) outlines these dynamic requirements.

These revised requirements seemed to be free of many of the characteristics for which the previous version was criticized. In particular, there was no longer any explicit mention of rate and attitude systems. This was implicitly taken care of in the response-matching integral criterion. At the same time, the response-matching integral criterion appeared to account for the combined influence of various relative locations of aperiodic and oscillatory roots on pilot ratings. Also since this criterion uses time histories and not estimated characteristics, it seemed well suited for compliance testing.

The review comments received on the revised dynamic requirements gave the general impression that the use of response matching in a specification was somewhat unexpected. Some comments were enthusiastic. One contractor applied the response matching technique to his own aircraft and sent graphs showing how the vehicle compared with the requirement. A few other comments mentioned that evaluating the integral criterion seemed to require complex calculations. For the most part, the comments made it clear that more experience in applying the method would be needed before detailed constructive comments could be made.

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There was, however, one thing that no doubt influenced the reaction to response matching. That was the presence of a unique alternate dynamic requirement developed in-house by the Air Force Flight Dynamics Laboratory. This alternate requirement specified the acceptability of an aircraft in terms of a calculated pilot rating. The scheme for calculating the rating utilizes a mathematical technique that minimizes R (R for rating) in the equation

$$R = \frac{\sigma_\alpha + 10\sigma_\phi - 0.8}{0.8} + 2.5\, T_{L_\theta} + T_{L_\alpha} - 1.0 \qquad (19)$$

Calculations are performed on an analytical representation of the closed-loop pilot/aircraft combination. Reference 45 describes the development of this technique.

The alternate dynamics requirement, being very new, very interesting and very different, probably had the effect of attracting much of the attention that otherwise would have been directed solely to critiqueing response matching. The gist of the review comments on the alternate dynamics requirement was that it had the potential to become a valuable tool and further work to perfect the technique should be carried out but its use in a specification would be premature considering its relatively short and limited developmental history.

During the final Government review cycle, i.e., the one leading to the dynamic requirement 3.2.2.1 of Reference 1, the decision was made to omit altogether the alternate dynamics requirement and to delete the response matching part of the basic or main requirement. Several factors entered into this decision. Foremost among these was that the lack of practical experience with the proposed new ideas introduced into the requirement made it difficult to foresee the nature of the possible problems that would inevitably arise during an aircraft procurement cycle. This factor was given heavy emphasis.

It is believed that the new requirements (3.2.2) are reasonably consistent with the data base in that the oscillatory mode boundaries provide a fairly good separation of the pilot ratings associated with the various levels of handling qualities. Figures 9(3.2.2.1), 10(3.2.2.1), and 11(3.2.2.1) show the s-plane representations of oscillatory mode boundaries for Levels 1, 2, and 3 respectively. Pilot rating data are shown on Figures 12(3.2.2.1) through 48 (3.2.2.1) along with portions of one or more of the oscillatory mode boundaries depending on the region in the s-plane covered by the data. (The next section provides a guide to the figures which should be helpful to the reader.)

By selecting a few graphs at random, one could find points labeled with pilot ratings that tend to be at odds with a boundary and with each other. It must be remembered that the data reflect several factors that influence pilot ratings. These factors include:

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- the differences between fixed-base and moving-base simulation,
- the differences in pilots,
- the differences in values of X_u , $M_u g$, Y_v , $L_v g$ and,
- the differences in wind and gust magnitudes.

Thus an interpretation of the data on any one graph should be guided by the character of the entire set of data. Particular care should be taken when looking at the Reference 34 results in which high values of X_u (Y_v) were evaluated in steady winds. Although the rating data are shown on an s-plane format, this does not mean that pilot opinion was influenced primarily by oscillatory characteristics. Proper interpretation must rely on a very important part of the data which is not contained here: the pilot comments. These comments are contained in References 34, 35, and 36. Reference 34 has an extensive amount of comments for each configuration while References 35 and 36 contain abbreviated comments. These comments were relied upon to interpret the data as a whole.

Although, as mentioned above, the new requirements are believed to be generally consistent with much of the data, they are also believed to have an inadequacy, namely, a vehicle that behaves like an acceleration system could satisfy Level 1. It has been established that vehicles with such dynamics are considered to have unsatisfactory flying qualities. Evidence of this fact can be seen on many of the control sensitivity vs. damping graphs presented in the section discussing the control sensitivity requirement (paragraph 3.2.3.2). These graphs show that for vehicles having very low (including zero) values of X_u (Y_v) and M_u ($L_v g$), ratings of 3.5 or better are not likely to occur at very low values of pitch (roll) damping. As a numerical example, consider a vehicle having natural derivatives as follows:

$$X_u = -.05$$
$$M_u g = .004$$
$$M_q = -.20$$

Solution of the characteristic equation

$$s^3 - (X_u + M_q)s^2 + X_u M_q s + M_u g = 0$$

yields the following roots

$$s_1 = -.268$$
$$s_{2,3} = .009 \pm .122$$

These roots satisfy the Level 1 requirements. However they are close to the origin of the s-plane indicating that the dynamic behavior of the vehicle is approximately that of a pure acceleration system. The values of the above derivatives are not considered to be those of a highly improbable special case. For comparison purposes, Table 5(3.2.2.1), based on the numbers

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in References 13 and 16, shows the values of X_u, M_{ug}, and M_q (unaugmented) for several VTOL's and helicopters. This range of values does include the values chosen for the sample calculation and it is therefore not unreasonable to expect them to occur.

Data presentation

The data for the various dynamic configurations are shown in the form of pilot rating correlations with s-plane oscillatory root locations on Figures 12(3.2.2.1) through 48(3.2.2.1). The s-plane boundaries which are defined by the new dynamic requirements are shown in Figures 9(3.2.2.1) to 11(3.2.2.1). Portions of one or more of these boundaries appear on the pilot rating correlation graphs depending on the region of the s-plane covered by the data points. It should be noted that different pilot rating scales were used in the different experiments. The Cooper scale [Table 6(3.2.2.1)] was used in the experiments of References 35 and 36. The rating scale of Table 7(3.2.2.1) was used in the experiment of Reference 34.

The following is a guide to the figures.

Figure 12(3.2.2.1)

Moving-base. Dynamics identical in both pitch and roll axes. $X_u = -.05$, $M_{ug} = .330$, wind = 10 knot, $\sigma_{u_g} = \sigma_{v_g} = 3.4$ fps.

Figure 13(3.2.2.1)

Moving-base. Dynamics identical in both pitch and roll axes. $X_u = -.20$, $M_{ug} = .330$. Compares effects of winds, turbulence, stick force gradient on ratings. Winds were 10 knots or zero. Turbulence was $\sigma = 3.4$ fps or zero in both axes.

Figures 14(3.2.2.1) to 17(3.2.2.1)

Moving-base longitudinal study. Lateral dynamics held constant. $Y_v = -0.1$, $L_{vg} = .164$, $L_p = -5.0$. Wind = 10 knots. $\sigma_{u_g} = \sigma_{v_g} = 3.4$ fps. Figures show X_u-M_{ug} families as indicated in the following table.

	$M_{ug} = .33$	$M_{ug} = 1.0$
$X_u = -.05$	Fig. 14	Fig. 15
$X_u = -.20$	Fig. 16	Fig. 17

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Figures 18(3.2.2.1) to 21(3.2.2.1)

Moving-base lateral study. Longitudinal dynamics held constant. X_u = - 0.1, $M_u g$ = .32, M_q = -3.0. Wind = 10 knots. σ_{u_g} = σ_{v_g} = 3.4 fps. Figures show $Y_v - L_v g$ families as indicated in the following table.

	$L_v g$ = -.33	$L_v g$ = 1.0
Y_v = -.05	Fig. 18	Fig. 19
Y_v = -.20	Fig. 20	Fig. 21

Figures 22(3.2.2.1) to 28(3.2.2.1)

Fixed-base longitudinal study. Lateral dynamics held constant. Y_v = - 0.1, $L_v g$ = - 1.1, L_p = - 3.0. No steady wind. Longitudinal turbulence, σ_{u_g} = 5.1 fps. Lateral turbulence, σ_{v_g} = 1.3 fps. Figures show families as indicated in the following table.

$X_u \rightarrow$	0	-.05	-.10	-.20	-.25	-.30
$\downarrow M_u g$	-	-	-	-	-	-
0.33	Fig. 22	Fig. 23	Fig. 22	Fig. 24	Fig. 22	Fig. 22
0.67	-	-	Fig. 25	-	-	Fig. 25
1.00	-	Fig. 26	-	Fig. 27	Fig. 28	Fig. 28

Figures 29(3.2.2.1) and 30(3.2.2.1)

Fixed-base longitudinal study. Lateral dynamics held constant. Y_v = - 0.1, $L_v g$ = - 0.1, L_p = - 3.0. No steady wind. No turbulence. σ_{u_g} = σ_{v_g} = 0. Figures show $X_u - M_u g$ families as indicated in the following table

	$M_u g$ = .33	$M_u g$ = 1.0
X_u = -.05	Fig. 29	Fig. 30
X_u = .25	Fig. 29	Fig. 30

Figures 31(3.2.2.1) and 32(3.2.2.1)

Fixed-base longitudinal study. Lateral dynamics held constant. Y_v = - 0.1, $L_v g$ = - 0.1, L_p = - 3.0. No steady wind. Longitudinal turbulence, σ_{u_g} = 2.6 fps. Lateral turbulence, σ_{v_g} = 1/4 σ_{u_g}. Figures show X_u - $M_u g$ families as indicated in following table.

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	$M_u g$ = .33	$M_u g$ = 1.0
X_u = -.05	-	Fig. 32
X_u = -.25	Fig. 31	Fig. 32

Figures 33(3.2.2.1) and 34 (3.2.2.1)

Fixed-base longitudinal study. Lateral dynamics held constant. Y_v = - 0.1, $L_v g$ = - 0.1, L_p = - 3.0. No steady wind. Longitudinal turbulence, σ_{u_g} = 7.7 fps. Lateral turbulence, σ_{v_g} = 1/4 σ_{u_g}. Figures show X_u - $M_u g$ families as indicated in following table.

	$M_u g$ = .33	$M_u g$ = 1.0
X_u = -.05	Fig. 33	Fig. 34
X_u = -.25	Fig. 33	Fig. 34

Figures 35(3.2.2.1) to 41(3.2.2.1)

Fixed-base lateral study. Longitudinal dynamics held constant. X_u = - .10, $M_u g$ = 0.67, M_q = - 3.0. No steady wind. σ_{v_g} = 5.1 fps. σ_{u_g} = 1.3 fps. Figures show Y_v - $L_v g$ families as indicated in following table.

$Y_v \rightarrow$	0	-.05	-.10	-.20	-.25	-.30
$\downarrow L_v g$						
-.33	Fig. 35	Fig. 36	Fig. 35	Fig. 37	Fig. 35	Fig. 35
-.67	Fig. 38	-	Fig. 38	Fig. 38	-	Fig. 38
-1.00	Fig. 39	Fig. 40	Fig. 39	Fig. 41	Fig. 39	Fig. 39

Figures 42(3.2.2.1) to 45(3.2.2.1)

Fixed-base longitudinal study. X_u = - .05, $M_u g$ values are 0.10, 0.67, 1.0, 2.0. Lateral dynamics held constant. Y_v = - 0.01, $L_v g$ = - 0.1, L_p = - 3.0. No steady winds. σ_{u_g} = 5.1 fps. σ_{v_g} = 1.3 fps.

Figures 46(3.2.2.1) to 48(3.2.2.1)

Fixed-base lateral study. Y_v = - .05, $L_v g$ values are -0.10, -1.0, -2.0. Longitudinal dynamics held constant. X_u = - .10, $M_u g$ = 0.67, M_q = - 3.0. No steady winds. σ_{v_g} = 5.1 fps. σ_{u_g} = 1.3 fps.

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TABLE 1(3.2.2.1)

SUMMARY OF RATE FEEDBACK STABILITY FEATURES

Characteristic equation: $s^3 - (X_u + M_q) s^2 + X_u M_q s + M_{ug} = 0$

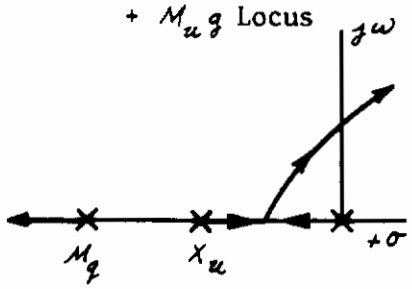
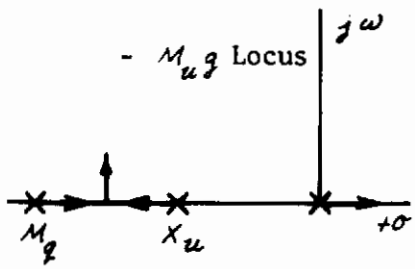
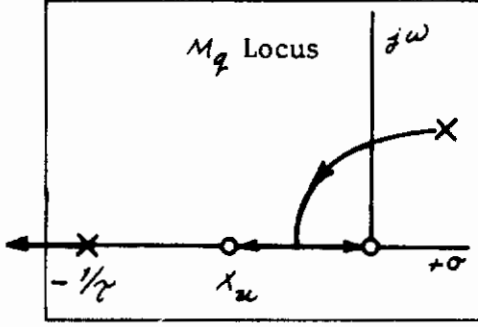
Root Locus	Features
	Increasing M_{ug} destabilizes oscillatory mode and stabilizes aperiodic mode. If M_{ug} = 0, pitch response is influenced only by M_q since equations of motion become uncoupled.
	Negative M_{ug} causes aperiodic divergence.
	More negative M_q has overall stabilizing effect. More negative X_u is similar because of symmetry in equations.

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TABLE 2(3.2.2.1)

SUMMARY OF ATTITUDE FEEDBACK STABILITY FEATURES

Characteristic equation: $s^3 - (X_u + M_q)s^2 + (X_u M_q - M_\theta)s + X_u M_\theta + M_u g = 0$

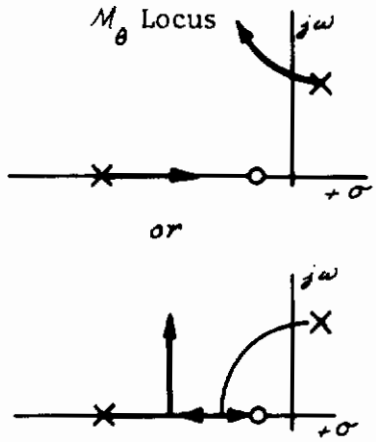
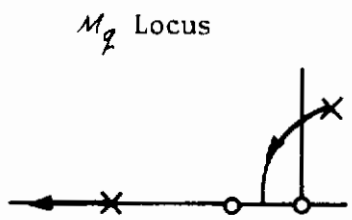
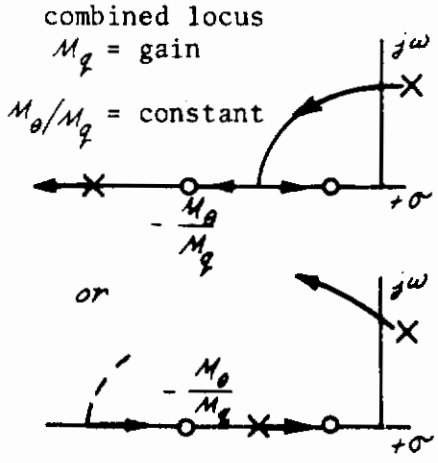
Root Locus	Features
	Stabilizing effect on oscillatory mode for low values of M_θ. Development of high-frequency oscillatory mode at high values of M_θ. Aperiodic mode becomes less stable.
	Similar to situation when $M_\theta = 0$, i.e., rate feedback system.
	Particular locus depends on ratio of M_θ to M_q. When ratio is low so that pole lies outside zeros, situation is similar to $M_\theta = 0$ or rate feedback system. When ratio of M_θ to M_q is high so that pole lies between zeros, oscillatory mode is highly stabilized. Aperiodic mode moves toward right (less stable)

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TABLE 3(3.2.2.1)

OUTLINE OF INITIAL VERSION OF DYNAMIC REQUIREMENTS (REFERENCE 3)

RATE SYSTEMS (AIRCRAFT WHICH DO NOT USE ATTITUDE OR INTEGRATED ANGULAR RATE STABILIZATION DEVICES)

OSCILLATORY MODE

LEVEL 1 PERIOD GREATER THAN 12 SECONDS: $\zeta > -0.10$
LEVEL 2 PERIOD GREATER THAN 7.5 SECONDS: $\zeta > -0.30$
LEVEL 3 PERIOD GREATER THAN 5 SECONDS: $\zeta > -0.50$

APERIODIC MODE

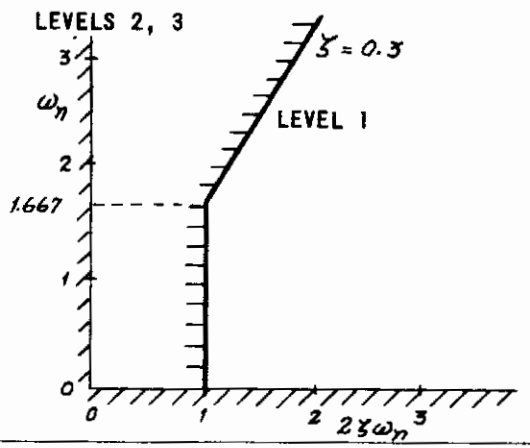
LEVEL 1 $T_{1/2}$ SHALL NOT EXCEED 0.70 SECONDS
LEVEL 2 NO DIVERGENCES IN PITCH. $T_{1/2}$ NOT OVER 1.4 SECONDS FOR ROLL
LEVEL 3 NO DIVERGENCES

ATTITUDE SYSTEMS (AIRCRAFT WHICH USE ATTITUDE OR INTEGRATED ANGULAR RATE STABILIZATION DEVICES)

SATISFY EITHER RATE SYSTEM REQUIREMENTS ABOVE OR THE FOLLOWING:

OSCILLATORY MODE

SATISFY BOUNDARIES OF FIGURE:



APERIODIC MODE

NO DIVERGENT TENDENCIES

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TABLE 4(3.2.2.1)

OUTLINE OF REVISED DYNAMIC REQUIREMENTS (REFERENCE 6)

Level 1 Aperiodic Mode

All aperiodic responses stable

Oscillatory Mode

All oscillations with $\omega_n > 0.5$ rad/sec stable
All oscillations with $\omega_n \leq 0.5$ rad/sec shall have $\zeta > -0.10$

Response Matching

The integral I shall be minimized

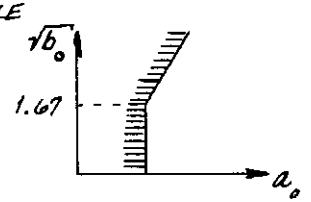
$$I = \int_0^{2.75} W(t)\left[\theta(t+\tau) - \theta_M(t)\right]^2 dt$$

$W(t)$ is a wieghting function, $W(t) = 3 + 2\cos\left(\frac{\pi t}{3}\right)$

$\theta(t)$ is actual response
$\theta(t+\tau)$ is shifted response to account for real abrupt step input
$\theta_M(t)$ is second-order response to ideal step input

$$\theta_M(t) = \frac{K}{s(s^2 + a_0 s + b_0)} \qquad K \text{ ADJUSTABLE}$$

values of a_0 and b_0 that minimize integral shall satisfy figure →



Level 2 Aperiodic Mode

Time to double not less than 12 seconds

Oscillatory Mode

Time to double not less than 10 seconds provided $\omega_n \leq 0.84$ rad/sec
if $\omega_n > 0.84$ rad/sec, oscillations shall be stable

Level 3 Aperiodic Mode

Time to double not less than 5 seconds

Oscillatory Mode

Time to double not less than 5 seconds provided $\omega_n \leq 1.25$ rad/sec
if $\omega_n > 1.25$ rad/sec, oscillations shall be stable

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TABLE 5(3.2.2.1)

HOVERING STABILITY DERIVATIVES FOR SEVERAL VTOL's AND HELICOPTERS

(Data From References 13 and 16)

Aircraft	X_u	M_{uq}	M_q
X-22A	-.201	.869	-.129
VZ-4DA	-.137	.409	-.045
XC-142A	-.153	.309	-.203
X-19	.0345	.106	-.248
XV-5A	-.110	.284	0
SH-3A	-.016	.0967	-.543
H-34A	0	.103	-.563
H-19	-.028	.193	-.610
UH-1D	-.009	.003	-.202
CH-46	-.022	-.03	-.982
HUP-1	-.019	.112	-1.980

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TABLE 6(3.2.2.1)

COOPER PILOT RATING SCALE

Operating Conditions	Adjective Rating	Numerical Rating	Description	Primary Mission Accomplished	Can Be Landed
Normal Operation	Satisfactory	1	Excellent, includes optimum	Yes	Yes
		2	Good, Pleasant to fly	Yes	Yes
		3	Satisfactory, but with some mildly unpleasant characteristics	Yes	Yes
Emergency Operation	Unsatisfactory	4	Acceptable, but with unpleasant characteristics	Yes	Yes
		5	Unacceptable for normal operation	Doubtful	Yes
		6	Acceptable for emergency condition only[1]	Doubtful	Yes
No Operation	Unacceptable	7	Unacceptable even for emergency condition[1]	No	Doubtful
		8	Unacceptable - dangerous	No	No
		9	Unacceptable - uncontrollable	No	No
	Catastrophic	10	Motions possibly violent enough to prevent pilot escape	No	No

[1] Failure of a stability augmenter.

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TABLE 7(3.2.2.1)

INTERIM COOPER-HARPER PILOT RATING SCALE

CONTROLLABLE Capable of being controlled or managed in context of mission with available pilot attention.	**ACCEPTABLE** May have deficiencies which warrant improvement, but adequate for mission. Pilot compensation, if required to achieve acceptable performance, is feasible.	**SATISFACTORY** Meets all requirements and expectations, good enough without improvement. Clearly adequate for mission	Excellent, highly desirable.	A1
			Good, pleasant, well behaved.	A2
			Fair. Some mildly unpleasant characteristics. Good enough for mission without improvement.	A3
		UNSATISFACTORY Reluctantly acceptable. Deficiencies which warrant improvement. Performance adequate for mission with feasible pilot compensation.	Some minor but annoying deficiencies. Improvement is requested. Effect on performance is easily compensated for by pilot.	A4
			Moderately objectionable deficiencies. Improvement is needed. Reasonable performance requires considerable pilot compensation.	A5
			Very objectionable deficiencies. Major improvements are needed. Requires best available pilot compensation to achieve acceptable performance.	A6
	UNACCEPTABLE Deficiencies which require mandatory improvement. Inadequate performance for mission even with maximum feasible pilot compensation.		Major deficiencies which require mandatory improvement for acceptance. Controllable. Performance inadequate for mission, or pilot compensation required for minimum acceptable performance in mission is too high.	U7
			Controllable with difficulty. Requires substantial pilot skill and attention to retain control and continue mission.	U8
			Marginally controllable in mission. Requires maximum available pilot skill and attention to retain control.	U9
UNCONTROLLABLE Control will be lost during some portion of the mission.			Uncontrollable in mission	10

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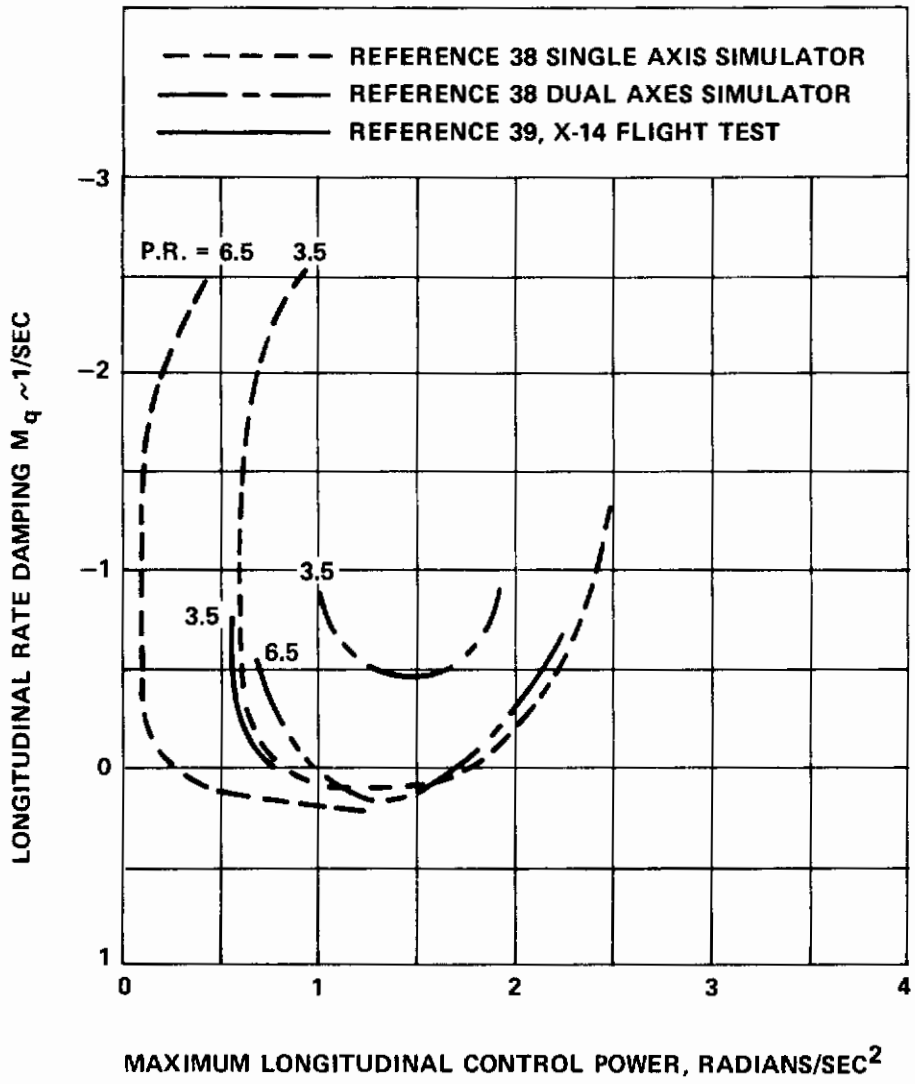

Figure 1 (3.2.2.1) COMPARISON OF FLIGHT AND SIMULATOR RESULTS FOR THE PITCH AXIS

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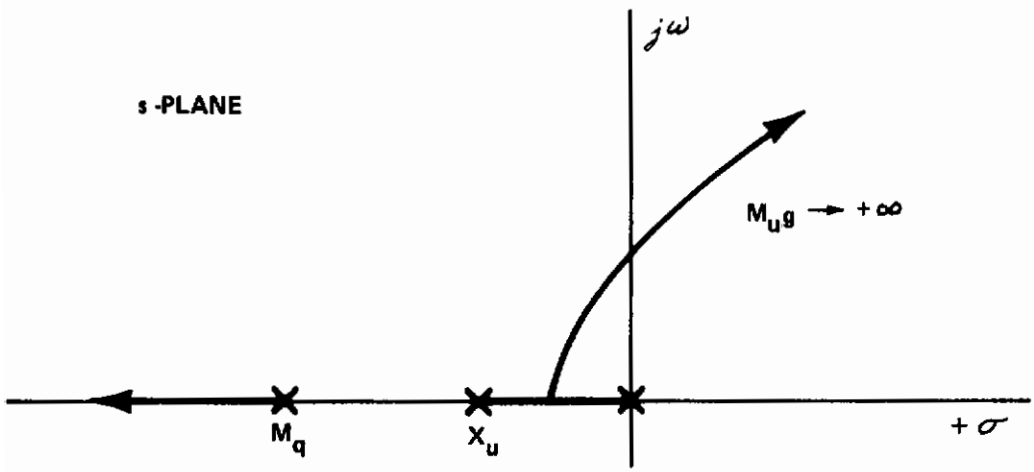
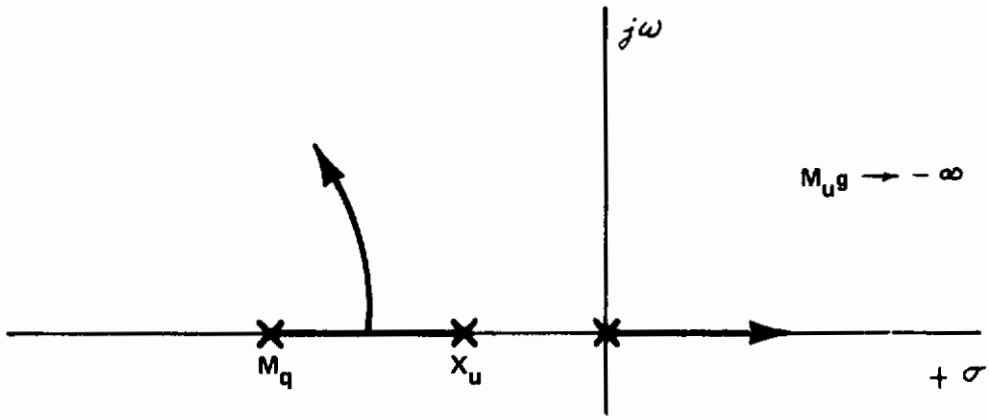


2(a)



2(b)

Figure 2 (3.2.2.1) $M_u g$ ROOT LOCI ($M_\theta = 0$)

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$$1 = \frac{M_q\, s(s - X_u)}{s^3 - X_u s^2 + M_u g}$$

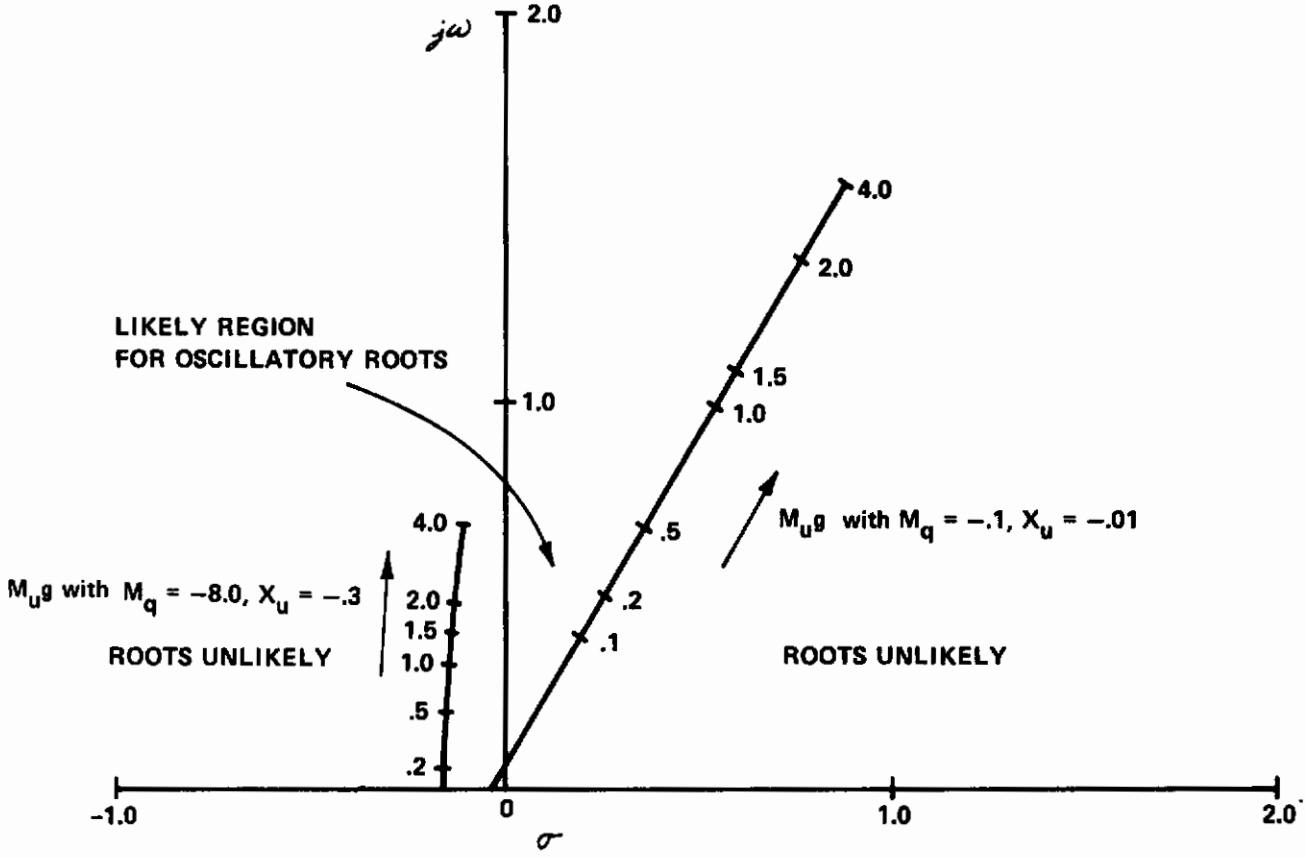

Figure 3 (3.2.2.1) $M_u g$ ROOT LOCI SHOWING LIKELY REGION OF OSCILLATORY ROOTS FOR A RANGE OF M_q AND X_u

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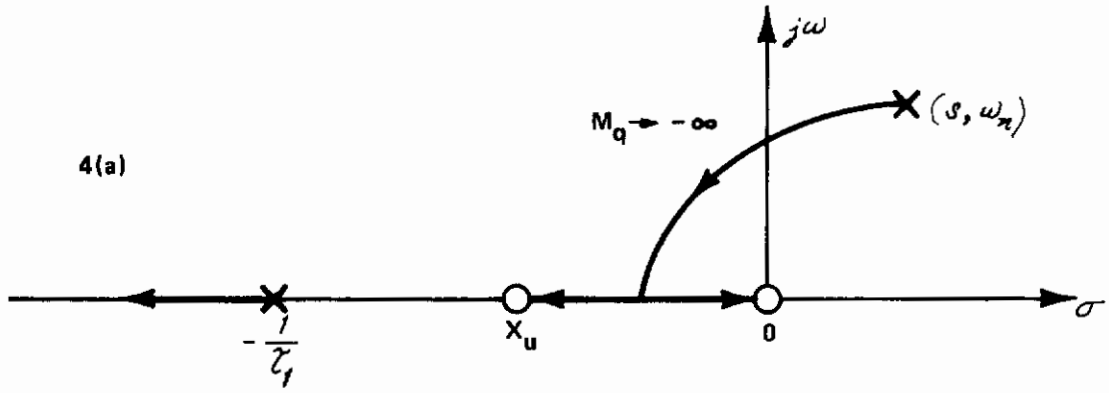
$$1 = \frac{M_q\, s(s - X_u)}{s^3 - X_u\, s^2 + M_w\, g} = \frac{M_q\, s(s - X_u)}{\left(s + \frac{1}{\tau}\right)\left(s^2 + 2\zeta\omega_n s + \omega_n^2\right)}$$

4(a)



$$1 = \frac{M_q\, s(s - X_u)}{s^3 - X_u\, s^2 + M_w\, g} = \frac{M_q\, s\,(s - X_u)}{\left(s + \frac{1}{\tau_1}\right)\left(s + \frac{1}{\tau_2}\right)\left(s + \frac{1}{\tau_3}\right)}$$

4(b)

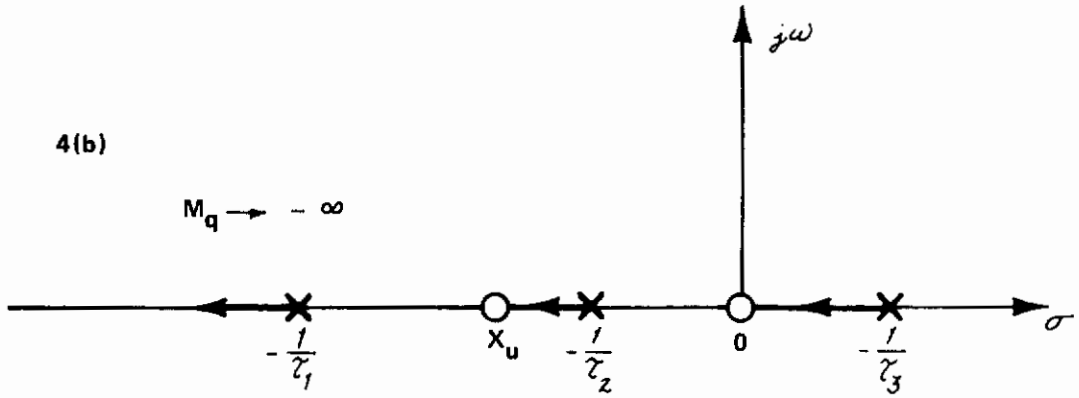


Figure 4 (3.2.2.1) M_q ROOT LOCI ($M_\theta = 0$)

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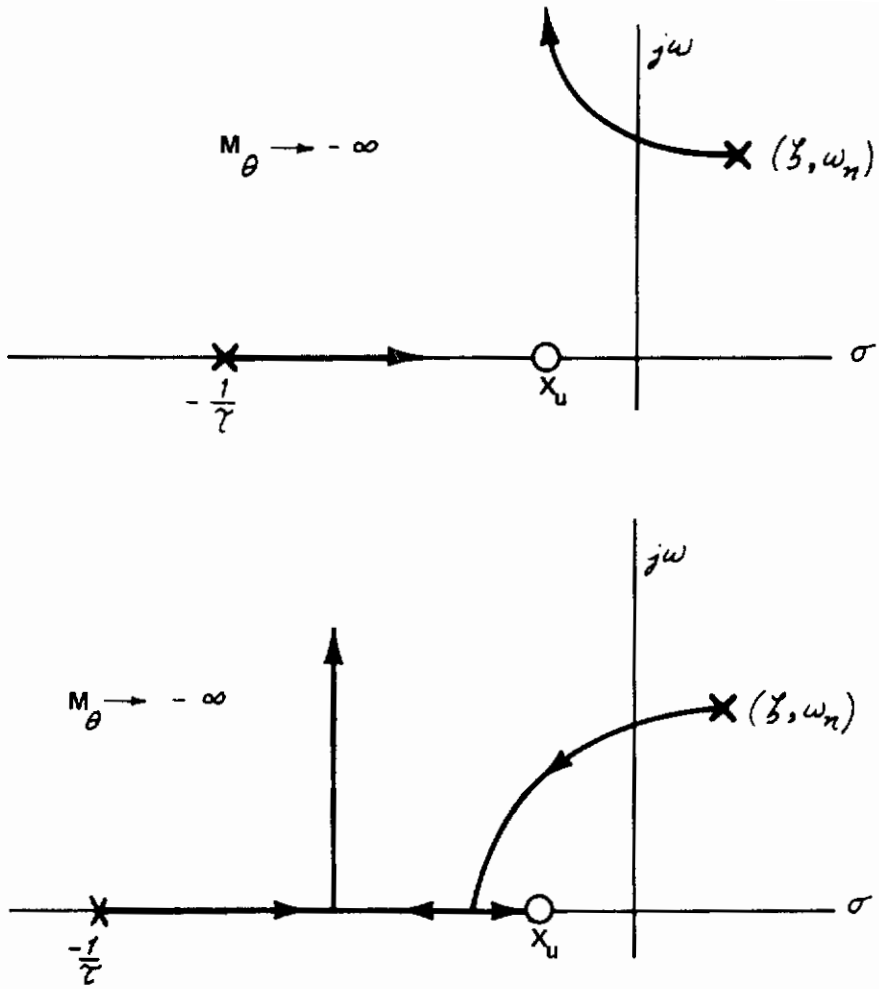

Figure 5 (3.2.2.1) M_θ ROOT LOCI

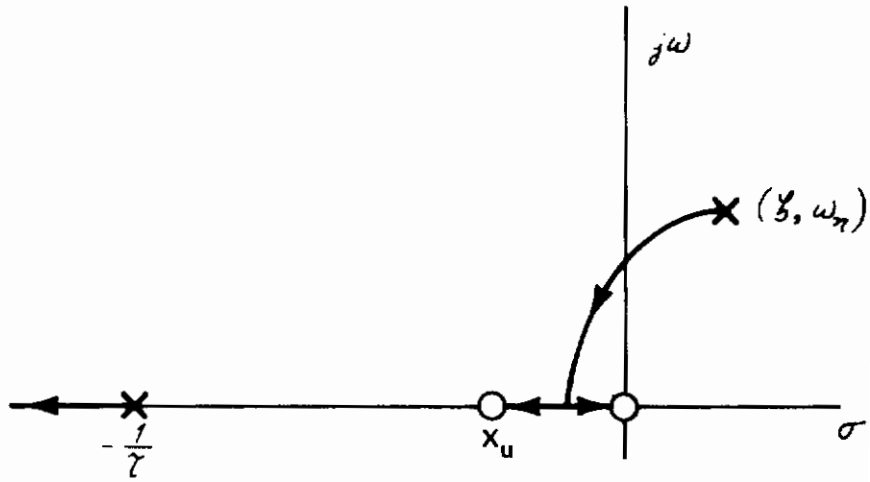

Figure 6 (3.2.2.1) M_q ROOT LOCI ($M_\theta \neq 0$)

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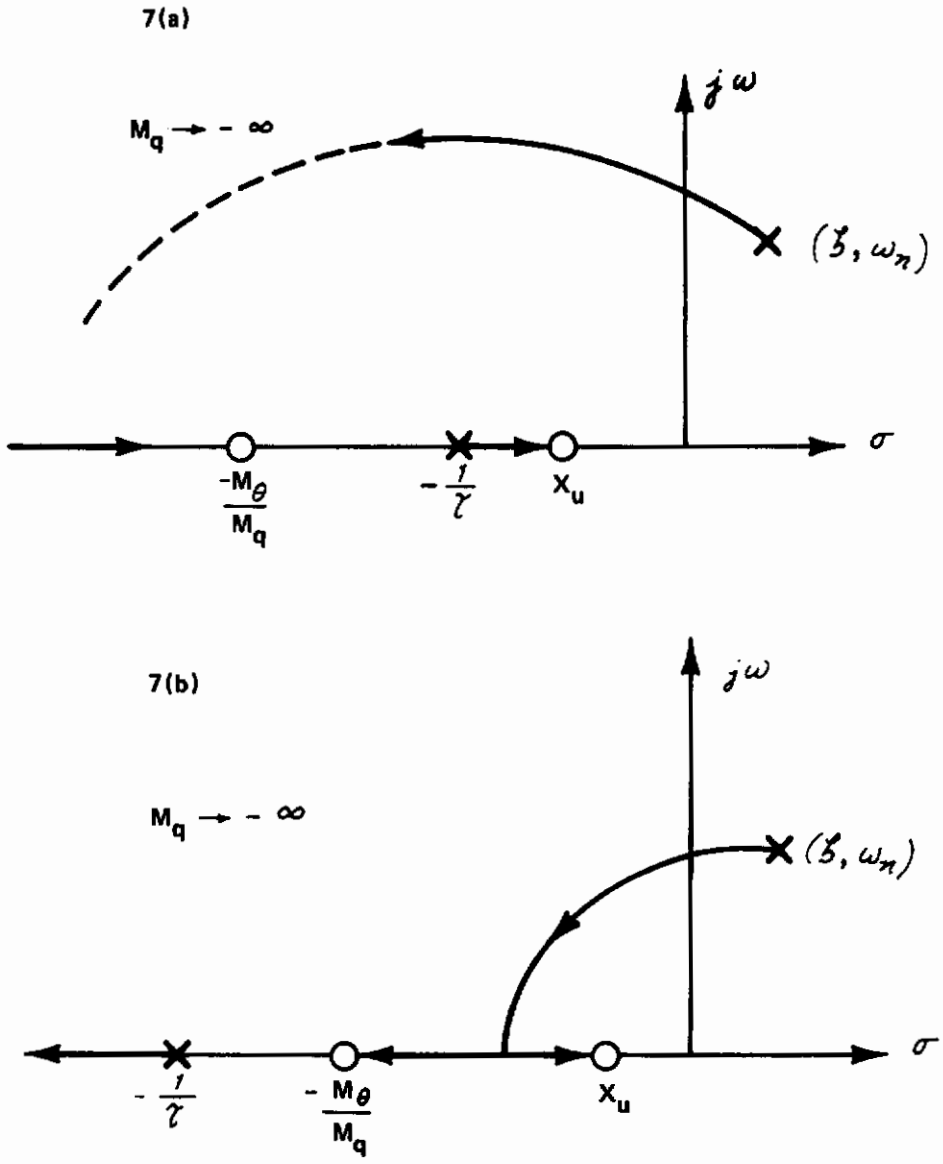

Figure 7 (3.2.2.1) COMBINED M_q AND M_θ LOCUS (M_θ VARIED TO KEEP M_θ/M_q CONSTANT)

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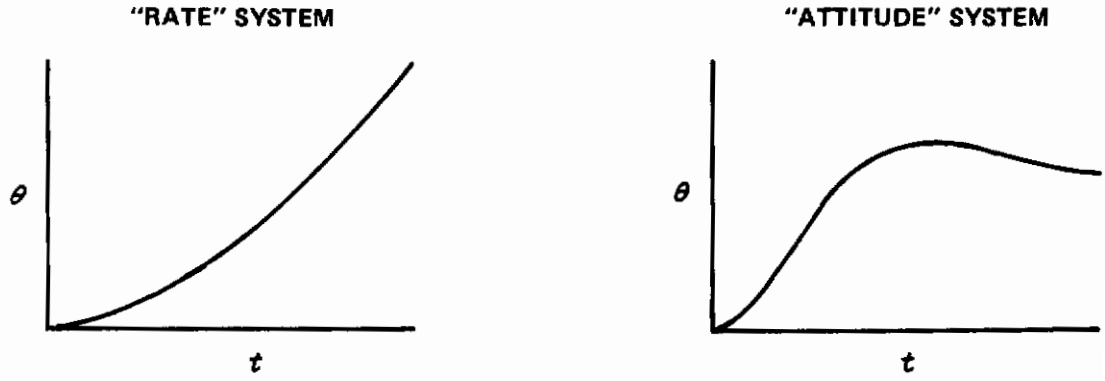

θ-TRANSFER FUNCTION:

$$\frac{\theta(s)}{\delta_{ES}(s)} = \frac{M_{\delta_{AS}}\left(s + \frac{1}{T_\theta}\right)}{\left(s + \frac{1}{T_{sp}}\right)\left(s^2 + 2\zeta_p\omega_p s + \omega_p^2\right)}$$

GENERAL EQUATION FOR θ-RESPONSE TO STEP INPUT $\left(\delta_{ES}(s) = \frac{\delta_{ES}}{s}\right)$:

$$\theta(t) = Ae^{-t/T_{sp}} + B + Ce^{-\zeta_p\omega_p t}\sin\left(\omega_p\sqrt{1-\zeta_p^2}\ t + \psi\right)$$

Figure 8 (3.2.2.1) SKETCHES OF ATTITUDE RESPONSES TO STEP INPUTS FOR THIRD-ORDER SYSTEMS

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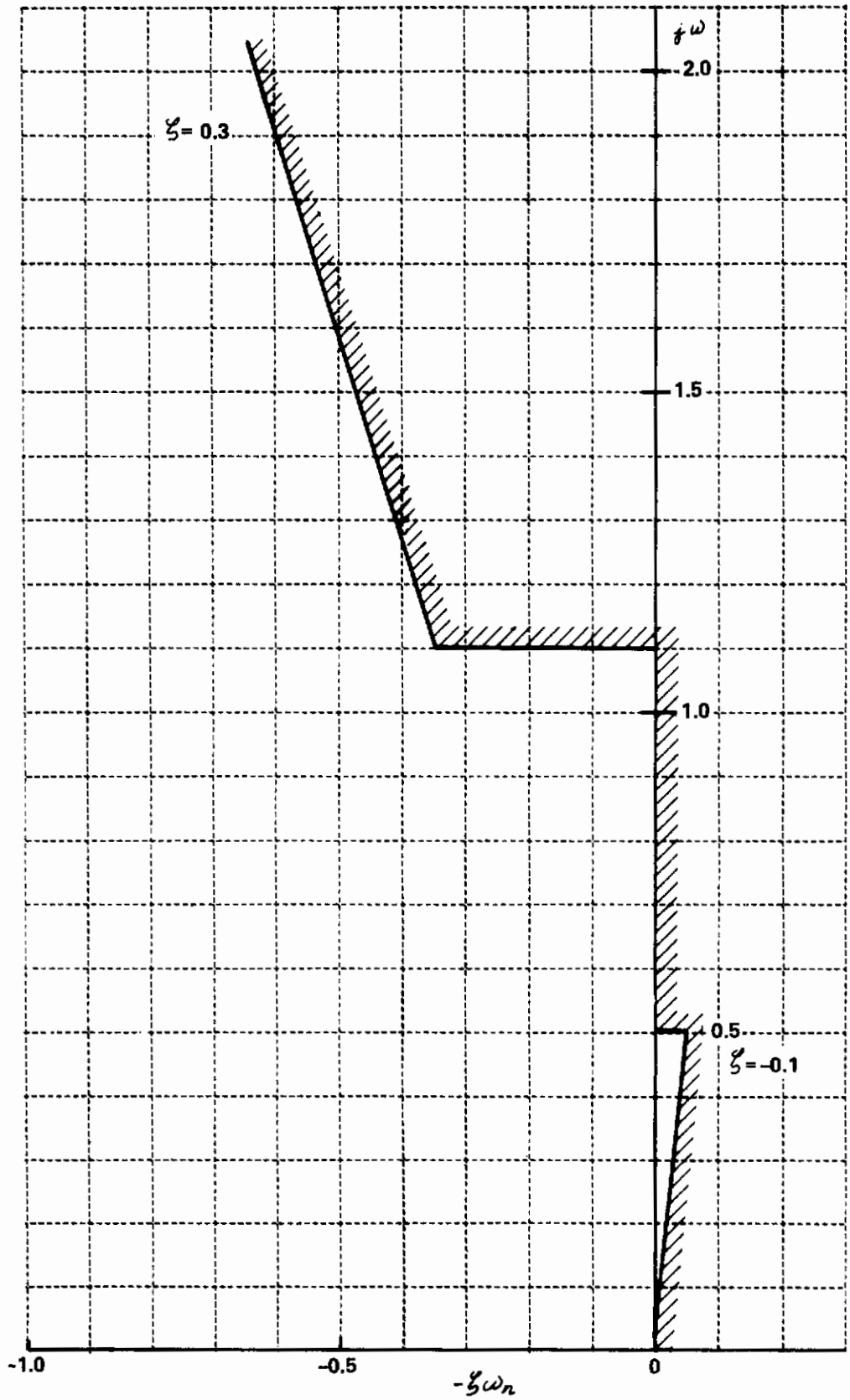

Figure 9 (3.2.2.1) LEVEL 1, AND LEVEL 2 IFR s-PLANE OSCILLATORY MODE BOUNDARY

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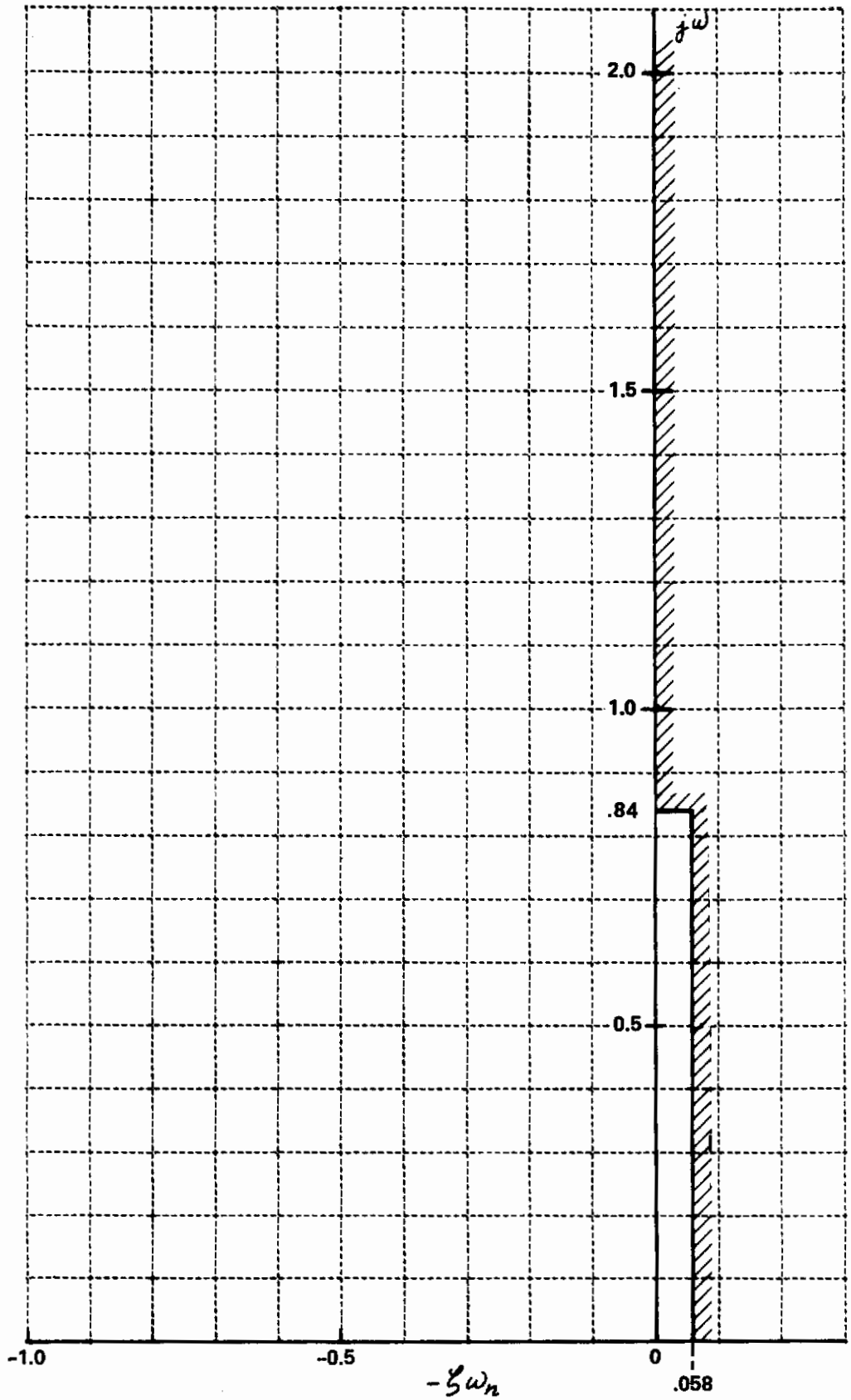

Figure 10 (3.2.2.1) LEVEL 2 VFR s-PLANE OSCILLATORY MODE BOUNDARY

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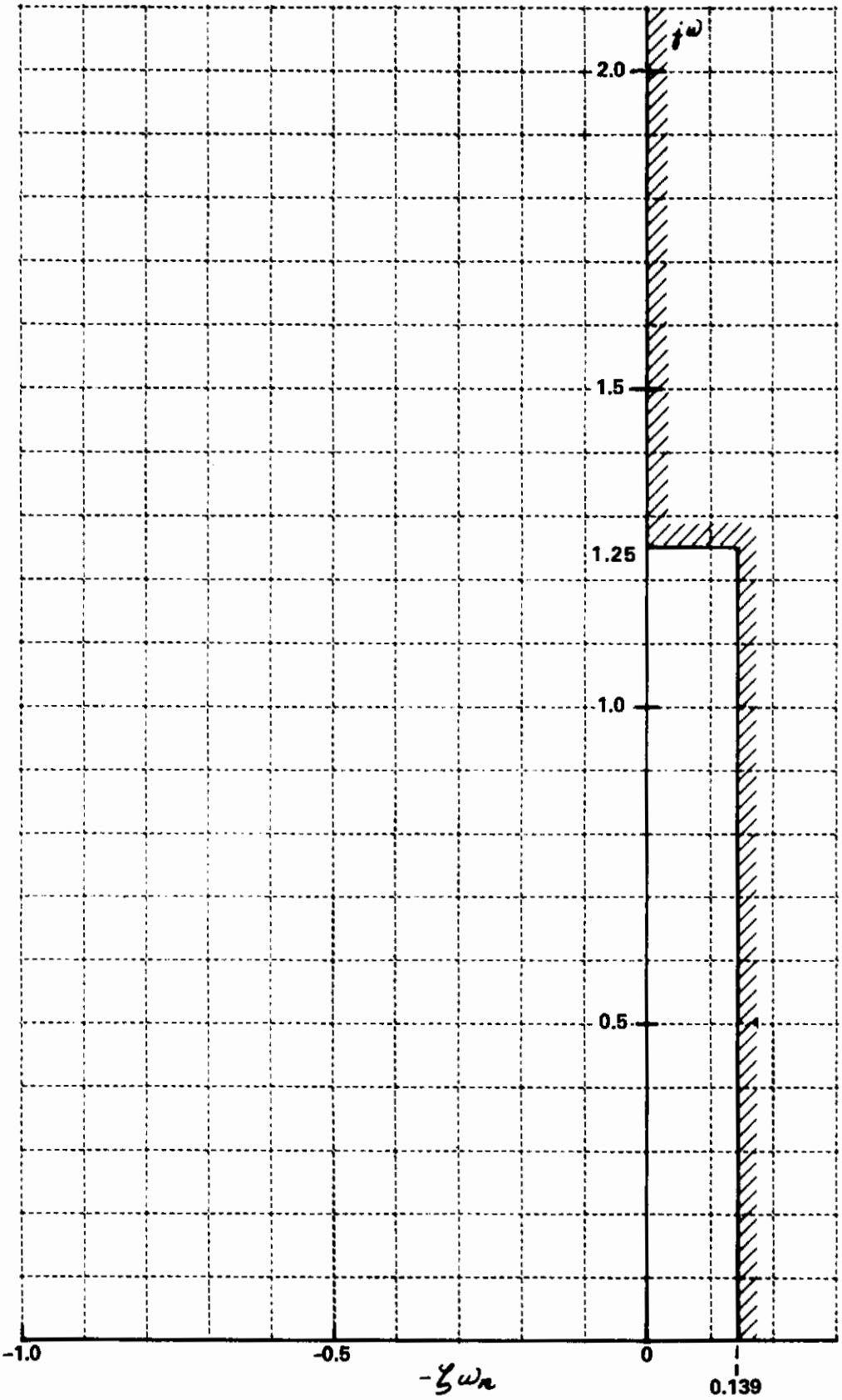

Figure 11 (3.2.2.1) LEVEL 3 s-PLANE OSCILLATORY MODE BOUNDARY

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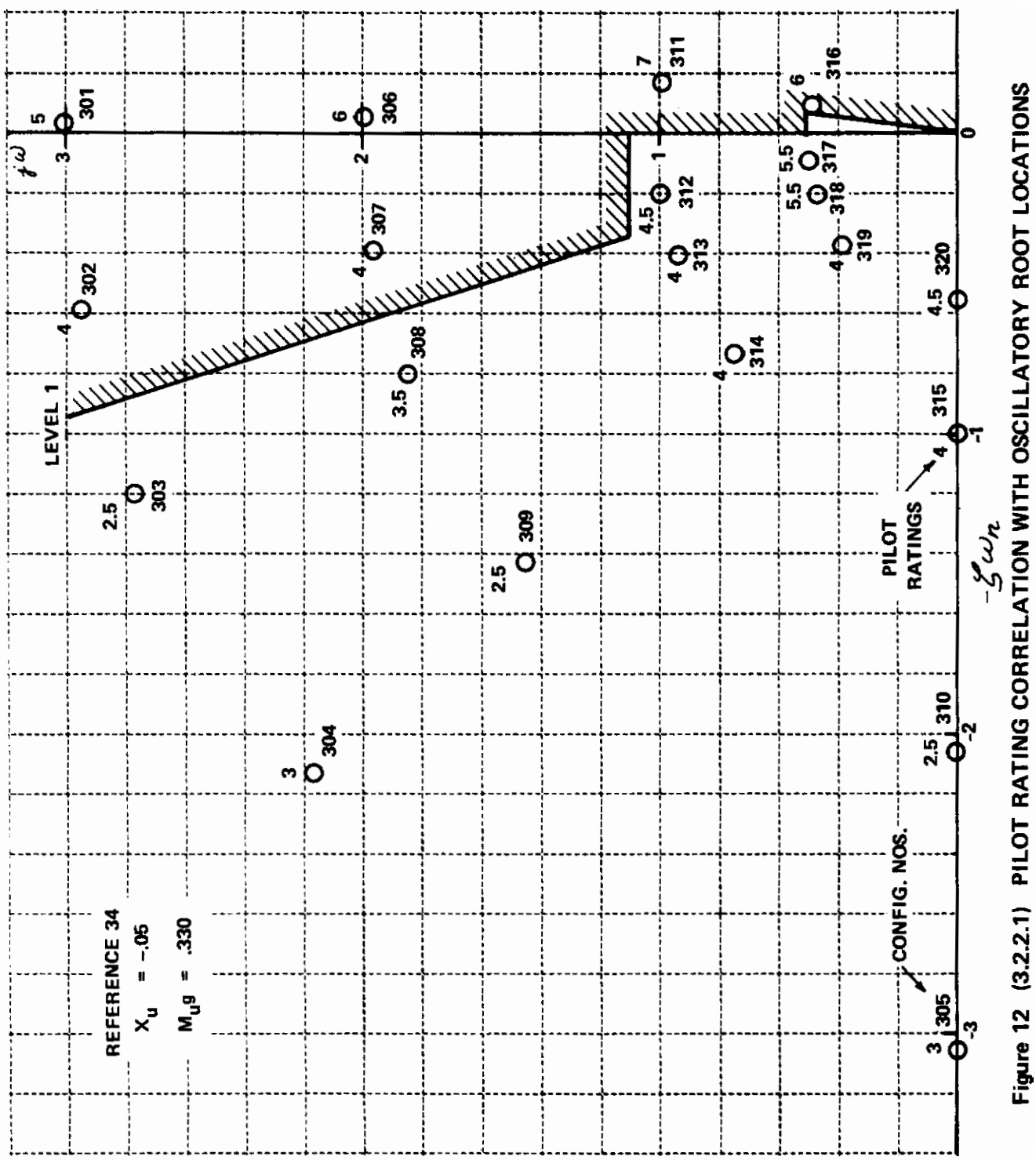

Figure 12 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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REF. 34 $X_u = -.20$ $M_u g = .33$

NOTE: POOR RATING CAUSED BY TRIM ATTITUDE RESULTING FROM LARGE X_u AND WIND

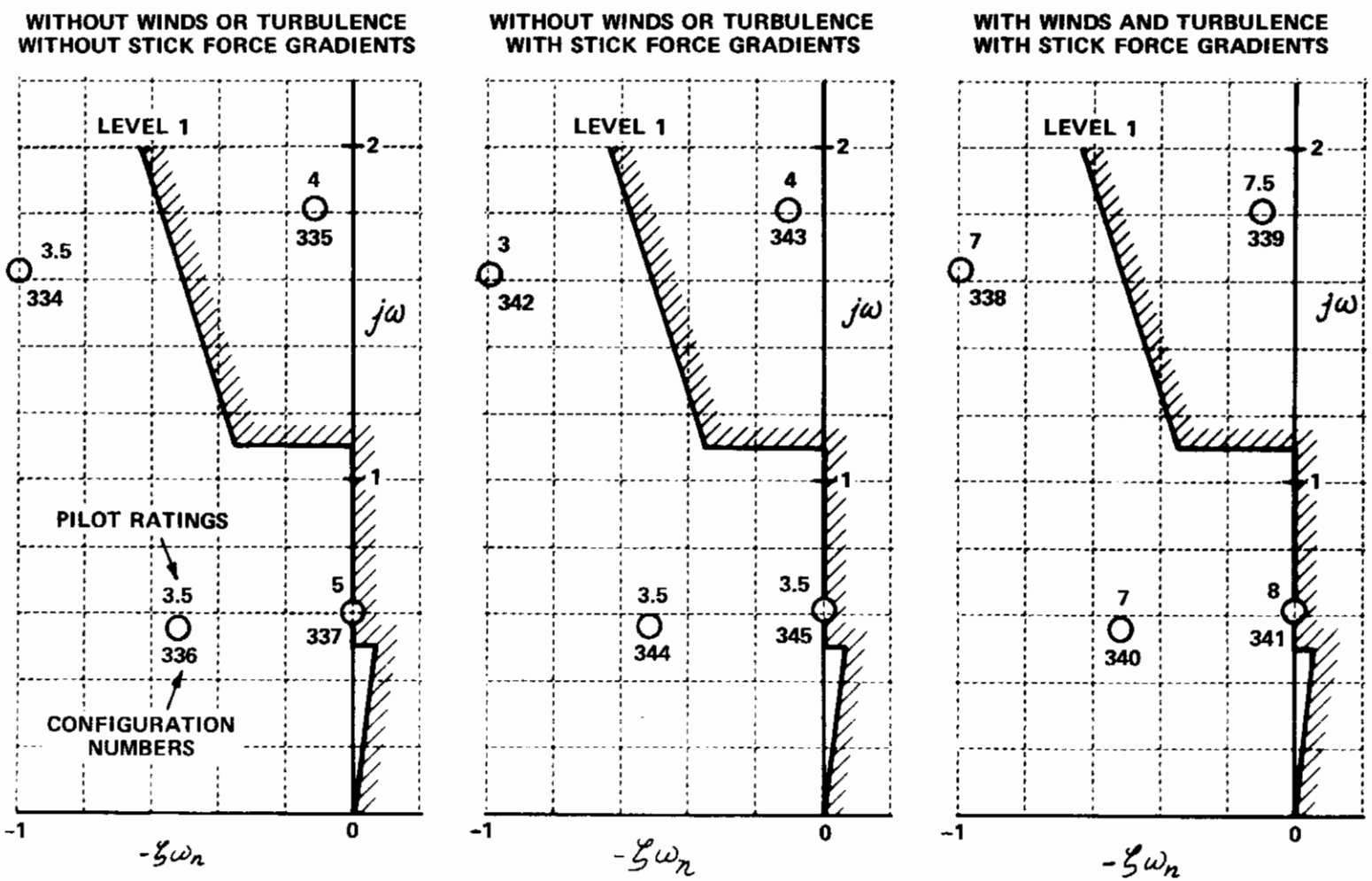

Figure 13 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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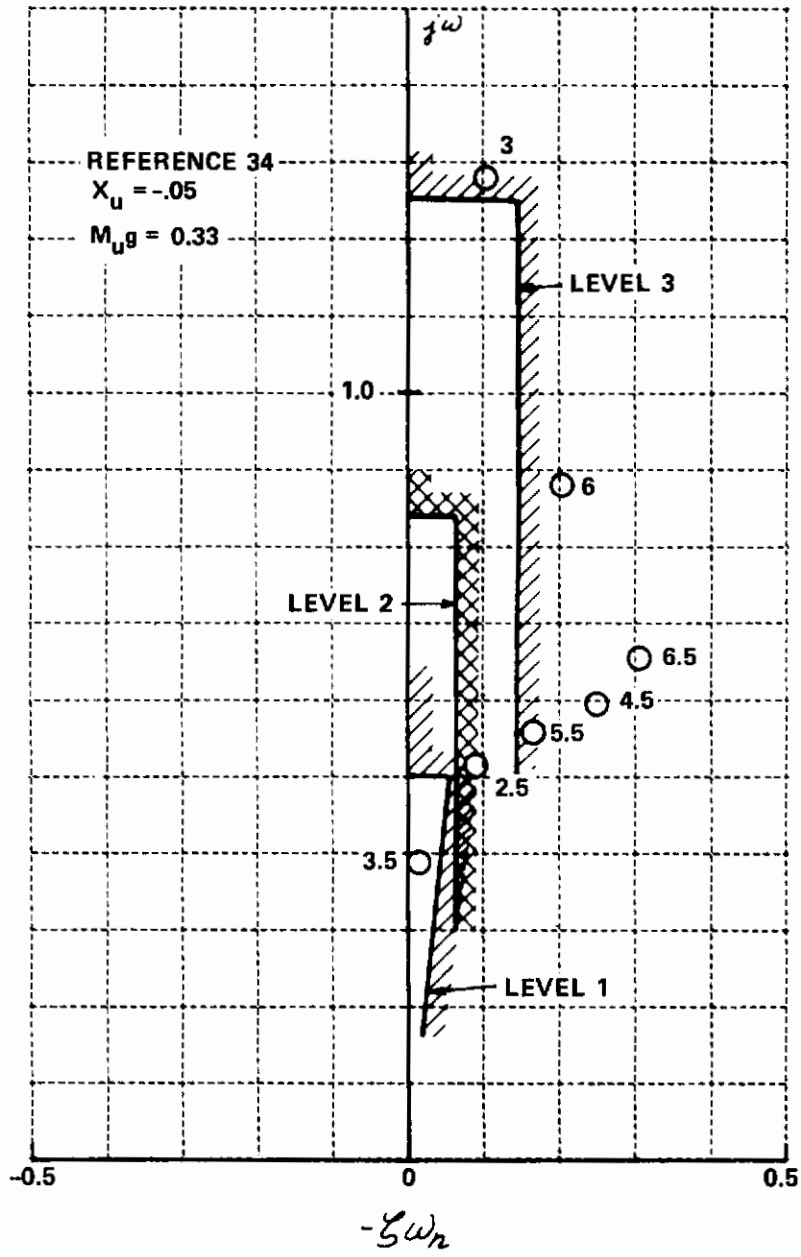

Figure 14 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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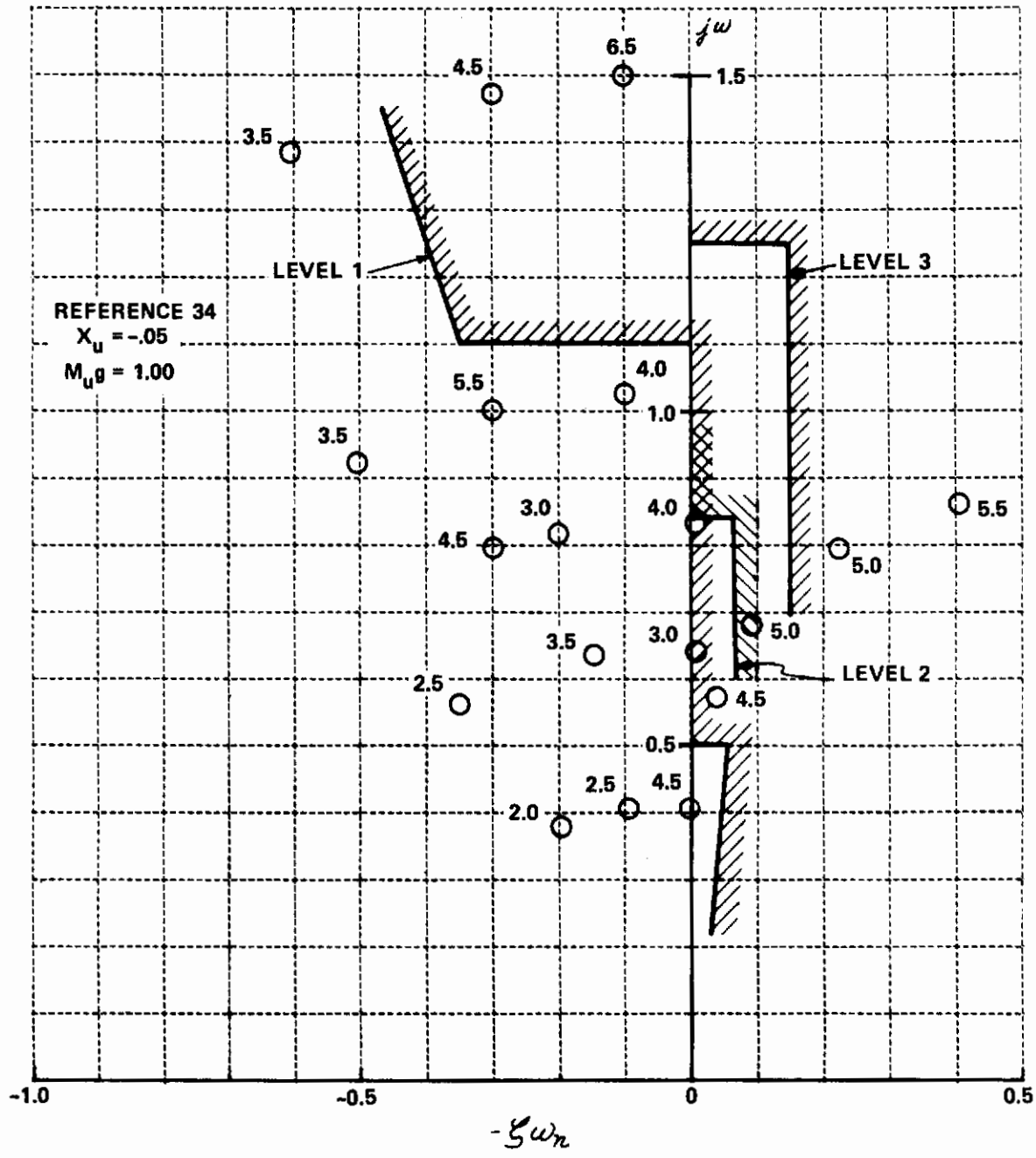

Figure 15 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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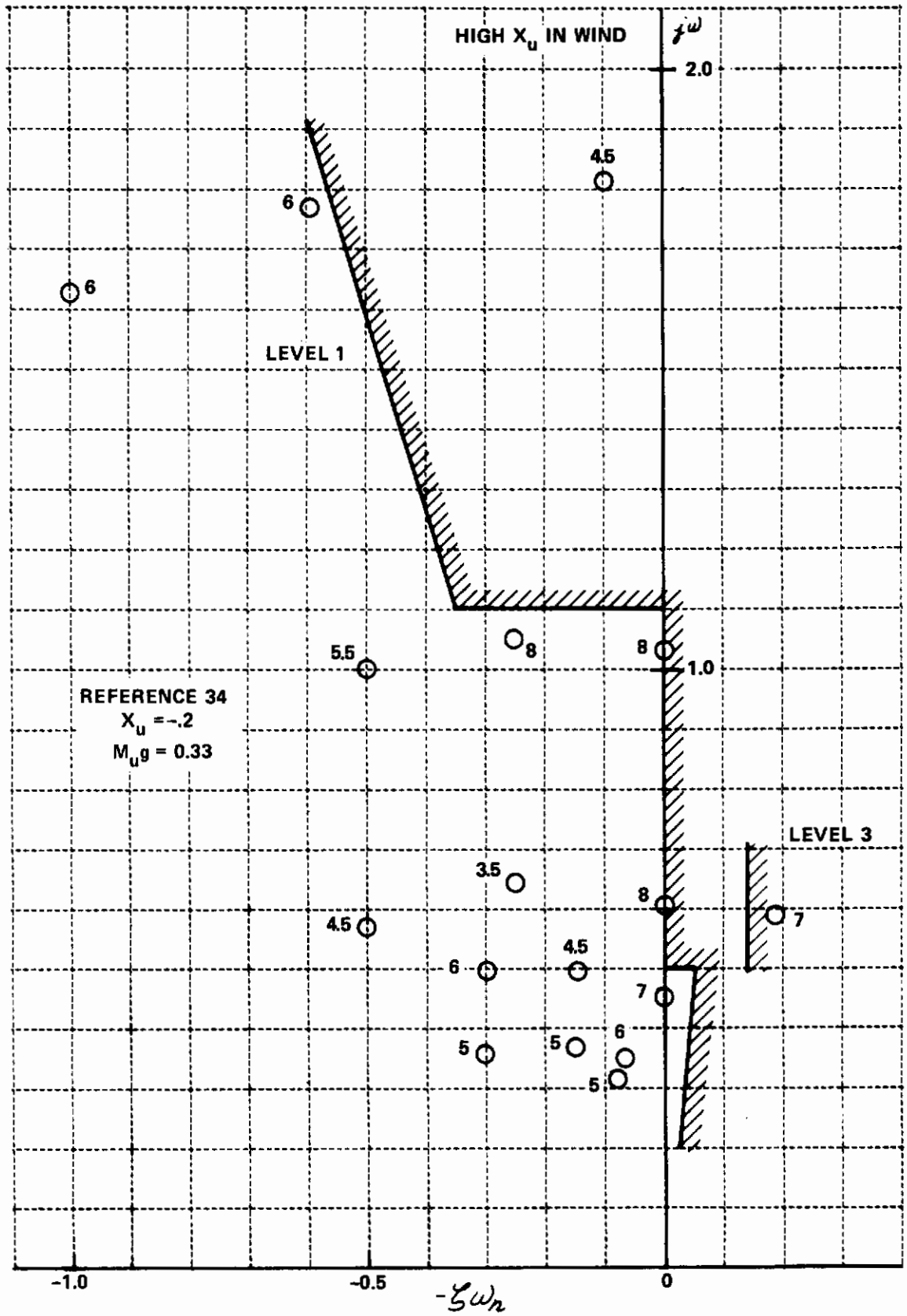

Figure 16 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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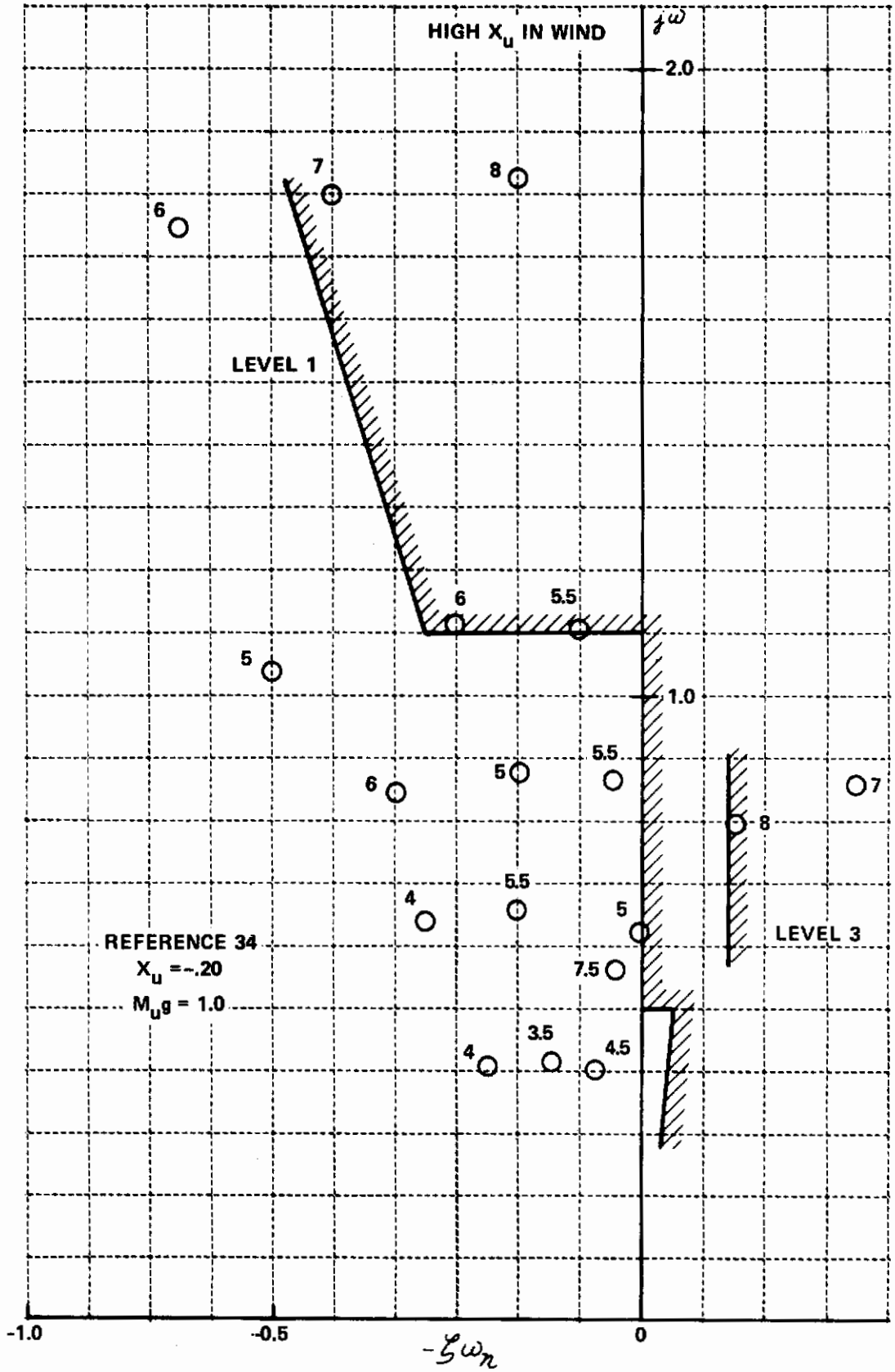

Figure 17 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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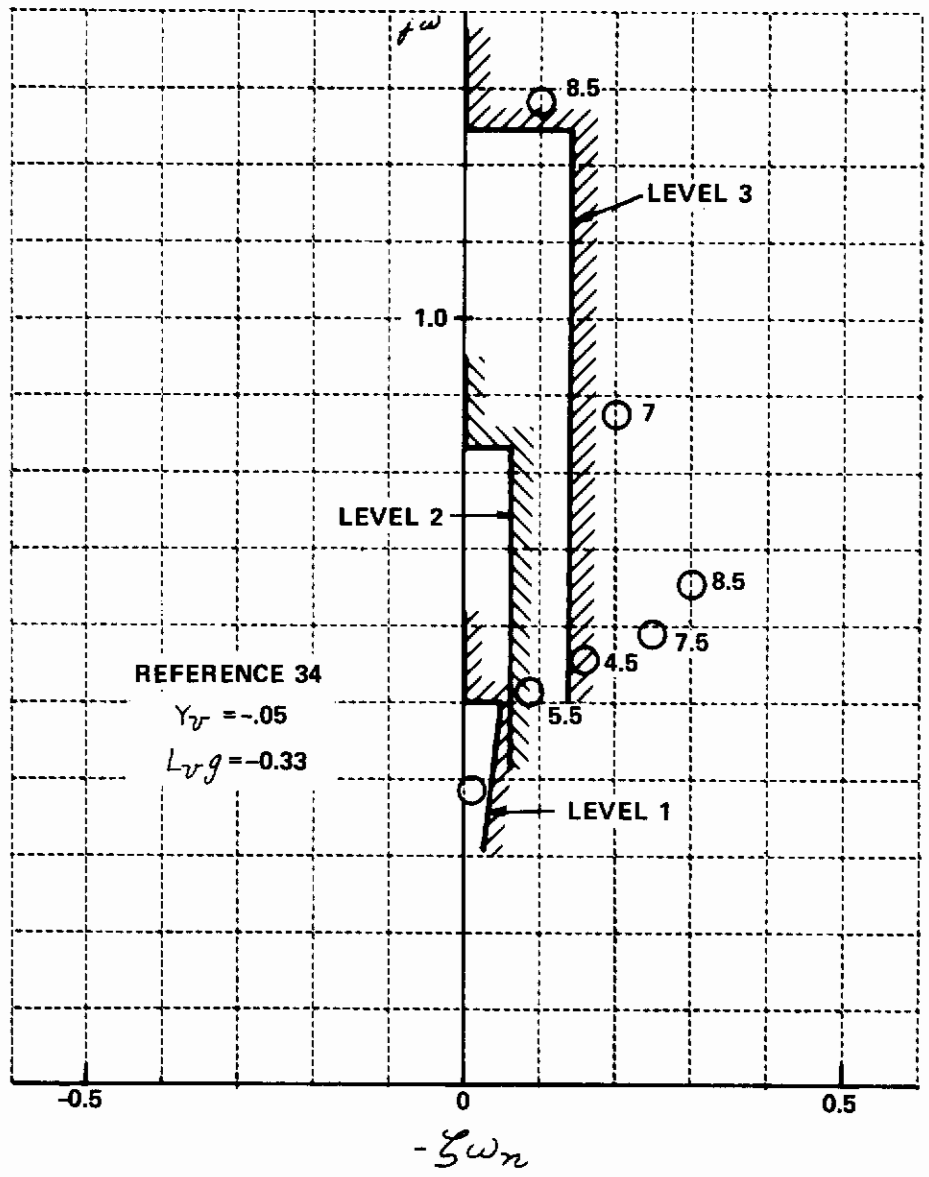

Figure 18 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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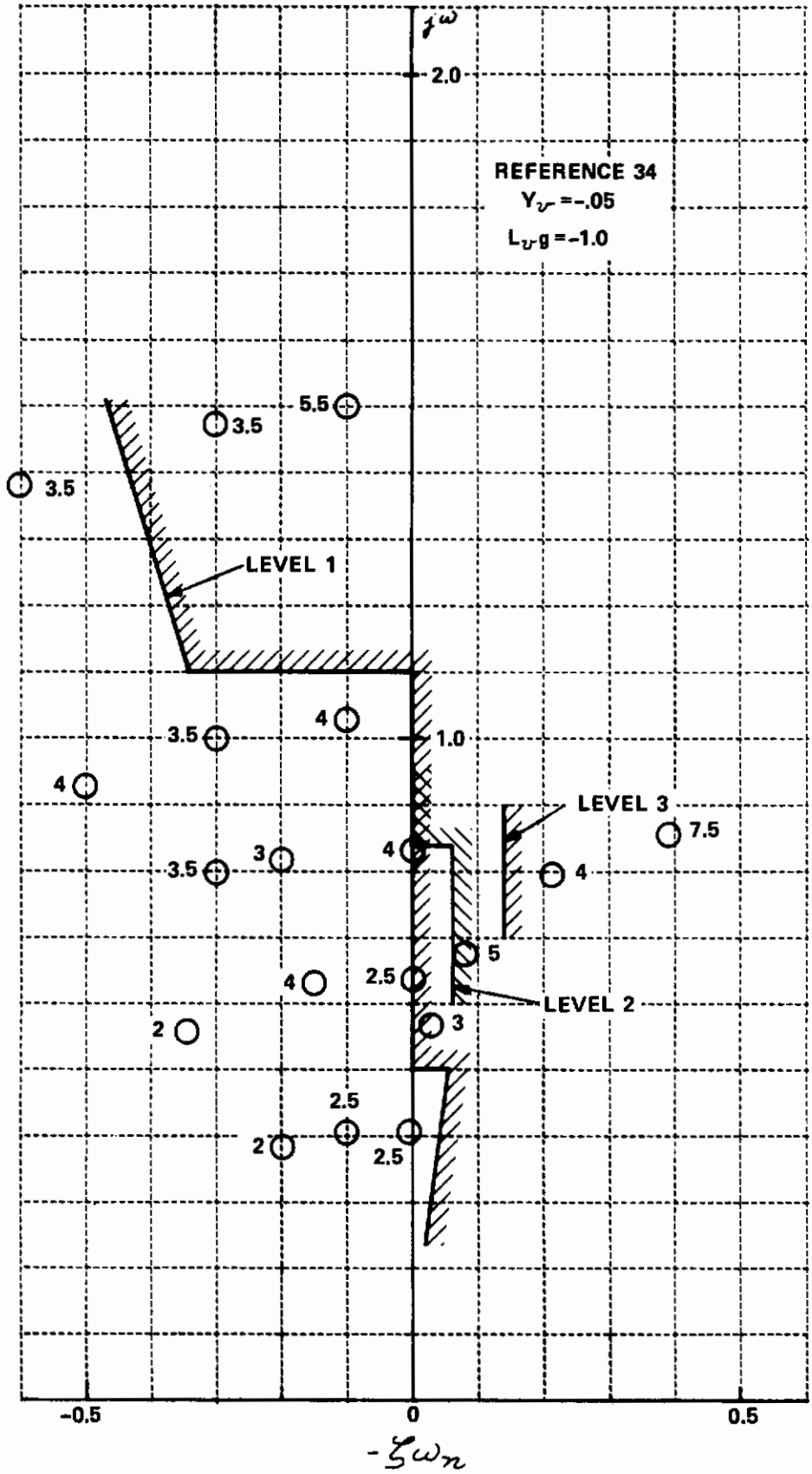

Figure 19 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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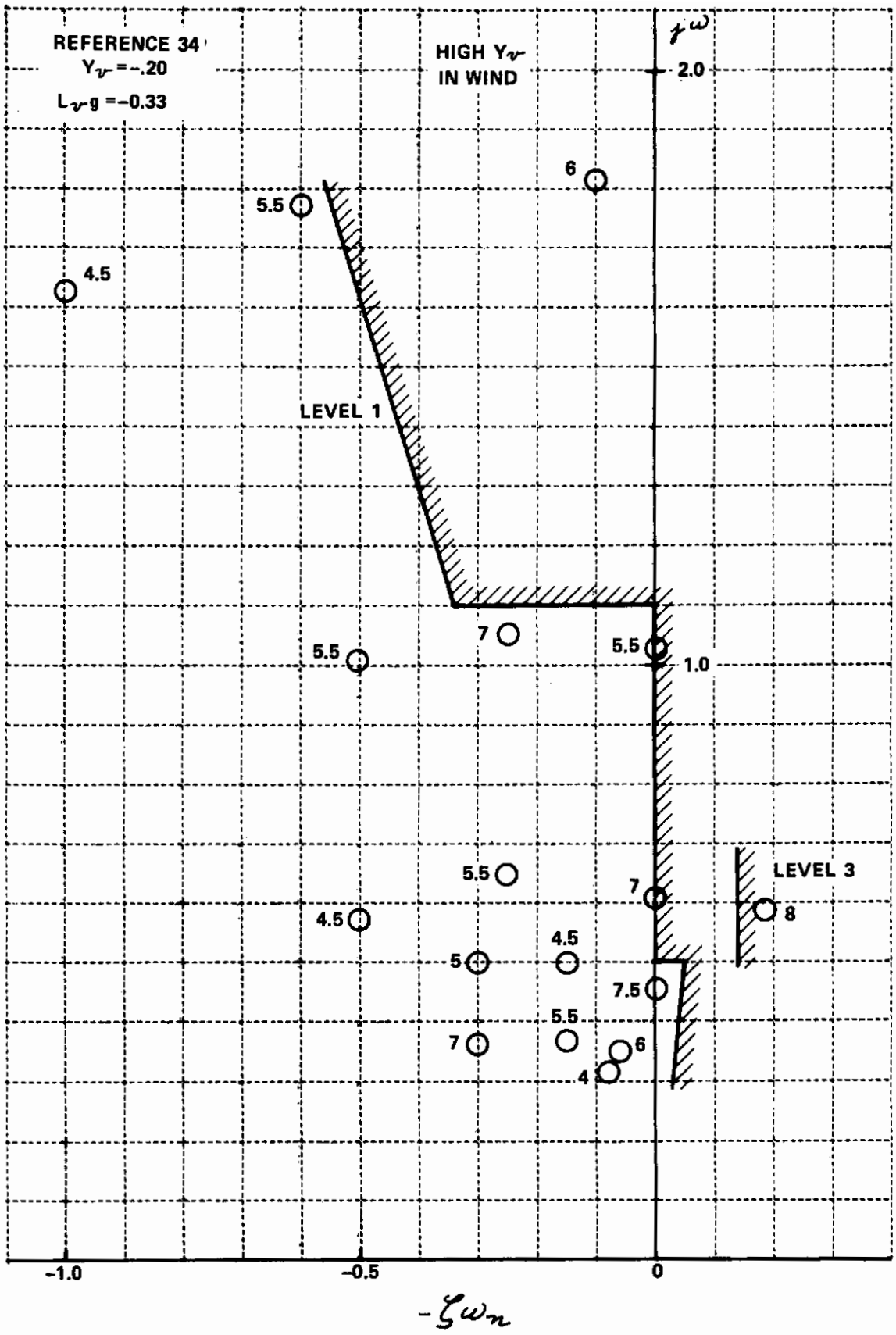

Figure 20 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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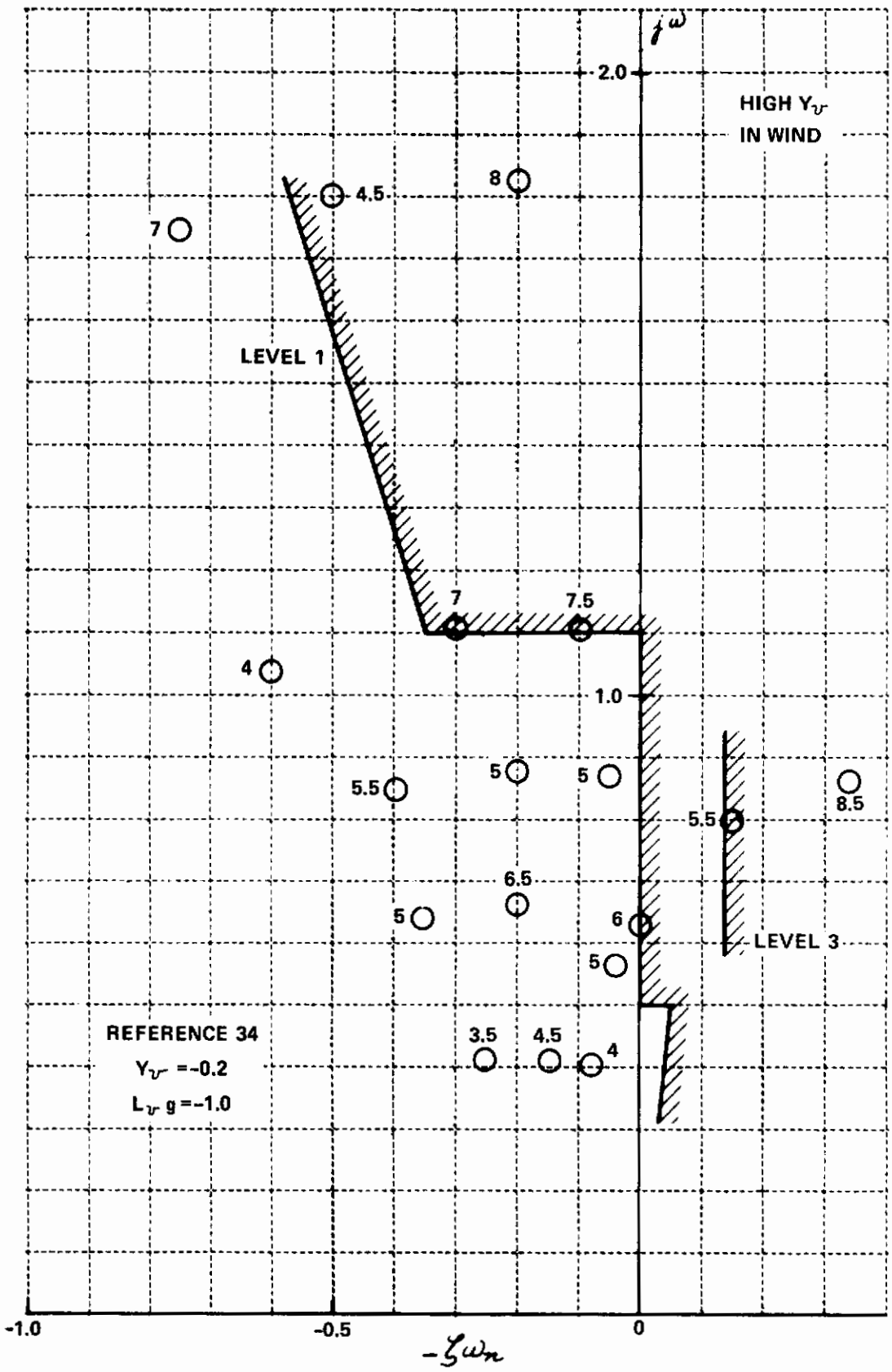

Figure 21 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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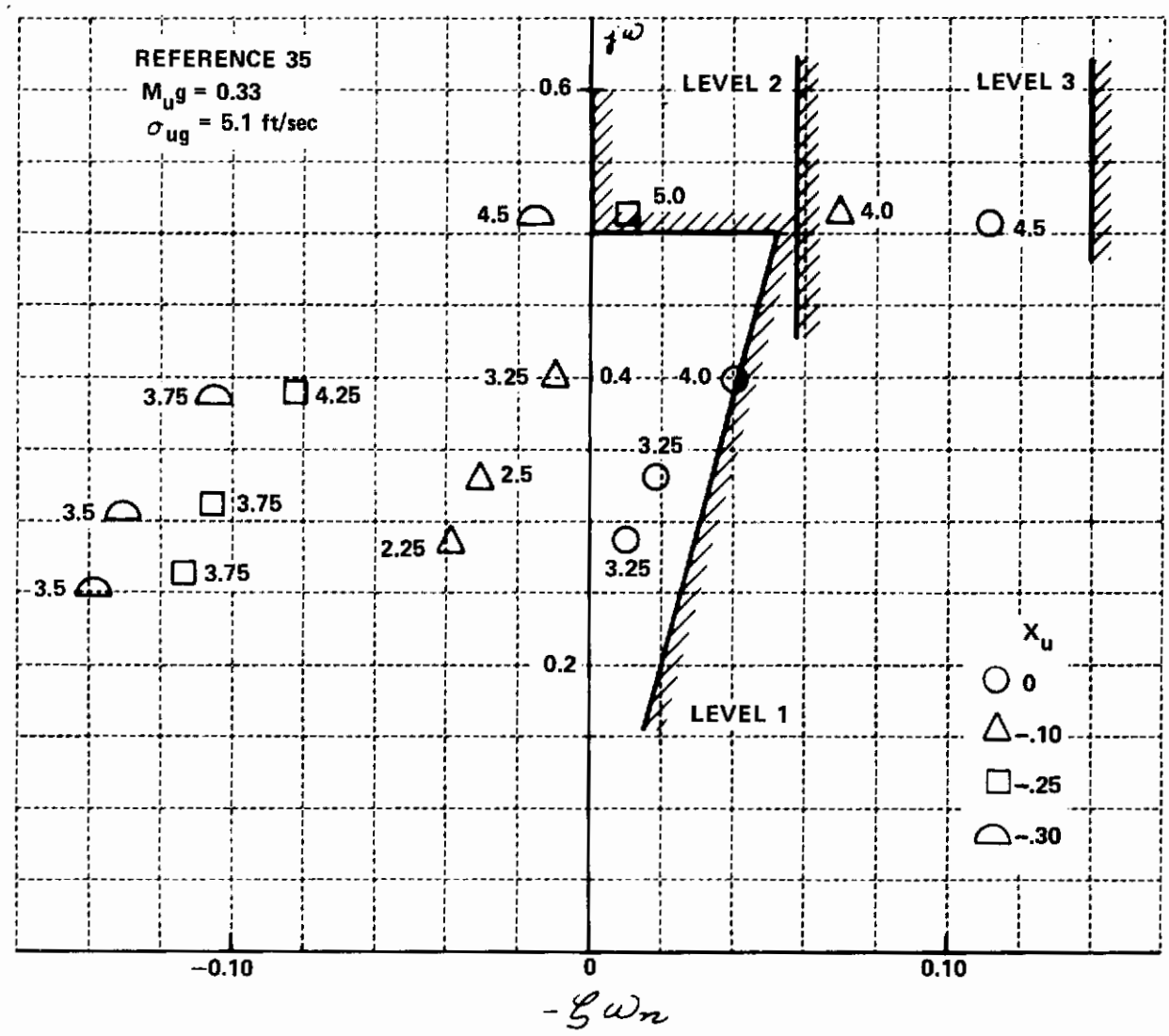

Figure 22 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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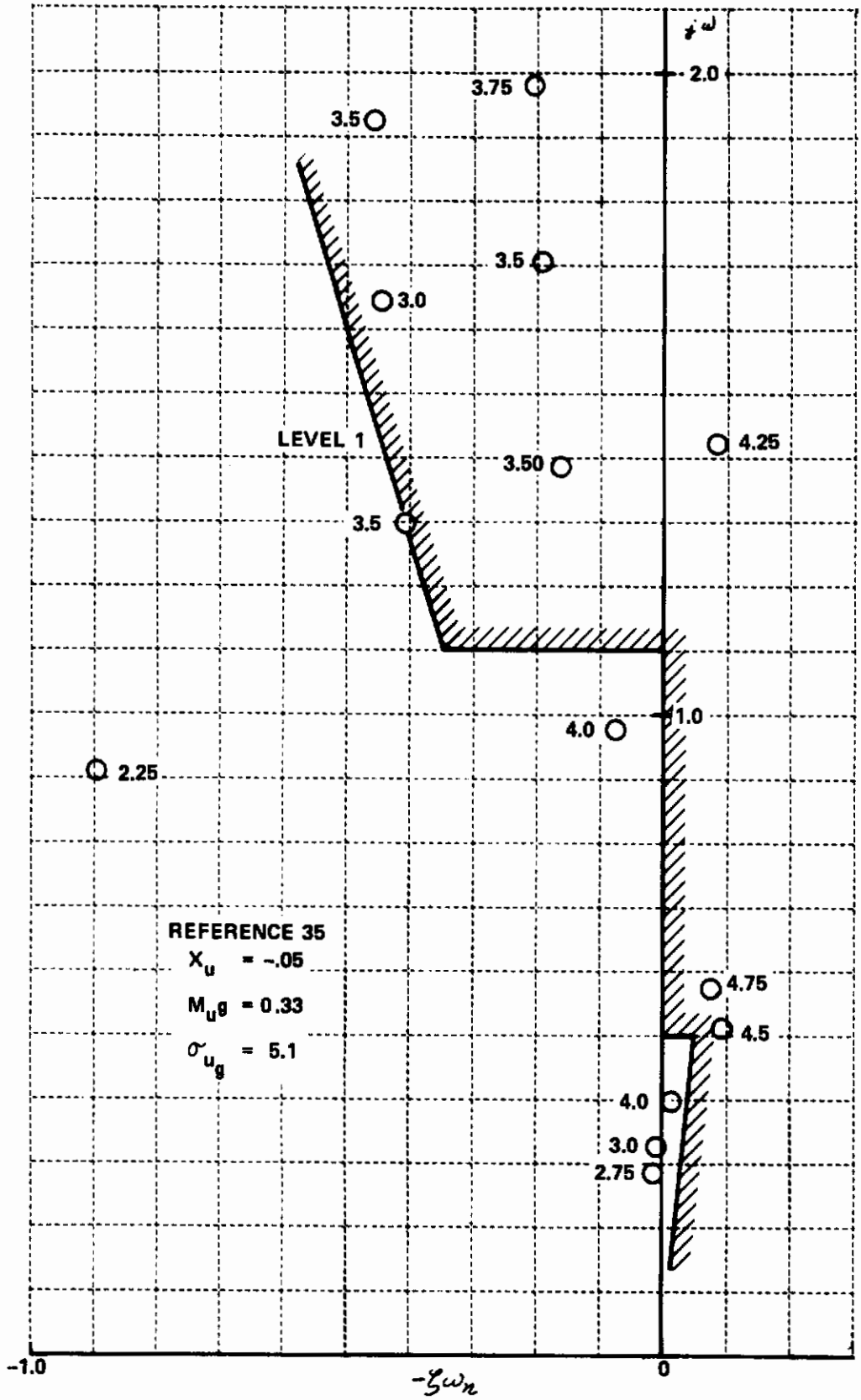

Figure 23 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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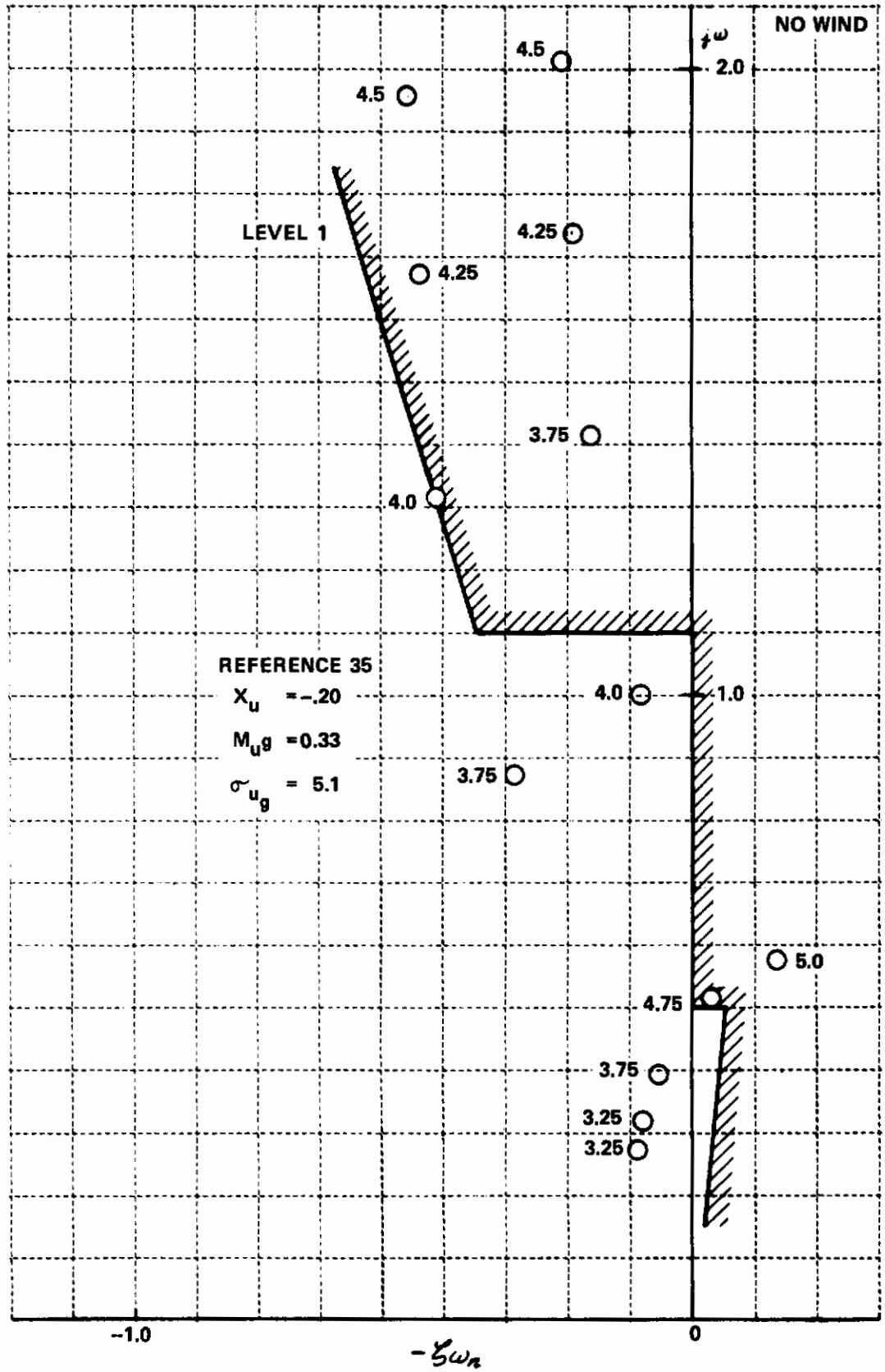

Figure 24 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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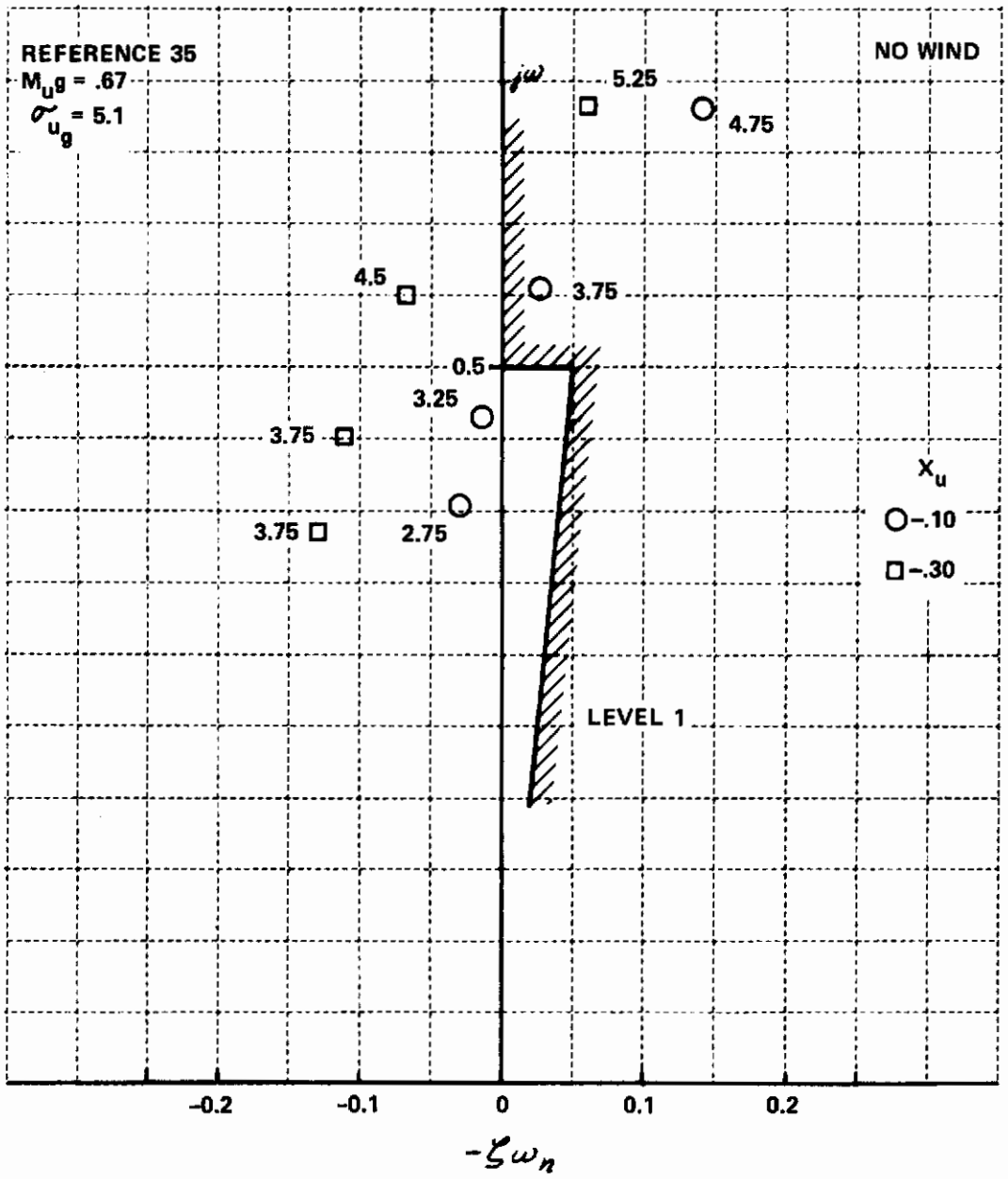

Figure 25 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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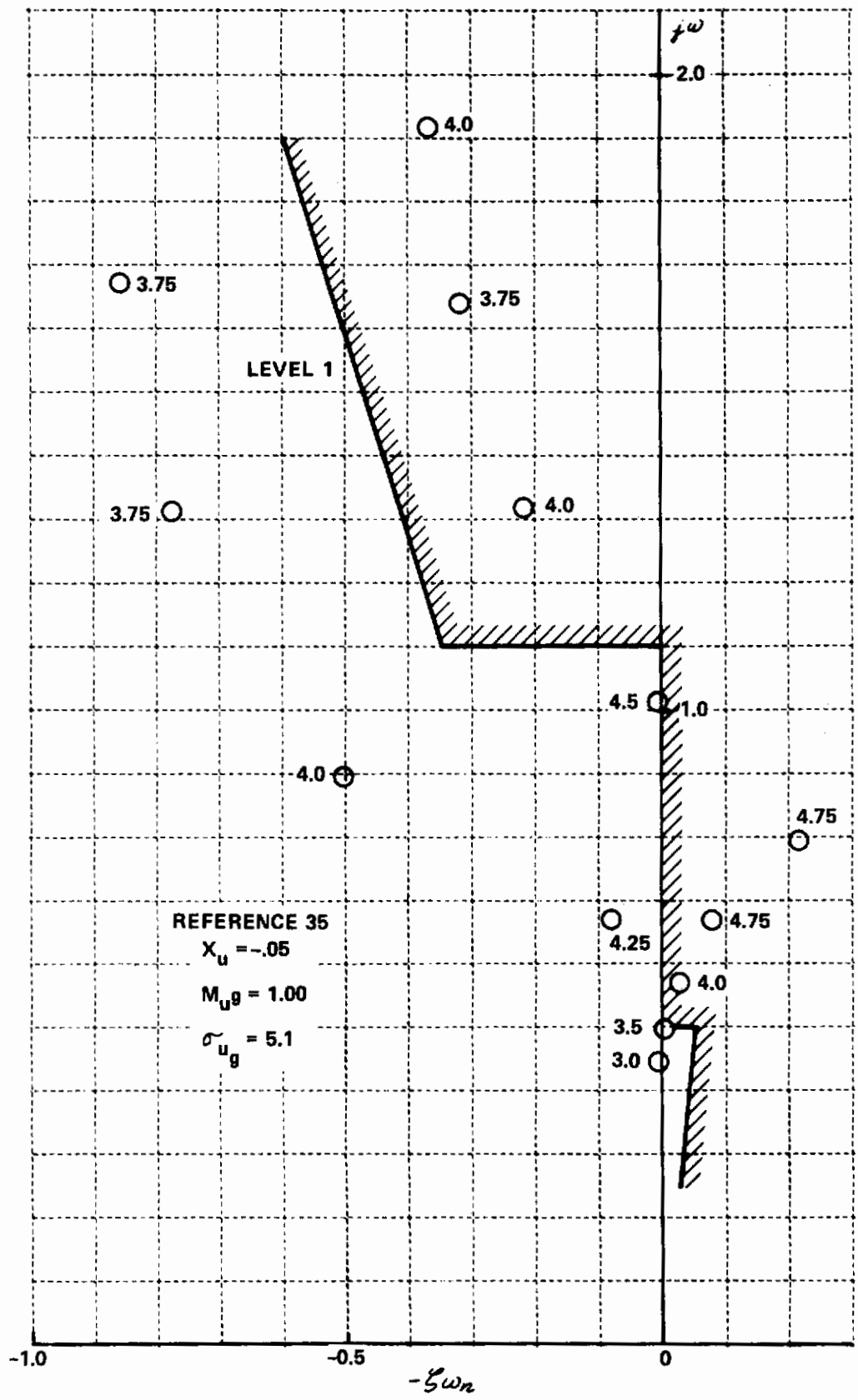

Figure 26 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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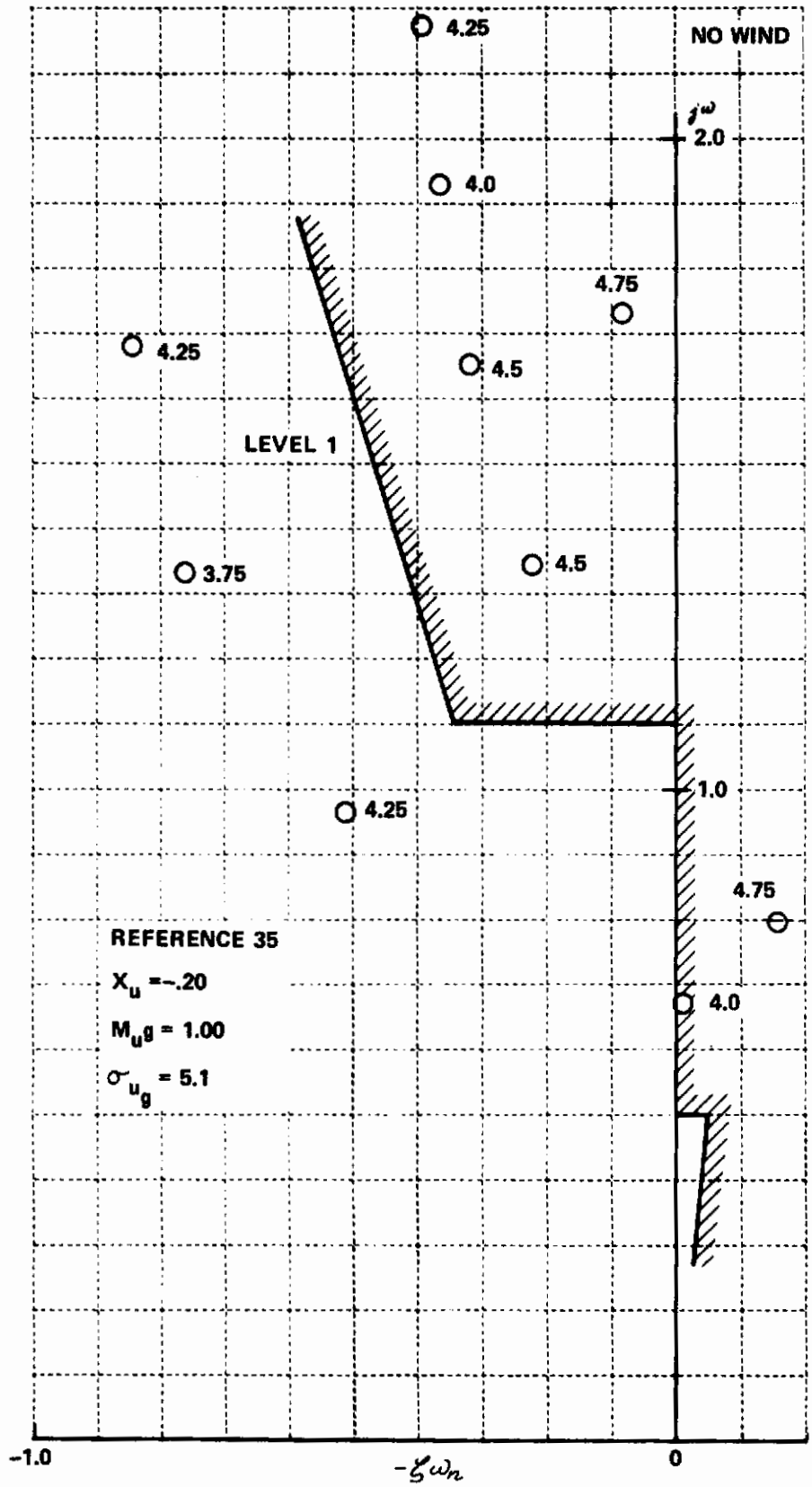

Figure 27 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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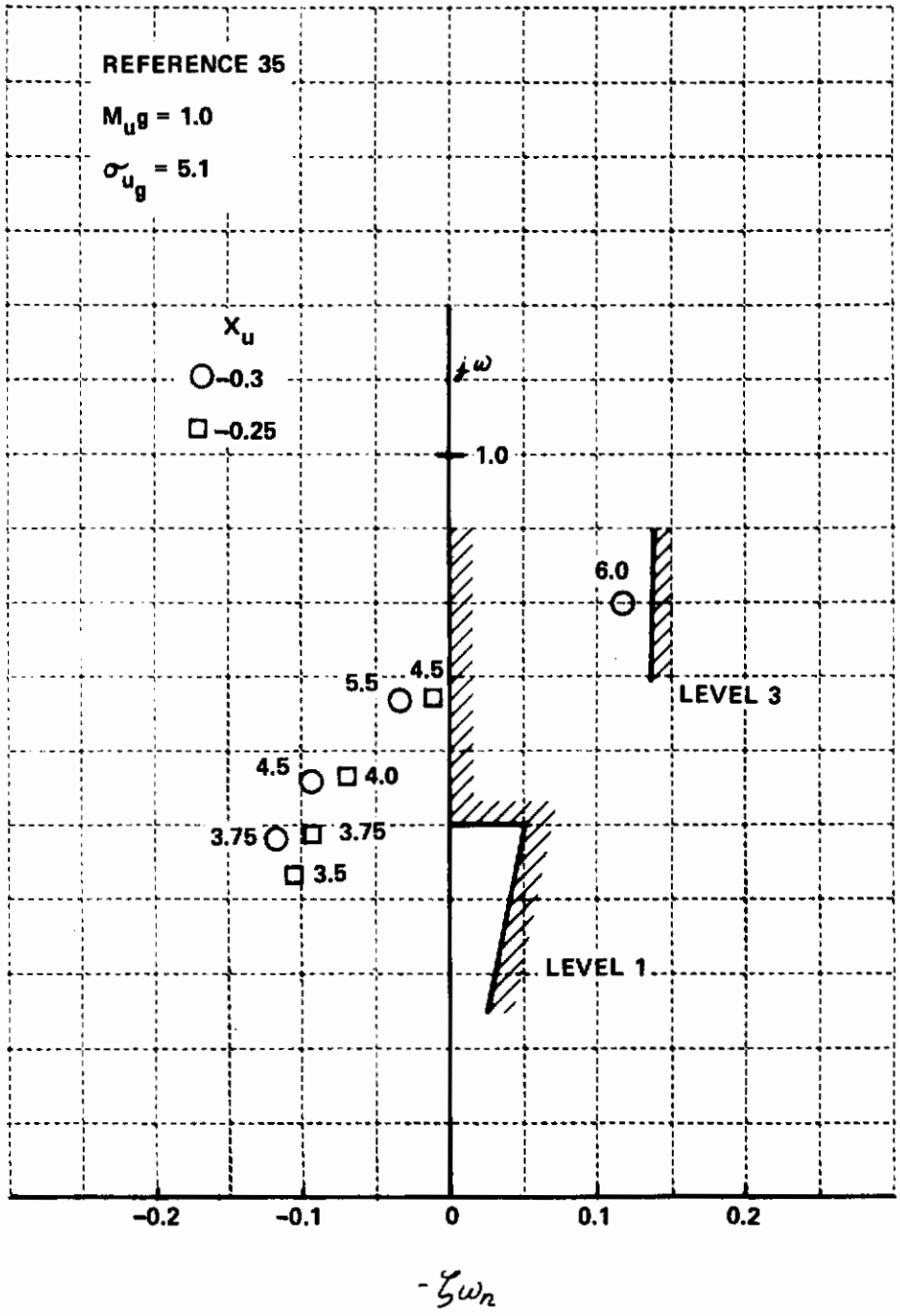

Figure 28 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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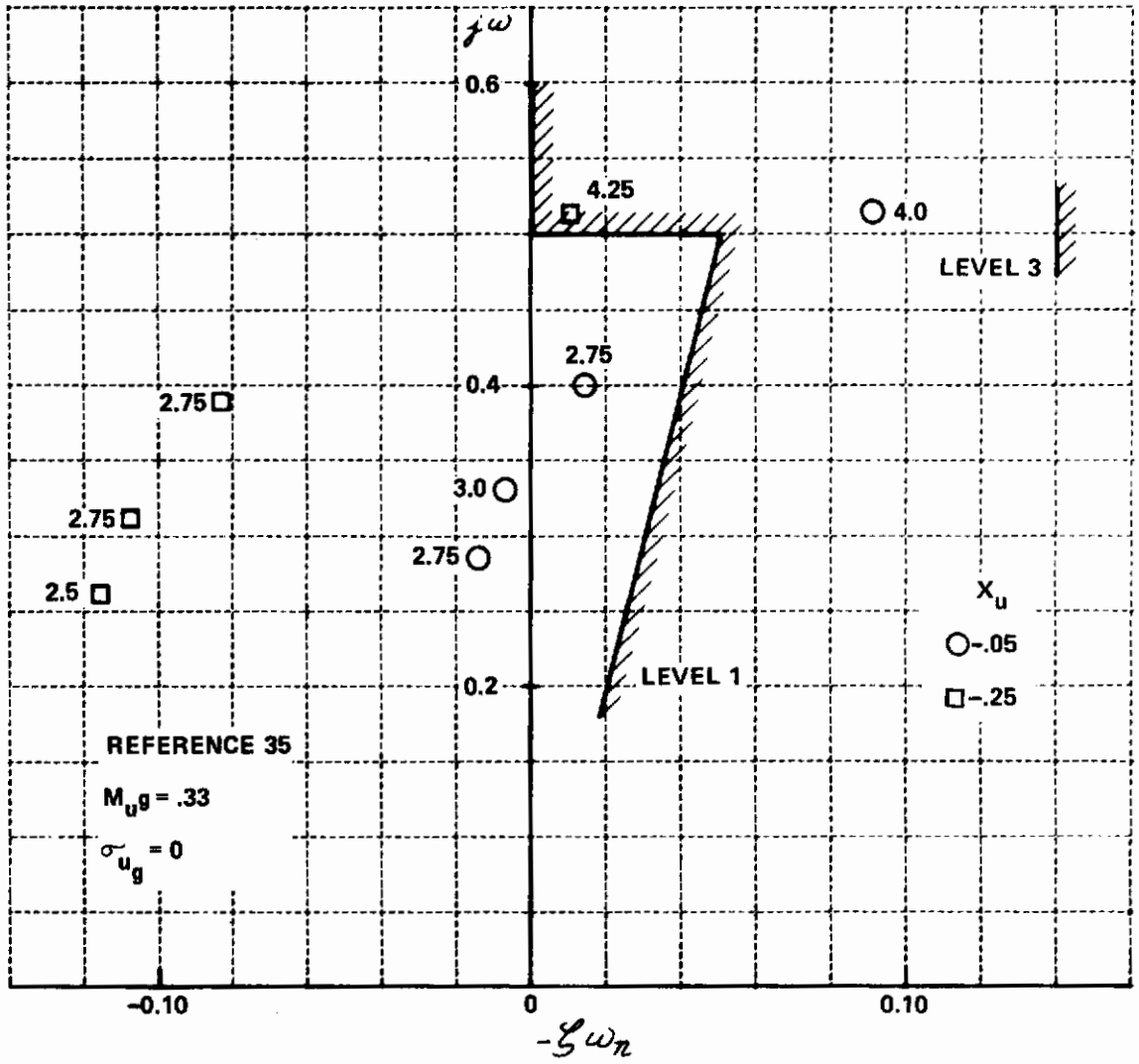

Figure 29 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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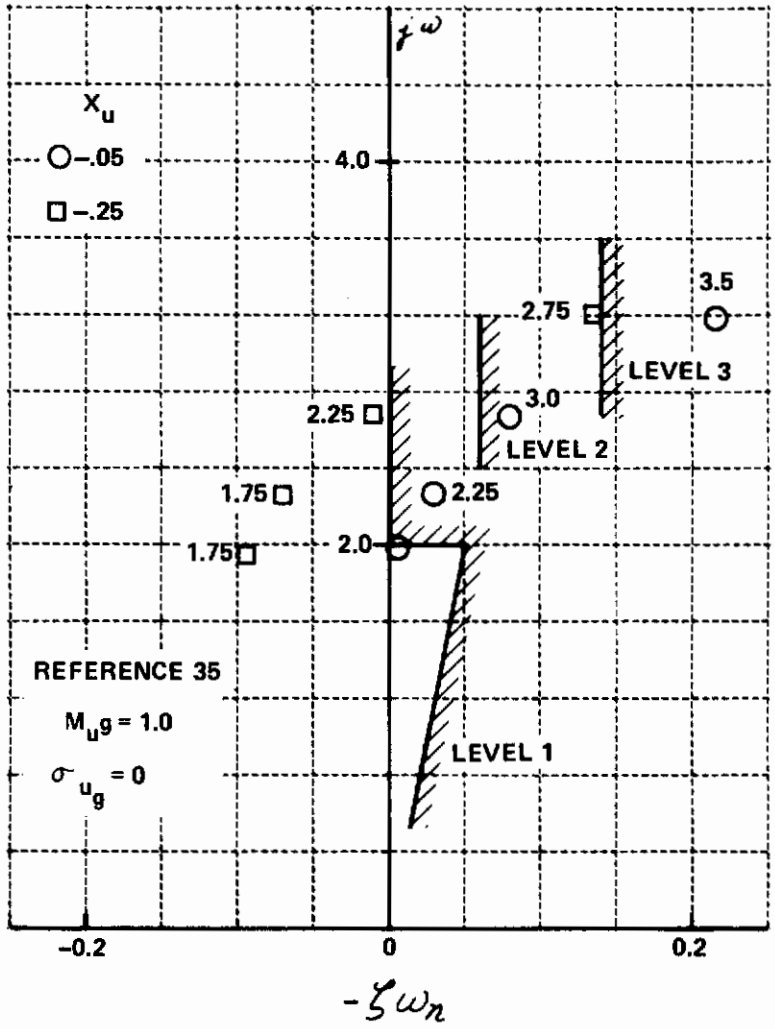

Figure 30 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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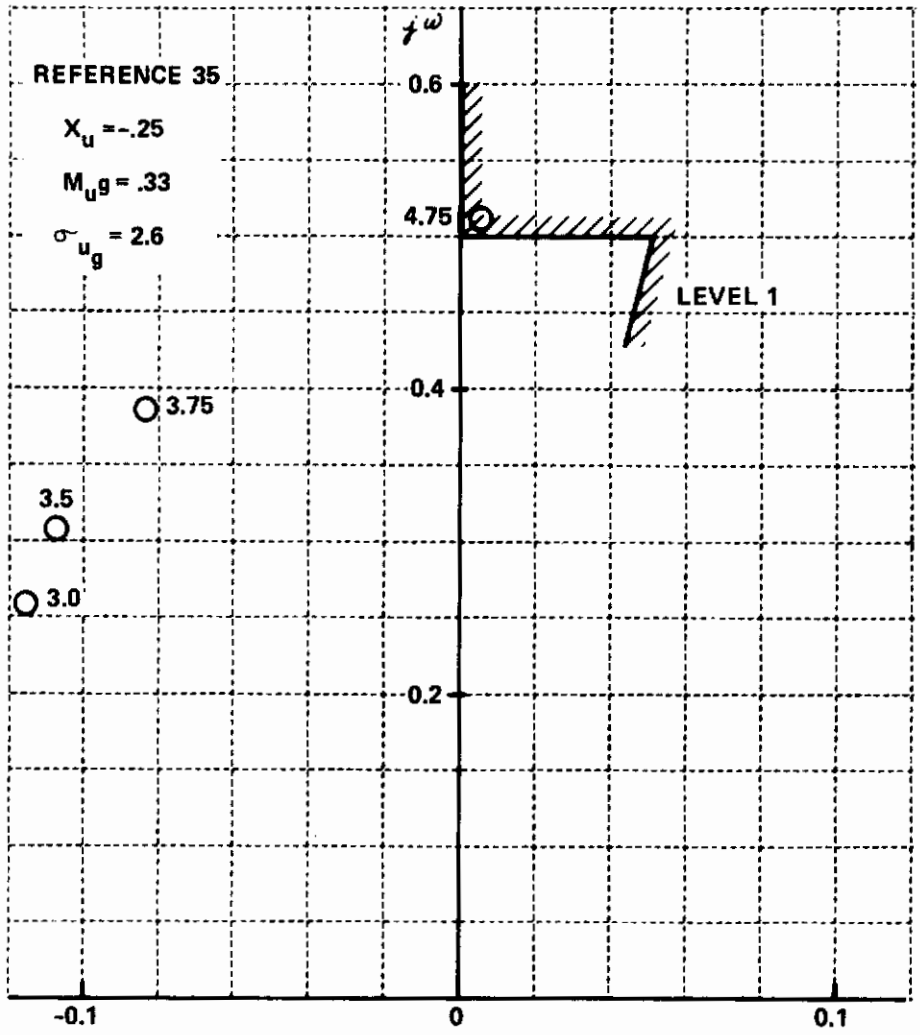

Figure 31 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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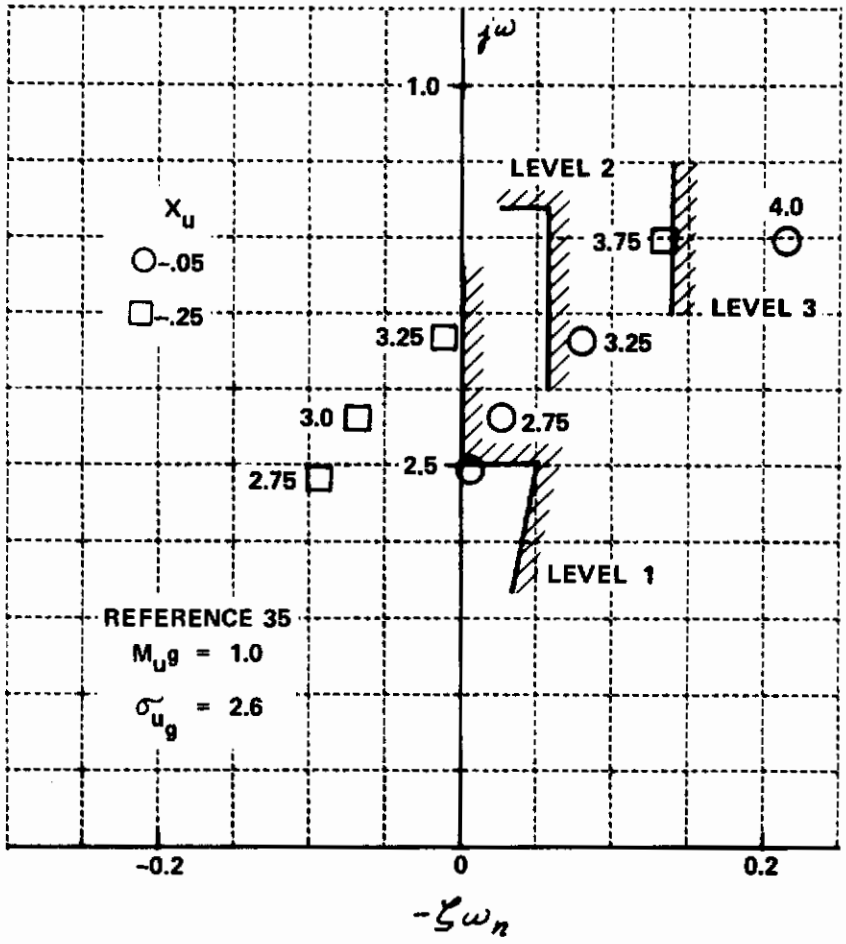

Figure 32 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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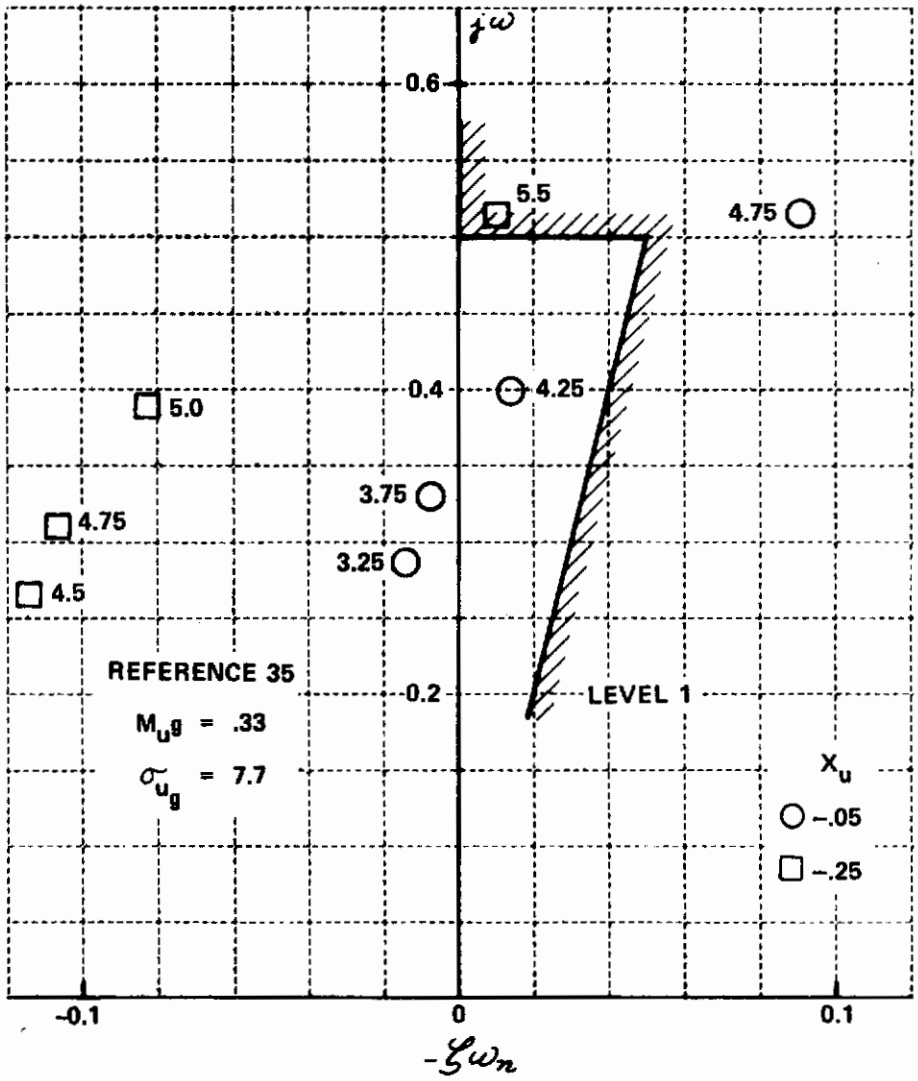

Figure 33 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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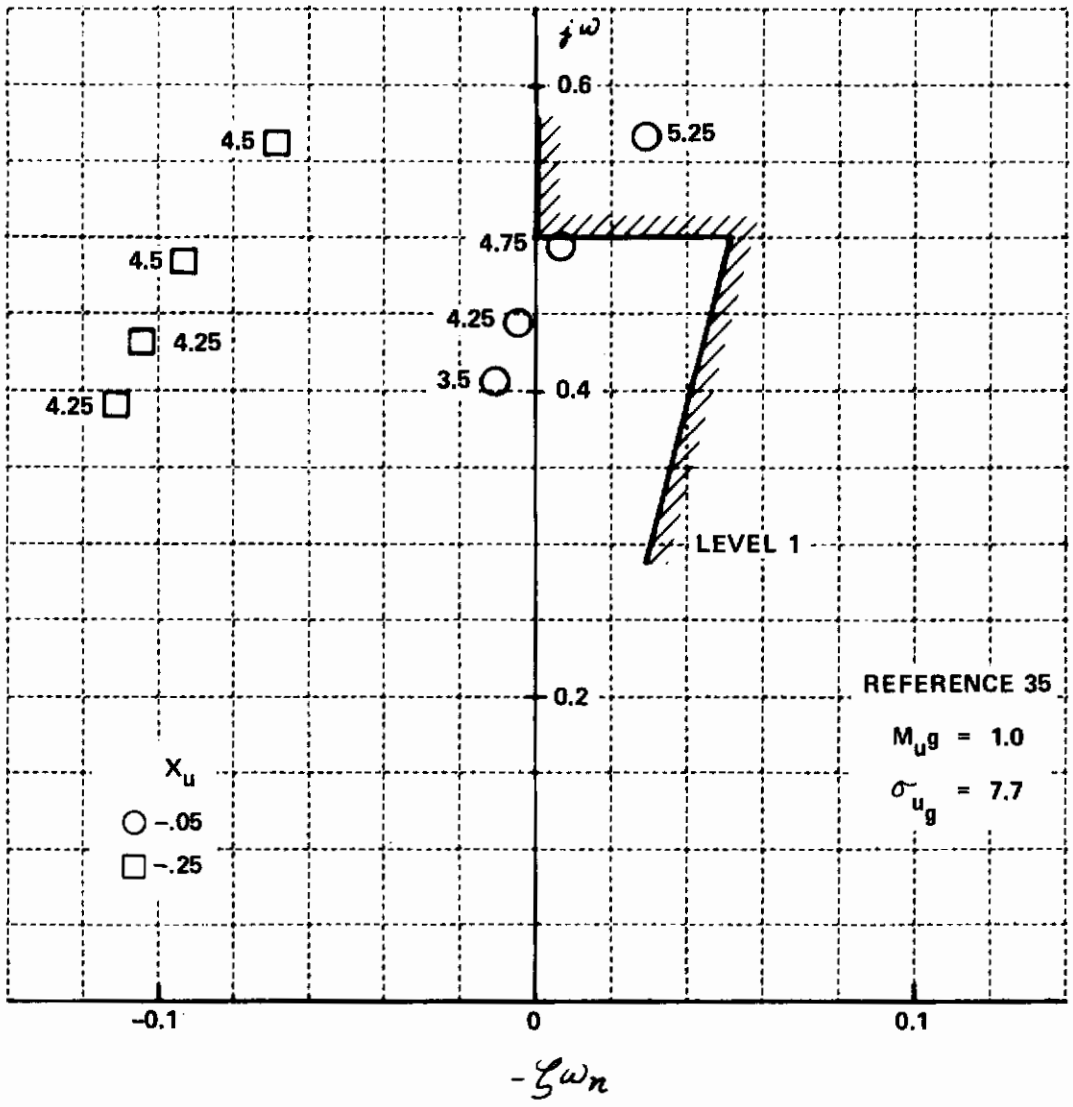

Figure 34 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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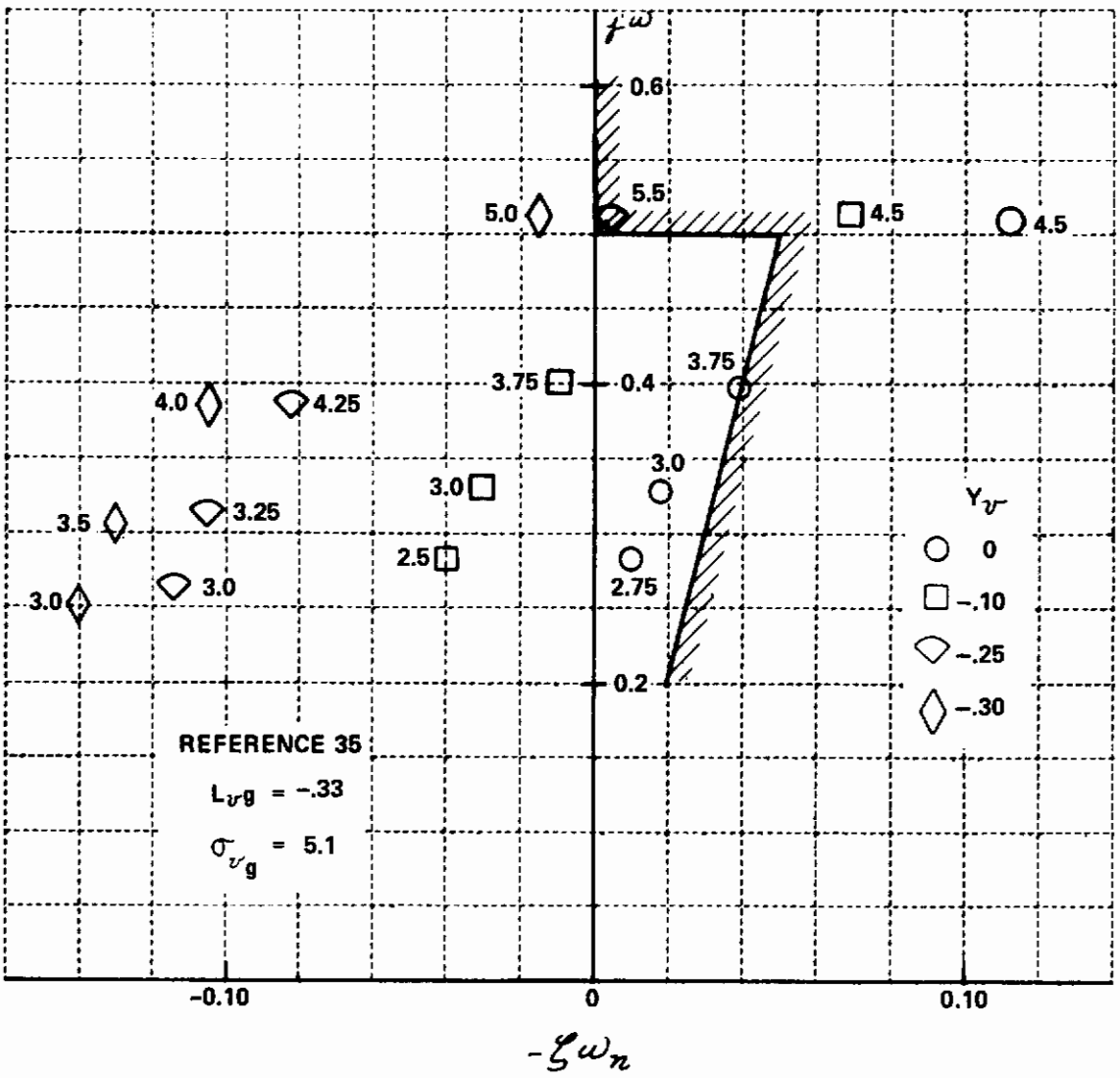

Figure 35 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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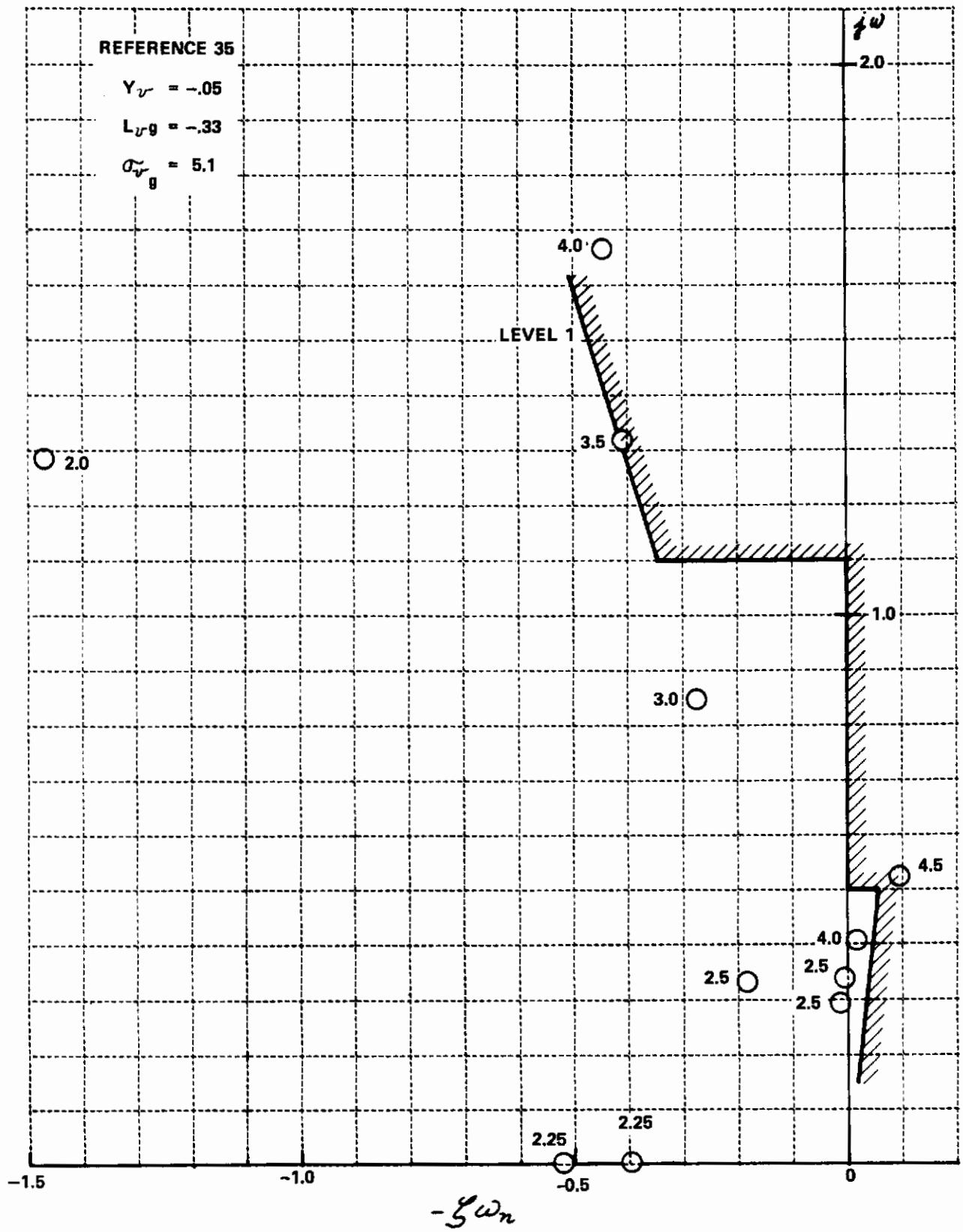

Figure 36 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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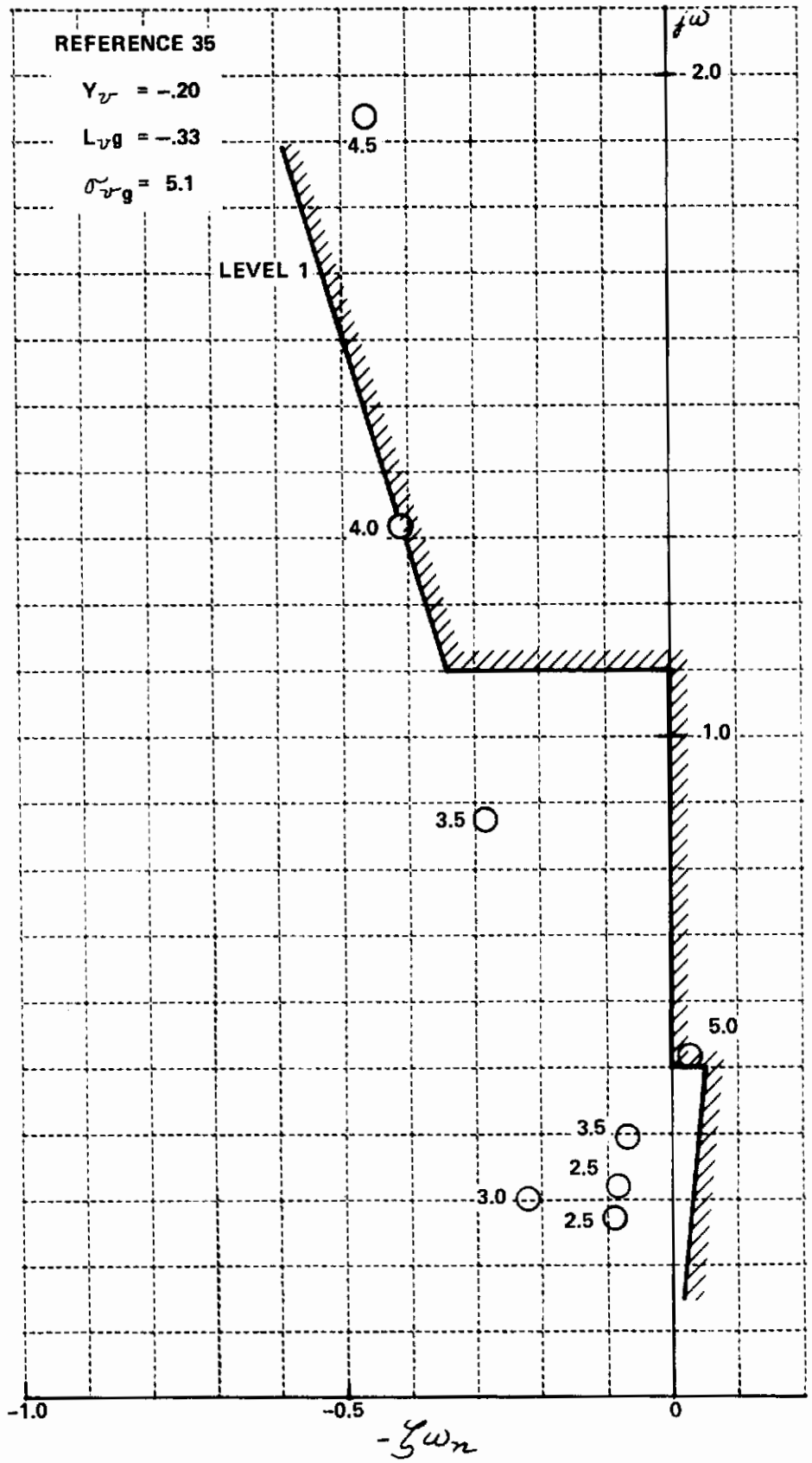

Figure 37 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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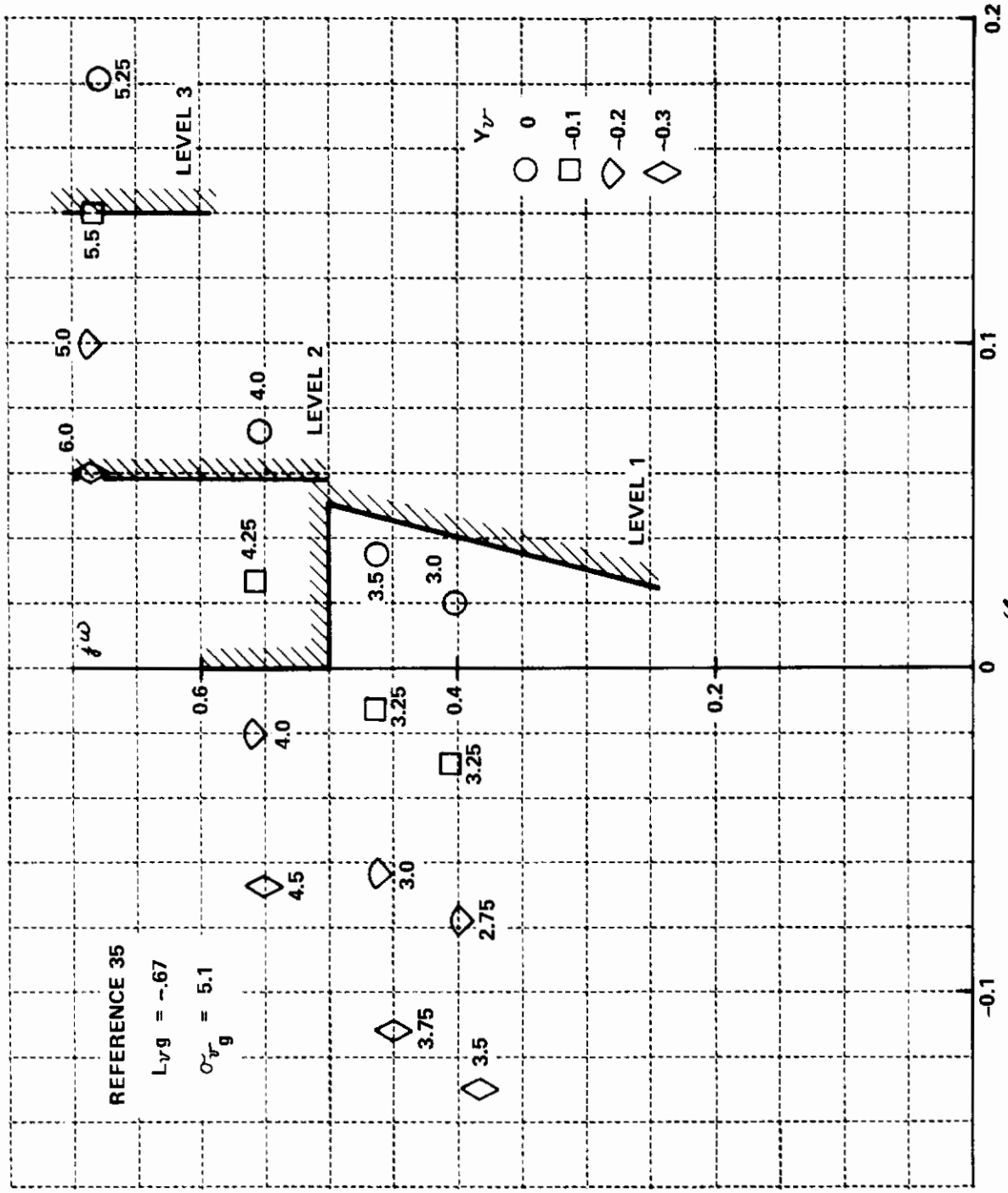

Figure 38 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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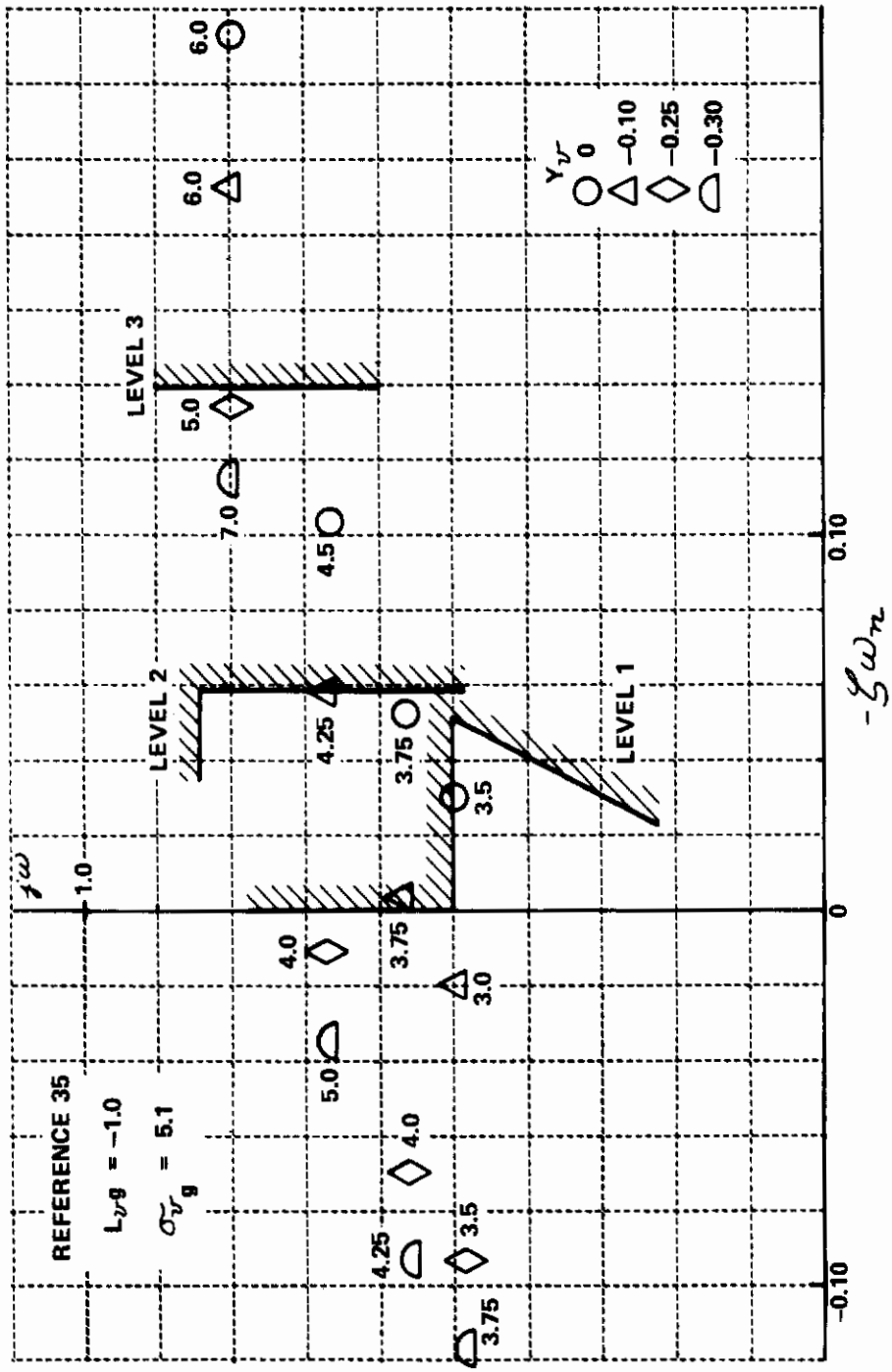

Figure 39 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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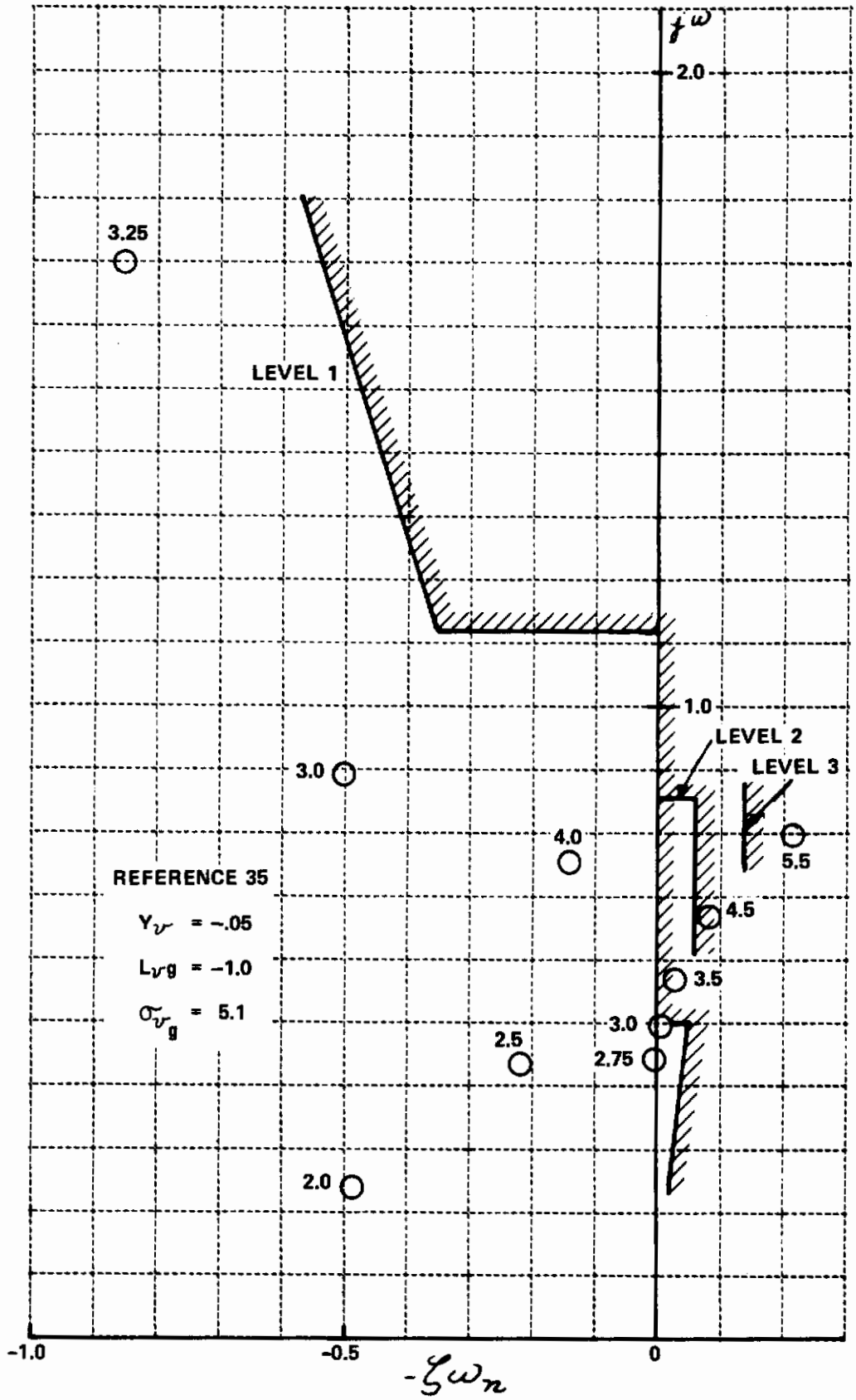

Figure 40 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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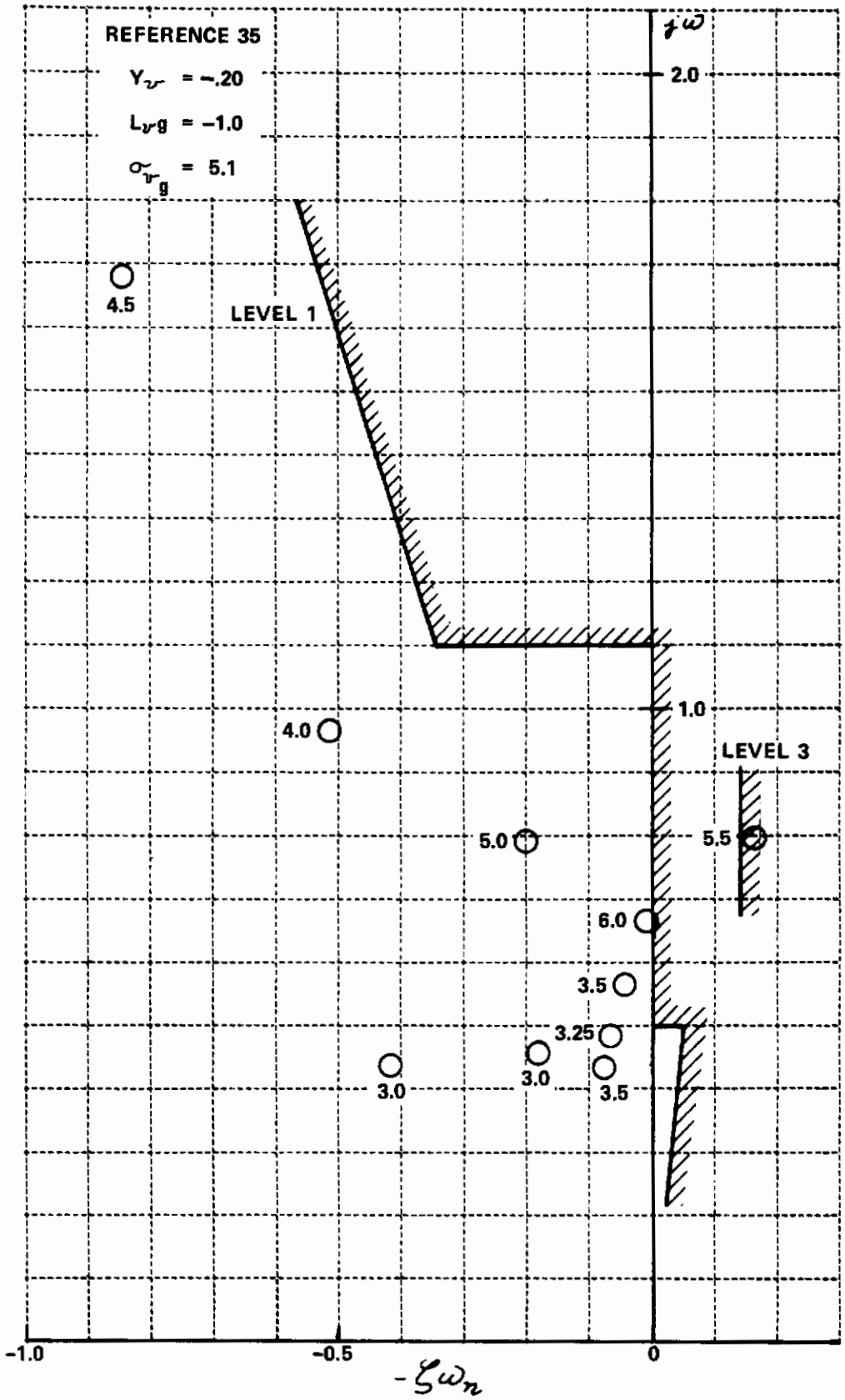

Figure 41 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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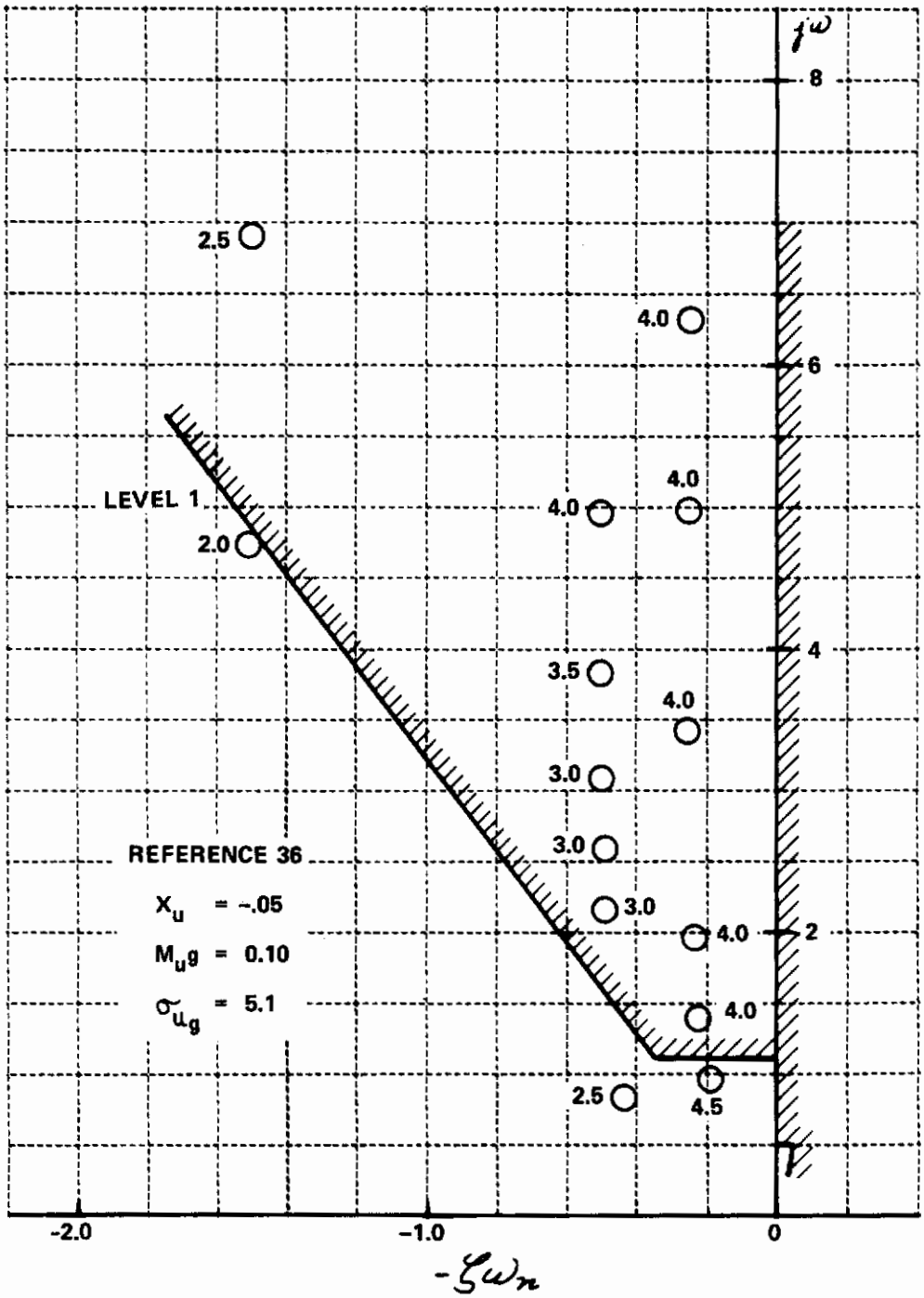

Figure 42 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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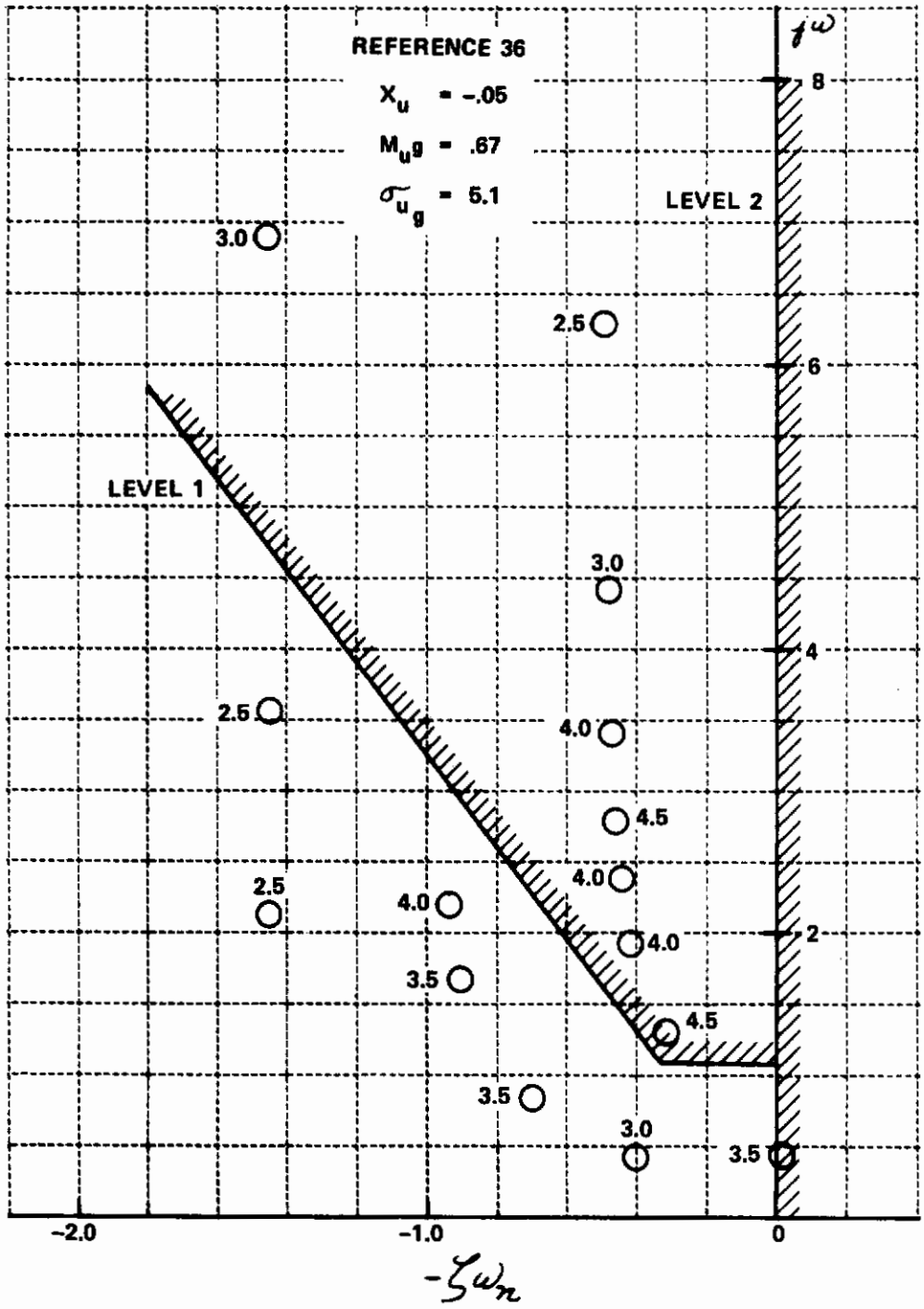

Figure 43 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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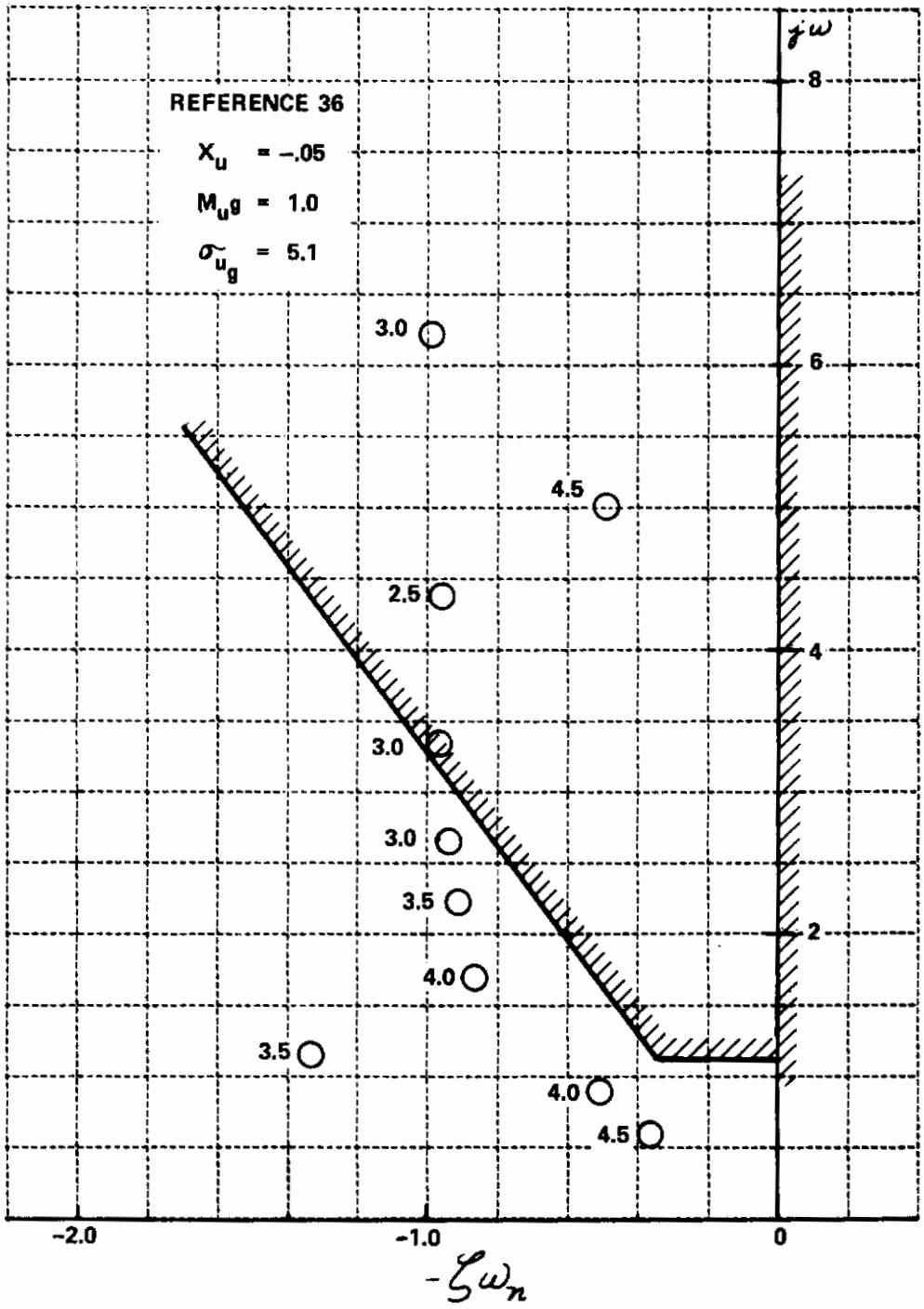

Figure 44 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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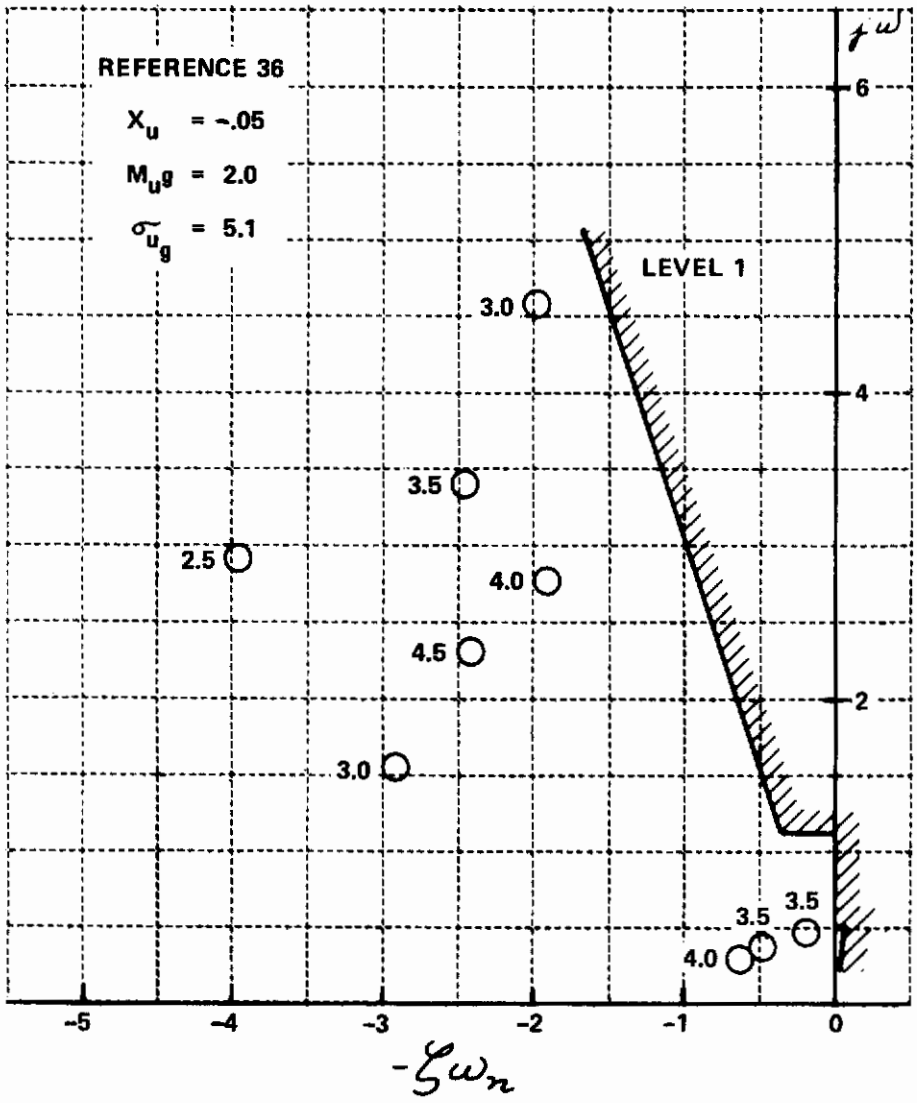

Figure 45 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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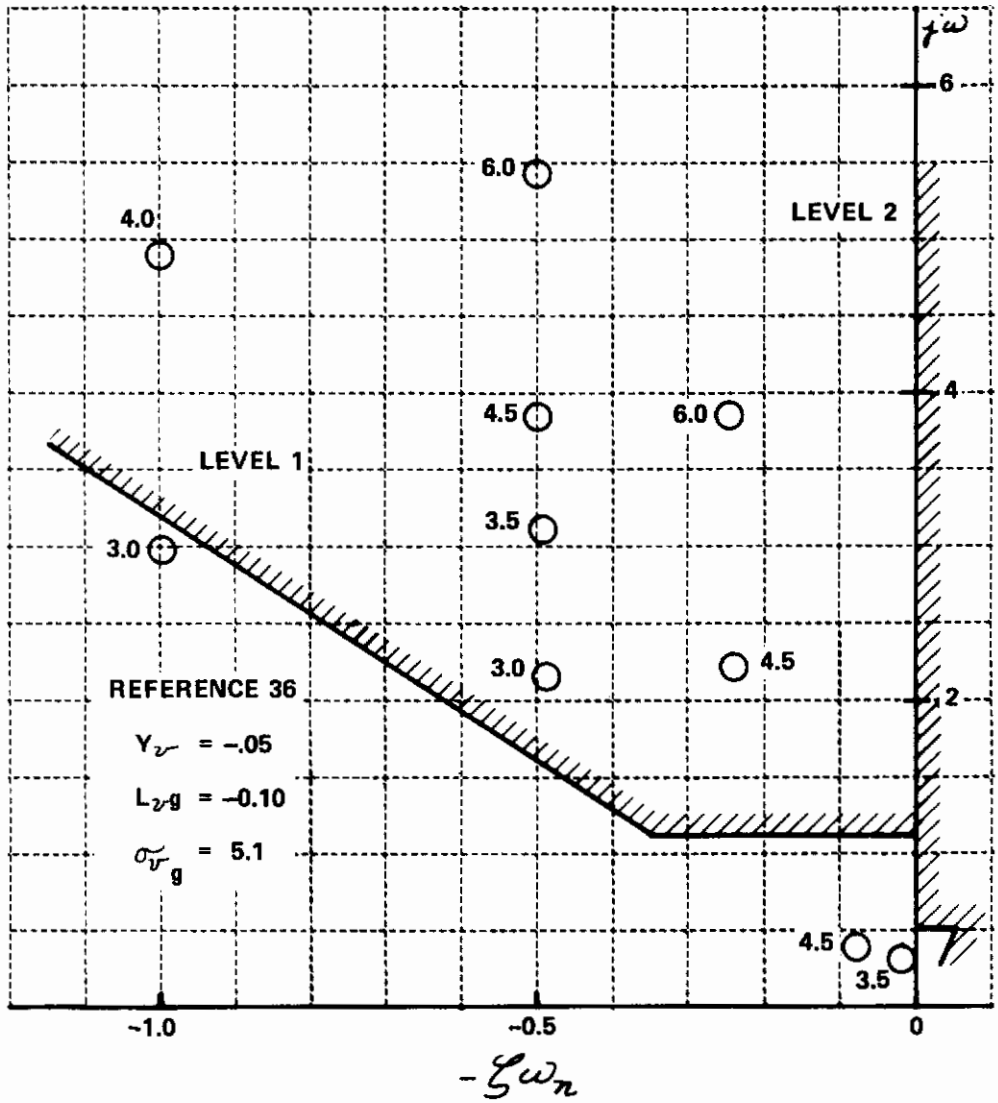

Figure 46 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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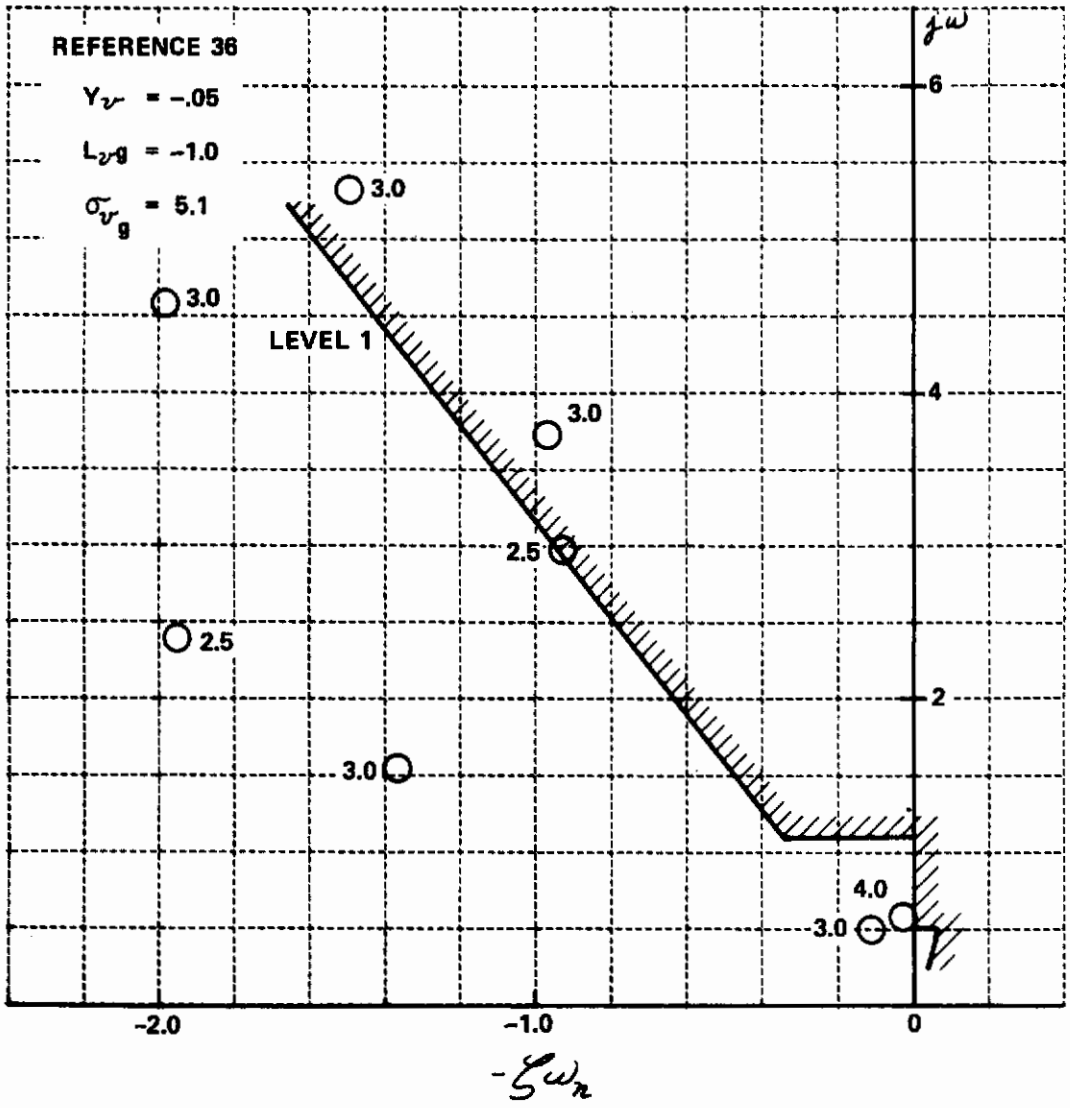

Figure 47 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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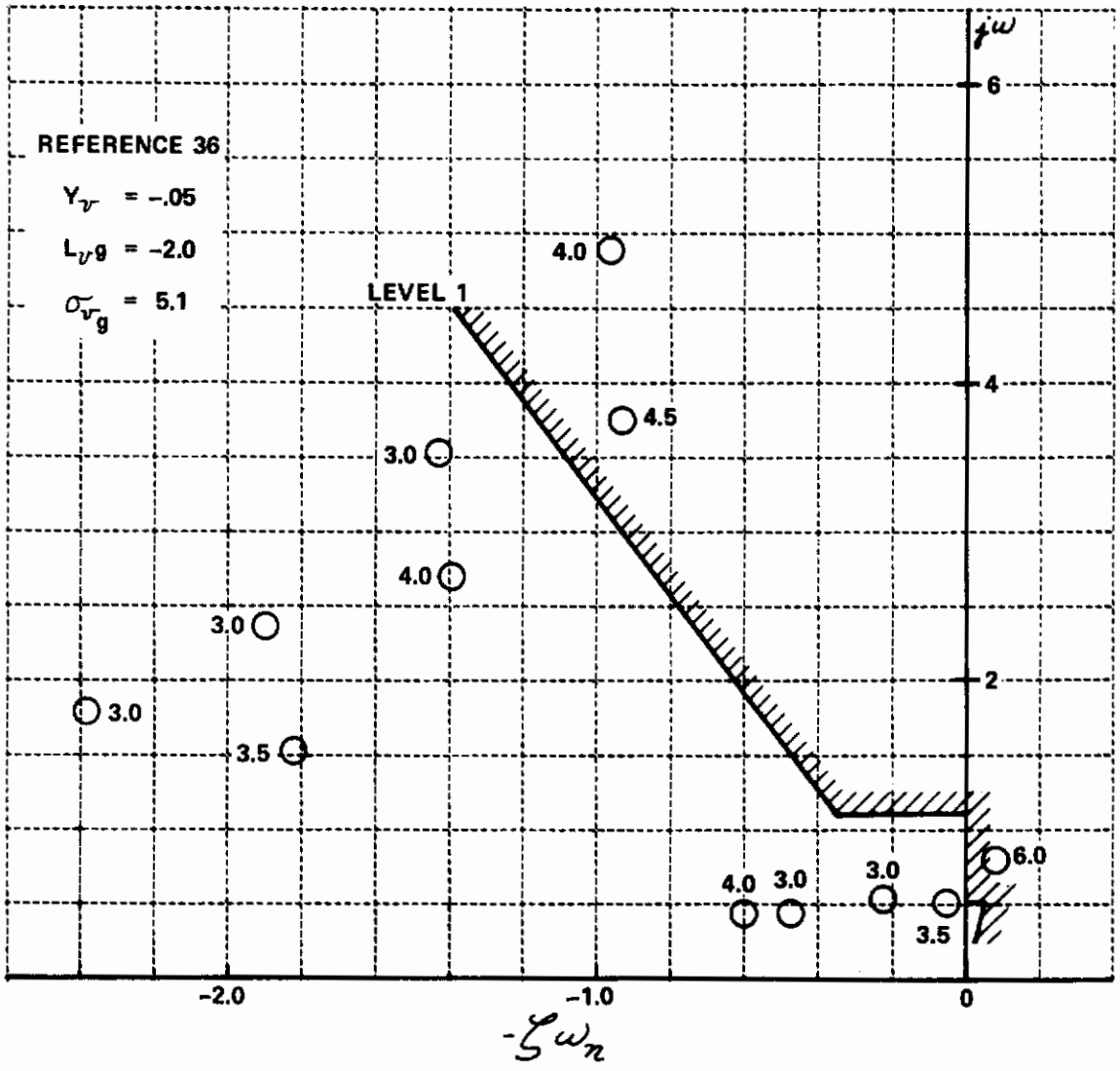

Figure 48 (3.2.2.1) PILOT RATING CORRELATION WITH OSCILLATORY ROOT LOCATIONS

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3.2.2.2 DIRECTIONAL DAMPING

REQUIREMENT

3.2.2.2 Directional damping. While hovering at zero airspeed, the yaw mode shall be stable and the time constant shall not exceed the following:

Level 1:	1.0 second
Level 2:	2.0 seconds

For Level 3 operation there shall be no tendency toward aperiodic divergence in yaw.

DISCUSSION

There has been considerable disagreement in the literature with respect to minimum acceptable yaw damping levels in hover. Figure 1(3.2.2.2) indicates some of these requirements including the present values. MIL-H-8501A and AGARD 408 (References 15 and 46) specify minimum yaw damping values as a function of I_z calling for angular velocity damping of $K(I_z)^{0.7}$ ft-lb (rad/sec) where K is 27.0 in MIL-H-8501A (VFR and IFR) and 14.0 in AGARD 408 (single failure). If $K(I_z)^{0.7}$ is divided by I_z, the result is

$$K(I_z)^{-0.3} \quad \text{sec}^{-1}.$$

The dimension of the latter is that of the yaw damping derivative, N_r. The time constant for the first-order yaw mode in hover can be approximated by

$$\tau = -\frac{1}{N_r}$$

RTM-37 (Reference 47) specifies a maximum value of 1.5 seconds for this first-order time constant. This is equivalent to specifying values of N_r more negative than -0.67.

The choice of minimum acceptable yaw damping appears to be a function of N_v as indicated in the experiment of Reference 48 which was conducted in the presence of a simulated steady wind and simulated turbulence [see Figures 20(3.2.3.2) to 23(3.2.3.2)]. At $N_v = 0.01$, a 3.5 rating was achieved with N_r as low as -0.5 for the VFR hover task (with optimum $N_{\delta_{RP}}$). This minimum damping increased to -1.2 at $N_v = 0.02$, to -3.5 at $N_v = 0.035$, to -5.0 at $N_v = 0.05$.

The following list (from References 13 and 16) gives values of N_v at hover for several VTOL's and helicopters.

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Aircraft	$N_{v'}$
X-22A	.005
VZ-4DA	0
XC-142A	-.00037
X-19	.0005
XV-5A	.002
SH-3A	.009
H-34A	.01
H-19	.035
UH-1D	-.023
CH-46	0

Note that the VTOL's have generally lower values of $N_{v'}$ than the helicopters. Thus the dependency of yaw damping on $N_{v'}$ is expected to be more pronounced on helicopters. This problem is in need of further research because $N_{v'}$ along with $Y_{v'}$, $L_{v'g}$, X_u and M_{ug} determine a vehicle's gust response characteristics. At this time there is no satisfactory manner of stating a requirement to ensure mutual compatibility of gust response and control response characteristics.

Some additional data for $N_{v'} = 0$ cases are shown in Figures 4(3.2.3.2) and 14(3.2.3.2) from References 38 and 49 respectively. In Figure 4(3.2.3.2) both the single- and two-axis results indicate a minimum N_r of -1.0 for a 3.5 rating and Level 2 boundaries around $N_r = 0$. Figure 14(3.2.3.2) indicates an N_r or -0.5 for a rating of 3.5.

Reference 50 summarizes the general state of the data base as follows: "The basic rotary damping requirements do not appear to be significantly size-, mission-, or weight-dependent. Basic requirements for minimum L_p and N_r appear to be about -1.0, but these may be low for high gust sensitivity and correspondingly high control effectiveness."

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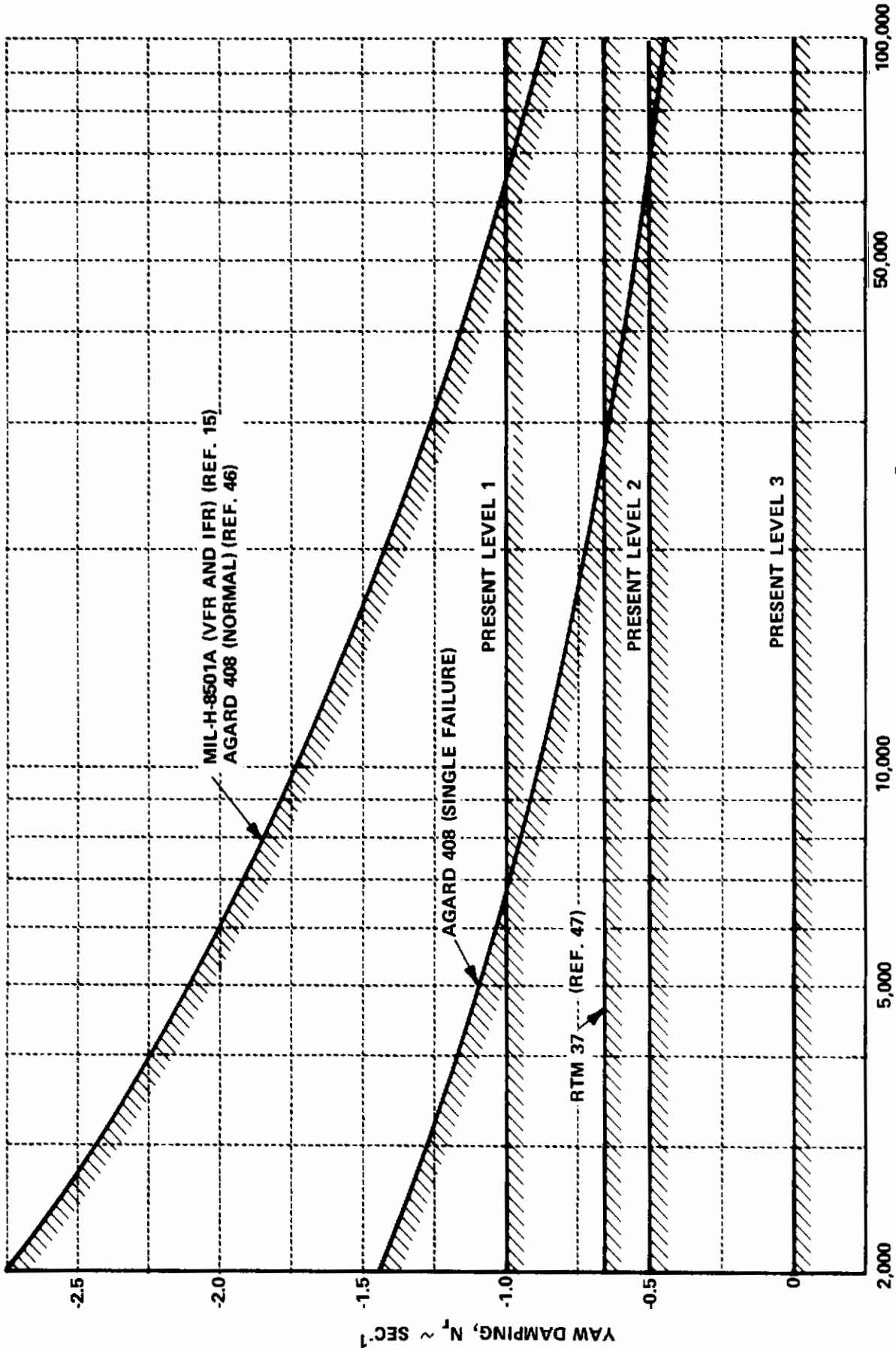

Figure 1 (3.2.2.2) DIRECTIONAL DAMPING REQUIREMENTS IN HOVER

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3.2.3 CONTROL CHARACTERISTICS

3.2.3.1 CONTROL POWER

REQUIREMENTS

3.2.3 Control characteristics. To ensure adequate hover and low-speed control characteristics, the following requirements shall be satisfied starting from flight at constant speed with zero angular rate.

3.2.3.1 Control power. With the wind from the most critical directions relative to the aircraft, control remaining shall be such that simultaneous abrupt application of pitch, roll and yaw controls in the most critical combination produces at least the attitude changes specified in table IV within one second from the initiation of control force application.

TABLE IV. Attitude Change in One Second or Less (Degrees)

	Pitch	Roll	Yaw
Level 1	±3.0	±4.0	±6.0
Level 2	±2.0	±2.5	±3.0
Level 3	±2.0	±2.0	±2.0

DISCUSSION

The amount of control power needed by an airplane can be generally classified as that required for

1. Trimming
2. Maneuvering
3. Stabilizing

Such a classification is very convenient for discussion purposes. Also it suggests that total control power be defined as the sum of the individual control power needs in each category. However, it would probably be better not to promote such a definition. Considering the current state of knowledge about control power, it would seem better to say that total required control power is a function of the individual control power needs in each category. Whether or not the function is equivalent to simple addition of the "components" remains to be established experimentally.

There is no particular problem in specifying the control power needed to trim. The easiest approach is to make use of the consequences of the laws of mechanics. For example, paragraph 3.2.3 refers to flight at constant speed and zero angular rate. This flight condition is one of trim and achieving it requires that the control moments balance all other moments.

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The maneuvering and stabilizing categories are not so straightforward as trim. First consider the maneuvering category. The problem of specifying control power requirements here can be best illustrated by starting with the basic equations of motion in vector form

$$\vec{F} = m\dot{\vec{V}} \quad \text{and} \quad \vec{M} = \dot{\vec{H}}$$

The control moments are included in $\vec{M}$. In order to specify what these control moments should be it is necessary to know the angular momentum vector, $\vec{H}$, as a function of time. In particular it may be necessary to consider several such functions, say $\vec{H}_1(t)$, $\vec{H}_2(t)$, and $\vec{H}_3(t)$, each over some appropriate time interval, before the condition of maximum control power application can be found. Each one of these functions is, of course, simply a different maneuver. The problem of writing requirements is that these functions are not known.

A survey of the literature will show that particular maneuvers are verbally distinguished but that precise descriptions of the maneuver kinematics are lacking. Before maneuver control requirements can be properly formulated it will be necessary to acquire the kinematic or motion data showing which maneuvers involve the largest accelerations, the fastest rates, and the greatest displacements, or the combinations thereof. Such kinematic data along with corresponding control usage data will go a long way in identifying those maneuvers which are most critical.

In lieu of precise kinematic data, the requirement of 3.2.3.1 used, as a measure of minimum maneuvering control power needs, certain values of attitude angles that should occur within 1.0 second after an abrupt control input. These values are thought to be sufficient for moderate maneuvering. Also, simultaneous application of controls is specified since certain types of VTOL control systems use engine compressor bleed air. For these types of systems, the moment available about one axis can depend on the moments being used about the other axes.

The control power used for stabilizing depends on three things:

1. turbulence spectrum,
2. vehicle dynamics, and
3. loop closures (i.e., the pilot-vehicle system).

Taking these three factors into account to specify control power usage is still a fertile field for both analytical and experimental research. It is felt that 3.2.3.1 accounts somewhat for turbulence in that portions of the data were acquired under conditions representing flight in gusty air.

In both MIL-H-8501A and AGARD 408 (References 15 and 46) the pitch control requirements were formulated in terms of the pitch angle response that can be developed in 1.0 second following an abrupt 1.0 inch control input (and also full control input). For a simple single-degree-of-freedom representation of pitch dynamics, the pitch angle response to a step input in δ_{ES} is given by

$$\theta(t) = \frac{M_{\delta_{ES}}\,\delta_{ES}}{M_q^2}\left(e^{M_q t} - 1 - M_q t\right) \qquad (1)$$

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If we substitute t = 1 second we get

$$\theta(1) = \frac{M_{\delta_{ES}} \delta_{ES}}{M_q^2} \left(e^{M_q} - 1 - M_q\right) \tag{2}$$

Eliminating pitch damping, M_q , from further consideration we see that $\theta(1)$ depends on both the sensitivity $M_{\delta_{ES}}$ and the applied control deflection δ_{ES} . This factor complicates the analysis of experimental data when using $\theta(1)$ as a correlating parameter because there are several interpretations that can be made using equation 2.

First, assume that δ_{ES} = 1.0 inch step. Then we get

$$\theta(1) = \frac{M_{\delta_{ES}}(1.0)}{M_q^2} \left(e^{M_q} - 1 - M_q\right) \tag{3}$$

From equation (3), it can be argued that for fixed damping, $\theta(1)$ is a measure of the sensitivity for 1.0 inch step inputs. However, it can also be argued that $\theta(1)$ is a measure of the amount of control power applied by a 1.0 inch step input, i.e., the product $M_{\delta_{ES}}(1.0)$.

Next consider the following form of equation (2):

$$\frac{\theta(1)}{\delta_{ES}} = \frac{M_{\delta_{ES}}}{M_q^2} \left(e^{M_q} - 1 - M_q\right) \tag{4}$$

This equation expresses the ratio of $\theta(1)$ to any δ_{ES} , and for fixed M_q we could use this ratio as a measure of sensitivity. Note however that the numerical value of $\theta(1) / \delta_{ES}$ computed from equation (4) is the same as the numerical value of $\theta(1)$ computed from equation (3). This is not particularly profound mathematics. But there are different implications in the case of writing sensitivity requirements depending on whether one's thoughts are being guided by equation (3) or equation (4). It can be seen that paragraph 3.2.3.2, which specifies the ratio of attitude change to control input, is based on equation (4). If equation (3) had been used as a guide, then the sensitivity requirements of paragraph 3.2.3.2 would have to specify that a 1.0 inch control input be used in demonstrating compliance. For configurations with high sensitivities, the use of a 1.0 inch input would result in some pretty large attitude excursions during demonstration flight testing.

As a third interpretation of the significance of $\theta(1)$ as a control requirement, we write equation (4) as

$$\theta(1) = \frac{M_c}{M_q^2} \left(e^{M_q} - 1 - M_q\right) \tag{5}$$

where $M_c = M_{\delta_{ES}} \delta_{ES}$ is the applied control power for an abrupt step control input of some unspecified size. From this equation we see that, for fixed pitch damping, $\theta(1)$ is a measure of control power applied to the vehicle. The distinction between control power applied to the vehicle and control power available to the pilot is an important one to make. There are cases in the literature where experimental results are presented in terms of boundaries on a graph of damping vs control power available on the vehicle. Without

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knowledge of how much of this available control power was being used by the pilot, care must be taken in drawing conclusions from such data. Similarly, one can find boundaries of damping vs sensitivity but without an indication of the control deflection and control power being applied to the vehicle by the pilot.

There is not a large amount of experimental data showing the actual control power applied to the vehicle. This lack of data and some suggestions on how to measure and present and interpret control power usage data are discussed in Reference 51.

However, the data that are available lead to some interesting conclusions. Figure 1 (3.2.3.1) shows some actual usage results for several vehicles developed under the VJ-101 program (Reference 52). In Figure 1 (3.2.3.1), $\dot{q}_{.99}$ is the control power which covers 99% of the total usage and lies between 0.20 and 0.30 rad/sec^2 for the several tasks performed with the attitude SAS operating on the three different vehicles. There is little, if any, variation of control power usage over the weight range of 2 to 8 tons.

Reference 53 contains many control moment distribution curves for the Do-31 hovering rig. One of the curves compares the pitch angular acceleration usage for a normal, smooth air flight with the usage when artificial disturbances were introduced. In smooth air, the probability of exceeding about ± 0.15 rad/sec^2 was essentially zero. With disturbances, on the other hand, the control usage increased to about ± 0.375 rad/sec^2 before probability of exceedance became very low.

Figures 2 (3.2.3.1) and 3 (3.2.3.1) also contain some control power data. However, the values indicated are those available on the vehicle and are not necessarily indicative of what the pilot used except perhaps at the lower values.

Figure 2 (3.2.3.1), from Reference 54, shows a control power of about .5 rad/sec^2 is needed to achieve a 3.5 rating and a rating of 6 is not encountered until the control power is reduced to about .2 rad/sec^2. The HIAD (Reference 55) limit line on Figure 2 (3.2.3.1) corresponds to a maximum cockpit deflection of ± 7 inches.

Figure 3 (3.2.3.1) taken from Reference 56 shows that at a control sensitivity of about 0.1 rad/sec^2-in., a control power of at least .25 rad/sec^2 is needed to achieve a 3.5 rating, and that ratings deteriorate rapidly if the control power falls much below .25 rad/sec^2.

Figures 4 (3.2.3.1) through 7 (3.2.3.1) contain some control power usage data from Reference 57. These figures show the percentage of time that a given value of control power is exceeded for several dynamic configurations under different wind, gust and task conditions. It is to be noted that the control power in these figures is that being applied to the vehicle and not just that being commanded by the pilot.

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Figure 4 (3.2.3.1) shows the effect on control power usage of SAS, turbulence and task for a configuration whose longitudinal and lateral dynamics are identical. Note that on this figure, control power (CP_o) is defined as the square root of the sum of the squares of pitch and roll control powers. Except for the condition where gusts are twice the nominal value, Figure 4 (3.2.3.1) shows that a CP_o value of 0.5 rad/sec^2 is exceeded less than 5% of the time for the hover task, the maneuver task, the nominal dynamics (M_q = - 3) and the dynamics with lower damping (M_q = - 1.0). Because of the way CP_o is defined we cannot determine the values of pitch and roll control power separately. However, we know that

$$M_c \leq CP_o \quad \text{and} \quad L_c \leq CP_o$$

and since σ_{u_g} = 5.1 ft/sec while σ_{v_g} = 1.3 ft/sec, a rough guess for the ratio of $M_{c_{max}}$ to $L_{c_{max}}$ would be about 4. The reference to maximum control powers in this last sentence is perhaps a bit casual. It would be more precise to use the value of pitch (roll) control power which is exceeded for some given percentage of time.

Figure 5 (3.2.3.1) shows the effect of task and wind direction on control power usage for a configuration whose dynamics are different from those of Figure 4 (3.2.3.1). For the three curves on Figure 5 (3.2.3.1), a control power value of 0.35 rad/sec^2 is exceeded less than 5% of the time.

Figure 6 (3.2.3.1) shows the effect of SAS level and task on the pitch control power usage. Except for the value of N_r, the basic dynamics are the same as Figure 5 (3.2.3.1). The two arrows pointing down from the caption in the center of Figure 6 (3.2.3.1) indicate the values of pitch control power needed to satisfy the Level 1 $\theta(1)$ response requirement of paragraph 3.2.3.1 of Reference 1. For the case where M_q = -.85, a pitch control power of .14 rad/sec^2 is required. This control power would be exceeded about 4% of the time in the hovering task and 7% of the time in the maneuvering task. For the case where M_q = -1.2, a pitch control power of about 0.155 rad/sec^2 is required. For the hover task this is exceeded about 1% of the time and for the maneuver task about 6% of the time.

Figure 7 (3.2.3.1) shows how control power limitations affect pilot rating. The upper part of this figure shows the measured control power usage for a hovering and maneuvering task when there is no limitation on the control power available to the pilot. The lower part of the figure shows pilot ratings when the control power available to the pilot is limited to the values shown. Note the rapid deterioration of rating at about 0.08 rad/sec^2 which, with more control, would be exceeded only 11% of the time. For pitch control power values of about 0.10 rad/sec^2 and more, control power deficiencies were noted only occasionally (probability of exceedance was about .06).

The data just presented provide reasonable evidence from which to conclude that control power usage for maneuvering and stabilizing should fall at least in the 0.15 to 0.5 rad/sec^2 range depending on the type of control system, the disturbances encountered, and of course the particular maneuvers involved.

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The interesting thing about this range of control powers, i.e., about 0.15 to 0.5 rad/sec^2, is that it is numerically equal to the range of numerical values of control sensitivities that is indicated by the experimental data to vary from minimum acceptable all the way up to maximum acceptable depending on the magnitude of pitch damping and speed stability.

Figure 8 (3.2.3.1), which compares the results of several experiments in the form of 3.5 boundaries on a pitch damping vs sensitivity graph, amply illustrates this. It is therefore not too unreasonable to relabel the horizontal axis with a title like "maximum control power applied by the pilot".

Going back to equation (5), suppose we substituted values of .15 to 0.5 for M_c , the applied control power, and then listed out the corresponding range of $\theta(t)$ so computed. This range of $\theta(t)$ would be the same as if we had used equation (3) and equation (4) along with a 0.15 to 0.5 range of control sensitivity, $M_{s_{ES}}$. Again this is not profound mathematics. But equating applied control power with sensitivity thus seems warranted by the experimental data, limited though it may be. Hence it appears that with proper qualification, control sensitivity data can be useful in formulating control power requirements.

The qualification deemed appropriate is that we can supplement actual control power usage data by using the lower values of control sensitivity, as indicated by pilot rating boundaries on control sensitivity vs damping graphs, to establish minimum control power requirements. However, it is necessary to interpret such sensitivity data in terms of control power needs.

One such interpretation seems best explained in terms of the simple X-force equation:

$$\dot{u}(t) = X_u u(t) - g\theta(t)$$

During the initial stages of a rapid pitching motion, $u(t)$ is small. Thus during some short time interval following an abrupt control input, the following equation is a satisfactory approximation.

$$\dot{u}(t) = -g\,\theta(t)$$

With this equation we can interpret the need for certain values of control power as follows: Control powers of a certain minimum level, say 0.15 to 0.5 rad/sec^2, are needed to command the rapid pitch angle changes so that the component of thrust provides the horizontal accelerations deemed adequate for useful horizontal translational maneuvers.

The particular horizontal accelerations needed would seem to depend not so much on the size of a vehicle as on the types of low-speed translational maneuvering the vehicle is called upon to perform. Size (fuselage, length, wing span) could, however, impose a practical constraint on the allowable horizontal accelerations. The studies of pitch response requirements for maneuvering reported in the literature have not, it seems, given enough attention to the linear accelerations associated with pitching motions.

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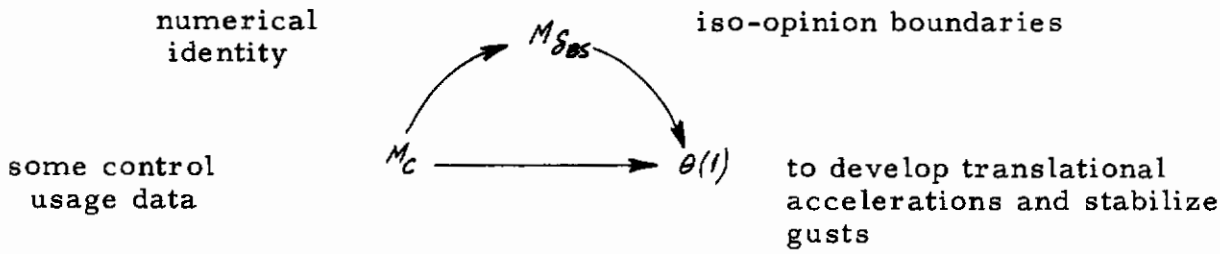
Assuming that $\theta(1)$ is a measure of this ability to develop horizontal accelerations, we can place minimum control power requirements in correspondence with minimum $\theta(1)$ requirements. Now constant $\theta(1)$ lines on a damping vs sensitivity graph that contains pilot rating boundaries tend to approximate the iso-opinion boundaries. Figure 8 (3.2.3.1) is a global picture showing the degree to which a constant $\theta(1)$ line of .05 radians approximates a part of the diverse pilot rating boundaries reported in several studies. If, as appears to be true, we can numerically equate minimum control power needs with sensitivity for these portions of the iso-opinion boundaries, then we can, in fact, use the sensitivity data as if it were control power data. This idea is indicated in the following sketch.



The Level 1 requirements of paragraph 3.2.3.1 were thus formulated by selecting the $\theta(1)$ lines that seemed to best match the overall data, much of which will appear in the next section. Although we have been using the symbolism and terminology for longitudinal control, essentially the same procedure and ideas were used in formulating the control power requirements for the roll and yaw axes.

In response to comments received during the review period, the Level 3 response requirements have been set at 2 degrees in one second or less, for the pitch, roll and yaw axes. Inasmuch as these control power levels are somewhat arbitrary and are in a region of high gradient of pilot rating with control power, these changes are considered reasonable.

As a closing remark, we call attention to the conditions under which demonstration of compliance is to be carried out. These conditions are: constant speed, and zero angular rate with the wind from the most critical direction relative to the aircraft. In particular they include ascending and descending flight. It is recommended that the reader temporarily turn to requirement paragraphs 3.2.5 to 3.2.5.2 which state specific climb and descent conditions under which control requirements must be satisfied. Although it could be argued that these conditions should have also been stated in 3.2.3.1, it is believed that the last sentence in 3.2.5 provided for more concise wording.

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In summary, the requirements of 3.2.3.1 are believed to provide realistic minimum control powers. There is still a great need for more research on this problem and in particular for data in a format that reflects how much control is being applied to the vehicle rather than just what is available.

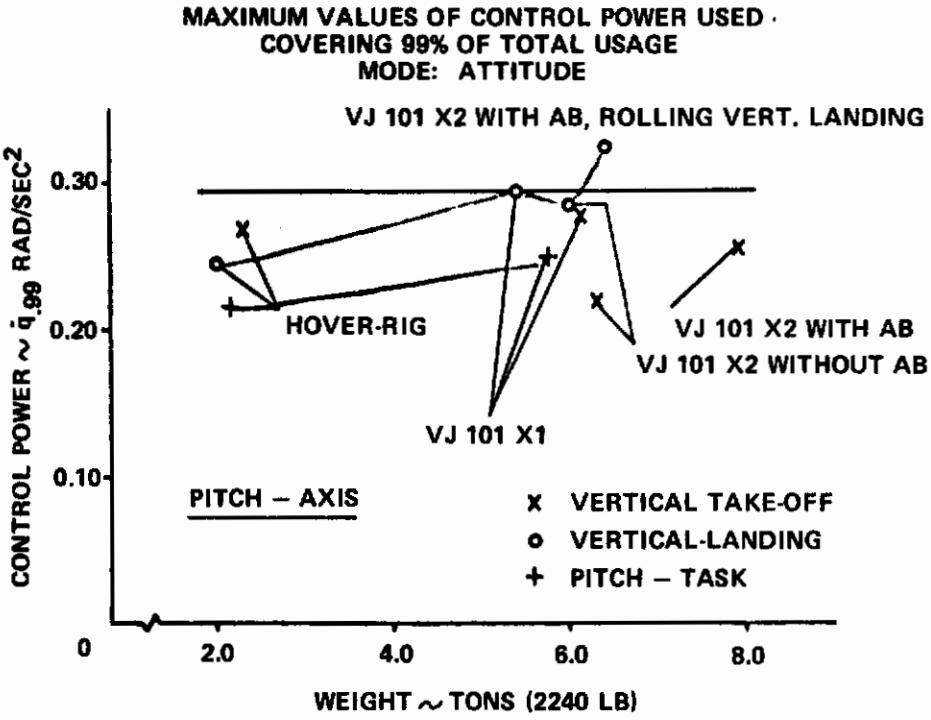

Figure 1(3.2.3.1) CONTROL POWER USAGE FROM FLIGHT TEST RESULTS OF REFERENCE 52.

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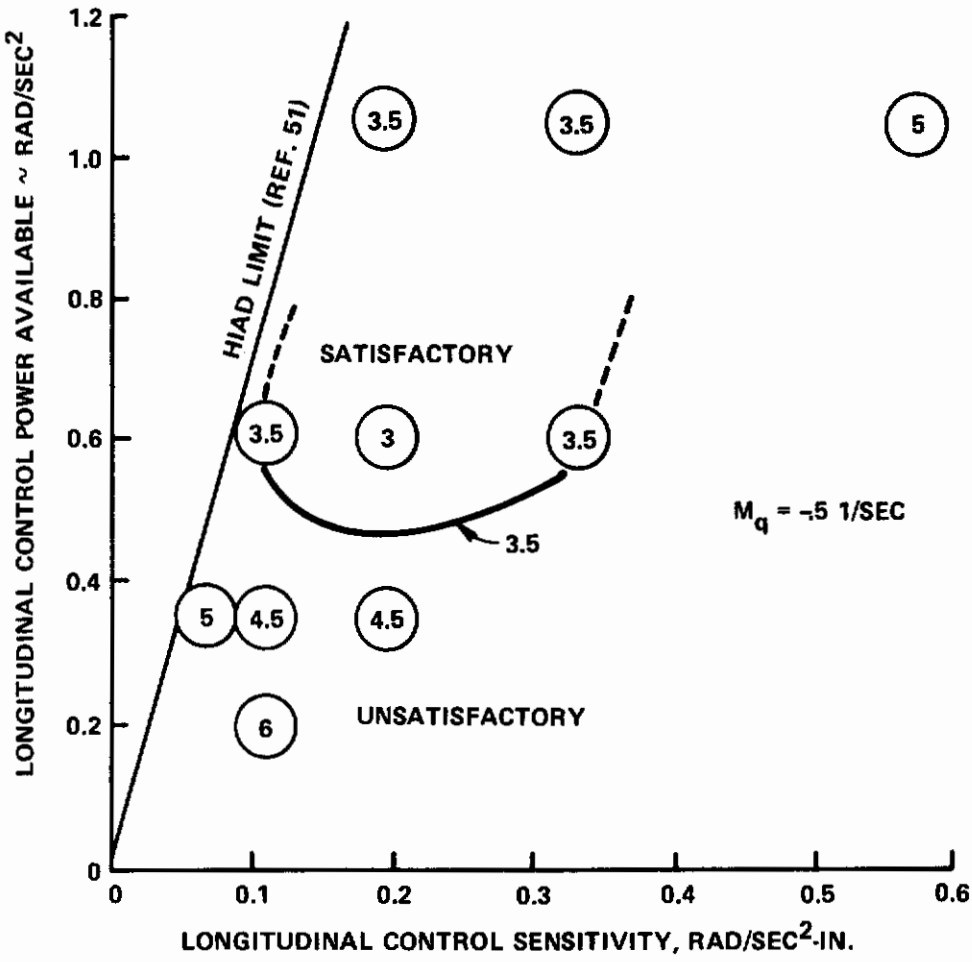

Figure 2 (3.2.3.1) CONTROL POWER AND CONTROL SENSITIVITY EFFECTS ON PILOT RATING FOR MANEUVERING TASK OF REF. 54

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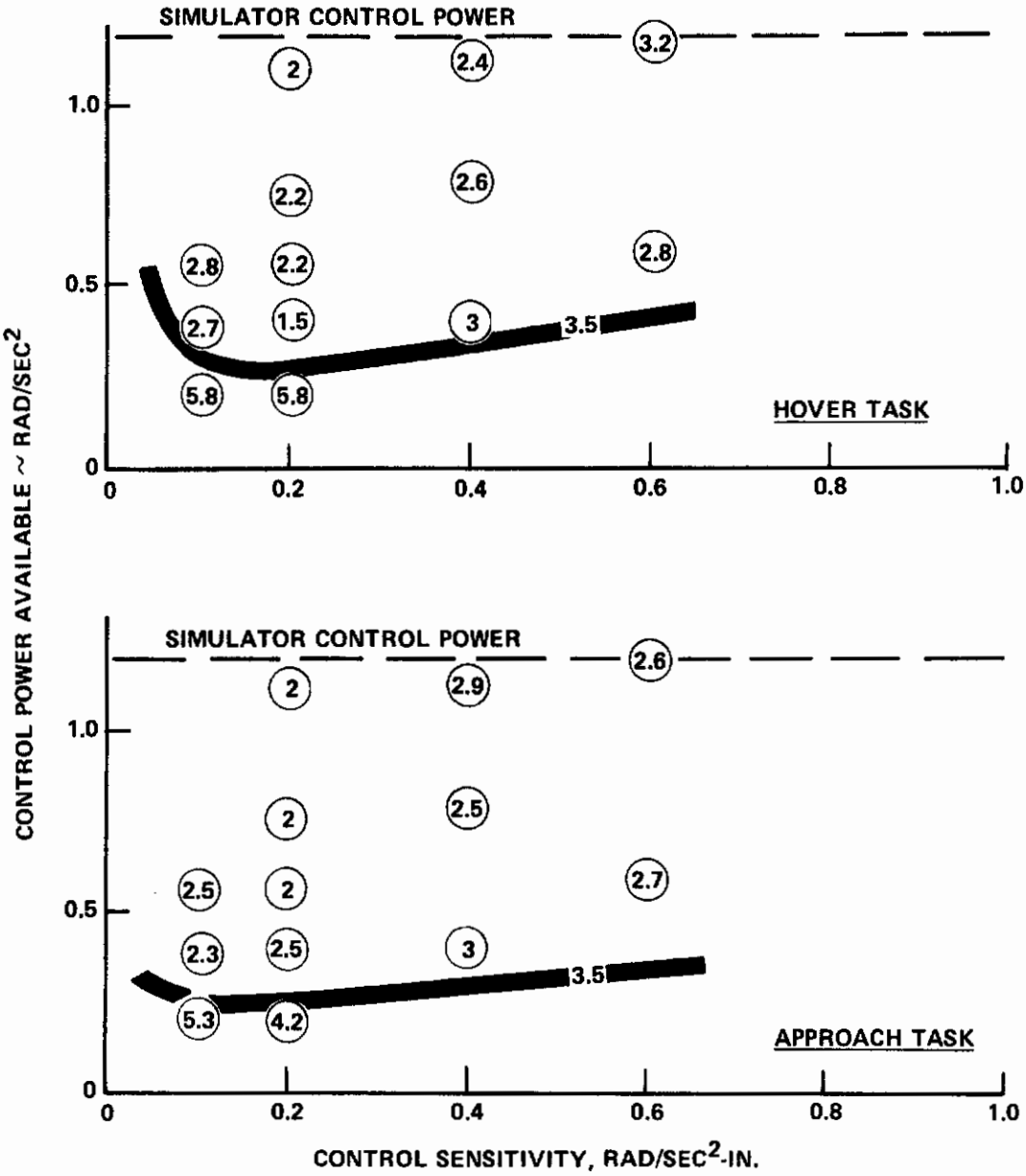

Figure 3 (3.2.3.1) CONTROL POWER AND CONTROL SENSITIVITY EFFECTS ON PILOT RATING FOR FLIGHT TESTS OF REFERENCE 56. (M_q = -0.5 1/SEC)

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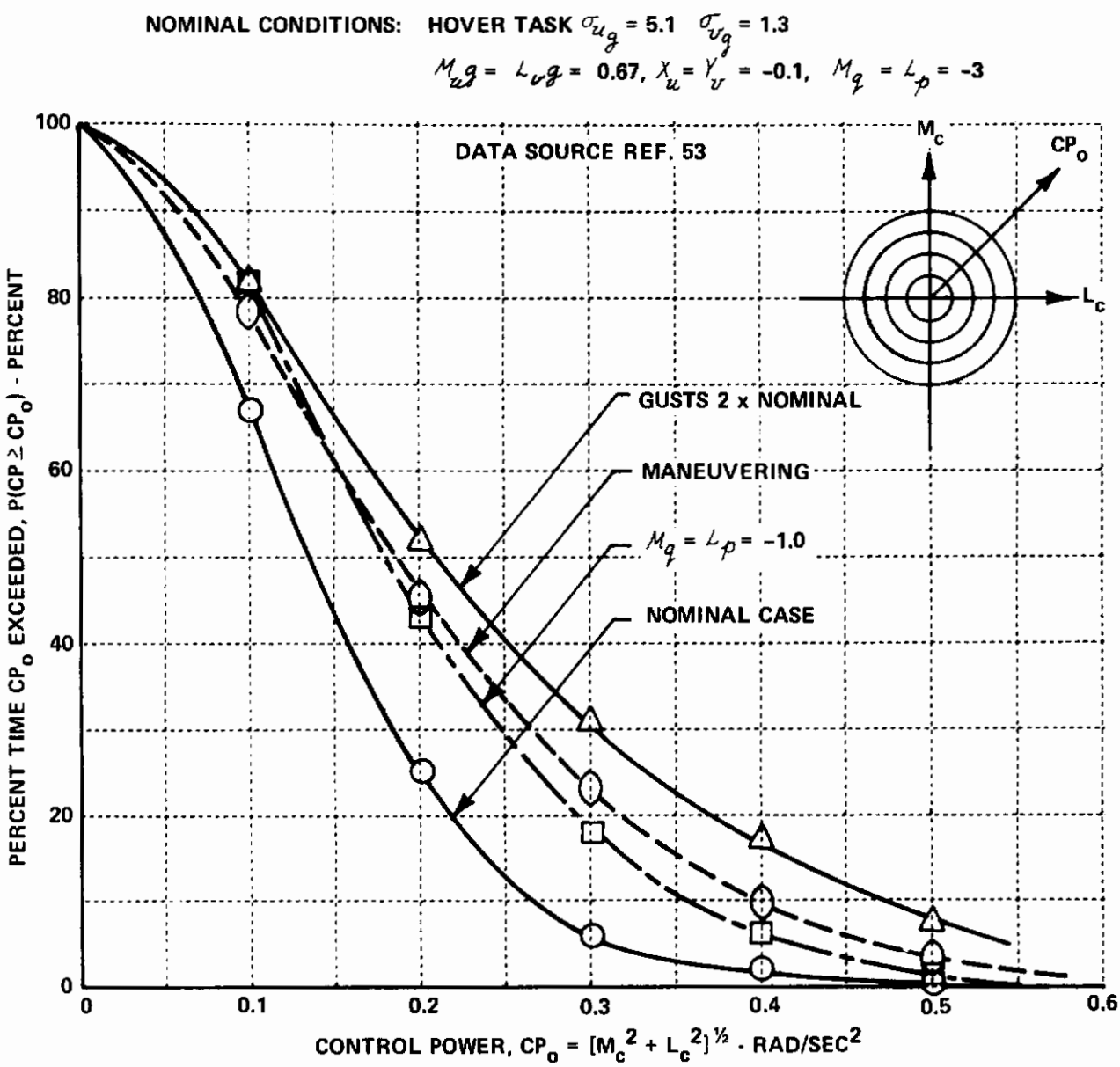

Figure 4 (3.2.3.1) EFFECT OF SAS, TURBULENCE AND TASK ON CONTROL POWER USAGE

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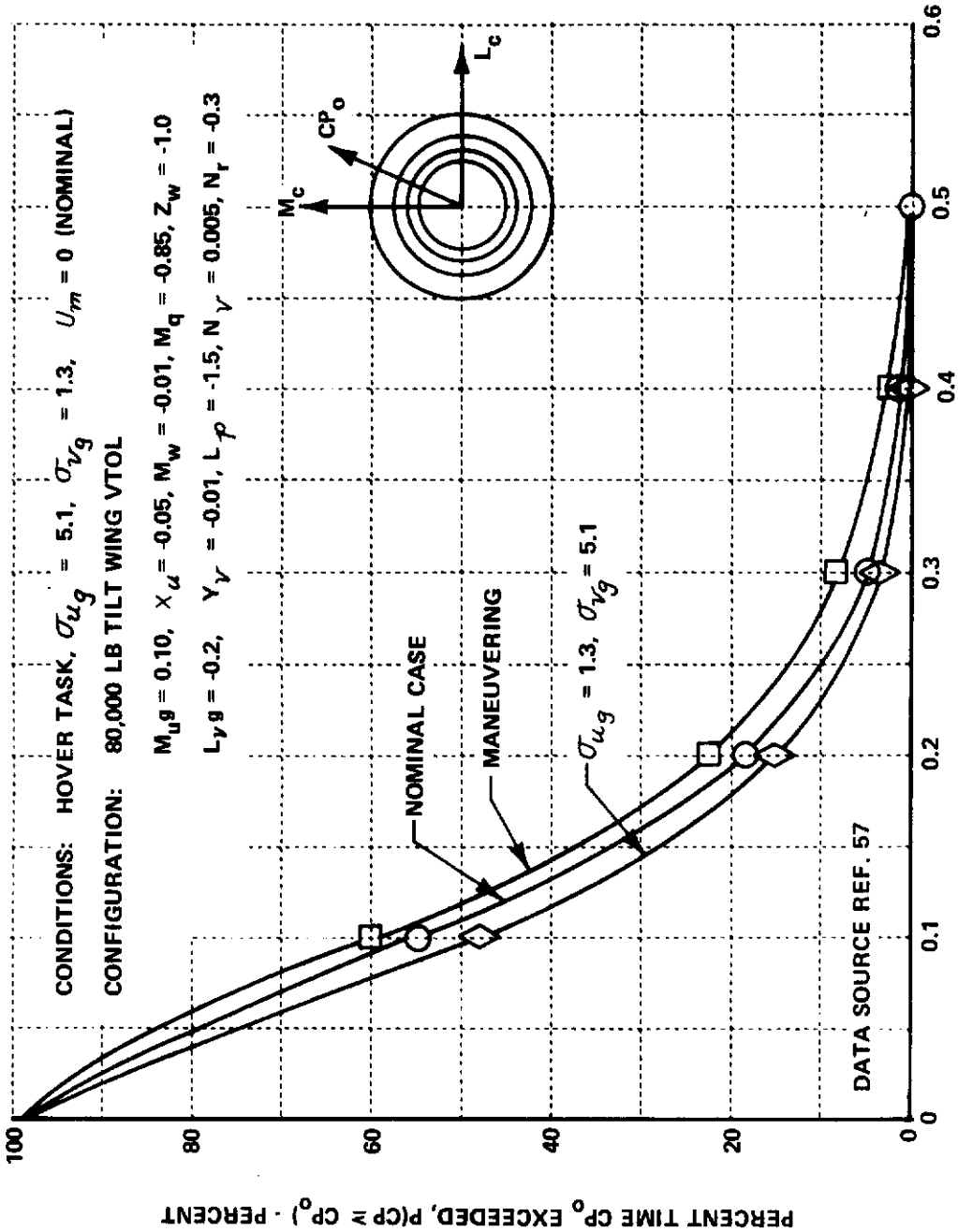

Figure 5 (3.2.3.1) EFFECT OF TASK AND WIND DIRECTION ON CONTROL POWER USAGE

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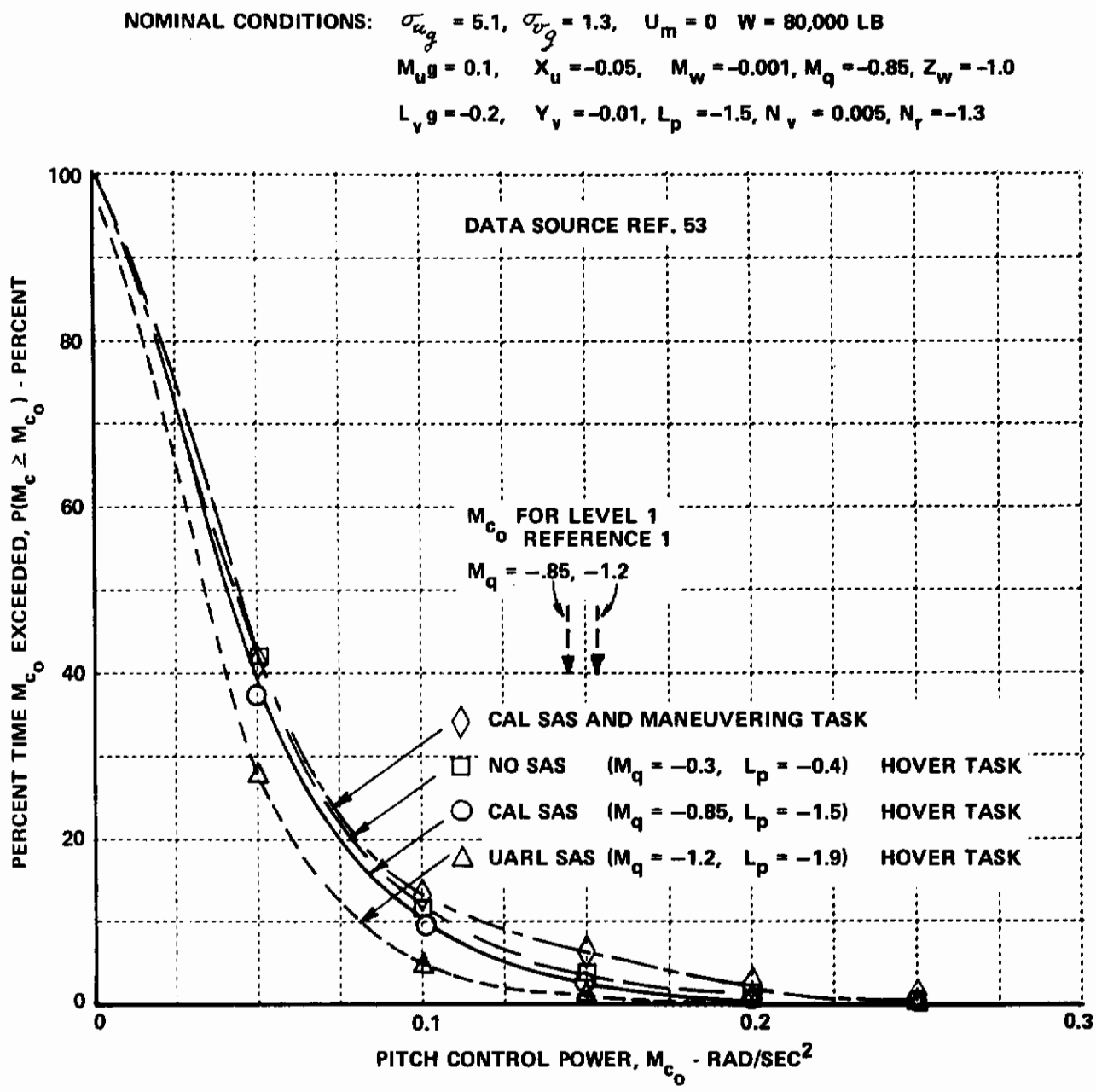

Figure 6 (3.2.3.1) EFFECT OF SAS LEVEL AND TASK ON PITCH CONTROL POWER USAGE OF TILT WING

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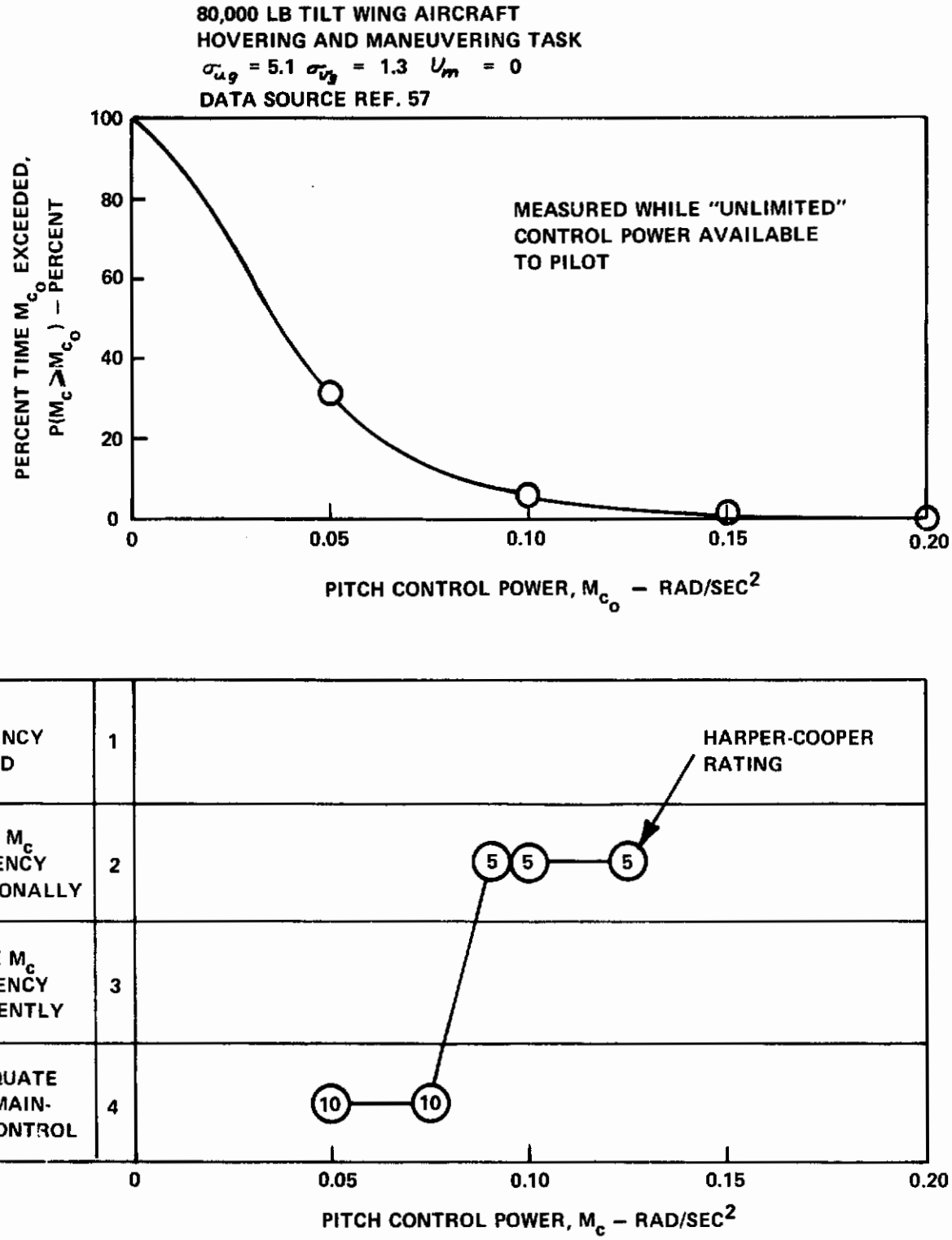

Figure 7 (3.2.3.1) COMPARISON OF PITCH CONTROL POWER USAGE WITH PILOT RATING OF LIMITED CONTROL POWER

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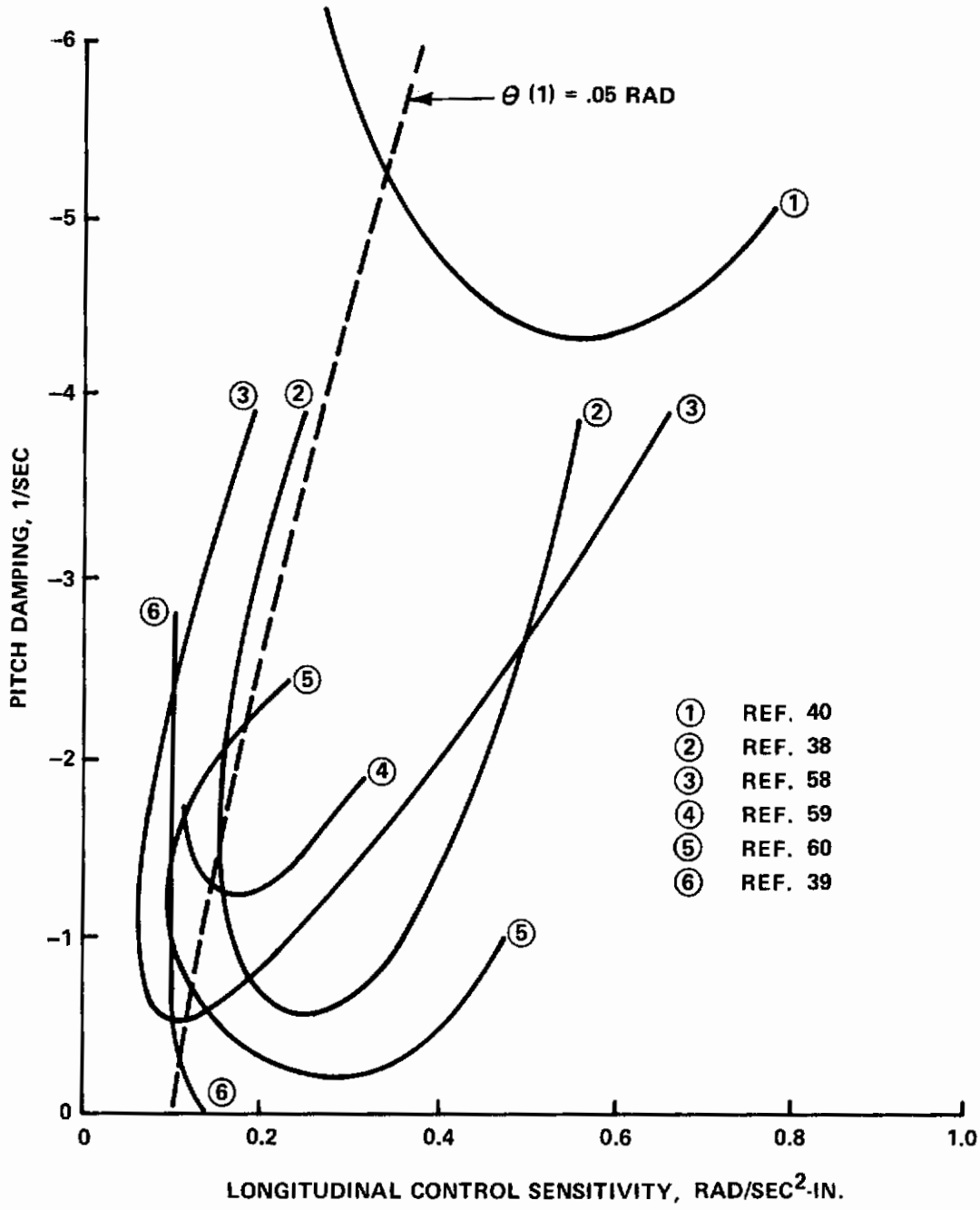

Figure 8 (3.2.3.1) DIVERSITY OF 3.5 PILOT RATING BOUNDARIES FOR PITCH RATE SYSTEMS

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3.2.3.2 RESPONSE TO CONTROL INPUT

REQUIREMENT

3.2.3.2 Response to control input. The ratio of the maximum attitude change, occurring within the first second following an abrupt step displacement of the appropriate cockpit control, to the magnitude of the cockpit control command shall lie within the bounds of table V. There shall be no objectionable nonlinearities in aircraft response to control deflections and forces.

TABLE V. Response to Control Input in One Second or Less (Degrees Per Inch)

Level	Pitch		Roll		Yaw	
	Min	Max	Min	Max	Min	Max
1	3.0	20.0	4.0	20.0	6.0	23.0
2	2.0	30.0	2.5	30.0	3.0	45.0
3	1.0	40.0	1.0	40.0	1.0	50.0

DISCUSSION

In addition to control sensitivity, that is, the acceleration per unit control input, the pilot's opinion of a vehicle's flying qualities is influenced by such factors as the vehicle's dynamics, control power, task, atmospheric disturbances, etc. However, control sensitivity is an important parameter in that improper selection of sensitivity can degrade the flying qualities of an otherwise satisfactory vehicle to an unacceptable level. Conversely, judicious selection of control sensitivities of a vehicle with marginal handling qualities can result in a considerable improvement in pilot opinion. Generally speaking, low sensitivities usually result in a vehicle with sluggish response characteristics while excessively high sensitivities tend to lead to overcontrolling.

Since control sensitivity is a difficult parameter to determine experimentally, the allowable range of sensitivities is specified indirectly in terms of attitude response within one second per unit control input. In addition to practical considerations, this method is considered reasonable because of the relationship between attitude response and maneuverability in translation over the ground.

VTOL airplanes do not require large normal accelerations to

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accomplish useful low-speed missions and to perform necessary flight tasks such as takeoff and landing. But the normal accelerations that are used must be developed by thrust control because the mechanism for developing lifting loads on conventional fixed wings is either absent or extremely weak. Thus, control of angle of attack, which is a fundamental variable in developing lift on fixed wings, is not a primary low-speed flight control consideration. Longitudinal and lateral moment control is still important, however, because it is effective in developing horizontal accelerations by tilting thrust vectors. Thrust vectors can also be tilted by configuration changes, but even in this case, longitudinal and lateral control will probably still be needed for gust stabilization.

In stabilizing and maneuvering a V/STOL aircraft, pilots tend to use rapid and frequent control inputs rather than long, steady control inputs. Thus, the pilot's awareness of the controllability and maneuverability of the vehicle is influenced primarily by its short term attitude response to control inputs. In particular, for the pitch and roll axes, attitude response directly influences the capability of the vehicle to rapidly develop translational accelerations in the horizontal plane.

For a given dynamic configuration, the minimum satisfactory control sensitivities are probably related to the magnitude of the control deflections required to perform the mission. Obviously pilot fatigue and discomfort will be aggravated by very low sensitivities which require large and frequent control deflections to control and maneuver the aircraft. Low sensitivities will be even more troublesome when high force gradients are present.

High control sensitivities tend to lead to problems of overcontrolling and PIO because of the difficulty of making precise control inputs. This situation is further aggravated when coupled with low force gradients.

In addition to the above considerations, the desirable level of control sensitivity is also dependent on the vehicle dynamics and the task. The majority of the data pertinent to the determination of allowable control sensitivities has been generated for rate type control systems, that is, vehicles which respond to steady control inputs with a constant rate of attitude change. However, a few experiments have considered vehicles whose pitch and roll response was predominantly that of an attitude system. Generally, the data indicates that higher control sensitivities are acceptable with attitude systems than with rate systems. Since no other requirements are directed toward specific types of control response dynamics, the control sensitivity requirements have been written to encompass the range of dynamic configurations which have been investigated. The following paragraphs discuss the data base for the specification.

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Specification Data Base

Rate Systems

An early in-flight experiment to investigate the relationship between angular damping and control sensitivity is described in Reference 61. The test vehicle was a small single rotor helicopter with a gross weight of 5,500 pounds modified to permit variations in pitch, roll, and yaw angular rate damping and control sensitivity. The evaluations included both VFR and IFR tasks although primary emphasis was placed on ILS approaches under blind flying conditions at 45 knots airspeed. While quantitative pilot ratings were not used in this experiment, the damping-sensitivity combinations tested were identified by adjectives ranging from desirable to unacceptable.

Assuming that a transfer function of the form

$$\frac{\theta}{\delta_{ES}}(s) = \frac{M_{\delta_{ES}}}{s(s-M_q)}$$

adequately describes the short term attitude response of this aircraft, the specification requirements can be represented as line boundaries in the $M_q \sim M_{\delta_{ES}}$ plane. These boundaries are compared to the experimental results in Figure 1 (3.2.3.2) and 2 (3.2.3.2). For the pitch and roll axes, the specification boundaries appear somewhat overly restrictive. In fact, the Level 1 boundaries closely approximate an optimum sensitivity line through the data. The preference for low sensitivities in these axes may be attributable to the tight trimming task of the experiment. In the yaw axis, however, the specification boundaries bracket the acceptable region reasonably well.

Reference 38 describes a moving-base (pitch and roll) simulation experiment to establish attitude control requirements for hovering flight. Various combinations of control sensitivity and damping were investigated for each of the three axes, one at a time. The tasks were rapid attitude changes under VFR conditions. In addition, a limited investigation of the effects of controlling two axes simultaneously (i.e., pitch-yaw and roll-yaw) was conducted. The results of these experiments are presented in Figures 3 (3.2.3.2) and 4 (3.2.3.2). In pitch and roll, the Level 1 specification boundaries are in reasonable agreement with the experimental boundaries for the more realistic two-axis control tasks. The yaw results of Figure 4 (3.2.3.2) correlate very well with the specification boundaries.

Figures 5(3.2.3.2) and 6(3.2.3.2), based on data of Reference 39, present the results of a handling qualities investigation utilizing the X-14A, a small deflected jet V/STOL aircraft. The evaluation tasks consisted of hovering and maneuvering,and of ground effect in VFR and under generally calm wind conditions. In addition, vertical takeoffs and landings were performed. The test aircraft was described as exhibiting "a high degree of hovering steadiness and an insensitivity to gust disturbances." It is concluded in Reference 39 that the low levels of rate damping and control sensitivity found satisfactory were attributable to the gust insensitivity of the vehicle.

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A moving-base simulator investigation of pitch, roll and yaw control requirements in hover is described in Reference 60. This experiment utilized "precision maneuvering" tasks as the basis for evaluation and was intended to fill the information gap between the tight trimming tasks of Reference 61 and the gross maneuvering tasks of Reference 38. The results are compared to the specification boundaries in Figures 7(3.2.3.2) through 9(3.2.3.2).

An investigation of the effects of variations in control power and sensitivity at fixed damping levels is documented in Reference 54. The test vehicle was a large tandem rotor helicopter with a model following variable stability system. Separate sets of pilot ratings were obtained for both precision and maneuvering tasks. These results are presented in Figures 10 (3.2.3.2) and 11 (3.2.3.2). When adequate levels of control power were available, the pilots tended to prefer higher minimum sensitivities for the maneuver tasks than for the precision tasks. The specification Level 1 boundary correlates best with the maneuver task data.

A large single rotor helicopter was utilized in Reference 62 to determine control sensitivity and damping requirements in hover. Both visual and instrument tasks were performed in the experiment. The visual tasks consisted of hovering, takeoffs and landings, turns, square patterns and point-to-point translations. The simulated instrument tasks were hooded ILS approaches at airspeeds of about 50 knots. The results are compared to the specification Level 1 boundaries in Figures 12(3.2.3.2) and 13(3.2.3.2). In both pitch and roll, the boundaries correlate best with the data for the ILS tasks. For the roll axis at the intermediate damping levels tested, the specification is somewhat overly restrictive.

Figure 14 (3.2.3.2) presents pilot opinion data for the yaw axis derived from a fixed-base simulation experiment, described in Reference 49. The data exhibits good correlation with the specification boundaries.

Attitude Control Systems

Assuming an uncoupled plunge mode, a linearized small perturbation representation of a V/STOL in hovering flight is given by

$$\frac{\theta}{\delta_{ES}}(s) = \frac{(s - X_u) M_{\delta_{ES}}}{s(s - X_u)(s - M_q) - M_\theta (s - X_u) + M_u g}$$

If the derivative M_u is zero the equation simplifies to:

$$\frac{\theta}{\delta_{ES}}(s) = \frac{M_{\delta_{ES}}}{s^2 + 2\zeta\omega_n s + \omega_n^2} \qquad \text{where} \quad 2\zeta\omega_n = -M_q, \quad \omega_n^2 = -M_\theta$$

For this case the requirements of this specification may be interpreted as indicated by the hatched lines in Figure 15 (3.2.3.2). These hatched boundaries correspond to the minimum and maximum control sensitivities required to achieve the Level 1 pitch attitude responses within one second as

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a function of natural frequency for a damping ratio of 0.3. The effect of increasing damping ratio, ζ, is to increase the control sensitivity required at a given frequency. The magnitude of the shift for a damping ratio of 0.7 is illustrated for the upper Level 1 boundary in Figure 15 (3.2.3.2).

At frequencies less than about 3 radians/second, it can be seen that the boundaries approach lines of constant control sensitivity. At frequencies above 3 radians/second, the maximum pitch attitude response has been reached in less than one second. As frequency is increased, maintaining constant $M_{\delta_{ES}}/\omega_n^2$ keeps the peak attitude constant. The dashed curves indicate the control sensitivity which would be required to achieve the required attitude changes at one second rather than within one second.

These characteristics are at least qualitatively correct as is evidenced by the results of References 63 and 64 plotted in Figure 16(3.2.3.2). The band taken from Reference 64 indicates that at low frequencies the pilots' preferred control sensitivities approach lines of constant control sensitivity while at larger frequencies (greater than about 3 radians/second) there is a tendency to approach lines of constant attitude response per inch of control. This is substantiated by the lower 3.5 pilot rating boundary of Reference 63. However, the upper 3.5 pilot rating curve of Reference 63 implies that there is a maximum satisfactory control sensitivity independent of frequency, probably due to overcontrolling, and a tendency to PIO.

General Case

In the general case the pitch or roll attitude response of a VTOL at hover may be represented by an equation of the form:

$$\frac{\theta}{\delta_{ES}}(s) = \frac{M_{\delta_{ES}}(s - X_u)}{(s+\lambda)(s^2 + 2\zeta\omega_n s + \omega_n^2)}$$

The influence of small values of X_u on the attitude response within one second is small and will be neglected in the subsequent analysis. Thus the assumed transfer function has the form

$$\frac{\theta}{\delta_{ES}}(s) = \frac{M_{\delta_{ES}} s}{(s+\lambda)(s^2 + 2\zeta\omega_n s + \omega_n^2)}$$

The effect of the transfer function pole at $s = -\lambda$ on the specification boundaries is to increase the required control sensitivity at all frequencies as illustrated in Figure 17(3.2.3.2).

An experiment to determine the influence of pitch rate damping, M_q, control sensitivity, $M_{\delta_{ES}}$, and speed stability, $M_u g$, on longitudinal handling qualities utilizing a tandem rotor helicopter is described in Reference 40. In addition, turbulence of 6.3 ft/second RMS and a mean wind of about 15 knots were artificially introduced to provide realistic disturbances to the aircraft. The evaluations consisted of precision VFR hovering tasks. Figure 18 (3.2.3.2) presents the pilot rating boundaries

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obtained for speed stability, $M_u g$, equal to 1.13. The significant differences between these and previous investigations are a relatively high level of minimum damping for satisfactory pilot ratings and, for constant damping, relative insensitivity of pilot rating to large increases in control sensitivity. The attitude response per unit control input is dominated by the parameters M_q and $M_{\delta_{ES}}$ despite the high speed stability. The Level 1 response boundary is plotted on Figure 18 (3.2.3.2) for comparison purposes. The desire for higher damping at high levels of speed stability is attributable to the increased gust sensitivity. Figure 19 (3.2.3.2) indicates that at constant control sensitivity, the minimum satisfactory damping level reduces markedly with decreased $M_u g$.

The high control sensitivities found satisfactory in this experiment are probably attributable to a combination of minimizing the stick deflection required for trimming in steady winds and maintaining reasonable controllability in the presence of high gust sensitivity due to $M_u g$.

The bulk of the data from which the present yaw response requirement was derived comes from Reference 48 in which hover experiments were performed in a small variable stability helicopter using simulated turbulence and steady winds. For four values of N_v, the control sensitivity damping plane was explored and pilot ratings obtained. Figures 20 (3.2.3.2) to 23 (3.2.3.2) present the results of these experiments. The primary effect of weathercock stability, N_v , appears to be an increase in the level of minimum damping required for satisfactory handling qualities. This is probably due to the increased gust sensitivity at high levels of N_v .

Reference 34 describes a moving-base simulator experiment in which VFR hovering and low speed tasks were performed. The evaluation flights consisted of hovering in steady winds and turbulence, flying square patterns, and hovering turns. For each dynamic configuration tested, the pilots selected the control sensitivities which they felt were most compatible with the mission requirements and vehicle dynamics. Figures 24 (3.2.3.2) through 27 (3.2.3.2) illustrate the correlation of the pilot selected control sensitivities with the specification Level 1 requirements for the pitch and roll axes. The data points and their corresponding Level 1 boundaries have been grouped according to the value of λ . The boundaries are shown as double lines corresponding to the minimum and maximum values of damping ratio, ζ , in each set of points. It is observed that the variation in selected sensitivities between pilots may vary by 4 or 5 to 1, while a given pilot may vary his selection by as much as 2 to 1.

A fixed-base ground simulator was utilized in the experiments described in Reference 35. Figure 28 (3.2.3.2) compares the pilot-selected control sensitivities from this experiment with the specification minimum Level 1 pitch axis boundaries. In relation to the previous data, the pilots of this simulation have tended to select sensitivities close to the minimum.

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In the simulations of Reference 34, a series of tests was conducted of configurations having predominantly attitude type control systems with $\zeta = 0.7$. In these tests the pilots were asked to rate given dynamic configurations with sensitivities ranging from 0.3 to 6.075 radians/sec^2-inch. At the highest sensitivities, control was extremely difficult and generally achieved ratings on the Cooper-Harper scale of 8 or greater as shown in Figures 29 (3.2.3.2) and 30 (3.2.3.2). These configurations are excluded by the Level 3 boundaries.

Little systematic data exist for configurations with natural frequencies above 3 or 4 radians per second. However, one recent source of such data is contained in Reference 36. Contrary to the results of References 63 and 64, the pilot-selected control sensitivities at high frequencies (larger than 3 radians/second) do not tend to produce constant attitude response per inch of stick but rather tend to continue to follow lines of constant control sensitivity, as shown in Figure 31 (3.2.3.2). However, pilot comments indicate that at low frequencies, the pilots selected control sensitivity on the basis of maneuverability while at higher frequencies the trim characteristics determined their selection. There are no comments regarding the loss in maneuverability but for the highest frequencies shown in Figure 31 (3.2.3.2) the steady-state attitude response per inch of stick is of the order of one or two degrees. Until further substantiating data is available, it is considered that no change in the specification is warranted to accommodate the above results.

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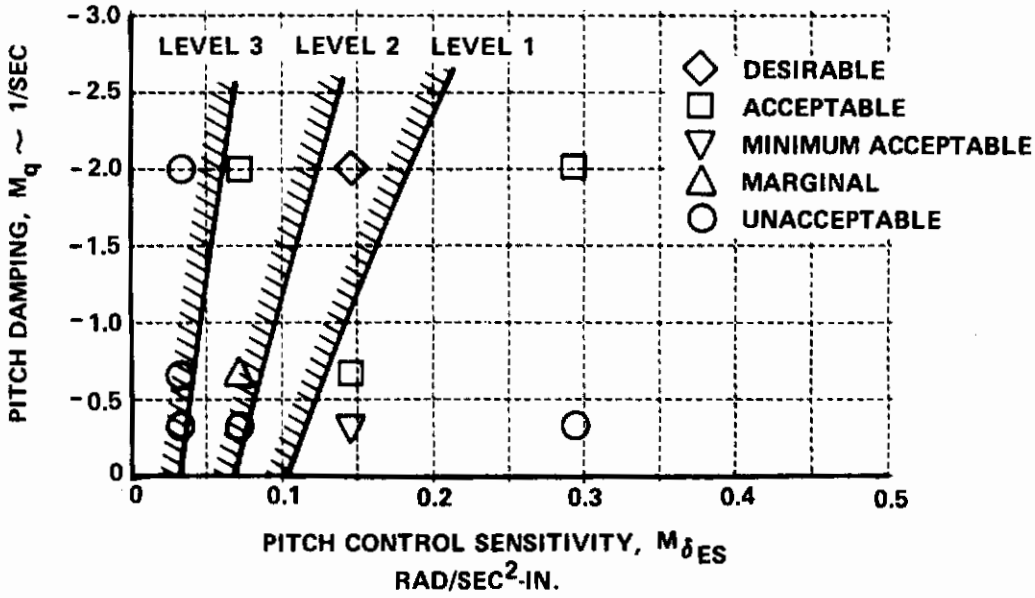

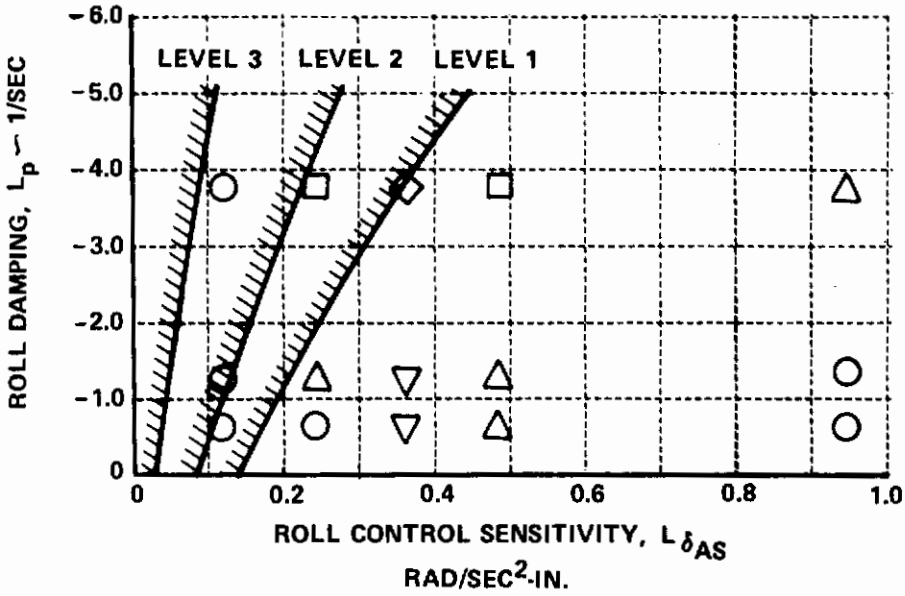

(DATA FROM REFERENCE 61)

Figure 1 (3.2.3.2) QUALITATIVE PILOT RATINGS OF HOVER HANDLING QUALITIES, PITCH AND ROLL AXES

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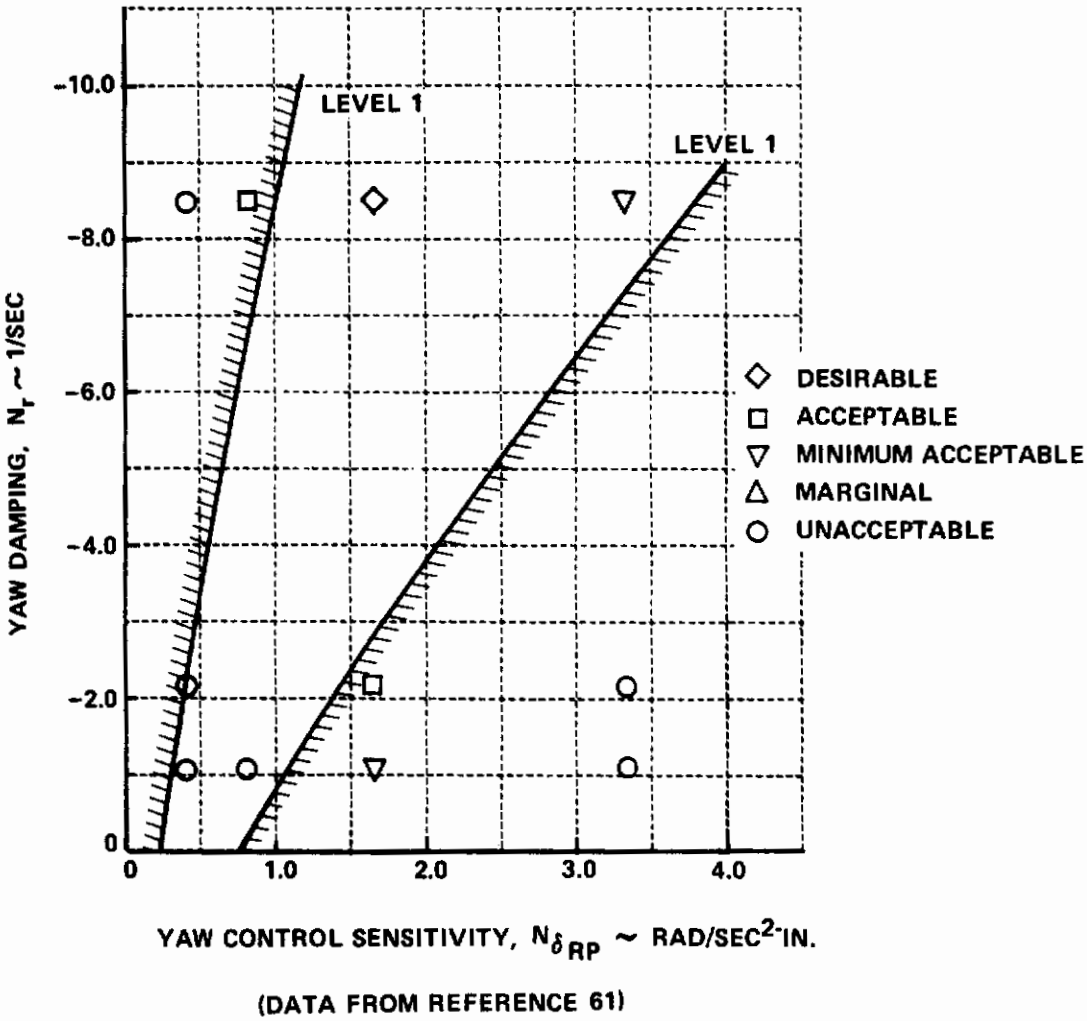

Figure 2 (3.2.3.2) QUALITATIVE PILOT RATINGS OF HOVER HANDLING QUALITIES, YAW AXIS

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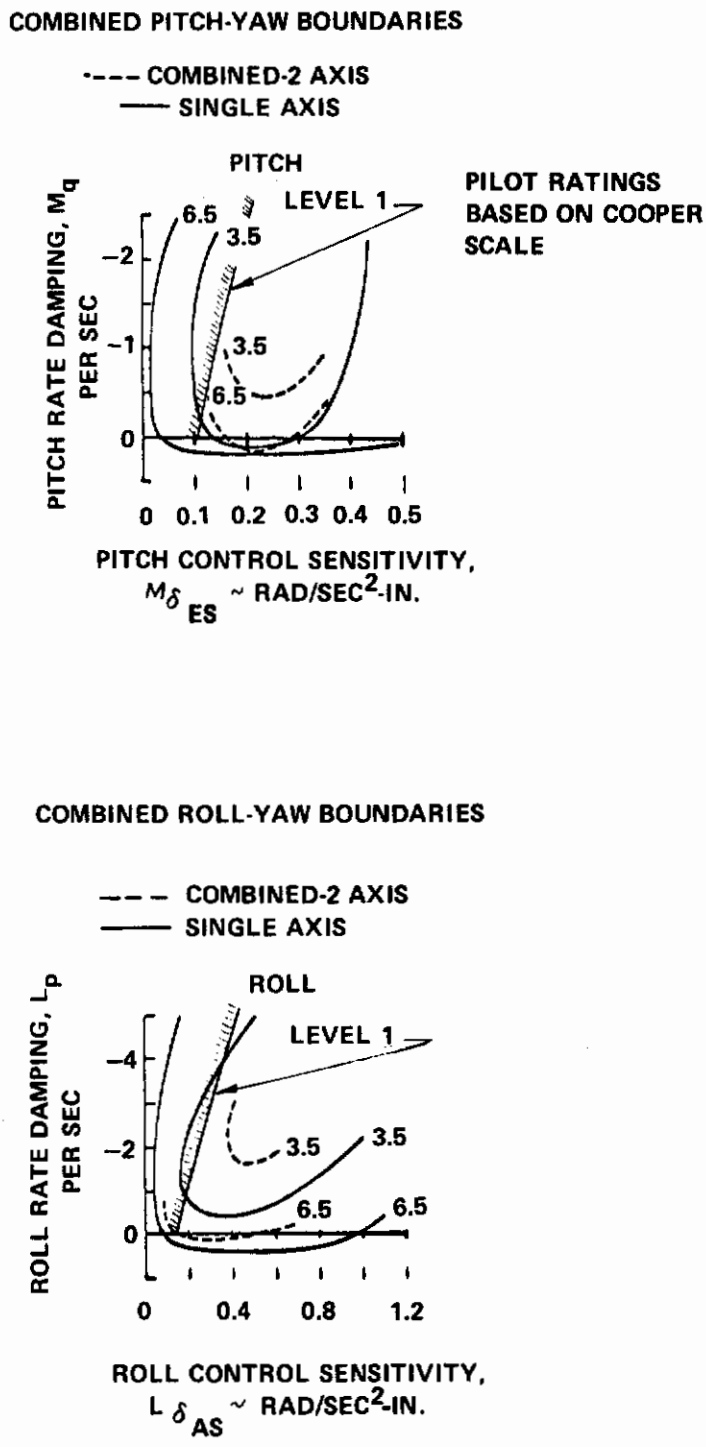

Figure 3 (3.2.3.2) PILOT RATING BOUNDARIES FOR HOVER SIMULATION, PITCH AND ROLL AXES (DATA FROM REFERENCE 38)

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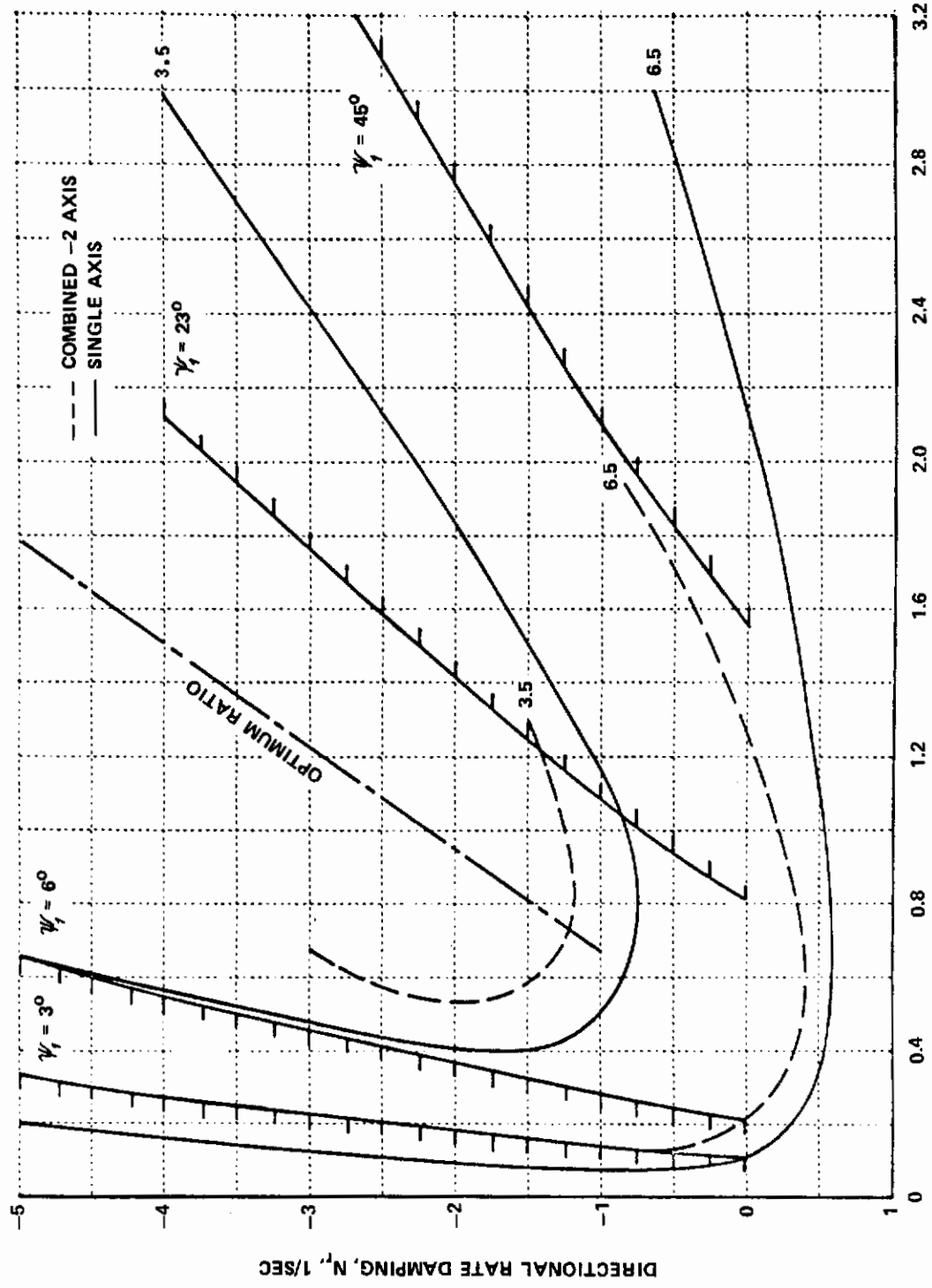

Figure 4 (3.2.3.2) YAW SENSITIVITY-DAMPING BOUNDARIES, COOPER SCALE
(DATA FROM REFERENCE 38)

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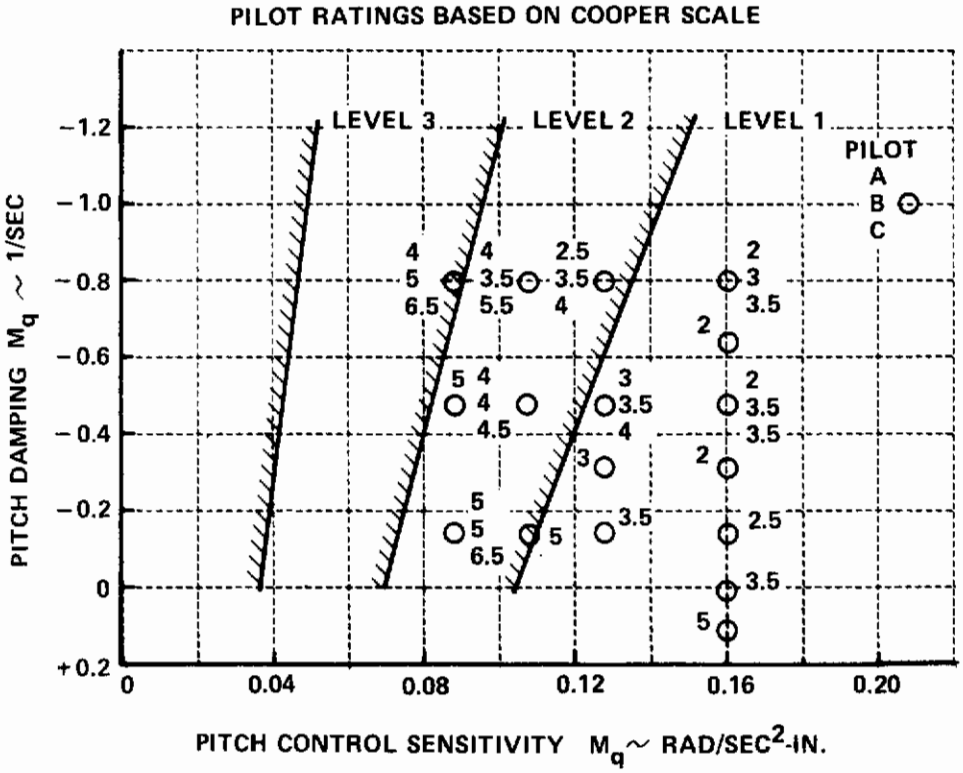

Figure 5 (3.2.3.2) PILOT RATINGS OF PITCH CONTROL SENSITIVITY AND DAMPING (BASED ON DATA OF REFERENCE 39)

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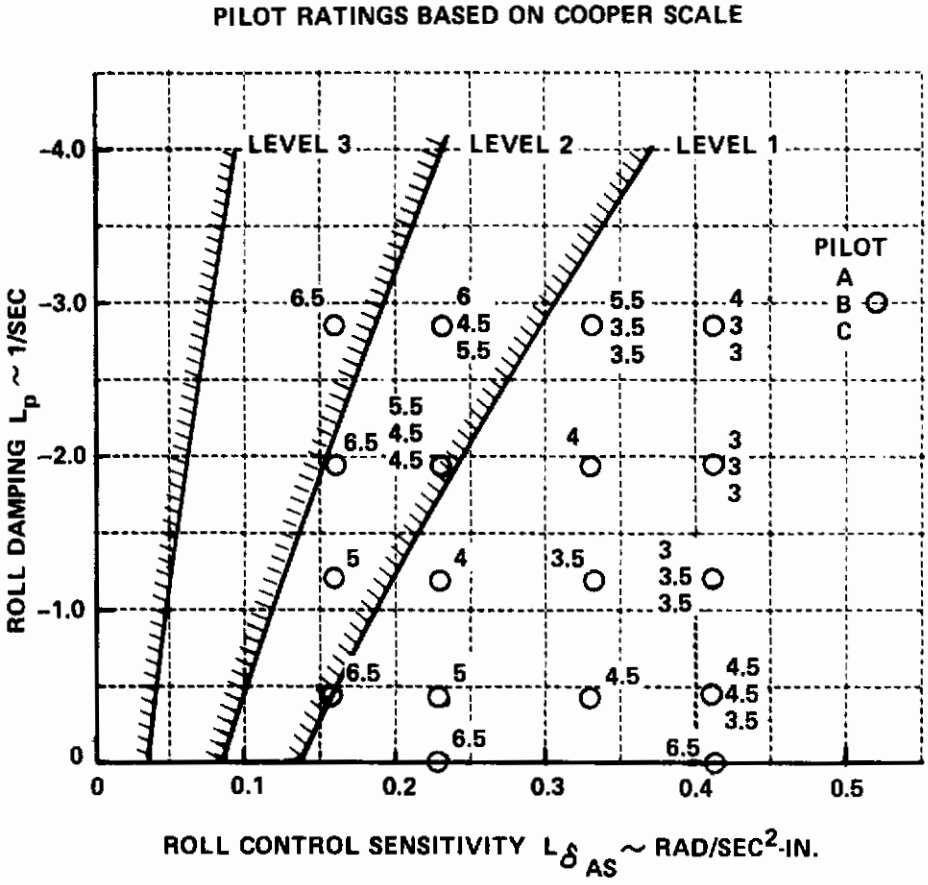

Figure 6 (3.2.3.2) PILOT RATINGS OF ROLL CONTROL SENSITIVITY AND DAMPING (BASED ON DATA OF REFERENCE 39)

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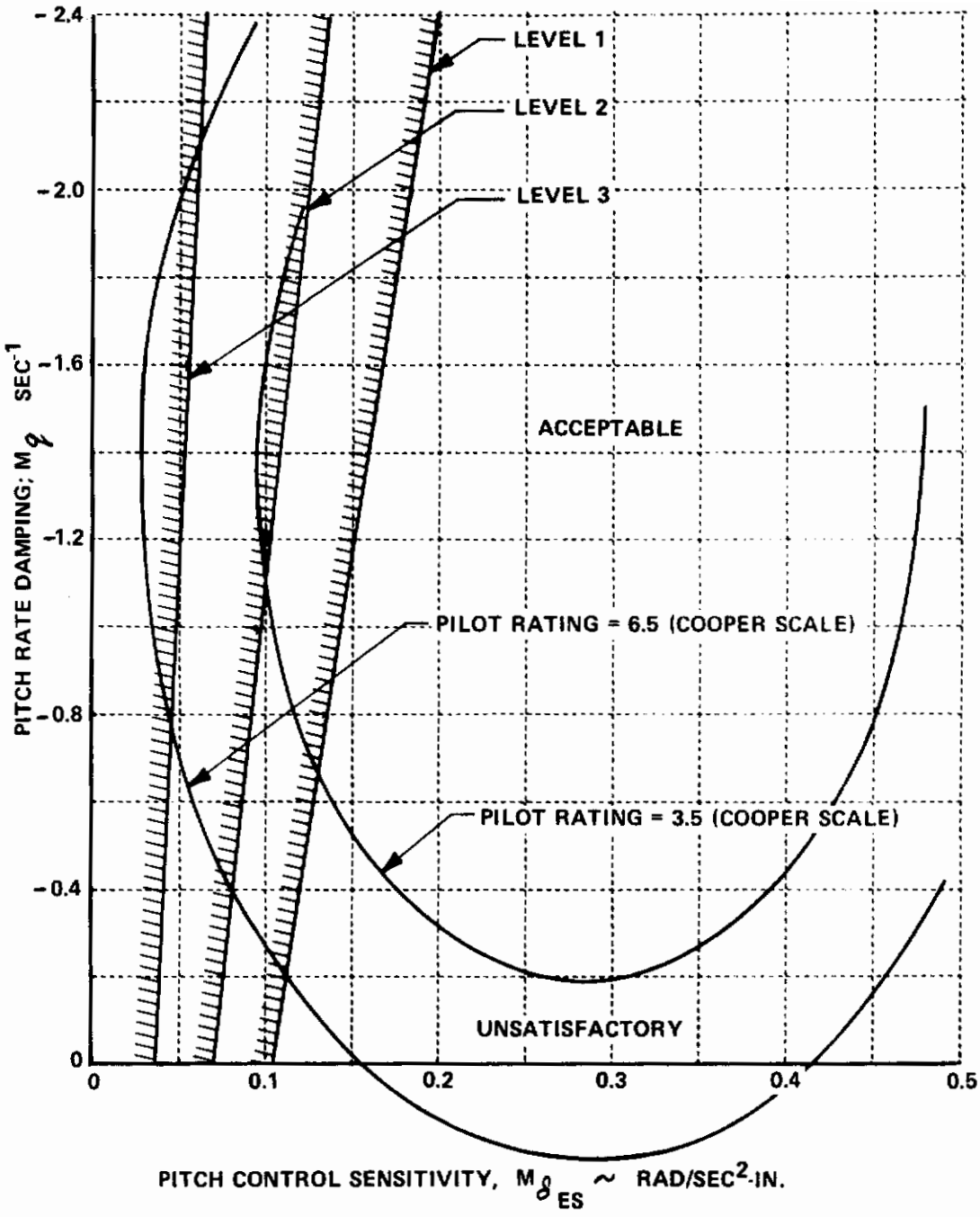

Figure 7 (3.2.3.2) BASIC DAMPING AND CONTROL SENSITIVITY BOUNDARIES (BASED ON DATA OF REFERENCE 60)

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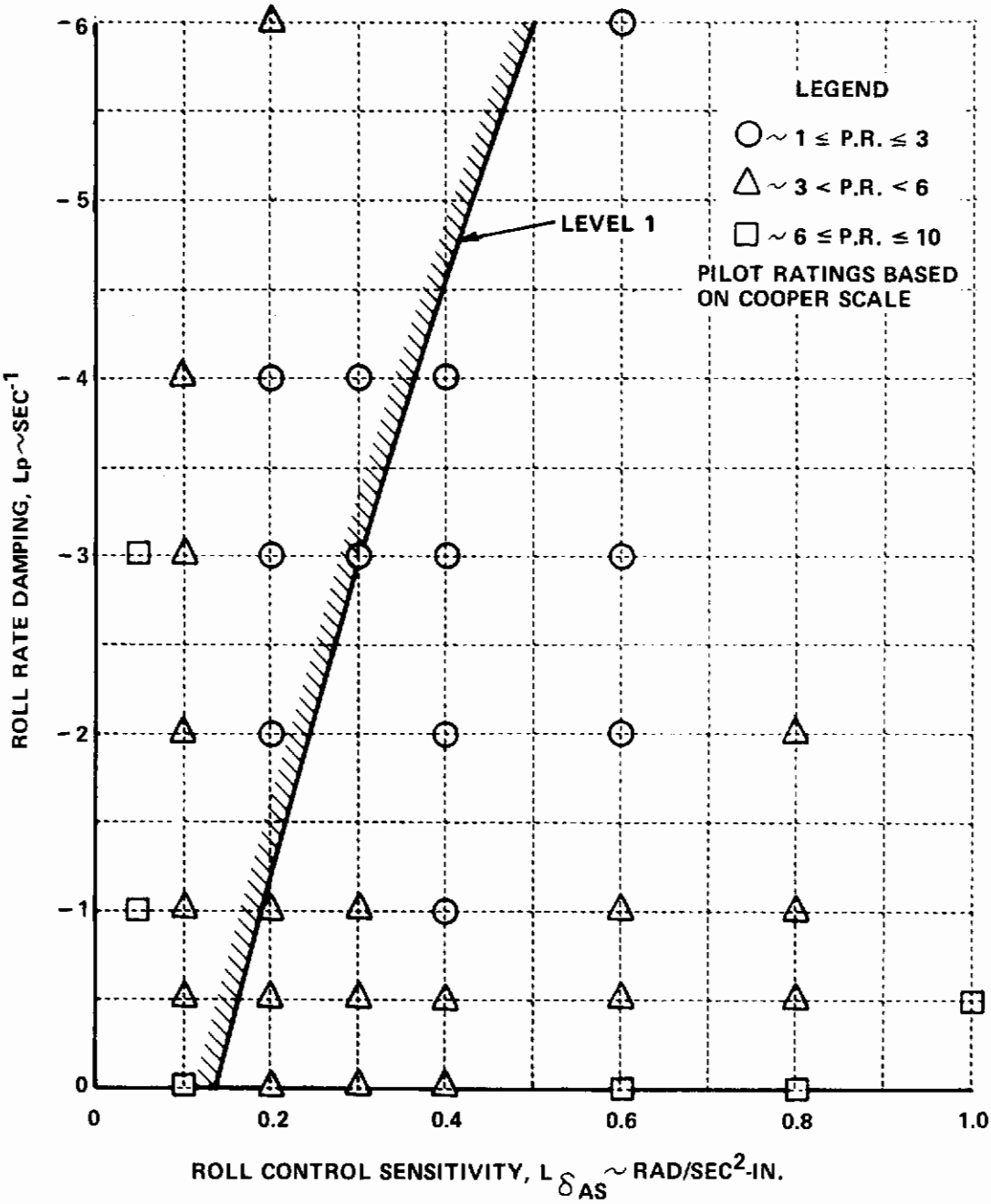

Figure 8 (3.2.3.2) BASIC DAMPING AND CONTROL SENSITIVITY BOUNDARIES (BASED ON DATA OF REFERENCE 60)

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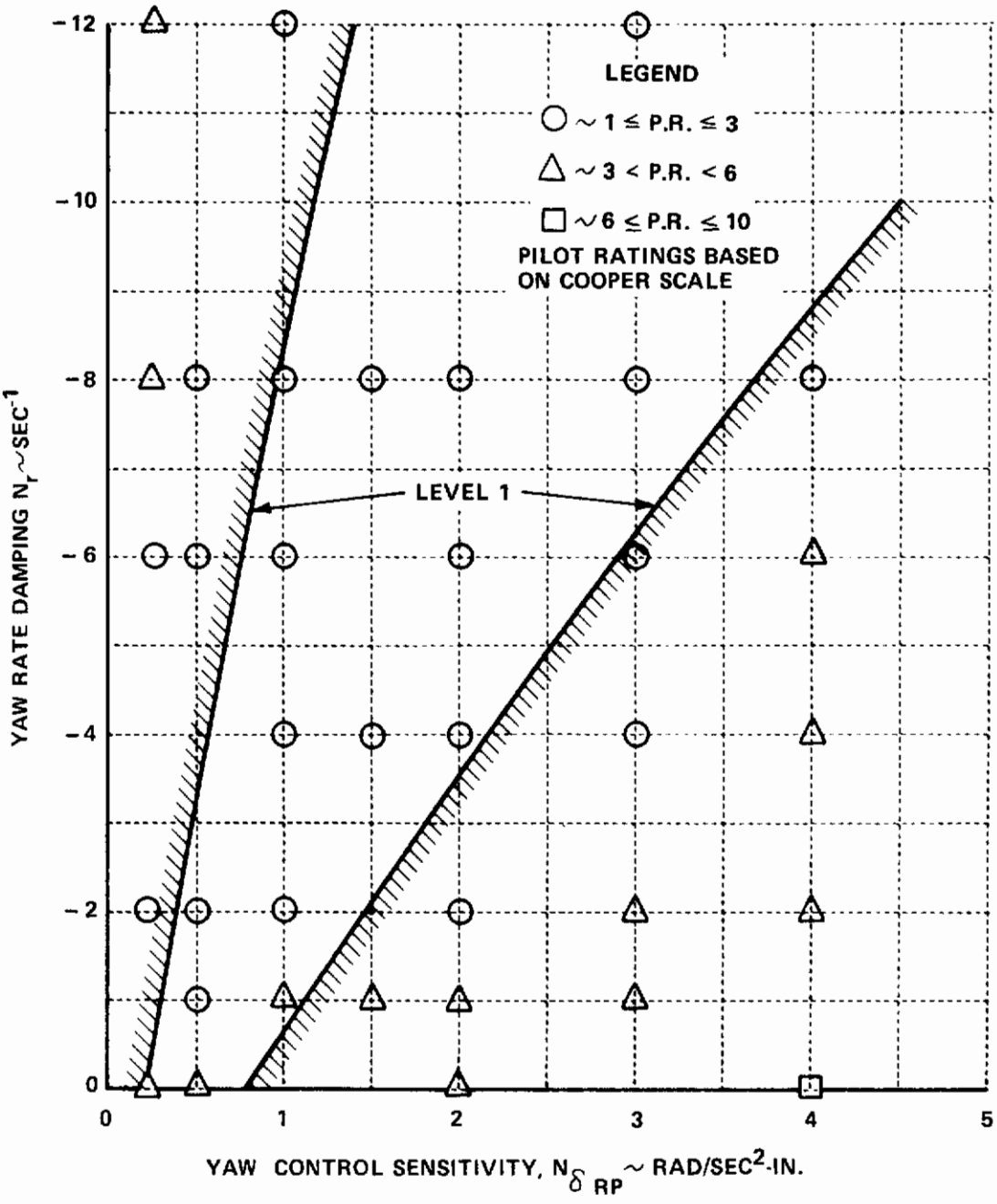

DIRECTIONAL HANDLING QUALITIES AT HOVER

Figure 9 (3.2.3.2) BASIC DAMPING AND CONTROL SENSITIVITY BOUNDARIES (BASED ON DATA OF REFERENCE 60)

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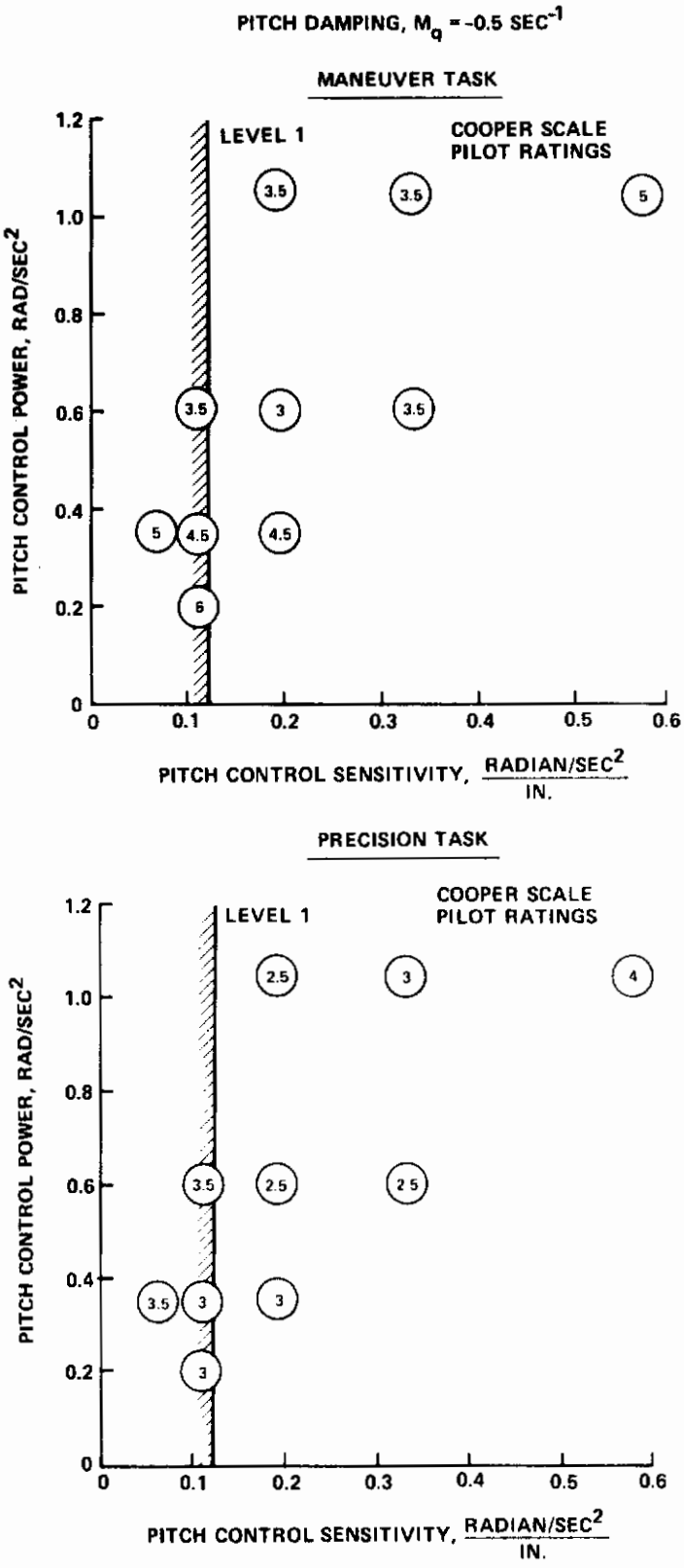

Figure 10 (3.2.3.2) CONTROL SENSITIVITY-POWER REQUIREMENTS FOR PRECISION AND MANEUVER TASKS - PITCH AXIS (BASED ON DATA OF REFERENCE 54)

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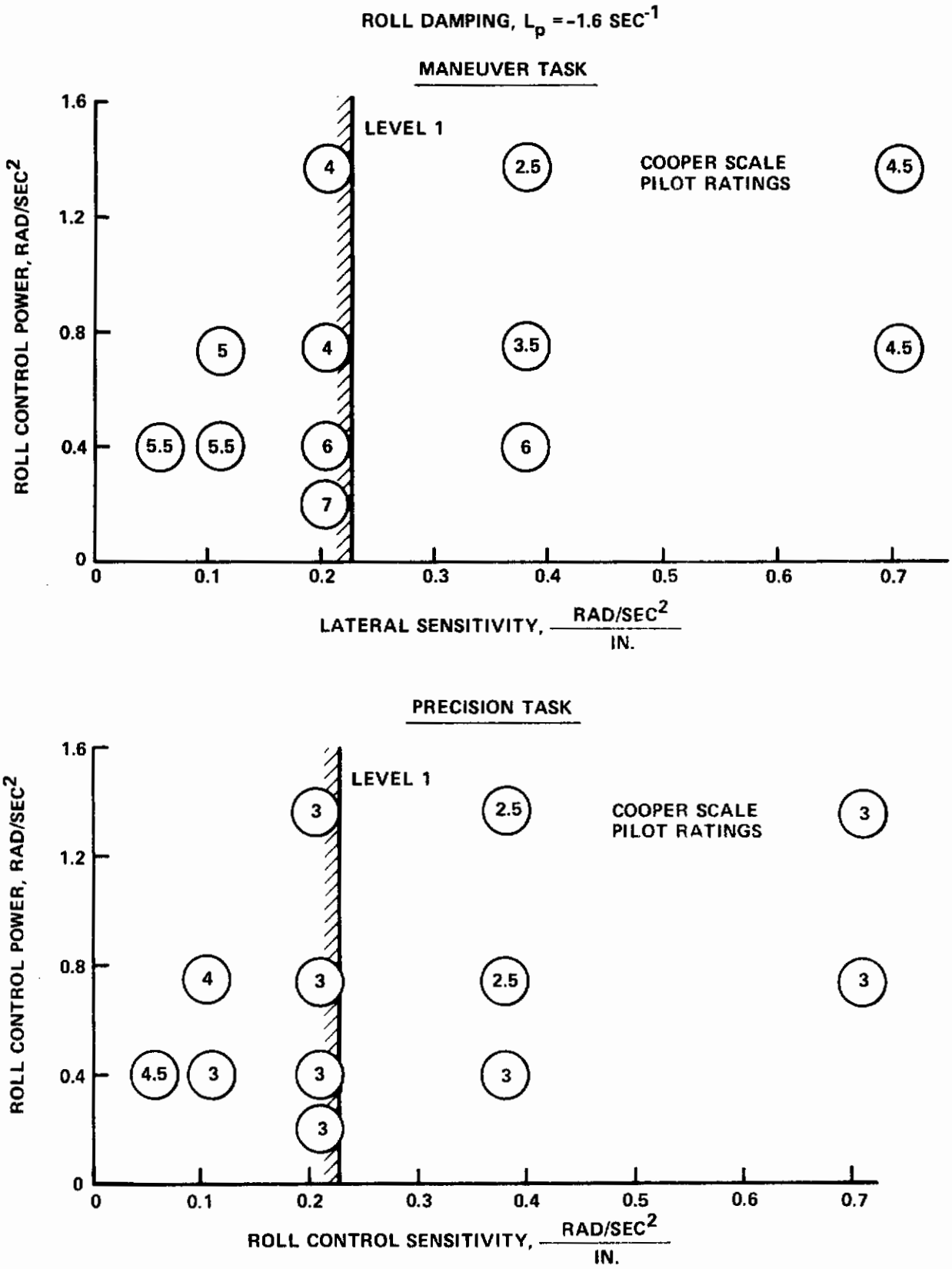

Figure 11 (3.2.3.2) CONTROL SENSITIVITY - POWER REQUIREMENTS FOR PRECISION AND MANEUVER TASKS - ROLL AXIS (BASED ON DATA OF REFERENCE 54)

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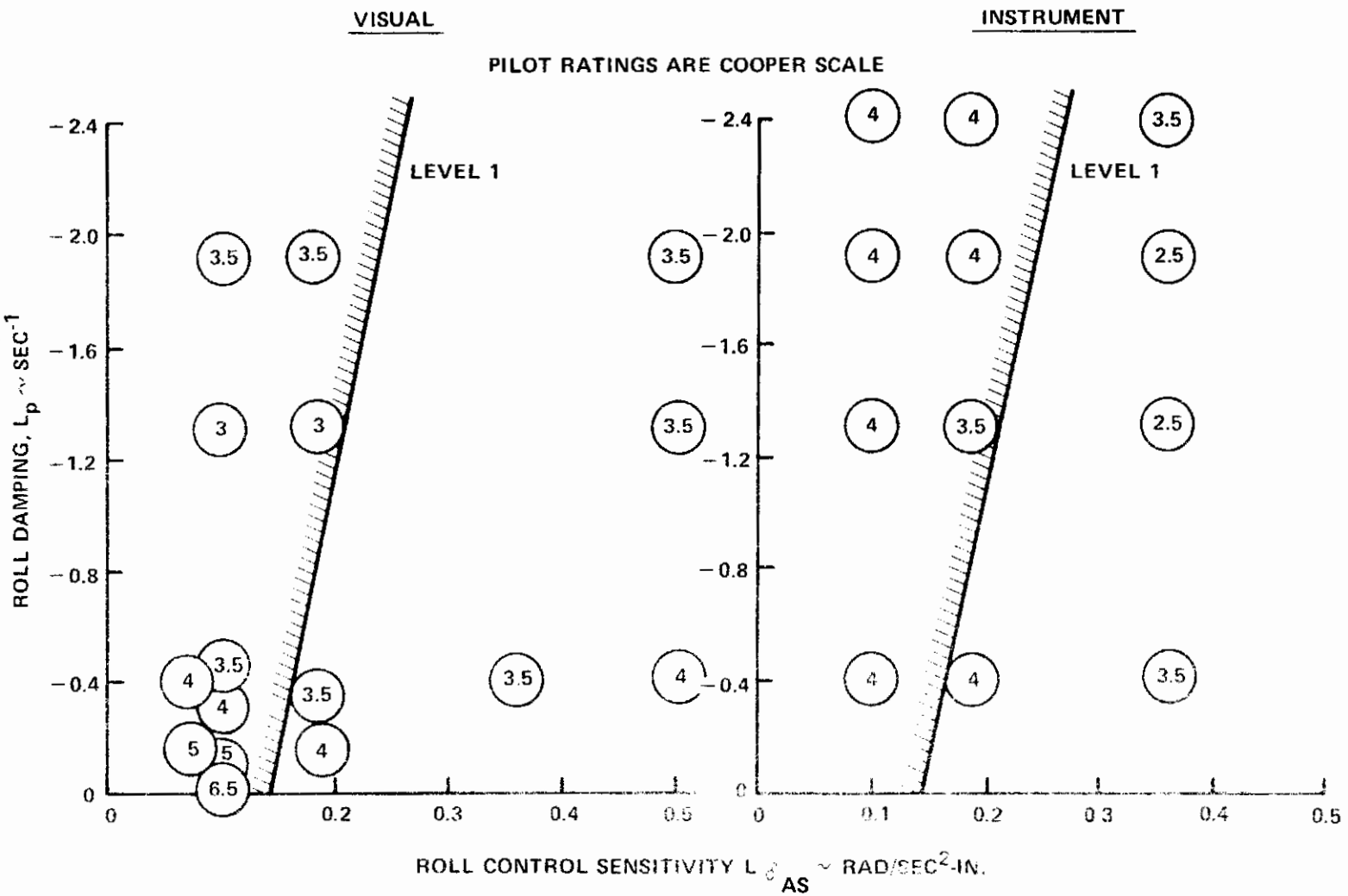

Figure 12 (3.2.3.2) PILOT OPINION RATINGS FOR CONTROL SENSITIVITY - DAMPING COMBINATIONS - ROLL AXIS (DATA FROM REFERENCE 62)

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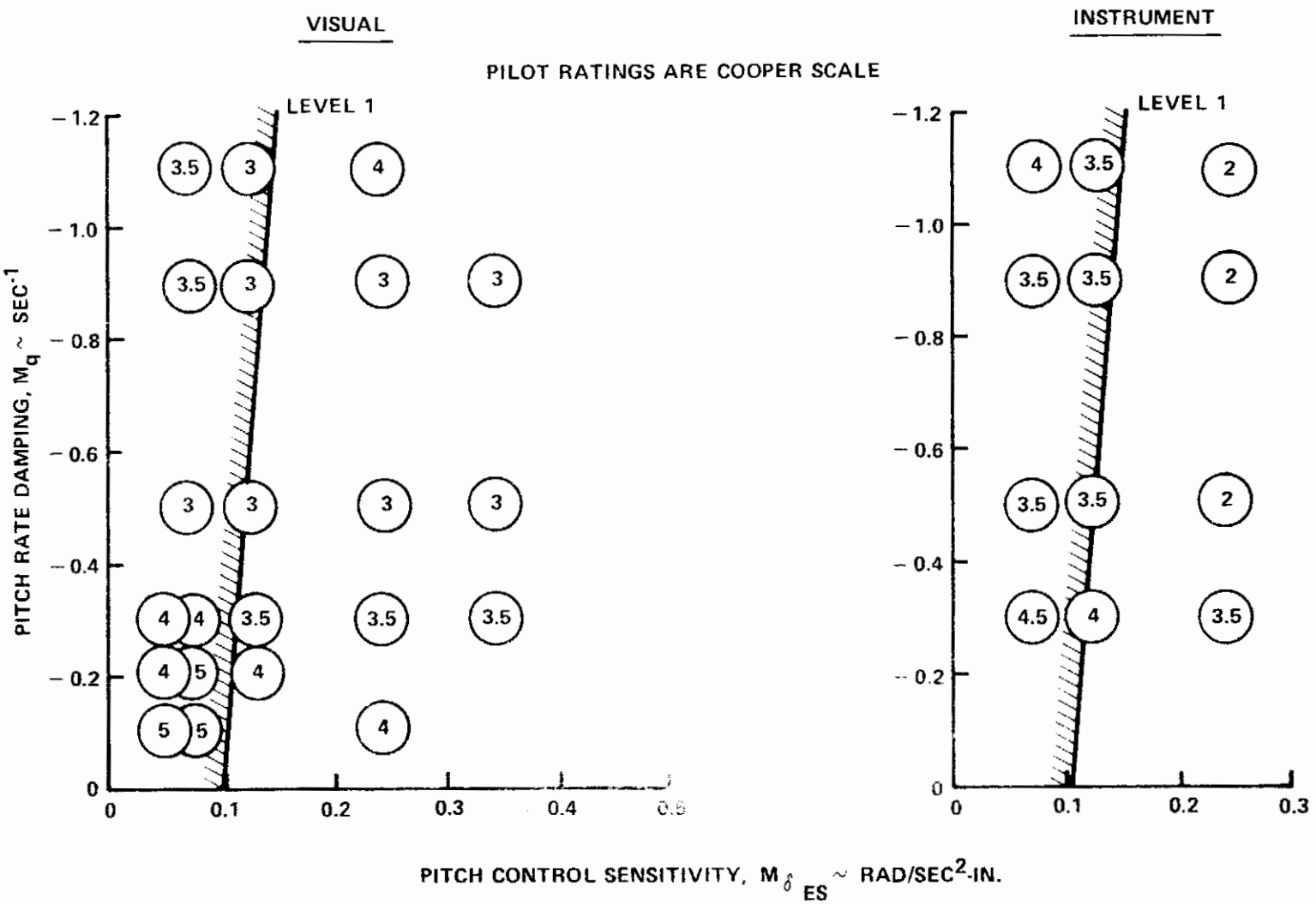

Figure 13 (3.2.3.2) PILOT OPINION RATINGS FOR CONTROL SENSITIVITY - DAMPING COMBINATIONS - PITCH AXIS (DATA FROM REFERENCE 62)

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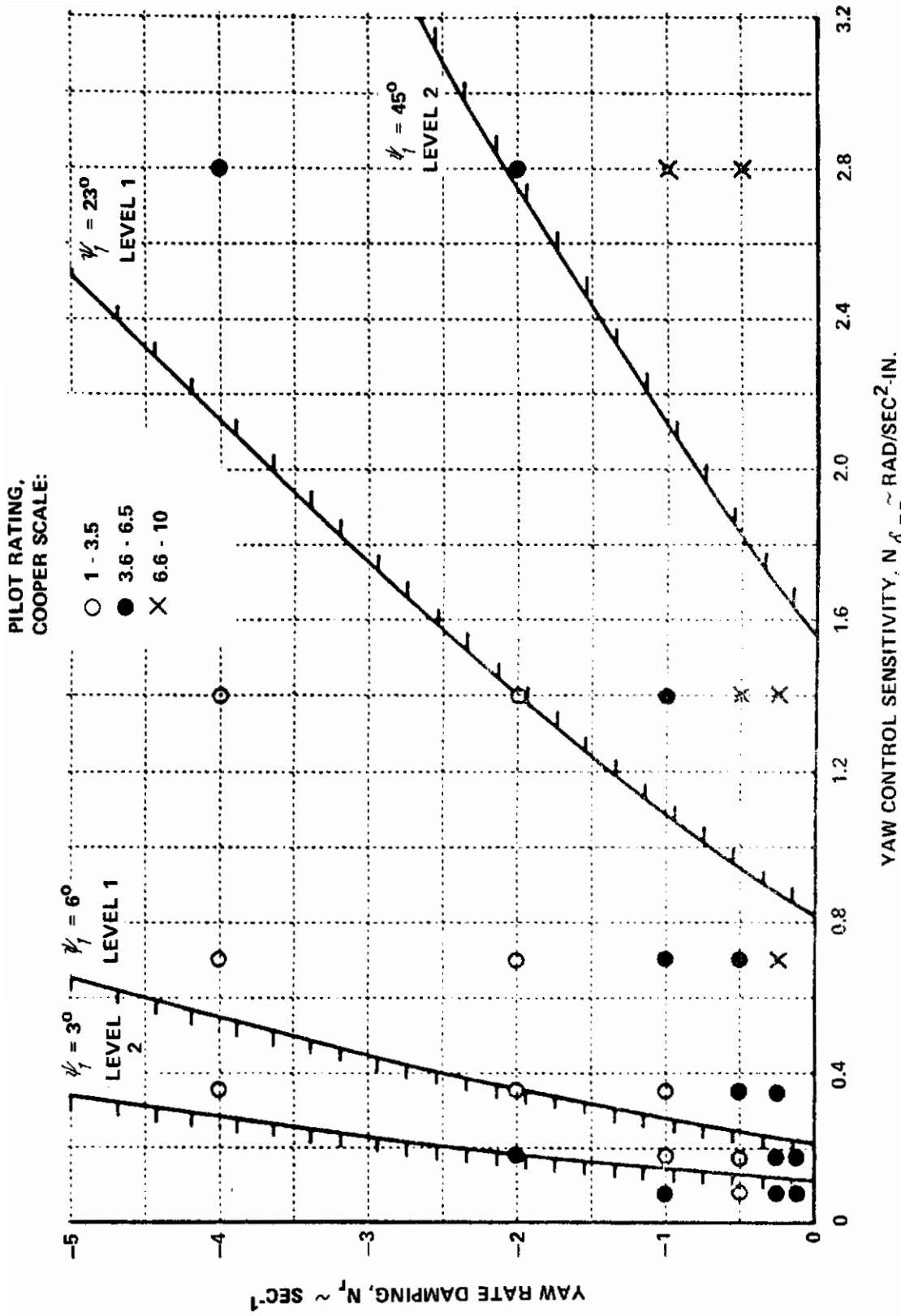

Figure 14 (3.2.3.2) YAW SENSITIVITY-DAMPING BOUNDARIES (DATA FROM REF. 49)

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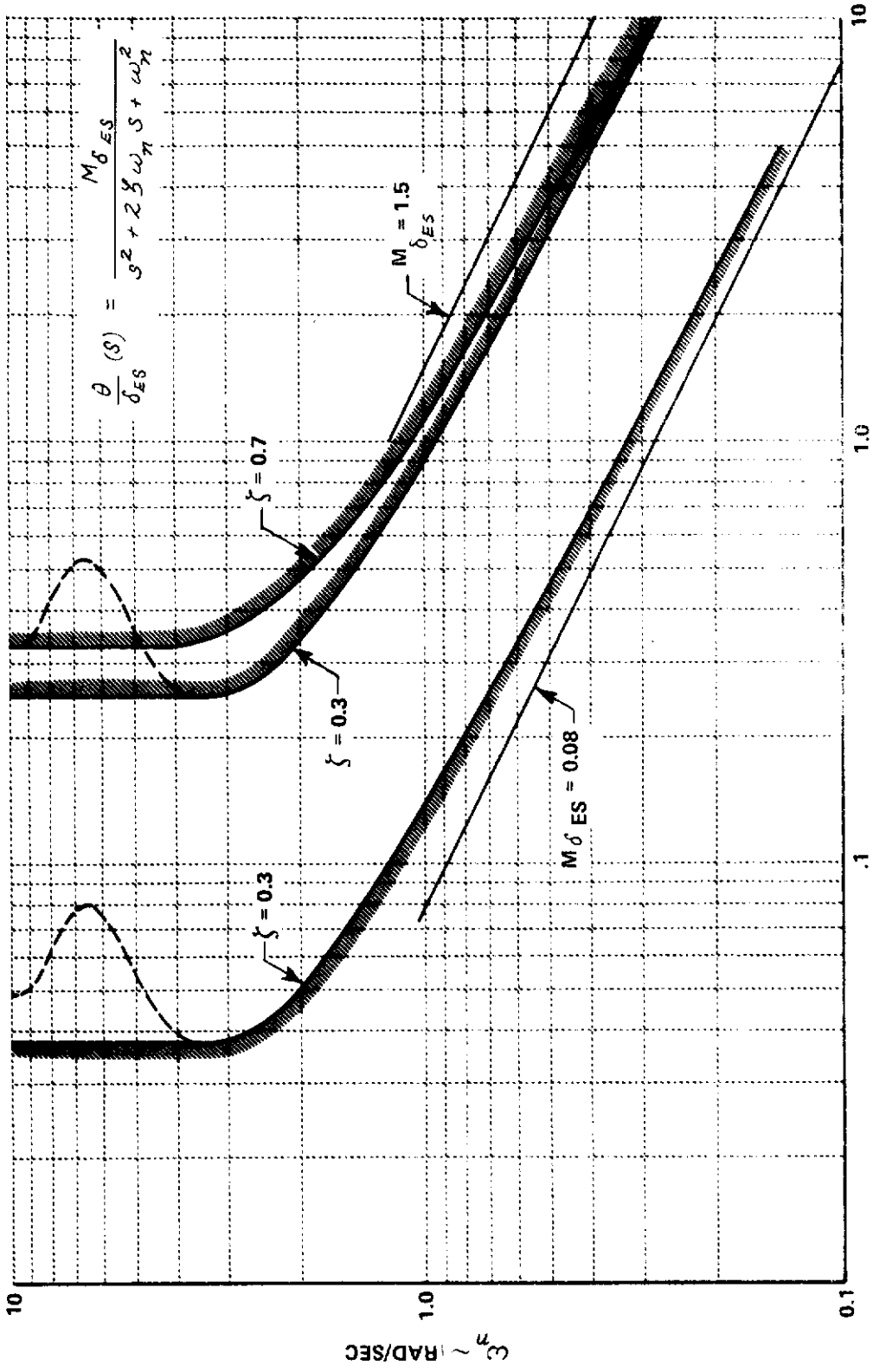

Figure 15 (3.2.3.2) LEVEL 1 PITCH ATTITUDE RESPONSE BOUNDARIES

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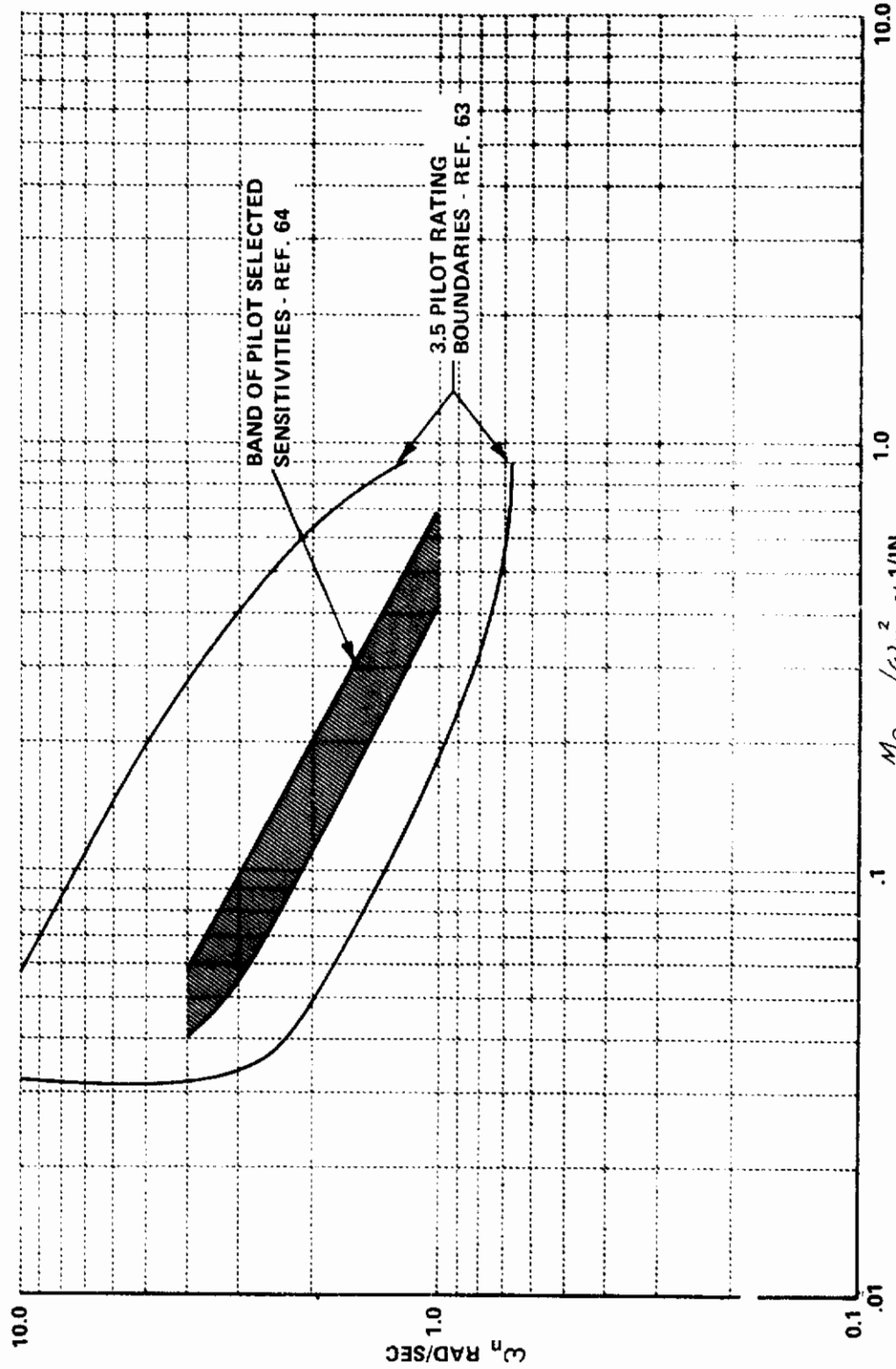

Figure 16 (3.2.3.2) PILOT PREFERRED CONTROL SENSITIVITY AS A FUNCTION OF NATURAL FREQUENCY (DATA FROM REFERENCES 63 AND 64)

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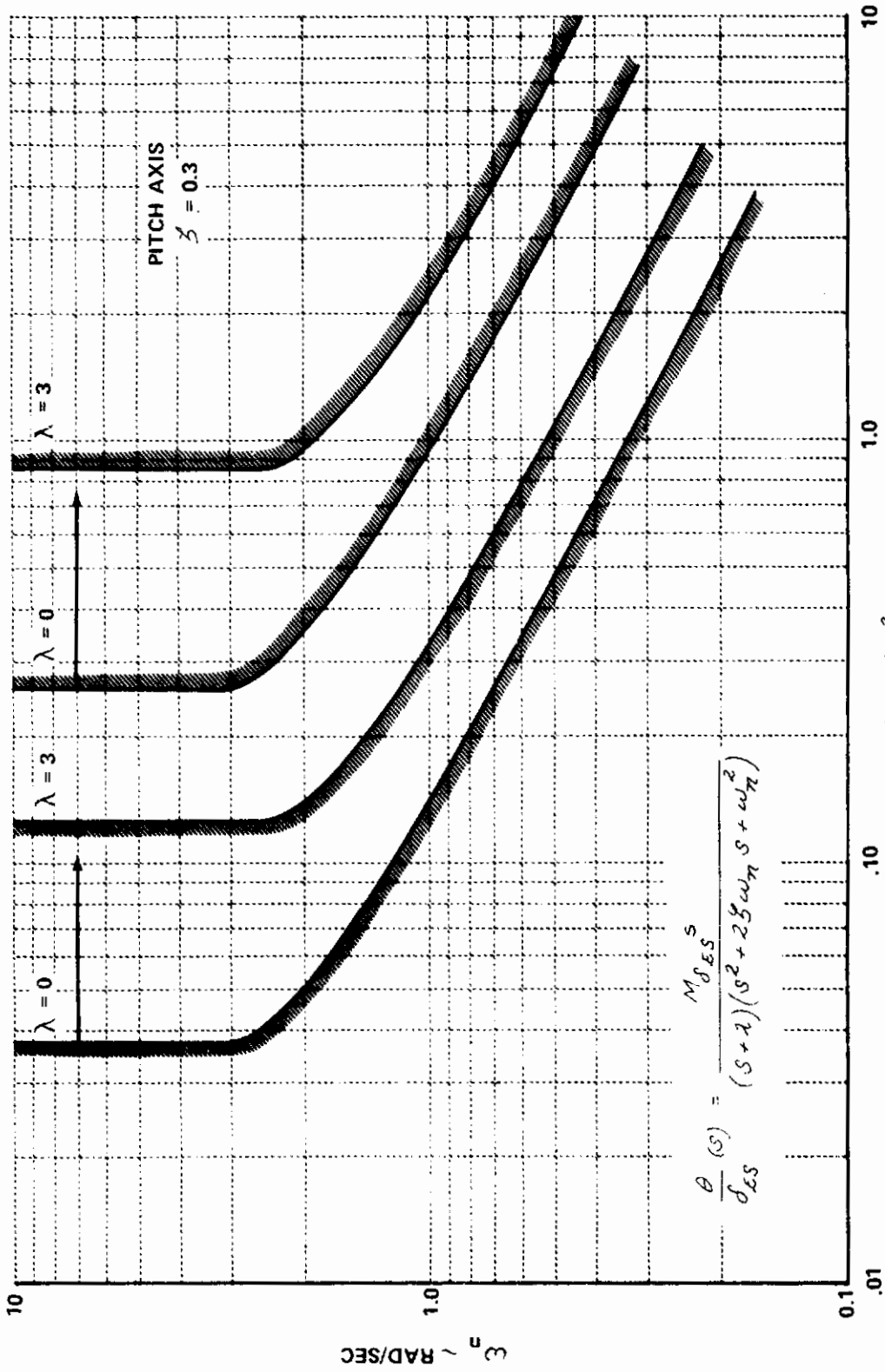

Figure 17 (3.2.3.2) INFLUENCE OF λ ON LEVEL 1 PITCH RESPONSE BOUNDARIES

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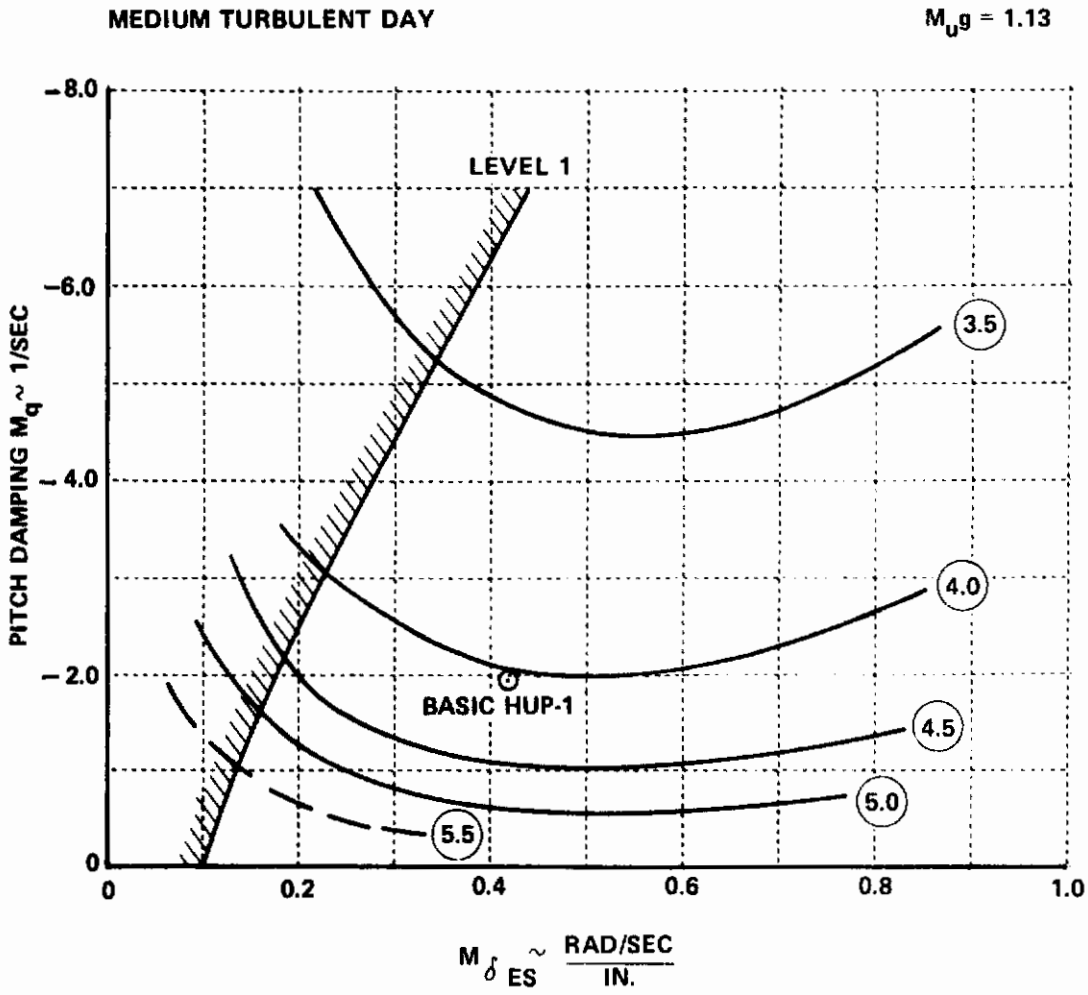

Figure 18 (3.2.3.2) HANDLING QUALITIES DATA FROM VARIABLE STABILITY HUP-1 TESTS (DATA FROM REFERENCE 40)

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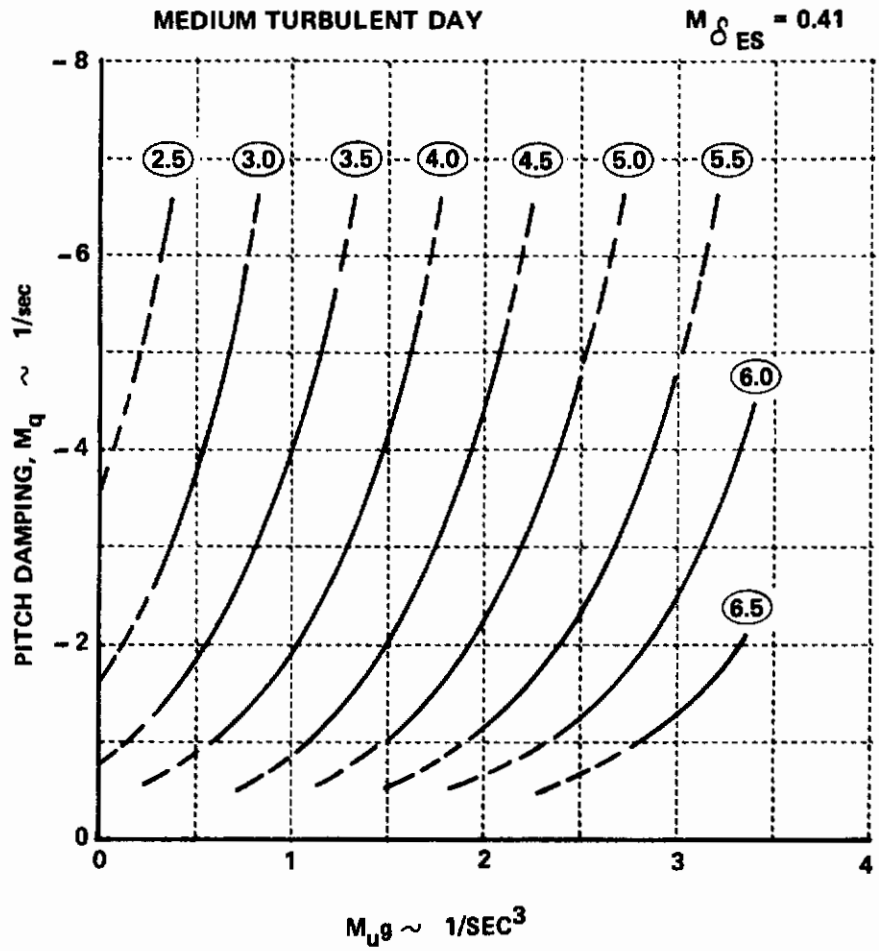

Figure 19 (3.2.3.2) INFLUENCE OF SPEED STABILITY ($M_u g$) ON DAMPING REQUIREMENTS (DATA FROM REFERENCE 40)

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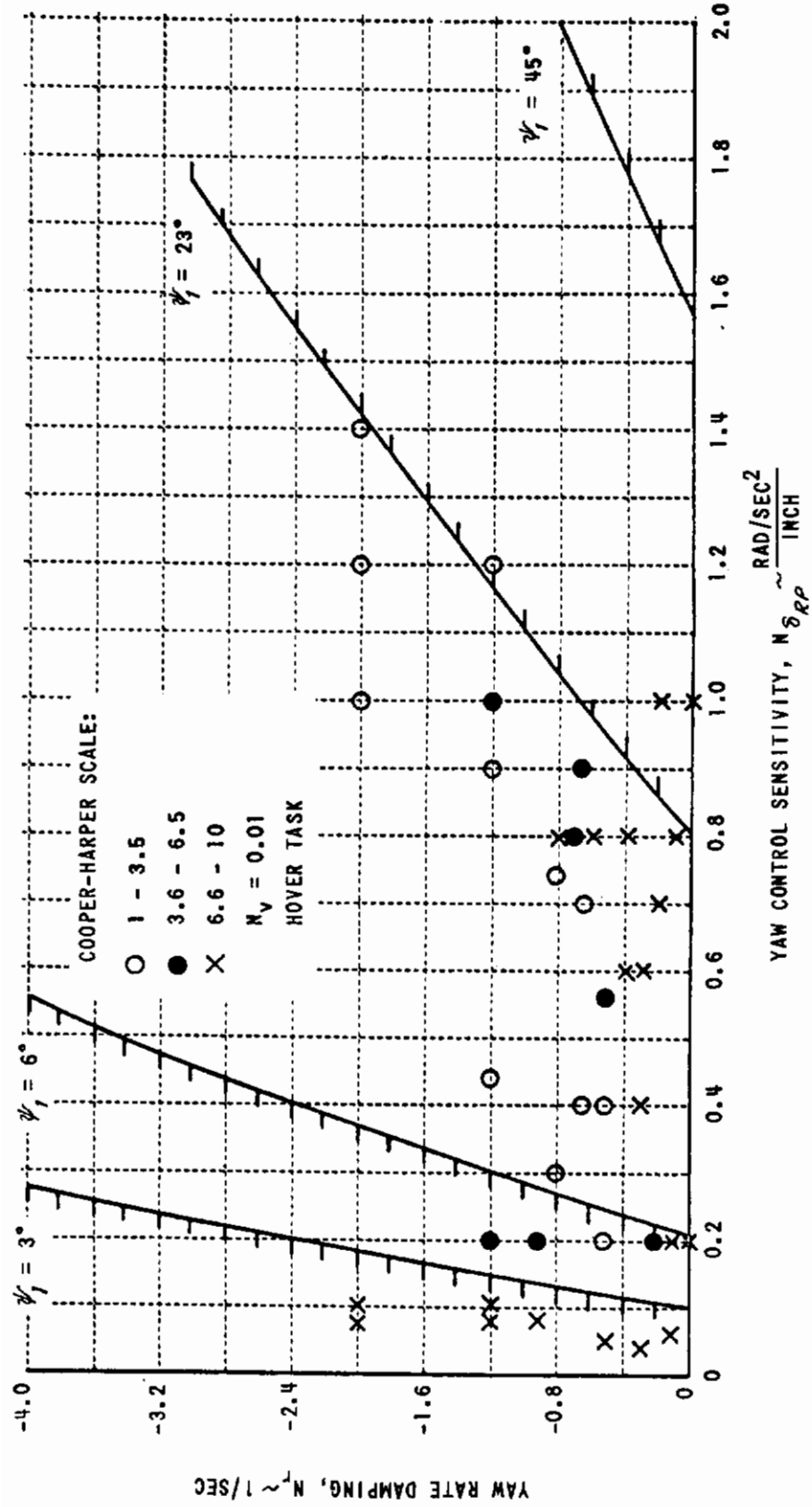

Figure 20(3.2.3.2) YAW SENSITIVITY-DAMPING BOUNDARIES (DATA FROM REF. 48)

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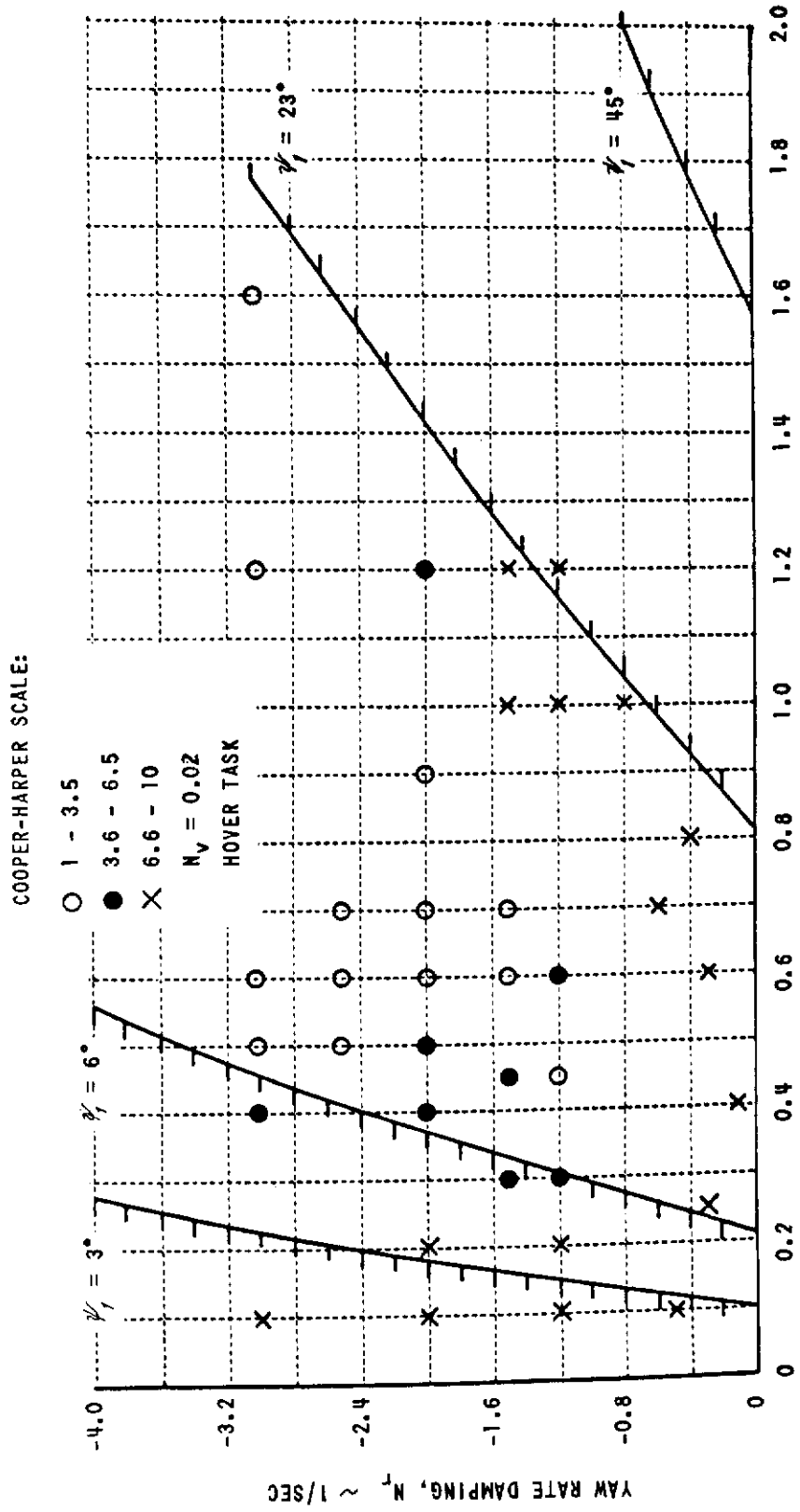

Figure 21(3.2.3.2) YAW SENSITIVITY-DAMPING BOUNDARIES (DATA FROM REF. 48)

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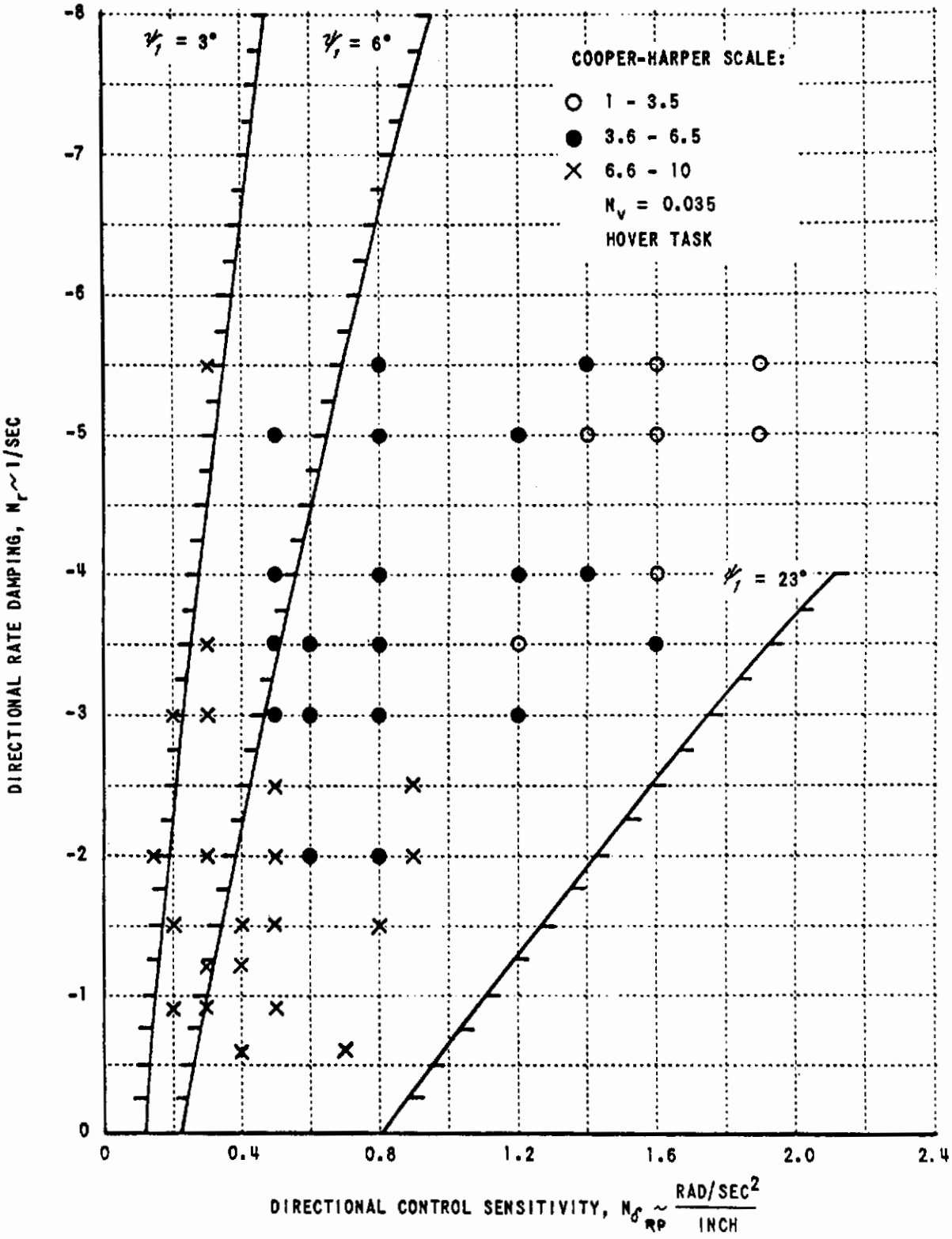

Figure 22(3.2.3.2) YAW SENSITIVITY-DAMPING BOUNDARIES (DATA FROM REF. 48)

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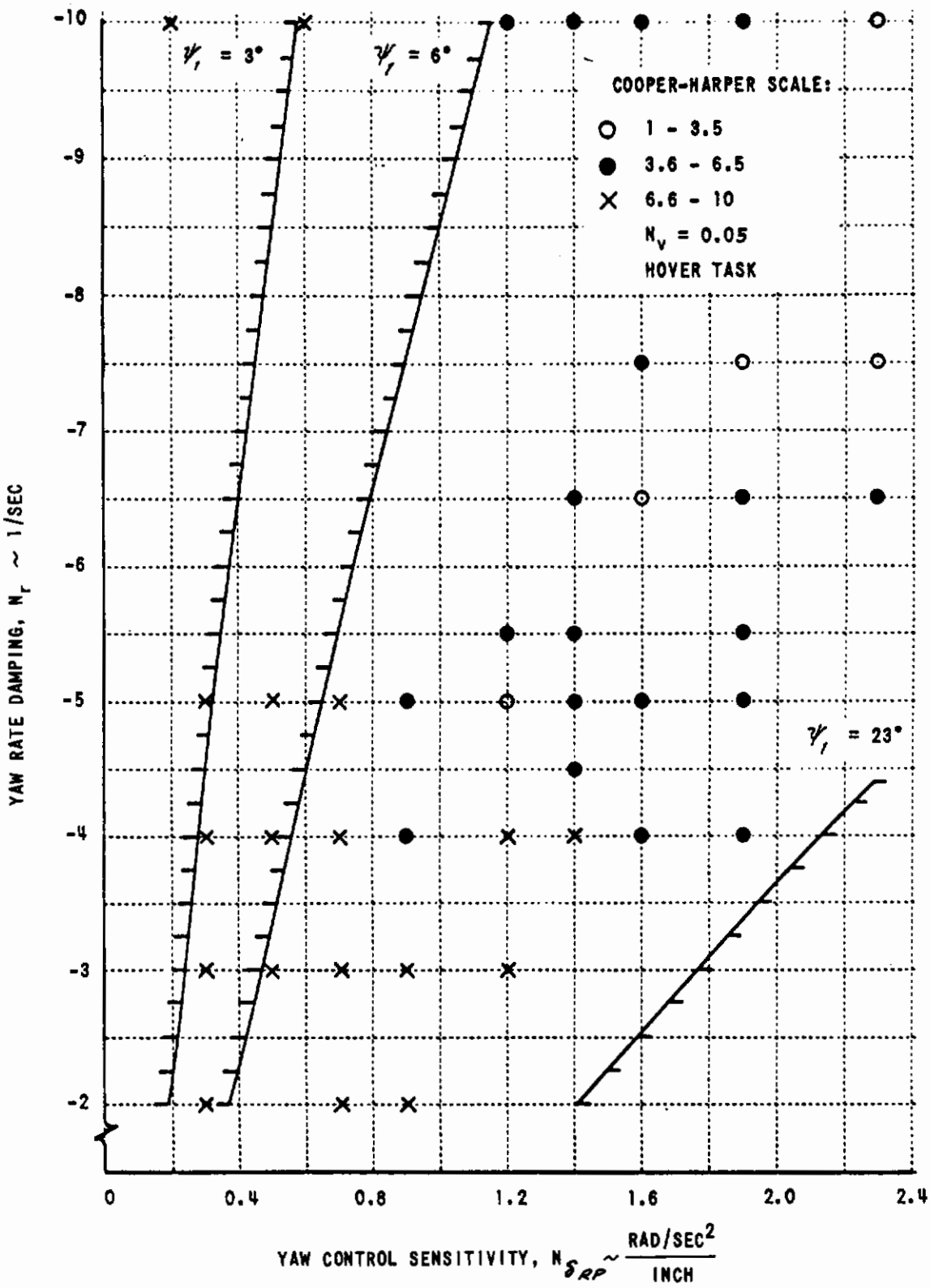

Figure 23(3.2.3.2) YAW SENSITIVITY-DAMPING BOUNDARIES (DATA FROM REF. 48)

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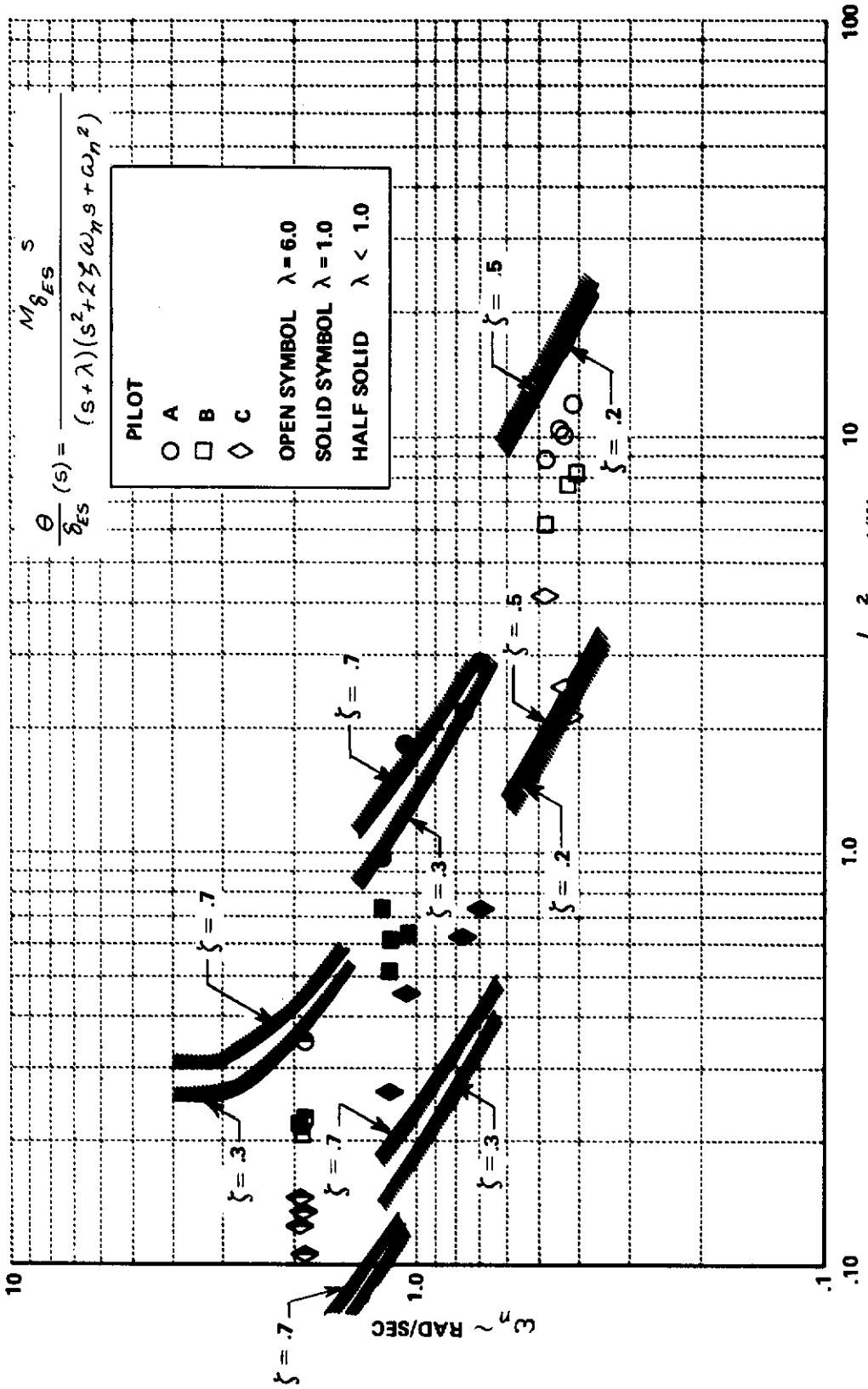

Figure 24(3.2.3.2) COMPARISON OF PILOT-SELECTED CONTROL SENSITIVITIES WITH LEVEL 1 PITCH AXIS BOUNDARIES (DATA FROM REF. 34)

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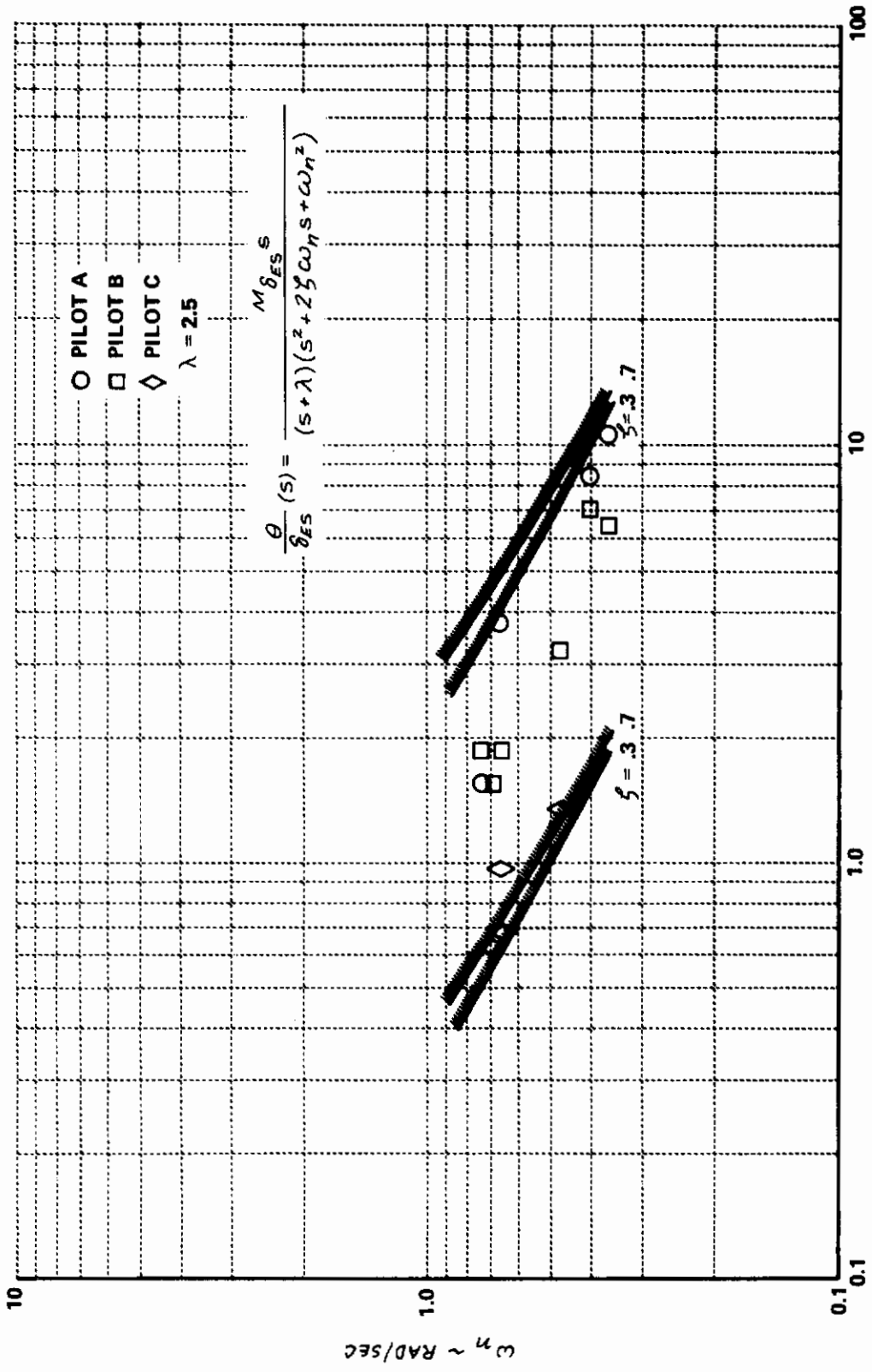

Figure 25(3.2.3.2) COMPARISON OF PILOT-SELECTED CONTROL SENSITIVITIES WITH LEVEL 1 PITCH AXIS BOUNDARIES (DATA FROM REF. 34)

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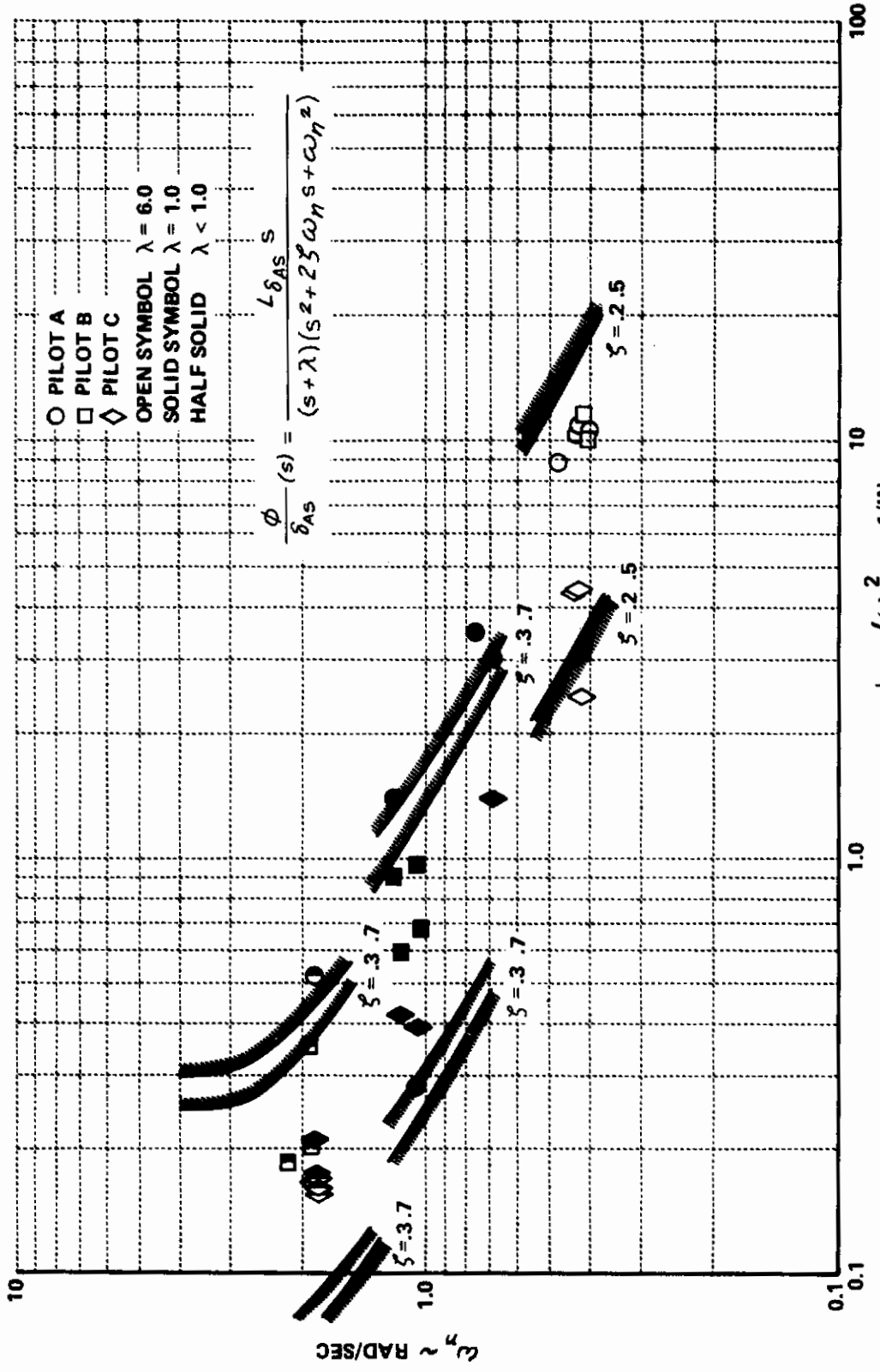

Figure 26(3.2.3.2) COMPARISON OF PILOT-SELECTED CONTROL SENSITIVITIES WITH LEVEL 1 ROLL AXIS BOUNDARIES (DATA FROM REF. 34)

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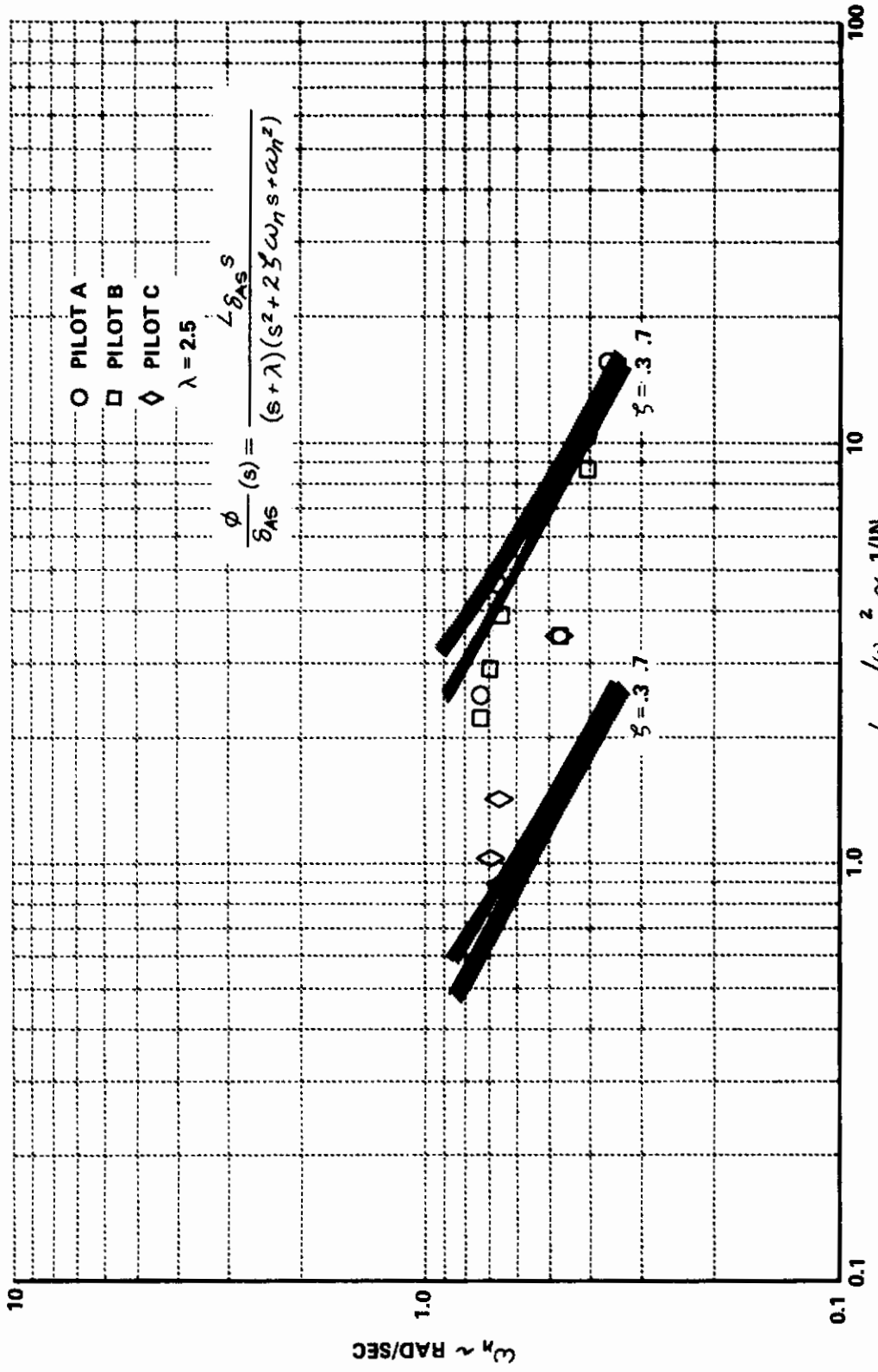

Figure 27(3.2.3.2) COMPARISON OF PILOT-SELECTED CONTROL SENSITIVITIES WITH LEVEL 1 ROLL AXIS BOUNDARIES (DATA FROM REF. 34)

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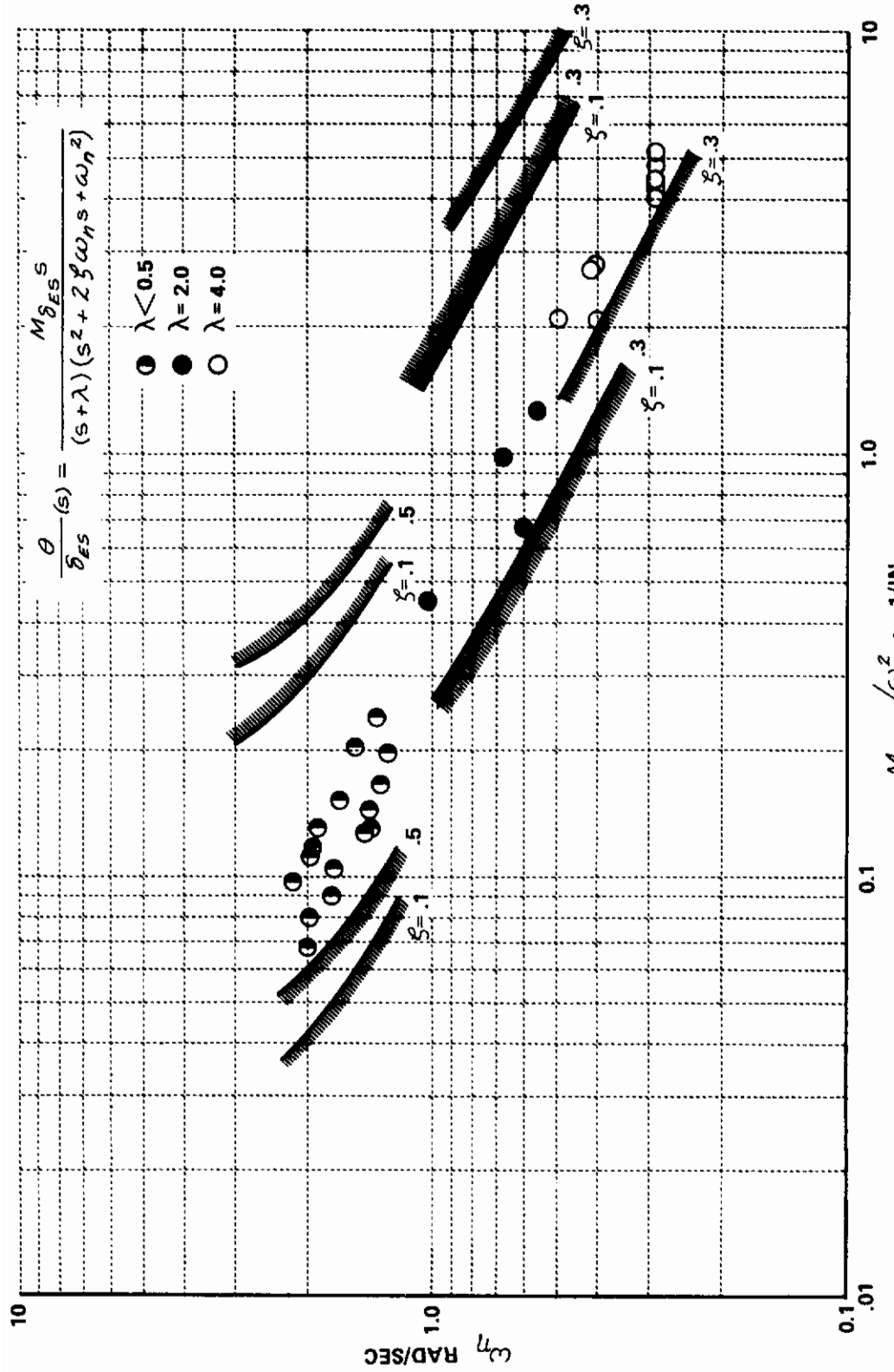

Figure 28(3.2.3.2) COMPARISON OF FIXED BASE SIMULATOR DATA WITH SPECIFICATION BOUNDARIES (BASED ON DATA OF REF. 35)

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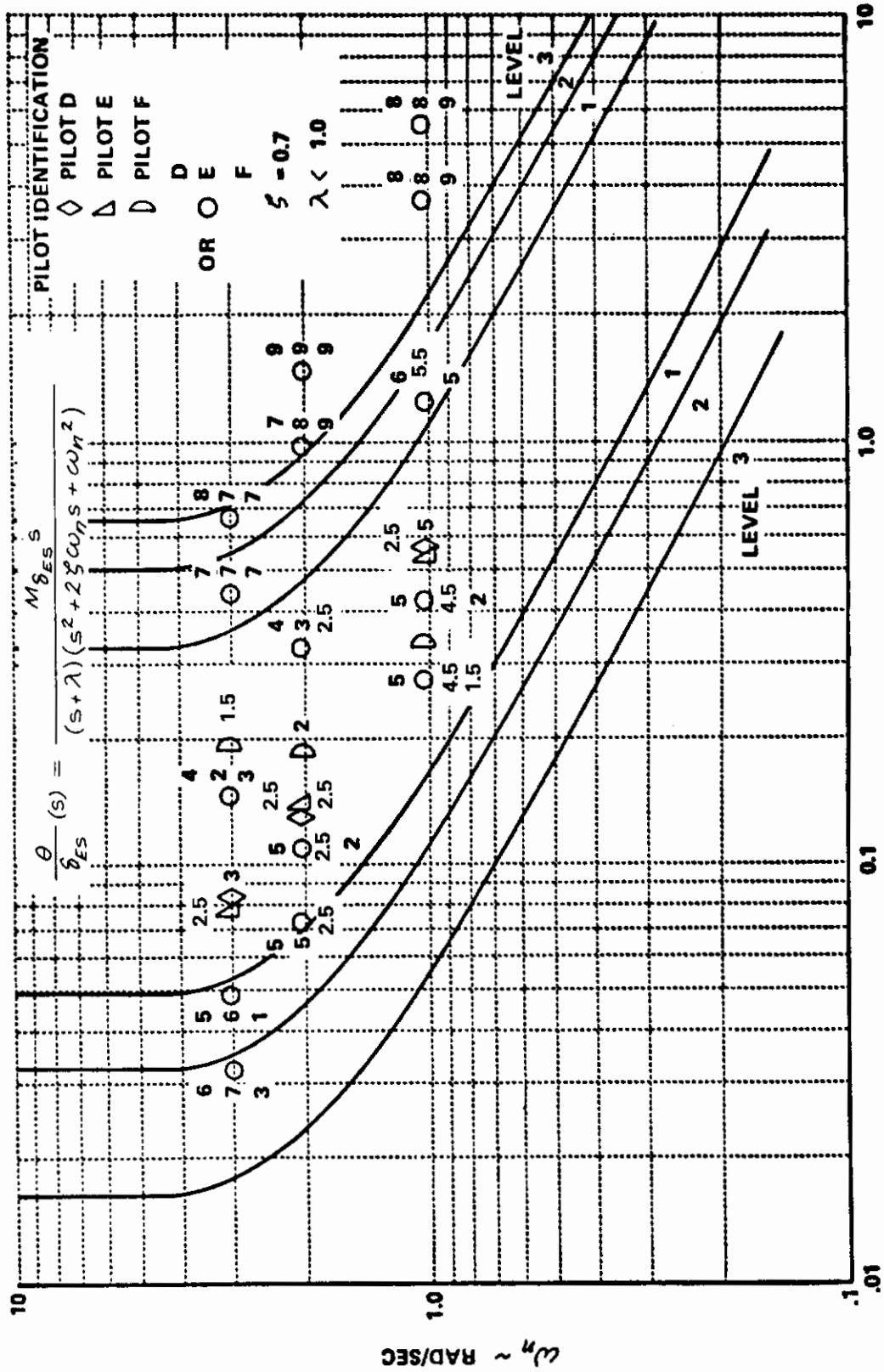

Figure 29(3.2.3.2) EFFECT OF CONTROL SENSITIVITY ON PILOT RATING, PITCH AXIS (BASED ON DATA OF REF. 34)

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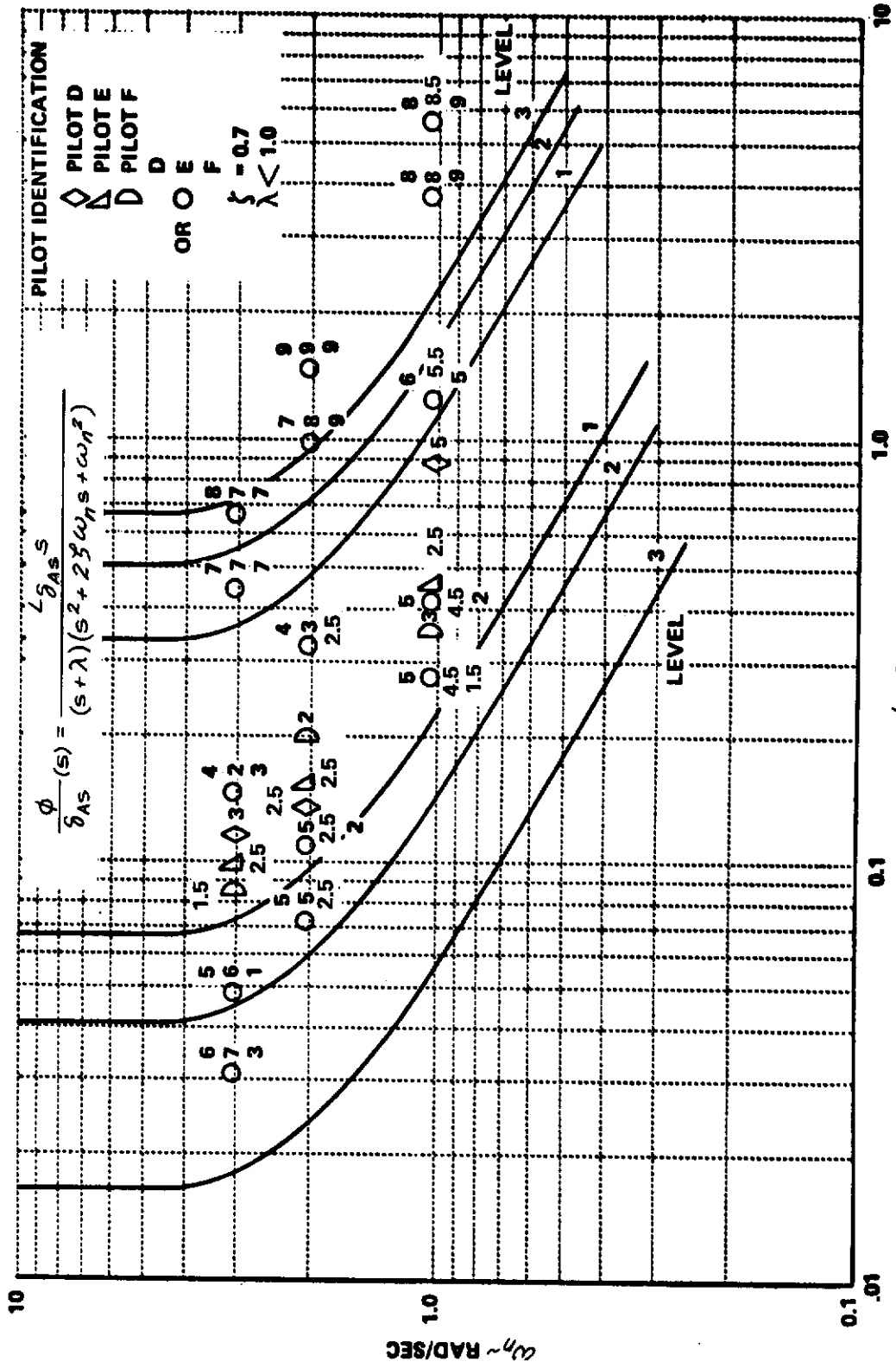

Figure 30(3.2.3.2) EFFECT OF CONTROL SENSITIVITY ON PILOT RATING, ROLL AXIS (BASED ON DATA OF REF. 34)

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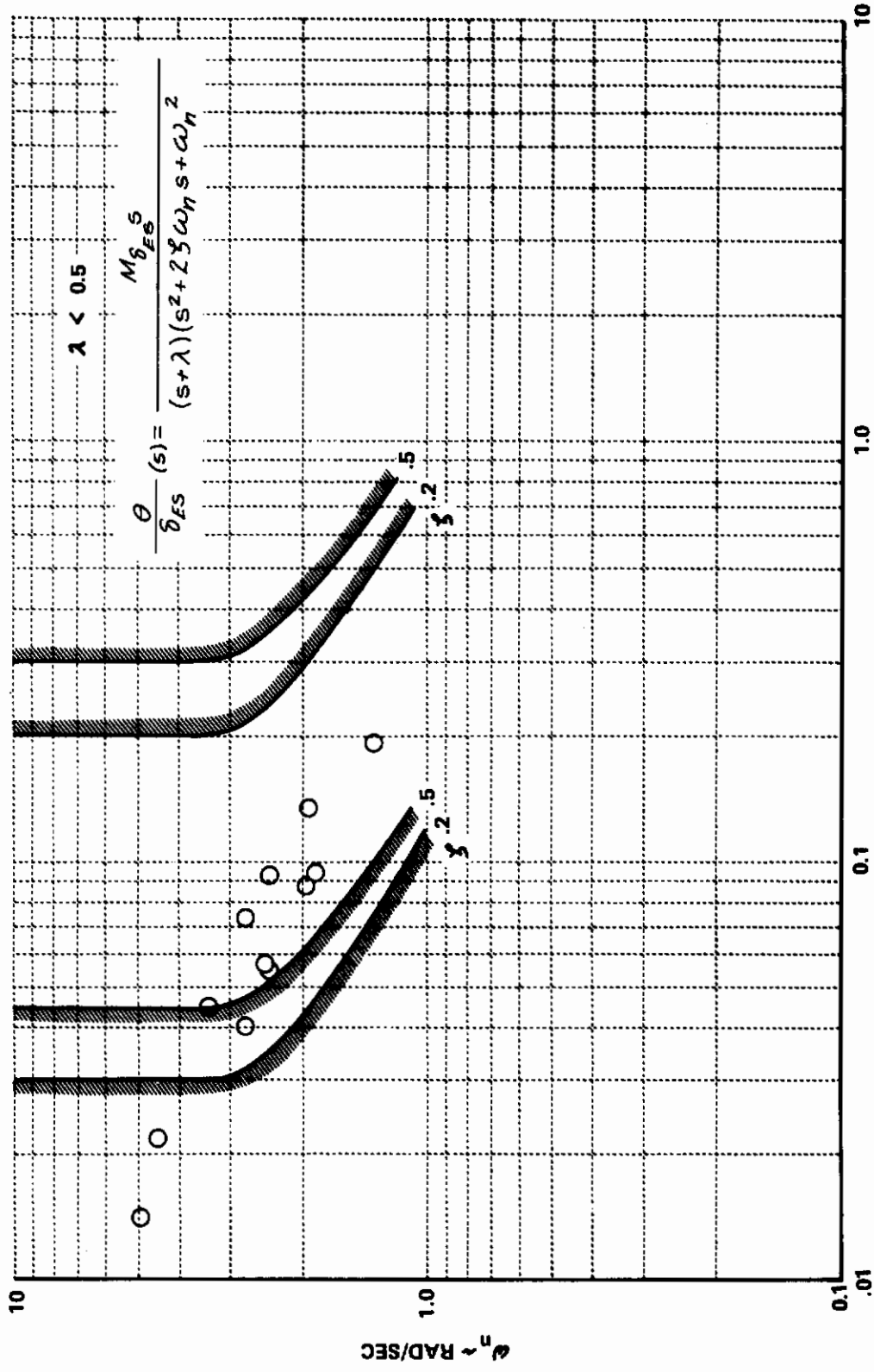

Figure 31(3.2.3.2) FLIGHT SIMULATOR DATA AND SPECIFICATION REQUIREMENTS AT HIGH FREQUENCY (BASED ON DATA FROM REF. 36)

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3.2.3.3 MANEUVERING CONTROL MARGINS

REQUIREMENT

3.2.3.3 Maneuvering control margins. When automatic stabilization and control equipment or devices are used to overcome an aperiodic instability of the basic aircraft, both the magnitude of the instability and the installed control power shall be such that at least 50 percent of the nominal control moment can be commanded by the pilot in the critical direction through the use of the cockpit controls. This requirement applies throughout the Service Flight Envelope within ± 15 knots TAS of the trim speed.

DISCUSSION

When an automatic device is utilized to overcome an aperiodic instability of the basic aircraft, there exists a possibility of being unable to recover the aircraft during maneuvering. This occurs because the augmentation device tends to use control power in the recovery direction to stabilize the unstable root. When such a situation exists, this requirement demands that throughout the Service Flight Envelope the authority of the device be such that at least 50 percent of the nominal control moment is available to the pilot in the recovery direction. Fifteen knots is considered to be a reasonable speed range about the trim speed for maneuvering without imposing unreasonable limits on augmentation system authority.

The requirement is intended to apply at all achievable combinations of flight conditions within the Service Flight Envelope, including climbing and descending flight and maneuvering.

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3.2.4 CONTROL LAGS

REQUIREMENT

3.2.4 Control lags. Starting from trimmed hovering or low-speed flight, the angular acceleration response in the commanded direction shall be developed within 0.1 seconds after the initiation of step displacements of the pitch, roll and yaw cockpit controls. In addition, the initial maximum angular acceleration shall be achieved within 0.3 seconds after the initiation of the cockpit control command. These requirements apply for input amplitudes of up to 0.5 inches.

DISCUSSION

Both the helicopter (MIL-H-8501A) and airplane (MIL-F-8785B) flying qualities specifications, References 15 and 10, recognize that some limitation of control lags is required to prevent serious degradation of handling qualities. The helicopter specification (paragraph 3.2.9) requires, that, following control inputs in pitch, roll or yaw, an angular acceleration response in the proper direction shall be developed within 0.2 seconds following the control input. This is essentially a restriction of pure transport time delays but places no limit on first or higher order lags in the control system.

The airplane specification approaches the problem of control system lags by limiting the phase lag between the cockpit control force and the deflection of the appropriate aerodynamic surface at frequencies up to the aircraft short period for pitch and the greater of the Dutch roll or the inverse of the roll mode time constant for roll and yaw.

The specification of control lags in terms of control surface response may not be satisfactory for V/STOL aircraft in the hovering and low speed flight regime since some of these aircraft will derive moment control through jet reaction devices. Thus, the appropriate control surface may be a valve or a divider in a duct or pipe. Limiting lags between the cockpit control and the control surface may not ensure good vehicle response characteristics in these cases because of inherent aerodynamic lags downstream of the control surface.

Since pilots cannot readily separate the effects of control system dynamics from the inherent or augmented vehicle dynamics, it would be most desirable to include the specification of control system lags in an overall requirement or criterion for the characteristics of vehicle response to control inputs. This specification should also include the effect of force feel system dynamics.

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A recent flight test program, described in Reference 65, was directed to gathering and analyzing data to develop such a criterion. In this experiment, a series of aircraft short period and control system dynamics was evaluated in a variable stability aircraft while performing tasks simulating the combat phase of a fighter's mission. A closed-loop analysis was used to determine the compensation required in a pilot model to meet certain performance criteria and this compensation was correlated with pilot ratings.

It is not intended to imply that closed-loop methods are the only techniques to be considered in developing control response criteria, but rather that additional effort should be directed to developing criteria which consider all the pertinent factors in a cohesive and unified manner.

The data presently available pertinent to the low speed and hovering flight regime generally consider relatively simple aircraft and control system dynamics. No information is presented regarding force feel system dynamics, thus precluding analyses of the type described above.

Both simulator and flight test data indicate that aircraft with better inherent dynamics can tolerate larger control system lags than vehicles with poorer dynamics. Consider, for example, the data of Figure 1 (3.2.4) obtained with a small variable stability single rotor helicopter.

Configuration 1 has pitch damping of $-2.5\ \text{sec}^{-1}$ and the basic aircraft is given a pilot rating of 2.5. Increasing the control system lag by 0.4 seconds results in a slight pilot rating degradation to 3.5. Configuration 2 has zero angular rate damping and the basic configuration is rated 4.0. A similar increase in control lag for this vehicle results in a pilot rating of 7.5. Clearly the degradation of pilot rating with lag is highly dependent on the inherent vehicle dynamics, in this case characterized by the damping of the basic configuration. Because pilot rating degradation with lag is greatest for vehicles with poor dynamic characteristics, a single lag criterion has been specified, regardless of the level of flying qualities exhibited by other vehicle characteristics. The lag specification was derived to minimize the degradation of flying qualities of vehicles which, without lags, would exhibit Level 1 dynamic characteristics. It is recognized, however, that the degradation of flying qualities may be more severe when the allowed lags are applied to vehicles with Level 2 and 3 characteristics.

An additional constraint is the limitation of transport type lags to 0.1 second following initiation of the control input. Little data exist for systems with combinations of transport and first-order lags. However, Figure 2 (3.2.4) indicates that even for a well damped system, the deterioration in pilot rating with increased transport lag is immediate and rapid, while for a first-order lag the gradient is nearly zero.

Specifying the lag in terms of the time to reach a peak acceleration response provides a performance index which is readily obtainable from flight test data and, in addition, permits larger lags with increased angular rate damping. Figure 3 (3.2.4) illustrates the variation in time to peak acceleration as a function of first-order control system lag and rate damping.

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A recent fixed-base simulation program (Reference 36) investigated the effects of lags in the control system with varying levels of pitch damping. A block diagram of the simulated aircraft is shown in Figure 4 (3.2.4). At each level of pitch damping (M_q = - 1, -3, -6), as the lag was increased, the pilot rating degraded. Lines of constant pilot rating were derived as functions of time to peak acceleration response following step control inputs and pitch damping [Figure 5 (3.2.4)]. A time to peak acceleration response of 0.3 seconds correlates reasonably well with the 3.5 pilot rating boundary except at the lowest levels of damping tested.

Although test data from earlier operational or research VTOL aircraft are extremely scarce, Reference 66 documents a flight test program in which the required parameter was measured. Figure 6 (3.2.4) shows plots of the time to maximum acceleration response from tests on the XV-5A aircraft in various hovering flight configurations. The quality of the step control input is not documented but the text of the report states that the control inputs were made manually without a jig. The scatter observed in the data probably reflects the manner in which the manual control input was made. However, the average time to peak acceleration is of the order of 0.3 to 0.4 seconds while the minimum time is about 0.2 seconds. No direct correlation of the control system lag characteristics with pilot rating is possible but it is stated that the overall handling qualities in hover received a pilot rating of 3. This pilot rating indicates that the level of lags present were at least satisfactory. Further, the data indicate that these rates of response to control inputs are not unrealistic design objectives.

There is a lack of systematic data concerning the effects of lags on VTOL aircraft with attitude stabilized or higher order dynamic systems. A flight test program (Reference 67), in which a variable stability aircraft was utilized to investigate the effects of control system dynamics in fighter aircraft in up-and-away flight and in the landing approach, lends support to the chosen lag criterion. Figures 7(3.2.4) and 8(3.2.4) illustrate the degradation in pilot rating as the time to achieve the peak acceleration response increases. Generally the data suggest that increasing system natural frequency and reducing the damping ratio makes the vehicle more sensitive to control system lags. Using 0.3 seconds as the maximum allowable time eliminates most of the configurations which received unsatisfactory ratings while accepting the satisfactory configurations. Only in the case of the landing approach configuration (LA), Figure 8(3.2.4), does the criterion appear overly restrictive.

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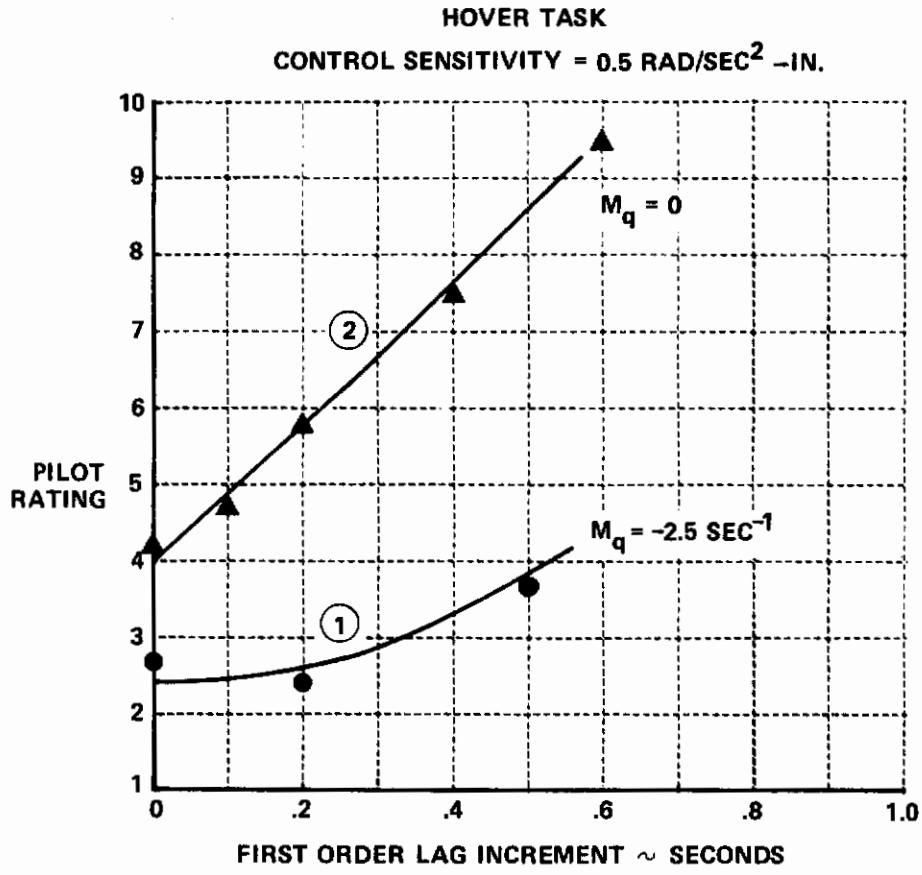

Figure 1 (3.2.4) INFLUENCE OF PITCH DAMPING AND CONTROL SYSTEM LAG ON PILOT OPINION (REFERENCE 68)

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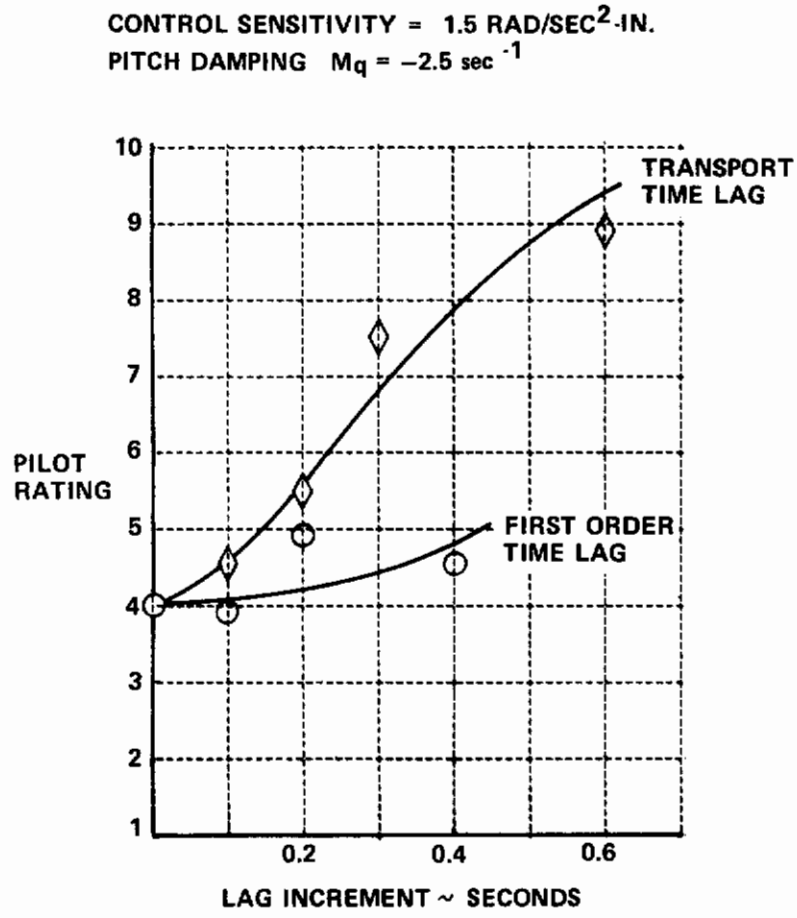

Figure 2 (3.2.4) COMPARISON OF EFFECTS OF FIRST ORDER AND TRANSPORT TIME LAGS ON PILOT OPINION (REFERENCE 68)

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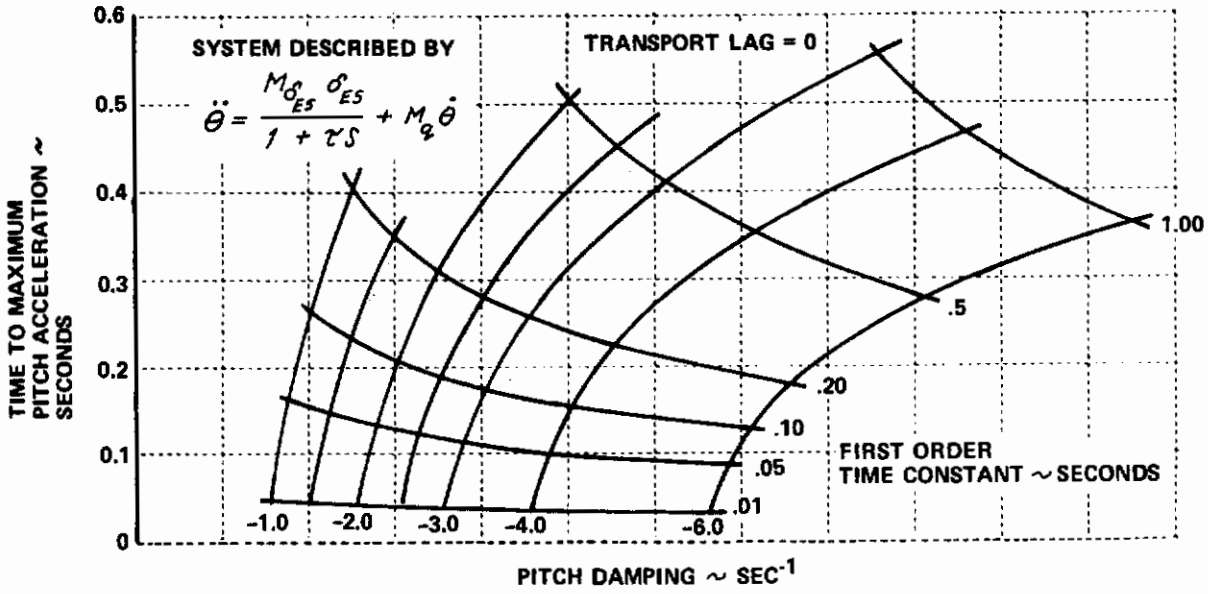

Figure 3 (3.2.4) TIME TO MAXIMUM PITCH ACCELERATION AS FUNCTION OF PITCH DAMPING AND CONTROL SYSTEM TIME CONSTANT

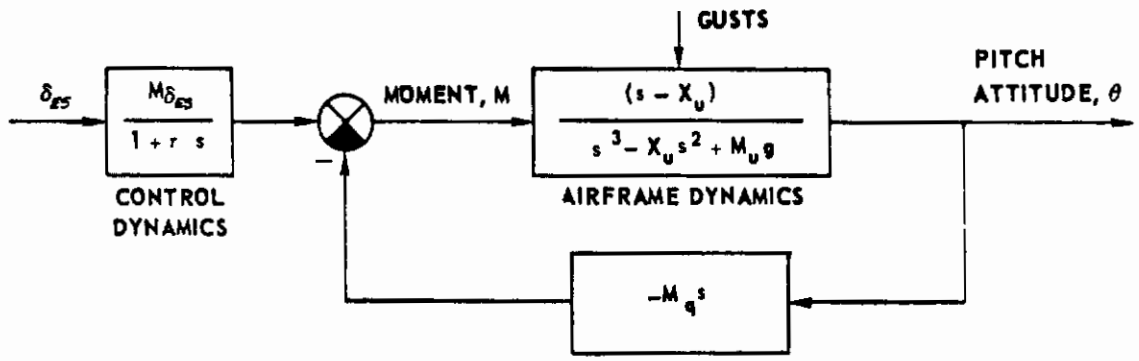

Figure 4 (3.2.4) MODEL FOR AIRCRAFT OPEN-LOOP LONGITUDINAL DYNAMICS WITH LAGS.

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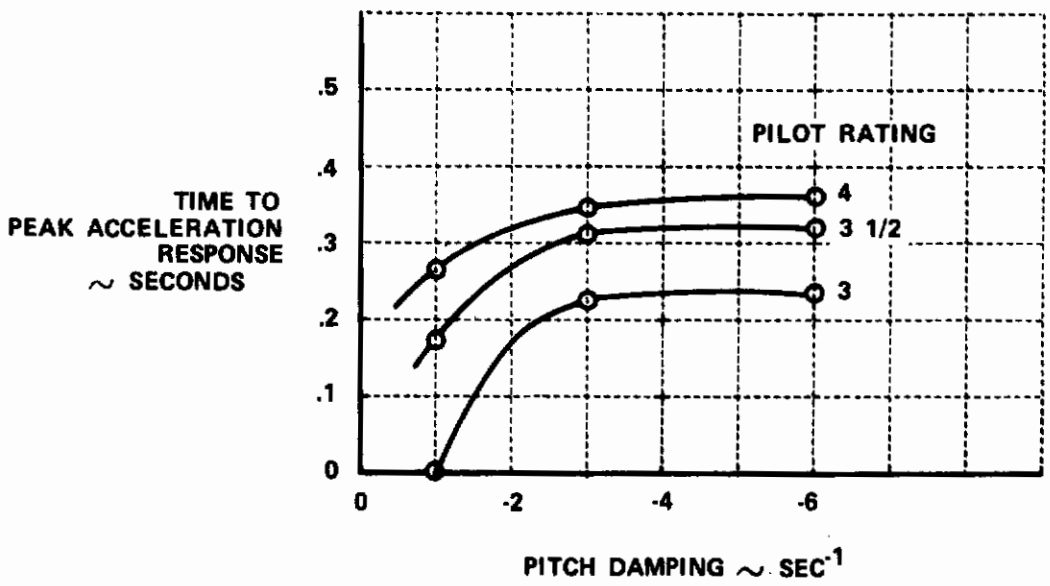

Figure 5 (3.2.4) CORRELATION OF PILOT RATING WITH TIME TO PEAK ACCELERATION RESPONSE (BASED ON DATA OF REFERENCE 36)

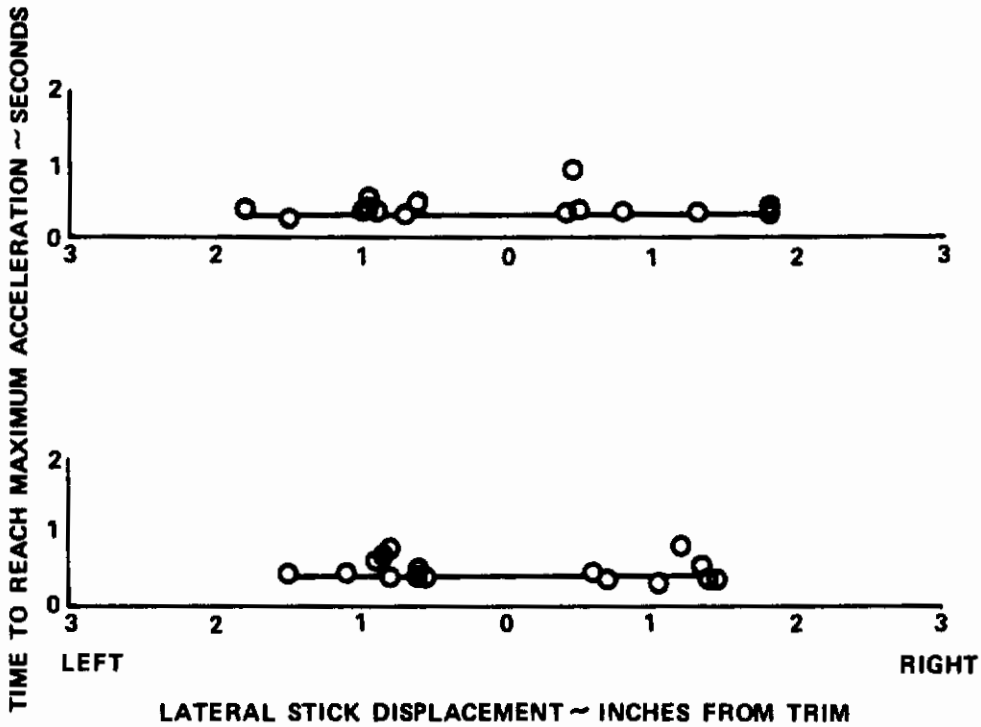

Figure 6 (3.2.4) TIME TO PEAK ANGULAR ACCELERATION RESPONSE FOLLOWING MANUAL STEP CONTROL INPUTS (BASED ON XV-5A DATA FROM REFERENCE 66)

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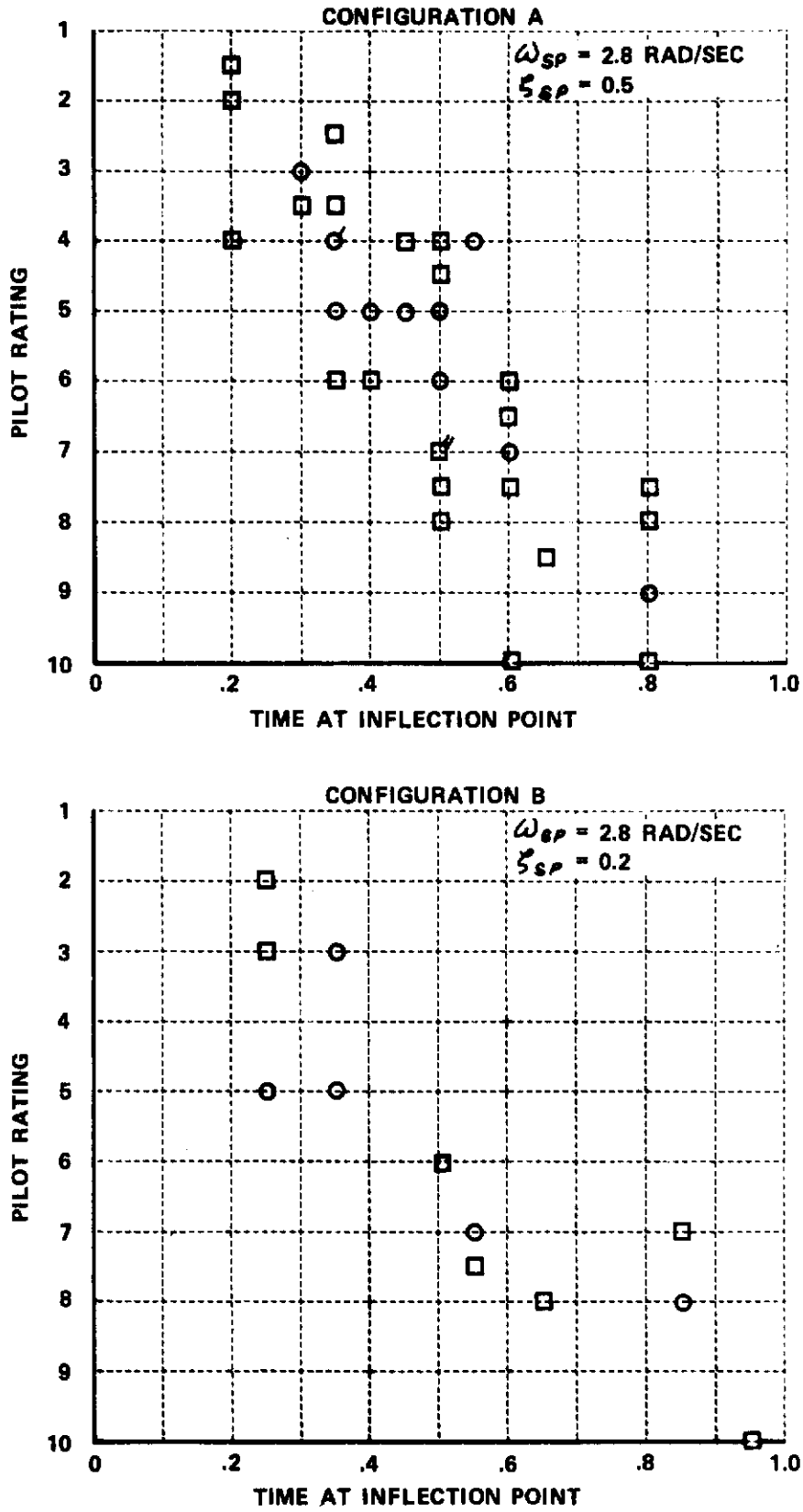

Figure 7 (3.2.4) RESULTS FROM FIGHTER SIMULATION (FROM REFERENCE 67)

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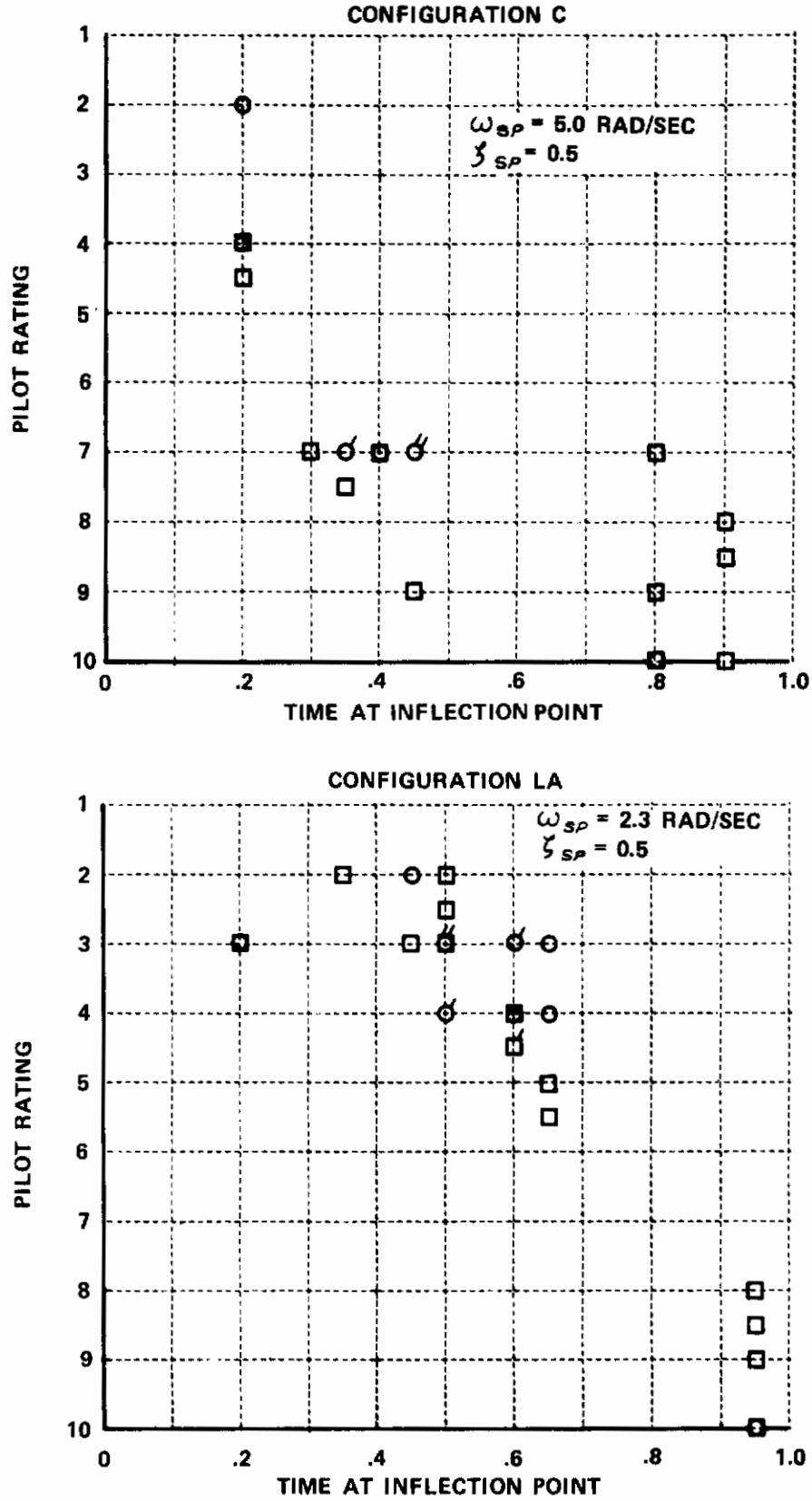

Figure 8 (3.2.4) RESULTS FROM FIGHTER SIMULATION (FROM REFERENCE 67)

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3.2.5 VERTICAL FLIGHT CHARACTERISTICS

3.2.5.1 HEIGHT CONTROL POWER

REQUIREMENT

3.2.5 Vertical flight characteristics. The requirements of 3.2.5.1 through 3.2.5.3 are applicable to ascending or descending flight with airspeeds up to 35 knots TAS. They shall be met while maintaining in reserve the attitude control power called for in 3.2.3.1.

3.2.5.1 Height control power. Starting from a steady descent rate of not greater than 4 feet per second, sufficient height control power shall be available to produce upward vertical accelerations of not less than those specified in table VI following an abrupt step input of the thrust magnitude control. In any case, the steady state thrust-to-weight ratio available, T/W, shall not be less than that specified in table VI.

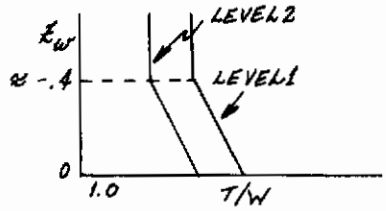
TABLE VI. Height Control Power Requirements

Level	Incremental Vertical Acceleration, g's	T/W
1	0.10	1.05
2	0.05	1.02
3		1.01

DISCUSSION

The requirements for height control power are based on the experimental data shown in Figure 1 (3.2.5). These data, which come from References 69 to 72, indicate that at high damping levels, the pilot rating boundaries are reasonably matched by a constant T/W ratio. However, at low vertical damping levels ($-Z_w < .4$), the pilot rating data show that reduced damping requires increased T/W ratios for acceptable handling qualities. From a handling qualities standpoint, it is important that this corner in the Z_w vs. T/W ratio plane be defined, since many existing VTOL aircraft lie in this region of low vertical damping.

On Figure 1 (3.2.5), the general disposition of ratings and the character of the iso-opinion curves suggest that requirement boundaries should be drawn like those indicated in the following sketch.



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The vertical portions of the boundaries so sketched offer no problem in stating requirements since they represent constant values of T/W. For a given Level, the values of T/W on the vertical portions of the boundaries in the above sketch are less than the value of T/W in the sloping portion. Hence the vertical portions represent absolute minimums of T/W. For Levels 1 and 2, the values selected for the requirement were based on Figure 1(3.2.5). For Level 3, the value selected was arbitrary.

The method by which 3.2.5.1 accounts for the sloping portions of the boundaries is explained in terms of the following trim equation:

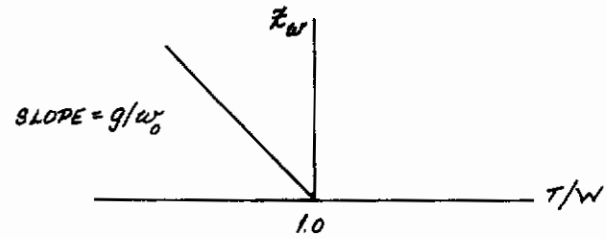
$$m Z_w w_0 - T + W = 0$$

This equation gives the balance of forces in steady vertical motion with velocity w_0 . It is important to mention that the term $m Z_w w_0$ represents the aerodynamic damping forces and not any damping that might come from a thrust SAS loop. (Vertical damping forces due to thrust SAS are already included in the T-term.)

If we write the above equation in the form

$$Z_w = \frac{g}{w_0}\left(\frac{T}{W} - 1\right)$$

then we have the equation for straight lines in the Z_w vs T/W plane passing through the point (Z_w , T/W) = 0, 1) and having a slope g/w_0 . For positive w_0 's, which represent descents, a typical graph appears as in the following sketch.



It was considered that the line representing w_0 = 4 ft/ sec was an adequate choice in terms of being parallel to the sloping straight line boundaries shown in the first sketch. Figure 2 (3.2.5), which is identical in part with Figure 1 (3.2.5), shows this. Lines L_1 and L_2 are parallel to the line labeled w_0 = 4 ft/ sec.

The horizontal distance between the w_0 = 4 ft/sec line and the line L_1 represents a constant increment in T/W independently of Z_w ; similarly for L_2 . This feature is the basis for the wording of 3.2.5.1 except that 3.2.5.1 refers to incremental vertical acceleration in g's rather than incremental T/W.

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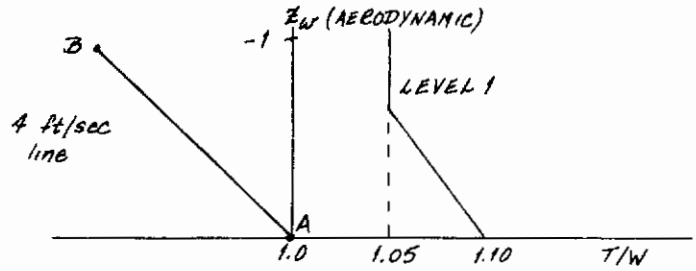
Although the use of a 4 ft/sec descent rate is somewhat of an expedient, it facilitated the writing of the requirement in what are considered to be simple yet meaningful terms.

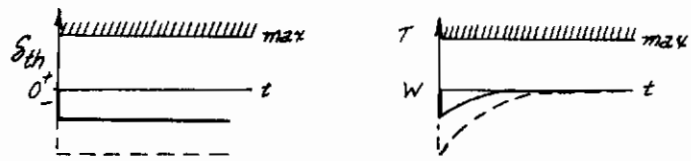
Now it could be argued that although the requirements appear to be consistent with the data base, they really may not be because other details which may be important are not specified in the data base. To expand on this, consider the following situation.

We are given two airplanes, say airplane A and airplane B. Suppose A has zero aerodynamic damping but a $Z_w = -1$ created by SAS. Suppose airplane B has zero SAS damping but an aerodynamic $Z_w = -1$. Thus both A and B have the same total damping. In a 4 ft/sec descent, airplane A would be trimmed at point A and airplane B at point B in the following sketch.



To meet Level 1, airplane A needs T/W = 1.10 while B needs 1.05. Thus it appears that airplane A is being penalized simply for having no natural or aerodynamic damping. It could be argued that since the total damping is the same for both airplanes, they should both meet the same T/W requirements. Note that airplane A will have an incremental T/W capability of 0.10 at any steady descent or climb rate. Airplane B will have an incremental T/W capability of 0.05 at zero descent rate and 0.175 at 4 ft/sec descent rate; it will have zero incremental T/W capability at a steady climb rate of 1.6 ft/sec.

A particularly important thing to consider is that the pilot of airplane A may be unaware that his height control power is limited to this constant value because his cockpit control position is not necessarily an indicator of the available control. For example, suppose airplane A is hovering and the pilot decides to establish a descent rate, say ω_1, by applying a step throttle command. The sketch below shows the cockpit input and the transient in the thrust.



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After the transient has decayed, the thrust is still equal to the weight. However the control position is lower than it was at hover. Now suppose the process is repeated only that a higher descent rate, say w_2 , is established. This is indicated by the dashed lines. Again, after the transient has decayed, the thrust is equal to the weight. But the control position is now even lower than it was for the descent rate w_1 . In both instances the amount of thrust that can be developed to arrest the descents is the same even though the cockpit control positions are different. Basically, this phenomenon is caused by the fact that the thrust is being controlled by two sources - the pilot and the SAS.

On the other hand, airplane B's incremental T/W capability varies with descent rate: at zero descent rate, this airplane would develop an incremental T/W of .05; at a 4 ft/sec descent rate the T/W incremental capability would be 0.175. Not only does this capability increase with higher descent rates, but the cockpit control position now does provide the pilot with a true measure of the vertical control power that he can apply to arrest a descent rate.

If both airplane A and airplane B just satisfy Level 1 requirements, then A would have a larger descent arresting capability but only for descent rates up to about 1.6 ft/sec, but aircraft B could arrest greater rates of descent more quickly.

It is apparent that further research on height control is needed particularly with regard to the following items.

1. The amount of thrust a pilot actually uses during stabilizing and vertical maneuvering. There is a direct analogy here with the status of the available data base for pitch, roll and yaw control power usage.

2. The kind of mathematical model and validity of assumptions to use in analyzing experimental data. For example, the discussion above indicates that even with a simple single-degree-of-freedom model, it is possible to follow different lines of reasoning depending on the mechanism by which height damping is developed.

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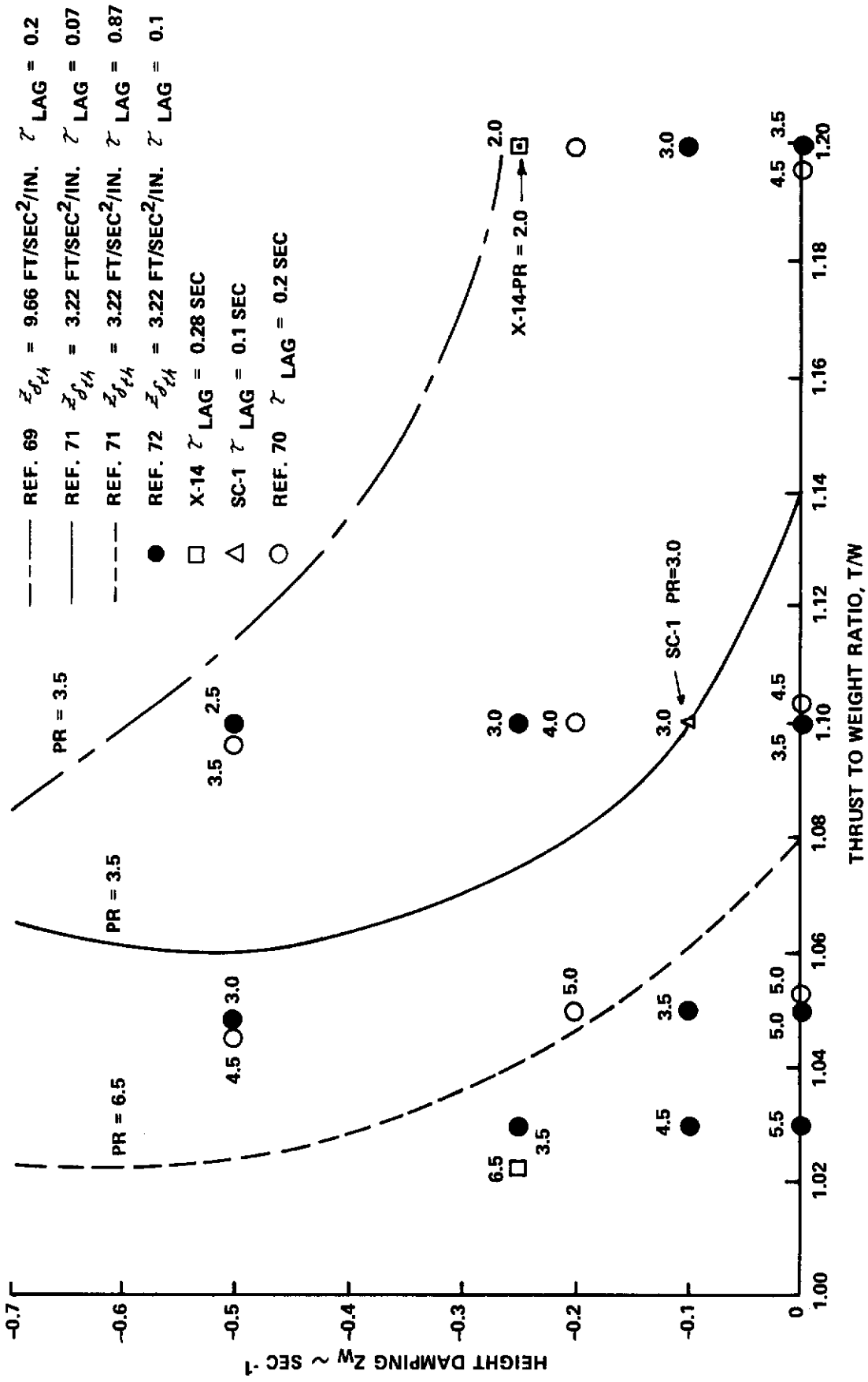

Figure 1 (3.2.5) HEIGHT CONTROL POWER VERSUS DAMPING REQUIREMENTS.

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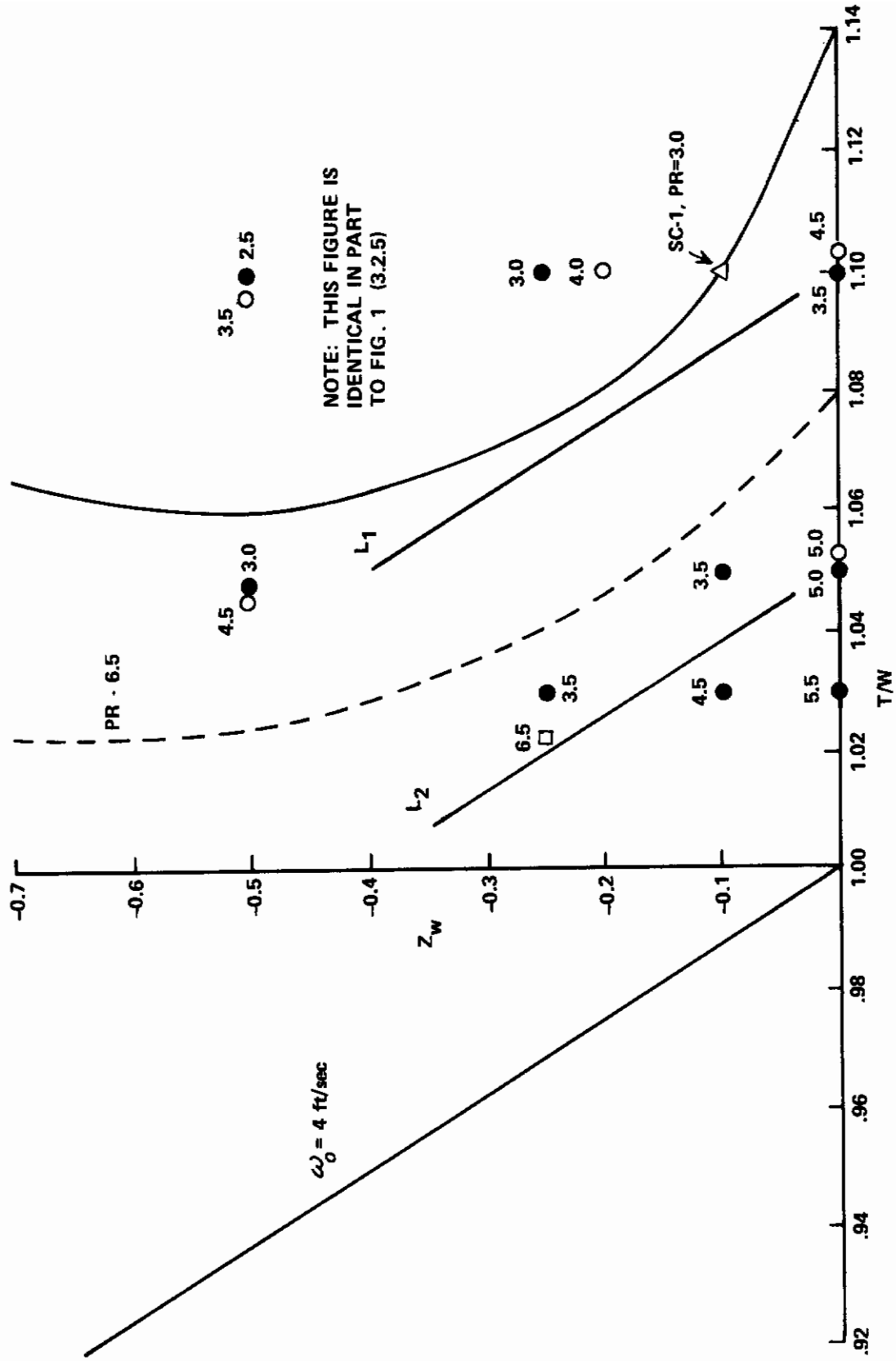

Figure 2 (3.2.5) HEIGHT CONTROL POWER CRITERIA

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3.2.5.2 THRUST MAGNITUDE CONTROL LAGS

REQUIREMENT

3.2.5.2 Thrust magnitude control lags. The following requirements shall be satisfied following an abrupt step input of the thrust magnitude control from the nominal setting corresponding to a steady descent rate of between 5 and 10 feet per second.

Level 1: It shall be possible to achieve 63 percent of a commanded incremental thrust of at least 0.05W in not more than 0.3 seconds.

Level 2: It shall be possible to achieve 63 percent of a commanded incremental thrust of at least 0.02W in not more than 0.6 seconds.

Level 3: It shall be possible to achieve 63 percent of a commanded incremental thrust of at least 0.01W in not more than 0.6 seconds.

DISCUSSION

Experimental evidence shows that the effect of increasing time lags between pilot inputs and commanded thrust levels is to cause a deterioration of pilot ratings. For the range of time lags investigated (0 to about 2.5 seconds), the variation of rating with lags as well as the absolute ratings depend on other characteristics of the height control system such as T/W and Z_w. It has been shown that the rate of deterioration is greater for the high T/W cases than the low T/W ones. This phenomenon stems largely from the fact that there is little control margin to permit much overcontrolling in the low T/W cases, regardless of the magnitude of delay. Also, at low T/W, the pilot is limited to much slower maneuvering, which minimizes the tendency to overshoot and the subsequent tendency toward PIO's. This result should not be construed as advocating a reduction in maximum T/W to cure control problems resulting from a control system delay. Rather, the pilot is limited to the use of smaller and more deliberate control motions when a great deal of precision is required. Results also indicate that increased damping is highly beneficial in enabling the pilot to cope with the delay.

A particularly pertinent experiment is described in Reference 71, wherein the pilots of the simulator were asked to maintain a constant altitude with control power of 1.15 g, $Z_w = 0$, $Z_{\delta_{th}} = 0.1$ g/inch. The time lag was then increased from 0.07 to 2.4 seconds while the pilots commented on their ability and difficulty in achieving the aims of MIL-H-8501A, i.e., hold height ± 1 ft with 1/2 inch or less of control motion. A noticeable decrease in hovering steadiness was observed as the time constant was increased to 0.3 seconds; the pilot rating deteriorated from 2.5 to 3.5. Overcontrolling was evident at 0.6 seconds, where the pilot rating had decreased to 4.5. At 1 second, the pilot rating was 6.5 and at 2.4 seconds, 9.5 and was considered too dangerous for actual flight because of large excursions in

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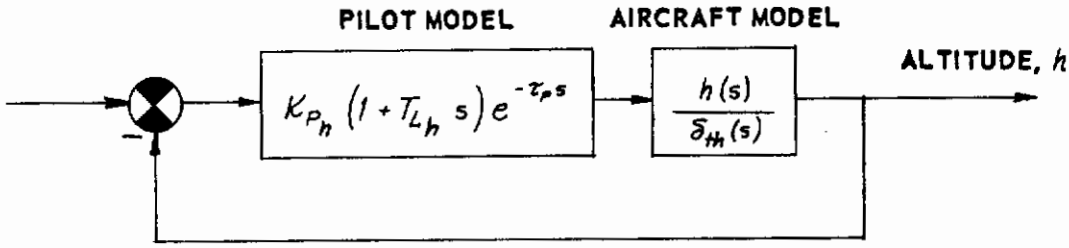
altitude. The referenced helicopter specifications were not met above 0.6 seconds. Figure 1 (3.2.5.2) from Reference 71 depicts this degradation.

In Reference 72 (in-flight), similar results were obtained. The presence of a time delay was considered to be most objectionable during the landing attempts, which, being a precision task, normally requires the pilot to increase his control activity as wheel height is reduced. For first-order lags greater than 0.5 seconds, the pilot found it necessary to alter his normal control technique for landing in order to reduce overcontrolling to a point where a reasonably safe touchdown could be made. Two of these techniques are described in Reference 72. However, it was considered unlikely that such techniques would be as beneficial when coupled with the kind of ground effect disturbances experienced by many VTOL's.

The separate effects of a first-order time lag between the pilot's input and the commanded thrust increment, and the first-order response in vertical velocity of the vehicle, are not always discernible to the pilot, nor are they easily separated in flight testing. The reason for this is apparent if we assume that the following transfer function is a satisfactory model of the combined airframe-control system dynamics.

$$\frac{h(s)}{\delta_{th}(s)} = \frac{Z_{\delta_{th}}}{s(s - Z_w)(1 + \tau_h s)}$$

In Reference 36, the closed-loop height control task was investigated both analytically and experimentally on a fixed-base simulator. Various combinations of height damping and thrust control lag were studied under the conditions where the sensitivity was selected by the pilot. In the analysis, a simple pilot model and loop closure were used as follows.



There are a number of pairs (τ_h , Z_w) that will result in a given loop closure as defined by values of crossover frequency, ω_c , and phase margin, PM_h . Reference 36 showed that these same pairs will result in a constant phase lag when excited at the crossover frequency, i.e., the phase angle ϕ'_h of the h/δ_{th} transfer function evaluated at ω_{c_h} is a constant for all these pairs. Figure 2 (3.2.5.2) shows three loci of these pairs. Each locus is for a different loop closure criterion as indicated on the figure. The interesting thing is that all three loci lie quite close to one another. They essentially divide the control lag-damping plane into two regions which separate satisfactory and unsatisfactory configurations. This result indicates the possibility of establishing flying qualities requirements in terms of phase

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lag as a function of frequency. With such an approach it would be possible to account for both lag and damping effects on pilot opinion, and provide for a reasonable means for demonstrating flight compliance. Such a criterion was suggested in Reference 36 and is shown in Figure 3(3.2.5.2).

Since it was felt that the phase lag approach needs further experimental substantiation it was not used in formulating 3.2.5.2. For example, all the satisfactory data points on Figure 3(3.2.5.2) have zero lag. The time constants selected for the requirement are based on the data of References 71 and 72.

Comments received during review cycles of early versions of the proposed specification (which specified only the time constants) indicated that the importance of nonlinear thrust characteristics of engines should be recognized in formulating height control response criteria. It is not realistic to expect fast response times for the complete usable range of engine operating conditions. However, it is apparently possible to equip power plants so that relatively small thrust changes can be made in a short time. In other words at some given operating point or for a restricted range about this operating point, it would be possible to have the kind of response times, say .3 to .6 seconds, as indicated to be desirable by some of the experimental handling qualities data. In paragraph 3.2.5.2 the range of initial descent rates and the values of commanded incremental thrust are considered to define operationally realistic "local" conditions under which thrust control response times have important effects on mission or task accomplishment.

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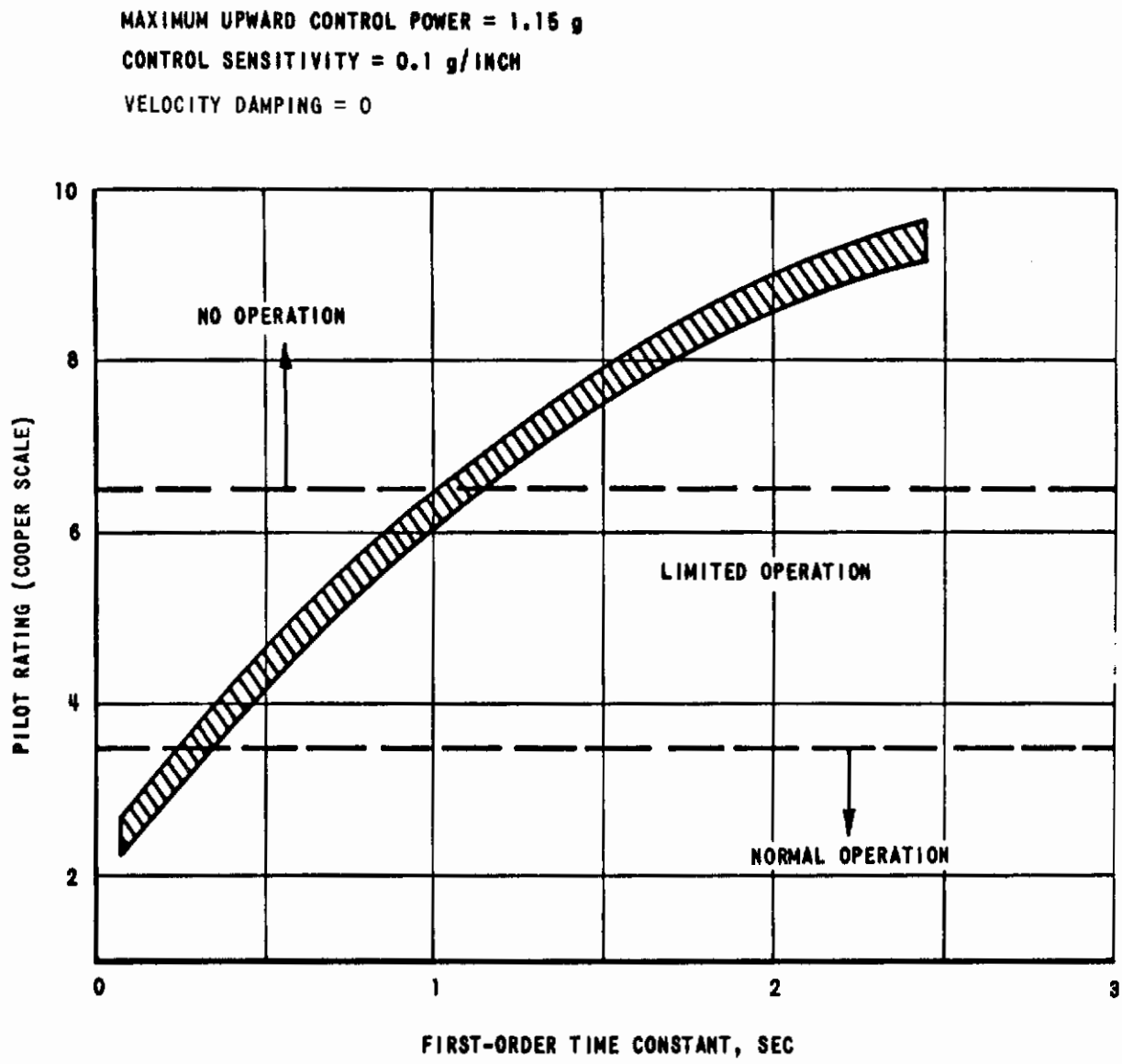

Figure 1 (3.2.5.2) VARIATION OF PILOT RATING WITH FIRST-ORDER TIME CONSTANT FOR A CONSTANT ALTITUDE HOVERING TASK. (REFERENCE 71)

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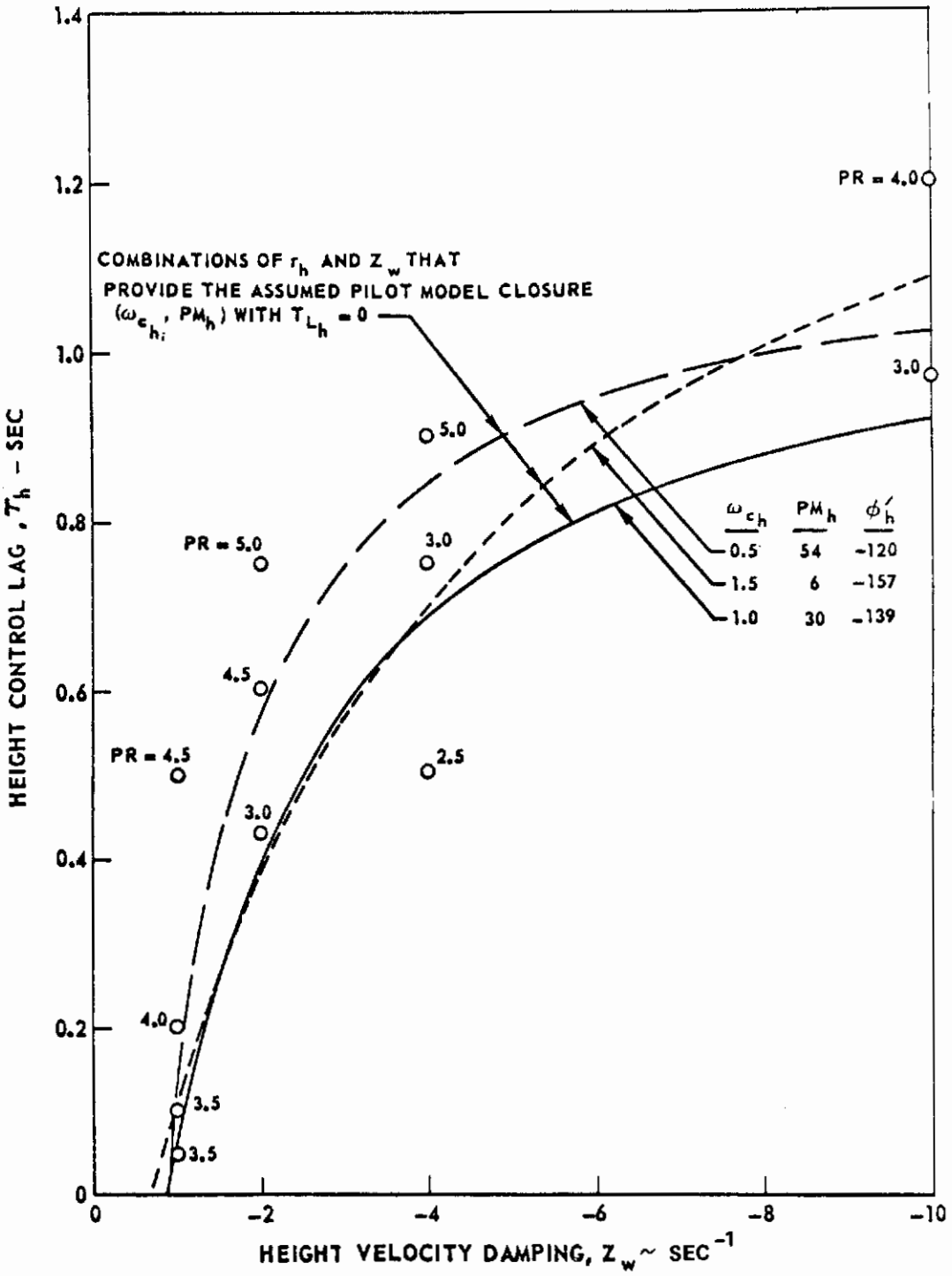

Figure 2 (3.2.5.2) COMPARISON OF FLIGHT SIMULATOR DATA WITH CLOSED-LOOP MODEL PREDICTION OF THE EFFECT OF CONTROL LAG ON Z_w REQUIRED FOR SATISFACTORY HEIGHT HANDLING QUALITIES. (FROM REFERENCE 36)

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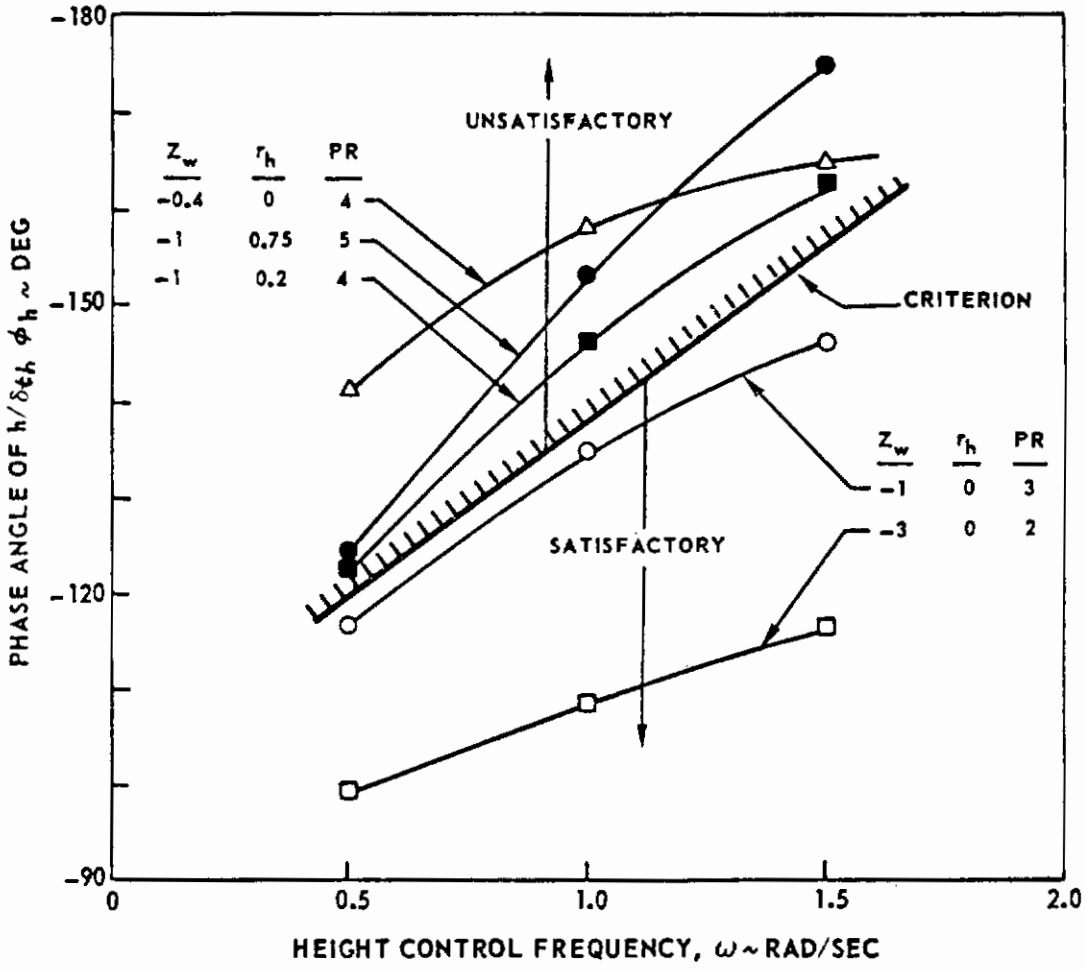

Figure 3 (3.2.5.2) DYNAMIC HEIGHT CONTROL CRITERION IN TERMS OF PHASE ANGLE OF HEIGHT RESPONSE TO SINUSOIDAL CONTROL INPUTS. (FROM REFERENCE 36)

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3.2.5.3 RESPONSE TO THRUST MAGNITUDE CONTROL INPUT

REQUIREMENT

3.2.5.3 Response to thrust magnitude control input. Following an abrupt step displacement of the thrust magnitude control, the ratio of the maximum rate of climb occurring within the first second to the magnitude of the cockpit control input shall lie within the bounds of table VII. This requirement is for hovering in still air and for inputs up to the maximum permissible.

TABLE VII. Response to Thrust Magnitude Control Input in One Second or Less (Climb Rate in Feet per Minute per Inch of Control Deflection)

Level	Minimum	Maximum
1	100	750
2	50	1200
3		2000

DISCUSSION

The effects of control sensitivity ($Z_{\delta_{th}}$) and damping (Z_w) have been treated in References 49, 69, and 70, and briefly in Reference 72. Reference 49 showed that the pilot rating boundaries were not significantly different for collective-type controllers as compared to throttle-type controllers, and hence most investigations thereafter dealt exclusively with collective-type controllers.

Most of the results in this area are shown in Figure 1 (3.2.5.3). Based on this data, the requirements of paragraph 3.2.5.3 were formulated in terms of the ratio

$$\frac{\text{maximum rate of climb occurring within first second}}{\text{magnitude of abrupt step displacement of cockpit control}} \left(\frac{\text{ft/min}}{\text{in.}}\right)$$

The values selected for this ratio are shown on Figure 1 (3.2.5.3) using a simple single-degree-of-freedom approximation to height dynamics. The lines on Figure 1 (3.2.5.3) representing the Level 1 and Level 2 values are considered to be reasonable equivalents to the pilot rating boundaries. For Level 3 no requirement is placed on a minimum while the maximum value is arbitrary.

Note that the requirement applies for inputs up to the maximum permissible which may be determined by the maximum control available, the development of excessive climb rates, structural limits, etc.

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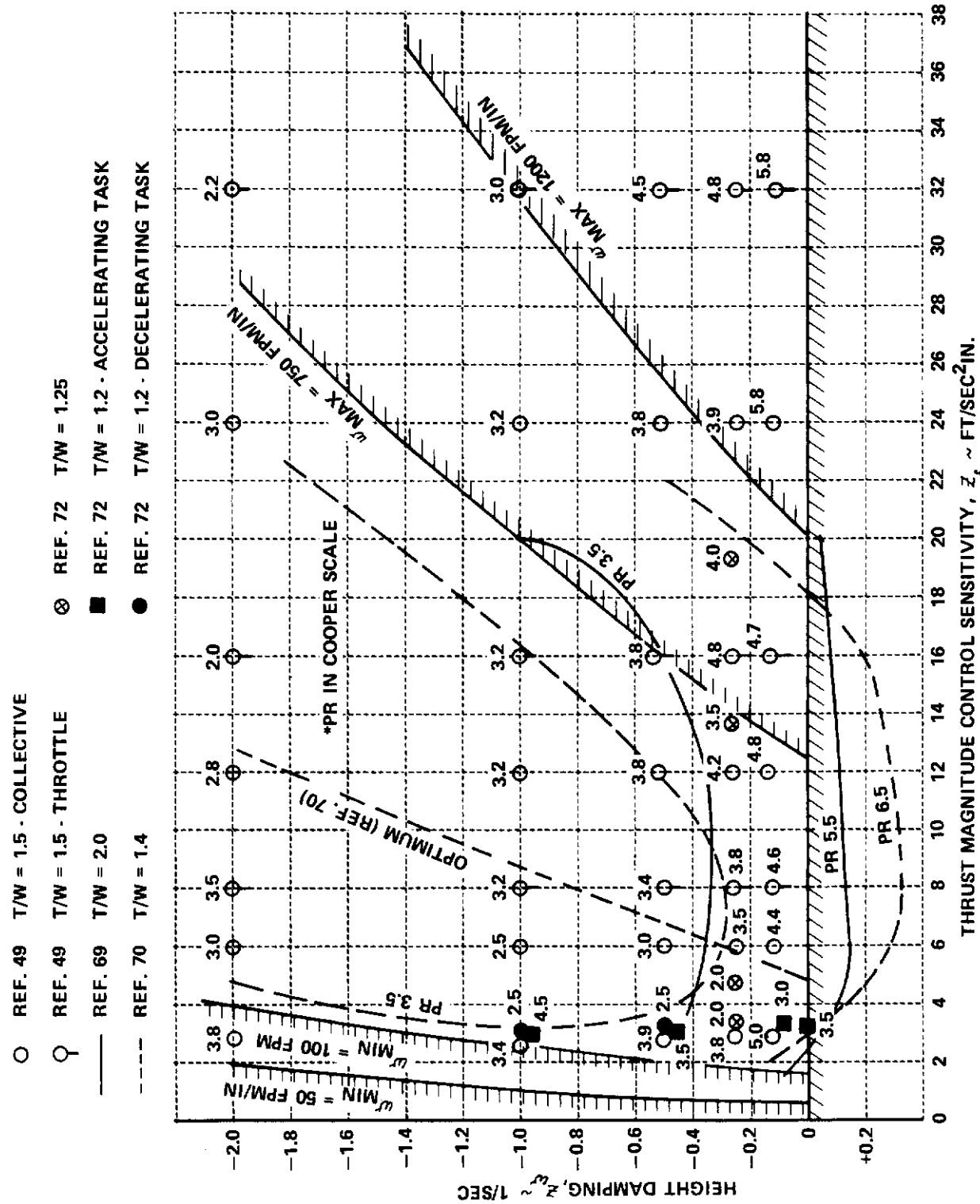

Figure 1 (3.2.5.3) VERTICAL VELOCITY RESPONSE CRITERIA

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3.2.5.4 VERTICAL DAMPING

REQUIREMENT

3.2.5.4 Vertical damping. The translational height damping in vertical flight (i.e., the vertical force proportional to vertical velocity) shall not be in the unstable sense.

DISCUSSION

This requirement is based on the data in Figure 1 (3.2.5.3), which indicates that some nonzero value of damping would be desirable especially for Level 1. However, this value is quite small. Because of the complex nature of vertical force characteristics at low speeds, it was considered that placing precise numerical requirements on height damping for different levels of flying qualities is not warranted.

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3.3 FORWARD FLIGHT

REQUIREMENT

3.3 Forward flight. The forward flight requirements apply to those Flight Phases of the operational missions of the aircraft which include equilibrium flight or maneuvering in the speed range of 35 knots TAS to V_{con}. The requirements of 3.5, 3.6, 3.7 and 3.8 are also applicable in this speed regime.

DISCUSSION

This requirement establishes the speed range wherein the following requirements apply. It should be noted that these requirements are related to equilibrium flight and maneuvering about a fixed operating point condition, and are not intended to apply during rapid acceleration or deceleration transition maneuvers (this aspect is discussed at some length in Section III, Specification Structure and Philosophy). In addition, the other sections of the V/STOL Specification relating to requirements on the characteristics of the flight control system (3.5), takeoff and landing (3.6), atmospheric disturbances (3.7), and miscellaneous requirements (3.8), must be applied in this speed range. The rationale for separation of the V/STOL Specification into the Hover and Low Speed Section and a Forward Flight Section is discussed in Section III, Specification Structure and Philosophy.

The essential philosophy behind the requirements of the Forward Flight Section of the V/STOL Specification is an attempt to enable an aircraft to be flown with satisfactory handling qualities at any speed between 35 knots and V_{con}. In the vicinity of V_{con}, it is desirable that the requirements of the Forward Flight Section of the V/STOL Specification blend with those of MIL-F-8785B (Reference 10). At the other end of the speed region it is desirable that the requirements of the Forward Flight Section blend with those of the Hover and Low Speed Section of the V/STOL Specification. While a reasonable amount of data are available around hover, and MIL-F-8785B requirements together with background data are available for $V>V_{con}$, there does exist a shortage of data when attention is directed at speeds between 35 knots forward and conventional landing approach speeds. In the original proposed V/STOL Specification (Reference 3) this data shortage coupled with the diverse requirements of such documents as AGARD 408A, RTM-37, and MIL-H-8501 (References 46, 47, and 15 respectively), and the desire to blend requirements established the basic philosophy that unless more specific data were available, V/STOL aircraft would be considered to be the equivalent of a landing approach configuration (Flight Phase Category C) of MIL-F-8785B.

Though this philosophy of blending requirements still manifests itself in the present V/STOL Specification (Reference 1), V/STOL aircraft are now considered on their own capabilities and are no longer directly tied to Flight Phase Category C requirements of MIL-F-8785B. The requirements of Reference 1 for the Forward Flight Section (3.3) reflect continued thought, re-evaluation of existing data, and analysis of new data and new reports (e.g., Reference 73), concerning V/STOL aircraft handling qualities.

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3.3.1 LONGITUDINAL EQUILIBRIUM

REQUIREMENT

3.3.1 Longitudinal equilibrium. With the aircraft trimmed at speeds from 35 knots forward to V_{con}, the following requirements shall be satisfied for perturbations of ± 10 knots from the trim speed except where limited by the boundaries of the Service Flight Envelope. The configuration and trim may be different at each trim condition, but shall remain fixed while determining the control gradients.

Level 1: The variations of pitch control force and control position with pitch attitude and airspeed shall be smooth and the local gradients stable or zero.

Level 2: For those Flight Phases of the operational missions of 3.1.1 for which IFR operation is required, the Level 2 requirement is the same as for Level 1. In all other cases, the Level 2 requirement is the same as for Level 3.

Level 3: The Level 1 requirements shall apply except that the local pitch control position gradients may be unstable. However, the change in the pitch control position shall not exceed one-half inch in the unstable direction over the speed range or pitch attitude range associated with the unstable gradient.

Stable pitch control gradients mean that incremental pull force and aft displacement of the cockpit control are required to maintain nose-up attitudes or slower airspeeds and the opposite to maintain nose-down attitudes or higher airspeeds. The term gradient does not include that portion of the control force or control position versus pitch attitude or airspeed curve within the preloaded breakout force or friction band.

DISCUSSION

The primary purpose of this requirement is to limit divergences in airspeed and aircraft attitude which might remain undetected by a busy pilot and could result in the aircraft entering an unsafe flight condition with insufficient control available for recovery. The Level 1 and 2 requirements are intended to prevent these divergences for IFR flight. The Level 3 requirements (and Level 2 VFR only) limits permissible static instability as determined from control position gradients with speed and attitude.

The requirements are directed at both force and position, although the aircraft industry maintains that force stability is sufficient. Discussions with pilots concerning IFR tasks and articles by pilots such as Reference 74 indicate the need for control position stability as well as control force stability.

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Other specifications have also recognized the desirability of providing both control force and control position stability (References 10, 15, 46, 47). This requirement provides a reasonable interface between the requirements of Section 3.2.1.3 of Reference 1 and those of Section 3.2.2.1 of Reference 10.

The general criterion for static stability (Reference 75) is that the constant term of a conventional aircraft characteristic equation be greater than zero. When this term is less than zero, aperiodic divergence results. The stability derivatives that result in aperiodic divergences also manifest themselves in the gradients of stick force and stick position with velocity or attitude. As discussed in Reference 76, the gradients are proportional to static stability and pilots are used to the relationships existing between control position, force and speed. For the V/STOL Specification, it was decided to address aperiodic divergences explicitly in the section on dynamic response requirements (3.3.2), and to address longitudinal equilibrium only through the device of stick gradients. This was felt necessary due to the various additional modes which might be introduced by complicated stability and control systems.

The two most common causes of static instability are the center of gravity being too far aft ($M_{\dot{w}}$ becoming positive), and M_u being negative. In both of these cases, a single real root will generally become unstable as the critical loading or flight condition is approached, as shown on Figure 1(3.3.1).

The following equations are representative of straight-and-level-flight about a fixed operating point written with respect to stability axes of the aircraft:

$$\begin{aligned}
(s-X_u)u &- X_w w &+ g\theta &= X_{\delta_{ES}}\delta_{ES} \\
-Z_u u &+ (s-Z_{\dot{w}})w &- u_o s\theta &= Z_{\delta_{ES}}\delta_{ES} \\
-M_u u &- (M_{\dot{w}}s + M_w)w &+ s(s-M_q)\theta &= M_{\delta_{ES}}\delta_{ES}
\end{aligned}$$

Thus the constant term of the characteristic equation may be expressed as: $g(Z_u M_w - Z_w M_u)$. As previously mentioned, the presence of an unstable aperiodic root can be determined analytically by the sign of the constant term of the characteristic equation. With the elevator fixed, for instance, the airspeed will diverge from trim when $Z_u M_w - Z_w M_u < 0$. With two unstable real roots (or any even number of such roots), the constant term of the characteristic equation can still be positive. In this case, the presence of divergent modes will be indicated by other coefficients, or Routh's discriminant, of the characteristic equation being negative and, of course, by factoring the characteristic equation. This is not a common occurrence, and therefore the criterion $Z_u M_w > Z_w M_u$ is usually sufficient to ensure that there are no first-order divergent modes present.

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When determining whether or not there is a divergent mode, stick free, the above relationship must be modified to account for pitch control surface movements caused by hinge moments, bobweights, downsprings, SAS, etc.

The foregoing discussion is concerned with the primary static stability requirement of 3.3.1 and is useful for design purposes. It is, however, extremely difficult to test for a divergent mode by disturbing the airplane from trim and then observing the airspeed response, the airplane may diverge for a while until it reaches a flight condition where the airplane is stable, and then hold some new speed. In such cases, the pilot would not know whether the airplane was unstable around the original trim speed or whether he was initially out of trim.

A straightforward way to detect slightly divergent modes in flight test is to measure control force and position variations with speed at constant throttle. A stable variation of control force with speed indicates the presence of a phugoid mode (stick-free). For example, if a steady push force is required to hold a speed above trim speed, release of the stick will cause the airplane to nose up and slow down, undershoot the trim speed, speed up again, etc. If zero push force is required to hold an off-trim speed, release of the stick will cause the airplane to maintain attitude and speed, i.e., there is no longer any phugoid oscillation (stick-free). When a pull force is required to maintain a speed above the trim speed, release of the stick will cause the airplane to pitch down, and the airspeed will diverge. The same kind of explanation relates the stick position required to hold off-trim airspeeds to the airplane's behavior when the stick is returned to its trim position and held fixed. The requirements on control gradients essentially ban divergences, but are stated in a form more directly useful for flight test purposes. Zero gradients have been permitted so that response-command systems are not disallowed.

The expression for the control position gradient with speed, for example, is:

$$\frac{d\delta_{ES}}{du} = \frac{Z_u M_w - M_u Z_w}{Z_w M_{\delta_{ES}} - M_w Z_{\delta_{ES}}}$$

A negative value of $Z_w M_{\delta_{ES}} - M_w Z_{\delta_{ES}}$ would result in a negative value of $(d\delta_{ES}/dn)$. That is, aft control motion would increase angle of attack in the normal manner, but the normal acceleration response would be downward, even in the short-term steady state. For all practical cases, therefore, $Z_w M_{\delta_{ES}} - M_w Z_{\delta_{ES}}$ will always be positive; and stable control gradients will ensure that no divergent aperiodic modes are present.

The explanation just presented is based on the assumption that the stability derivatives remain essentially constant and may become less sound at trim speeds much below V_{con}. At these lower speeds, V/STOL configurations can exhibit wide variations in stability derivatives over relatively small speed ranges. In addition, any tendency toward nonlinear aerodynamics would limit the generality of conclusions based on linear equations. This by no means detracts from the validity of the requirement of the need for "static" longitudinal stability.

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It may be noticed that the specification requires both attitude and speed to have stable gradients with pitch control. Usually, if the attitude gradient with stick position is stable, then the speed gradient with stick position will be also, and vice versa. If this was always the case it would be logical to specify that either can be demonstrated and leave it to the contractor to choose the most convenient. Unfortunately this is not so; an unusual combination of derivatives could result in stable speed-stick position stability but unstable attitude-stick position stability or vice versa. The pitch control force derivatives (e.g., $X_{\delta_{ES}}$) are most likely to cause such a phenomenon.

An example of such a configuration is quoted in Reference 42 (Table B1). This configuration had the following derivatives

$$X_u = -.15,\ M_u = +.105,\ Z_u = -.82,\ M_q = -.99,\ \frac{X_{\delta_{ES}}}{M_{\delta_{ES}}} = -1.88$$

Then the following approximations to the u/δ_{ES} and θ/δ_{ES} responses give

$$\frac{\theta}{\delta_{ES}}(s) = \frac{+M_{\delta_{ES}}\left(s - X_u + \frac{X_{\delta_{ES}}}{M_{\delta_{ES}}} M_u\right)}{\Delta} = \frac{+M_{\delta_{ES}}(s - .047)}{\Delta}$$

$$\frac{u}{\delta_{ES}}(s) = \frac{+M_{\delta_{ES}}\left(-g - \frac{X_{\delta_{ES}}}{M_{\delta_{ES}}}(-s^2 + M_q s + M_\theta)\right)}{\Delta} = \frac{-1.88 M_{\delta_{ES}}\left(s^2 + .99 s + \frac{32.2}{1.88}\right)}{\Delta}$$

where $\Delta = (s + 1.9)\left[s^2 + 2(-.30)(1.28)s + 1.28^2\right]$

For this case, the attitude-stick position gradient will be unstable and the speed-stick position gradient will be stable.

For Level 1 and Level 2 IFR, the requirements in 3.3.1 of Reference 1 are essentially the same as 3.2.1.1 of Reference 10. The Level 3 requirement of Reference 1 is similar to the VFR requirement of Reference 15 for control position instability. The speed range required for demonstration is limited to ± 10 knots about the trim speed, since it is assumed that beyond this speed range the vehicle would probably be retrimmed. Although there is a significant shortage of data for the speed range from 35 knots to V_{con}, examination of comments in References 37 and 77 indicates that pilot rating can rapidly deteriorate with changes from stable to unstable stick position and force gradients with either speed or attitude.

It should be noted that the intent of this requirement is not to discourage various types of V/STOL aircraft. Many vehicle concepts (e.g., tilt-wing) do not inherently possess the stability characteristics necessary to meet the requirements until well above 35 knots forward airspeed, as indicated in Reference 78. Thus, for these type of aircraft, augmentation systems would be required to meet the requirements for Level 1 and Level 2 IFR.

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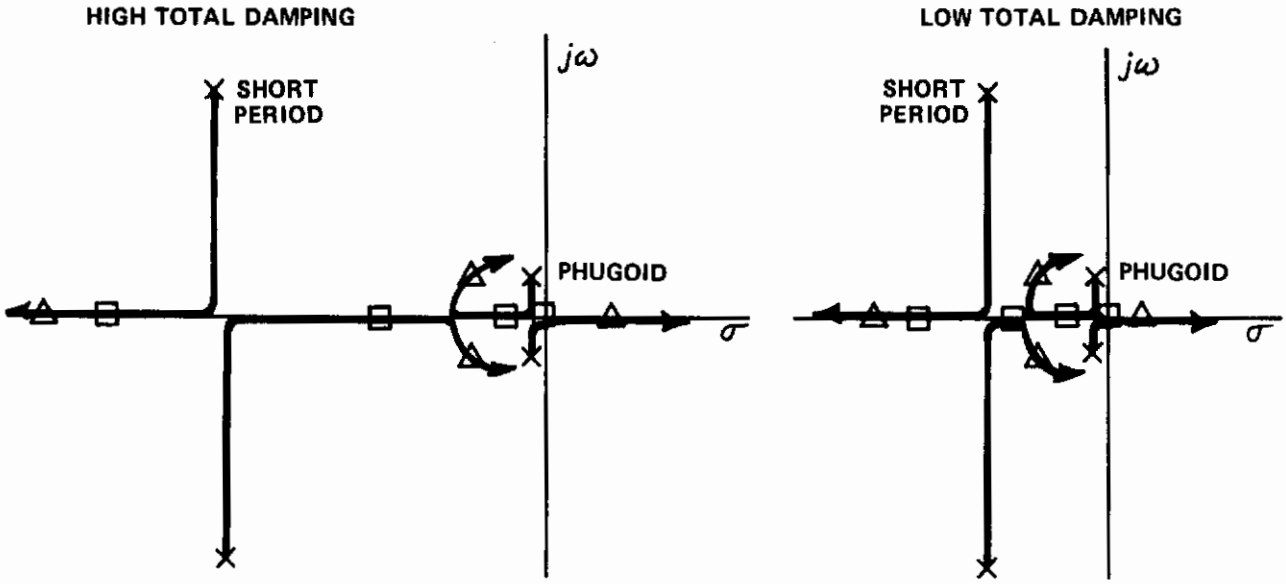
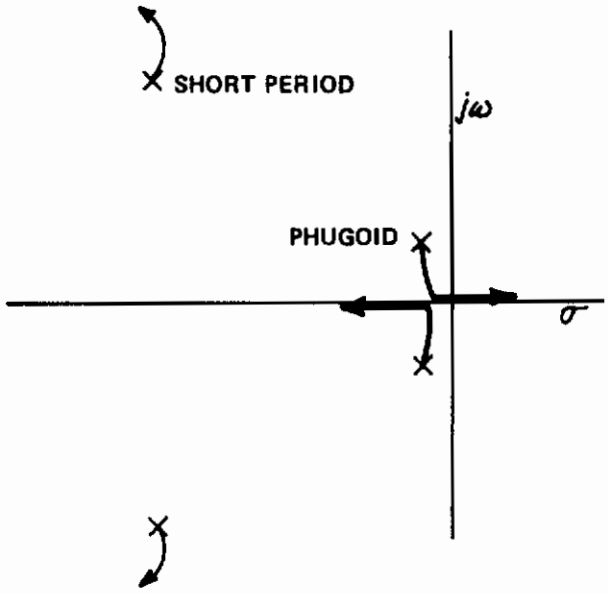
× LOCATION OF ROOTS WITH ANGLE OF ATTACK STABILITY, $M_w < 0$

□ LOCATION OF ROOTS WITH NEUTRAL ANGLE OF ATTACK STABILITY, $M_w = 0$

△ LOCATION OF ROOTS WITH $U_o M_w = Z_w M_q > 0$



(a) EFFECT OF M_w BECOMING POSITIVE, ($M_u = 0$)



(b) EFFECT OF M_u BECOMING NEGATIVE (NORMAL VALUES OF M_w)

Figure 1 (3.3.1) EFFECT OF ANGLE OF ATTACK STABILITY (M_w) AND SPEED STABILITY (M_u) ON THE ROOTS OF THE LONGITUDINAL CHARACTERISTIC EQUATION

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3.3.2 LONGITUDINAL DYNAMIC RESPONSE

REQUIREMENT

3.3.2 Longitudinal dynamic response. The following requirements shall apply to the dynamic response of the aircraft with the pitch control free and with it fixed. These requirements apply following a disturbance in smooth air, and following abrupt pitch control inputs in each direction, for responses of any magnitude that might be experienced in operational use. If the oscillations are nonlinear with amplitude, the requirements shall apply to each cycle of the oscillation.

Level 1: The response of the aircraft shall not be divergent (i.e., all roots of the longitudinal characteristic equation of the aircraft shall be stable). In addition, the undamped natural frequency, ω_n, and damping ratio, ζ, of the second-order pair of roots (real or complex) that primarily determine the short-term response of angle of attack following an abrupt pitch control input shall meet the Level 1 requirements of figure 1.

Level 2: For those Flight Phases of the operational missions of 3.1.1 for which IFR operation is required, the Level 2 requirement is the same as for Level 1. In all other cases, for Level 2, divergent modes of aperiodic response shall not double amplitude in less than 12 seconds. Oscillatory modes may be unstable provided their frequency is less than or equal to 0.84 radians per second and their time to double amplitude is greater than 12 seconds. In addition, the undamped natural frequency and damping ratio of the second-order pair of roots (real or complex) that primarily determine the short-term response of angle of attack following an abrupt pitch control input shall meet the Level 2 requirements of figure 1.

Level 3: Divergent modes of aperiodic response shall not double amplitude in less than 5 seconds. Oscillatory responses shall be stable; however, an instability will be permitted provided its frequency is less than 1.25 radians per second and its time to double amplitude is greater than 5 seconds.

DISCUSSION

This requirement is directed at all possible response modes of the aircraft. It is perhaps of some value to discuss briefly why the terms phugoid and short period dynamic characteristics do not explicitly appear in this requirement. For several V/STOL aircraft, the basic aerodynamics of the aircraft complicate the approximations that normally give use to the "conventional" phugoid and short period modes of response. The separation of these modes may in certain cases become nonexistent without augmentation,

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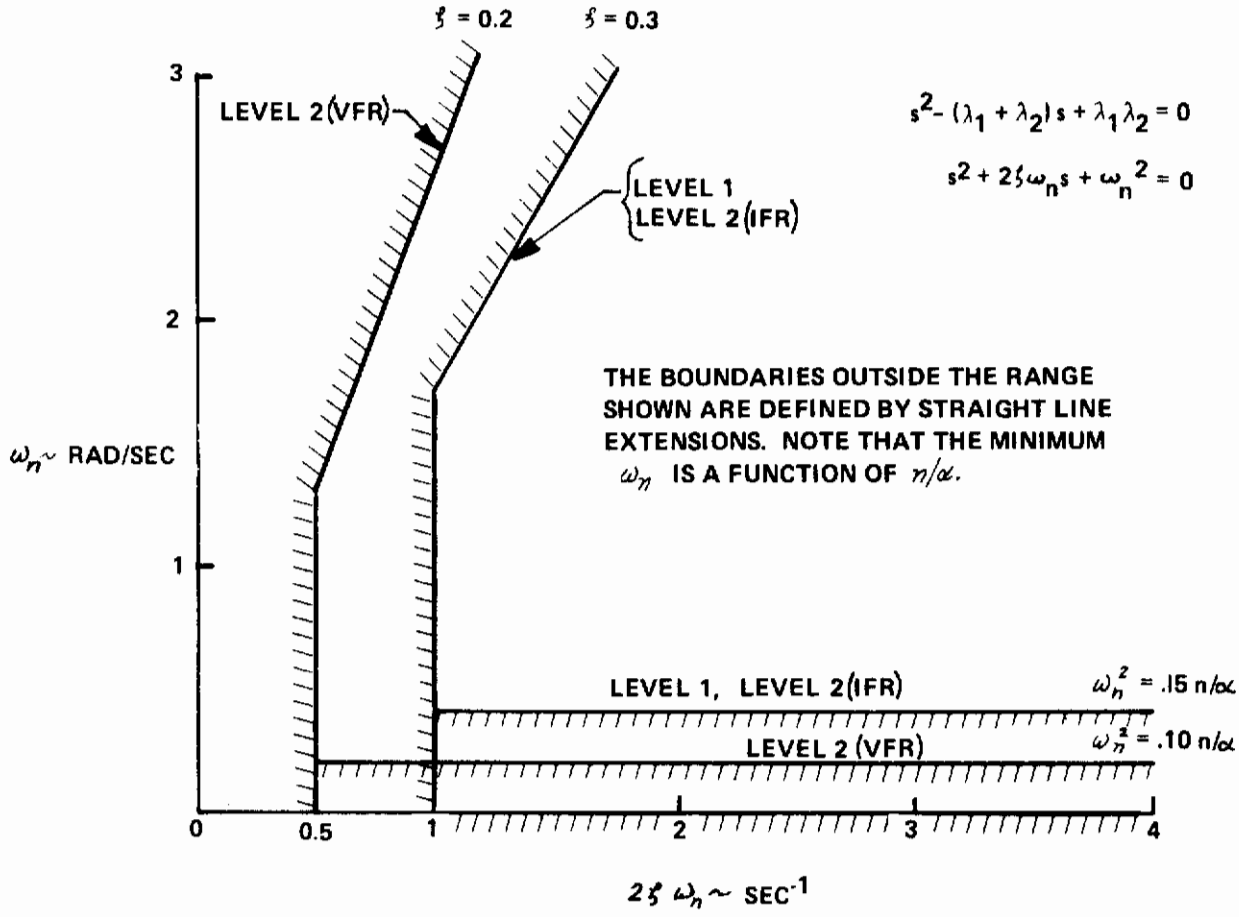

Figure 1 SHORT-TERM LONGITUDINAL RESPONSE REQUIREMENTS (FROM REFERENCE 1)

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and both modes may have a significant influence on the response of the aircraft either for a disturbance in smooth air or to an abrupt input. For those vehicles which do not have modal separation, the short term requirement can be directly interpreted as a short period requirement, while the remainder of the requirement is directed at the longer period response (phugoid mode). For these aircraft where the modes are not sufficiently separated to distinguish "short period" from "phugoid" the phrasing of the requirement directs attention at the short term response (high frequency) of angle of attack which under conventional circumstances would be associated with the short period mode of the aircraft. There are unfortunately no simple hard and fast rules to determine when the standard "short period" and "phugoid" approximations apply to a particular V/STOL aircraft. Information concerning the effects of changes in stability derivatives for V/STOL aircraft and considerations necessary to develop approximate transfer functions for various V/STOL configurations are discussed in Reference 79. Many current V/STOL aircraft rely on augmentation systems in order to provide modal separation and to achieve dynamic characteristics for satisfactory handling qualities. It is recognized that augmentation systems can, and very often do, introduce additional response modes; thus the requirement is written to also account for additional aperiodic or oscillatory modes created through the use of stability augmentation and control systems.

The basic philosophy of this requirement is to create a reasonable change in the longitudinal response parameters from the hover and low speed requirements (3.2.2) of Reference 1 to the requirements 3.2.1.2 (Phugoid Stability) and 3.2.2.1 (Short Period Response) of MIL-F-8785B, Reference 10.

It is now valuable to compare the requirements for hover and low speed flight as well as the requirements for $V > V_{con}$ with the V/STOL forward flight requirements for the longitudinal dynamics. The following table illustrates the various requirements on aperiodic stability.

	Level 1	Level 2	Level 3
Hover & Low Speed (3.2.2)	Stable	Stable(IFR) T_2 > 12 sec(VFR)	T_2 > 5 sec
Forward Flight (3.3.2)	Stable	Stable(IFR) T_2 > 12 sec(VFR)	T_2 > 5 sec
MIL-F-8785B (Reference 10)	Stable	Stable	Stable

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Aperiodic response is related to longitudinal equilibrium and the previous discussion of requirement 3.3.1 of the V/STOL Specification describes the effects of positive M_w and negative M_u on the roots of the characteristic equation. The most significant changes in the requirements on aperiodic response between the V/STOL Specification for forward flight and MIL-F-8785B are in Level 3. The Level 3 requirements of the V/STOL Specification recognize that the basic airframe for many V/STOL aircraft may be aperiodically unstable and that some instability is acceptable for Level 3 flying qualities. Data in Reference 80 indicate that time to double amplitudes (T_2) on the order of one second are acceptable for Level 3 flying qualities.

Certain data supplied by industry on a simulation of a STOL aircraft indicate that an aperiodic instability in the pitch mode with $T_2 < 5$ seconds was essentially rated Level 2, however, the exact details of the piloting task were not described.

Data from Reference 81 are plotted on Figure 1(3.3.2) using the approximations $\omega_n^2 = - M_\alpha + M_q Z_w$, and $2\zeta\omega_n = - (M_q + Z_w)$. The task in the experiment involved IFR (hooded) flight consisting of climbs, descents and turns at 40 knots and 70 knots. For this task, it can be seen in Figure 1(3.3.2) that values of T_2 between one and two seconds would satisfy Level 3 flying qualities requirements.

Data obtained during the X-22A MPE (Reference 37) showed pure aperiodic divergences with values of T_2 between 3 and 5 seconds. This degree of instability was considered unsatisfactory by the pilots. Hands-off flight for more than a few seconds was not possible, and instrument flight was considered not feasible due to excessive pilot workload.

Thus, although there exists a body of data which indicates that $T_2 <$ 5 seconds could be acceptable for Level 3 flying qualities, the result of actual flight test evaluation of the X-22A indicates that $T_2 = 5$ seconds is an appropriate level for aperiodic divergences. Discussions with military pilots at V/STOL Specification review meetings further reflected the desirability to limit the time to double amplitude of aperiodic aircraft responses to 5 seconds for Level 3, especially for IFR tasks.

The frequency and damping ratio requirements of Section 3.3.2 of the V/STOL Specification are not written in terms of "phugoid" and "short period" modes of response as in MIL-F-8785B; however, in order to compare the requirements, it is helpful to interpret the short term requirements of the V/STOL Specification as a short period mode. Other V/STOL specifications (e.g., AGARD 408A) write requirements in terms of frequency and damping ratio and use concave downward requirements to direct attention at maneuver margin, or as shown in Reference 76, short period dynamics.

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The short term requirements of the V/STOL Specification are essentially a blend of the Flight Phase B (e.g., cruise) and Flight Phase C (e.g., landing approach) requirements of MIL-F-8785B (Reference 10). The lower limit imposed on natural frequency of the short term response is a function of n/α and, as shown on Figure 1(3.3.2), separates modal parameters in a similar nature to the concave downward requirements on normal acceleration of MIL-H-8501A (Reference 15). The lower limit established for Levels 1 and 2 for $2\zeta_{sp}\omega_{sp}$ is based upon data presented in Reference 81 and illustrated on Figure 1(3.3.2). As mentioned previously, the data illustrated were plotted based on the assumptions that $\omega_{sp}^2 = - M_\alpha + M_q Z_w$ and $2\zeta_{sp}\omega_{sp} = - (M_q + Z_w)$, and the experiment involved a task wherein speed was varied between 40 and 70 knots.

Figure 2(3.3.2) compares some results from Reference 82 with the short term V/STOL requirements. The Level 1 boundary for approach-path stability and control (Flight Phase Category C) appears to be bounded better by the minimum damping requirement of Flight Phases A and C of MIL-F-8785B ($\zeta_{sp} = 0.35$) than by the more lenient requirement of the V/STOL Specification ($\zeta_{sp} = 0.30$). In addition, the Level 3 boundary for the data in Reference 82 is more restrictive than the V/STOL Specification. These data illustrate that the V/STOL requirement on short term dynamic requirements is perhaps the most lenient requirement that can be specified based upon the existing data and suggest the need for further investigation of the short period dynamics of V/STOL aircraft by Flight Phase Category.

Figures 3(3.3.2) through 5(3.3.2) indicate a comparison of the V/STOL Specification short term dynamic requirements for forward flight with those of References 46, 47, and 15, and also serve to indicate the differences in the dynamic requirements of other V/STOL Specifications.

The short term response requirements of the V/STOL Specification are reasonably substantiated by the little available data and are written to blend reasonably with the low speed requirements (3.2.2) of the V/STOL Specification at 35 knots and with the short period requirements of MIL-F-8785B at V_{con}. It should be noted that one significant difference between the V/STOL Specification and MIL-F-8785B is that the V/STOL requirements for short term dynamic response parameters are open-ended. That is, they state only the minimum while both the maximum and the minimum are specified in MIL-F-8785B for conventional aircraft. This is a result of the shortage of experimental data to justify such additional requirements. In addition it should be remembered that for many V/STOL aircraft the short term dynamics are in reality the combination of aperiodic roots, and in the speed range from 35 knots to V_{con}, it is very probable that the aircraft will possess a short period damping ratio in excess of values considered desirable for conventional flight. Thus to impose a maximum damping ratio requirement might be physically unrealistic. The minimum frequency requirement should help to prevent configurations that would be too sluggish in response to control inputs.

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The previous discussions were related to aperiodic modes and short-term (e.g., short period, high frequency) modes of aircraft response. The following is directed at the low frequency, phugoid-type modes.

Again the fundamental philosophy is to establish requirements that will blend between the low speed requirements (3.2.2) of the V/STOL Specification and the phugoid requirements (3.2.12) of MIL-F-8785B. The following table illustrates the various requirements for low frequency oscillatory modes.

	Level 1	Level 2	Level 3
Hover & Low Speed (3.2.2)	$\omega > .5 \frac{rad}{sec}$, $\zeta > 0$ $\omega < .5$, $\zeta > -.1$	IFR: same as Level 1 VFR: $\omega > .84$, $\zeta > 0$ $\omega < .84$, $T_2 > 12$	$\omega > 1.25$, $\zeta > 0$ $\omega < 1.25$, $T_2 > 5$
Forward Flight (3.3.2)	Stable	IFR: same as Level 1 VFR: $\omega > .84$, $\zeta > 0$ $\omega < .84$, $T_2 > 12$	same as 3.2.2
MIL-F-8785B (Reference 10)	$\zeta_p \geq 0.04$	$\zeta_p \geq 0$	$T_2 \geq 55$

Thus the low frequency forward flight longitudinal oscillatory requirement represents a reasonable interface between the hover and low speed requirements of Reference 1 and the requirements on phugoid stability of Reference 10. Comments in Reference 73 indicate that many of the STOL aircraft evaluated by NASA had low frequency modes with near neutral damping and period greater than 20 seconds and caused no problems to the pilot in the evaluation programs. For Levels 2 and 3, specifying for forward flight the same requirements on the low frequency oscillatory mode as for hover and low speed establishes a lower bound on phugoid stability.

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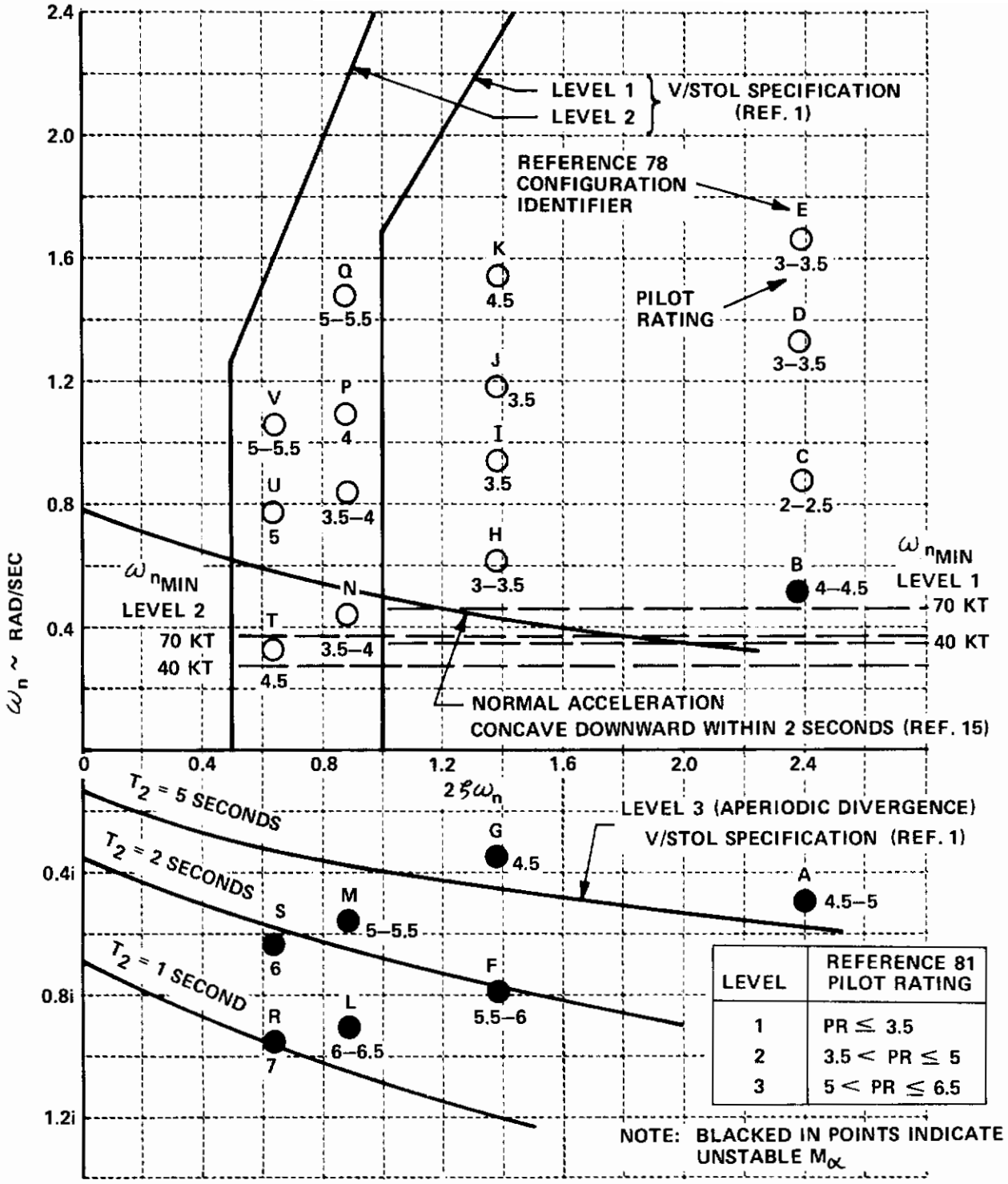

Figure 1 (3.3.2) COMPARISON OF RESULTS OF REFERENCE 81 WITH SHORT TERM LONGITUDINAL RESPONSE REQUIREMENTS OF V/STOL SPECIFICATION

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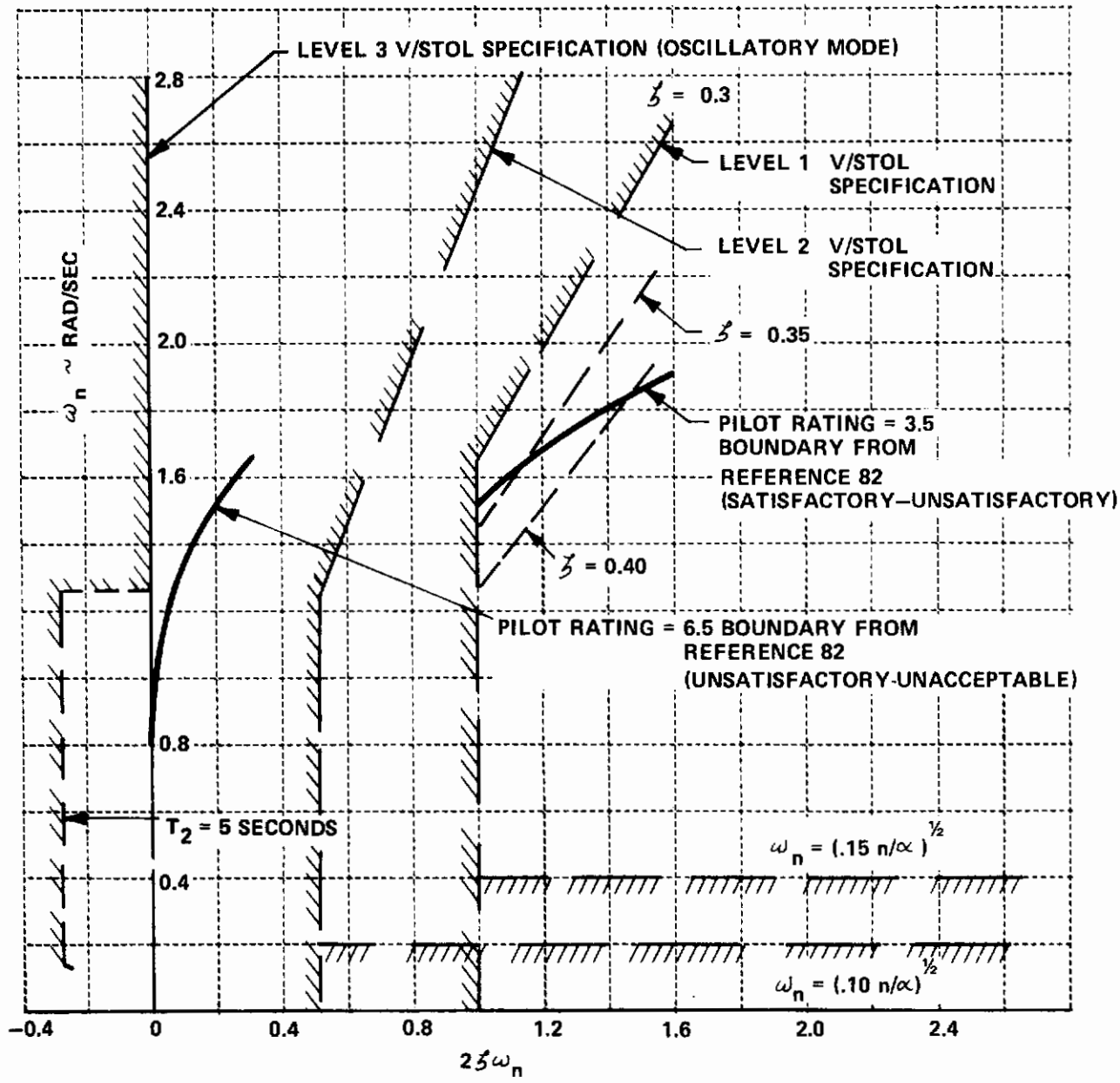

Figure 2 (3.3.2) COMPARISON OF PILOT OPINION BOUNDARIES OF REFERENCE 82 WITH V/STOL SPECIFICATION SHORT TERM RESPONSE REQUIREMENTS

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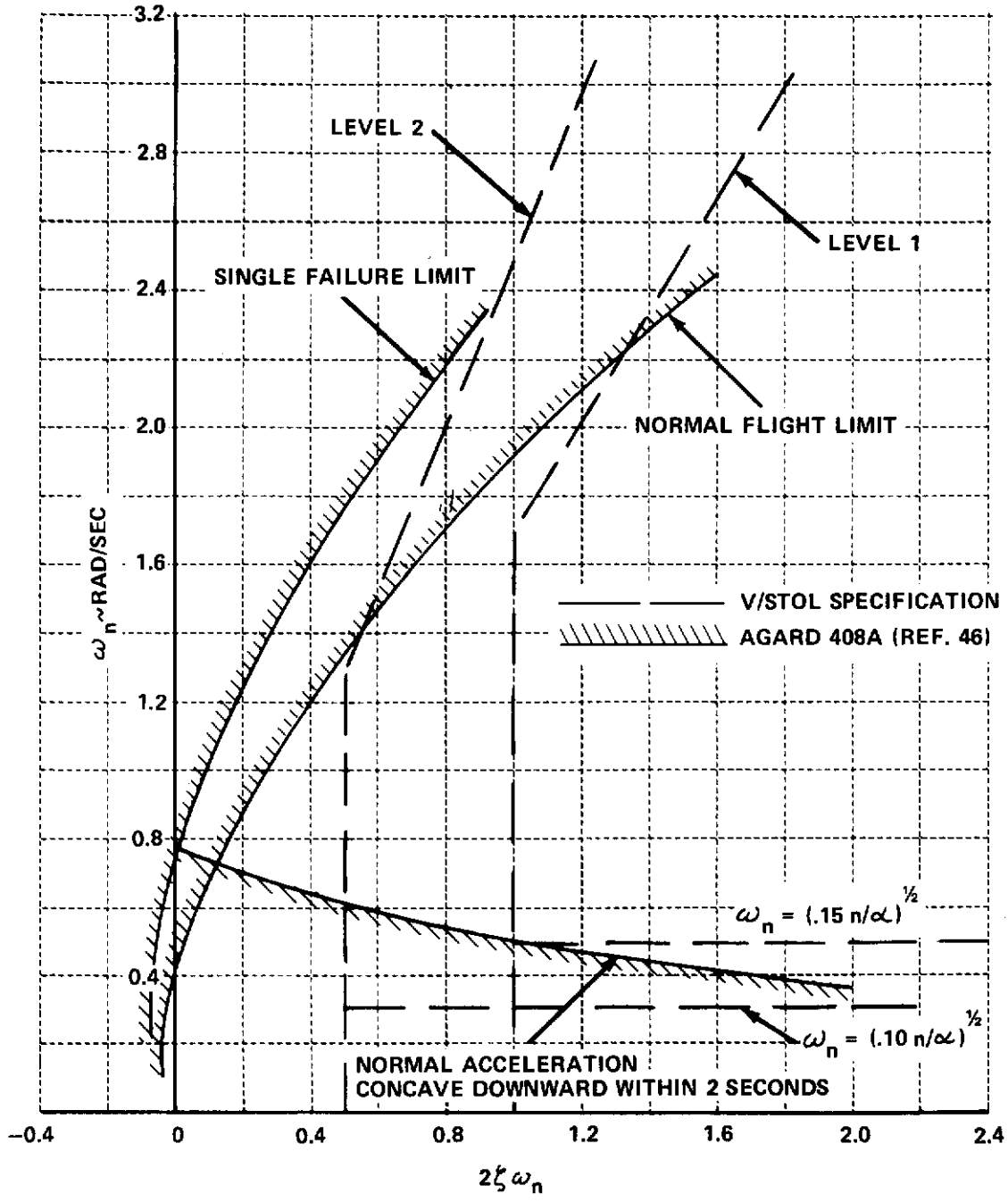

Figure 3 (3.3.2) DYNAMIC REQUIREMENTS OF AGARD 408A (REFERENCE 46) COMPARED TO V/STOL SPECIFICATION SHORT TERM RESPONSE REQUIREMENTS

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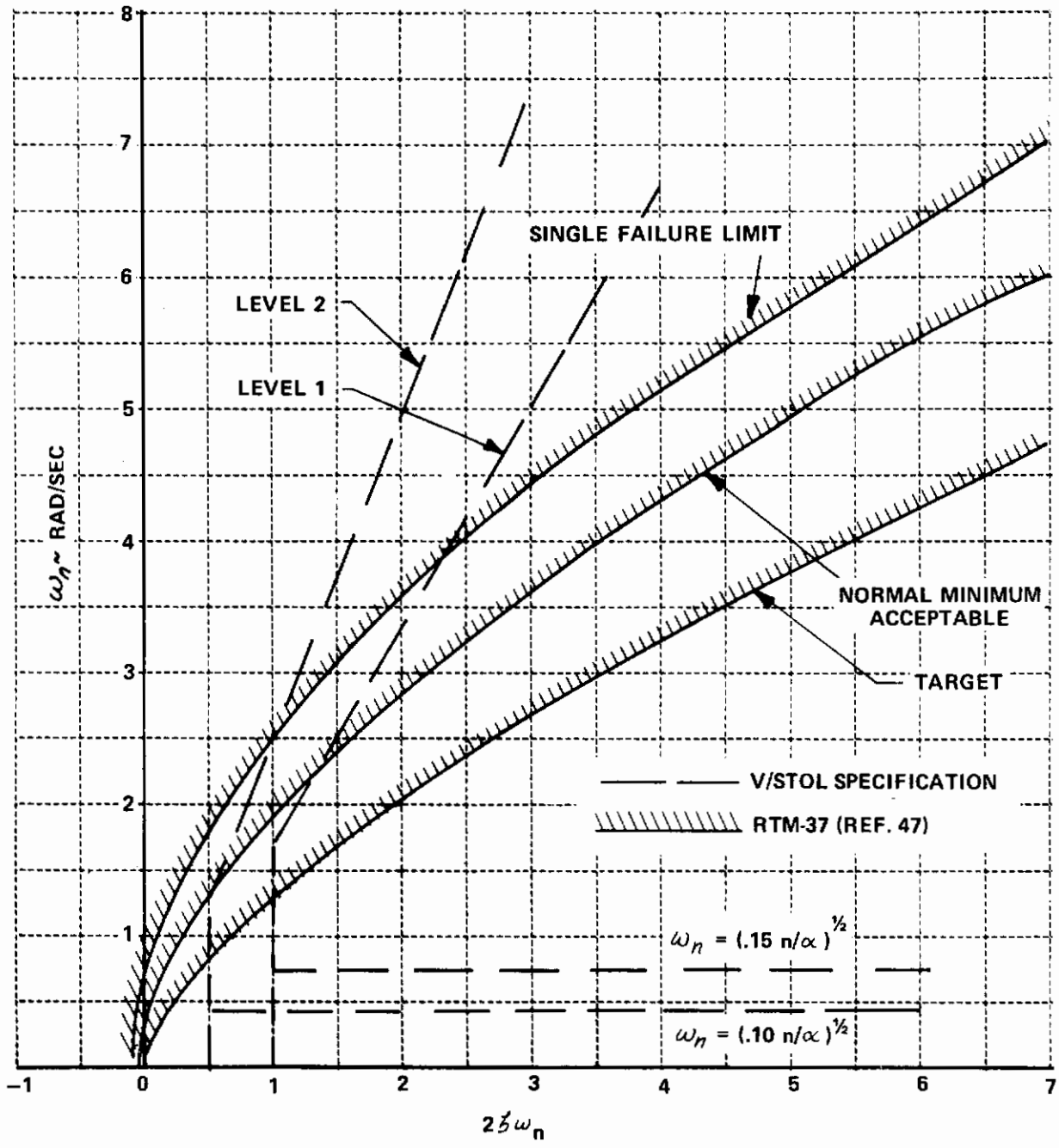

Figure 4 (3.3.2) DYNAMIC REQUIREMENTS OF RTM-37 (REFERENCE 47) COMPARED TO V/STOL SPECIFICATION SHORT TERM RESPONSE REQUIREMENTS

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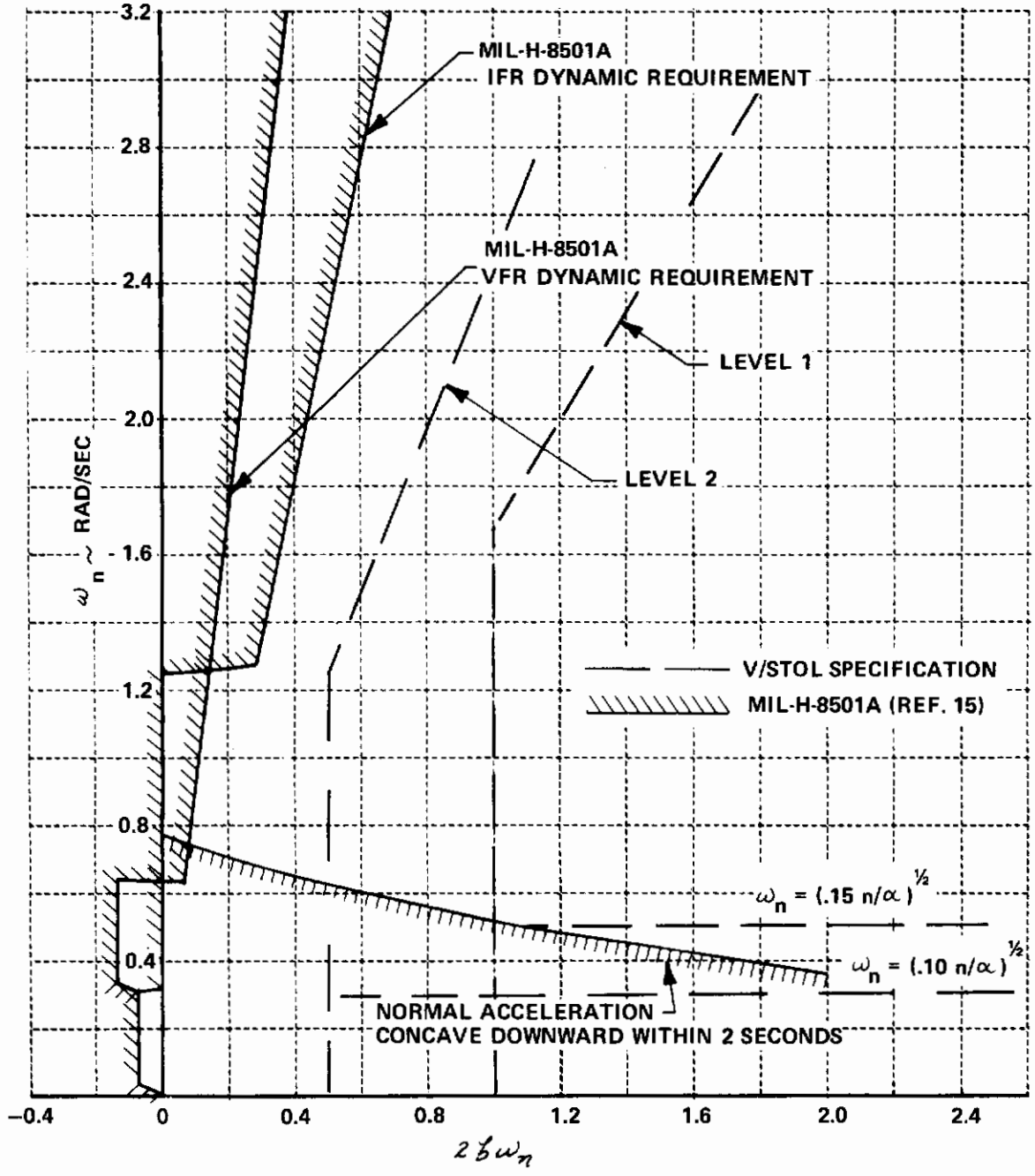

Figure 5 (3.3.2) DYNAMIC REQUIREMENTS OF MIL-H-8501A (REFERENCE 15) COMPARED TO V/STOL SPECIFICATION SHORT TERM RESPONSE REQUIREMENTS

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3.3.3 RESIDUAL OSCILLATIONS

REQUIREMENT

3.3.3 Residual oscillations. Any sustained residual oscillations shall not interfere with the pilot's ability to perform the tasks required in operational use of the aircraft. For Levels 1 and 2, oscillations in normal acceleration at the pilot's station greater than ± 0.05g will be considered excessive for any Flight Phase. These requirements shall apply with the pitch control fixed and with it free.

DISCUSSION

This requirement is essentially the same as 3.2.2.1.3 of MIL-F-8785B (Reference 10). The primary purpose of the requirement is to direct attention at limit cycles in the control system and structural oscillations which might affect pilot performance in the tactical mission, cause pilot discomfort, etc.

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3.3.4 PITCH CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

REQUIREMENT

3.3.4 Pitch control feel and stability in maneuvering flight. In steady turning flight, and in pullups at constant speed, increased pull forces and aft displacement of the cockpit pitch control shall be required to maintain increases in angle of attack, normal acceleration and nose-up pitch rate throughout the range of angle of attack and load factor in the Service Flight Envelope. Increases in push forces and forward displacement of the cockpit pitch control shall be required to maintain reductions of angle of attack and normal acceleration in pushovers at constant speed.

3.3.4.1 Pitch control forces in maneuvering flight. In steady turning flight, in pullups, and in pushovers, at constant speed, the variation in pitch control force with steady-state normal acceleration or angle of attack shall be approximately linear. In general, 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. For Levels 1 and 2 the local value of the pitch control force gradient with normal acceleration shall never be less than 3 pounds per g. There shall be no undesirable inputs to the pitch control system due to changes in linear or angular accelerations produced by gusts or thrust magnitude control inputs. The term gradient does not include that portion of the force versus normal-acceleration or angle-of-attack curve within the preloaded breakout force or friction band.

DISCUSSION

The intent of these requirements is to provide the pilot with proper cues while he maneuvers the aircraft about a fixed operating point. Although at low speeds, normal acceleration might not be as important to the pilot as pitch rate, stable control force and position variations with normal acceleration at constant speed ensure that the aircraft has a short-term mode that will exhibit a restoring tendency which will return the aircraft to 1-g flight following a disturbance. This relationship is analogous to that between control force and position variation with speed and the low-frequency (phugoid) mode of aircraft response. These requirements are similar to those appearing in MIL-F-8785B (Reference 10) and AGARD 408A (Reference 46). In addition, the concave downward requirement on normal acceleration of MIL-H-8501A is equivalent to maneuver stability for conventional aircraft (Reference 83). Although it is stated in Reference 83 that the "divergence" criterion should not be considered as an alternate criterion for oscillatory stability, the analysis in Reference 76 illustrates that there are root locations in the s-plane which would be prohibited by the concave downward requirement on normal acceleration as stated in MIL-H-8501A. There appears to be a difference of opinion as to the ease of measuring acceleration trim gradients between References 76 and 83, although the intent of the MIL-H-8501A requirement is an attempt to write a requirement directed at acceleration trim gradients (i.e., maneuver margin). Reference 50 summarizes several of the existing reports relating to maneuvering stability and recommends that

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the concave downward requirement (3.2.11.1) of MIL-H-8501A (Reference 15) be modified, since the intent of this requirement is to ensure constant-speed maneuvering stability. Based on this background, it was decided to formulate the V/STOL Specification requirement in terms of constant speed maneuvering stability, essentially based on the related requirements of MIL-F-8785B. Thus requirement 3.3.4 of the V/STOL Specification requires positive stick-fixed and stick-free maneuver margin.

The minimum value of pitch control force gradient with normal acceleration of 3 pounds per g for Level 1 and 2 further defines the desired minimum Level of positive stick-free maneuver margin. This value is the minimum value acceptable to satisfy the MIL-F-8785B requirements on maneuvering flight and is necessary to ensure that if the aircraft can develop load factor by application of the pitch control, the response of the vehicle can be adequately controlled. It should be noted that the V/STOL requirement makes no distinction between stick controllers and wheel controllers, since it has been assumed that due to the general complexity of the piloting task for V/STOL aircraft, only one hand would be available to make pitch control inputs for maneuvering.

For many V/STOL aircraft between 35 knots and V_{con}, it is quite probable that load factor changes will be primarily dependent upon application of the thrust magnitude and thrust angle controls. Thus, for these aircraft n/F_s for a pitch control input can approach zero (i.e., $[F_s/n] \rightarrow \infty$). For this reason no upper limit has been established on the acceleration trim gradient at constant speed.

Reference 37 indicates that as speed is reduced, normal acceleration cues to the pilot become less perceptible and pitch rate becomes a more obvious cue to the pilot. Thus at low speeds it may be more desirable to place requirements on $F_s/\dot{\theta}$ in order to establish limits on force gradient in constant speed maneuvers. An upper limit would be useful, since too large a force gradient would increase pilot workload and could result in a sluggishly responding aircraft. AGARD 408A (Reference 46) requires an upper limit of 20 lb/g, while Reference 47 expresses requirements in terms of control force per g as a function of design positive limit load factor and also in terms of $\dot{\theta}$ per pound of control force. Unfortunately there are insufficient data on V/STOL aircraft to place a requirement on either a maximum value of F_s/n or $F_s/\dot{\theta}$ at speeds between 35 knots and V_{con}.

It should be remembered that the requirement places a limit on the minimum value of F_s/n, to aid in maneuvering control of the aircraft and to hopefully avoid pilot-induced oscillations in longitudinal maneuvers. This should not be interpreted as a desired level of the stick force acceleration gradient. There are several considerations relating to a desired level of stick force/g. For example, if an aircraft had a lightly damped,high frequency, short period mode it might be necessary to provide a stick force acceleration gradient in excess of 3 lb/g for Levels 1 and 2 in an attempt to avoid abrupt response, excessive sensitivity, and tendencies toward pilot-induced oscillations. Further the Flight Phase Category and Aircraft Class could influence

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the desired level of stick force/g. Based on the data presented in Reference 84 as background to the MIL-F-8785B requirements on maneuver margin, the selection of 3 lb/g appears to be a reasonable lower limit for Levels 1 and 2 for V/STOL aircraft.

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3.3.5 PITCH CONTROL EFFECTIVENESS IN MANEUVERING FLIGHT

REQUIREMENT

3.3.5 Pitch control effectiveness in maneuvering flight. When the aircraft is trimmed in unaccelerated flight at any speed and altitude in the Operational Flight Envelope, it shall be possible to develop at the trim speed the limiting attitude or angle of attack of the Operational Flight Envelope.

3.3.5.1 Maneuvering control margins. When automatic stabilization and control equipment or devices are used to overcome an aperiodic instability of the basic aircraft, both the magnitude of the instability and the installed control power shall be such that at least 50 percent of the nominal control moment is available to the pilot in the critical direction through the use of the pitch control. This requirement applies throughout the Service Flight Envelope within ± 15 knots TAS or 15 percent of the trim speed, whichever is greater.

3.3.5.2 Speed and flight-path control. The aircraft dynamic characteristics, together with the effectiveness and response times of the pitch, thrust magnitude, and thrust angle controls, shall be such that adequate control of the flight path and airspeed can be maintained at all permissible angles of attack and load factors.

DISCUSSION

Requirement 3.3.5 of Reference 1 is essentially the same as 2.13 of AGARD 408A (Reference 46), except that the pitch control effectiveness requirement is now only imposed within the Operational Flight Envelope rather than at all permissible speeds. Thus, outside the Operational Flight Envelope, what falls out of the design is now considered acceptable for pitch control effectiveness. The requirement of 3.3.5 is related to the requirements of 3.2.3.2 of MIL-F-8785B (Reference 10) except that at the speeds consistent with application of the V/STOL Specification, pitch control effectiveness is directed at attitude and angle of attack rather than load factor.

Requirement 3.3.5.1 is equivalent to some of the content in both 2.2 and 2.4 of AGARD 408A (Reference 46). Aperiodic instabilities in particular have been singled out because of their insidious nature. If an aircraft with aperiodic instability is flown away from trim, the control surface deflection needed to return to trim is in the same direction as that used to maintain the perturbation. Thus in this situation an automatic stabilizing device could use all the control power available and thus leave the pilot with no means to recover. Since the augmentation device is providing stick position stability, the stick could move forward, for example, while the control surface moves in the opposite sense and the pilot would not be aware of the reduction in control power in the recovery direction. It should be noted that this requirement is intended to apply only when an aperiodic instability exists and it is necessary to augment the aircraft in order to achieve a desired Level of flying qualities. The speed region about trim wherein this requirement applies has

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been restricted based on practical considerations of the regions of aperiodic instability associated with V/STOL aircraft, augmentation system design, and anticipated maneuvering requirements about a fixed operating point in the speed range from 35 knots forward to V_{con}.

The intent of 3.3.5.2 is obvious, and has been expressed as a very general, qualitative requirement. The requirement as stated exemplifies the need for additional research on the multi-input, multi-output control situations associated with V/STOL aircraft and the possible changing nature in the modes of aircraft response to control inputs as the aircraft is flown between 35 knots forward and V_{con}.

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3.3.6 PITCH CONTROL IN SIDESLIPS

REQUIREMENT

3.3.6 Pitch control in sideslips. With the aircraft trimmed for straight flight with zero sideslip, the pitch control force required to maintain constant speed in steady, constant-heading sideslips, shall not exceed one-third of the limits of 3.5.3 in the pull direction or one-sixth of the limits of 3.5.3 in the push direction. This requirement shall be met for sideslips up to the magnitude specified in 3.3.11 or the magnitude which can be generated by 50 pounds of yaw control force, whichever is less. 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. In addition, in the steady sideslip conditions specified above, a margin of at least 20 percent of the nominal control moment in pitch shall be available as an allowance for the control of gust disturbances. This requirement applies in level flight and in climbs and descents to the limits of the appropriate Flight Envelopes.

DISCUSSION

This requirement is essentially a combination of the ideas in 2.18 of AGARD 408A (Reference 46) and 3.2.3.7 of MIL-F-8785B.

There are two primary reasons for writing requirements on the maximum pitch control force in sideslips. The first is to ensure that small amounts of sideslip inadvertently developed during normal operations do not result in excessive or possibly dangerous angle-of-attack or pitch attitude changes. The second reason is simply to limit longitudinal corrections required when the pilot intentionally changes the sideslip angle, as in crosswind operation. The intent of the control margin portion of the requirement is to enable the pilot to correct attitude changes resulting from atmospheric disturbances which could compromise the precision of flight path control, and it is based on the AGARD 408A (Reference 46) requirement.

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3.3.7 LATERAL-DIRECTIONAL CHARACTERISTICS

GENERAL

Before discussing the lateral-directional requirements of the V/STOL Specification (Reference 1), it is perhaps useful to examine the philosophy adopted in treating the whole problem of lateral-directional control at speeds between 35 knots and V_{con}. It has been assumed that the "conventional" modes of motion have emerged from the hovering cubic situation, and that "airplane-like" responses and hence "airplane-like" handling qualities parameters are appropriate, modified as necessary by low-speed considerations. This allows requirements to be placed on the modal parameters, i.e., the roots of the characteristic equation. However, because of coupling between lateral and directional motions, each lateral-directional requirement can have implications in many areas of flying qualities. Conversely, each flying qualities area is generally a function of many different parameters. For example, the rolling moment acting on an aircraft following a roll control command input is a function of:

- directional stability
- yaw due to aileron
- dihedral characteristics
- roll effectiveness characteristics, etc.

Another example is that the desired directional stability characteristics of an aircraft can be dependent upon the dynamic effects of coupling between roll and sideslip following a roll control command input.

The way such interactions are studied is by changing one parameter at a time while keeping all the other parameters at values known to be good or near optimum. Thus the Dutch roll mode characteristics specified in 3.3.7.1 of the V/STOL Specification are "basic" requirements, that is, they have been developed from data that were not significantly degraded by either coupling or turbulence effects. Other documents describe additional requirements, for example on the total damping of the Dutch roll mode, based on considerations of turbulence (e.g., References 10 and 84) or on the additional damping that is a function of lateral-directional coupling (e.g., Reference 85). For the V/STOL Specification, as in MIL-F-8785B (Reference 10), the requirements associated with roll-sideslip coupling in response to roll control commands are written in terms of the desired aircraft responses rather than in terms of additional requirements on modal parameters (3.3.8 of V/STOL Specification, Reference 1). Another example of this approach is that the spiral stability requirements are written in terms of the time it takes for the bank angle to double rather than in terms of a first-order time constant. Thus the requirement includes control system and trim effects which are considered to be more representative of the spiral behavior, as noted by the pilot, than the first-order spiral mode time constant. This approach yields considerable insight into the effects of the coupling parameters, and is useful to the aircraft designer when developing design trade-offs and to the

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activity responsible to check compliance since the measures used are easily obtainable from flight test records and are directed at what the pilot experiences in flight.

The philosophy of developing requirements on a particular parameter when the effects of other parameters have been minimized then leads to an interesting question. While this approach may be valid if the vehicle meets all Level 1 requirements, what is the Level of flying qualities for example for the roll-sideslip requirements if the Dutch roll mode is Level 2? In other words, if the vehicle meets the Level 1 ϕ_{osc}/ϕ_{4V} requirement but has Level 2 Dutch roll mode characteristics, what would be the resulting Level of flying qualities for the aircraft? This question in reality is related to the more general question, when two or more parameters are degraded together (e.g., to Level 2) what is the resultant overall Level of flying qualities, (Level 2 or Level 3)? In general, this question cannot be answered based on the presently available data. Some indication of the resulting pilot rating from combining requirements for configurations examined in Reference 86 is presented in the discussion of requirement 3.3.8.1 of the V/STOL Specification, and illustrated on Figure 11(3.3.8.1). These data show that two Level 2's can indeed give a pilot rating equivalent to Level 3, though this is not always the result. This example serves to show that a configuration having a series of Level 2 characteristics should be investigated very carefully and that it is certainly possible that an aircraft designed to satisfy most of the Level 2 requirements could actually be a Level 3 aircraft.

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3.3.7.1 LATERAL-DIRECTIONAL OSCILLATIONS (DUTCH ROLL)

REQUIREMENT

3.3.7.1 Lateral-directional oscillations (Dutch roll). The frequency, ω_{n_d}, and damping ratio, ζ_d, of the lateral-directional oscillations following a disturbance input, for example a yaw control doublet, shall exceed the minimums specified on figure 2. The requirements shall be met with controls fixed and with them free for oscillations of any magnitude that might be experienced in operational use. If the oscillation is nonlinear with amplitude, the requirements shall apply to each cycle of the oscillation. Residual oscillations may be tolerated only if the amplitude is sufficiently small that the motions are not objectionable and do not impair mission performance. With control surfaces fixed, $\omega_{n_d}^2$ shall always be greater than zero.

DISCUSSION

The intent of this requirement is to establish Dutch roll modal parameters for the forward flight speed regime, that are not only compatible with the hover and low speed requirements (3.2.2) of the V/STOL Specification and the requirements of 3.3.1.1 of MIL-F-8785B at V_{con}, but in addition, reflect the limited data available for V/STOL aircraft in this speed region.

The requirement is primarily based on the results of a recent in-flight investigation of lateral-directional handling qualities in low speed maneuvering flight (Reference 86). Figure 1(3.3.7.1) presents the pilot ratings for several of the configurations investigated in Reference 86. Those data points presented are for configurations where pilot comments indicated that the rating was not downgraded by requirements appearing in other sections of the V/STOL Specification (e.g., the roll-sideslip coupling requirements of Section 3.3.8) and primarily reflect the influence of the Dutch roll modal parameters. It should be noted that the roll mode and spiral mode time constants were held fixed during the in-flight investigation at values compatible with the Level 1 requirements of 3.3.7.2 and 3.3.7.3 respectively of the V/STOL Specification. The low frequency (ω_{n_d} = .25 radian/second) data obtained during the in-flight experiment indicates that for Level 1 flying qualities in tasks that require precise flight-path control (Flight Phase Categories A and C), a minimum level of ω_{n_d} should be specified. For the V/STOL Specification this minimum ω_{n_d} was selected at .25 rad/sec. The value selected is based on the data presented in Reference 86, that in general, for maneuvering flight, a value of ω_{n_d} = .25 rad/sec was not rated Level 1 regardless of damping ratio[Figure 1(3.3.7.1)]. MIL-F-8785B (Reference 10) establishes minimum values of ω_{n_d} of either 0.4 or 1.0 rad/sec based on Class and Flight Phase Category. The limited data available for V/STOL aircraft do not permit establishment of minimum values of ω_{n_d} based on Class.

The purpose of requiring that $\omega_{n_d}^2 > 0$ with control surfaces fixed is to prescribe a short-term restoring tendency in yaw for the basic, unaugmented aircraft, i.e., "weathercock" stability ($N_\beta > 0$) for the bare airframe.

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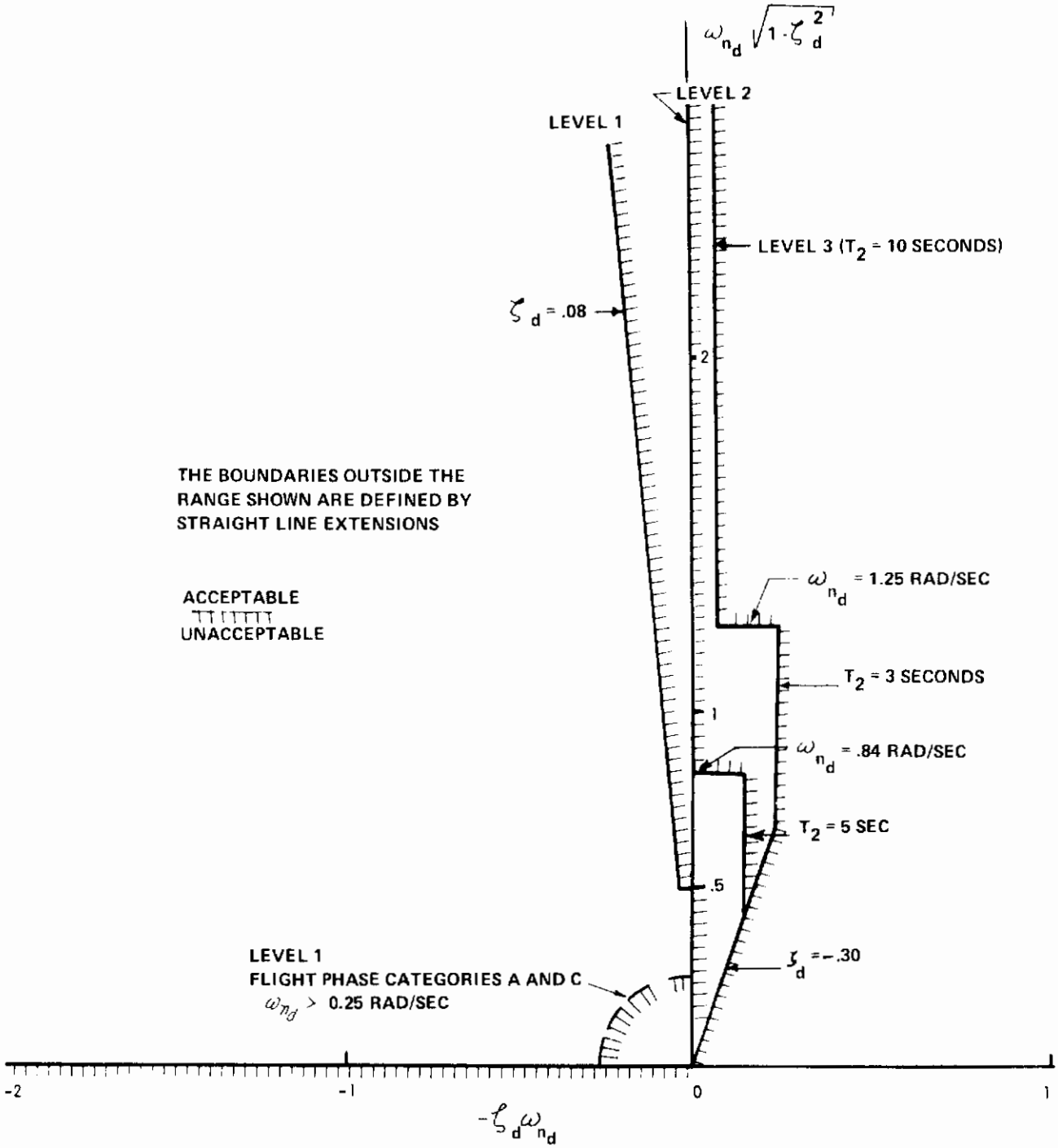

Figure 2 LATERAL - DIRECTIONAL OSCILLATORY REQUIREMENTS (FROM REFERENCE 1)

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Although using the device of $\omega_{n_d}^2$ to prescribe N_β might appear at first glance to be an unfortunate choice of notation, it should be remembered that for a first approximation to the Dutch roll mode obtained by setting $\phi = 0$ and the assumption of straight line motion, $\omega_{n_d}^2 = N_\beta$ (Reference 76). Flight tests conducted by NASA (Reference 87) indicated that the minimum value of "static" directional stability is a function of dihedral effect, however, the report concluded that a minimum value of "static" directional stability (i.e., weathercock stability) for the tests conducted, was $N_\beta = .3 \frac{rad/sec^2}{rad}$. Since the data available are limited, the V/STOL Specification only requires positive stability without specifying a desired value for "weathercock" stability.

Reference 73 contains a compilation of various NASA investigations into the airworthiness of STOL aircraft. This report recommends that, for satisfactory operation, the Dutch roll period should be less than 12 seconds, and the time to half amplitude for the Dutch roll mode should be less than 8 seconds; for safe operation, the Dutch roll mode should be stable.

The data contained in Reference 73 for Dutch roll modal parameters are illustrated on Figure 2(3.3.7.1), together with the Level 1 boundary of the V/STOL Specification. It should be noted that, as explained in Reference 73, many of the poorer ratings are associated with roll-sideslip coupling or unsatisfactory spiral mode characteristics and do not entirely reflect problems associated with the Dutch roll mode. In addition, the data indicate the results of utilizing improper augmentation (e.g., for the NC-130B using augmentation, the Dutch roll mode damping ratio was improved but resulted in large sideslip angles in steady turns). In general, the NASA criteria are overly restrictive for long period Dutch roll modes when compared with the results of Reference 86 shown on Figure 1 (3.3.7.1). In addition, the criteria of Reference 73 reject the augmented 367-80 and the sideslip rate augmented NC-130B, both of which had pilot ratings of 3.5. The data presented by NASA in Reference 73 are in general agreement with the V/STOL Specification at moderate Dutch roll frequencies.

Dutch roll oscillation data for several investigations have been collected and analyzed in Reference 85. The author separates Dutch roll modal requirements into "basic" and "additional". The additional requirements are generally directed at desired aircraft response, while the "basic" requirements are essentially directed at open-loop modal behavior. Reference 85 concludes that "basic" damping requirements appear to be specified better in terms of total damping then damping ratio, i.e., $\zeta_d \omega_{n_d}$, rather than ζ_d. In addition, for satisfactory handling qualities (Level 1) the value of $\zeta_d \omega_{n_d}$ should be greater than 0.2 to 0.3 for all frequencies between 0.8 and 6 radians/second, and unsatisfactory pilot ratings will occur for the same frequencies when the aircraft is neutrally damped.

An examination of the data in Reference 85 from the point of view of ζ_d rather than $\zeta_d \omega_{n_d}$ shows that the correlation of satisfactory pilot rating with $\zeta_d \geq 0.08$ is at least as good as $\zeta_d \omega_{n_d} > 0.2$. In fact as frequency

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increases the data indicates a trend requiring increased damping ratio rather than keeping $\zeta_d \omega_{n_d}$ constant.

The possible need for increasing ζ_d as ω_d increases above about 2.5 radians/second is also indicated by data in Reference 88. This study examined several configurations with Dutch roll frequency of 3 radians/second and pilot ratings were 3.9 to 4.3 on the Cooper scale, when ζ_d = 0.1; however, when ζ_d was increased to 0.4, pilot rating varied between 3.1 and 3.4. It should be noted that these ratings were relatively insensitive to $|\phi/\beta|_d$.

The specified value of ζ_d = .08 for $\omega_{n_d}\sqrt{1-\zeta_d^2} > .5$ (Level 1) is generally consistent with existing data as previously indicated and is compatible with MIL-F-8785B requirements. It appears that the required level of Dutch roll damping might be insufficient at Dutch roll frequencies above approximately 2.5 rad/sec due to turbulence response, however, insufficient data are available for V/STOL aircraft to impose any additional requirement at this time and further research is required.

Figure 3(3.3.7.1) illustrates data obtained from both simulation studies and in-flight research, conducted by NASA (Reference 89), on the lateral-directional characteristics of transport aircraft operating at STOL airspeeds. This illustrates that, although pilot rating can degrade significantly as the Dutch roll mode becomes unstable, mild instability is acceptable for Levels 2 and 3 flying qualities in the landing approach at speeds between 80 and 90 knots. This information tends to support the data obtained from the in-flight experiment (Reference 86) which was conducted at approximately 50 knots.

Figure 4(3.3.7.1) is a comparison of the Dutch roll modal characteristics of the X-22A with the requirements of the V/STOL Specification, based on taking a mean line through the data presented on Figure 16 of Reference 37. Although pilot rating data were not presented in Reference 37 for the Dutch roll characteristics, it is noted that increasing or decreasing airspeed from 80 knots (the 30 degree duct angle condition) improved Dutch roll damping characteristics, and Reference 90 reports that the stability characteristics of the Dutch roll oscillatory mode were acceptable for the flight research role.

Dutch roll oscillatory characteristics of the AH-1G (Huey Cobra) are presented on Figure 5(3.3.7.1) based on the data obtained in a Phase B test reported in Reference 91. The reference describes the Dutch roll mode as excellent with SCAS-ON. With SCAS-OFF, the low lateral-directional damping was not considered sufficiently severe to limit the safe flight envelope of the helicopter, however, it is noted that mission effectiveness with SCAS-OFF can be degraded. These comments are quite consistent with the Level 2 definition, and the data presented agree quite well with the V/STOL requirements.

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Figures 6(3.3.7.1) through 12(3.3.7.1) indicate how several existing V/STOL aircraft and helicopters compare to the Dutch roll requirement of the V/STOL Specification. These data are based upon the modal parameters tabulated in References 13 and 16. This information is presented to indicate general trends of existing V/STOL aircraft and cannot be evaluated because of the lack of pilot ratings and pilot comments.

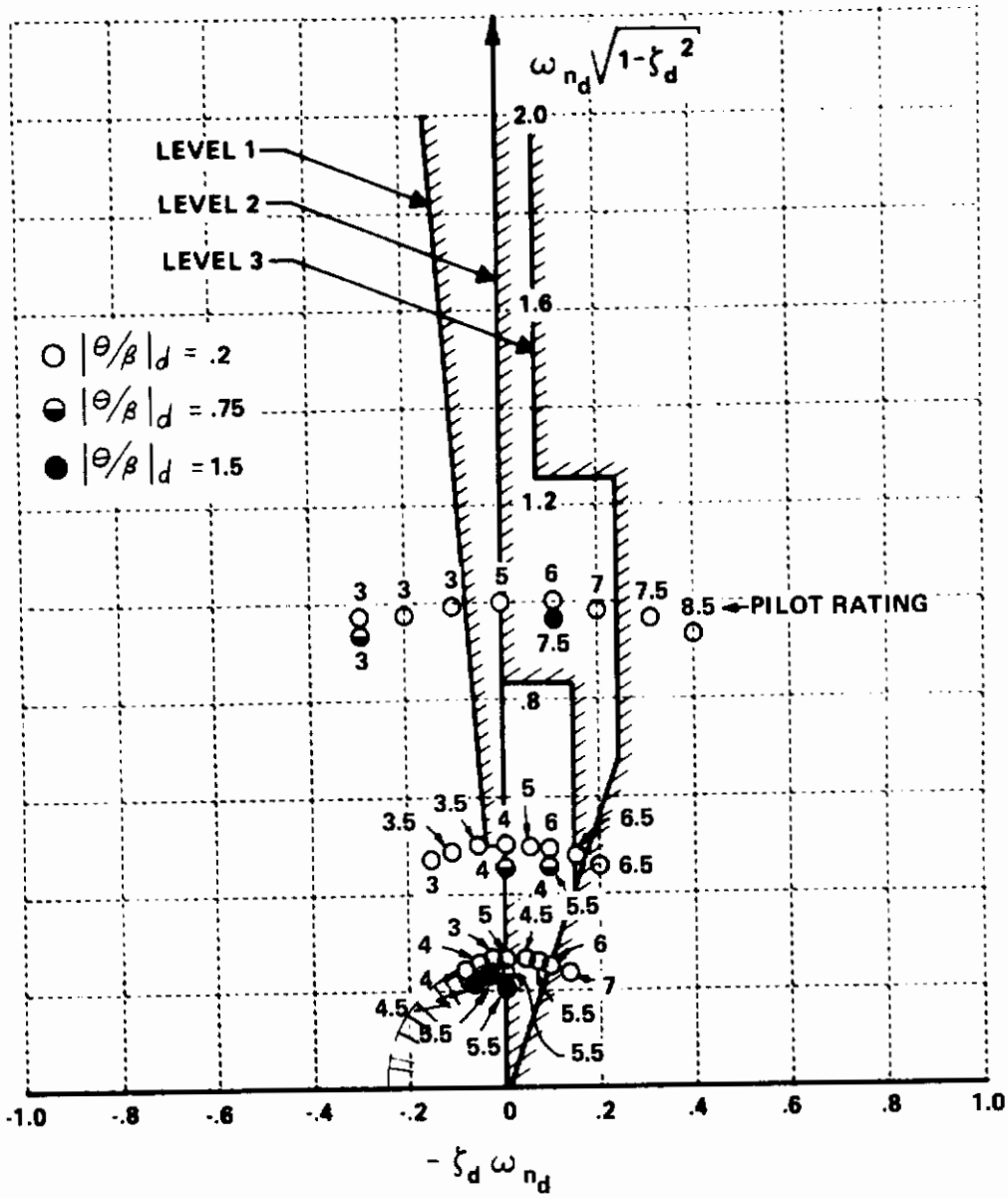

Figure 1 (3.3.7.1) AVERAGE PILOT RATINGS VS DUTCH ROLL REQUIREMENTS (BASED ON DATA FROM REFERENCE 86)

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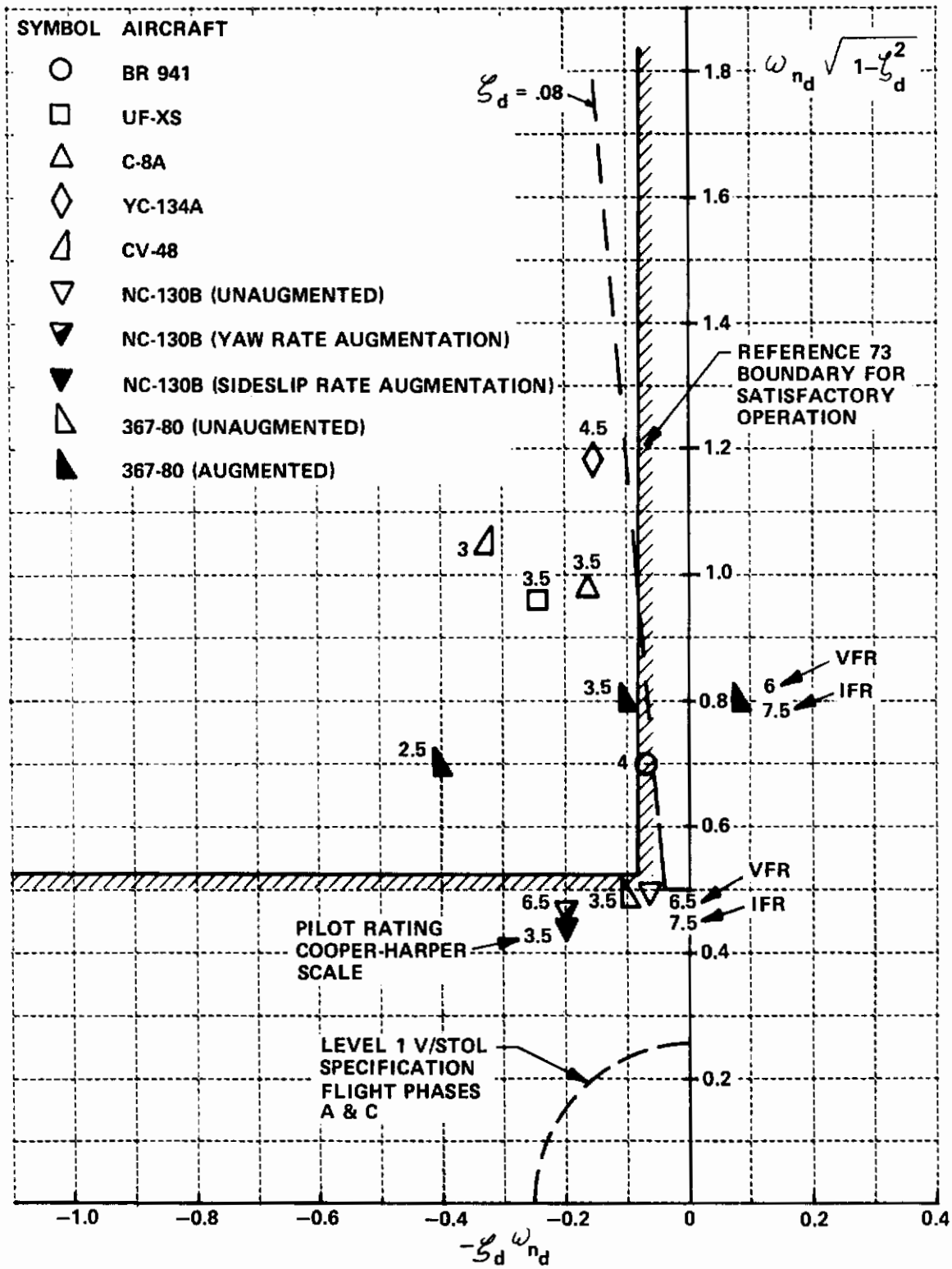

Figure 2 (3.3.7.1) DUTCH ROLL MODAL DATA FOR STOL AIRCRAFT (DATA FROM REFERENCE 73)

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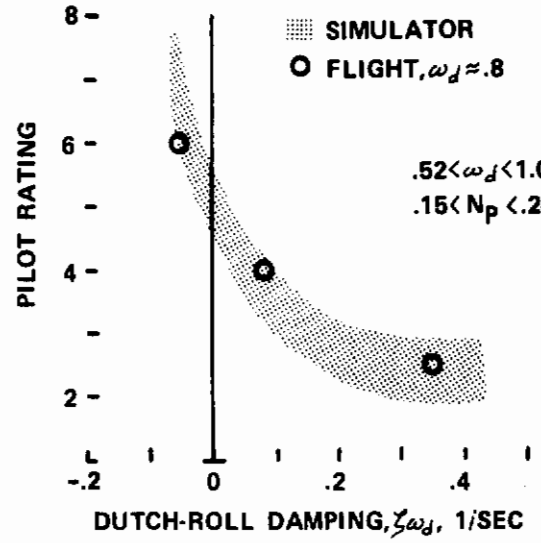

Figure 3 (3.3.7.1) PILOT OPINION OF DUTCH-ROLL DAMPING (FROM REFERENCE 89)

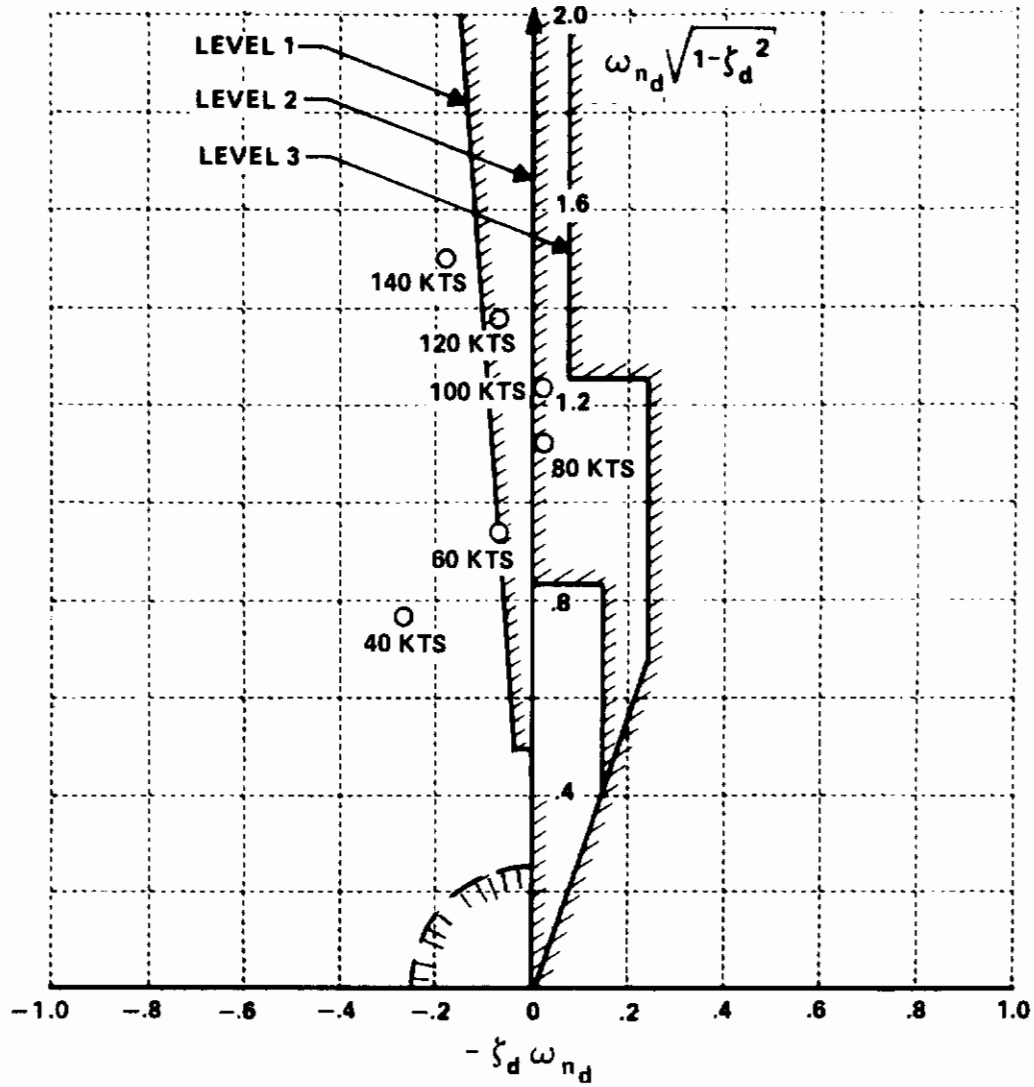

Figure 4 (3.3.7.1) DUTCH ROLL FREQUENCY-DAMPING CHARACTERISTICS OF X-22A (CONFIGURATION A, AUGMENTED) (DATA FROM REFERENCE 37)

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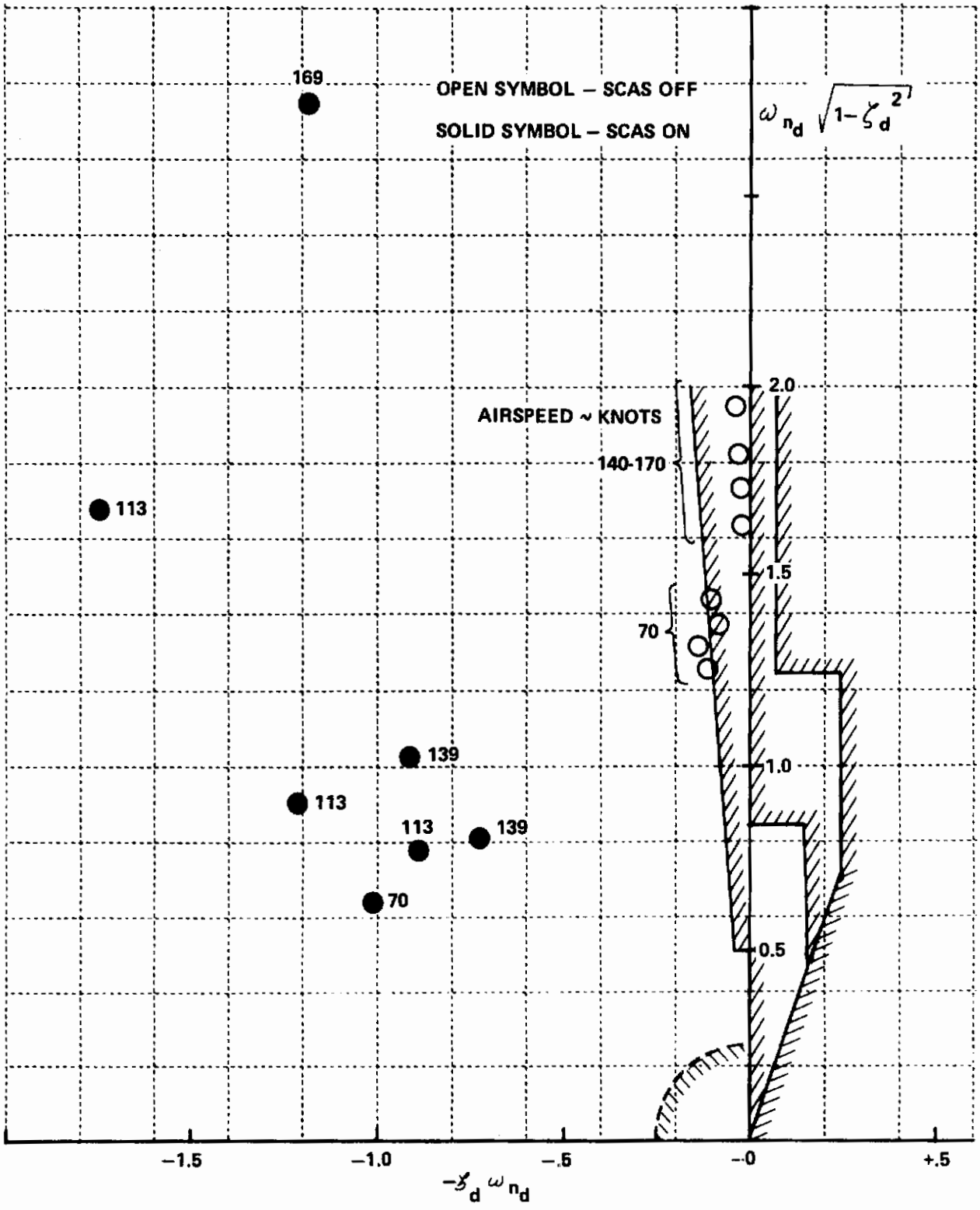

Figure 5 (3.3.7.1) DUTCH ROLL FREQUENCY–DAMPING CHARACTERISTICS OF AH–1G (DATA FROM REFERENCE 91)

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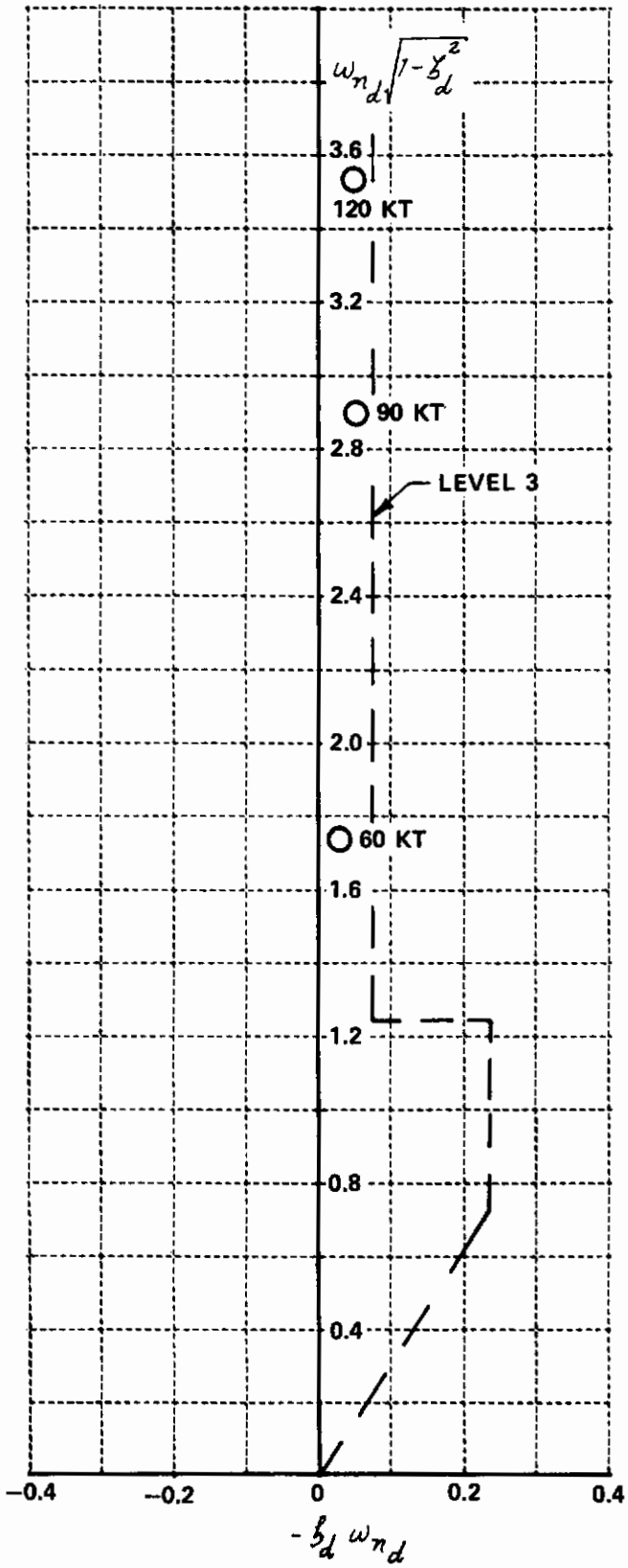

Figure 6 (3.3.7.1) DUTCH ROLL FREQUENCY–DAMPING CHARACTERISTICS OF UH–ID (DATA FROM REFERENCE 16)

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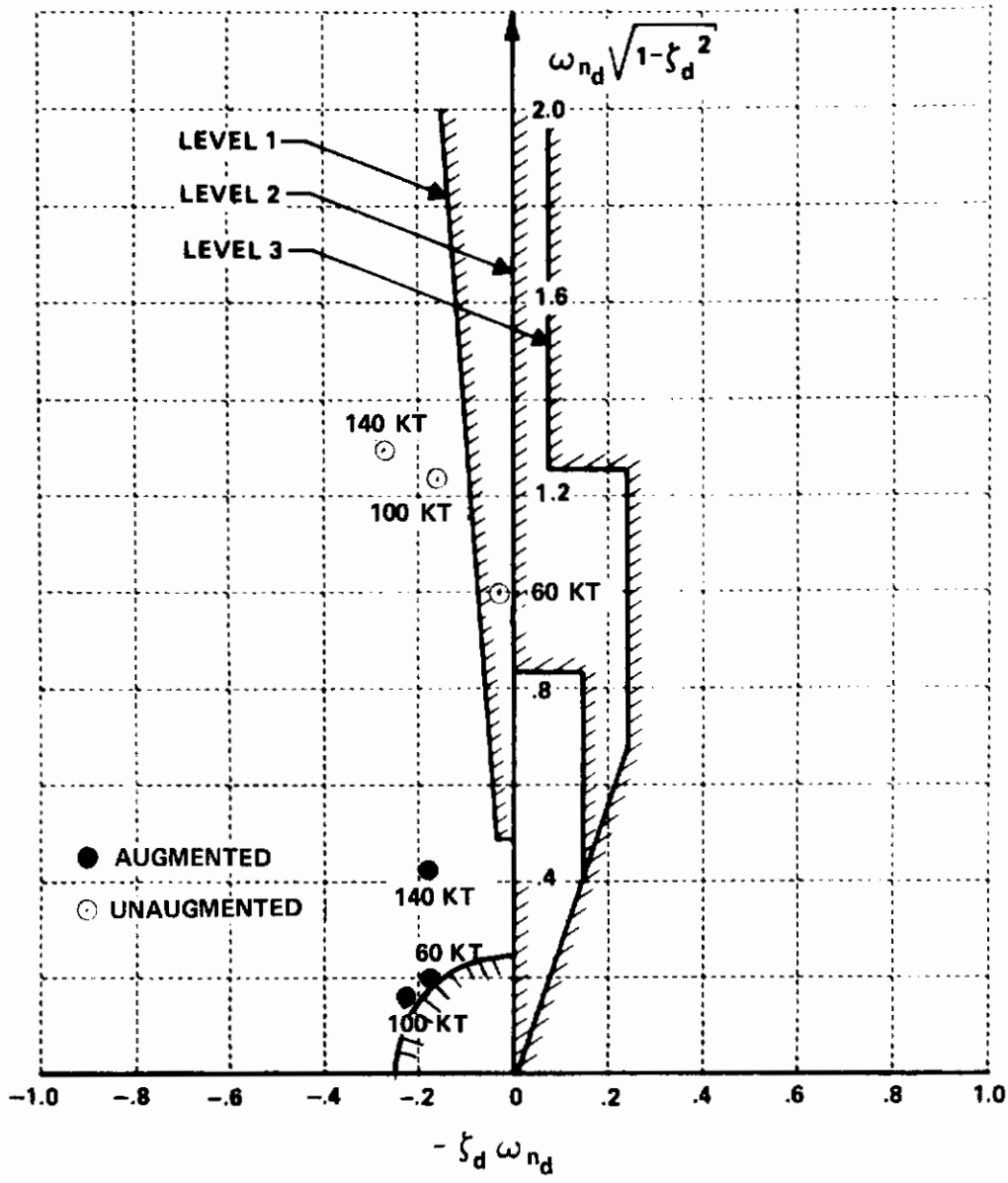

Figure 7 (3.3.7.1) DUTCH ROLL FREQUENCY-DAMPING CHARACTERISTICS OF SH-3A (DATA FROM REFERENCE 16)

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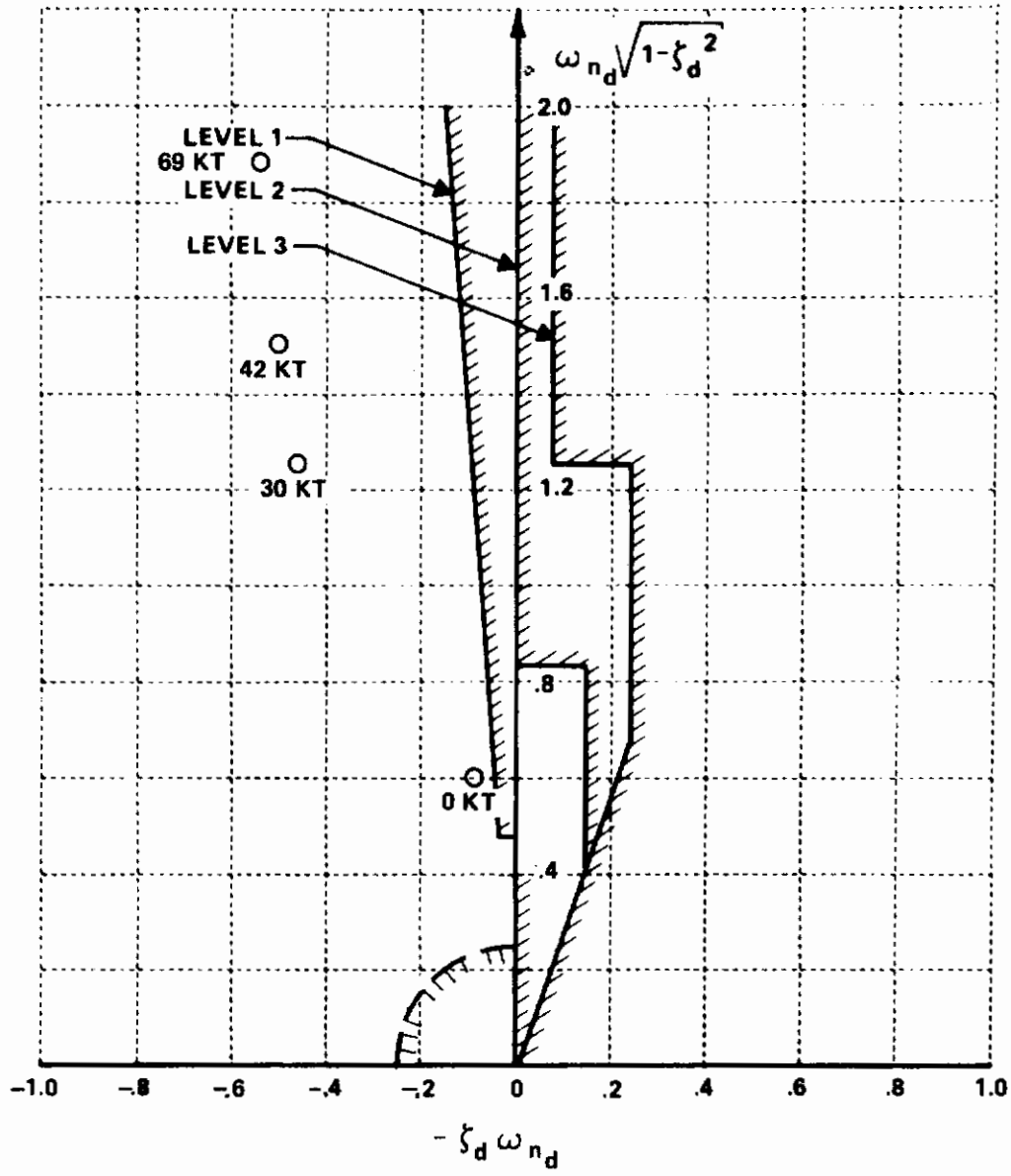

Figure 8 (3.3.7.1) DUTCH ROLL FREQUENCY-DAMPING CHARACTERISTICS OF H-19 (DATA FROM REFERENCE 16)

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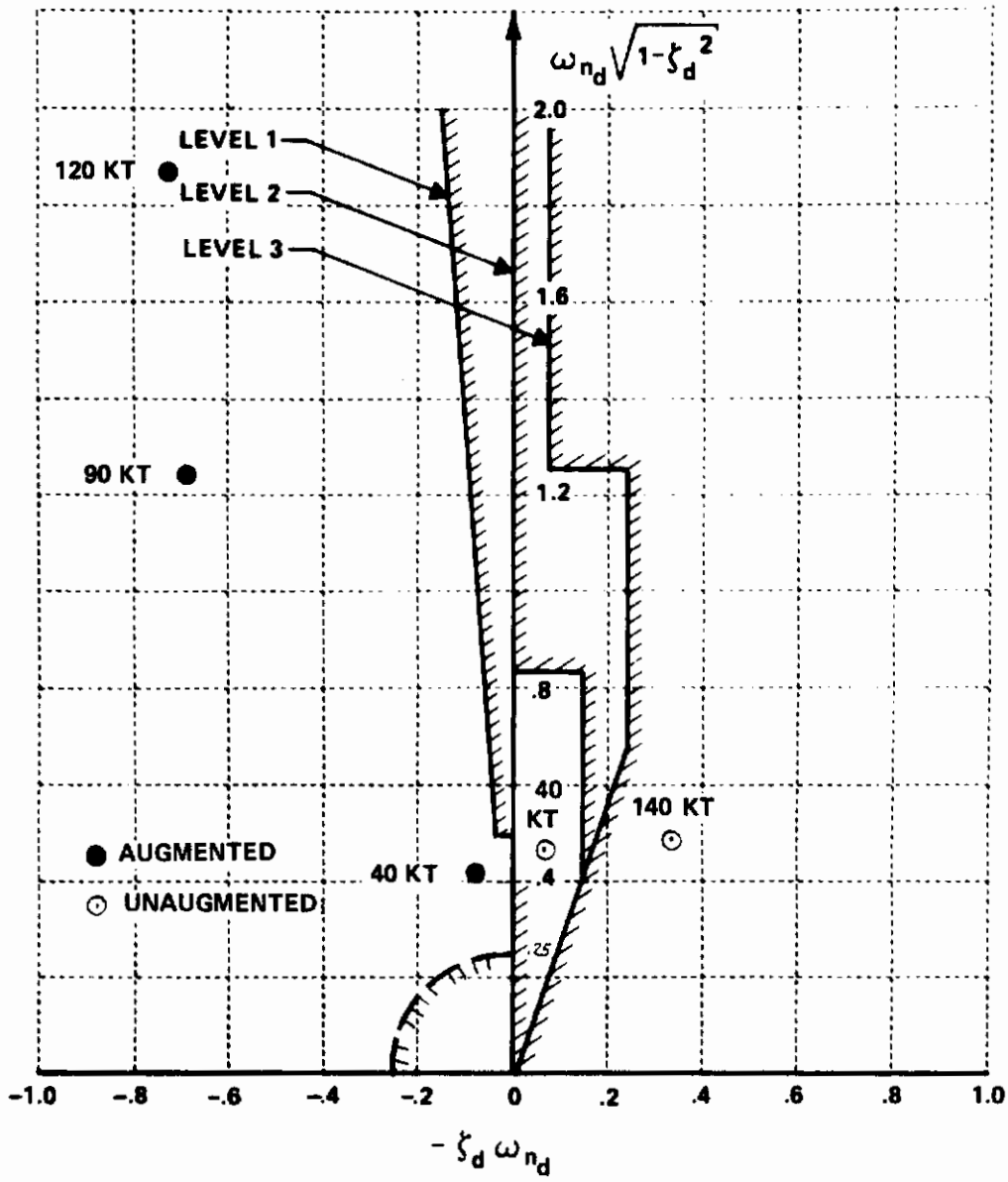

Figure 9 (3.3.7.1) DUTCH ROLL FREQUENCY-DAMPING CHARACTERISTICS OF CH-46 (DATA FROM REFERENCE 16)

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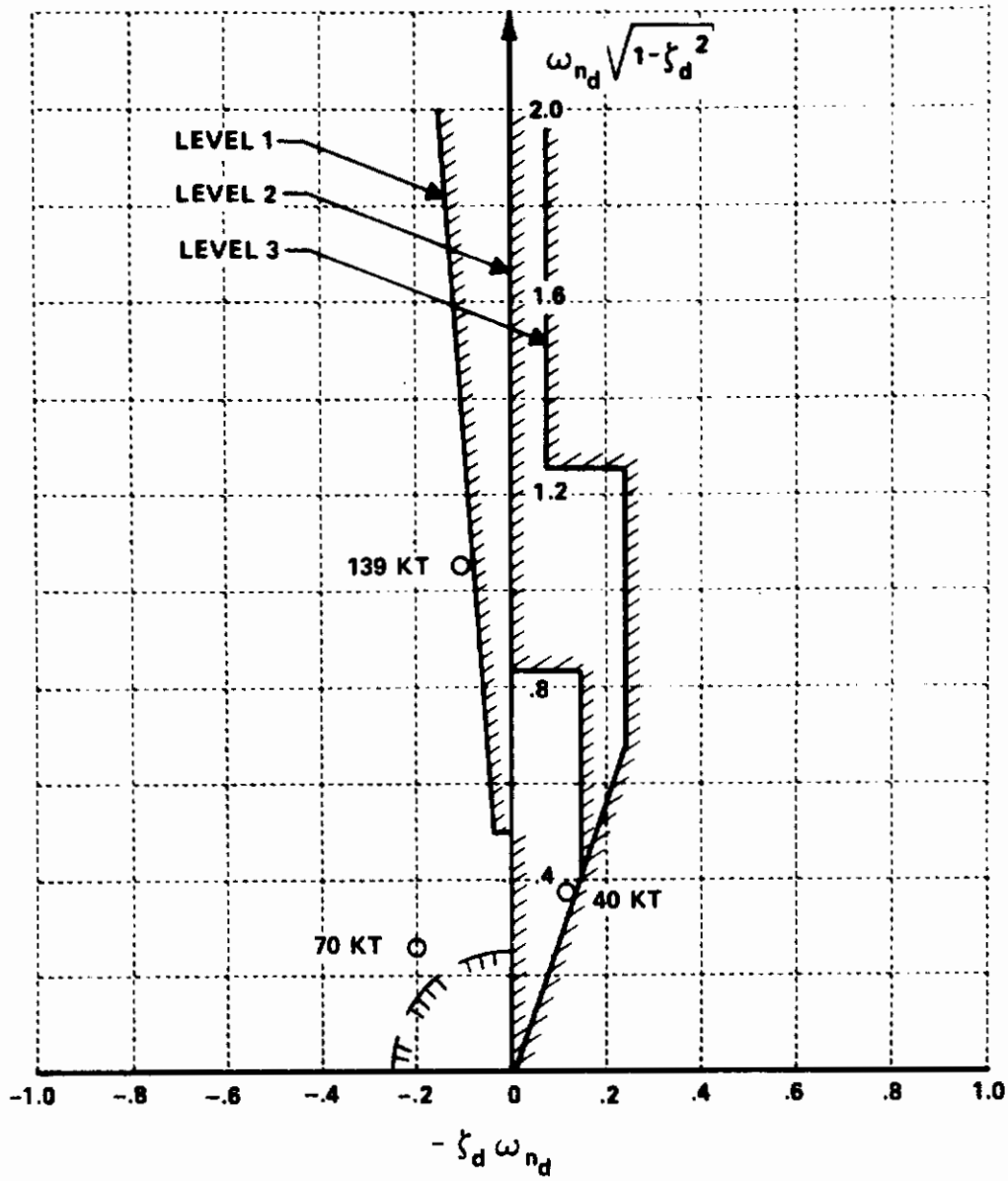

Figure 10 (3.3.7.1) DUTCH ROLL FREQUENCY-DAMPING CHARACTERISTICS OF XC-142A (DATA FROM REFERENCE 13)

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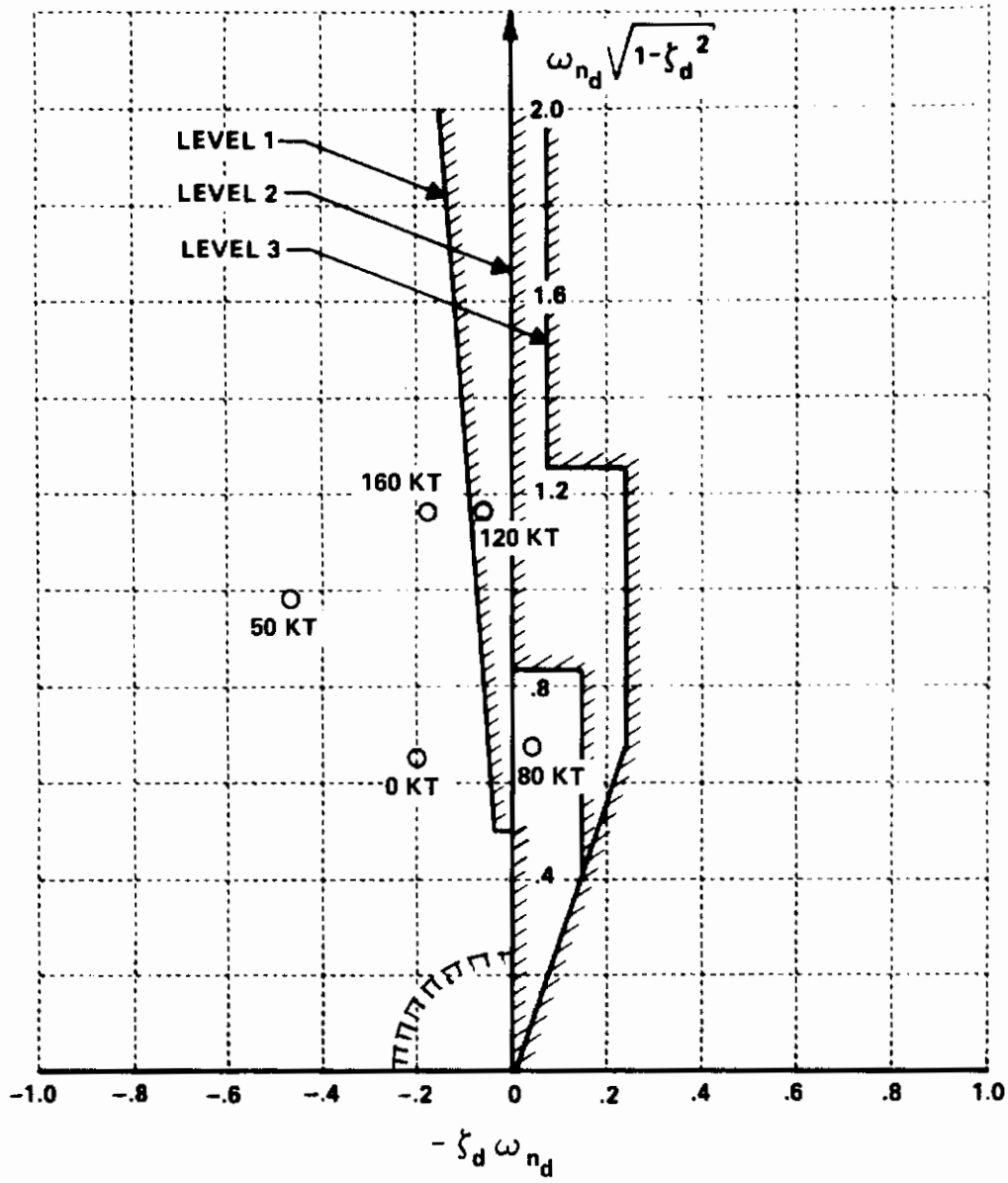

Figure 11 (3.3.7.1) DUTCH ROLL FREQUENCY-DAMPING CHARACTERISTICS OF X-19 (DATA FROM REFERENCE 13)

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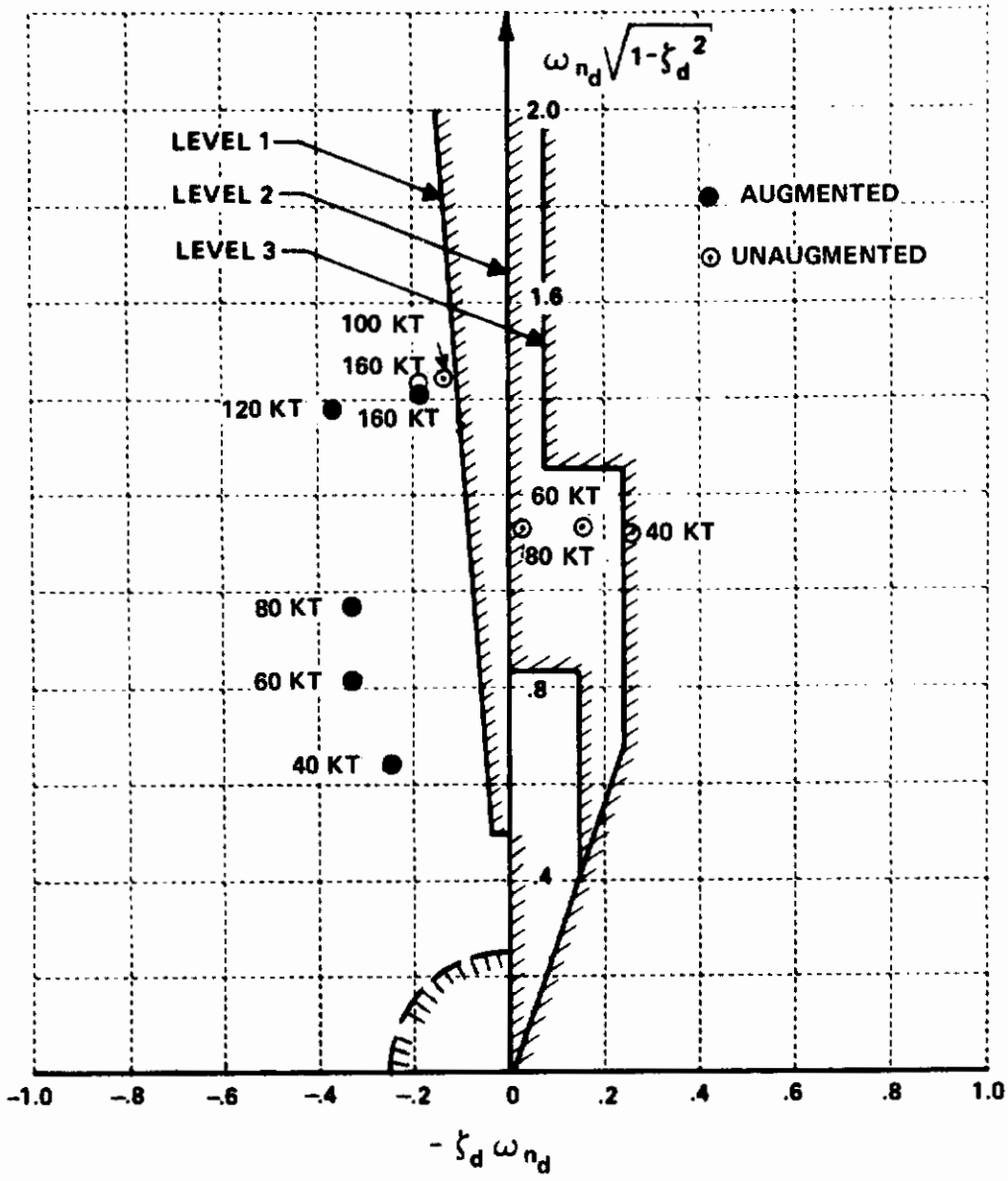

Figure 12 (3.3.7.1) DUTCH ROLL FREQUENCY-DAMPING CHARACTERISTICS OF X-22A (DATA FROM REFERENCE 13)

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3.3.7.2 ROLL MODE TIME CONSTANT

REQUIREMENT

3.3.7.2 Roll mode time constant. The roll mode shall be stable and the time constant, τ_R, shall be less than the following:

Level 1:	1.4 seconds
Level 2:	3.0 seconds
Level 3:	10.0 seconds

DISCUSSION

This requirement, which is directed at precision of control, is essentially similar to 3.3.1.2 of MIL-F-8785B(ASG) (Reference 10). The amount of roll damping has considerable effect on the roll response of an aircraft both for pilot inputs and in the presence of gusts. A considerable number of experiments have been conducted to illustrate that pilot rating is a function of roll damping. Roll damping is generally expressed in terms of the first-order roll mode time constant, τ_R, of the roll rate response following a lateral command step input. Thus, the requirement has been specified on roll mode time constant.

Reference 92 presents the results of several experiments to evaluate the effect of roll mode time constant on pilot rating. These data are presented on Figure 1(3.3.7.2). This reference concludes that no improvement in pilot rating results for roll mode time constant variations between 0.5 to 1.0 seconds, and that the maximum value of roll mode time constant considered necessary for a pilot rating of 3.5 in about 1.3 seconds. Reference 76 indicates that a "critical" value of roll mode time constant is approximately one second, and that as τ_R increases above this value, rapid roll response cannot be achieved by the pilot in a simple closed-loop task without overshooting, and undesirable oscillations.

Reference 73 describing airworthiness considerations for STOL aircraft presents roll mode data obtained by NASA on various experiments. This is illustrated on Figure 2(3.3.7.2). Based on this information, NASA suggests as a criteria in Reference 73 that for "satisfactory operation" the roll mode time constant should be less than 2 seconds, while for "safe operation" the roll mode time constant should be less than 4 seconds.

Examination of the data presented on Figure 1(3.3.7.2) and Figure 2(3.3.7.2) indicates that values of τ_R equal to approximately 1.4 seconds and 3 seconds are consistent with the relationship between pilot rating and boundaries for Level 1 and Level 2 flying qualities respectively.

The Level 3 value of τ_R = 10 seconds is based on the requirements of MIL-F-8785B(ASG) (Reference 10). While this value is arbitrary, it does legislate against unstable roll modes while not eliminating acceleration-like responses to control inputs.

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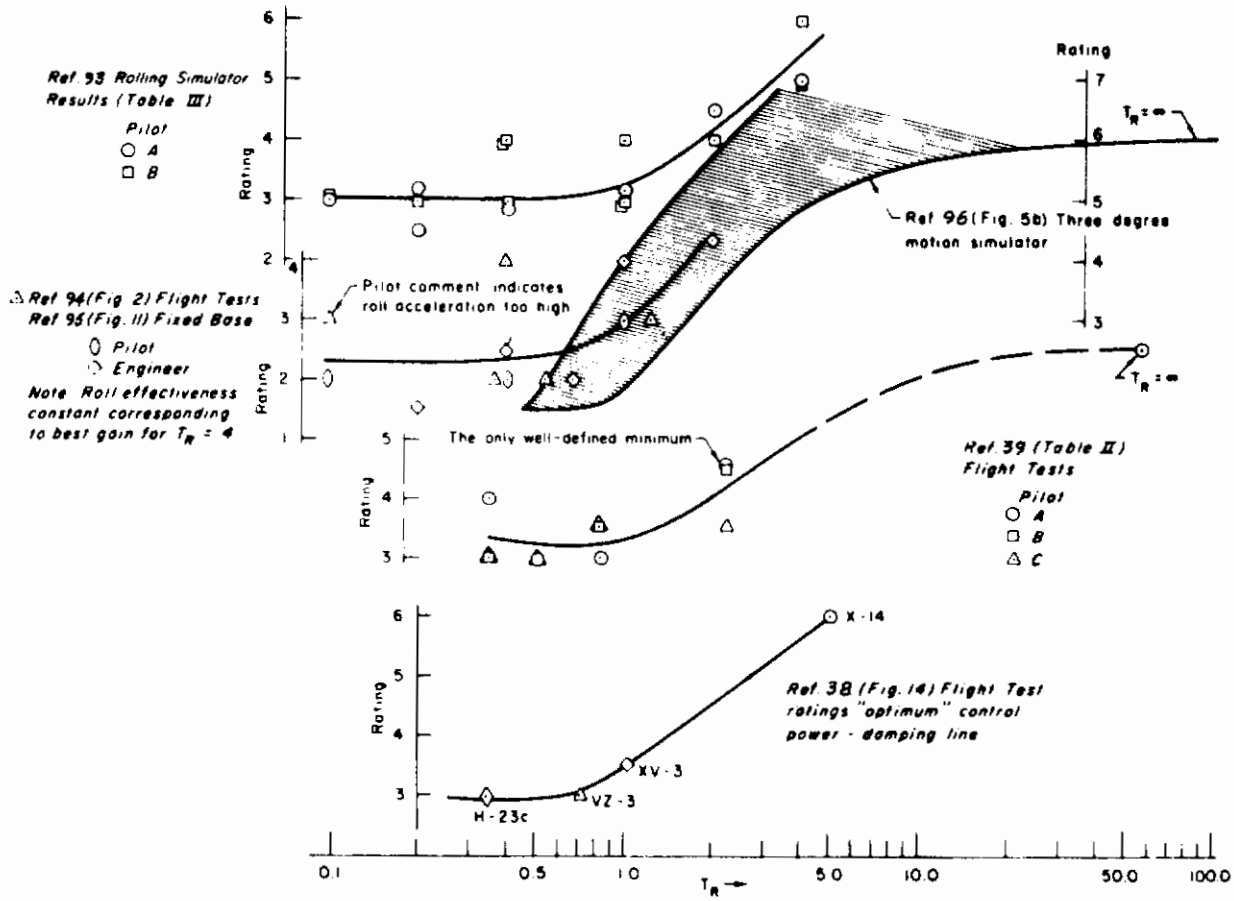

Figure 1 (3.3.7.2) PILOT RATING VERSUS ROLL DAMPING – FLIGHT TEST, MOVING-BASE, FIXED-BASE WITH RANDOM INPUT (FROM REFERENCE 92)

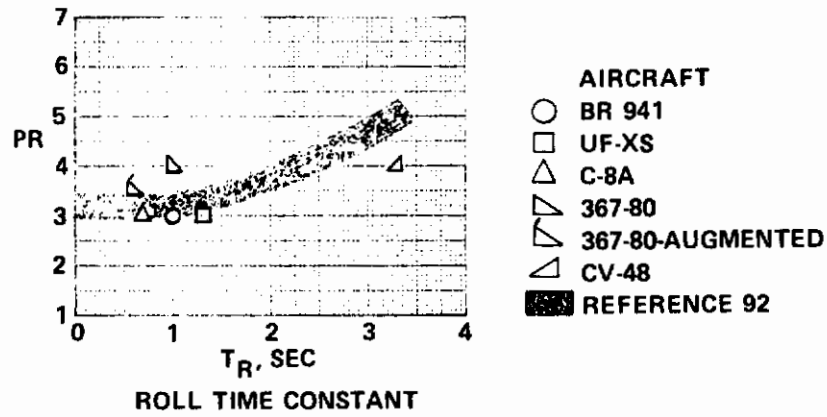

Figure 2 (3.3.7.2) ROLL MODE TIME CONSTANT OF STOL AIRCRAFT (DATA FROM REFERENCE 73)

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3.3.7.3 SPIRAL STABILITY

REQUIREMENT

3.3.7.3 Spiral stability. The combined effect of spiral stability, flight-control system characteristics, and trim change with speed shall be such that following a disturbance in bank of up to 10 degrees, the time for the bank angle to double shall be greater than the following:

Level 1:	20.0 seconds
Level 2:	12.0 seconds
Level 3:	4.0 seconds

These requirements shall be met with the cockpit controls free and the aircraft trimmed for zero-bank-angle, zero-yaw-rate flight.

DISCUSSION

The requirements on spiral stability are aimed primarily at ensuring low divergence rates from a zero-bank-angle flight path during periods of pilot inattention or when task requirements might momentarily divert pilot attention from flight path control. The requirement is essentially similar to 3.3.1.3 of MIL-F-8785B (Reference 10) in that the requirement includes the combined effects of the flight control system, lateral trim change with speed as well as the constant-speed spiral stability characteristics. This is more representative of the "apparent" spiral behavior noted by the pilot than the constant-speed spiral stability effects alone, and the requirement is written to include these effects rather than place limits on τ_5 , the first-order spiral mode time constant.

The present requirement is applied to bank angle disturbances "up to 10 degrees" rather than the 20 degrees of MIL-F-8785B(ASG) in order to be more compatible with the lower flight speeds under consideration in the V/STOL Specification and is based on comments in Reference 73 and requirement 3.7.4.5 of Reference 47.

Reference 73 recommends that the spiral stability, as measured by time to double amplitude of bank angle, should not be less than 20 seconds for satisfactory operation, nor less than 5 seconds for safe operation. The data used in Reference 73 are presented on Figure 1(3.3.7.3), and compared with the requirements of the V/STOL Specification.

Reference 97 presents the results of a recent in-flight investigation primarily directed at determining the effect of the spiral mode on the response of an aircraft in cruising flight. It also investigated the relationships between acceptable spiral stability limits and lateral-directional dynamics and control characteristics. Although the configurations evaluated do not meet the Level 1 Dutch roll frequency - damping requirements, (3.3.7.1) of the V/STOL Specification, the results do indicate that pilot rating can be acceptable for T_2 > 20 seconds and degrades significantly as T_2 decreases. The results of this experiment are in general agreement

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with the spiral stability requirements of the V/STOL Specification.

It should be noted that the V/STOL requirements on spiral stability are equal to the most lenient of the spiral stability requirements (3.3.1.3) of MIL-F-8785B (Reference 10). That is, the V/STOL Specification requirement on spiral stability is the same as the Class II and Class III requirement for all Flight Phases and Class I and Class IV requirement for Flight Phase Categories B and C.

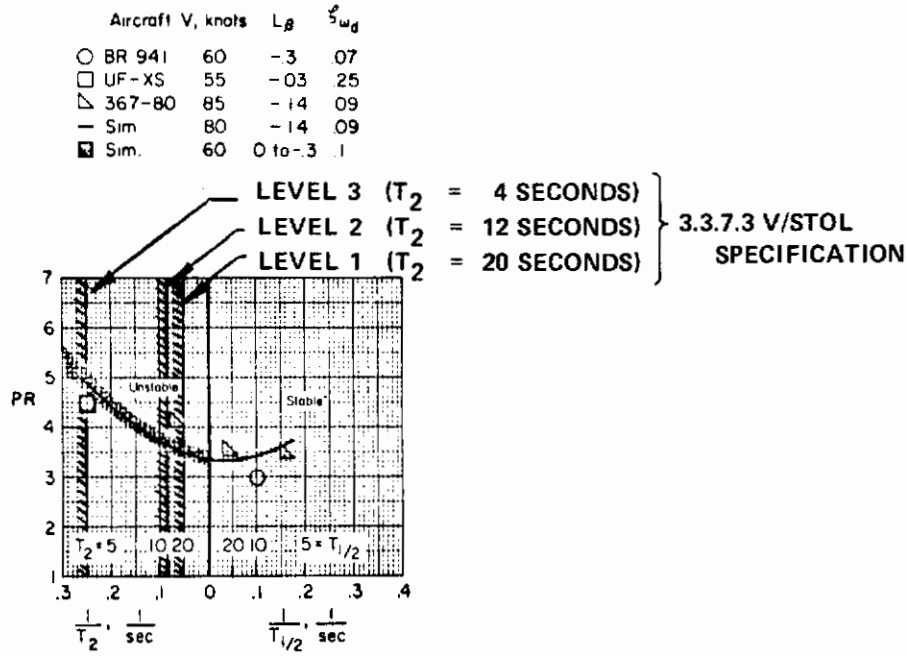

Figure 1 (3.3.7.3) VARIATION OF PILOT RATING WITH SPIRAL STABILITY (FROM REFERENCE 73)

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3.3.8 ROLL-SIDESLIP COUPLING

REQUIREMENT

3.3.8 Roll-sideslip coupling. The requirements on roll-sideslip coupling are stated in terms of allowable bank angle oscillations, sideslip excursions, roll control forces and yaw control forces that occur during specified rolling and turning maneuvers. The requirements of 3.3.8.1 and 3.3.8.2 apply for both right and left roll control commands for all magnitudes up to the magnitude required to meet the roll performance requirements of 3.3.9, unless otherwise stated.

DISCUSSION

This section of the V/STOL Specification places both quantitative and qualitative requirements on the coupling that can exist between roll and sideslip for turns and moderate bank angle change maneuvers such as turn entry. In contrast to most other requirements which specify desired response to control inputs, the requirements of this section are directed at restricting undesired responses resulting from initiation of the previously mentioned maneuvers. These unwanted couplings between the roll and sideslip responses can detract from precise flight path control, increase pilot workload by placing additional demands on control coordination, and can contribute to PIO tendencies. From a flying qualities viewpoint, roll-sideslip coupling manifests itself in various ways dependent upon the ratio of amplitudes of the bank-angle and sideslip-angle envelope in the Dutch roll mode ($|\phi/\beta|_d$).

For relatively low values of $|\phi/\beta|_d$, sideslip per se is perhaps the most important cue to the pilot. For these cases, if roll rate or roll control commands excite excessive sideslip, the flying qualities can be degraded by such motions as an oscillation of the nose on the horizon during a turn, a lag or initial reversal in yaw during a turn entry, or by pilot difficulty in quickly and precisely acquiring a desired heading. In addition, the pilot cannot successfully damp out Dutch roll oscillations excited by application of the roll control by use of the roll control alone.

As the value of $|\phi/\beta|_d$ increases, the pilot generally complains less about sideslip and essentially directs his comments at any difficulty in precisely controlling roll rate or in acquiring a desired bank angle. The coupling of β with φ and ϕ for these values of $|\phi/\beta|_d$ results primarily in large oscillations in the φ or ϕ response. For extremely large values of $|\phi/\beta|_d$ the sensitivity of the aircraft to turbulence may be so great that the aircraft may never be considered to be very good.

Another factor that has been considered in addressing the problems associated with roll-sideslip coupling is the ability of the pilot to control or prevent undesired motions by coordination of yaw control commands. If the aircraft is relatively easy to coordinate using normal piloting techniques, then the pilot may be more tolerant of large undesired responses since he can effectively control these responses. On the other hand, if coordination

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is difficult, the pilot may only tolerate small unwanted motions since he must accept these motions and attempts to coordinate might only aggravate the undesired response. The parameter ψ_β is introduced as a factor to indicate the difficulty of coordination, where ψ_β is a measure of the Dutch roll mode phasing in the response of sideslip to an aileron command.

The bank angle oscillation and sideslip requirements were derived empirically from experimental in-flight and flight test data generated from aircraft with conventional modal characteristics. The theoretical discussion contained in the following few sections of this report, which is based on linear conventional responses, is included to give some insight on why it is possible to correlate the empirical data with the selected parameters.

It is necessary to tie these requirements to a specific input and specific measures of the responses in order to specify them precisely and unambiguously. However, it is recognized that other techniques might be available whereby the required data can be obtained. If the phase angle, $\angle p/\beta$, cannot be measured, the requirements can still be applied as long as the sign of the dihedral effect is known. This is illustrated in Reference 84; when dihedral effect is positive, p will lead β by 45° to 225° in a free Dutch roll excitation, and p will lead β by 225° through 360° to 45° for negative effective dihedral.

During the development of the roll-sideslip coupling requirements (3.3.8, 3.3.8.1, 3.3.8.2) questions were raised as to whether or not placing requirements on ϕ_{osc}/ϕ_{AV}, $|\Delta\beta/\phi_1|$ and $|\Delta\beta/\phi_1| \times |\phi/\beta|_d$ would result in overspecification. This is questionable especially since the requirements are essentially directed at the same coupling derivatives, and the demarcation between sideslip problems and bank angle problems for V/STOL aircraft operating between 35 knots forward airspeed and V_{con} may not be as marked for aircraft operating above V_{con}. Figure 1(3.3.8) presents a plot of $|\Delta\beta/\phi_1| \times |\phi/\beta|_d$ versus ϕ_{osc}/ϕ_{AV} for typical configurations examined in Reference 86. As can be noted from the figure, the sideslip criteria appear to be a more sensitive measure as the response changes from slightly adverse to proverse, while the ϕ_{osc}/ϕ_{AV} metric appears to be most sensitive as the response becomes increasingly adverse, that is, $\frac{\Delta(|\Delta\beta/\phi_1| \times |\phi/\beta|_d)}{\Delta N'_{\delta_{AS}}} > \frac{\Delta(\phi_{osc}/\phi_{AV})}{\Delta N'_{\delta_{AS}}}$ when $N'_{\delta_{AS}}$ increases in a positive (proverse) sense and vice versa when $N'_{\delta_{AS}}$ increases in a negative (adverse) sense. It is then reasonable to assume that the measure which experiences the greatest change would tend to serve as a better parameter to correlate with pilot rating, in order to determine the dependency of the pilot rating with yaw due to aileron. Thus all measures are considered necessary to fully understand the responses of a particular aircraft.

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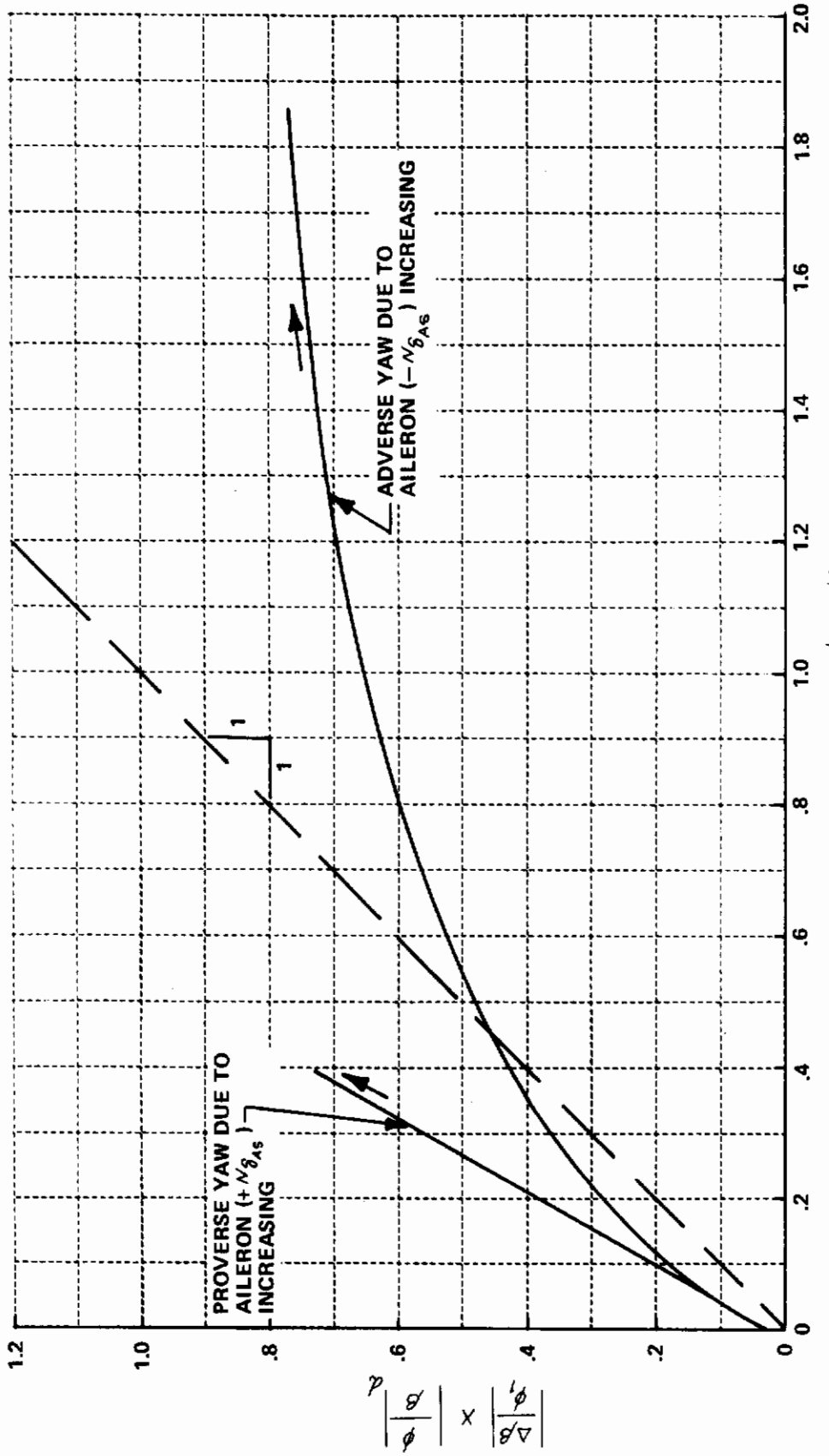

Figure 1 (3.3.8) SENSITIVITY OF ROLL-SIDESLIP COUPLING PARAMETERS (CONFIGURATION LH___+20+___, REFERENCE 86)

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3.3.8.1 BANK ANGLE OSCILLATIONS

REQUIREMENT

3.3.8.1 Bank angle oscillations. The value of the parameter ϕ_{osc}/ϕ_{AV} following a yaw-control-free impulse roll control command shall be within the limits specified in figure 3 for Levels 1 and 2. The impulse shall be as abrupt as practical within the strength limits of the pilot and the rate limits of the roll control system. For Levels 1 and 2, ϕ_{AV} shall always be in the direction of the roll control command.

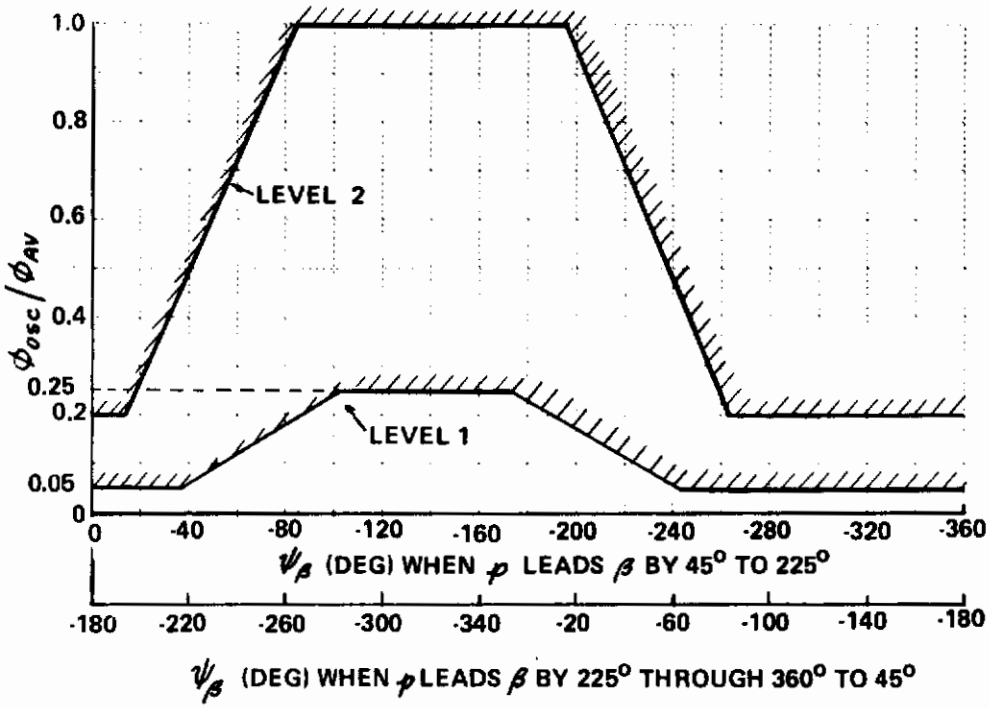

Figure 3 BANK ANGLE OSCILLATION LIMITATIONS

DISCUSSION

This requirement is directed at precision of control of bank angle, and is related to requirements 3.3.2.2.1 and 3.3.2.3 of MIL-F-8785B(ASG) (Reference 10).

In applying requirement 3.3.2.2.1 of MIL-F-8785B(ASG) to V/STOL configurations with long Dutch roll period (T_d > 6 sec) and light damping ratio, it was found that in order to determine p_{osc}/p_{AV} for these configurations, excessive bank angle changes could result during step responses. For a linear representation, an equivalent requirement that would not result in excessive bank angle changes is to specify requirements on ϕ_{osc}/ϕ_{AV} following a roll command pulse input rather than p_{osc}/p_{AV} for a roll command step input. This was also recognized in MIL-F-8785B(ASG) and

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gave rise to requirement 3.3.2.3 in that specification. The advantages of using a pulse, rather than a step, are:

- much larger inputs may be used since large bank angles do not result, and,
- ψ_β may be easier to measure since the β trace will not tend to ramp.

From an examination of linear transfer functions it can be determined that the magnitude of ϕ_{osc} / ϕ_{AV} for a pulse is identical to the magnitude of p_{osc} / p_{AV} for a step command. Also, the phase, ψ_β, for a pulse is $(90 + \sin^{-1} \zeta_d)$ degrees more positive than ψ_β for a step.

Thus, the bank angle oscillation requirements of the V/STOL Specification are based on 3.3.2.3 of the MIL-F-8785B. A detailed theoretical discussion of 3.3.2.2.1 and 3.3.2.3 (Reference 10) is presented in Reference 84, and will not be repeated in this document.

The parameters ϕ_{osc} / ϕ_{AV} and ψ_β will not give a measure of the relative location of the numerator of the ϕ / δ_{AS} transfer function with respect to the Dutch roll mode, but are relatively easy to obtain from flight test records. From a root locus analysis of a closed-loop compensatory tracking task it can be shown that the locations of the zeroes of the numerator of the ϕ/δ_{AS} transfer function with respect to the Dutch roll mode are measures of closed-loop stability and difficulties of rudder coordination in coordinated turn entries or exits (e.g., References 76 and 84).

The following discussion will yield some additional insight into the requirements and the effects of the stability derivatives on the requirement:

A commonly used form of the ϕ/δ_{AS} transfer function is as follows:

$$\frac{\phi}{\delta_{AS}}(s) = \frac{L'_{\delta_{AS}} \left(s^2 + 2\zeta_\phi \omega_\phi s + \omega_\phi^2\right)}{\left(s + \frac{1}{\tau_s}\right)\left(s + \frac{1}{\tau_R}\right)\left(s^2 + 2\zeta_d \omega_{n_d} s + \omega_d^2\right)}$$

Additional insight can be obtained if the following relationships are introduced:

$$2\zeta_\phi \omega_\phi = 2\zeta_d \omega_{n_d} + \Delta_1$$
$$\omega_\phi^2 = \omega_{n_d}^2 + \Delta_2$$

and for the sake of simplicity assume that the spiral mode is at the origin $(1/\tau_s = 0)$, thus ϕ/δ_{AS} becomes

$$\frac{\phi}{\delta_{AS}}(s) = \frac{L'_{\delta_{AS}} \left(s^2 + 2\zeta_d \omega_{n_d} s + \omega_{n_d}^2\right) + L'_{\delta_{AS}} \left(\Delta_1 s + \Delta_2\right)}{s\left(s + \frac{1}{\tau_R}\right)\left(s^2 + 2\zeta_d \omega_{n_d} s + \omega_{n_d}^2\right)}$$

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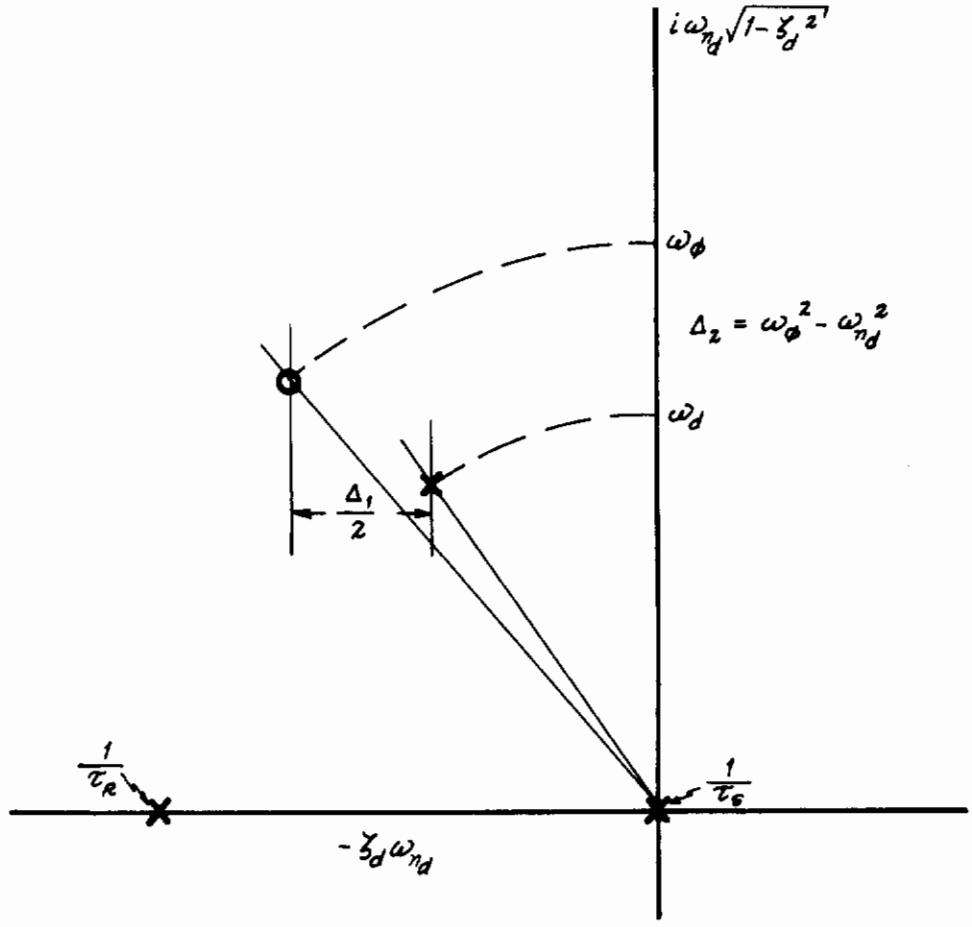
or

$$\frac{\phi}{\delta_{AS}}(s) = \frac{L'_{\delta_{AS}}}{s\left(s+\frac{1}{\tau_R}\right)} + \frac{L'_{\delta_{AS}}}{s\left(s+\frac{1}{\tau_R}\right)}\left(\frac{\Delta_1 s + \Delta_2}{s^2 + 2\zeta_d\omega_{n_d}s + \omega_{n_d}^2}\right)$$

Thus the response of bank angle can be thought of as the classical first-order approximation and an additional response. It is this additional response that introduces oscillatory behavior into the bank angle response, and this behavior is related to the offset terms(Δ_1 , Δ_2) previously defined, see sketch.



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The following approximations (Reference 98) may be used to examine Δ_1 and Δ_2.

$$\Delta_1 = 2(\zeta_\phi \omega_\phi - \zeta_d \omega_{n_d})$$

$$= \frac{N'_{\delta_{AS}}}{L'_{\delta_{AS}}} L'_r + \frac{1}{L'_p}\left[\frac{L'_\beta}{L'_p}\left(N'_p + \frac{g}{U_0}\right) - L'_r N'_p\right] - \lambda_s$$

$$\Delta_2 = \omega_\phi^2 - \omega_{n_d}^2 = -\frac{N'_{\delta_{AS}}}{L'_{\delta_{AS}}}(L'_\beta + Y_\beta L'_r) + \frac{L_\beta}{L_p}\left(N'_p - \frac{g}{U_0}\right) + \frac{Y_\beta L'_r N'_p}{L'_p}$$

With the assumption of the spiral mode at the origin, $\lambda_s = \frac{g(N'_r L'_\beta - N'_\beta L'_r)}{\lambda_R \omega_{n_d}^2} = 0$, thus $L'_r = \frac{L'_\beta}{N'_\beta} N'_r$ and the expressions for Δ_1 and Δ_2 become:

$$\Delta_1 = \frac{L'_\beta}{N'_\beta}\left[\left(\frac{N'_{\delta_{AS}}}{L'_{\delta_{AS}}} - \frac{N'_p}{L'_p}\right) N'_r + \frac{N'_\beta}{L'_p}\left(\frac{N'_p - \frac{g}{U_0}}{L'_p}\right)\right]$$

$$\Delta_2 = \frac{L'_\beta}{N'_\beta}\left[Y_\beta N'_r\left(\frac{N'_p}{L'_p} - \frac{N'_{\delta_{AS}}}{L'_{\delta_{AS}}}\right) + N'_\beta\left(\frac{N'_p - \frac{g}{U_0}}{L'_p} - \frac{N'_{\delta_{AS}}}{L'_{\delta_{AS}}}\right)\right]$$

Thus from the simplified expressions for Δ_1 and Δ_2 it becomes obvious that for a given set of modal parameters the position of the ϕ/δ_{AS} numerator zero with respect to the Dutch roll pole is strongly dependent upon a relationship between yaw due to aileron ($N'_{\delta_{AS}}$) and yaw due to roll rate (N'_p). This relationship is examined in the following discussion.

Figures 1(3.3.8.1) through 4(3.3.8.1) examine the effects of yaw due to aileron for several configurations evaluated in Reference 86. (Configurations LH+ 100 ____ + 20 ____ , with $N'_\beta > 0$).

Figure 1(3.3.8.1) illustrates the effect of yaw due to aileron ($N'_{\delta_{AS}}/L'_{\delta_{AS}}$) on the magnitude of the first peak of the bank angle response to a roll control pulse (ϕ_1). This figure illustrates that adverse yaw due to aileron ($N'_{\delta_{AS}} < 0$) tends to decrease ϕ_1 , however, the effect is more dependent on the sign of $\frac{N'_{\delta_{AS}}}{L'_{\delta_{AS}}} - \frac{N'_p - (g/U_0)}{L'_p}$ than on the sign of $N'_{\delta_{AS}}$. Also, when $\frac{N'_{\delta_{AS}}}{L'_{\delta_{AS}}} = \frac{N'_p - (g/U_0)}{L'_p}$, ϕ_1 is approximately the value that would be obtained from a first-order approximation of bank angle response.

Figure 2(3.3.8.1) illustrates the effect of yaw due to aileron on ϕ_{osc}/ϕ_{AV}. It should be noted that ϕ_{osc}/ϕ_{AV} tends to a minimum value when $\frac{N'_{\delta_{AS}}}{L'_{\delta_{AS}}} = \frac{N'_p - (g/U_0)}{L'_p}$.

From Figures 1 (3.3.8.1) and 2 (3.3.8.1), it can be concluded that increasing proverse yaw due to aileron, with respect to the value required

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to satisfy the relationship $\frac{N'_{\delta_{AS}}}{L'_{\delta_{AS}}} = \frac{N'_p - (g/U_o)}{L'_p}$ for a given configuration will increase ϕ_1 significantly and has relatively minor effects on ϕ_{osc}/ϕ_{AV}. On the other hand, making the value of yaw due to aileron adverse with respect to the value required to satisfy the relationship $\frac{N'_{\delta_{AS}}}{L'_{\delta_{AS}}} = \frac{N'_p - (g/U_o)}{L'_p}$ tends to minimize ϕ_1 and strongly increases ϕ_{osc}/ϕ_{AV}.

The preceding discussion has examined the effects of yaw due to aileron on ϕ_1 and ϕ_{osc}/ϕ_{AV}. This is only a part of the story since changing $N'_{\delta_{AS}}$ will also change the phasing of the Dutch roll excited in the response of sideslip to aileron (roll control) commands, and thereby change the complexity of pilot required rudder coordination in turn entries and exits. The question remaining then is: what is the effect of yaw due to aileron on (the measure of coordination difficulties)? Reference 99 illustrates that the rms of yaw control required in a turn entry maneuver is minimized when $N'_{\delta_{AS}}/L'_{\delta_{AS}} = (N'_p - g/U_o)/L'_p$. The effects of yaw due to aileron on ψ_β are illustrated on Figure 3(3.3.8.1), with respect to the ϕ_{osc}/ϕ_{AV} boundaries of the V/STOL Specification. As expected, minimization of ϕ_{osc}/ϕ_{AV} for a given configuration results in ψ_β of approximately -180° or -360° depending on the sign of the yaw due to aileron needed to set $N'_{\delta_{AS}}/L'_{\delta_{AS}} = (N'_p - \frac{g}{U_o})/L'_p$. Figure 4(3.3.8.1) transforms the previously presented data into the s-plane, and illustrates the mapping of the Level 1 ϕ_{osc}/ϕ_{AV} criteria into the s-plane for the configurations examined.

Figure 5(3.3.8.1) from Reference 86 is presented for comparison with Figure 4(3.3.8.1). It should be noted from Figure 5(3.3.8.1) that pilot comments regarding adverse or proverse yaw due to aileron application do not adhere to either a line of $\omega_\phi/\omega_d = 1$ or $N'_{\delta_{AS}} = 0$ as would be expected from the simple approximation to ω_ϕ/ω_d, but rather conform to the relationship $N'_{\delta_{AS}}/L'_{\delta_{AS}} = (N'_p - \frac{g}{U_o})/L'_p$. It should also be noted that the 3.5 pilot rating iso-opinion line of Figure 5(3.3.8.1) is more restrictive than the boundary shown on Figure 4(3.3.8.1) for Level 1 ϕ_{osc}/ϕ_{AV}. This indicates that the primary factor in pilot rating for the configurations evaluated was not necessarily precise control of bank angle. This point will be picked up during the discussion of the sideslip excursion requirement of the V/STOL Specification (Paragraph 3.3.8.2).

The previous discussion is supplemental to that appearing in Reference 84 and it is suggested that a review of that discussion will be most useful to the complete understanding of the requirement.

The most significant difference between the V/STOL Specification requirement on ϕ_{osc}/ϕ_{AV} and that of MIL-F-8785B(ASG) (Reference 10) is that the regions of acceptable flying qualities are not separated by Flight Phase Category. Thus the Level 1 ϕ_{osc}/ϕ_{AV} requirement of the V/STOL Specification is equivalent to the Level 1 requirement for Flight Phase

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Categories A and C of Reference 10, while the Level 2 V/STOL requirement is equivalent to the Flight Phase B Level 2 requirement of MIL-F-8785B(ASG) [see Figure 6(3.3.8.1)]. Pilot rating data obtained from References 86, 88, 100, and 101 are presented in Figures 6(3.3.8.1) through 8(3.3.8.1) plotted on a grid of ϕ_{osc}/ϕ_{AV} and ψ_β computed for a pulse roll control command. These figures provide a comparison between the V/STOL requirement and available experimental data.

Recently, attention has been directed to the changes in ψ_β when nonideal command inputs are used (Reference 102). The perfect impulse command is replaced by a sequence of ramp inputs, such that the result is a triangular command input. The results of the analysis are illustrated on Figure 9(3.3.8.1). The time, T, is one-half of the duration of the triangular command input. The figure illustrates that differences in ψ_β are a function of the pulse duration and the Dutch roll period and damping ratio. Examination of Dutch roll mode characteristics for typical V/STOL aircraft (References 13 and 16) indicate that the minimum Dutch roll period in the speed region of interest is of the order of 3.5 seconds. Examination of various flight test programs conducted by CAL and discussions with pilots indicate that a pilot is capable of applying a manual pulse in less than 0.6 seconds, thus for V/STOL aircraft flying between 35 knots forward and 120 knots a critical value of T/T_d is approximately 0.086. Examination of Figure 9(3.3.8.1) indicates that $\Delta\psi_\beta$ would be less than 2 degrees. In addition, Reference 102 examined the effects of a ramp and hold input in comparison to a perfect step for p_{osc}/p_{AV}. These results are shown on Figure 10(3.3.8.1), and for the critical value of T/T_d previously determined, the effect of an imperfect step input on p_{osc}/p_{AV} is negligible.

For the imperfect inputs selected, the triangular pulse is not the derivative of the ramp and hold input, however, the integral of the triangular input is approximately a ramp and hold with an additional time delay. Thus the magnitude of $\Delta(\phi_{osc}/\phi_{AV})$ for the triangular input is of the same order of magnitude as $\Delta(p_{osc}/p_{AV})$ for the ramp and hold. Since $\Delta(p_{osc}/p_{AV})$ is negligible for the values of T/T_d normally associated with V/STOL aircraft below V_{con}, then $\Delta(\phi_{osc}/\phi_{AV})$ should also be negligible.

Figure 11(3.3.8.1) is presented to illustrate the pilot rating for configurations evaluated in Reference 86 when Level 2 Dutch roll mode characteristics (3.3.7.1 of V/STOL Specification, Reference 1) are combined with various Levels of ϕ_{osc}/ϕ_{AV}. In general the pilot rating associated with these configurations is still at least Level 3, but no simple rules of thumb can be developed to predict the Level of flying qualities associated with such configurations.

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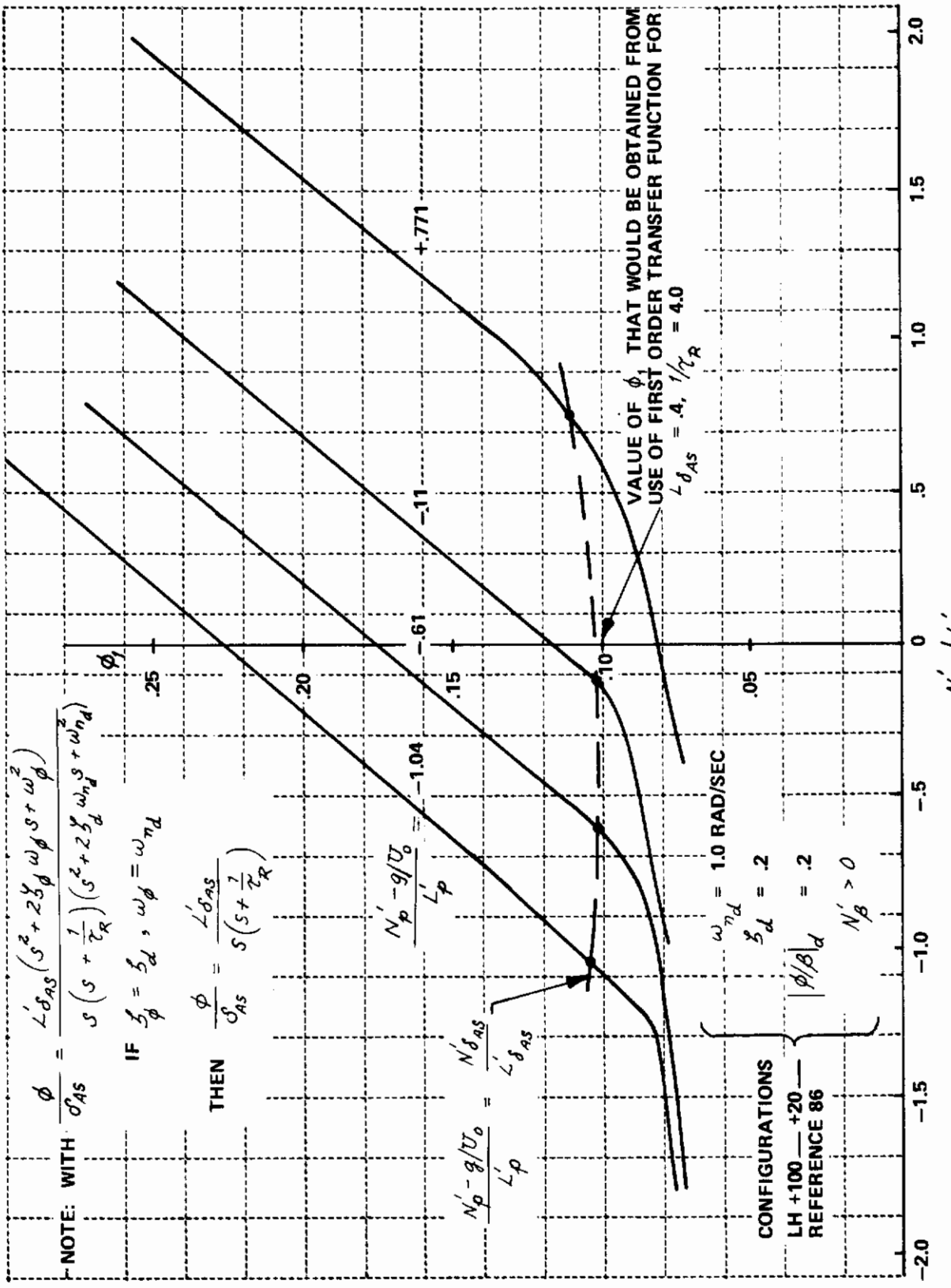

Figure 1 (3.3.8.1) EFFECT OF YAW DUE TO AILERON ON ϕ_1

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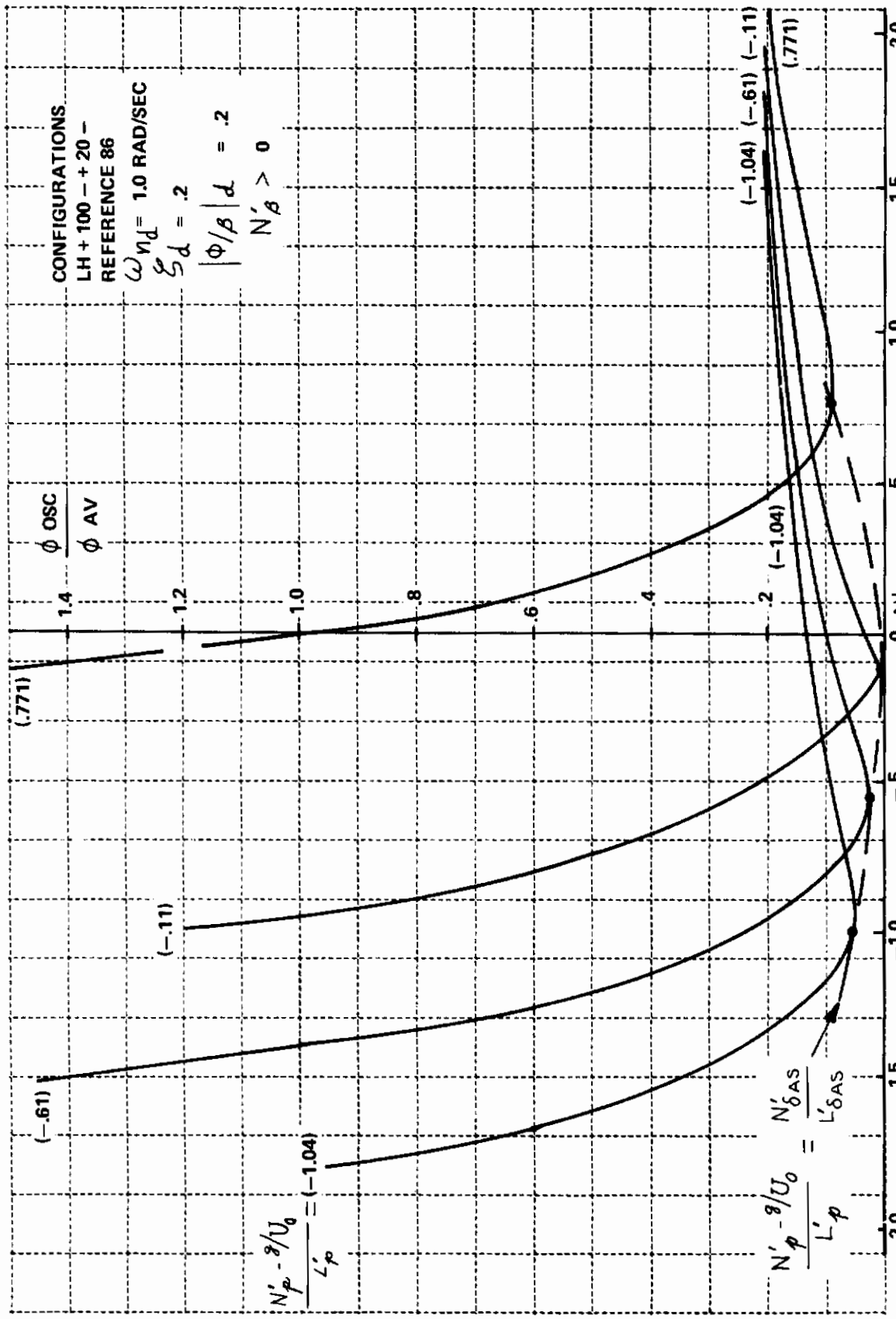

Figure 2 (3.3.8.1) EFFECT OF YAW DUE TO AILERON ON ϕ OSC/ϕ AV

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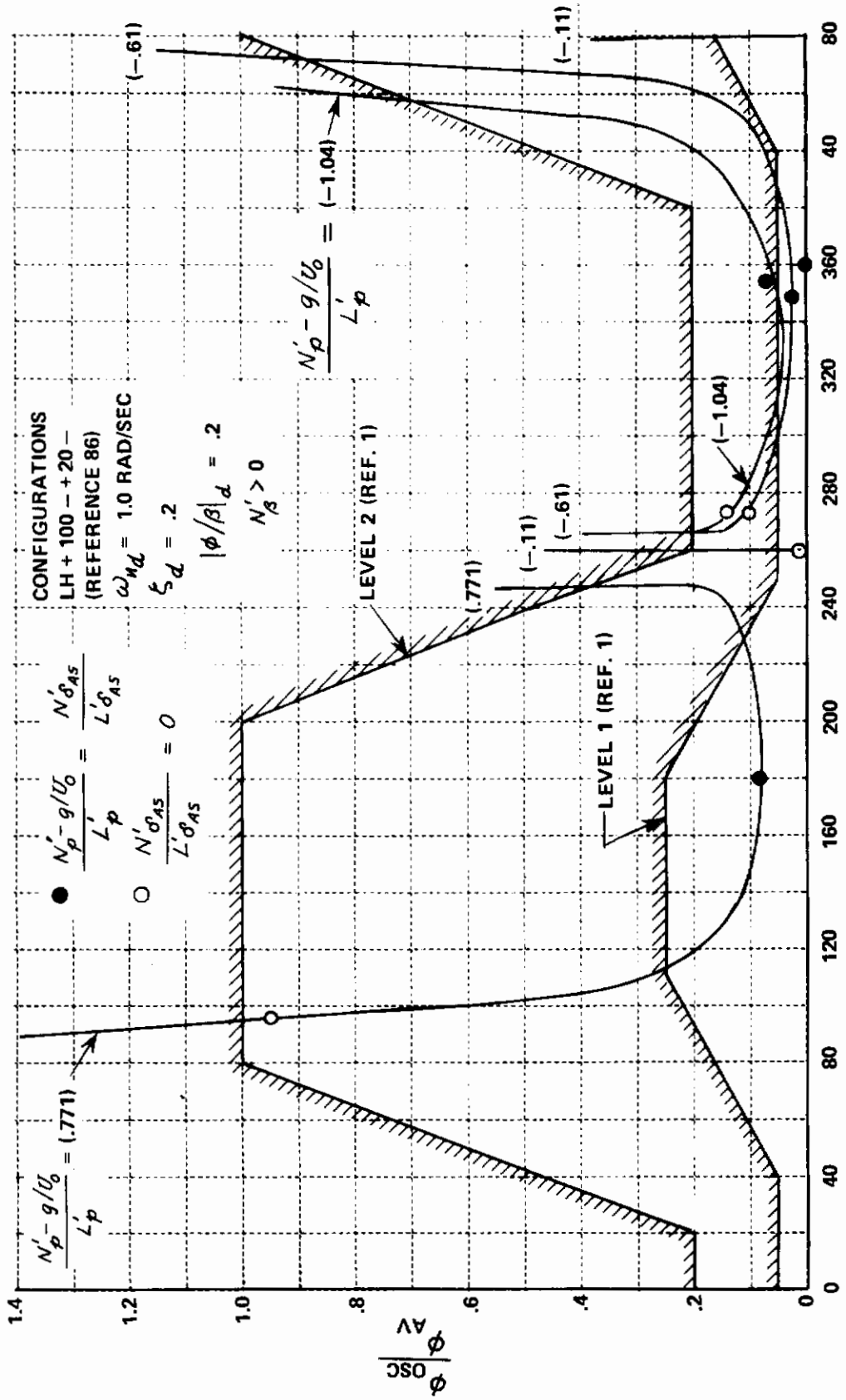

Figure 3 (3.3.8.1) EFFECT OF YAW DUE TO AILERON ON ψ_β AND ϕ_{OSC}/ϕ_{AV}

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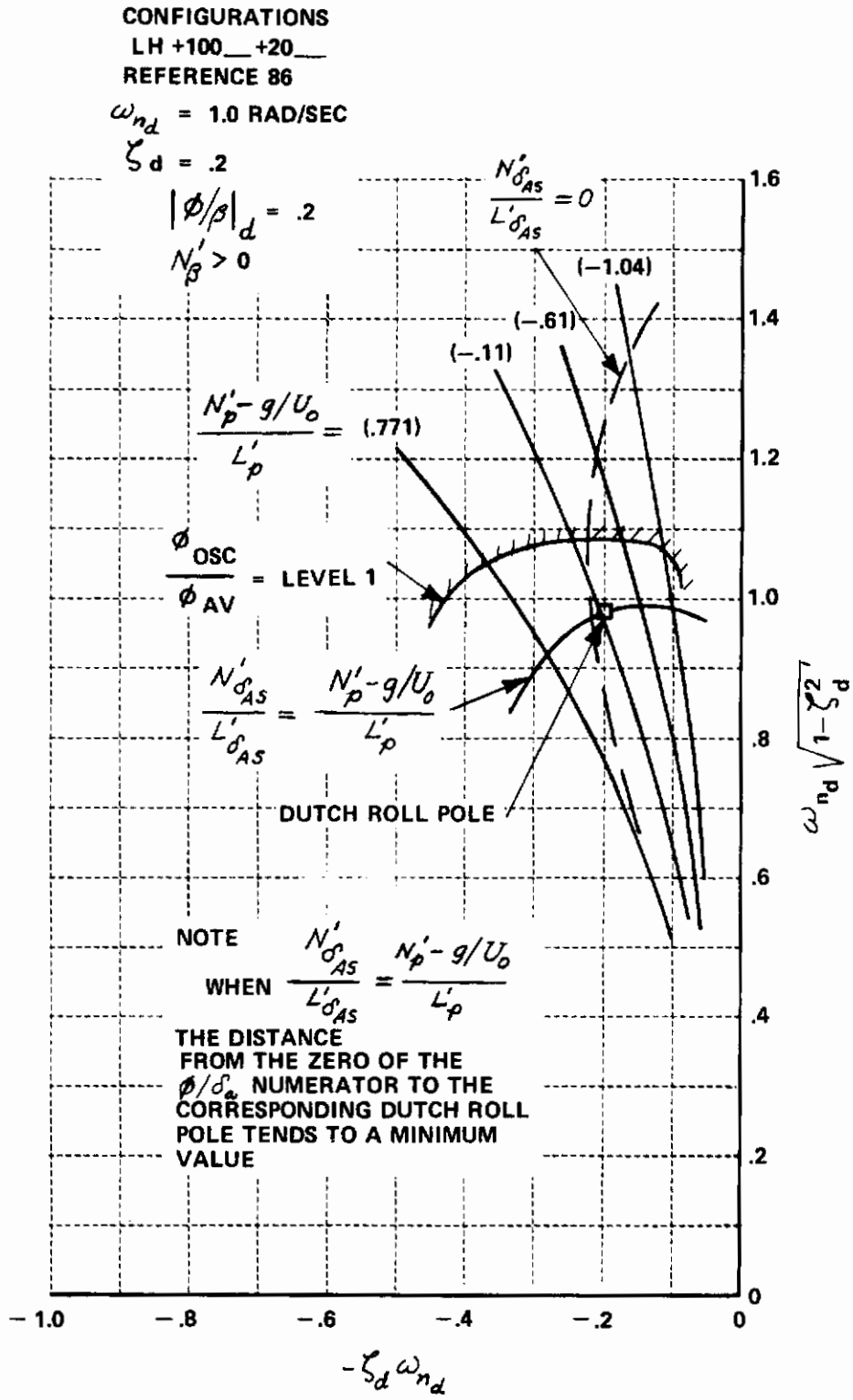

Figure 4 (3.3.8.1) EFFECT OF YAW DUE TO AILERON ON LOCATION OF NUMERATOR ZEROS OF ϕ/δ_{AS} TRANSFER FUNCTION

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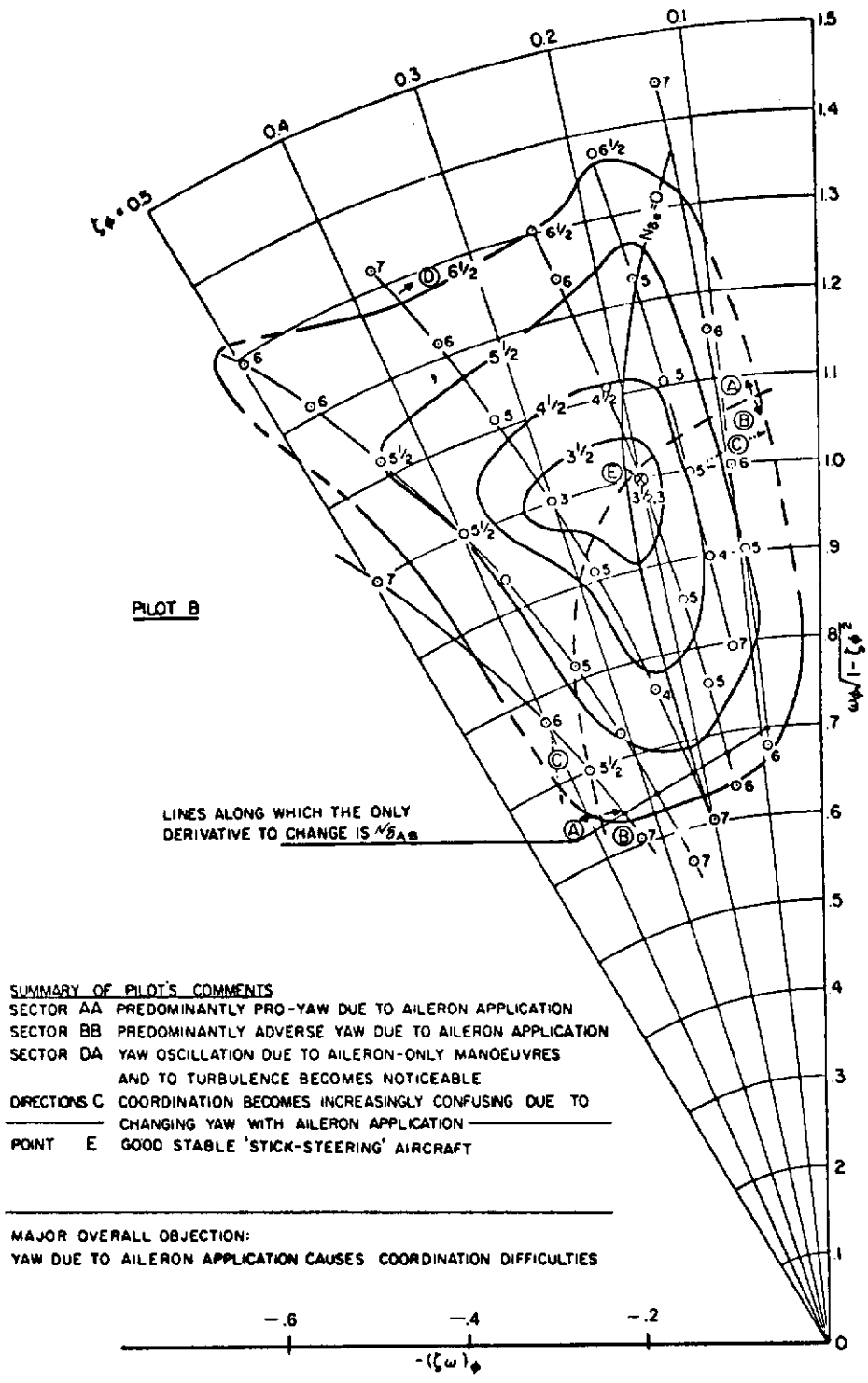

Figure 5 (3.3.8.1) EFFECT OF CHANGING ZEROS OF ϕ/δ_{AS} TRANSFER FUNCTION ABOUT $\omega_d = 1.0$, $\zeta_d = 0.2$ ON PILOT B'S RATINGS; $\left|\frac{\phi}{\beta}\right|_d = 0.2$ (FROM REFERENCE 86)

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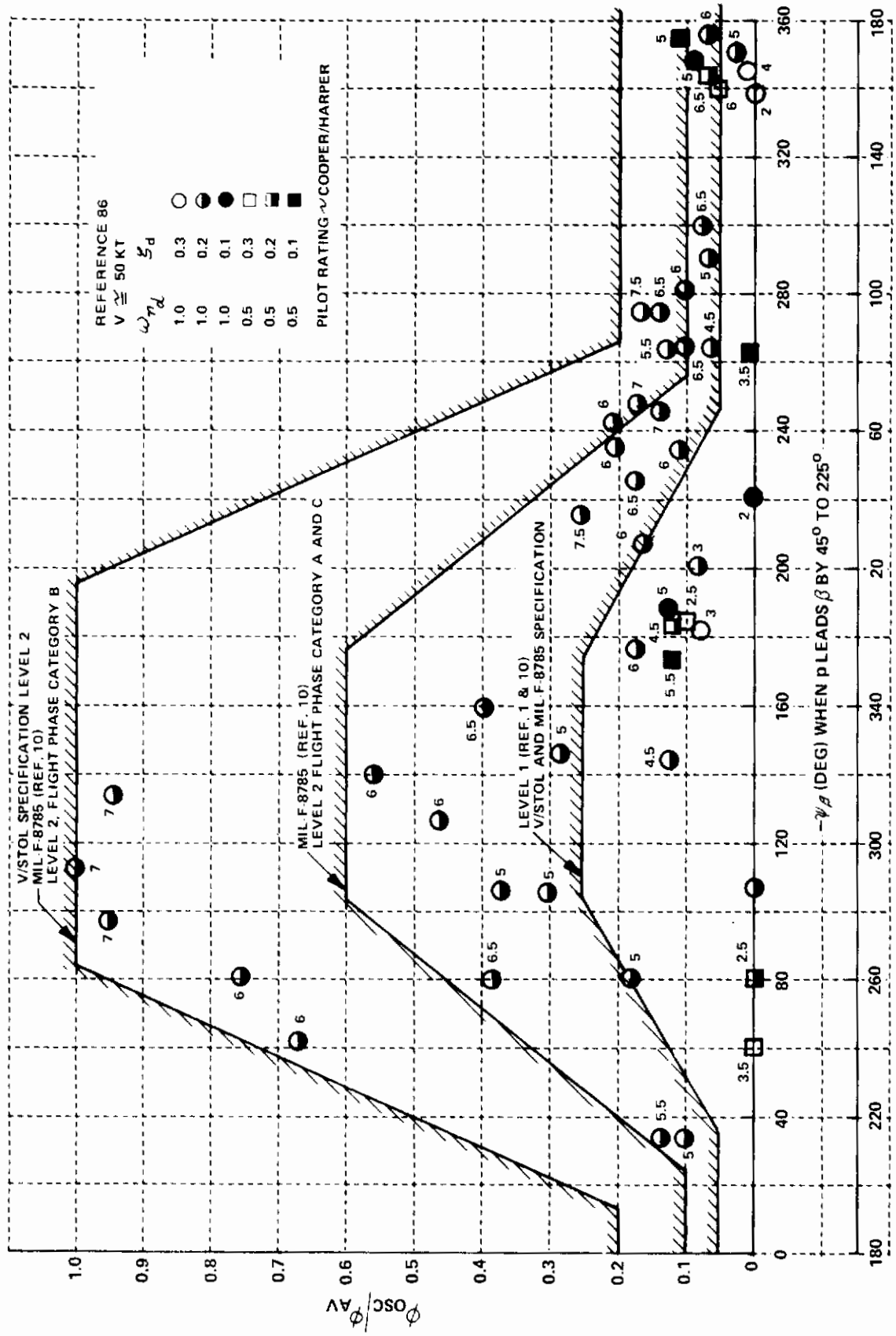

Figure 6 (3.3.8.1) COMPARISON OF ϕ_{OSC}/ϕ_{AV} FOR CONFIGURATIONS TESTED IN REFERENCE 86 WITH REQUIREMENTS OF REFERENCE 1 & 10

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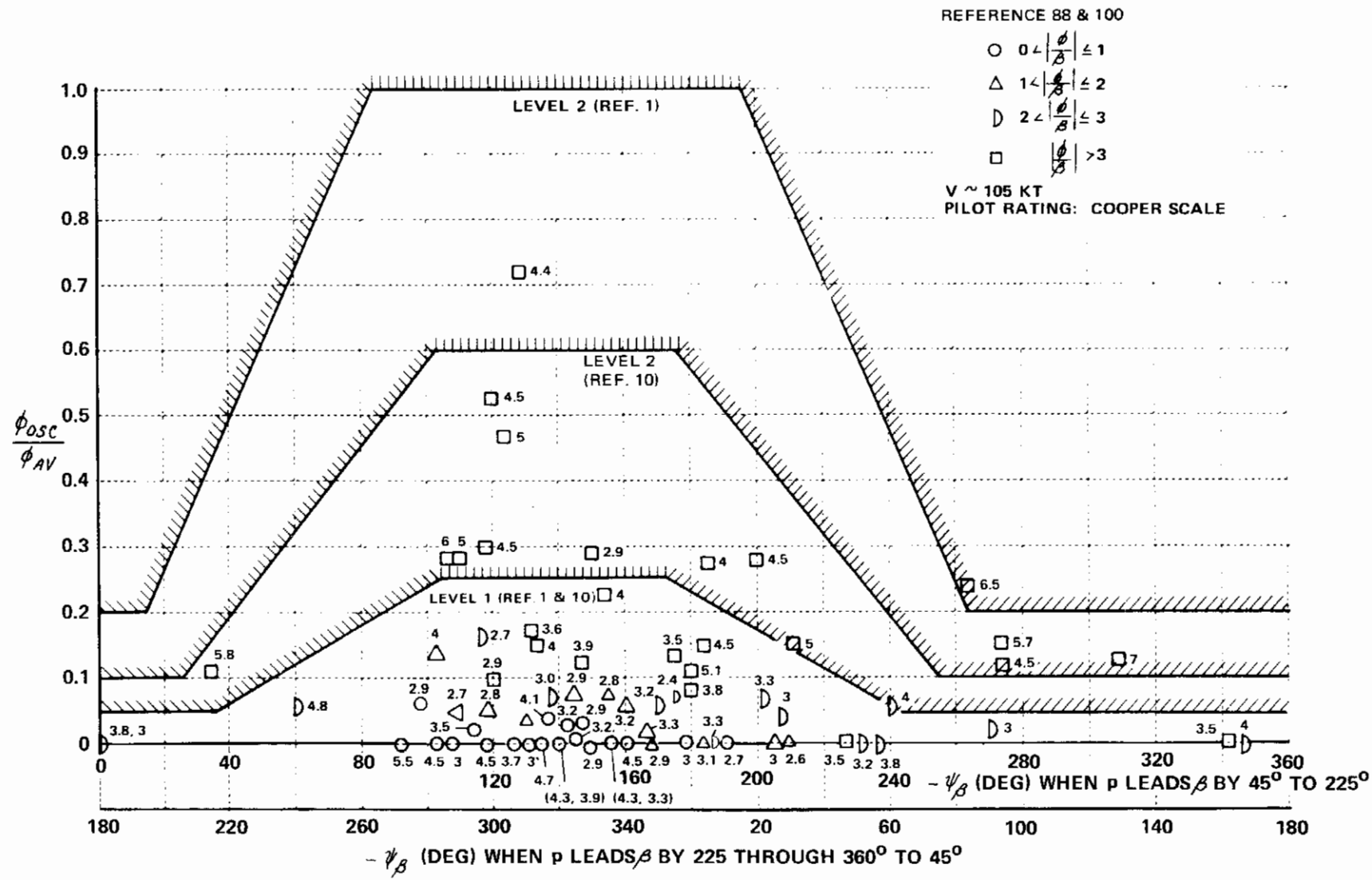

Figure 7 (3.3.8.1) COMPARISON OF ϕ_{OSC}/ϕ_{AV} FOR DATA FROM REFERENCES 88 & 100 WITH REQUIREMENTS OF REFERENCES 1 & 10

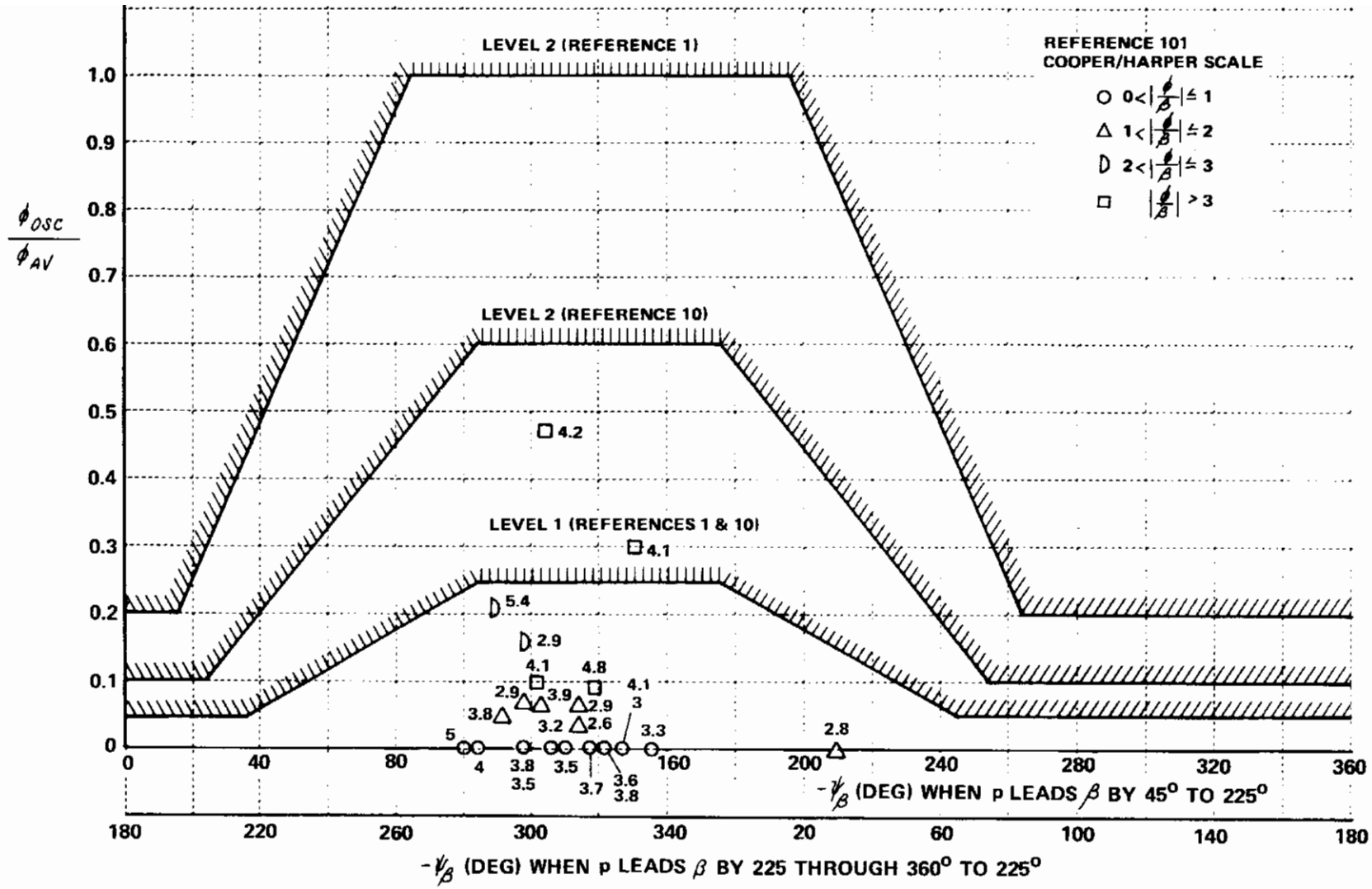

Figure 8 (3.3.8.1) COMPARISON OF ϕ_{OSC}/ϕ_{AV} FROM DATA OF REFERENCE 101 WITH REQUIREMENTS OF REFERENCES 1 & 10

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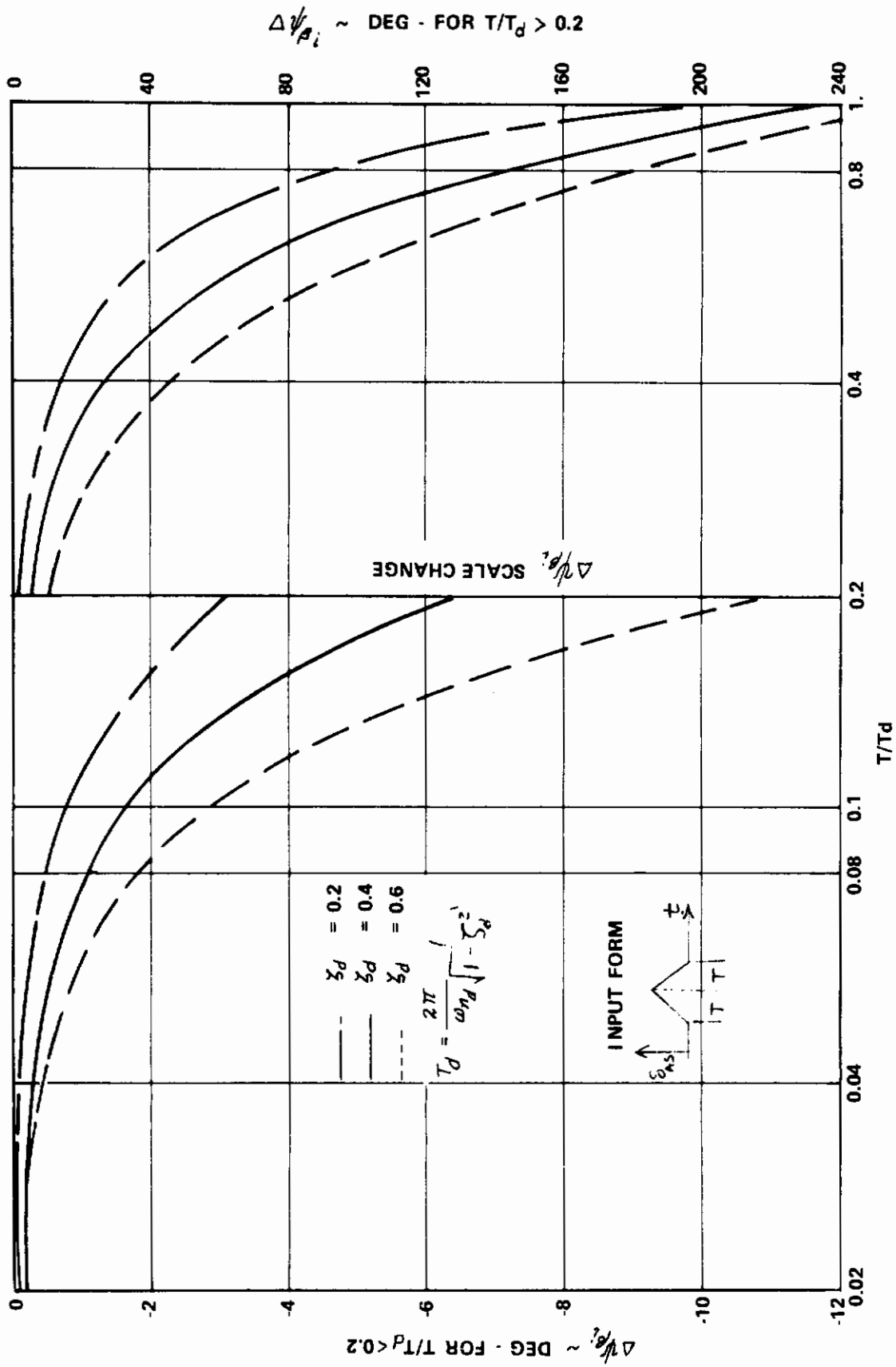

Figure 9 (3.3.8.1) CHANGES DUE TO NONIDEAL IMPULSE INPUT (FROM REFERENCE 102)

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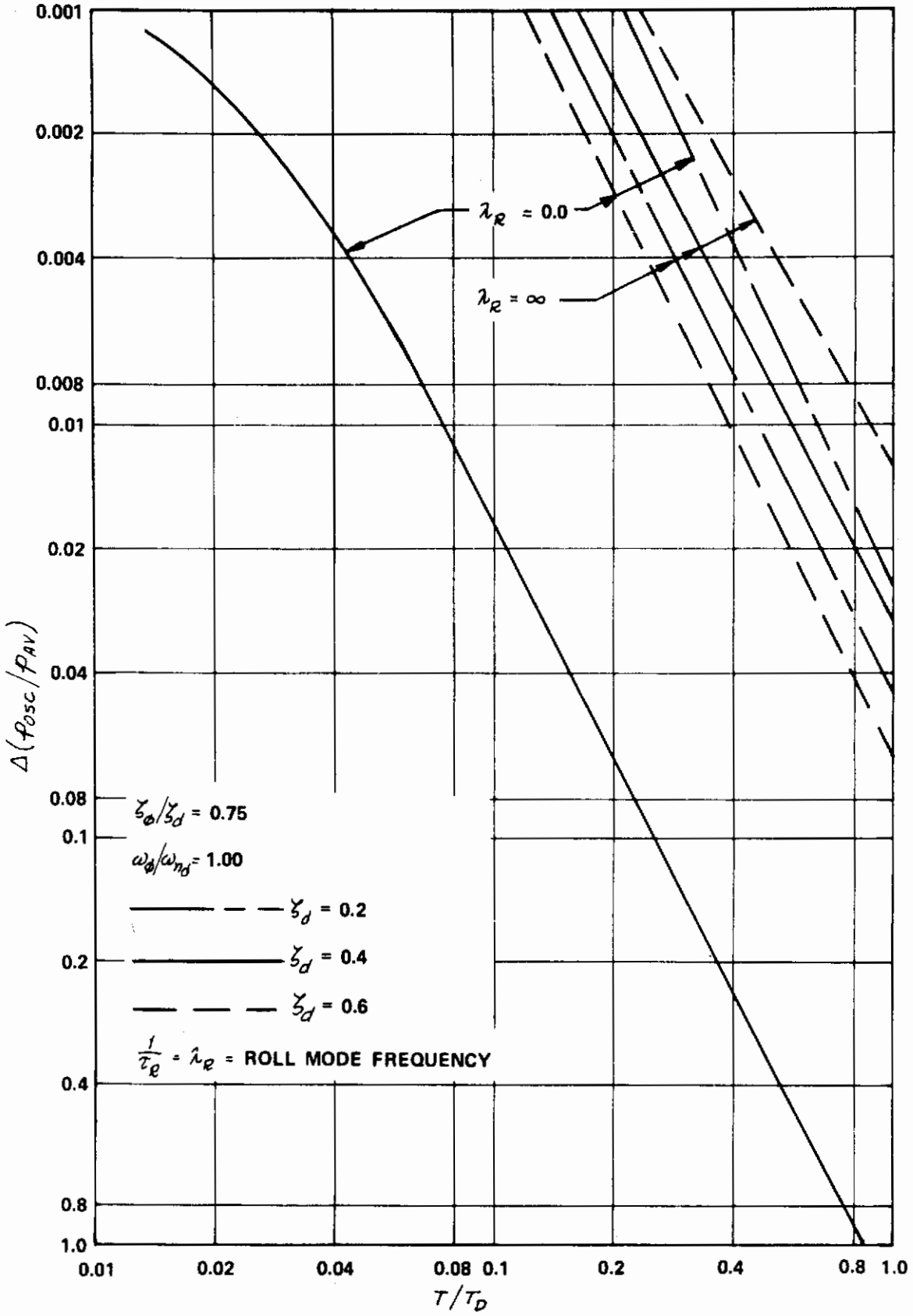

Figure 10 (3.3.8.1) EFFECTS OF DUTCH ROLL CHARACTERISTICS ON $\Delta(p_{osc}/p_{RV})$ AT ROLL MODE FREQUENCIES OF 0.0 AND ∞ (FROM REFERENCE 102)

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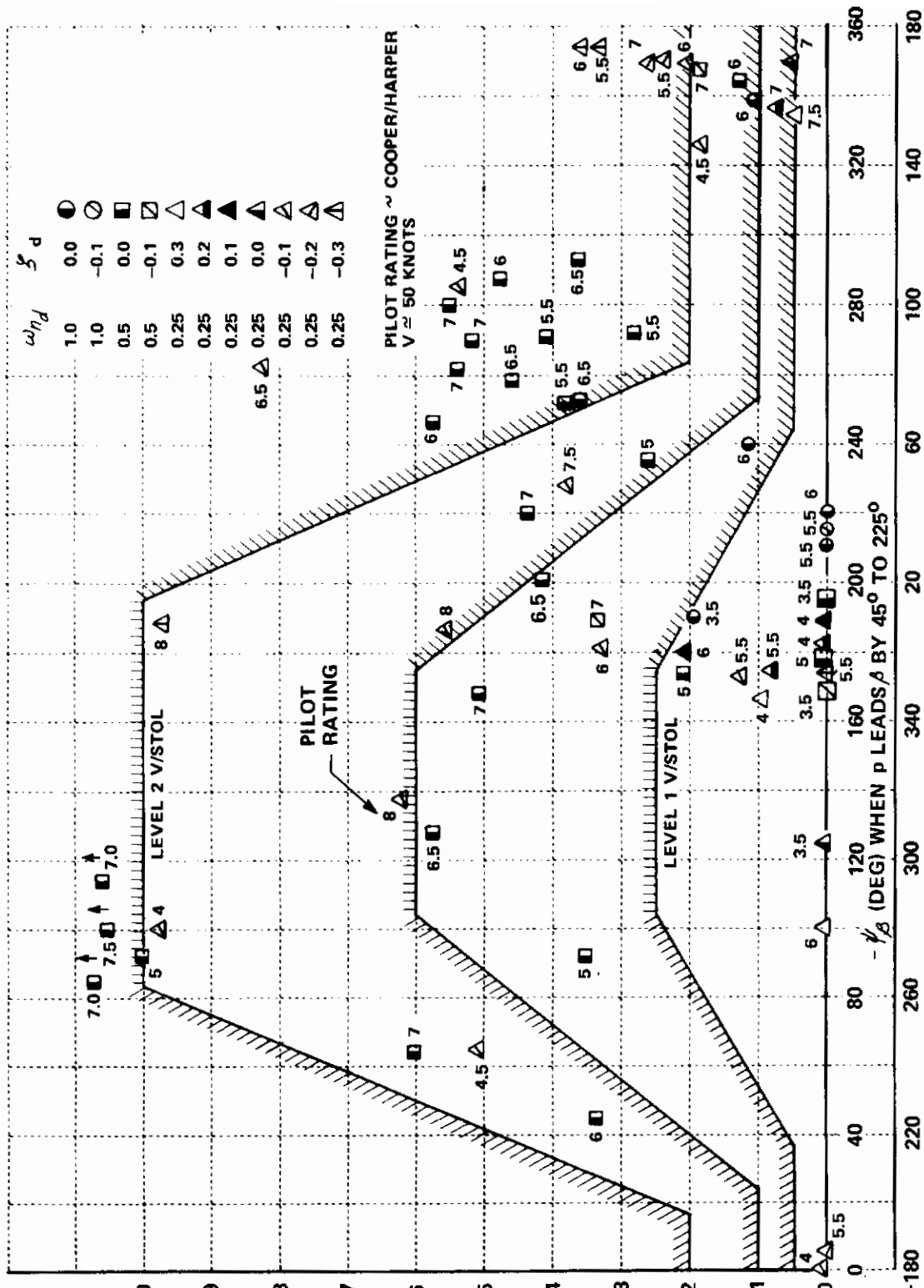

Figure 11 (3.3.8.1) EFFECTS OF VARIOUS LEVELS OF ϕ_{OSC}/ϕ_{AV} WITH LEVEL 2 DUTCH ROLL MODE CHARACTERISTICS (DATA FROM REFERENCE 86)

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3.3.8.2 SIDESLIP EXCURSIONS

REQUIREMENT

3.3.8.2 Sideslip excursions. The amount of sideslip resulting from abrupt roll control commands shall not be excessive or require complicated or objectionable rudder coordination. For Flight Phase Categories A and C, the ratio of the maximum change in sideslip angle to the initial peak magnitude in roll response, $|\Delta\beta/\phi_1|$, for an abrupt roll control pulse command shall not exceed the limit specified on figure 4. In addition, $|\Delta\beta/\phi_1| \times |\phi/\beta|_d$ shall not exceed the limit specified on figure 5.

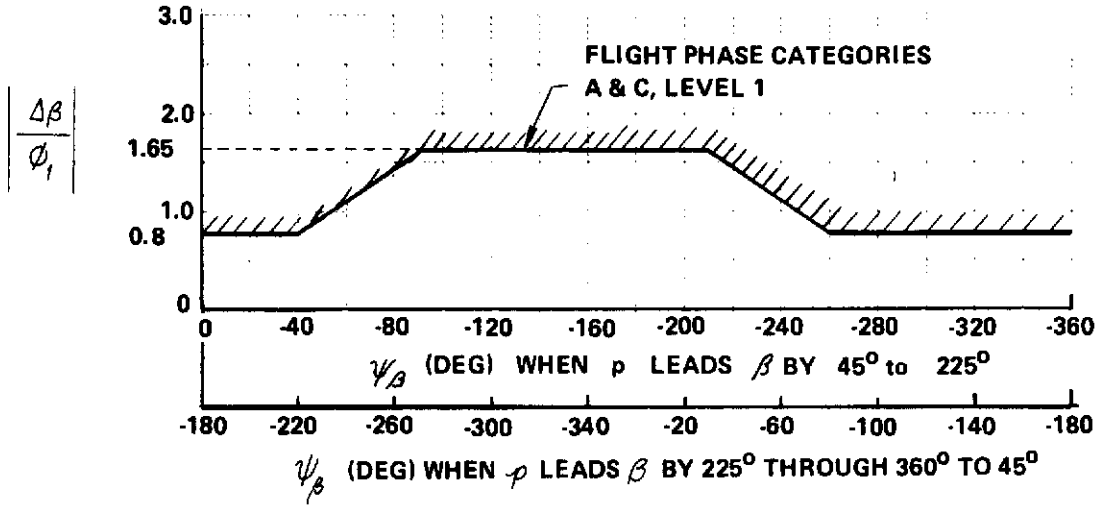

Figure 4 SIDESLIP EXCURSION LIMITATIONS

(BOUNDARY FOR $\left|\frac{\Delta\beta}{\phi_1}\right|$)

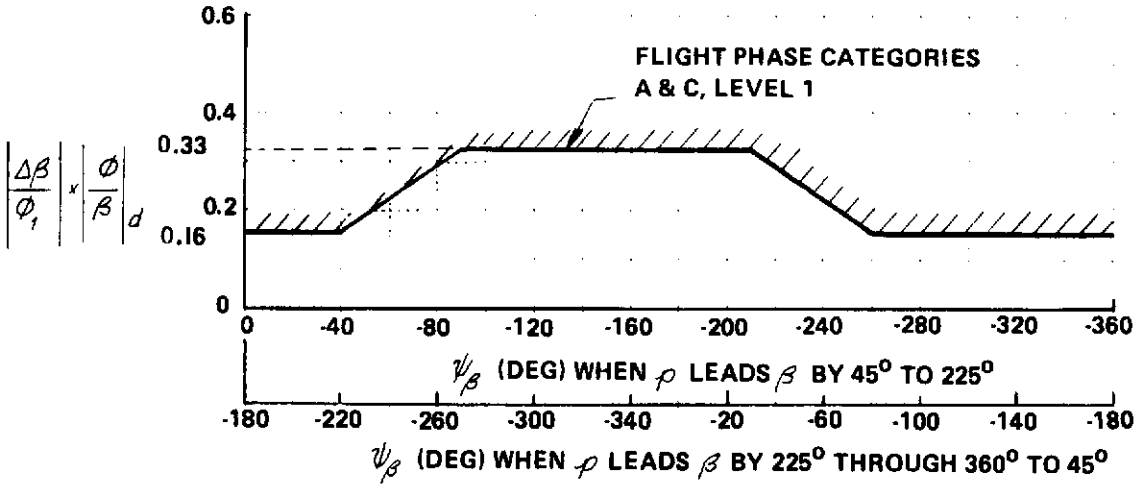

Figure 5 SIDESLIP EXCURSION LIMITATIONS

(BOUNDARY FOR $\left|\frac{\Delta\beta}{\phi_1}\right| \times \left|\frac{\phi}{\beta}\right|_d$)

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DISCUSSION

This requirement is directed at precision of control, and in particular at limiting the large sideslip angles normally generated in the speed regime between 35 knots and V_{con} when commanding ϕ . These large sideslip angles are a result of yaw due to roll control application ($N'_{\delta_{AS}}$) and yaw due to roll rate (N'_{p}). The previous discussion for requirement 3.3.8.1 indicated how these stability derivatives affect bank angle control. The discussion of this requirement (3.3.8.2) will reflect on the influence of these same derivatives on the sideslip angle response to roll control commands. The necessity to limit sideslip response is long standing and various requirements have been directed at suggesting reasonable limits. For example, AGARD 408A (Reference 46) and RTM 37 (Reference 47) have an identical requirement under the heading "Adverse Yaw" which essentially limits sideslip angle during an abrupt rolling maneuver to 15 degrees SAS on, and 20 degrees SAS off. MIL-F-8785B (Reference 10) has requirements to limit sideslip excursion, however, it is necessary to use a scaling parameter that is dependent upon the roll control effectiveness requirements to demonstrate compliance. While scaling of the undesired response (sideslip excursion) with some measure of the desired response (bank angle change) for a roll control command is considered necessary to the development of a pertinent requirement, the dependence on roll performance requirements may introduce unnecessary and unwarranted complications in a requirement designed to limit sideslip excursion.

NASA investigations into the lateral-directional requirements for aircraft operating at STOL airspeeds, Reference 89, utilize the ratio of peak sideslip angle to peak bank angle in "rapid turn entries," as the cross-coupling parameter. The results of this investigation are illustrated in Figure 1(3.3.8.2). This data, and additional data discussed in Reference 73 illustrated on Figure 2(3.3.8.2), had led NASA to prescribe recommendations that $\Delta\beta/\Delta\phi$ be not greater than 0.3 for satisfactory operation (Pilot Rating 3.5) and 0.6 for safe operation (Pilot Rating 6.5). As defined, this allows any amount of proverse yaw since $\Delta\beta/\Delta\phi$ will then be negative. Figure 1(3.3.8.2) suggests that only small amounts of proverse yaw are acceptable for safe operation so presumably the intention of Reference 73 is to exclude proverse yaw entirely and restrict adverse yaw to $0 < \Delta\beta/\Delta\phi < 0.3$ for satisfactory operation and $0 < \Delta\beta/\Delta\phi < 0.6$ for safe operation. While the ratio $\Delta\beta/\Delta\phi$ appears to be a pertinent parameter, certain problems exist because of the possibly vague definitions given for $\Delta\beta$ and for a "rapid turn entry maneuver with yaw control fixed." One problem is that the definition of $\Delta\beta_{max}$ i.e., peak sideslip excursion, needs clarification for configurations where the sideslip response to roll control commands does not exhibit the clearly adverse yaw condition illustrated on Figure 3(3.3.8.2), from Reference 103.

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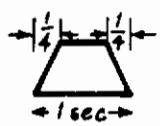
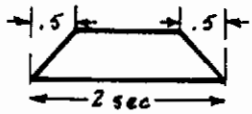
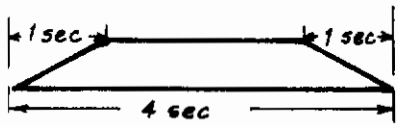
Another problem is the definition of the "rapid turn entry" maneuver used to determine roll-sideslip coupling. The abruptness and duration of the step-in step-out roll control command, which has generally not been precisely defined in the NASA investigations, can significantly influence the ratio $\Delta\beta/\Delta\phi$. The effect of the input on $\Delta\beta/\Delta\phi$ becomes important as the duration of the input approaches the period of the Dutch roll mode. This effect is illustrated by values of $\Delta\beta/\Delta\phi$ calculated for the following configuration selected from the in-flight experiment (Reference 86), configuration LH 124 + 20 + 24, for which $|\phi/\beta|_d = .2$, $\omega_{n_d} = 1.0$ rad/sec, $\omega_\phi = 1.24$ rad/sec, $\zeta_d = .2$, $\zeta_\phi = .24$ and Pilot Rating = 6, 7. Thus for geometrically similar inputs:

Input	$\Delta\beta/\Delta\phi$ (as described in Reference 73)
Pure Impulse	1.021
	0.998
	0.924
	0.670

From the above, it becomes obvious that the change in the parameter $\Delta\beta/\Delta\phi$ associated with input duration can exceed that normally anticipated from experimental errors, and the NASA claim that the parameter $\Delta\beta/\Delta\phi$ is generally not dependent on control input rapidity is highly questionable. As illustrated, a configuration which would not satisfy the Reference 73 criteria for a very abrupt turn entry, could be made to satisfy the criteria by nothing more than a modification of pilot control technique. It may be noticed that the time history reproduced on Figure 3(3.3.8.2) shows an aileron input duration of almost 3.0 seconds, which could give about 20% error in $\Delta\beta/\Delta\phi$.

The preceding discussion illustrates the importance of requiring a specific input in order to determine sideslip excursions. Figure 4(3.3.8.2) compares pilot rating data obtained in Reference 86 with the criteria of Reference 73. The value of $\Delta\beta/\Delta\phi$ computed for the configurations examined is based on the Reference 73 definitions. The input command used is a perfect impulse of the roll control with yaw control fixed. Examination of Figure 4(3.3.8.2) indicates that the values of $\Delta\beta/\Delta\phi$ recommended by Reference 73 would be overly restrictive for many of the configurations tested, and as $\Delta\beta/\Delta\phi$ increases (for adverse yaw) it is apparent that the sensitivity of pilot rating to $\Delta\beta/\Delta\phi$ diminishes. This is related to the sensitivity

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discussion presented previously on the discussion of 3.3.8.

Figures 5(3.3.8.2) and 6(3.3.8.2) present a summary of data on the relationship between $N'_{\delta_{AS}}/L'_{\delta_{AS}}$, and $(N'_p - g/U_o)/L'_p$ and pilot rating. As previously discussed in 3.3.8.1, setting $\frac{N'_p - (g/U_o)}{L'_p} = \frac{N'_{\delta_{AS}}}{L'_{\delta_{AS}}}$ minimizes ϕ_{osc}/ϕ_{AV}. As shown on Figures 5(3.3.8.2) and 6(3.3.8.2), this does not necessarily result in Level 1 pilot ratings. However, in general, approaching this relationship does tend to improve pilot rating. Examination of the numerator of a conventional β/δ_{AS} transfer function indicates that $N'_{\delta_{AS}}/L'_{\delta_{AS}}$ and $(N'_p - g/U_o)/L'_p$ are important parameters in the higher order terms of the β/δ_{AS} transfer function numerator, thus the initial response of sideslip to a roll control impulse should serve as an important measure of roll sideslip coupling. This measure of initial β response, defined in the V/STOL Specification as $\Delta\beta$, was examined for the same configurations previously described in section 3.3.8.1[Figures 1(3.3.8.1) through 4(3.3.8.1)] and the variation in $\Delta\beta$ as a function of $N'_{\delta_{AS}}/L'_{\delta_{AS}}$ and $(N'_p - g/U_o)/L'_p$ is illustrated on Figure 7(3.3.8.2).

As mentioned previously, it is probably desirable to relate the undesired response (sideslip) to the desired response (bank angle) for the roll control command in order to examine pilot opinion as a function of the undesired response. Thus it is necessary to scale the $\Delta\beta$ response by a pertinent measure of the bank angle response, and ϕ_1 was selected. Thus the metric for sideslip excursion is related to $|\Delta\beta/\phi_1|$, and it should be noted that this measure is similar to the NASA metric $\Delta\beta/\Delta\phi$; however, as defined in the V/STOL Specification $\Delta\beta$ is applicable to adverse yaw, proverse yaw and combinations thereof.

Figure 8(3.3.8.2) illustrates the effect of yaw due to aileron on $|\Delta\beta/\phi_1|$. Figure 9(3.3.8.2) indicates the effects of yaw due to aileron on both $|\Delta\beta/\phi_1|$ and ψ_β. Also presented on Figure 9(3.3.8.2) are hatched lines representing pilot ratings of 3.5 based on the Cooper-Harper rating scale used by NRC. These ratings were obtained by plotting pilot rating for each set of configurations presented on Figure 9(3.3.8.2) against $N'_{\delta_{AS}}/L'_{\delta_{AS}}$ and using nonlinear regression analysis to fit the pilot rating data. Figure 10(3.3.8.2) compares the results of this analysis with the $|\Delta\beta/\phi_1|$ requirement of the V/STOL Specification.

The preceding analysis illustrated the development of $|\Delta\beta/\phi_1|$, however it used data for a $|\phi/\beta|_d = 0.2$ which is quite low [see Figure 11(3.3.8.2)]. Examination of the simplified expressions for Δ_1 and Δ_2 previously described in the discussion on ϕ_{osc}/ϕ_{AV} (paragraph 3.3.8.1) indicates that a more pertinent parameter should also include the effect of L'_β/N'_β (e.g., $|\phi/\beta|_d$). It was decided to investigate the product $|\Delta\beta/\phi_1| \times |\phi/\beta|_d$ as a parameter to correlate pilot rating with sideslip excursion. However, as $|\phi/\beta|_d \to 0$ this new metric approaches zero and a requirement on the product would become meaningless. Thus it is necessary to place requirements not only on $|\Delta\beta/\phi_1| \times |\phi/\beta|_d$ but also $|\Delta\beta/\phi_1|$ in particular for small values of $|\phi/\beta|_d$. Figures 12(3.3.8.2) through 16(3.3.8.2) present a correlation of the sideslip excursion parameter with pilot rating. Since the data base used to develop this requirement did not include Flight Phase

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Category B, and, in addition, since the sensitivity of $|\Delta\beta/\phi_1|$ decreases with large amounts of adverse yaw due to aileron, no quantitative sideslip requirement has been established for either Flight Phase Category B or Level 2 flying qualities. For the present time it is anticipated that the qualitative statement in the sideslip excursion requirement and the ϕ_{osc}/ϕ_{AV} requirement will adequately cover these areas. However, additional data are required to fully describe and restrict roll-sideslip coupling and to establish if the Class of the aircraft has a significant effect on the selected measures of roll-sideslip coupling in the speed range from 35 knots forward to V_{con}.

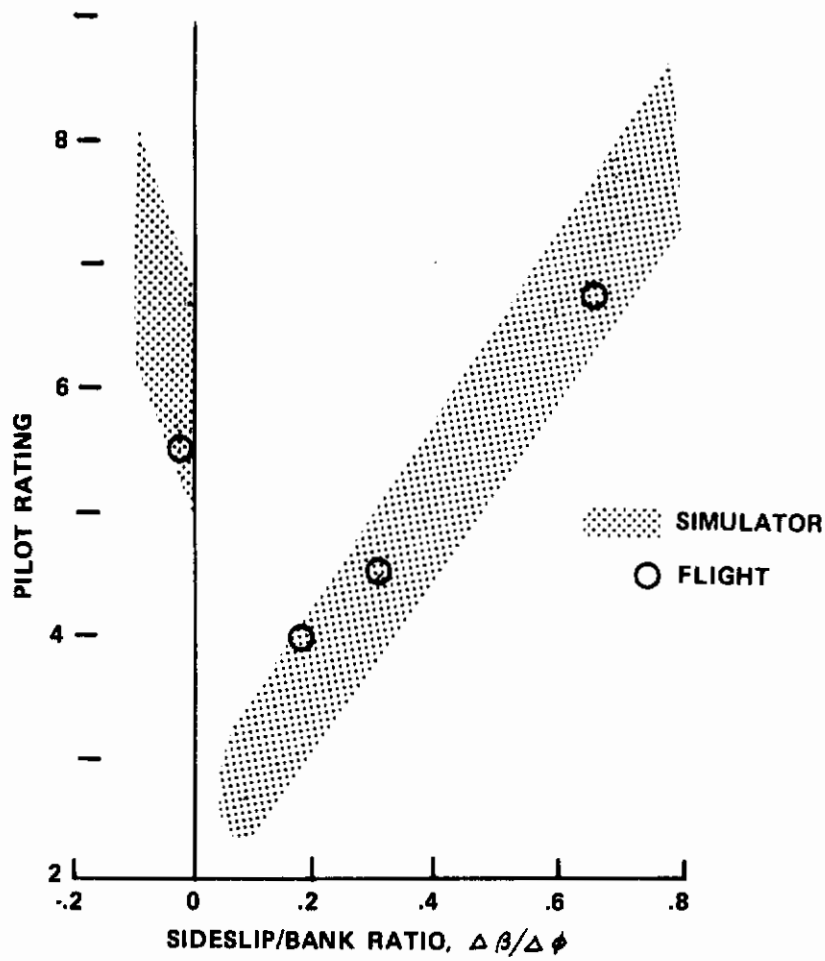

Figure 1 (3.3.8.2) EFFECT OF SIDESLIP ON PILOT RATING
$\zeta_d\ \omega_{n_d} \approx .09,\ .52 < \omega_d < 1.05$, **(FROM REFERENCE 89)**

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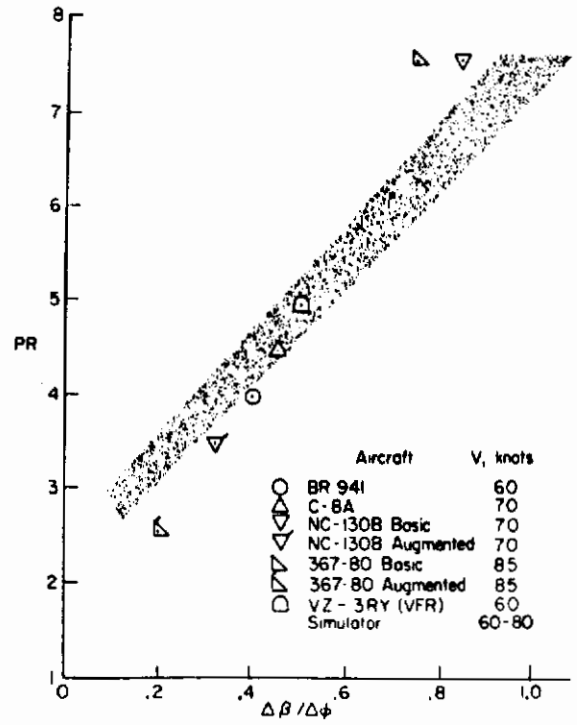

Figure 2 (3.3.8.2) RELATION OF TURN ENTRY COORDINATION AND PILOT OPINION IN IFR (FROM REFERENCE 73)

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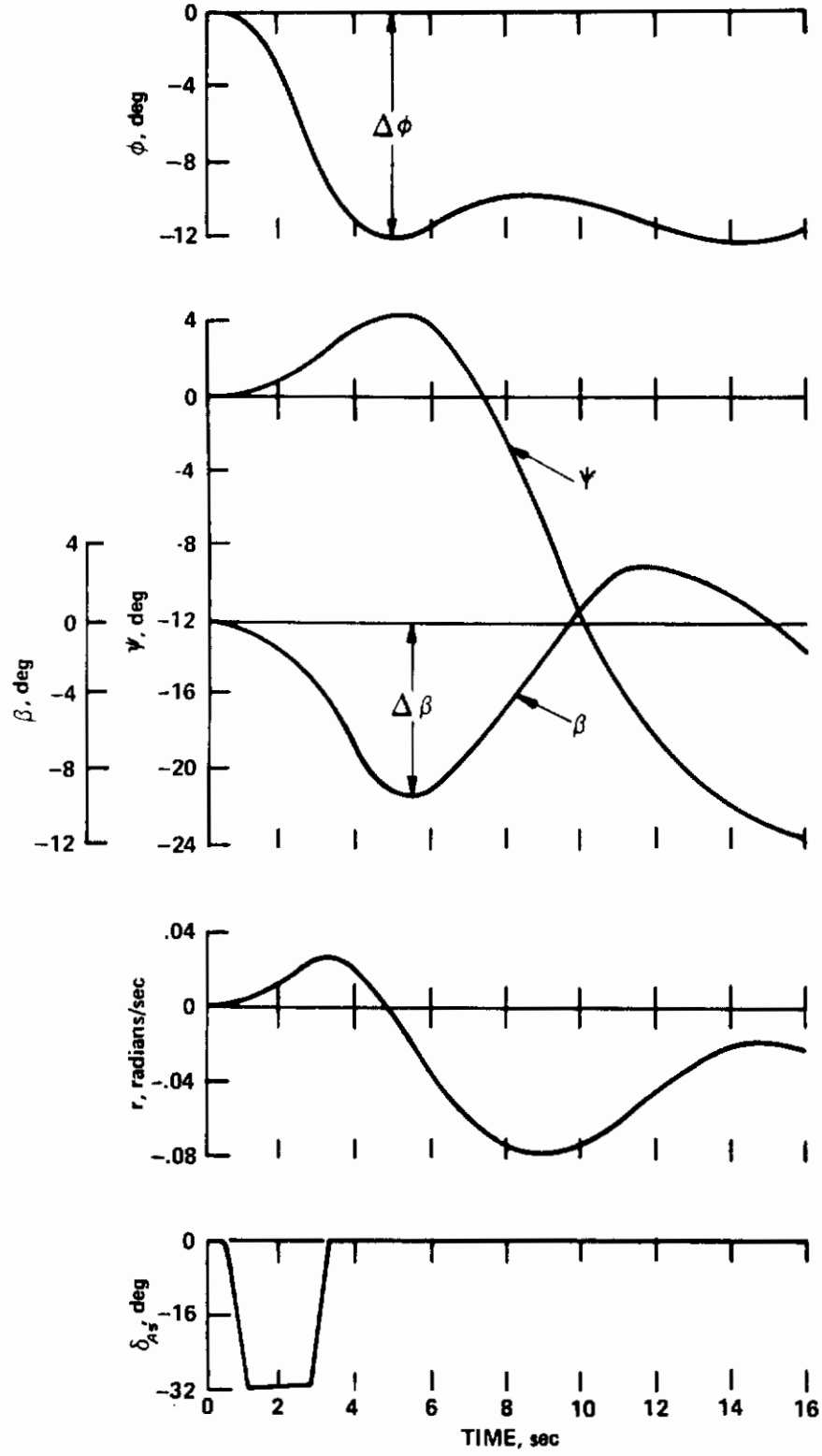

Figure 3 (3.3.8.2) TIME HISTORY OF THE RESPONSE TO A LATERAL CONTROL INPUT; V ≈ 70 KNOTS (FROM REFERENCE 103)

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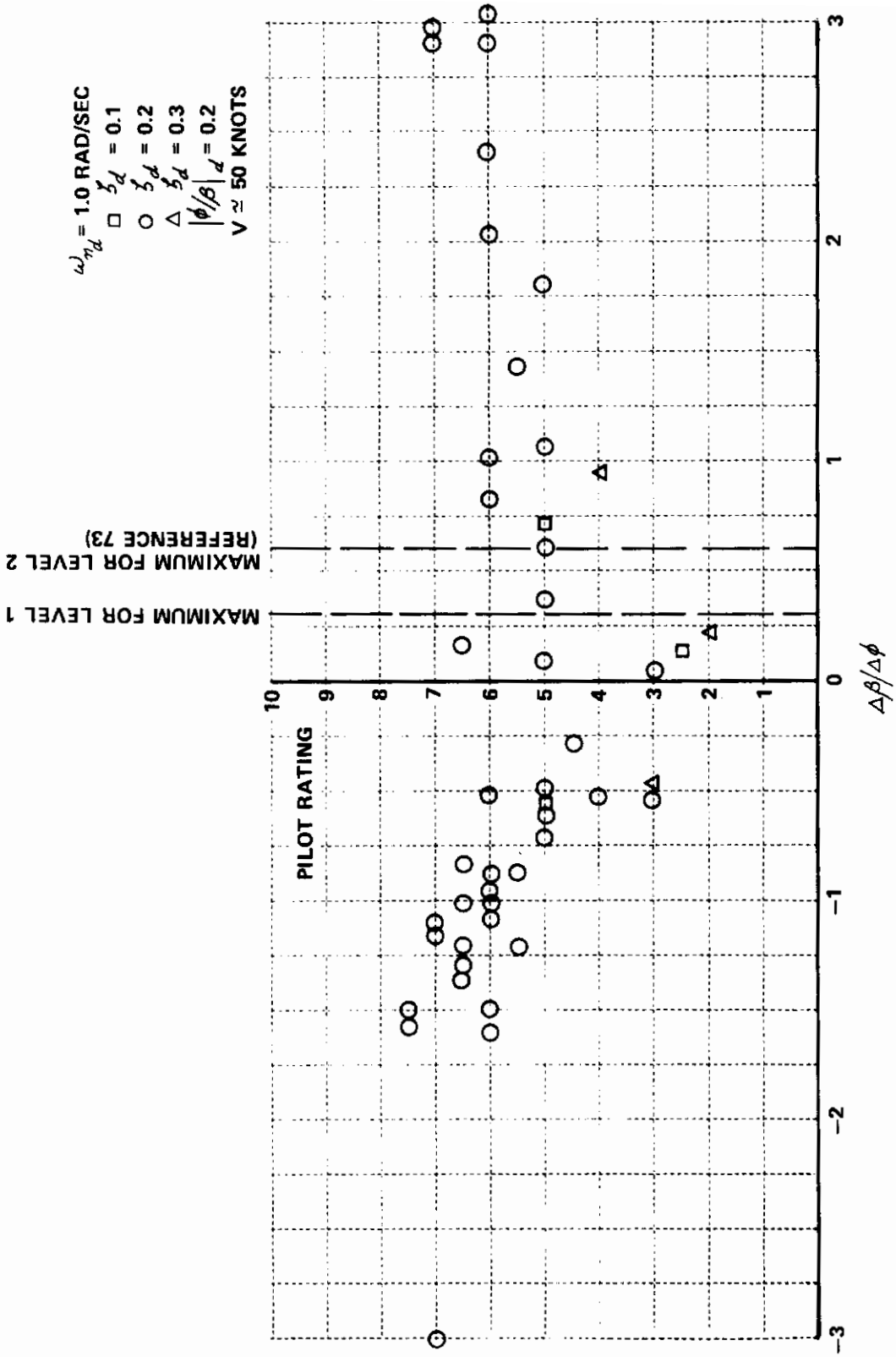

Figure 4 (3.3.8.2) EFFECT OF ROLL/SIDESLIP COUPLING ON PILOT RATING (DATA FROM REFERENCE 86)

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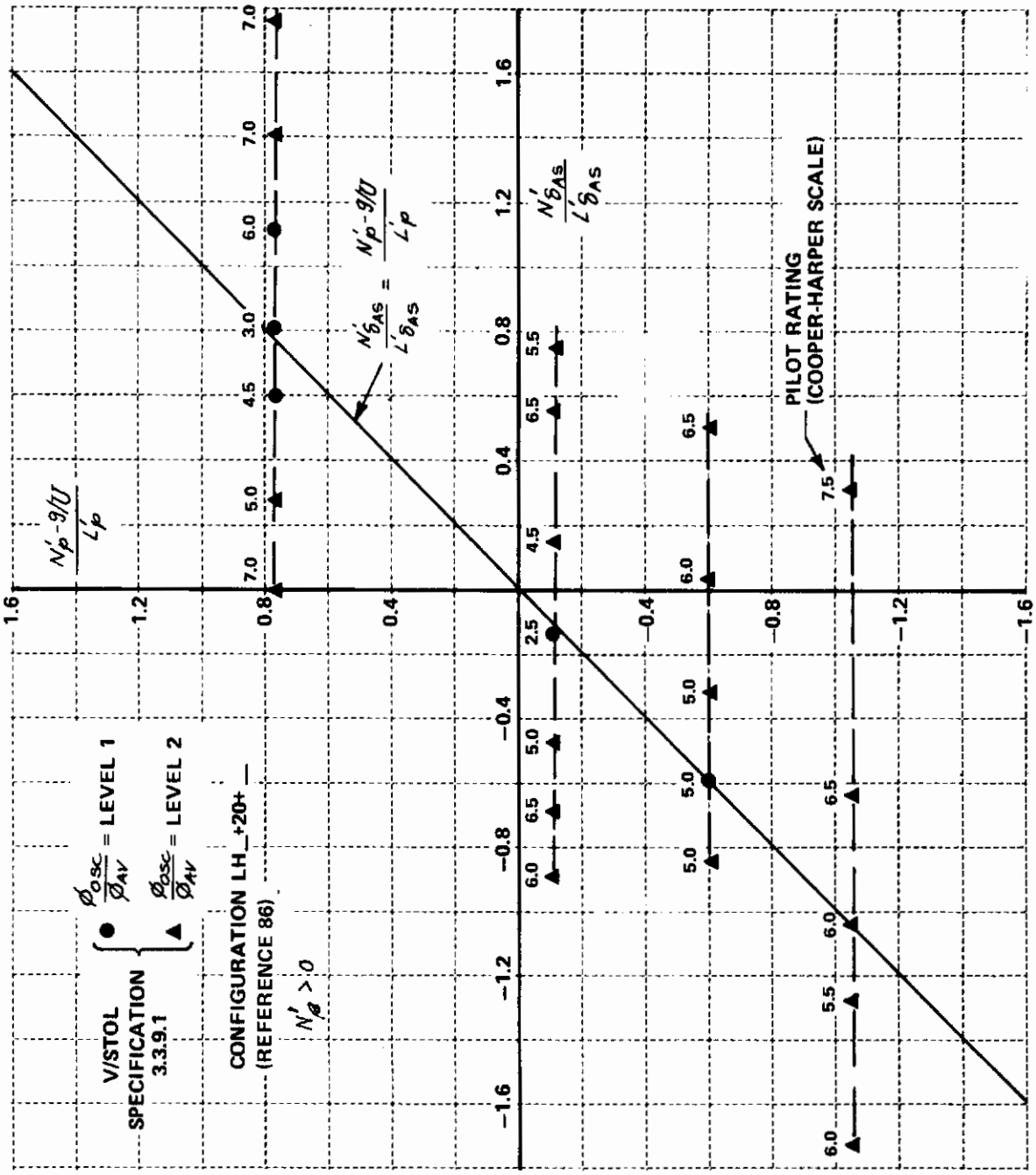

Figure 5 (3.3.8.2) EFFECT OF $N'_{\delta_{AS}}/L'_{\delta_{AS}}$ AND $(N'_p - g/U)/L'_p$ ON AVERAGE PILOT RATING FOR FIXED MODAL PARAMETERS (DATA FROM REFERENCE 86)

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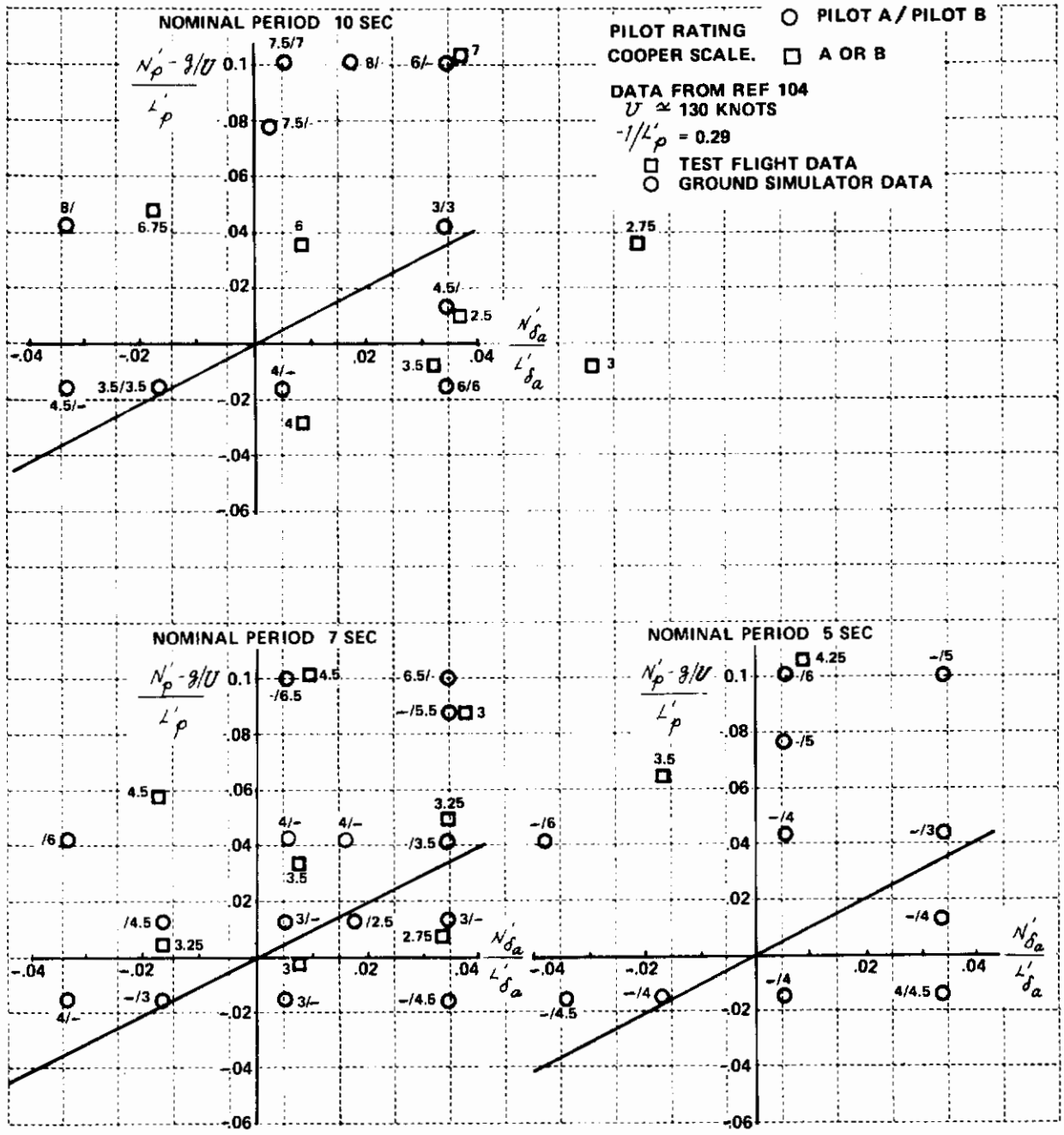

Figure 6 (3.3.8.2) COMPARISON OF PILOT RATINGS FROM GROUND SIMULATOR AND FLIGHT TEST AS RELATED TO $N'_{\delta AS}/L'_{\delta AS}$ AND $\frac{L'_p - g/U}{L'_p}$ (DATA FROM REFERENCE 104)

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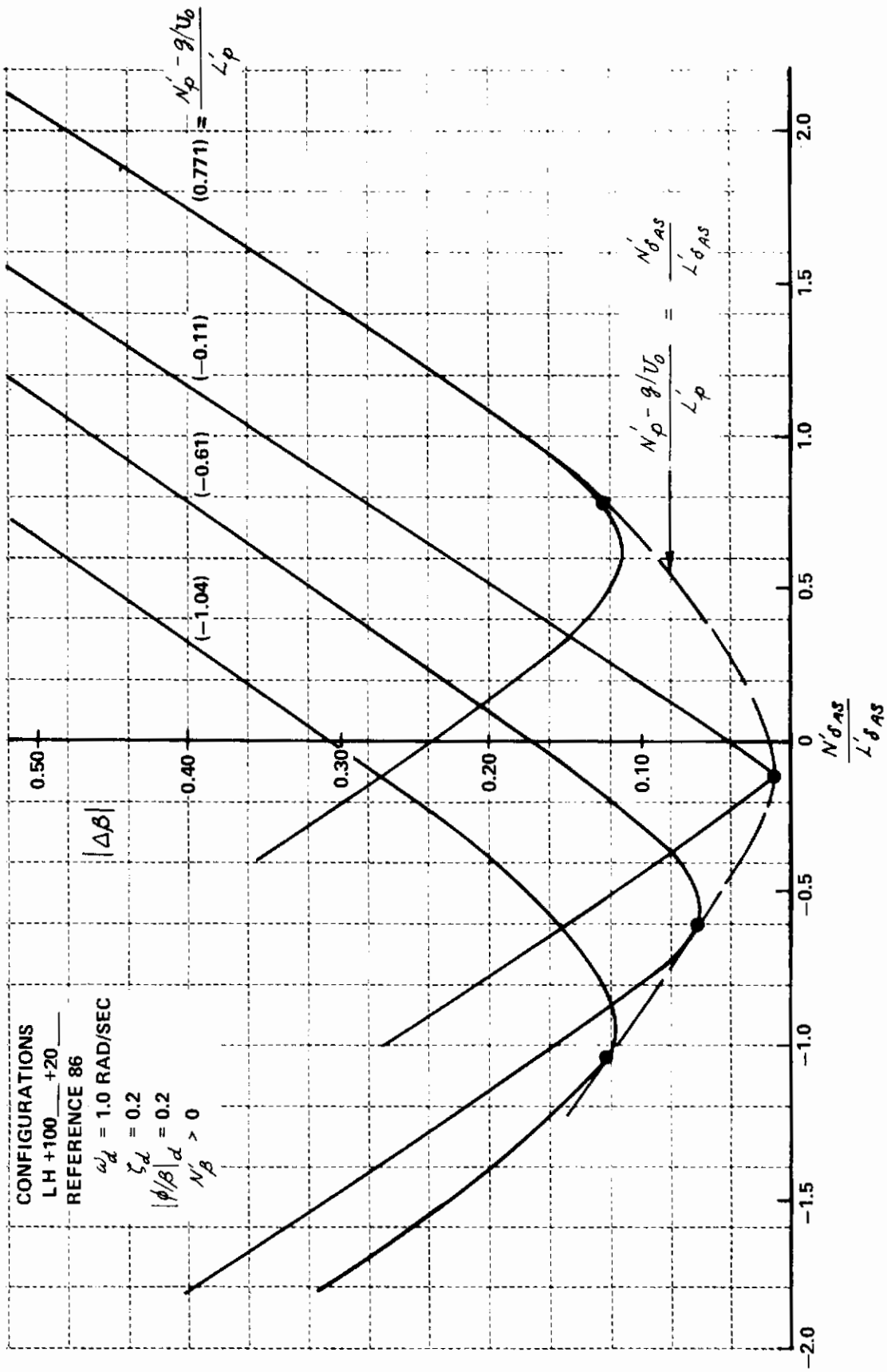

Figure 7 (3.3.8.2) EFFECT OF YAW DUE TO AILERON ON $|\Delta\beta|$

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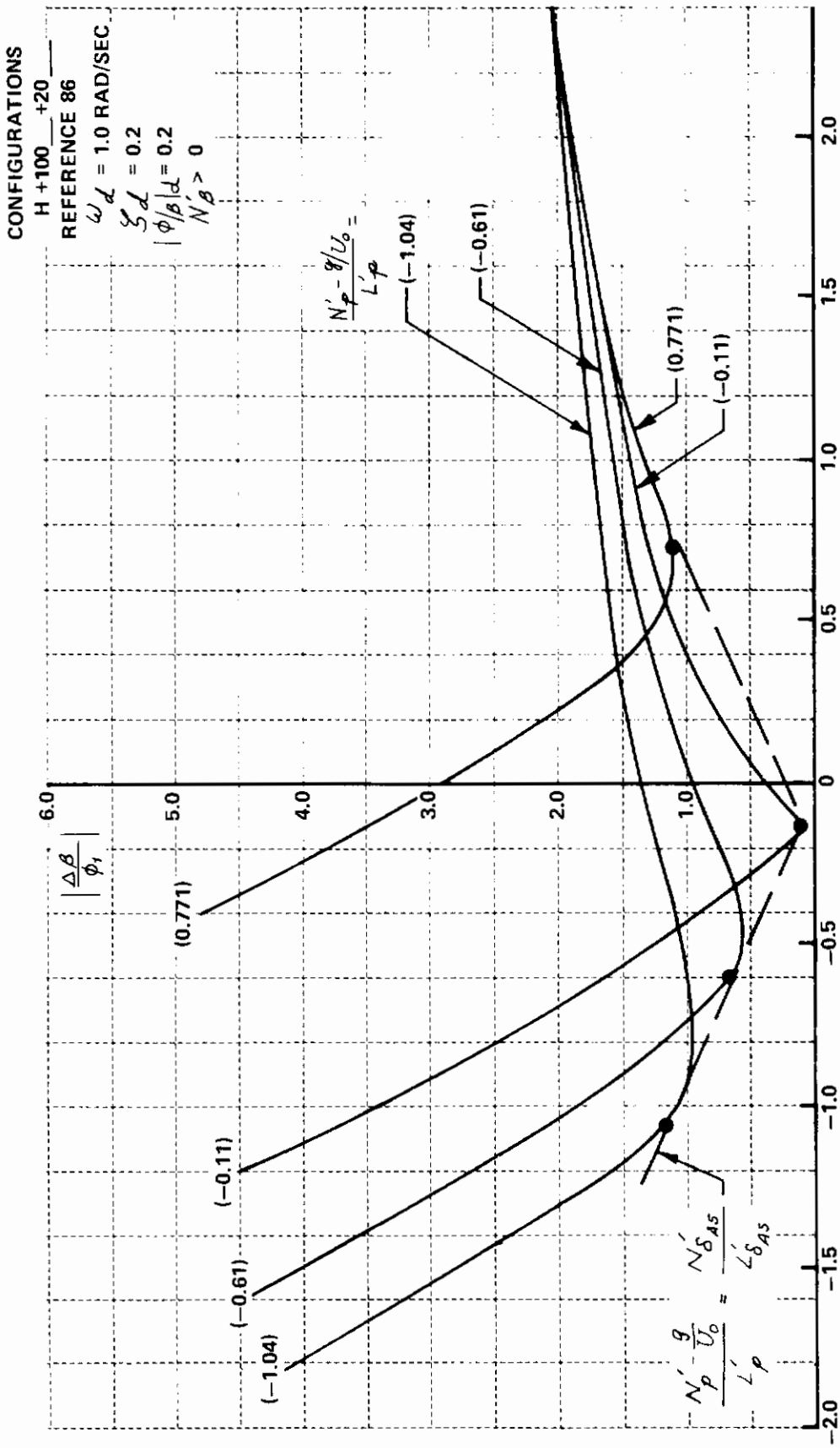

Figure 8 (3.3.8.2) EFFECT OF YAW DUE TO AILERON ON $\left|\frac{\Delta\beta}{\phi_1}\right|$

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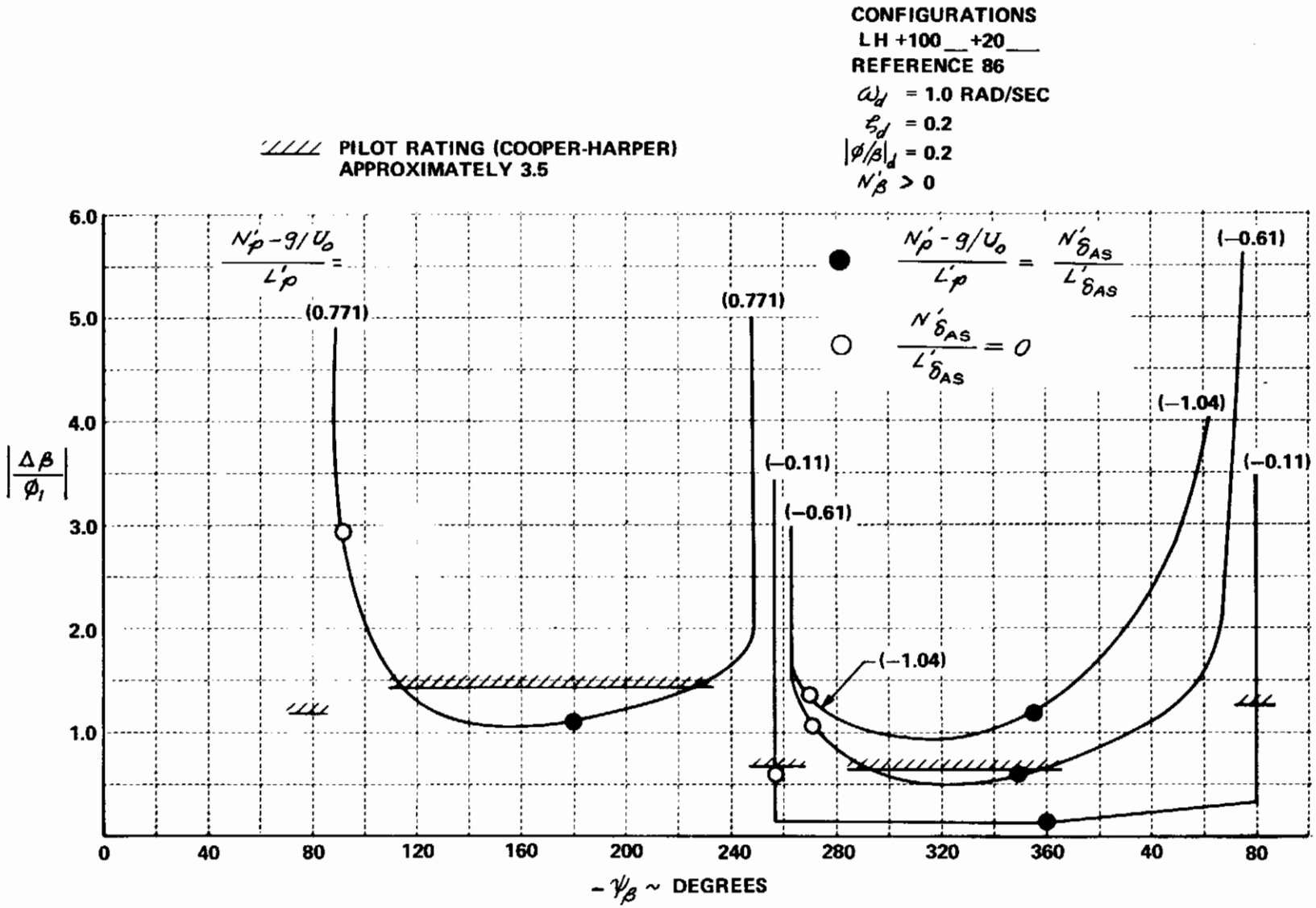

Figure 9 (3.3.8.2) EFFECT OF YAW DUE TO AILERON ON ψ_β AND $|\Delta\beta/\phi_1|$

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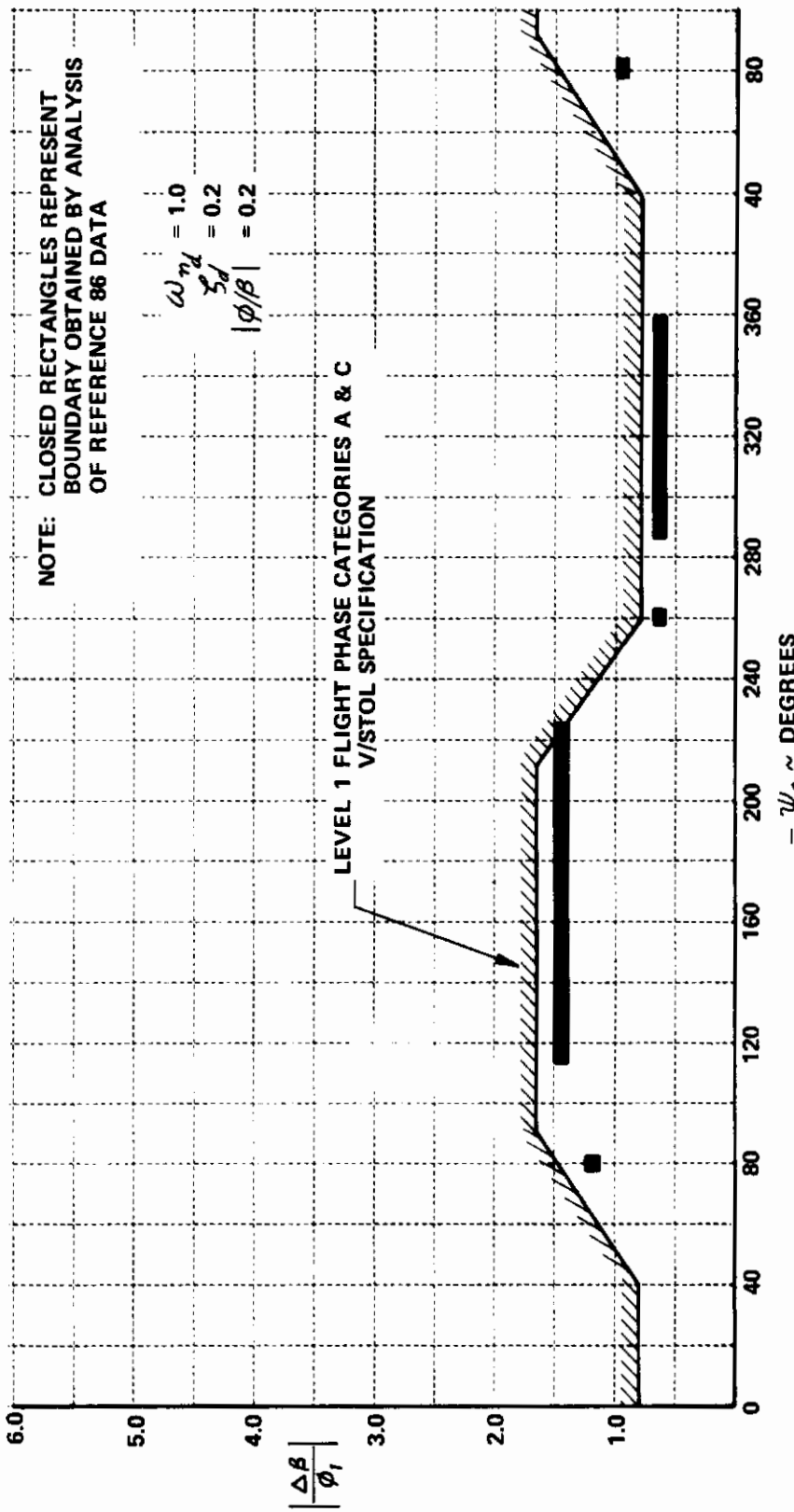

Figure 10 (3.3.8.2) COMPARISON OF V/STOL SPECIFICATION ON $|\Delta\beta/\phi_1|$ WITH NONLINEAR REGRESSION ANALYSES OF PILOT RATINGS FOR SELECTED CONFIGURATIONS OF REFERENCE 86

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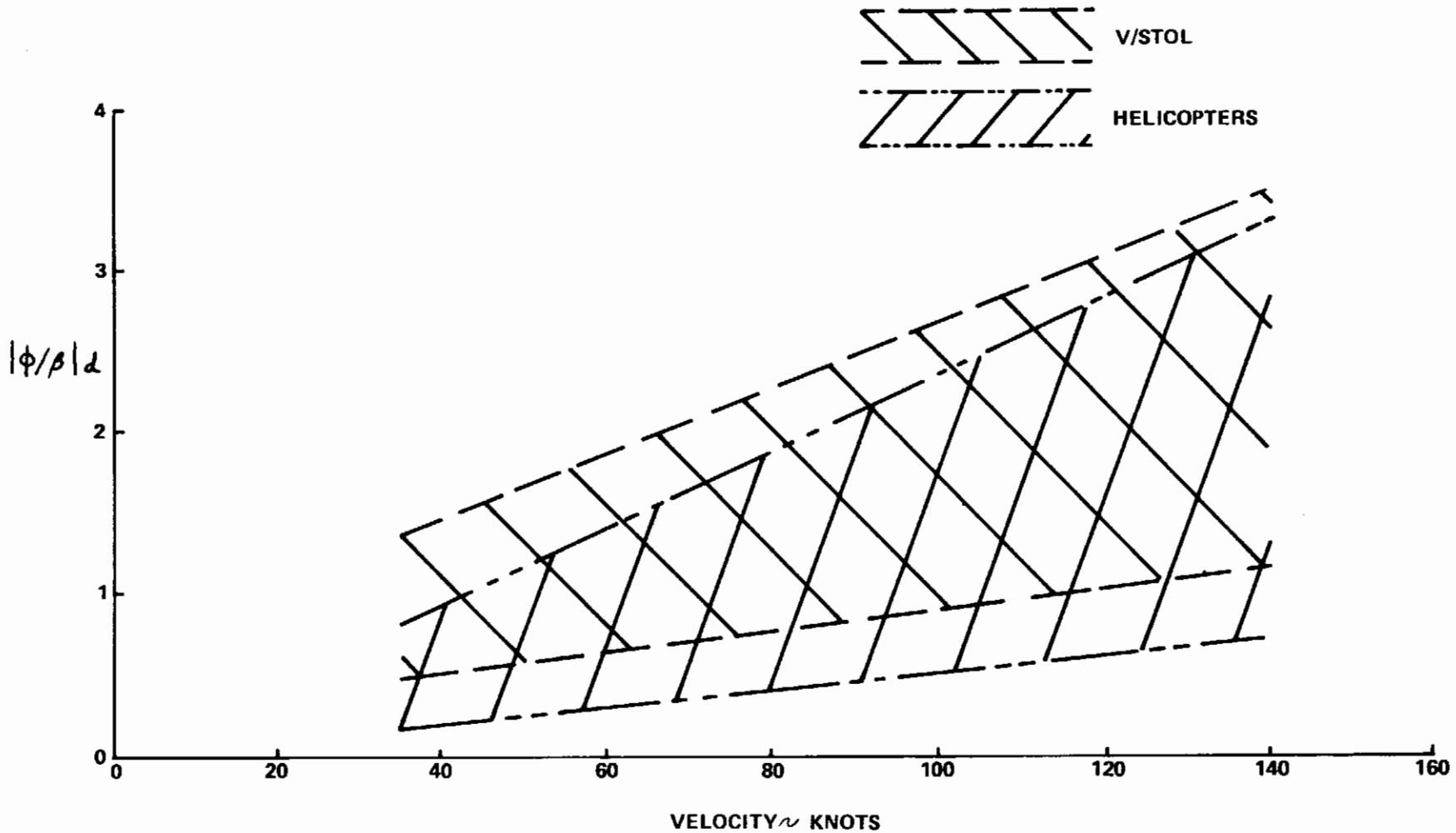

Figure 11 (3.3.8.2) VARIATION IN THE VALUES OF $|\phi/\beta|_d$ ASSOCIATED WITH V/STOL AIRCRAFT (BASED ON DATA IN REFERENCES 13 & 16)

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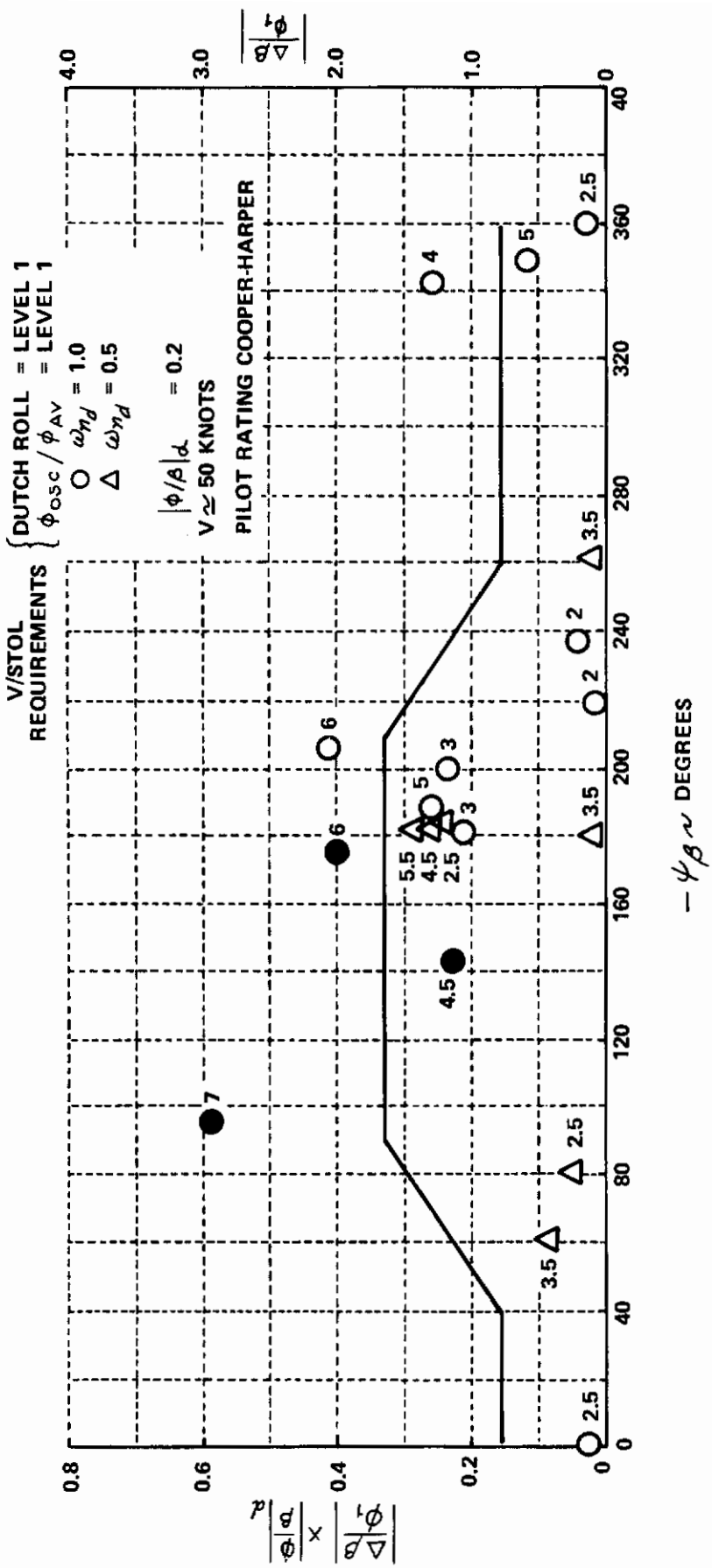

Figure 12 (3.3.8.2) COMPARISON OF SIDESLIP EXCURSION REQUIREMENT AND PILOT RATING (DATA FROM REFERENCE 86)

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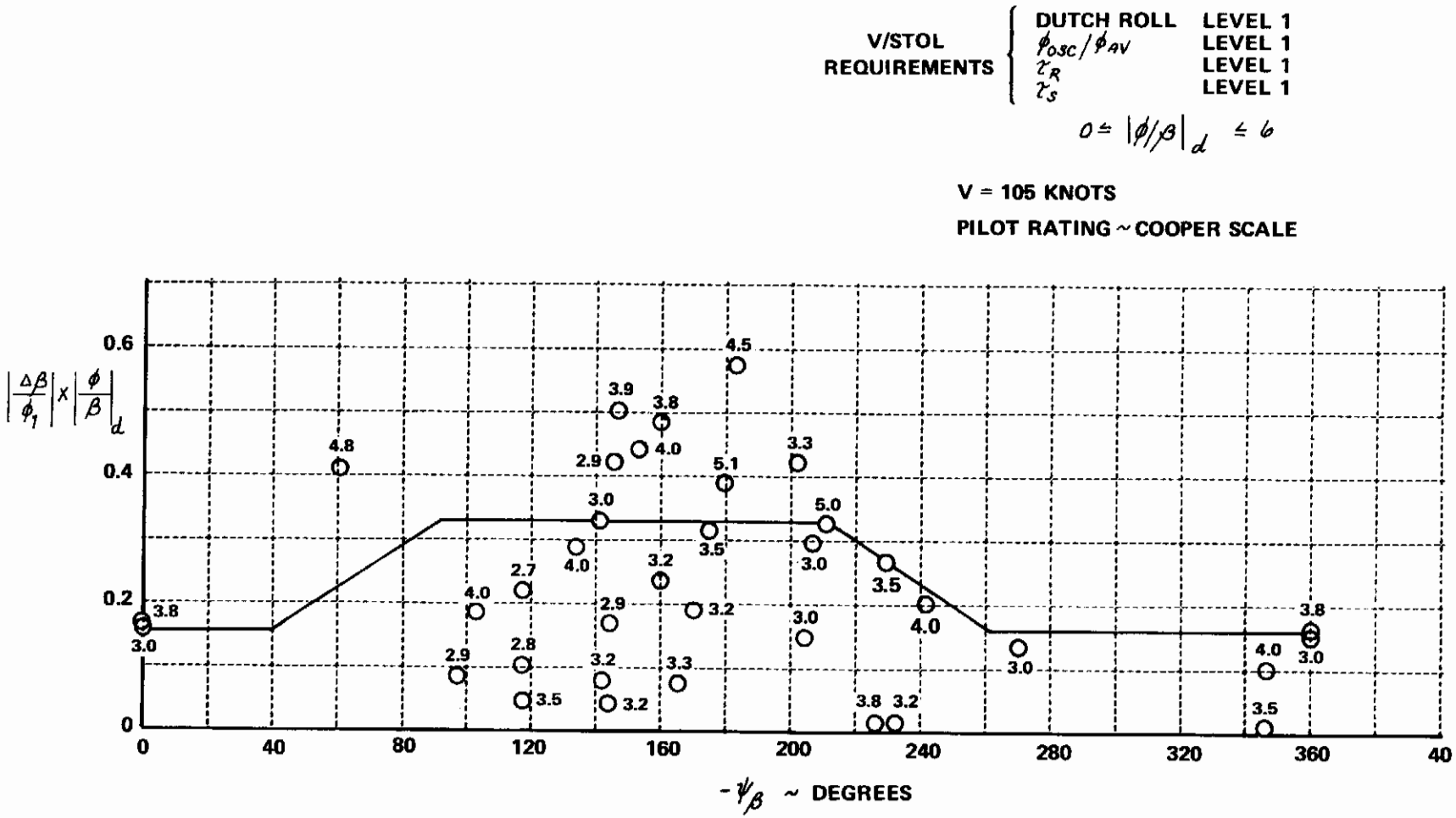

Figure 13 (3.3.8.2) COMPARISON OF SIDESLIP EXCURSION REQUIREMENT AND PILOT RATING (DATA FROM REFERENCE 100)

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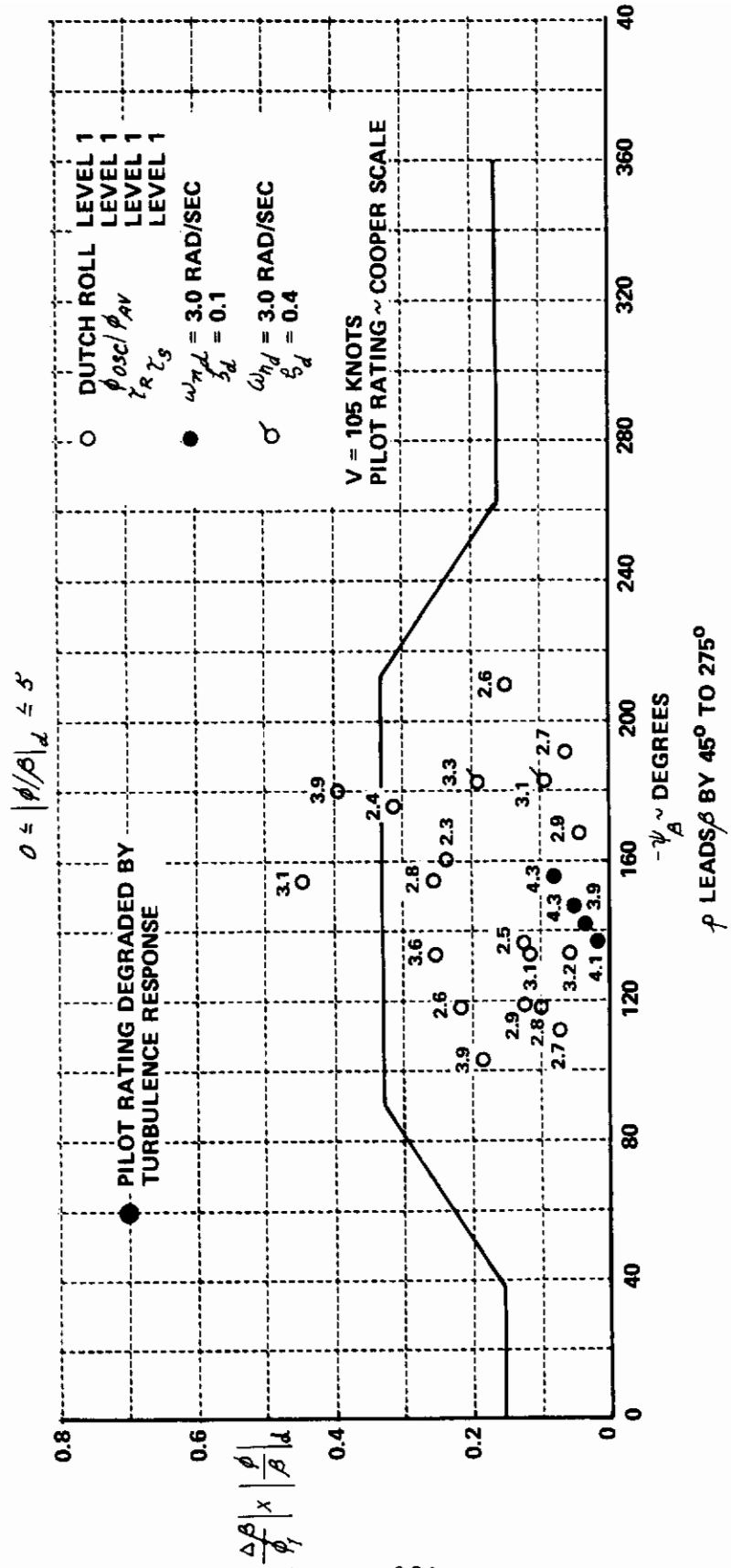

Figure 14 (3.3.8.2) COMPARISON OF SIDESLIP EXCURSION REQUIREMENT WITH PILOT RATING (DATA FROM REFERENCE 88)

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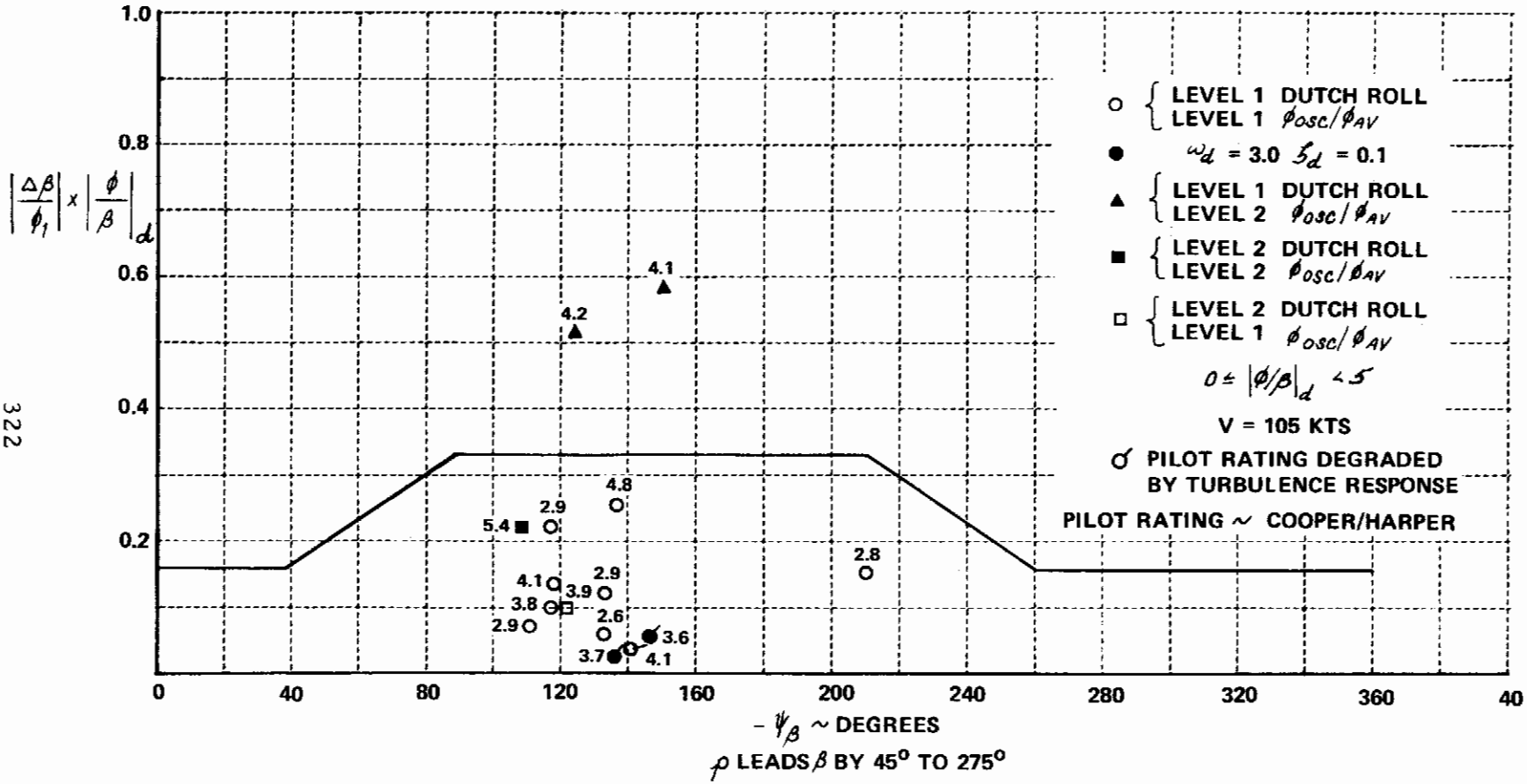

Figure 15 (3.3.8.2) COMPARISON OF SIDESLIP EXCURSION REQUIREMENT WITH PILOT RATING (DATA FROM REFERENCE 101)

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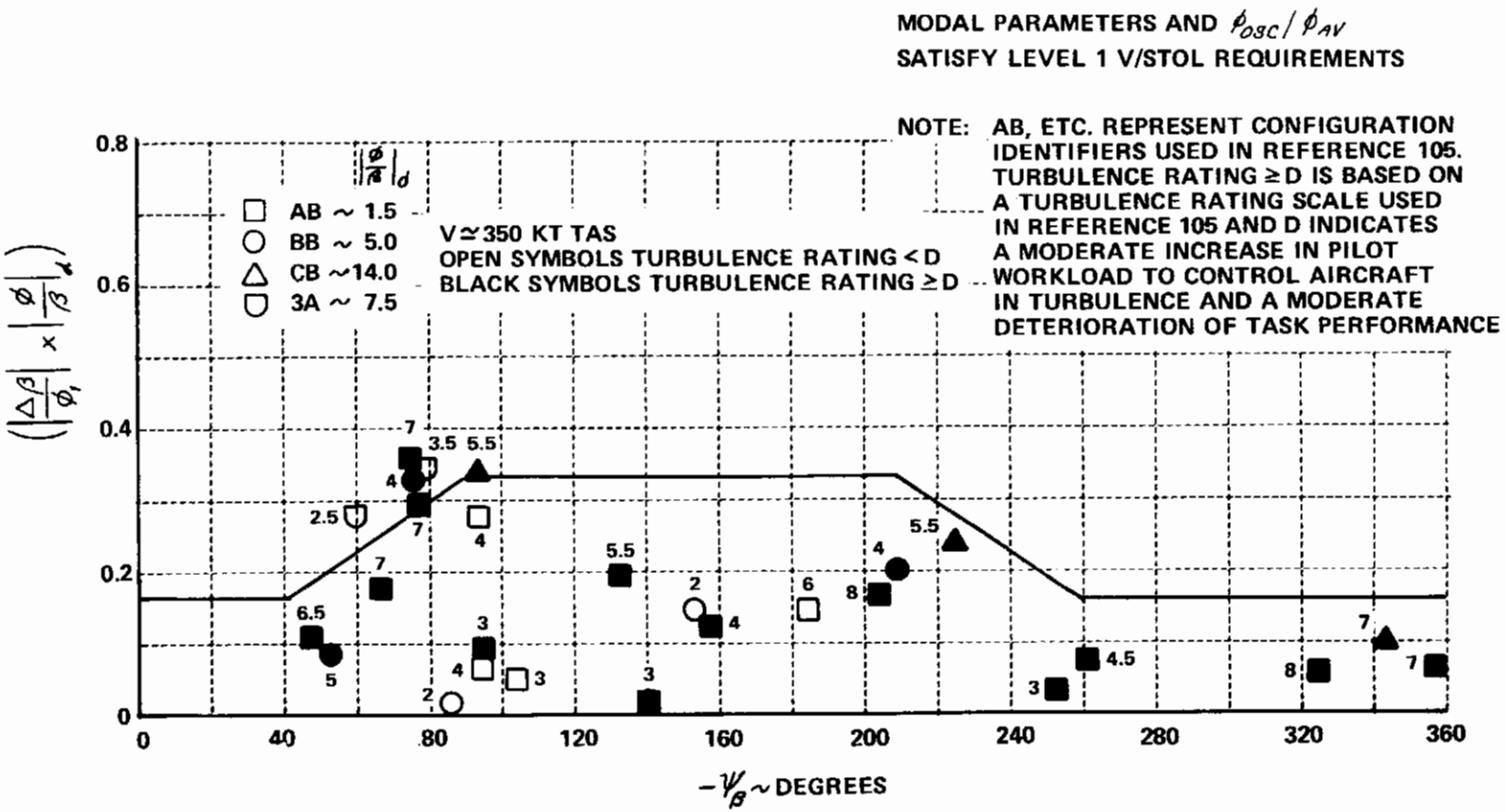

Figure 16 (3.3.8.2) COMPARISON OF SIDESLIP EXCURSION REQUIREMENT WITH PILOT RATING (DATA FROM REFERENCE 105)

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3.3.8.3 CONTROL OF SIDESLIP IN ROLLS

REQUIREMENT

3.3.8.3 Control of sideslip in rolls. In the rolling maneuvers described in 3.3.9, yaw-control effectiveness shall be adequate to maintain zero sideslip with yaw control forces not exceeding those of 3.5.3. This requirement applies to rolling maneuvers of magnitude up to the required roll performance of 3.3.9. For inputs smaller than those required to meet the roll performance requirements of 3.3.9, the resultant forces shall be divided by the ratio of the bank angle obtained at the time specified in 3.3.9 to the bank angle required, and the results compared with the limits of 3.5.3 for compliance.

DISCUSSION

This requirement simply places reasonable limits on yaw-control forces required to maintain zero sideslip when performing prescribed rolling maneuvers. The scaling statement is included to facilitate demonstration of compliance with this requirement . Thus it is required to demonstrate that the yaw control is sufficiently effective to maintain zero sideslip for the Class of aircraft and Level of flying qualities necessary for compliance with roll effectiveness requirements, without excessive yaw-control forces. For smaller roll-control inputs, the yaw-control force for zero sideslip should be at least proportionately less, although linearity is not required.

This requirement is similar to 3.3.2.5 of MIL-F-8785B(ASG) (Reference 10); no similar requirement is noted in AGARD 408A (Reference 46), MIL-H-8501A (Reference 15) or RTM 37 (Reference 47).

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3.3.8.4 TURN COORDINATION

REQUIREMENT

3.3.8.4 Turn coordination. It shall be possible to maintain steady constant-altitude coordinated turns in either direction, using bank angles up to either that required to produce a turn rate of 10 degrees per second or a bank angle of 60 degrees for Class IV aircraft, 45 degrees for Class I and II aircraft, or 30 degrees for Class III aircraft. Yaw control forces shall not exceed 40 pounds and roll control forces shall not exceed 5 pounds. This requirement applies to Level 1, with the aircraft trimmed for zero-bank-angle straight flight.

DISCUSSION

The objective of this requirement is to ensure that only modest yaw control and roll control forces are required for Level 1 operation when performing coordinated turning maneuvers.

The steepness of the turn is a function of aircraft Class to correspond with normal anticipated operational usage. The limiting turn rate specified is based on considerations of normal operation around 35 knots, and is not intended to reflect "ability" maneuvers or nap-of-the-earth maneuver requirements. The relationship between turn radius, turn rate, velocity and bank angle is illustrated on Figure 1(3.3.8.4).

This requirement is essentially the same, as velocity increases, as 3.3.2.6 of MIL-F-8785B(ASG), Reference 10.

It should be noted that both spiral stability and control gradients will have a significant effect on forces required to hold an aircraft in a turn (Reference 97). In addition this requirement can have implications on augmentation system design.

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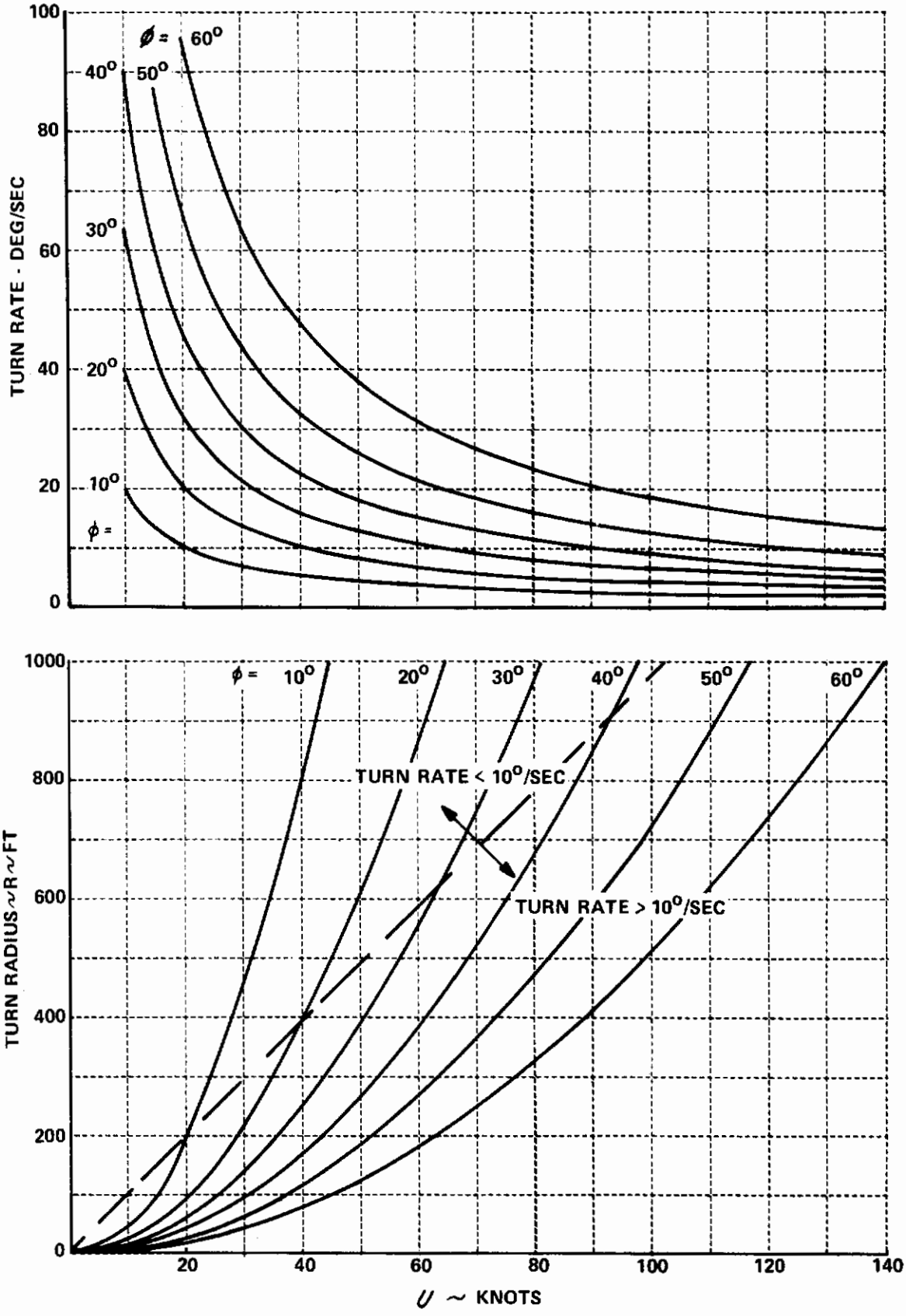

Figure 1 (3.3.8.4) RELATION BETWEEN BANK ANGLE AND TURN RATE IN STEADY TURNS

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3.3.9 ROLL CONTROL EFFECTIVENESS

REQUIREMENT

3.3.9 Roll control effectiveness. The time to change bank angle by 30 degrees (t_{30}) to the right or left from a trimmed zero-roll-rate condition shall not exceed the value specified in table VIII. The time shall be measured from the initiation of roll control force application. Yaw control may be used to reduce sideslip that retards roll rate (not to produce sideslip that augments roll rate), provided that yaw control inputs are simple, easily coordinated with roll control inputs, and are consistent with piloting techniques for the aircraft in its mission. Roll control shall be sufficiently effective, in combination with other normal means of control, to balance the aircraft laterally throughout the Service Flight Envelope in the atmospheric environments of 3.7.

TABLE VIII. Roll Control Effectiveness

Class	t_{30} - seconds		
	Level 1	Level 2	Level 3
I	1.3	1.8	2.6
II	1.8	2.5	3.6
III	2.5	3.2	4.0
IV	1.0	1.3	2.0

DISCUSSION

Roll control effectiveness is a parameter of fundamental importance since it determines the maneuverability of an aircraft in roll. The amount of roll control power desired by a pilot is dependent on several factors. Among these are control power necessary to trim the aircraft laterally, desired maneuvering characteristics and correcting for disturbances resulting from atmospheric environment (i.e., gusts, crosswinds) and operational environment (e.g., operation within ground effect).

The portion of control power required to control gust effects and to trim the aircraft laterally (e.g., trim in presence of crosswinds) is dependent on the turbulence environment, the aircraft dynamics, the augmentation system used, and the sensitivity of the augmentation system to the atmospheric environment. Control power requirements are so dependent on these factors that it has been necessary to write them in a qualitative manner subject to the atmospheric environments of Section 3.7. Reference 106 may serve to

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indicate some design guidance on the relationships between atmospheric turbulence, required control power, and the augmentation system.

Other important factors which influence control power requirements are the aircraft operational mission (e.g., Flight Phase) and the Aircraft Class. In general, for mission effects, it is reasonable to assume that control power requirements for attack missions, rescue missions, etc. which require rapid changes in flight path would be larger than those required for more routine tasks when either smaller or slower changes would be acceptable. Unfortunately there are little data available to establish requirements by Flight Phase Category. Those data which are available for STOL aircraft (e.g., Reference 73) are primarily concerned with landing approach, i.e., Flight Phase Category C.

The quantitative requirements on roll control effectiveness are expressed in terms of the maximum time to change bank angle by 30 degrees (t_{30}) from a trimmed zero-roll-rate condition. Thus they reflect the lateral control power necessary to maneuver the aircraft about trim. The requirements on t_{30} are identical to those appearing in MIL-F-8785B (Reference 10) for Flight Phase Category C, except that for Class II aircraft no distinction is made between land- or carried-based aircraft. As previously mentioned, these requirements may be lenient for Flight Phase Category A; additional guidance on the effects of mission can be found in Reference 84. Section 3.3.4, which supplies the background data for the roll control effectiveness requirements of MIL-F-8785B, Reference 10.

The lateral control power requirements of both MIL-H-8501A (Reference 15) and AGARD 408A (Reference 46) distinguish requirements by aircraft size. This is achieved by using a general relationship between the cube root of aircraft weight and aircraft size, and then placing requirements on lateral control power as a function of the cube root of weight. The lateral control power requirements of RTM-37 (Reference 47) are not specified in terms of aircraft size or the weight law. The authors of Reference 47 express their basic philosophy on size vs. mission in Reference 107, and suggest that mission and not size is a primary parameter.

It should be noted that the cube root of weight attempts to explain the same concept as Class structure. This concept is that the weight, size, inertia, etc. effects should be used to establish requirements when applicable. The shortcoming of just relating control effectiveness to size is that the mission of an aircraft is an important variable and must be considered. The requirements of MIL-H-8501A (Reference 15) and AGARD 408A (Reference 46) allow decreasing bank angle changes with increasing weight, size, etc., which is equivalent to allowing decreasing bank angle changes with Class for a given Level of flying qualities. Since bank angle changes are essentially proportional to time for a step command, the requirements of the V/STOL Specification have selected a specific bank angle change and allow the time to reach this desired bank angle to change with Class of the aircraft, in a manner equivalent to the concepts used to formulate the "weight law".

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Although the recent NRC experiment, Reference 86, was not directed at this requirement, data obtained in this experiment indicates that pilot rating improved with an increase in roll control effectiveness. In particular, for NRC configuration MH100+30+30 ($|\phi/\beta|_d = .75$, $\omega_{n_d} = \omega_\phi =$ 1 rad/sec, $\zeta_d = \zeta_\phi = .3$), as the roll control sensitivity increased from .4 rad/sec^2 - inch to .6 rad/sec^2 - inch, pilot rating improved from 5.5 to a pilot rating of 3, and t_{30} decreased from 2 seconds to 1.35 seconds. These results are reasonably consistent with the values in Table VIII of the V/STOL Specification for Class I and Class II aircraft.

Reference 73 also uses t_{30} as a requirement on roll control effectiveness. This reference points out that t_{30} criteria can normally be easily measured and evaluated, and that the effects of damping and control system lag and dynamics while being included, are not so important as they are in a ϕ_1 (bank angle in one second) criteria which may be strongly affected by control system lags and the shape of the control input. This is illustrated in Figure 1(3.3.9). Although both ϕ_1 and t_{30} are affected by the shape of the control input, the control power required to meet a ϕ_1 requirement is certainly more dependent on the input shape than is t_{30}. Consider the following example: suppose the ϕ_1 requirement was 10° of bank angle change, then if $\tau_R = 1$ sec, $\ddot{\phi}_o$ would be approximately .47 rad/sec^2 for a step, but .8 rad/sec^2 for the indicated ramp input. This example illustrates that, for demonstration of compliance for a lateral control power requirement, it is extremely important to use the correct input, and that the effect on a ϕ_1 requirement is stronger than the effect on a t_{30} requirement. The relationships used to develop these curves are based on a single-degree-of-freedom representation of the rolling equation such that for a step input

$$\frac{\phi}{\ddot{\phi}_o} = 57.3\left[\tau_R t + \tau_R^2\left(e^{-t/\tau_R} - 1\right)\right] \qquad \text{deg/(rad/sec}^2\text{)}$$

while for a ramp input

$$\frac{\phi}{\ddot{\phi}_o} = 57.3\left\{\tau_R t - \tau_R^2 - \frac{t_a \tau_R}{2} - \frac{\tau_R^3}{t_a}\left[e^{-t/\tau_R} - e^{(t_a/\tau_R)-(t/\tau_R)}\right]\right\} \qquad \text{deg/(rad/sec}^2\text{)}$$

Of course, transport lag is included after the response is calculated by these equations.

Tests conducted on various STOL aircraft by NASA indicate that a rapid roll control command was reasonably approximated by a 0.1 second transport lag and a ramp time of 0.3 seconds. These calculations and data obtained are listed in Reference 73 and resulted in recommendations on t_{30} which are illustrated on Figure 2(3.3.9). The NASA data agree reasonably well with the V/STOL requirements on t_{30} for Class II and Class III aircraft.

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Reference 100 examined control power requirements based on a simulation of a carrier landing approach. The characteristics simulated would be related to Class I and Class IV aircraft as defined in the V/STOL Specification. Figure 3(3.3.9) presents data obtained in Reference 100. Superimposed on this plot are the Level 1 and Level 2 roll control effectiveness requirements for Class I and Class IV aircraft and the Level 1 and Level 2 roll mode time constant requirements (3.3.7.2) of the V/STOL Specification. The calculations for t_{30} are based on a single-degree-of-freedom approximation with the 0.2 second ramp input utilized in Reference 100. As can be seen on Figure 3(3.3.9), reasonable agreement exists between the Class I and Class IV roll effectiveness requirements, the roll mode time constant requirements, and the pilot ratings obtained by Princeton, especially for Level 1.

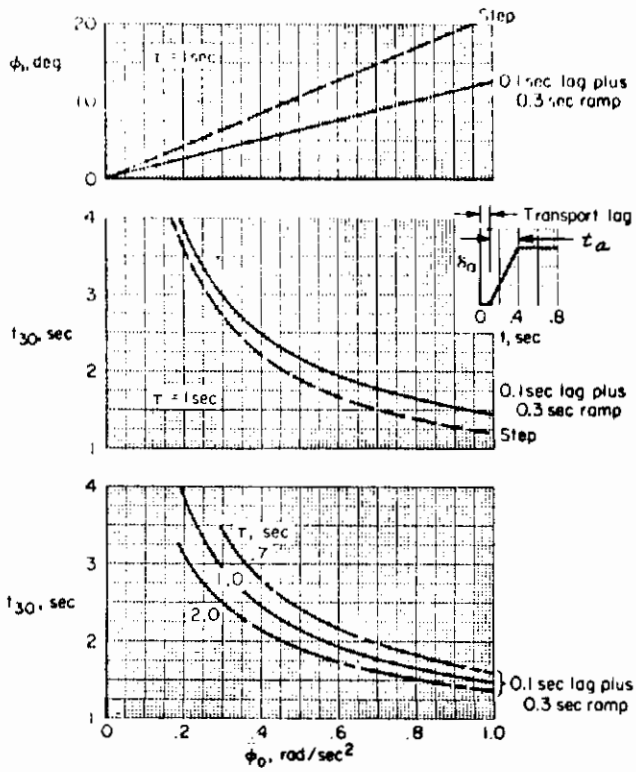

Figure 1 (3.3.9) BANK ANGLE AND TIME TO BANK (FROM REFERENCE 69)

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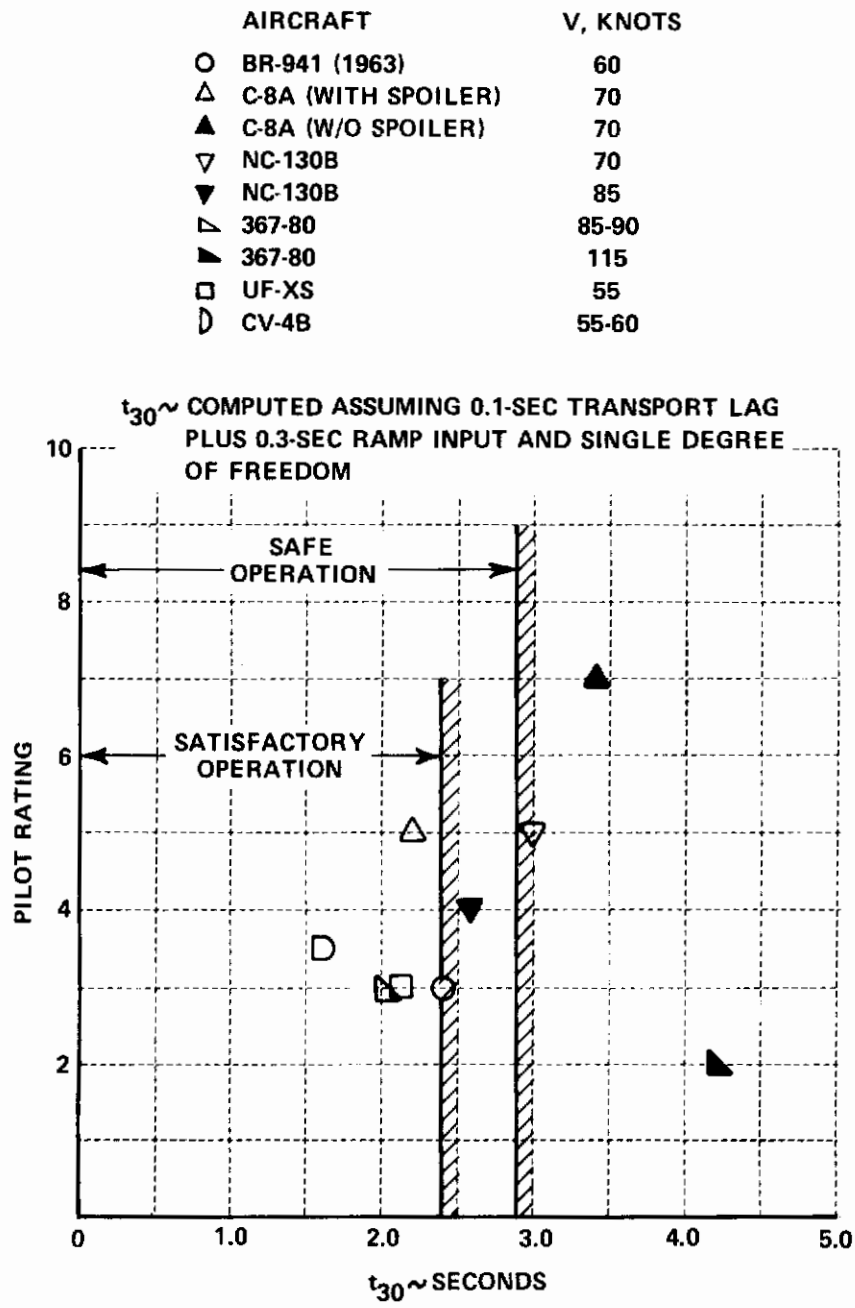

Figure 2 (3.3.9) ROLL CONTROL EFFECTIVENESS OF VARIOUS STOL AIRCRAFT (DATA FROM REFERENCE 73)

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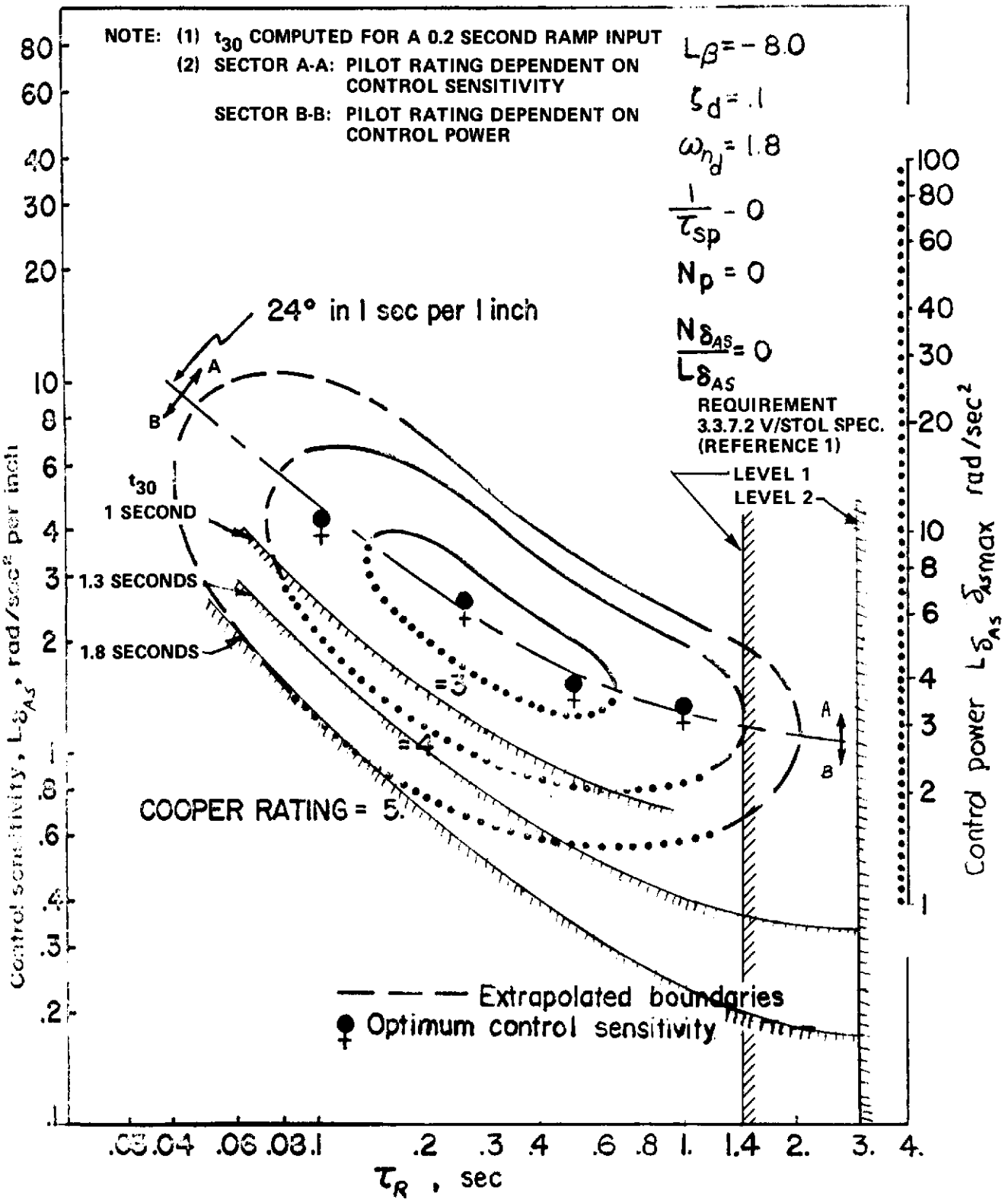

Figure 3(3.3.9) LATERAL CONTROL BOUNDARIES
(DATA FROM REFERENCE 93)

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3.3.9.1 ROLL CONTROL FORCES

REQUIREMENT

3.3.9.1 Roll control forces. The roll control forces required to obtain the rolling performance specified in 3.3.9 shall lie between the maximums and minimums of table IX.

TABLE IX. Roll Control Forces (Pounds)

	Level 1	Level 2	Level 3
Maximum	15.0	20.0	25.0
Minimum	3.3	3.0	0.5

DISCUSSION

The three levels of maximum roll control forces specified here are the same as the limit forces of Table XIII, Section 3.5.3 of the V/STOL Specification for $V > 35$ kt and are thought to be compatible with one-handed operation of the roll control to perform the required bank angle maneuvers of 3.3.9. AGARD 408A (Reference 46) specifies a maximum lateral control force of 20 pounds for both normal operation and after a failure in a power control system for their rolling performance maneuvers. RTM-37 (Reference 47) has similar force requirements to AGARD 408A for the failure condition. NASA in Reference 73 recommends a maximum control force of 20 pounds for satisfactory operation, while allowing 40 pounds of control force for safe operation. Reference 108 recommends either 20 or 25 pounds for peak lateral control force essentially as a function of Flight Phase for a stick controller. MIL-F-8785B(ASG), Reference 10, has a matrix of requirements on aileron control force as a function of Class, Flight Phase Category and type of controller ranging from 20 through 35pounds for a stick controller and from 20 to 70 pounds for a wheel controller. Since the philosophy of the V/STOL Specification has assumed one-handed operation, both stick and wheel forces would be the same.

The minimum forces specified are intended to guard against oversensitive roll control. In place of specifying sensitivity requirements, the decision was made to specify minimum control force based on the results indicated in Reference 105. It appears that as long as stick displacements are not uncomfortably large, the pilot is more aware of control forces than control displacements. The values specified essentially follow the minimum control force relationship specified in Section 3.3.4.2 of MIL-F-8785B(ASG) (Reference 10). The value specified for Level 1 minimum control force was also based on scaling the maximum force by the ratio of the allowable control force gradients of Table XII, Section 3.5.1.2 of the V/STOL Specification.

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3.3.9.2 LINEARITY OF ROLL RESPONSE

REQUIREMENT

3.3.9.2 Linearity of roll response. There shall be no objectionable nonlinearities in the variation of roll response with roll control deflection or force.

DISCUSSION

This requirement is directed at control precision and is essentially the same as 3.3.4.3 of MIL-F-8785B(ASG), Reference 10. The requirement is qualitative since it has not been possible to specify values on objectionable nonlinearities.

Reference 73 requires that for satisfactory operation, roll acceleration per unit stick deflection should not increase. As noted in Reference 73, when there is an abrupt increase in roll acceleration per unit stick deflection as the control deflection is increased, there exists a tendency for the pilot to overcontrol the aircraft in roll; this tendency was not noted for nonlinearities that reduced the roll acceleration per unit stick deflection. Thus the pilot's objection to certain nonlinearities appears to be related to his ability to make precise roll control inputs to control the flight path. An abrupt increase in roll response per unit control deflection could tend to result in a situation where the pilot would have to continually correct his input to achieve the desired response. For example, if the pilot desired to increase his bank angle, the input, if it went into the nonlinear portion of the relationship between control power and control displacement, would tend to give the pilot more response than desired. To correct for this increase in the response, he would have to correct his input. The degree of nonlinearity could result in a situation whereby the pilot would have to repeat this process several times to achieve the desired level of response. This would, of course, increase pilot workload and possibly detract from mission effectiveness. On the other hand, if the sensitivity decreases as the control displacement increases, only relatively minor corrections in control displacement would be required to achieve the desired change in bank angle for either maneuvers or when compensating for gust disturbances.

The extent of nonlinearity which can usefully be allowed will no doubt depend on the aircraft lateral dynamics, the aircraft Class, and Flight Phase Category (e.g., mission). At this time, insufficient data are available to place quantitative requirements on nonlinearities of aircraft response.

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3.3.9.3 WHEEL CONTROL THROW

REQUIREMENT

3.3.9.3 Wheel control throw. For all aircraft with wheel controllers, the wheel throw necessary to meet the roll performance requirements specified in 3.3.9 shall not exceed 60 degrees in either direction. For completely mechanical systems, the requirements may be relaxed to 80 degrees.

DISCUSSION

This requirement is the same as 3.3.4.4 of MIL-F-8785B(ASG) (Reference 10). AGARD 408A (Reference 46) requires that wheel throw should not exceed 60 degrees for normal operation and, in addition, for a failure in a power control system. Reference 73 recommends no more than 60° wheel throw for satisfactory operation, and no more than 90° wheel throw for safe operation. The value of 60° for wheel throw is consistent with one-handed operation.

A wheel throw of 80 degrees for completely mechanical systems has been specified in deference to the aircraft designer.

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3.3.9.4 YAW CONTROL INDUCED ROLLS

REQUIREMENT

3.3.9.4 Yaw-control-induced rolls. For Levels 1 and 2 the application of right yaw control force shall not result in left rolls and the application of left yaw control force shall not result in right rolls.

DISCUSSION

This requirement is a restatement of 3.3.4.5 of MIL-F-8785B(ASG) which requires the ability to "raise the wing with the rudder." The desirability or necessity to have this capability at low speeds is not clear cut. Princeton University data (References 88, 100, and 101) indicate that with satisfactory lateral-directional modal characteristics, L_β can vary significantly without any adverse effect on pilot rating in the landing approach task [Figure 1(3.3.9.4)]. In general, it appears that if the aircraft can roll through application of the yaw control, the sense should be in the proper direction (i.e., the aircraft can have zero effective dihedral, but should not have negative effective dihedral). This requirement does direct attention to this capability, although it does not ensure stable dihedral since it would be possible to use a rudder/aileron interconnect ($L_{\delta_{RP}}$) to accomplish the intent of the requirement.

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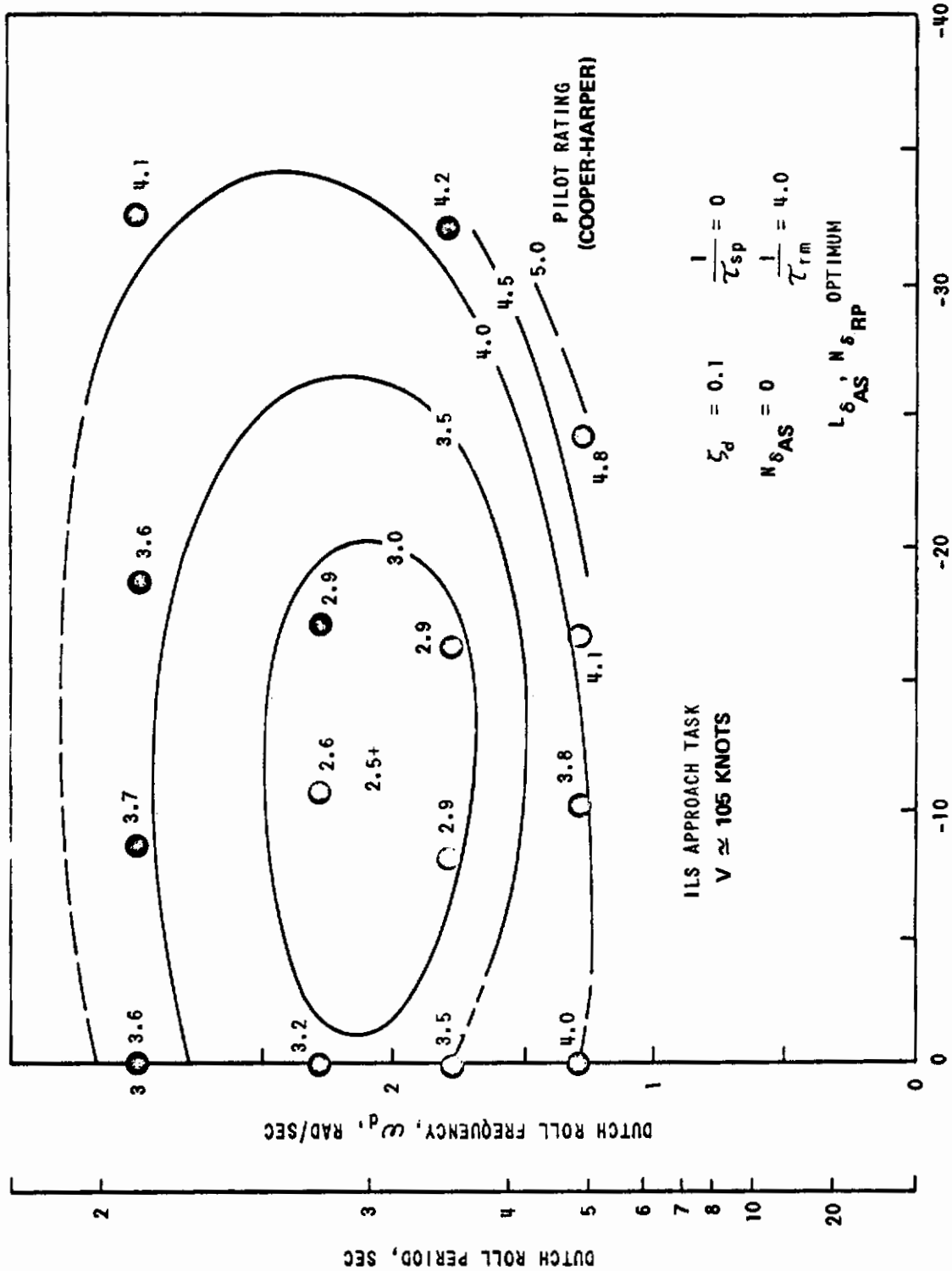

Figure 1(3.3.9.4) PILOT RATING CONTOURS, ω_d VS L_β (REFERENCE 101)

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3.3.10 DIRECTIONAL CONTROL EFFECTIVENESS

REQUIREMENT

3.3.10 Directional control effectiveness. Yaw control shall be sufficiently effective, in combination with other normal means of control, to balance the aircraft directionally throughout the Service Flight Envelope in the atmospheric environments of 3.7.

DISCUSSION

This is a general catch-all requirement on balancing yawing moments, and is similar to requirement 3.3.5 of MIL-F-8785B(ASG), Reference 10; 3.15 and 3.16 of AGARD 408A, Reference 46; and similar requirements of RTM-37, Reference 47. The requirements of AGARD 408 and RTM-37 impose margins on aircraft. Perhaps the most critical combination of conditions for V/STOL aircraft to establish directional control effectiveness requirements is the ability to maintain a desired heading in the presence of a crosswind with an asymmetric thrust condition in a landing approach task. The general phrasing of the requirement to balance the aircraft in the Service Flight Envelope in the various atmospheric environments to be specified will determine critical conditions and the directional control effectiveness required.

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3.3.10.1 DIRECTIONAL RESPONSE TO YAW CONTROL INPUT

REQUIREMENT

3.3.10.1 Directional response to yaw control input. The yaw attitude change within the first second following an abrupt step displacement of the yaw control shall not be less than:

Level 1:	6.0 degrees
Level 2:	3.0 degrees
Level 3:	1.0 degree

This requirement applies with all other cockpit controls fixed.

DISCUSSION

Yaw control power is required to develop steady state sideslips during maneuvers such as crosswind landing approaches, for changing heading during maneuvers such as decrabbing prior to landing, and for reducing unwanted sideslip angles in maneuvering flight. Sufficent yaw control power to develop steady sideslips is demanded by the requirements of paragraph 3.3.11 of Reference 1. At low speeds, the aircraft is required to develop at least 15 degrees of steady state sideslip. Since directional stiffness, N_β, generally varies linearly with dynamic pressure, the yaw control power required for compliance with 3.3.11 varies directly as speed squared. If 3.3.11 were the only yaw control power requirement, the ability to change heading during low speed flight might be severely limited. Thus, the intent of this requirement (3.3.10.1) is to assure that yaw control power exists in low speed flight where sideslip requirements may not assure adequate maneuverability.

The statement of this requirement parallels that of the corresponding hover and low speed requirement (paragraph 3.2.3.1). However, unlike hover and low speed, the initial trim conditions for demonstration of compliance are not specified.

Based on flight test experiments with a group of STOL aircraft, NASA has proposed a somewhat similar requirement for maneuvering control power (Reference 73). Their requirement demands sufficient control power to change heading by 15 degrees within 2.2 seconds for satisfactory operation and within 3.1 seconds for safe operation. The NASA requirement was derived based on a control input having the form of a 0.1 second transport time lag followed by a 0.3 second ramp. Assuming a configuration whose short term yaw attitude response is described by $\psi/\delta_{RP}(s) = N_{\delta_{RP}}/s(s-N_r)$ the NASA requirement is compared to the V/STOL Specification requirements in Figure 1 (3.3.10.1).

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At low damping levels ($N_r \rightarrow 0$) the V/STOL Specification, Level 1, is seen to be more demanding than the NASA satisfactory criterion. At damping levels of about 2.3 sec^{-1} and greater, however, the NASA criterion is more demanding. The relationship of the NASA safe boundary to the specification Level 2 or 3 boundaries is more difficult to assess. Figure 1(3.3.10.1) indicates that at low damping the safe boundary lies between Levels 2 and 3 while at higher damping it lies between Levels 1 and 2.

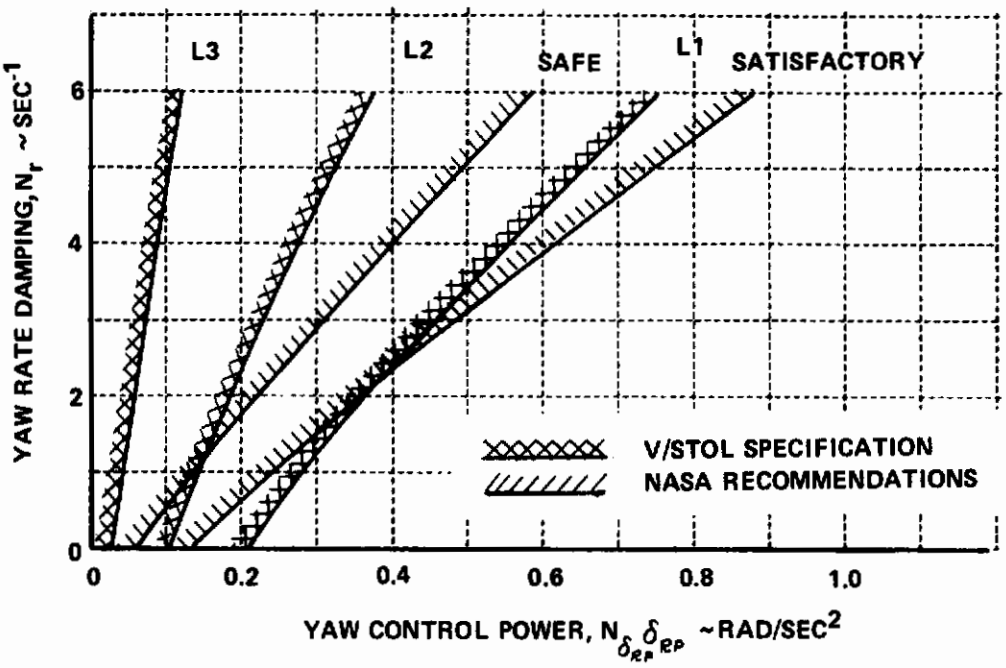

Figure 1 (3.3.10.1) COMPARISON OF V/STOL SPECIFICATION CONTROL POWER REQUIREMENTS WITH NASA RECOMMENDATIONS

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3.3.10.2 LINEARITY OF DIRECTIONAL RESPONSE

REQUIREMENT

3.3.10.2 Linearity of directional response. There shall be no objectionable nonlinearities in the variation of directional response with yaw control deflection or forces. Excessive sensitivity or sluggishness in response to small yaw control deflections or forces shall be avoided.

DISCUSSION

This requirement is essentially the counterpart of the linearity requirement on roll response in the V/STOL Specification. Again, the requirement is qualitative since sufficient data are not available to set limits on nonlinearities. Avoiding excessive sensitivity or sluggishness for small control inputs will tend to minimize either pilot workload or tendency to overcontrol when the pilot is required to make small heading corrections.

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3.3.10.3 DIRECTIONAL CONTROL WITH SPEED CHANGE

3.3.10.3.1 DIRECTIONAL CONTROL WITH ASYMMETRIC LOADING

REQUIREMENT

3.3.10.3 Directional control with speed change. With the aircraft initially trimmed directionally with symmetric power, it shall be possible to maintain zero-bank-angle straight flight over a speed range of ± 30 percent of the trim speed or ± 20 knots whichever is less (except where limited by the boundaries of the Service Flight Envelope). The yaw control forces shall not be greater than those of table XIII for propeller-driven aircraft and not greater than one-half those of table XIII for other aircraft. These requirements must be satisfied without retrimming.

3.3.10.3.1 Directional control with asymmetric loading. With the aircraft initially trimmed directionally with any asymmetric loading specified in the contract 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 forces not exceeding those of table XIII, without retrimming.

DISCUSSION

These requirements are based on requirements 3.3.5.1 and 3.3.5.1.1 of MIL-F-8785B(ASG), Reference 10. They place quantitative limits on yaw control forces for the specific cases of speed changes and maintaining straight flight in the Operational Flight Envelope with an asymmetric loading. The speed change requirements have been made in response to comments from industry and governmental agencies to reflect the type of propulsion system, to reflect the effects of either the presence or lack of propeller slipstream effects on yaw control forces.

The purpose of the asymmetric loading requirement is an attempt to keep control trim forces within limits during the Flight Phases associated with the Operational Flight Envelope.

Both 3.3.10.3 and 3.3.10.3.1 could be discussed in a fair amount of detail using some basic equations. However, in order to do so it would be necessary to first formulate the proper mathematical model. In particular, some of the basic assumptions usually made in the process of deriving the usual linearized equations of motion would have to be re-examined. For example, if an aircraft had an aerodynamic asymmetry caused by propeller slipstream effects, it would no longer be valid to assume that derivatives such as, say, N_u are equal to zero (N_u is yawing moment due to forward speed change). The presence of this derivative couples the longitudinal and lateral-directional equations of motion. Also, if there is inertial asymmetry (as distinct from aerodynamic asymmetry), c.g. locations will not be in the plane of geometric symmetry as is usually assumed. Thus, additional inertial terms will appear in the equations.

The above comments have been made to point out that a reasonably adequate analytical treatment of the flying qualities aspects toward which

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3.3.10.3 and 3.3.10.3.1 are directed would probably require some time-consuming preparatory mathematical model development. The fact that full analytical treatments have not appeared in the general literature indicates that the associated flying qualities aspects have not been extremely acute. The requirements, however, recognize the importance of these flying qualities aspects and should help to prevent acute problems from developing.

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3.3.11 LATERAL-DIRECTIONAL CHARACTERISTICS IN STEADY SIDESLIPS

REQUIREMENT

3.3.11 Lateral-directional characteristics in steady sideslips. The requirements of 3.3.11.1 through 3.3.11.3.1 are expressed in terms of characteristics in yaw-control-induced steady zero-yaw-rate sideslips with the aircraft trimmed for zero-bank-angle straight flight. Sideslip angles to be demonstrated shall be the lesser of 25 degrees or $\sin^{-1}$ (30/airspeed in knots), or those limited by structural limitations, or the yaw control and roll control force limits of table XIII. In any event, the minimum sideslip to be demonstrated shall be the lesser of 15 degrees or $\sin^{-1}$ (30/airspeed in knots).

DISCUSSION

The purpose of this requirement is to establish the limiting sideslip angle to be considered in determining the compliance of an aircraft to the requirements of 3.3.11.1 through 3.3.11.3.1. It must be possible to reach one of these limits to meet the requirements.

This requirement is related to 3.3.6 of MIL-F-8785B, Reference 10; 3.1 of AGARD 408A, Reference 46; and 3.7.4.2 of RTM-37, Reference 47. However, the qualitative requirements of AGARD 408A on sideslip angles required in normal tactical employment are now quantitatively specified in the V/STOL Specification.

The sideslip angle related to a 30 knot, 90 degree crosswind condition is based on considerations of the operational environment and is related to the requirement of 3.3.7 of MIL-F-8785B(ASG), although Reference 73 indicates that existing STOL aircraft would be limited by a 25 knot crosswind condition. The value of sideslip equal to 25 degrees is intended to relax the 30 knot crosswind requirement for STOL aircraft at speeds below approximately 70 knots and is based on the capability of the BR-941 in a wing-down approach as indicated in Reference 73. The additional conditions that determine what sideslip angles are to be considered reflect practical design and operational considerations for limiting sideslip capability. Sideslip angles corresponding to control force limits are considered operationally possible to achieve. However, these angles must not be so low as to detract from the capability to generate reasonable sideslip angles.

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3.3.11.1 YAWING MOMENTS IN STEADY SIDESLIPS

REQUIREMENT

3.3.11.1 Yawing moments in steady sideslips. For the sideslips specified in 3.3.11, right yaw control deflection and force shall be required in left sideslips and left yaw control force and deflection shall be required in right sideslips.

For Levels 1 and 2, the following requirements apply. The variation of sideslip angle with yaw control deflection and force shall be essentially linear for sideslip angles between + 15 degrees and - 15 degrees. For larger sideslip angles, an increase in yaw control deflection shall always be required for an increase in sideslip and, although a reduction of yaw control force gradient is acceptable outside this range, the following requirements shall apply:

Level 1: The gradient of sideslip angle with yaw control force shall not reverse slope

Level 2: The gradient of sideslip angle with yaw control force is permitted to reverse slope provided the sign of the yaw control force does not reverse.

The term gradient does not include that portion of the yaw control force versus sideslip-angle curve within the preloaded breakout force or friction band.

DISCUSSION

This requirement is related to 3.3.6.1 of MIL-F-8785B(ASG), Reference 10; 3.3 and 3.4 of AGARD 408A, Reference 46; 3.7.4.3 of RTM-37, Reference 47; and 3.3.9 of MIL-H-8501A, Reference 15. The intent of the requirement is to describe the static directional characteristics of the aircraft in terms normally considered by the pilot (i.e., the variation of cockpit control deflection and force with sideslip angle). Linearity of control deflection and force is desirable as a pilot cue in controlling sideslip within the range of sideslip angles specified. Relaxation on gradient requirements beyond ± 15 degrees of sideslip angle is permitted based on practical considerations of V/STOL aircraft design. Reference 73 places similar requirements on the linearity of sideslip angle with control deflection. Because of possible control cross-coupling, meeting this requirement alone will not necessarily assure static directional stability; however, the requirements of 3.3.8.1 of the V/STOL Specification are also directed at static directional stability.

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3.3.11.2 BANK ANGLE IN STEADY SIDESLIPS

REQUIREMENT

3.3.11.2 Bank angle in steady sideslips. For the sideslips specified in 3.3.11, an increase in right bank angle shall accompany an increase in right sideslip, and an increase in left bank angle shall accompany an increase in left sideslip.

DISCUSSION

This requirement is directed at the desirability of providing a proper sense of bank angle variation with sideslip angle as a possible cue to the pilot to indicate the sense and magnitude of the sideslip angle. The requirement is related to 3.3.6.2 of MIL-F-8785B(ASG), Reference 10, and 3.7 of AGARD 408A, Reference 46. It should be noted that satisfying the requirement has implication on Y_β, however, as shown in Reference 76 the relationship between ϕ and β in a steady sideslip is also dependent upon L'_β and control derivatives. It is possible, though relatively unlikely, to design an aircraft that will not meet this requirement.

There is some evidence to indicate that zero bank in straight sideslips is not objectionable. However, in general, opposite bank seems to be undesirable. It should be noted that for many V/STOL configurations at low speed, the gradient of ϕ to β is normally quite small, although in the proper sense, and it would be difficult for the pilot to sense sideslip due to the lack of side forces related to ϕ. Reference 73 examined the gradient of ϕ to β in steady sideslips for several STOL aircraft. In general, for these aircraft, although the gradient was in the proper sense, little side force due to bank angle was noted. This was not considered to be a problem by the pilots.

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3.3.11.3 ROLLING MOMENTS IN STEADY SIDESLIPS

REQUIREMENTS

3.3.11.3 Rolling moments in steady sideslips. For the sideslips specified in 3.3.11, left roll control deflection and force shall be required in left sideslips, and right roll control deflection and force shall be required in right sideslips. For Levels 1 and 2, the variation of roll control deflection and force with sideslip angle shall be essentially linear.

DISCUSSION

This requirement is directed at effective positive dihedral, although it would be possible to meet such a requirement by use of control cross-coupling. The requirement is similar to 3.3.6.3 of MIL-F-8785B(ASG), Reference 10; 3.3.9 of MIL-H-8501A, Reference 15; 3.5 and 3.6 of AGARD 408A, Reference 46; and 3.7.4.3 of RTM-37, Reference 47. While these documents generally agree that positive dihedral effect is desirable, Princeton University data (References 88, 100 and 101) indicate that with proper vehicle modal parameters the effect of positive dihedral on pilot rating is insignificant [see Figure 1(3.3.9.4)]. Data in References 73 and 89 indicate that the pilot's desired level of effective dihedral is related to such problems as low directional damping, turn coordination, turbulence response, and spiral instability. In general, the designer must select L'_β based on a trade-off of spiral stability and Dutch roll mode stability. Increasing positive effective dihedral increases Dutch roll mode frequency and reduces Dutch roll mode damping but tends to stabilize the spiral mode, while the opposite occurs for negative effective dihedral ($L'_\beta > 0$). The effect this can have on pilot rating is illustrated on Figure 1(3.3.11.3) for a simulator study discussed in Reference 110.

Reference 111 examined the effect of dihedral on V/STOL directional handling qualities. The results are described in terms of directional rate damping (N_r), and directional control sensitivity ($N_{\delta_{RP}}$), see Figure 2(3.3.11.3). The results of this investigation indicate that dihedral had no significant influence on pilot rating when the combinations of N_r and $N_{\delta_{RP}}$ would result in an aircraft that is acceptable only for emergency operation. For combinations of N_r and $N_{\delta_{RP}}$ that resulted in a pilot rating of 3.5, increased amounts of both damping and control sensitivity were required for slight increases in positive effective dihedral. For relatively large values of positive dihedral, an additional increase in the dihedral effect had a relatively insignificant effect on damping, control sensitivity, and pilot rating. Similar results on the effect of positive dihedral are discussed in Reference 87. Figure 3(3.3.11.3) illustrates the data presented in Reference 73 on the influence of dihedral on pilot rating.

The requirement on linearity of roll control deflection and force with sideslip angle is imposed to provide a consistent cue to the pilot of the magnitude of sideslip as an aid to precision control of the flight path.

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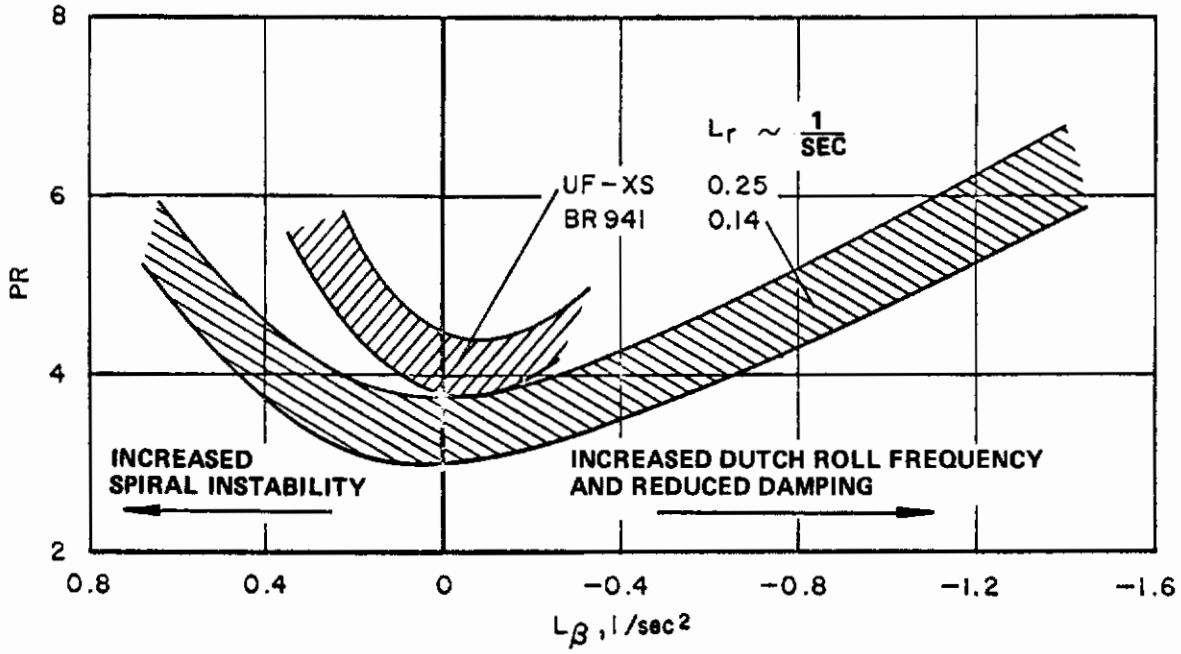

Figure 1(3.3.11.3) EFFECT OF DIHEDRAL ON PILOT OPINION; ASE OFF, V = 50 TO 60 KNOTS, N_β = 0.5 RAD/SEC^2, N_r = 0.3 SEC^{-1} (FROM REFERENCE 110)

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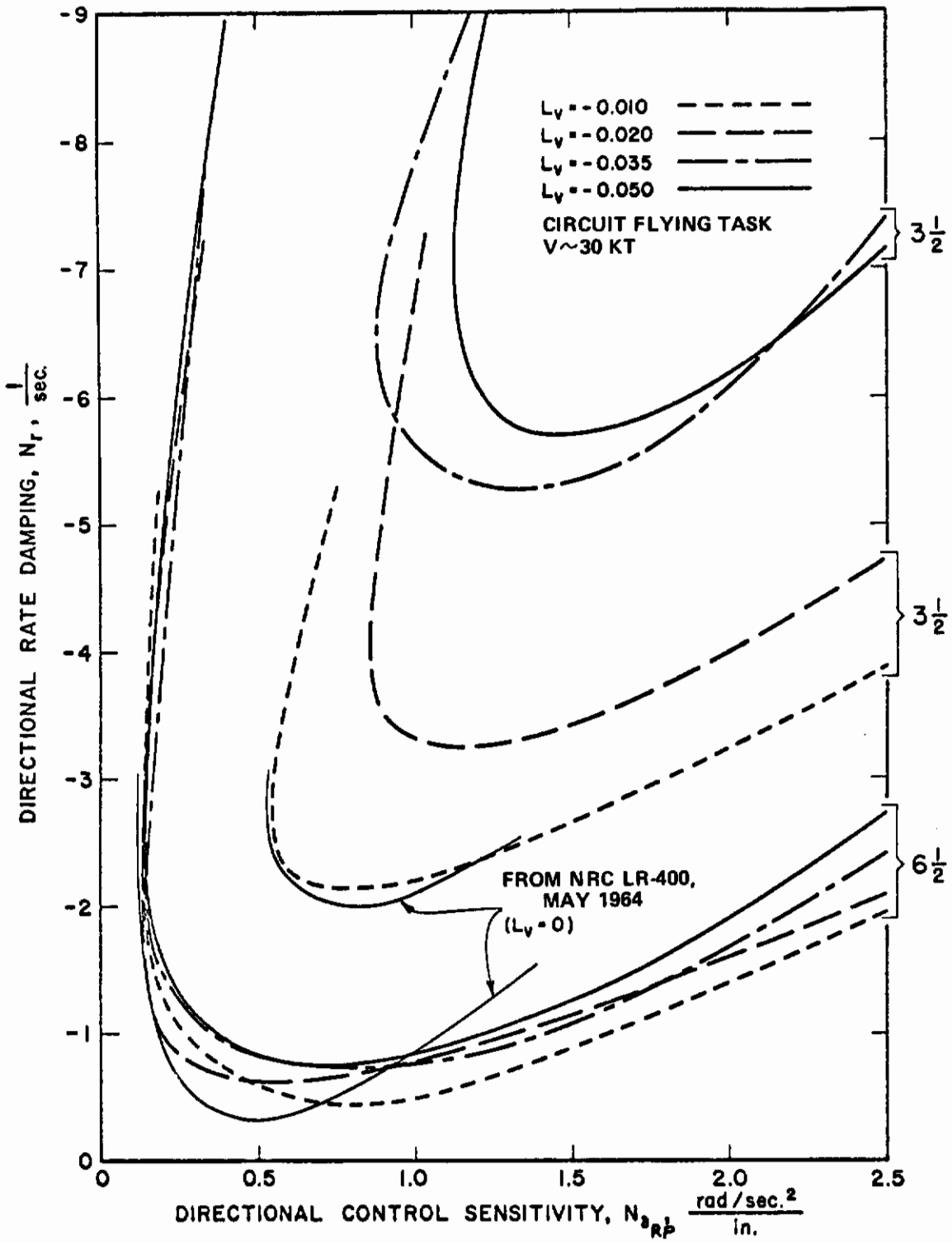

Figure 2 (3.3.11.3) COMPARISON OF HANDLING QUALITIES BOUNDARIES FOR VARIOUS VALUES OF DIHEDRAL EFFECT (FROM REFERENCE 111)

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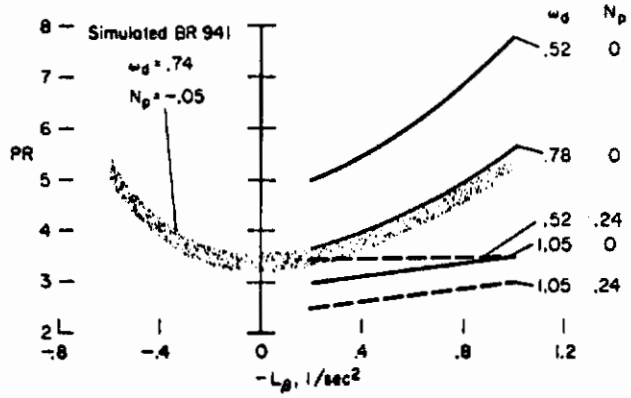

Figure 3(3.3.11.3) EFFECT OF DIHEDRAL ON PILOT RATING (DATA FROM REFERENCE 73)

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3.3.11.3.1 POSITIVE EFFECTIVE DIHEDRAL LIMIT

REQUIREMENT

3.3.11.3.1 Positive effective dihedral limit. For Level 1, positive effective dihedral (right roll control for right sideslip and left roll control for left sideslip) shall never be so great that more than 50 percent of the roll control power available to the pilot and no more than 7.5 pounds of roll control force are required for sideslip angles which might be experienced in service employment. The corresponding limits for Level 2 shall be 75 percent and 10 pounds.

DISCUSSION

The intent of this requirement is to limit the positive effective dihedral in terms of roll control power and roll control forces required in steady sideslips. This requirement is similar to 3.3.6.3.2 of MIL-F-8785B(ASG), Reference 10; 3.5 and 3.6 of AGARD 408A, Reference 46, and 3.7.4.3 of RTM-37, Reference 47.

It is necessary to restrict the amount of effective positive dihedral based on considerations of turbulence response and cross-wind flight where large effective dihedral will tend to use up the available lateral control power. Figure 3(3.3.11.3) indicates that the pilot's acceptance of large positive dihedral effect is also dependent upon the vehicle dynamics, and turn coordination characteristics, however, in general, pilot rating worsens with increased effective dihedral.

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3.4 TRANSITION

REQUIREMENT

3.4 Transition. The transition requirements are applicable to the accelerating or decelerating transition maneuver itself, and not to the maneuvering capability when operating about a fixed operating point defined by some trim speed lying in the range between hover and V_{con}. For operation around such fixed operating points, the requirements of 3.2 and 3.3 shall apply. Compliance shall be demonstrated when performing transition profiles as defined by the mission requirements. The transition maneuver requirements shall be met for all applicable Aircraft States except Aircraft Special Failure States.

DISCUSSION

This general statement defines the conditions for which the requirements of 3.4 apply. A discussion of how the transition requirements (3.4) fit into the specification structure as a whole, is given in Section III. Also given in Section III is an explanation of what is meant by the term transition, and some background philosophy on the transition requirements.

It has been pointed out in the discussion of paragraph 3.1 that the definitions of Flight Envelopes are concepts which do not apply during transition. The question, then, is how to treat transition. An attempt will be made to clarify the desired intent as the specific requirements are discussed. Overall it should be remembered that the concept of Levels does apply. There are transition Flight Phases; during such transitions, Level 1 flying qualities are required. There are no quantitative requirements in 3.4 with a breakdown for Levels, hence the Level concept will be applied in its qualitative sense defined in paragraph 1.5, e.g., Level 1: flying qualities are clearly adequate for the mission Flight Phase. This is required even though for a particular aircraft the transition may pass through a region of the speed, altitude and load factor space in a configuration which would, for fixed operating point operation, be considered outside the Operational Flight Envelope (or even outside the Permissible Flight Envelope).

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3.4.1 ACCELERATION-DECELERATION CHARACTERISTICS

REQUIREMENT

3.4.1 Acceleration-deceleration characteristics. From every possible fixed operating point at speeds below V_{con}, with the aircraft trimmed at the operating point, it shall be possible to accelerate rapidly and safely to V_{con} at approximately constant altitude and also on any other flight path as required by the operational missions of 3.1.1. From trimmed steady, level, unaccelerated flight at V_{con}, it shall be possible to decelerate rapidly and safely, at approximately constant altitude and also on any other flight path as required by the mission, to all fixed operating points at speeds below V_{con}. The time taken for these maneuvers and the altitudes flown shall be those designated by the mission requirements. It shall be possible to execute these maneuvers without restriction due to factors such as pitch, roll, or yaw control power, pitch trim, stalling or buffeting, or thrust response characteristics. All controls required to effect a transition shall be easily operated by one pilot.

DISCUSSION

An aircraft which has an Operational Flight Phase in the speed range V to V_{con} should be able to accelerate to V_{con} (and above) as required by the operational missions. Also, it should be able to decelerate from any appropriate fixed operating point (FOP) in the Service Flight Envelope in the speed range below V_{con}. This concept is illustrated in Figure 1(3.4.1). The phrase "every possible fixed operating point" in paragraph 3.4.1 has been restricted to points in the Service Flight Envelope since it is considered that the following requirement (3.4.2) adequately covers points in the Permissible Flight Envelope.

Transitions between points in the Operational Envelope corresponding to Flight Phase A and points in the Operational Flight Envelope corresponding to Flight Phase B, if designated as a required Flight Phase (e.g., TT or NT), should be flyable with Level 1 flying qualities in the sense defined in paragraph 1.5.

The intent of this requirement is to ensure that the transition maneuvers dictated by operational requirements can be performed safely and quickly without excessive pilot attention to aircraft attitude, airspeed, trim or other factors which would reduce the ability to control the aircraft about a chosen flight path. No requirements have been imposed on the time required to perform the selected transition maneuver since this parameter is dependent upon the mission of the aircraft. However, the rapidity of the transition should not be limited by aircraft controllability about any axis.

It is desirable that the pilot need only operate primary flight controls, power setting and/or thrust angle controls throughout the transition. Inadvertent operation of automatic devices operated during transition should be avoided and the performance of such devices should be easily monitored by

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the pilot. To minimize the pilot workload it will be necessary to ensure that the phasing of controls necessary to perform the transition is simple and that a large number of separate tasks are avoided so that the pilot is not oversaturated by the information necessary to accomplish the transition maneuver and control the flight path of the aircraft.

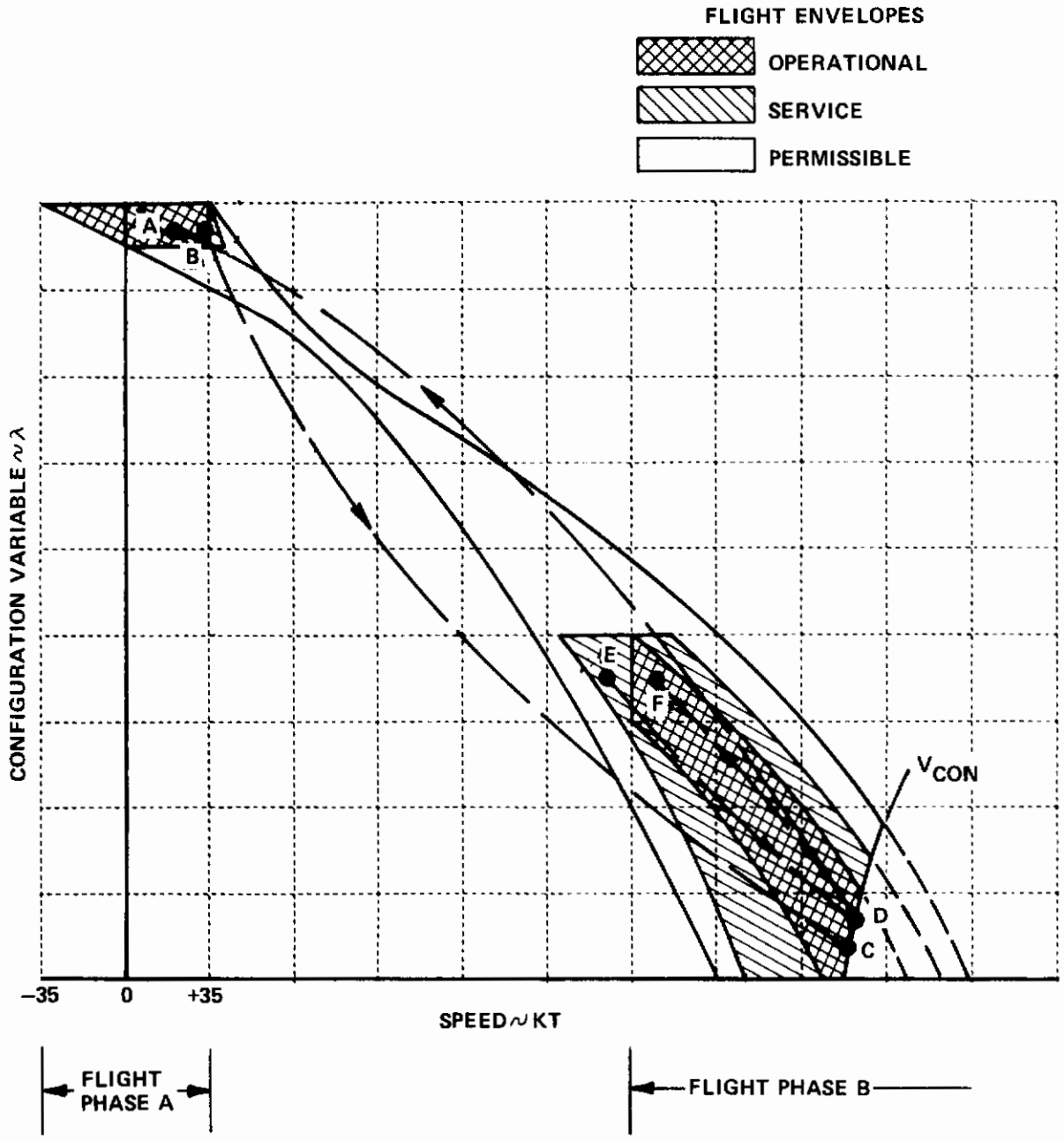

Figure 1(3.4.1) EXAMPLES OF TRANSITIONS FOR WHICH 3.4.1 APPLIES (B→C; E→D, D→F, D→A)

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3.4.2 FLEXIBILITY OF OPERATION

REQUIREMENT

3.4.2 Flexibility of operation. At any time during a transition it shall be possible for the pilot to quickly and safely stop the transition maneuver and reverse its direction.

DISCUSSION

The purpose of this requirement is to provide the pilot with the option of changing his mind; it should not be mandatory for a transition to be completed once it has been initiated. The concept required is illustrated by Figure 1(3.4.2). An accelerating transition A to E should be capable of being aborted at any point. For example, if it is aborted at B, the accelerations should diminish to zero when the fixed operating point C is reached. Point C should be within the Permissible Flight Envelope. From C it should be possible to return back into the Operational Flight Envelope corresponding to Flight Phase A (point D). It should also be possible to reverse the sense of the transition at point B (rather than stopping the transition), and return directly to D. A similar argument holds for decelerating transitions which should be capable of being aborted at F with the option of returning directly to H or holding at G and then returning to H.

The need for this requirement is based both on possible operational usage and for flight safety. From operational or tactical considerations it may be desirable to stop the transition or reverse the transition. From safety considerations, the transition could involve automatic devices which in the event of a malfunction could result in dangerous and unrecoverable aircraft attitudes unless the pilot can stop the transition and either operate about a fixed operating point or reverse the transition. The requirement that any stopped transition results in fixed operating point flight within the Permissible Flight Envelope means that Level 3 flying qualities will be available and that the aircraft can be landed safely.

Reference 112 indicates that in the P.1127 V/STOL, the pilot's ability to stop and/or reverse the transition maneuver at any point is a feature which significantly contributed to the pilot's comfort and confidence during the performance of the transition maneuver.

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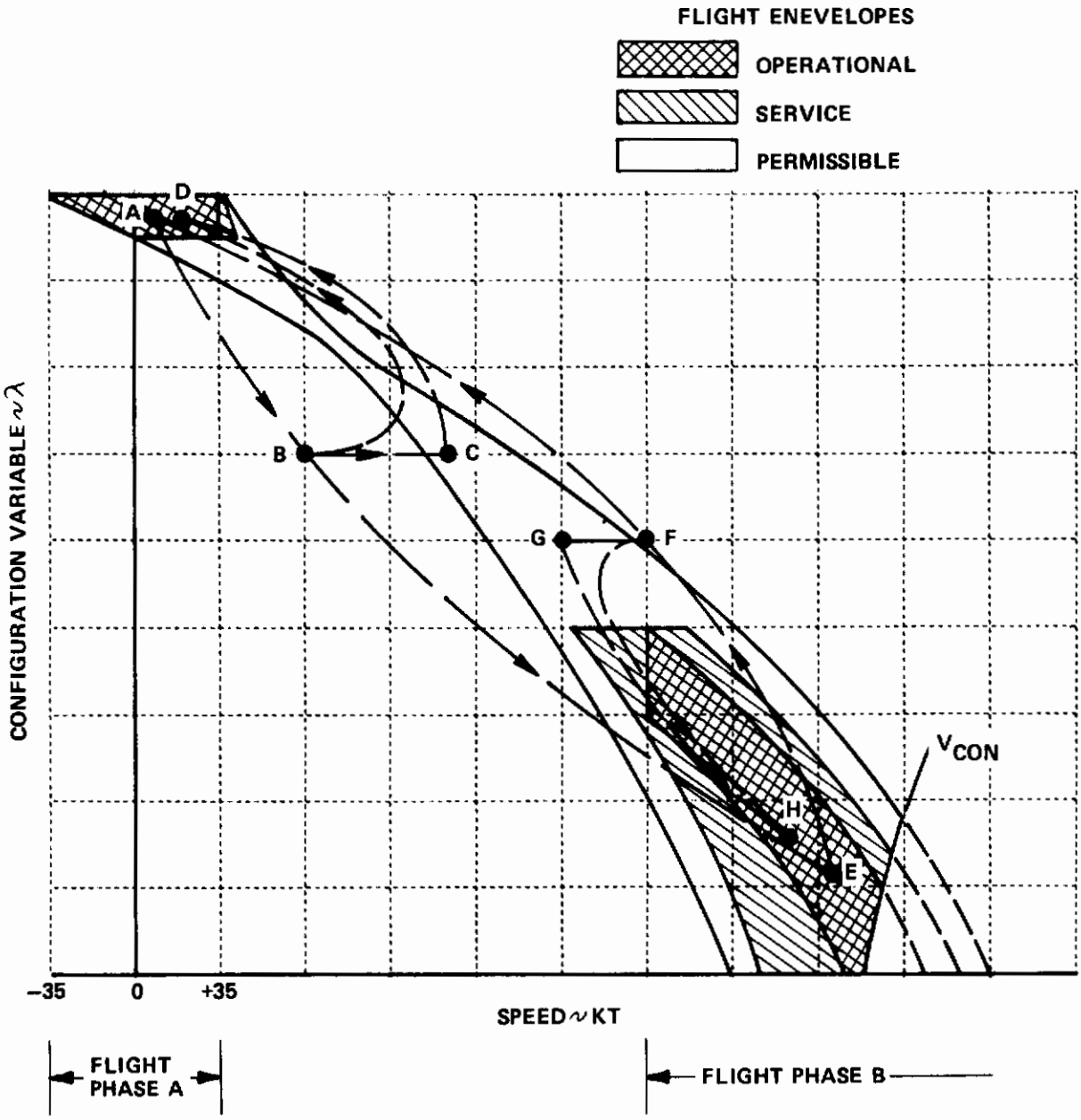

Figure 1(3.4.2) EXAMPLES OF TRANSITIONS ILLUSTRATING FLEXIBILITY OF OPERATION

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3.4.3 TOLERANCE IN TRANSITION PROGRAM

REQUIREMENT

3.4.3 Tolerance in transition program. It shall be possible to change from hover or minimum speed to conventional flight, and vice versa, safely and easily. There shall be no need for precise programming by the pilot of engine power, fuselage attitudes, wing or lift engine tilt, etc., in terms of speed or time, such as to demand excessive pilot skill and attention.

DISCUSSION

The basic intent of this requirement is to ensure that excessive pilot skill and attention, and precise pilot programming of engine power, etc., are not required when the pilot is performing transition. Although precise programming may be required of the devices used to transition, manual control of transition shall not be complex and result in significant increases in pilot workload.

The need for precise programming would arise if the allowable transition corridor was too narrow. This is illustrated on Figure 1(3.4.3). At just over 35 knots there is a very restricted range of speeds allowed as the configuration changes. To pass through such a narrow corridor would require the pilot to exercise very careful control over the transition, making the aircraft difficult to fly and diverting the pilot's attention from other tasks. Ideally the corridor will be very wide and natural phenomena such as unusual attitudes (not dangerous attitudes) will alert the pilot that he is too fast, too slow, or needs more or less power, etc. Pilot control of transition should not divert the pilot's attention from any operational task required to be performed during the transition in such a manner as to compromise the operational effectiveness of the aircraft.

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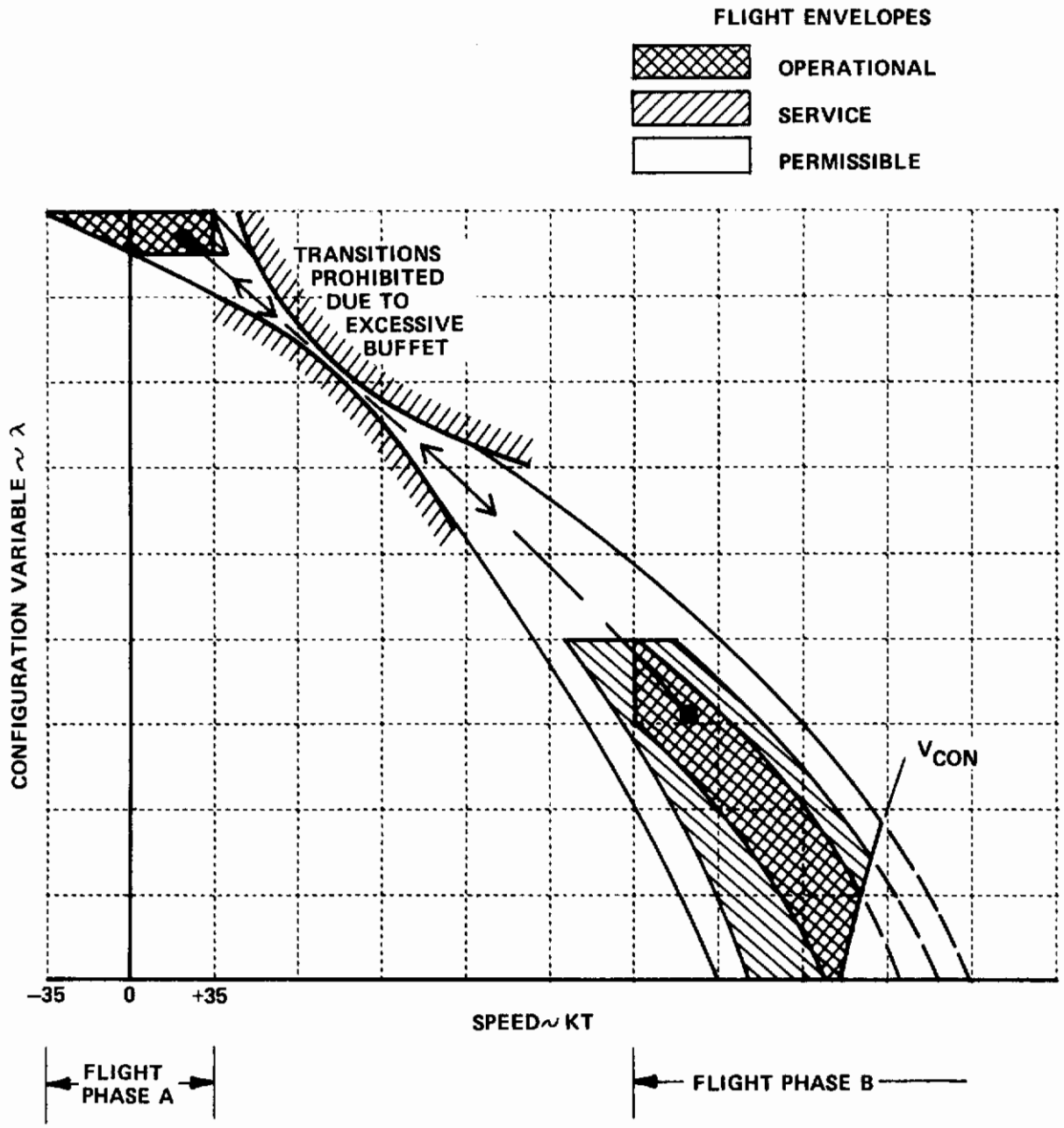

Figure 1 (3.4.3) ILLUSTRATION OF TRANSITION REQUIRING PRECISE PROGRAMMING

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3.4.4 CONTROL MARGIN

REQUIREMENT

3.4.4 Control margin. To allow for disturbances and for maneuvering, the margin of control power remaining at any stage in the transition shall not be less than 50 percent of the nominal pitch, roll and yaw control moments available.

DISCUSSION

The intent of this requirement is to provide sufficient control power margin to allow the pilot to control gust disturbances and to maneuver the aircraft as required during the transition. This requirement is similar to 5.5 of AGARD 408A (Reference 46) except that the value of control margin has been changed to 50% and extended to the roll and yaw axes.

Reference 112 indicates that, although the P.1127 suffers from a lack of stability about the pitch axis in transition, the abundance of control power and the precise response of the control system helped compensate for the lack of stability at low speeds.

Reference 77 on the CL-84 aircraft reports that the aircraft has an undesirable tendency to settle (50 foot loss in height) during the terminal phase of constant altitude conversions, however the ability to rotate the nose up could minimize this condition.

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3.4.5 TRIM CHANGES

REQUIREMENT

3.4.5 Trim changes. All trim changes throughout the transition shall be small. Without retrimming, the pitch control forces shall not exceed 15 pounds pull or 7 pounds push.

DISCUSSION

The purpose of this requirement is to limit the forces which have to be applied by the pilot to maintain the aircraft on the nominal transition. It is very likely that automatic phasing of controls (e.g., gearing of elevator angle) will be necessary to counteract the moment changes received by configuration changes such as tilting the wing. The designer can do this relatively easily for a certain transition. However, perturbations from this transition could result in errors; these the pilot will have to control. The forces that the pilot should be expected to encounter while performing any transition are limited by this requirement.

This requirement is similar to 5.6 of AGARD 408A, Reference 46. However, the trim change requirement is now extended to all axes, and the pitch control forces have been modified. Flight test reports that have documented forces during transition show that they are generally moderate. For example, Reference 116 shows maximum forces of approximately 6 pounds pull and push during XC-142 transitions. Light control forces of less than 6 pounds at full displacement in transition are reported for the XV-6A (P.1127) in Reference 112. However, Reference 66 indicates that for conversion from conventional to fan powered flight during transition, the XV-5A required forces of approximately 15 pounds pull to prevent altitude loss and these maneuvers were assigned a pilot rating of 2.5. The force limits of this requirement reflect the above data.

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3.4.6 RATE OF PITCH CONTROL MOVEMENT

REQUIREMENT

3.4.6 Rate of pitch control movement. During transition, with the maximum available rate of change of forward speed, the rate of pitch control movement to maintain trim shall not exceed 1 inch per second.

DISCUSSION

The pilot has a two part task during transition:

a. to maintain the aircraft on the nominal transition trajectory of speed and flight path, and

b. to control perturbations about this nominal.

The control position required to perform (a) may be thought of as the trim variation (though of course the aircraft is never trimmed in the sense of being in equilibrium). The significance of this trim variation will be from three aspects:

a. how rapidly the aircraft diverges due to errors in trim,

b. the magnitude of the trim forces required, and

c. the size of the trim control displacements.

The rapidity of divergence is probably the most important of these three effects. However, it is dependent not only on the out-of-trim moment, but also on the dynamics of the aircraft. As discussed in Section III, there are presently no quantitative data on which to base requirements for the dynamics desired during transition. Nor are there any quantitative data on out-of-trim moments. As a result, limits have been placed on the magnitude of the trim forces and the rate of control displacements since these are closely tied to the rate of divergence that is being controlled. It appears reasonable to limit divergences to such levels as can be controlled by stick deflections of 1 inch/second.

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3.5 CHARACTERISTICS OF THE FLIGHT CONTROL SYSTEM

REQUIREMENT

3.5 Characteristics of the flight control system. These requirements are concerned with those aspects of the flight control system which are directly related to flying qualities, and are imposed in addition to the requirements of the applicable control system design specifications, e.g., MIL-F-9490 for Air Force procurements or MIL-C-18244 for Navy procurements. Meeting the following requirements separately will not necessarily ensure that the overall system will be satisfactory; the mechanical characteristics must be compatible with the nonmechanical portions of the control system and with the aircraft dynamic characteristics. The requirements apply at all speeds up to V_{con}.

DISCUSSION

Because V/STOL aircraft generally exhibit reduced stability and damping characteristics in hovering and low speed flight, it is extremely important that the characteristics of the flight control system do not further degrade the vehicle's flying and handling qualities. Although the requirements, for the most part, are specified separately, consideration should be given to the overall system compatibility to assure satisfactory accomplishment of the aircraft mission.

Section 3.5 deals with the whole flight control system including, as defined in MIL-F-9490, the primary, secondary, and automatic flight control systems. No separation of the requirements along these lines is made since there might be unnecessary controversy in classifying some of the unique control systems characteristic of V/STOL aircraft. These requirements are, of necessity, largely qualitative since either a scarcity of data exists or the complex interplay of the various control system characteristics makes selection of specific values for a given parameter difficult or unrealistic. As a result the requirements rely heavily on the use of the Level structure as outlined in 1.5.

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3.5.1 MECHANICAL CHARACTERISTICS

REQUIREMENT

3.5.1 Mechanical characteristics. Some of the important mechanical characteristics of control systems (including servo valves and actuators) are: friction and preload, lost motion, flexibility, mass imbalance and inertia, nonlinear gearing, and rate limiting. Requirements for these characteristics are contained in 3.5.1.1 through 3.5.1.7.

DISCUSSION

This paragraph outlines some of the important mechanical characteristics which must be considered collectively in commenting on the acceptability of a flight control system and forms an introduction to the following seven paragraphs.

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3.5.1.1 CONTROL CENTERING AND BREAKOUT FORCES

REQUIREMENT

3.5.1.1 Control centering and breakout forces. Pitch, roll, and yaw controls shall exhibit positive centering in flight at any normal trim setting. 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., refer to the cockpit control force required to start movement of the control surface in flight. The requirements for breakout force are given in tables X and XI. Table X applies for all speeds less than 35 knots. At V_{con}, the values shown in table XI apply for Levels 1 and 2; for Level 3 the maximum values may be doubled. Between 35 knots and V_{con}, the breakout force may increase to but not exceed the V_{con} value provided the change in breakout force with speed is not objectionable.

TABLE X. Allowable Breakout Forces, Pounds, V< 35 knots

Cockpit Control		Level 1 min/max	Level 2 min/max	Level 3 max
Pitch		0.5/1.5	0.5/3.0	6.0
Roll		0.5/1.5	0.5/2.0	4.0
Yaw		2.0/7.0	2.0/7.0	14.0
Thrust Magnitude	throttle type	1.0/3.0	1.0/3.0	3.0
	collective type	1.0/3.0	1.0/3.0	6.0

TABLE XI. Allowable Breakout Forces at V_{con}, Pounds

Control	Classes I, II, IV		Class III	
	min	max	min	max
Pitch	0.5	3.0	0.5	5.0
Roll	0.5	2.0	0.5	4.0
Yaw	2.0	7.0	2.0	14.0

The minimum thrust-magnitude-control breakout force may be measured with adjustable friction set. Measurement of breakout forces on the ground will ordinarily suffice in lieu of actual flight measurement, provided qualitative agreement between ground measurement and flight observation can be established.

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DISCUSSION

Positive centering is defined as the tendency for a cockpit control to return to its initial position following release from a deflected position. With absolute centering, a cockpit control will always return, unassisted, exactly to its trim position when released.

Positive centering aids the pilot in gauging the magnitude of control displacements from trim and in returning to trim following maneuvering. This characteristic is extremely desirable in V/STOL aircraft in hovering and low speed flight where their inherent aerodynamic stability is lowest. References 114 and 115 note that positive centering is also a desirable characteristic during instrument flight conditions.

The following discussion is presented to illustrate the intent of this requirement. It is not intended to be a comprehensive analysis of the myriad of possible force feel system configurations.

Figure 1a(3.5.1.1) illustrates the cockpit control static force-deflection characteristic obtained with a simple linear spring force-feel system. It is assumed there is no force feedback from the control surfaces. This system has absolute centering since, upon removal of the applied force, the control returns to its initial position.

The more usual situation in an aircraft control system is to have some residual friction which results in a force-feel characteristic as illustrated in Figure 1b(3.5.1.1). This control system possesses positive but not absolute centering. If the control is displaced by a force from the initial position to point A and released, the control tends to return, but due to friction stops at B, short of the trim position. If displaced to position C and released, the control will not return but remain deflected at D with zero force. In fact, the control may occupy deflected positions from E to B with zero force. The greater the magnitude of this zero force displacement band relative to the total control deflection, the less positive the centering characteristic. Over this band, the lack of control force provides no cue to the pilot as to the magnitude of the control deflection. For this example, the centering characteristic can be made more positive by increasing the force gradient or decreasing the friction. However, control gradients can only be increased subject to the limitations of paragraph 3.5.1.2 and small amounts of residual friction may be difficult to eliminate.

It is possible, however, to further reduce the trim control displacement band through the introduction of preloaded springs to the force-feel system. The improvement of the centering characteristic with preload is indicated in Figure 1c(3.5.1.1). However, preload cannot be used indiscriminantly to achieve good control centering because of the possibility of excessive breakout forces. Excessive breakout forces tend to make precise maneuvering difficult and can lead to overcontrolling especially when combined with relatively low force gradients. The cockpit control breakout force is taken to be the force which must be overcome to start movement of

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the cockpit control, starting from a control position midway through the trim displacement band. In the example of Figure 1c(3.5.1.1), the breakout force is, therefore, equal to one-half the friction force, plus the preload.

No quantitative limits are specified for control centering characteristics because of the lack of a suitable data base. However, the specification does require that the effects of centering, together with other control system characteristics, shall not produce objectionable flying qualities.

This requirement places quantitative limits on breakout forces in terms of the forces required to start movement of the appropriate control surface(s). This breakout force will be equal to the cockpit control breakout force only if there is a unique control surface position corresponding to each cockpit control position. This unique relationship will not exist, for example, if there is free play (as defined in Paragraph 3.5.1.3) in the control system between the cockpit control and the appropriate control surface. Since the vehicle cannot respond to control inputs until the control surface moves, it is considered that this definition of breakout force yields a more meaningful force-feel system parameter.

The specification allows the use of ground measured breakout forces when qualitative agreement between ground and in-flight measured characteristics can be established. Ground measurements will generally not be adequate when there is significant control surface aerodynamic force feedback to the cockpit controls or when the characteristics of control surface actuators are altered by aerodynamic force feedback.

There is general agreement among handling qualities investigators that a minimum value of breakout force is desirable to reduce the probability of inadvertent control inputs, while an upper limit is necessary to facilitate precise maneuvering and reduce the tendency to overcontrol. The specification takes cognizance of the fact that lighter breakout forces are desirable in hover and low speed flight, as opposed to conventional cruise conditions, to minimize pilot fatigue.

Table 1(3.5.1.1) lists the breakout force characteristics specified in AGARD 408A (Reference 46), MIL-H-8501A (Reference 15) and RTM-37 (Reference 47),for comparison with the hover and low speed requirements of the present V/STOL specification (Reference 1). The Level 1 values are identical to those of MIL-H-8501A with the exception of the minimum yaw control breakout force.

The requirements of RTM-37 are seen to be in reasonable agreement with the V/STOL specification if Normal Operation is taken to correspond to Level 2 handling qualities and After Failure of Power Control to correspond to Level 3.

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Table 2(3.5.1.1) summarizes breakout force data for existing V/STOL aircraft. Aircraft with satisfactory characteristics are generally within the Level 1 requirements of Table X and where objectionable characteristics are cited, the breakout forces correspond to Level 2 or 3 values.

Table 1(3.5.1.1)
COMPARISON OF BREAKOUT FORCES

Cockpit Control	Min/Max Breakout Force, Pounds			
			RTM-37 (Ref. 47)	
	AGARD 408A (Ref. 46)	MIL-H-8501A (Ref. 15)	Normal Operation	After Failure of Power Control
Elevator	0.5/2.5	0.5/1.5	0.5/2.5	5.0
Aileron	0.5/2.0	0.5/1.5	0.5/2.0	4.0
Rudder	1.0/10.0	3.0/7.0	1.0/10.0	15.0
Thrust - throttle	1.0/3.0	-	1.0/3.0	3.0
- collective	1.0/3.0	1.0/3.0	1.0/3.0	5.0

Table 2(3.5.1.1)

MEASURED BREAKOUT FORCES

Aircraft	Breakout Force, Pounds				Remarks
	Reference	Elevator	Aileron	Rudder	
P.1127(XV-6A)	112	0.5-1.0	0.5-1.0	5.5	Desirable
XV-5A	66	4.0	3.0	5.0	Lower in Flight - Pilots
XV-5A	113	1.0	1.0	2.0	-
XC-142A	116, 117	1.0-2.0	1.0-2.0	10.0-14.0	Direct. too high
X-22A	117	0.7	0.6	8.0	Long. and lat. too low Dir. acceptable for research role
X-22 Sim.	117	0.5	0.5	3.0	Optimum from Sim.
CL-84	77	0.9	0.9	5.0	Boost on
CL-84	77	3.0	3.0	5.0	Boost off
SC-1	118	1.0	1.0	5.0	-
CH 46, 47	117	1.0-1.5	1.0	9.0-13.0	Long. & Dir. too high for hover
Kaman Co.	117	0.5	0.5	-	Considered optimum
AH-1G	91	3	2	-	Breakout high

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The specification recognizes that above V_{con}, the requirements of MIL-F-8785B(Reference 10) will be imposed. Therefore at V_{con}, the most lenient values demanded by MIL-F-8785B have been imposed. Between 35 knots and V_{con}, an unobjectionable phasing is required. The Level boundaries are indicated in Figure 2(3.5.1.1).

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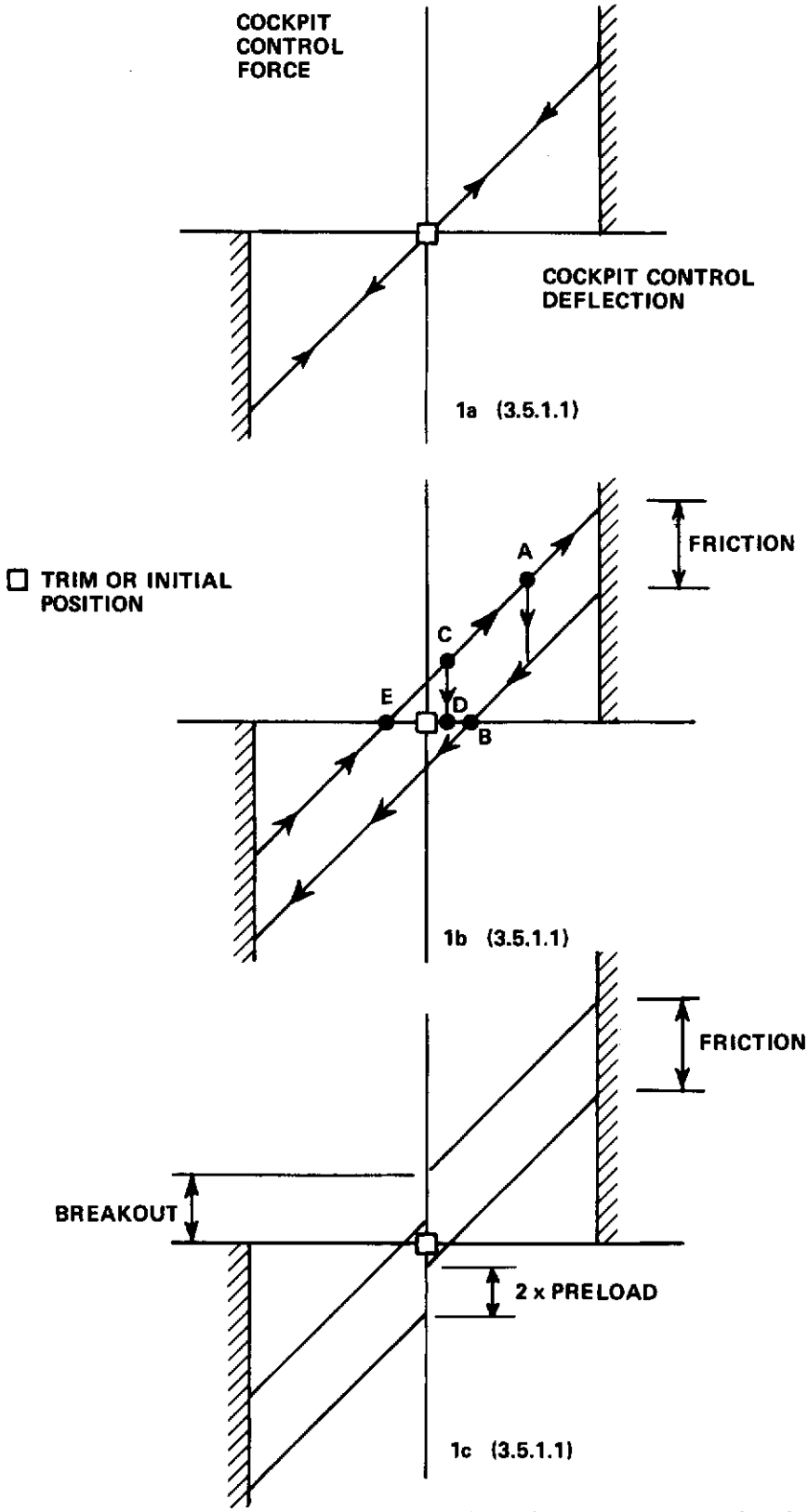

Figure 1 (3.5.1.1) EXAMPLES OF COCKPIT CONTROL FORCE-DEFLECTION CHARACTERISTICS

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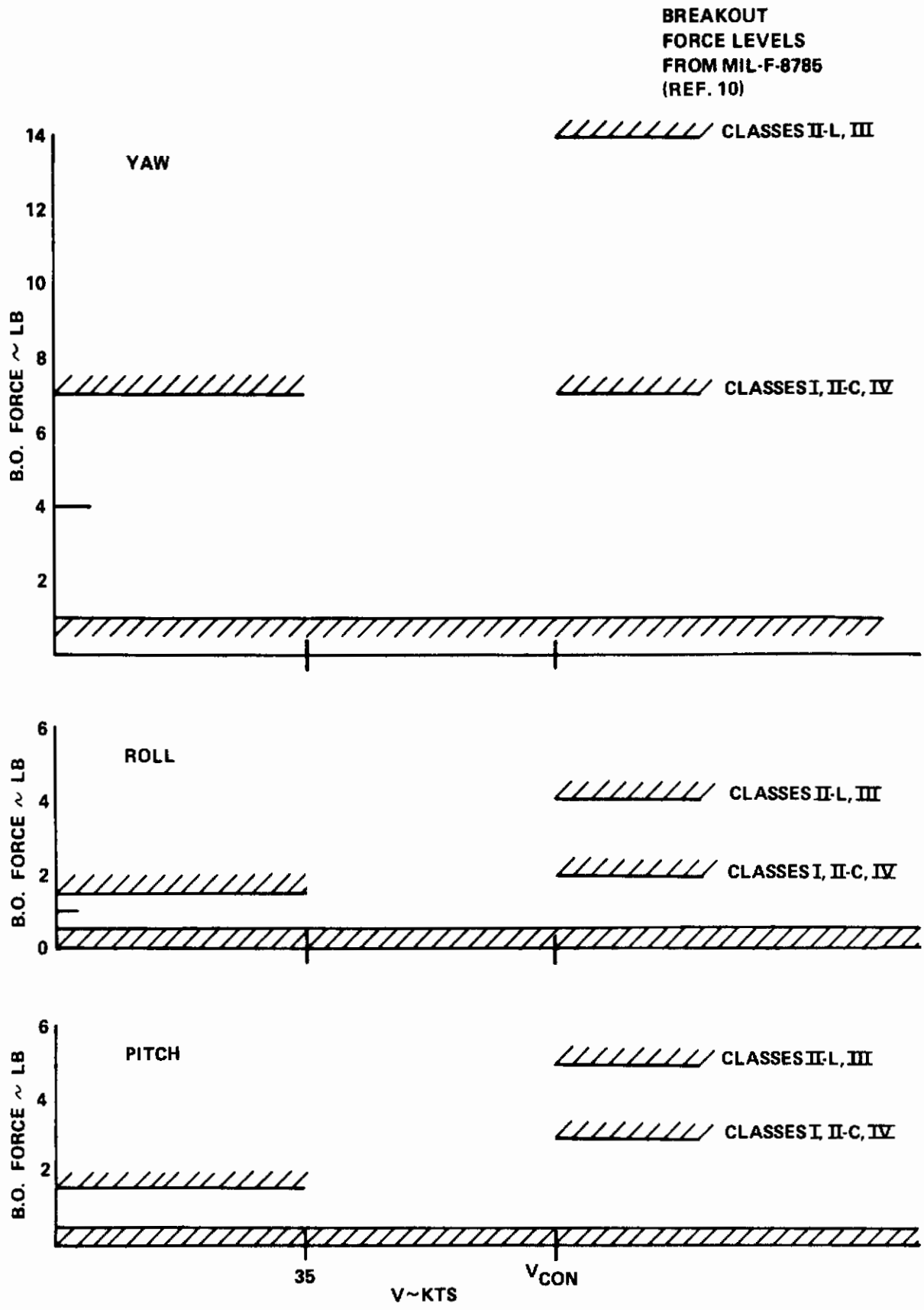

Figure 2 (3.5.1.1) LEVEL I BREAKOUT FORCE REQUIREMENTS

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3.5.1.2 COCKPIT CONTROL FORCE GRADIENTS

REQUIREMENT

3.5.1.2 Cockpit control force gradients. At speeds up to 35 knots, the pitch, roll and yaw control force gradients shall be within the range specified in table XII throughout the range of control deflections. From 35 knots to V_{con}, transition of the gradients to the values required to comply with MIL-F-8785 at V_{con} is allowed in any manner which is not objectionable to the pilot. In addition, the force produced by a 1-inch travel from trim by the gradient chosen shall not be less than the breakout force. For the remaining control travel, the local gradients shall not change by more than 50 percent in one inch of travel. The thrust magnitude control should preferably have zero force gradient.

TABLE XII. Allowable Control Force Gradients, Pounds/Inch

Control	Level 1		Level 2	
	min	max	min	max
Pitch	0.5	3.0	0.5	5.0
Roll	0.5	2.5	0.5	5.0
Yaw	5.0	10.0	5.0	20.0

DISCUSSION

The control force gradient characteristics specified in Table XII for hover and low speed flight ($V < 35$ knots) are based on the results of simulator studies and on the gradients selected for existing V/STOL aircraft. Table 1(3.5.1.2) summarizes these data. The control gradient values are somewhat lower than those usually associated with conventional aircraft. This preference for lower gradients is probably attributable to the higher control activity associated with the stability characteristics of V/STOL aircraft in hovering and low speed flight.

Table 2(3.5.1.2) summarizes the control gradients specified by AGARD 408A (Reference 46) and MIL-H-8501A (Reference 15) for comparison. The Level 1 requirements of the present specification are seen to lie between the minima of MIL-H-8501A and the maxima of AGARD 408A.

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Table 1(3.5.1.2)
MEASURED FORCE GRADIENTS

Aircraft	Force Gradient, Pounds/Inch				Remarks
	Reference	Elevator	Aileron	Rudder	
P.1127(XV-6A)	112	2.5	1.7	8.0	Hover Values - Desirable
XV-5A	66	1.5	1.3	8.0	Fan Power Mode
XV-5A	113	1.0	1.0	1.7	-
XC-142A	116	-	-	60.0	Too High
X-22A	117	1.2	1.2	5.0	For 50 knots and below
X-22 Sim.	117	0.9	1.0	5.0	Optimum
CL-84	77	1.3	1.7	8.0	Boost on
CL-84	77	2.8	3.0	8.0	Boost off
SC-1	118	2.0	1.1	5.0	Hover Values
CH-46	117	0.56	0.75	3.3	-
CH-47	117	0.75	0.70	3.9	-
Lockheed Rigid Rotor	117	4.0-5.0	2.0	-	Long:Lat should be 2:1

Table 2(3.5.1.2)

COMPARISON OF FORCE GRADIENT REQUIREMENTS

Cockpit Control	Min/Max Force Gradients, Pounds/Inch	
	AGARD 408A (Ref. 46)	MIL-H-8501A (Ref. 15)
Elevator	1.0/2.5	0.5/2.0
Aileron	1.0/2.5	0.5/2.0
Rudder	5.0/15.0	-

The requirement that the force produced by a one-inch travel from trim should be greater than the breakout force is imposed to help prevent overcontrolling and PIO which tends to arise when low gradients are used in combination with relatively high breakout forces. In these cases, having applied a force sufficient to overcome the breakout, only a small increase in force results in relatively large control inputs.

In addition, control gradient changes are restricted to less than ± 50 percent per inch of stick travel. Little quantitative data regarding the effects of nonlinear force-deflection characteristics exists. However, it is

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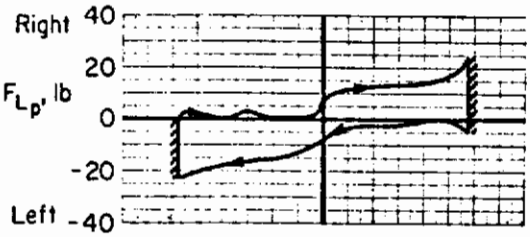
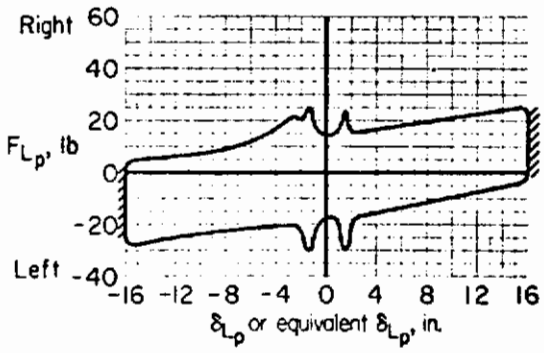
known that when large nonlinearities are present they are noticeable to pilots and the control system is downrated for this reason. Examples of this are the roll control force-deflection characteristics of the STOL NC-130B and the YC-134A described in Reference 73 and reproduced here as Figure 1(3.5.1.2).

NC-130B



1. Wheel deflection ± 90°, too large
2. Powered control system with high friction between wheel and actuators, ground measured forces
3. Unsatisfactory force characteristics
4. Unsatisfactory aircraft response, probably due to low sensitivity and high friction

YC-134A



1. Wheel deflection ± 120°, too large
2. Mechanical control system, ground measured forces not really indicative of in-flight forces
3. Unsatisfactory forces
4. Unsatisfactory nonlinear forces and response
5. High inertia

V=80knots,

Figure 1 (3.5.1.2) LATERAL CONTROL FORCE-DISPLACEMENT CHARACTERISTICS OF TWO STOL AIRCRAFT (FROM REFERENCE 73)

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3.5.1.3 COCKPIT CONTROL FREE PLAY

REQUIREMENT

3.5.1.3 Cockpit control free play. The free play in each control, that is, any motion of the cockpit control which does not move the appropriate moment- or force-producing device in flight, shall be compatible with the required Level of flying qualities.

DISCUSSION

The amount of tolerable free play in a given configuration is a complex function of the vehicle and control system dynamics, the force-feel system characteristics, etc. It is recognized that at the present time, absolute limits cannot realistically be placed on the magnitude of the free play allowed. Therefore, the problem of determining compliance is referred to the defined Level structure.

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3.5.1.4 RATE OF CONTROL DISPLACEMENT

REQUIREMENT

3.5.1.4 Rate of control displacement. The ability of the aircraft to perform the operational maneuvers required of it, or to operate in an atmospheric disturbance environment consistent with the operational missions of 3.1.1, shall not be limited by the rate of movement of the moment- or force-producing devices. For powered or boosted controls, the effect of engine speed and duty cycle of any part of the flight control system, together with pilot control techniques, shall be included when establishing compliance with this requirement.

DISCUSSION

The intent of this paragraph is straightforward. The second sentence points out that powered or boosted devices may use up significant portions of the available power during critical phases of the mission. For example, actuation of landing gear, flaps, slats, etc., during the landing approach or takeoff 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, that same hydraulic system might be more than adequate. For example, flight testing described in Reference 77 revealed that the CL-84 does not have sufficient hydraulic power to actuate the wing tilt mechanism while the landing gear is being retracted.

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 119, 120, and 121). This technique is likely to be used when the short-term longitudinal response is too slow, or if the phugoid is unstable. With critical flight characteristics such as these, it is especially important that rate limiting of the control system does not restrict the pilot's control technique.

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.

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3.5.1.5 ADJUSTABLE CONTROLS

REQUIREMENT

3.5.1.5 Adjustable controls. When a cockpit control is adjustable for pilot physical dimensions or comfort, the control forces defined in 6.2.4 shall refer to the mean adjustment. A force referred to any other adjustment shall not differ by more than 10 percent from the force referred to the mean adjustment.

DISCUSSION

When adjustable controls are utilized, some variation in control force characteristics must be tolerated due to different moment arms to control pivots, etc. Ten percent (plus or minus) appears to be a reasonable limit to the allowed variation in control forces defined in 6.2.4.

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3.5.1.6 CONTROL HARMONY

REQUIREMENT

3.5.1.6 Control harmony. The control forces, displacements, and sensitivities of the pitch, roll and yaw controls shall be compatible, and their responses shall be harmonious.

DISCUSSION

This paragraph, together with paragraph 3.5.1.7, is intended to assure ease of control in maneuvering the aircraft. Lack of an adequate data base precludes a quantitative requirement. "Control harmony" implies a satisfactory relationship between the pitch, roll and yaw controls in terms of angular response of the vehicle per unit control force, deflection, etc. Several aspects of this relationship are described below.

A desired characteristic 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. Further, 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, an aircraft with roll control forces which are much too high with respect to pitch control forces may be difficult to control in pitch when rolling rapidly into a turn. The intent of 3.5.1.6 is to prevent situations such as these.

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3.5.1.7 MECHANICAL CROSS-COUPLING

REQUIREMENT

3.5.1.7 Mechanical cross-coupling. Displacement of one cockpit control shall not produce objectionable forces or displacements at any of the other cockpit controls.

DISCUSSION

The requirement to limit mechanical cross-coupling arises from the need for easy one-handed control of the aircraft, particularly in hovering and low speed flight. The presence of high levels of force or displacement cross-coupling tends to produce unwanted aircraft motions, making coordinated flight difficult. Paragraphs 3.5.1.6 and 3.5.1.7 are intended to prevent these difficulties.

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3.5.2 DYNAMIC CHARACTERISTICS

REQUIREMENT

3.5.2 Dynamic characteristics. The control deflection shall not lead the control force for any frequency or force amplitude. This requirement applies to the pitch, roll, yaw and thrust magnitude controls.

DISCUSSION

This requirement is similar in intent to 3.5.3.1 of MIL-F-8785B (Reference 10) and is directed to bobweights and other devices which feed back the aircraft's response into the feel system. Specific mention of allowable lags in the feel system is not made since no supporting data exist.

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3.5.2.1 DAMPING

REQUIREMENT

3.5.2.1 Damping. All control system oscillations shall be well damped, unless they are of such an amplitude, frequency, or phasing that the cockpit-control or airframe oscillations resulting from abrupt maneuvers or flight in atmospheric disturbances are compatible with the required Level of flying qualities as determined from 3.1.10.

DISCUSSION

The paragraph is based on requirement 3.5.3.2 of MIL-F-8785B (Reference 10).

V/STOL aircraft, including those utilizing boosted control systems and stability and control augmentation devices may be prone to objectionable control system oscillations. This difficulty often arises when inadequate separation is provided between the airframe natural frequencies and the frequencies of the control system modes. It is aggravated by any slop or deadbands in the system. In addition, the designer is reminded that feedback into the augmentation system from vibratory and aeroelastic modes should not be overlooked as a source of objectionable control system and airframe oscillations. In Reference 84, it is pointed out that certain types of adaptive control systems employing modes with low damping and natural frequency may also give rise to objectionable control-system oscillations.

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3.5.3 LIMIT COCKPIT CONTROL FORCES

REQUIREMENT

3.5.3 Limit cockpit control forces. Unless otherwise specified in particular requirements, the maximum control forces required, without retrimming, for any maneuver consistent with service use, shall not exceed the values stated in table XIII.

TABLE XIII. Limit Cockpit Control Force Values, Pounds

Cockpit Control	V< 35 Knots			35 Knots < V < V_{con}		
	Level 1	Level 2	Level 3	Level 1	Level 2	Level 3
Pitch	10.0	20.0	40.0	30.0	35.0	40.0
Roll	7.0	15.0	20.0	15.0	20.0	25.0
Yaw	30.0	40.0	80.0	75.0	100.0	125.0
Thrust Magnitude						
Throttle type	3.0	3.0	3.0	3.0	3.0	3.0
Collective type	7.0	7.0	7.0	7.0	7.0	7.0

DISCUSSION

The limit cockpit control forces (pitch, roll and yaw) for Level 1 ($V < 35$ knots and 35 knots $< V < V_{con}$) are the values recommended in RTM-37 (Reference 47).

The Level 3 values for $V < 35$ knots correspond to those of AGARD 408A (Reference 46) with a failed power control system. The Level 3 values for 35 knots $< V < V_{con}$ are taken from RTM-37 except that the maximum longitudinal control force has been reduced from 50 to 40 pounds. It is the experience of test pilots from the ARPS at Edwards AFB that these Level 3 values are consistent with the physical capabilities of most pilots under emergency conditions such as failures of powered control systems. It was found that experienced pilots asked to land with one engine out (in a B-57) frequently lost control. The limit forces encountered were 50, 30, and 160 lb for the pitch, roll, and yaw respectively.

The Level 2 force limits are interpolations between the Level 1 and Level 3 values.

The limit forces for collective type thrust magnitude controls are those of MIL-H-8501A, Reference 15. For throttle type controls, the force limits are identical to the maximum breakout forces recommended by AGARD 408A (Reference 46) after failure of the power control system.

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3.5.4 AUGMENTATION SYSTEMS

REQUIREMENT

3.5.4 Augmentation systems. Normal operation of stability augmentation and control augmentation systems and devices shall not introduce any objectionable flight or ground handling characteristics.

DISCUSSION

This requirement is identical to paragraph 3.5.4 of MIL-F-8785B (Reference 10). The intent is to assure that the introduction of stability and control augmentation devices to improve the dynamic characteristics of an aircraft does not produce undesirable side effects. Several examples are cited below to illustrate the problems which may arise with these devices.

Depending on authority, pitch rate dampers may tend to saturate in steady banked turns resulting in a loss of longitudinal stability. The use of a high pass filter to wash out such steady-state signals may, in turn, reduce the longitudinal stability.

The control response characteristics of an aircraft may be heavily influenced by the force-feel and control system dynamics. "Shaping" of the control commands by the force-feel and control system may completely mask the inherent response characteristics of the basic aircraft.

Improperly located augmentation system response sensors may result in undesirable vehicle dynamic characteristics. Placing an acceleration sensor at a structural anti-node may generate augmentation system instabilities particularly with wide bandwidth augmentation systems.

Additional examples may be found in AFFDL-TR-69-72 (Reference 84).

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3.5.4.1 PERFORMANCE OF AUGMENTATION SYSTEMS

REQUIREMENT

3.5.4.1 Performance of augmentation systems. Any degradation of the performance of augmentation systems during flight in a severe atmospheric disturbance environment consistent with the operational missions of 3.1.1, or because of structural vibrations, shall be taken into account in demonstrating compliance with the required Level of flying qualities. In addition, any limits on the authority of augmentation systems or saturation of equipment shall not produce flying characteristics inconsistent with the required Level of flying qualities.

DISCUSSION

This requirement combines 3.5.4.1 and 3.5.4.2 from MIL-F-8785B (Reference 10) and incorporates these requirements into the qualitative Level structure philosophy. The first part of the requirement is aimed at recent self-adaptive control systems which depend upon automatic gain changes to keep the loop gains as high as possible without driving the system unstable. Some of these systems have a tendency to drive the loop gains down when the airplane flies in turbulence or when a structural mode is excited, resulting in poor system performance.

The second part serves as a reminder to the designer that limiting the authority of augmentation devices for safety purposes also may limit the effectiveness for improving flying qualities. For instance, a limited-authority pitch-rate damper may improve pitch response in light turbulence and for precision tracking tasks, but the nonlinearity of the airplane's response in maneuvering tasks due to saturation of the rate damper might be extremely objectionable.

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3.5.5 FAILURES

REQUIREMENT

3.5.5 Failures. Special provisions shall be incorporated to preclude any critical single failure of the flight control system including trim devices or stability augmentation system which may result in flying qualities which are dangerous or intolerable. Failure-induced transient motions and trim changes resulting either immediately after failure or upon subsequent transfer to alternate control modes shall be small and gradual enough that dangerous flying qualities never result. In addition, the crew member concerned shall be provided with immediate and easily interpreted indications whenever failures occur in the flight control system.

DISCUSSION

Experience with the present generation of V/STOL aircraft indicates that most of these vehicles require some form of stability and/or control system augmentation and partially or fully boosted control systems for satisfactory flying qualities. This requirement directs the designer to assure that a single failure of any component within these augmentation devices cannot result in unacceptable flying qualities. Further, it is not sufficient to assure that acceptable flying qualities exist in the failed state; the transients following failure shall also be sufficiently small that dangerous conditions do not result during the failure recovery.

The final sentence requires that any failure of these devices shall be indicated to the appropriate crew member, even if no transient motions of the aircraft accompany the failure. For example, a failure of a single channel of a dual stability and control augmentation system may produce no warning transients but result only in reduced authority. Continued straight and level flight is possible but turbulence encounters or gross maneuvering will saturate the augmentation system. In this situation, the crew should be warned of the failure so that appropriate action may be taken. Paragraph 3.5.5 is also intended to apply to automatic devices which function only in the event of a failure. For example, paragraph 3.8.10.1 permits the use of such a device to maintain control of the aircraft following a thrust or powered lift loss on the ground. Any failure of this device must be indicated to the appropriate crew member.

In general, hidden failures of any automatic flight control system or stability and control augmentation or trim devices are not permissible.

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3.5.5.1 CONTROL FORCE TO SUPPRESS TRANSIENTS

REQUIREMENT

3.5.5.1 Control force to suppress transients. Without retrimming, the cockpit control forces required to suppress transients following a failure in any part of the flight control system shall not exceed one-half the Level 1 limit control force values in table XIII.

DISCUSSION

This requirement specifies the allowable cockpit control forces required to comply with 3.8.9.1, that is, to avoid dangerous conditions by pilot corrective action following a failure. Table 1(3.5.5.1) compares the force limits of MIL-F-8785B (Reference 10) with the V/STOL specification limits (Reference 1).

TABLE 1(3.5.5.1) Transient Force Limits ~ lb

	V/STOL Specification (Ref. 1)		MIL-F-8785B (Ref. 10)
	V<35 knots	35<V<V_{con} knots	Paragraph 3.5.5.2
Pitch	5	15	20
Roll	3.5	7.5	10
Yaw	15.	37.5	50
Thrust Magnitude			
Throttle type	1.5	1.5	-
Collective type	3.5	3.5	-

The limit control forces increase in magnitude as the appropriate speed regime increases. This characteristic is considered reasonable since pilots tend to prefer the lightest control forces in hover and low speed flight. The values specified apply for all Levels since no data are available to support separate values. These values are speed dependent since Level 1 limit control forces are specified for both speed regimes in table XIII of Reference 1.

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3.5.6 TRANSIENTS AND TRIM CHANGES

REQUIREMENT

3.5.6 Transients and trim changes. This requirement applies to all Aircraft State changes made under conditions representative of operational procedure by activation of the Aircraft State selectors and controls available to the pilot. With the aircraft initially trimmed at a fixed operating point, the peak pitch, roll, and yaw control forces required to suppress the transient aircraft motions resulting from the change and maintain the desired heading, attitude, altitude, rate of climb or descent, or speed without use of the trimmer control, shall not exceed one-third of the appropriate limit control force in table XIII. This applies for a time interval of at least 5 seconds following completion of the pilot action initiating the change. The magnitude and rate of trim change after this period shall be such that the forces can be trimmed as required in 3.5.7. There shall be no objectionable buffeting or oscillations of the control device during the change.

DISCUSSION

This section is intended to cover the same area as 3.6.3 and 3.6.3.1 of MIL-F-8785B (Reference 10), with the exception that the configurations to be tested are not specified since this cannot be done in general terms for the variety of control systems used in VTOL aircraft. The trim force requirements are again somewhat arbitrary but are felt to be compatible with the MIL-F-8785B values and are Level dependent. For example, MIL-F-8785B allows peak elevator cockpit control forces of 10 pounds for Level 1, and 15 pounds for Levels 2 and 3.

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3.5.6.1 TRANSFER TO ALTERNATE CONTROL MODES

REQUIREMENT

3.5.6.1 Transfer to alternate control modes. The transients and trim changes caused by the intentional engagement or disengagement of any portion of the flight control system consistent with normal service use, such as selection of a particular augmentation mode, shall not be objectionable. Additional requirements are contained in MIL-F-9490 for Air Force procurements.

DISCUSSION

The intent of this paragraph is straightforward. No quantitative requirement is possible because of the great variety of flight control system configurations which are possible with V/STOL aircraft.

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3.5.7 TRIM SYSTEM

REQUIREMENT

3.5.7 Trim system. At all steady flight conditions within the Operational Flight Envelope, the trimming devices shall be capable of reducing the pitch, roll, and yaw control forces to zero for Levels 1 and 2. At all steady flight conditions within the Service Flight Envelope, the untrimmable cockpit control forces shall not exceed 10 pounds pitch, 5 pounds roll, and 20 pounds yaw. For Level 3, the untrimmed cockpit control forces shall not exceed 10 pounds pitch, 5 pounds roll, and 20 pounds yaw. The failures to be considered in applying the Level 2 and 3 requirements shall include trim sticking and runaway in either direction. It is permissible to meet the Level 2 and 3 requirements by providing the pilot with alternate trim mechanisms or override capability. Additional requirements on trim rate and authority are contained in MIL-F-9490 for Air Force procurements and MIL-F-18372 for Navy procurements.

DISCUSSION

This requirement is similar to paragraph 3.6.1 of MIL-F-8785B (Reference 10). The untrimmable cockpit control forces allowed for flight outside the Operational Flight Envelope but within the Service Flight Envelope, and for Level 3 are somewhat arbitrary, but should be small enough to be held for some length of time on rare occasions.

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3.5.7.1 RATE OF TRIM OPERATION

REQUIREMENT

3.5.7.1 Rate of trim operation. Trim devices shall operate rapidly enough to enable the pilot to maintain the pitch and roll control forces less than one-third of the appropriate limit forces in table XIII during any maneuver consistent with service use, but not ever to operate so rapidly as to cause oversensitivity or trim precision difficulties.

DISCUSSION

This requirement is similar to 3.6.1.2 in MIL-F-8785B (Reference 10) except that the specific force values were stated in terms of the limit forces of 3.5.3. These values seem reasonable relative to the MIL-F-8785B values which allow ± 10 pounds elevator cockpit control force, but no real substantiation exists.

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3.5.7.2 TRIM SYSTEM IRREVERSIBILITY

REQUIREMENT

3.5.7.2 Trim system irreversibility. All trimming devices shall maintain a given setting indefinitely unless changed by the pilot, by a special automatic interconnect such as to the 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 completion of each interconnect or augmentation operation.

DISCUSSION

This requirement is the same as paragraph 3.6.1.4 of MIL-F-8785B (Reference 10). The requirement is considered logical and necessary.

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3.6 TAKEOFF, LANDING AND GROUND HANDLING

REQUIREMENT

3.6 Takeoff, landing and ground handling. These requirements shall be satisfied within the Operational Flight Envelope for all applicable Category C Flight Phases.

DISCUSSION

This is an introductory paragraph which sets forth the conditions under which subsequent requirements are to be satisfied. Mention of the Operational Flight Envelope and the Category C Flight Phases makes it unnecessary to specifically define particular types of landings and takeoffs such as vertical, rolling, short, long, etc. Such definitions properly belong under the more general category of mission definition. The assumption is that the requirements apply to all types of takeoffs and landings that an aircraft is expected to perform. Special cases may arise where detailed definitions are necessary. For these cases, the procuring activity will supply or approve detailed Flight Phase definitions (see last sentence of paragraph 1.4).

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3.6.1 PITCH CONTROL EFFECTIVENESS IN TAKEOFF

REQUIREMENT

3.6.1 Pitch control effectiveness in takeoff. The effectiveness of the pitch control shall not restrict the takeoff performance of the aircraft and shall be sufficient to prevent over-rotation to undesirable attitudes. Satisfactory takeoffs shall not be dependent on the use of the trimmer control or on complicated control manipulation by the pilot.

DISCUSSION

The intent of this requirement should be self-evident.

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3.6.2 PITCH CONTROL FORCES IN TAKEOFF

REQUIREMENT

3.6.2 Pitch control forces in takeoff. With the trim setting optional but fixed, the cockpit pitch control forces shall not exceed one-half of the limits of table XIII in the pull direction or one-fourth of the limits of table XIII in the push direction at any time during the takeoff Flight Phases.

DISCUSSION

This requirement is based on requirement 2.15 of AGARD 408A (Reference 46), except that the force limits are somewhat different. If we equate AGARD's 408A normal operation with Level 1, then the differences are indicated in the following table.

		max pull force	max push force
Paragraph 3.6.2, Level 1	V < 35	5	2.5
	V > 35	15	7.5
AGARD 408A normal operation --		10	5

If we equate AGARD 408A, failure case, with Level 3, then there are no differences.

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3.6.3 PITCH CONTROL EFFECTIVENESS IN LANDING

REQUIREMENT

3.6.3 Pitch control effectiveness in landing. For Levels 1 and 2 the pitch control shall be sufficiently effective that the geometry-limited attitude or the guaranteed landing speed can be obtained at touchdown in the landing Flight Phase. For rolling landings this requirement must be met with the aircraft trimmed at the recommended approach speed for the approach Flight Phase. For Level 3 the pitch control shall be sufficiently effective to permit a safe landing.

DISCUSSION

Paragraph 3.2.3.4 of MIL-F-8785B (Reference 10) has been reworded slightly. The basic change is to trim at the recommended approach speed rather than a specific speed such as $V_S(L)$. The change was obviously necessary because $V_S(L)$ becomes unrealistic as an approach speed for V/STOL's. It could be mentioned that some manufacturers consider the requirement to fly conventional airplanes near the ground at $V_S(L)$ unnecessarily strict. Because of the imprecise nature of the landing flare maneuver, however, it is quite probable for a pilot to intentionally or unintentionally hold the airplane off the ground during the landing flare until the speed is well below the normal landing speed. In this event, it is essential that the pilot have enough pitch control to prevent the nose wheel from hitting the runway before the main gear.

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3.6.4 PITCH CONTROL FORCES IN LANDING

REQUIREMENT

3.6.4 Pitch control forces in landing. The cockpit pitch control forces required to meet the landing requirements of 3.6.3 shall not exceed one-half of the limits of table XIII in the pull direction or one-fourth of the limits of table XIII in the push direction at any time during the landing Flight Phase.

DISCUSSION

The comments made in the discussion of paragraph 3.6.2 are also applicable here.

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3.6.5 CROSSWIND OPERATION

3.6.5.1 LANDING AND TAKEOFF

REQUIREMENT

3.6.5 Crosswind operation

3.6.5.1 Landing and takeoff. It shall be possible to execute all of the takeoff and landing Flight Phases in crosswinds by using normal pilot skill and technique. The pitch, roll and yaw controls in conjunction with other means of control shall be adequate to maintain a straight path on the landing surface during takeoff runs and landing rollouts with cockpit control forces not exceeding the values specified in table XIII. These requirements apply in 90-degree crosswinds, right and left, of 30 knots.

DISCUSSION

An initial version of this requirement called out a 35-knot crosswind. The change to 30 knots was prompted by a number of review comments received from industry pointing out that a 35-knot crosswind requirement for V/STOL aircraft would be more severe than the comparable requirement for conventional airplanes in MIL-F-8785B. Also, it was pointed out that with low takeoff and landing speeds, these aircraft could be pointed out more into the wind than could a conventional aircraft.

However, there is a counter argument to this last point. The primary purpose of a V/STOL aircraft is to be able to use small landing areas. In forward battle areas it may be extremely inconvenient to clear other than a single, minimum width runway. It may also be necessary to limit direction to avoid over-flying an area occupied by hostile troops. Similarly in more developed areas, STOL aircraft will be required to operate from sites which are inside populated areas or at least closely bordering such areas. Buildings, bridges, high tension lines, other airways, etc. may in fact impose quite strict limitations on choice of takeoff and landing headings.

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3.6.5.2 FINAL APPROACH

REQUIREMENT

3.6.5.2 Final approach. Yaw and roll control shall be adequate to permit development of at least 15 degrees of steady, zero-yaw-rate sideslip in the power approach with yaw control forces not exceeding the values specified in table XIII. Roll control shall not exceed either 7.5 pounds of force or 50 percent of available control power (for the same configuration and flight condition), as applied manually or automatically or both, for Level 1. The limits are 10 pounds or 75 percent for Level 2. For Level 3, the roll control force shall not exceed 20 pounds.

DISCUSSION

This requirement is similar to requirement 3.3.7.1 of MIL-F-8785B (Reference 10). However it is a bit more strict because at the low speeds associated with STOL operation, moderate crosswinds can require fairly large crab and sideslip angles. Directional control in particular can become critical when trimming to compensate for a crosswind component in the approach. Because accurate estimates of crosswinds that might be encountered is unknown, it is felt that a requirement more severe than that imposed by MIL-F-8785B is warranted. The table below outlines the differences.

	Paragraph 3.6.5.2	MIL-F-8785B
Sideslip requirement	15 deg	10 deg
Max yaw control force, lb		
Level 1	75	100
Level 2	100	180
Level 3	125	180
Max roll control force, lb		
Level 1	7.5	10
Level 2	10	20
Level 3	20	20
Roll control remaining, % of available roll control	50	25

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3.6.5.3 COLD- AND WET-WEATHER OPERATION

REQUIREMENT

3.6.5.3 Cold- and wet-weather operation. The requirements of 3.6.5.1 shall be applicable on wet runways for all aircraft, and on snow-packed and icy runways for aircraft intended to operate under such conditions. If the demonstration for specification compliance is not accomplished under these adverse runway conditions, directional control shall be maintained by use of pitch, roll and yaw controls alone for all airspeeds above 30 knots. For very slippery runways, the requirement need only apply for crosswind components up to that at which the force tending to blow the aircraft off the runway is equal to the opposing tire-runway frictional force with the tires supporting all the aircraft's weight.

DISCUSSION

Some aircraft having large side area would tend to be blown sideways on very slippery runways if they operated in high crosswinds. Therefore it would be unreasonable to expect the aircraft to take off. When crosswinds are reduced to a value such that frictional forces can effectively contribute to the balance of aerodynamic forces, the task of maintaining a straight path on the ground is eased considerably. An analysis of an aircraft design for compliance with this requirement should consider the expected variations in lift, side force, and cornering force (tire-runway side force).

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3.6.6 POWER RUN-UP

REQUIREMENT

3.6.6 Power run-up. In all vertical takeoff configurations it shall be possible, without the use of wheel chocks or other restraints, to maintain a fixed position on a level surface during power run-up using only cockpit controls, in wind conditions to be specified by the procuring activity.

DISCUSSION

This requirement is based on an equivalent requirement in MIL-H-8501A (Reference 15). A fundamental difference is the necessity to refer to the vertical takeoff configuration. Physically, a helicopter is always in a vertical takeoff and landing configuration. VTOL airplanes, however, can rotate their thrust vectors. With thrust vector in a horizontal position, a full power run-up could result in the aircraft skidding over the ground because of the large thrust that is developed. The phrase "power run-up" deserves some clarification since it was of concern to some who reviewed earlier versions of the proposed specification. As used in the requirement, it refers to whatever procedures must be followed by the pilot to satisfactorily check the engines, transmission, etc., prior to takeoff.

As a ground handling requirement, this paragraph does not apply to thrust settings equal to, or greater than, the aircraft gross weight. It is not intended to require an anchor to hold the aircraft down during run-up.

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3.6.7 GROUND HANDLING

REQUIREMENT

3.6.7 Ground handling. It shall be possible to perform all required ground handling maneuvers, including taxiing, without damage to rotating components or any part of the structure. In addition, in winds up to 35 knots it shall be possible to taxi in a straight line at any angle to the wind and to make 360-degree taxiing turns in either direction within a circle whose radius equals the major dimension of the aircraft.

DISCUSSION

Considering that aircraft must transport themselves to various locations while on the ground, this requirement seems self-explanatory. It is a compilation of similar requirements in MIL-F-8785B, MIL-H-8501A, AGARD 408A and RTM-37, References 10, 15, 46, and 47 respectively.

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3.7 ATMOSPHERIC DISTURBANCES

REQUIREMENT

3.7 Atmospheric disturbances. Some requirements are written in terms of a steady wind speed, in which case, compliance with the requirement should be demonstrated in flight, in that wind condition. Other requirements are written with reference to operation in all potential atmospheric environments. For such cases the atmospheric disturbances such as discrete gusts, wind shear and turbulence to be used shall be chosen by the contractor subject to the approval of the procuring activity. Compliance shall be demonstrated by suitable analysis, test, or both, as determined by the procuring activity.

DISCUSSION

This section is almost self-explanatory, however, some words are in order as to how the specification has incorporated the effects of turbulence on V/STOL flying qualities. Unlike MIL-F-8785B, no mathematical model for turbulence has been offered. Experience in a number of ground and flight experiments, as well as experience with helicopters and experimental VTOL vehicles, indicates that aircraft response-to-turbulence characteristics are very important to the vehicle's flying qualities, and in some cases, are the most dominant effect. As much as possible, the detailed requirements have been selected by emphasizing the results of experiments which have been conducted in representative atmospheric environments. It should, however, be an aim of further research to identify mathematical models of turbulence to be encountered in V/STOL missions, and to develop criteria for adequate vehicle response in that turbulence. It is anticipated that work currently being conducted in this area by the Air Force will contribute to this purpose. It remains for future effort to incorporate such results in flying qualities requirements.

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3.8 MISCELLANEOUS REQUIREMENTS

3.8.1 APPROACH TO DANGEROUS FLIGHT CONDITIONS

REQUIREMENT

3.8 Miscellaneous requirements.

3.8.1 Approach to 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. Final determination of the adequacy of all warning of impending dangerous flight conditions will be made by the procuring activity, considering functional effectiveness and reliability. Devices may be used to prevent entry to dangerous conditions only if the criteria for their design, and the specific devices, are approved by the procuring activity.

3.8.1.1 Warning and indication. Warning or indication of approach to a dangerous condition shall be clear and unambiguous. For example, a pilot must be able to distinguish readily among: warning of loss of aerodynamic lift (which may require increased thrust), engine acceleration buffet (which may require decreased thrust) and normal aircraft vibration (which may not require a thrust change). If a warning or indication device is required, functional failure of the device shall be indicated to the pilot.

3.8.1.2 Prevention. As a minimum, dangerous-condition-prevention devices shall perform their function whenever needed, but shall not limit flight within the Operational Flight Envelope. Hazardous operation of these devices, normal or inadvertent, shall never be possible. For Levels 1 and 2, neither hazardous nor nuisance operation shall be possible. For Level 3, hazardous inadvertent operation shall not be possible.

DISCUSSION

These requirements are essentially the same as paragraphs 3.4.1, 3.4.1.1, and 3.4.1.2 of MIL-F-8785B (Reference 10). Similar requirements for V/STOL aircraft are clearly required.

Of necessity, the requirements for warning of approach to dangerous flight conditions are qualitative and wide latitude has been given to the procuring activity for approval or disapproval. The suitability of natural warning and/or the need for artificial warning is difficult to predict in the design phase and may not become apparent until late in the development program. Close coordination is, therefore, required between the contractor and the procuring activity to define these requirements at the earliest practical time. Paragraph 3.8.1.1 further requires that no hidden failures of these artificial warning devices be possible.

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Paragraph 3.8.1.2 recognizes that automatic devices may be used to prevent entry into dangerous conditions. The Level 1 and 2 requirement that neither hazardous nor nuisance operation be possible is intended to assure that these devices do not create additional problems through their malfunction.

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3.8.2 LOSS OF AERODYNAMIC LIFT

REQUIREMENT

3.8.2 Loss of aerodynamic lift. These requirements are related to those conditions where a loss of lift would result in a loss of altitude or partial loss of control.

3.8.2.1 Warning. The approach to a loss of aerodynamic lift shall be accompanied by an easily perceptible warning. Such a warning shall be provided artificially, subject to approval by the procuring activity, when natural warning is not feasible. The increase in warning intensity with further loss of lift shall be sufficiently marked to be noted by the pilot.

3.8.2.2 Prevention of loss of aerodynamic lift. It shall be possible to prevent loss of aerodynamic lift by normal use of the controls at the onset of the warning indication.

3.8.2.3 Control and recovery following loss of aerodynamic lift. In the event of loss of aerodynamic lift, it shall be possible to maintain control and recover by normal use of the controls, with control forces not exceeding the Level 3 requirements of table XIII. Recovery shall be accomplished without experiencing pitch, roll, or yaw attitude changes in excess of 20 degrees or excessive loss of altitude or buildup of speed. It is desired that no pitch-up tendencies occur; however, a mild nose-up pitch may be acceptable if no pitch control force reversal occurs and if no dangerous, unrecoverable, or objectionable flight conditions result.

DISCUSSION

These requirements are essentially similar to the requirements of 3.4.3 through 3.4.3.4.1 of MIL-F-8785B (Reference 10).

The requirements are intended to apply to those conditions where loss of lift requires corrective action by the pilot. Some pertinent examples of the intended application of this requirement are as follows:

(1) conventional "stall" where increases in angle of attack result in loss of aerodynamic lift,

(2) conditions where strong coupling exists between the lateral-directional response and the longitudinal response where, for example, sudden changes in sideslip result in a significant loss of aerodynamic lift, and

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(3) conditions where small perturbations of thrust magnitude or thrust angle controls by the pilot may result in a significant or undesired change in the aircraft aerodynamic lift.

These requirements are not intended to apply to conditions where loss of lift is intentional or normal (i.e., during transfer from aerodynamic lift from a wing to direct lift from lifting engines), provided no undesirable conditions result as a consequence of lift transfer.

Paragraph 3.8.2.1 is based on Section 3.4.2.2 of MIL-F-8785B (Reference 10) but modified to reflect the more general terminology appropriate to V/STOL aircraft. It is intended to ensure that the pilot is given clear and adequate warning of the approach of a loss of aerodynamic lift condition so that he may take appropriate corrective action in time to avoid a dangerous flight condition if it should exist. The requirement is only intended to apply for those conditions where the loss of such lift requires pilot action, and is not intended to be required in conditions where the loss of lift is intentional, or normal, and does not call for special pilot consideration. (For instance, loss of aerodynamic lift from tilt ducts may be compensated for by increased direct lift as ducts are rotated up. Provided no buffet or other undesirable conditions result as a consequence of this loss of aerodynamic lift, no warning need be given the pilot.)

The intent of Section 3.8.2.2 is self-explanatory and is meant to ensure that the condition of lost aerodynamic lift is not catastrophic and recovery can be executed without exceptional pilot skill or strength, before the aircraft loses excessive altitude or builds up excessive speeds.

Section 3.8.2.3 is intended to assure that upon entry into a loss of aerodynamic lift condition, the subsequent recovery of the aircraft is not limited by stalling of aerodynamic surfaces or "masking" of propulsion systems, etc. The control force limitations are reasonable and compatible with the capabilities of a pilot under emergency conditions.

Requirements similar to 3.8.2.2 and 3.8.2.3 are found in AGARD 408A (Reference 46) and RTM-37 (Reference 47) as follows:

AGARD 408A- Section 6.4.7 - "It should be possible to avoid the attainment of the minimum flight speed by normal use of the controls at the onset of the warning. In the event of attaining V_{min} it should be possible to recover by normal use of the controls, with engine power used as necessary, and without excessive loss of altitude or increase in speed. Control forces should not exceed 20 pounds for lateral control, 40 pounds for longitudinal control or 80 pounds for directional control."

RTM 37 - Part of Section 3.7.6.7 - "It shall be possible to prevent stall by normal use of the controls at the onset of stall warning. In the event of a complete stall, it shall be possible to recover (by normal use of the controls with reasonable control forces) without excessive loss of altitude or buildup of speed."

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3.8.3 PILOT-INDUCED OSCILLATIONS

REQUIREMENT

3.8.3 Pilot-induced oscillations. 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 about any control axis or combination of control axes.

DISCUSSION

This requirement is similar to paragraphs 3.2.2.3 and 3.3.3 of MIL-F-8785B (Reference 10). However, the requirement now applies to pilot-induced oscillations (PIO) tendencies about or along any axis, such as vertical PIO.

The requirement is, of necessity, qualitative because of the multitude of factors contributing to PIO susceptibility. Such factors as short-period dynamics, control and force-feel system dynamics, pilot body and body support dynamics, control system friction and free play and structural and aeroelastic modes, etc, have all been identified as sources of PIO problems. Reference 84, for example, presents a description of PIO difficulties related to the use of bobweights in control systems.

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3.8.4 BUFFET

REQUIREMENT

3.8.4 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.

DISCUSSION

This requirement is identical to Section 3.4.6 of MIL-F-8785B (Reference 10). Obviously, objectionable buffet can seriously detract from mission effectiveness both directly by interfering with precision tasks and indirectly through fatigue which reduces crew proficiency. The lack of suitable data precludes a quantitative requirement at this time.

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3.8.5 RELEASE OF STORES

3.8.6 EFFECTS OF ARMAMENT DELIVERY AND SPECIAL EQUIPMENT

REQUIREMENT

3.8.5 Release of stores. 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.

3.8.6 Effects of armament delivery and special equipment. Operation of movable parts such as bomb bay doors, cargo doors, armament pods, refueling devices, rescue equipment, or firing of weapons, release of bombs, extension of lift engines, 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 condition. These requirements shall be met for Levels 1 and 2.

DISCUSSION

These two requirements are identical to paragraphs 3.4.7 and 3.4.8 of MIL-F-8785B (Reference 10). These requirements are necessary although the variety of possibilities makes it necessary to phrase the requirement qualitatively.

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3.8.7 CROSS-COUPLED EFFECTS

REQUIREMENT

3.8.7 Cross-coupled effects. Control inputs or aircraft motions about a given aircraft axis shall not induce objectionable control forces or aircraft motions about any other axis. Specifically, the requirements of 3.8.7.1 and 3.8.7.2 shall apply.

3.8.7.1 Gyroscopic effects. Gyroscopic moments caused by rotating components shall not result in objectionable flight or ground handling characteristics. In flight, the elimination of the cross-coupled response during the maneuvers required to demonstrate compliance with this specification shall require less than 10 percent of the maximum control moment available about the cross-coupling axis for Level 1, and less than 20 percent for Level 2.

3.8.7.2 Inertial and aerodynamic cross-coupling. The application of any cockpit control input necessary to meet any pitch, roll or yaw performance requirement of this specification shall not result in any objectionable aircraft attitudes or angular rates about the axes not under consideration. In addition, undesired altitude changes shall be minimal.

DISCUSSION

Excessive cross-coupling can produce serious degradation of flying qualities both in hovering and in forward flight. It was, therefore, considered appropriate to place the requirements of 3.8.7 in the miscellaneous section to avoid repetition in Sections 3.2 and 3.3.

These requirements are essentially those of Sections 6.2.1 and 6.2.3 of AGARD 408A (Reference 46). Section 3.8.7.2 above has been generalized to restrict objectionable coupling about any of the control axes. The maneuvers necessary to demonstrate compliance with the pitch, roll and yaw maneuver performance requirements of Sections 3.2 and 3.3 are considered realistic for demonstration of compliance with 3.8.7.2.

Various cross-coupling requirements can be found in other documents. For example, MIL-F-8785B (Reference 10) Section 3.4.4, RTM-37 (Reference 47) Sections 3.6.4 and 3.7.4.13, and MIL-H-8501A (Reference 15) Sections 3.3.14, 3.5.11, and 3.5.11.1. All the concepts of these requirements are adequately covered by the present requirements, and there are no apparent conflicts.

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3.8.8 FAILURES

REQUIREMENT

3.8.8 Failures. No single failure of any component or system shall result in dangerous or intolerable flying qualities; Special Failure States (3.1.6.2.1), including certain propulsion failures (3.1.10.3.4 and 3.8.9) are excepted. The crew members concerned shall be provided with immediate and easily interpreted indications whenever failures occur that require or limit any flight-crew action or decision.

3.8.8.1 Transients following failures. The aircraft motions following sudden aircraft system or component failures which might occur during maneuvering flight or unattended trimmed flight 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. 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.

DISCUSSION

The need for these two requirements, taken from MIL-F-8785B (Reference 10), is self-evident.

Certain failures inherently provide warning of their occurrence through aircraft transient motions, characteristic noises, etc. Other failures may be more subtle. For example, a stability augmentation system failure may result only in a loss of authority with no accompanying transient motions. Although the aircraft remains stable for small perturbation motions, subsequent gross maneuvering or encounters with turbulence may result in SAS saturation and severe degradation in flying qualities. Thus, 3.8.8 requires that adequate and timely warning be provided to the crew whenever crew action or decision is required in the event of a failure.

Paragraph 3.8.8.1 is intended to remind the designer that in determining the allowable time delays to corrective action following failures, consideration must be given to the adequacy of the natural or artificial warning and the magnitude of the transient motions.

In addition, consideration should be given to the nature of the control corrections required to effect recovery. That is to say, failures which require unnatural or unusual control corrections will probably require greater time delays since the pilot will not be able to react "instinctively" in these cases.

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3.8.9 CONTROL FOLLOWING LOSS OF THRUST/POWERED LIFT

REQUIREMENT

3.8.9 Control following loss of thrust/powered lift. Thrust, powered lift, or both may be lost from many factors including engine failure, inlet unstart, propeller failure, propeller-drive failure and boundary-layer control system failure. The requirements of 3.8.9.1 through 3.8.9.2.3 apply to loss of thrust or lift on one or more engines, propellers and segments of a powered lift system, caused by all single factors except structural failure of propellers or rotors. The effect of the failure or malperformance of all powered or driven subsystems shall also be included. In demonstrating compliance with 3.8.9.1 through 3.8.9.2.3, a realistic time delay (3.8.8.1) shall be incorporated between the thrust loss and pilot action.

3.8.9.1 Thrust/powered lift loss on the ground. During all takeoffs and landings of the aircraft, it shall be possible without exceptional pilot skill to maintain control following a sudden loss of thrust, powered lift, or both. No failures, beyond those required by 3.8.9, need be considered. For running takeoffs and landings it shall further be possible, after loss, to achieve and maintain a straight path while on the ground without a deviation of more than 30 feet from the path originally intended. Control forces shall not exceed the values for Level 3 in table XIII.

Additional controls such as nosewheel steering if operated by the yaw control, differential braking, and automatic devices which normally operate in the event of a thrust loss may be used. For aborted takeoffs, the requirements apply up to the maximum takeoff speed for the configuration. For continued takeoffs these requirements apply to thrust/powered lift loss at speeds from the lowest refusal speed for the configuration to the maximum takeoff speed, and the requirements of 3.8.9.2 apply once the aircraft is airborne.

3.8.9.2 Thrust/powered lift loss in flight. The aircraft motions following a sudden loss of thrust, powered lift, or both shall be such that dangerous conditions can be avoided by corrective action without undue pilot skill. From considerations of operational requirements, the procuring activity will designate which of the following requirements apply after the loss.

3.8.9.2.1 Continued mission. Aircraft required to proceed from Category C Flight Phases to Category A or B Flight Phases shall meet at least Level 2 flying qualities requirements following the thrust or powered lift loss. Aircraft required to complete Category A Flight Phases shall meet at least Level 2 flying qualities requirements following the thrust or powered lift loss. Aircraft required to terminate Category A Flight Phases and to complete Category B Flight Phases shall meet at least Level 3 flying qualities requirements following the thrust loss.

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3.8.9.2.2 Safe landing. Aircraft required to perform a safe landing shall meet at least Level 3 flying qualities requirements following the thrust loss. If the landing is completed using autorotation, the autorotation requirements of 3.8.10 through 3.8.10.3 must also be satisfied.

3.8.9.2.3 Crew escape. When there are no operational requirements after a loss of thrust/powered lift, the aircraft shall not diverge so rapidly that the ability of the crew to escape is impaired.

DISCUSSION

These paragraphs state the requirements for the specific failure state of loss of thrust or powered lift. Paragraph 3.8.9 is intended to point out that many types of failures can give rise to thrust/powered lift losses. In addition, all side effects of the failure must be considered. For example, a turbine failure may result in complete or partial loss of bleed air for a reaction control system, hydraulic pressure for a boosted control system, and electrical power. Any analysis of the consequences of this turbine failure which did not consider all the related side effects could result in extremely optimistic conclusions.

The intent of paragraph 3.8.9.1 is to assure that deviations of the aircraft's ground track following the failure shall be controllable so that the pilot can either abort or continue the takeoff without leaving the side of the runway. The use of additional controls is permitted provided the pilot is not required to remove his hands or feet from the pitch, roll and yaw cockpit controls to operate them. For example, requiring the pilot to operate a separate tiller bar for nosewheel steering would require his releasing either the throttle or the stick. If the normal crew complement includes a copilot, he may assume the control which the pilot had released. However, this situation is awkward and potentially dangerous. With only a single crew-man, the operation of separate controls may be catastrophic.

Upon becoming airborne following a thrust loss on the ground, or if the failure occurs in flight, the requirements of 3.8.9.2 to 3.8.9.2.3 apply.

The rationale for these requirements parallels the reasoning of Schairer (Reference 122). He argues that there are basically three possible failure situations:

> "Firstly, there are one and two engine combat aircraft which ordinarily fly with only one or two people aboard. It appears acceptable practice to make no provision in the powerplant number or size to provide for engine failure during takeoff or landing. The crew is protected by ejection seats which can operate down to zero speed.

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"Secondly, there are aircraft without stored energy and with too many passengers aboard to consider ejection seats. These aircraft must be designed to survive a failure of any engine at any time during takeoff, landing or enroute. Usually they will be designed to be able to continue to intended destination. The power of the remaining active engines must keep the aircraft in flight and under control and usually without drawing from any of the remaining engines sufficient power to require their removal or replacement.

"Thirdly, there are those helicopters, tilt rotors and tilt wings, which have been designed to hold enough stored energy in their propulsion systems to permit an emergency landing following a powerplant failure. Sometimes the aircraft will be damaged in the emergency landing but ordinarily there will be no personnel injuries. For long range operation there is an unanswered question about the requirement for continued flight to the destination such as might be required over water."

These possibilities, in their most general form, may be summarized as:

1. Crew escape
2. Continue mission
3. Land

It is not the purpose of a flying qualities specification to decide which of these possibilities should be applied, rather it is the job of the specification to decide what flying qualities should be available to the chosen possibility. Therefore, the procuring activity will specify which of 3.8.9.2.1 to 3.8.9.2.3 shall apply.

The most stringent requirement that can be applied is that the aircraft have the capability of continuing its mission following a thrust loss. Under paragraph 3.8.9.2.1 the procuring activity may specify any of three failure characteristics.

First, it may be required that an aircraft have the capability of proceeding from a Category C Flight Phase to Category A or B Flight Phases with Level 2 flying qualities. By definition, this is the minimum level of flying qualities for accomplishment of the mission Flight Phase. For example, imposition of this requirement to a VTOL ground attack aircraft would demand the capability to suffer a thrust loss during a vertical takeoff (Category C) and with the remaining thrust available to cruise to its target (Category B) and make ground attacks (Category A) with at least Level 2 flying qualities.

Secondly, the procuring activity has the option of specifying a less stringent requirement which is intended to assure that an aircraft has the

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capability of terminating its mission and returning safely. That is to say, it may be required that following a thrust loss, the aircraft can safely terminate Category A and complete Category B Flight Phases. This is, in fact, the definition of Level 3 flying qualities.

When there is no necessity to continue the mission, the procuring activity may require that the aircraft have the capability to make a safe landing (3.8.9.2.2). This requirement might be imposed on a personnel transport, for example, where the number of passengers aboard makes it impossible to consider ejection. For an aircraft with little stored energy, this would require the capability to make an unpowered or partial power descent to a landing site (if available) and execute a landing. Rates of descent under these conditions should be such that ordinarily there will be no personal injuries. Sometimes, however, the aircraft may be damaged in the emergency landing. Aircraft with sufficient stored energy, such as rotary wing vehicles, may use their autorotative capability to execute the landing.

Finally, when there is no requirement to land the aircraft, as when the crew complement is such that ejection is feasible, the procuring activity may specify 3.8.9.2.3. No specific Level of handling qualities is required; however, it must be possible to maintain the aircraft's transient motions within the operating limits of the particular crew escape or ejection mechanism provided.

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3.8.10 AUTOROTATION

REQUIREMENT

3.8.10 Autorotation. All aircraft required by the procuring activity to demonstrate an autorotative capability shall meet the following requirements.

3.8.10.1 Autorotation entry. The aircraft shall be capable of entry into autorotation (power off) at all speeds from hover to V_{con}. Following power failure a delay of 1 second prior to pilot corrective action is mandatory, and a delay of 2 seconds is desired. During the delay, no dangerous flight conditions or excessive changes in aircraft attitude or altitude shall occur. Changes in aircraft attitudes shall be considered excessive if they exceed 20 degrees in 2 seconds following complete loss of power with controls fixed. During the transition from powered flight to autorotative flight, the control forces shall not exceed the Level 2 maximums of table XIII, and 20 percent of the nominal control power must remain for maneuvering.

3.8.10.2 Autorotative descent and landing. All aircraft with an autorotative capability requirement shall be capable of descending and landing (power off) safely. The pitch, roll and yaw dynamic stability requirements of this specification shall apply in autorotation at any speed. Touchdown speeds and landing zone environment will be specified by the procuring activity.

DISCUSSION

Requirements similar to 3.8.10 are found in MIL-H-8501A and RTM-37 (References 15 and 47 respectively).

It is recognized that during the entry into a controlled autorotation following power failure, some altitude will generally be lost. When the initial altitude of the vehicle is insufficient to permit recovery before impacting the ground, the aircraft is considered to be flying within the critical height-velocity regime. The requirements for power failure characteristics within the critical height-velocity regime are stated in paragraph 3.1.10.3.4 of Reference 1.

The specification of a mandatory delay time before pilot control corrections is intended to assure that the rate of divergence of the aircraft motions following power failure is compatible with the pilot's capability to detect and diagnose the failure and make appropriate control corrections. Obviously, the maximum delay time which can be demonstrated is a function of the vehicle's configuration and flight regime, the adequacy of natural or artificial failure warnings and the physiology of the pilot.

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MIL-H-8501A (Reference 15) requires that it be possible to transition safely to autorotation when the collective pitch control correction has been delayed for 2 seconds. No minimum time delay is specified for the other controls (roll, pitch or yaw). Simulated power failure testing of a high performance helicopter, the AH-1G Huey Cobra, documented in Reference 91, indicate that, for this vehicle, the collective delay times are related to the degree of cyclic flare used in the recovery. The maximum collective delay time reported was 2.2 seconds following a throttle chop at 163 knots CAS. However, for this case, the cyclic flare (pitch control) was initiated at about 0.6 seconds while a yaw control correction was made almost immediately following the power loss. It seems, therefore, unrealistic to limit the demonstration of time delay to the collective control only, when much shorter reaction times are required to control the other degrees of freedom of the vehicle.

A more realistic approach has been taken in RTM-37 (Reference 47) in which a reaction delay time of 2 seconds for all controls is considered desirable while 1 second is mandatory for demonstration of compliance. A similar requirement has been adopted for the V/STOL Specification (Reference 1).

Little or no data base exists to substantiate this requirement since military helicopters to date have generally been procured and tested against MIL-H-8501A. However, under operational conditions, the transients associated with power failures may be much more critical than can be effectively simulated by throttle chops as in flight testing. It is considered, therefore, that a one-second mandatory time delay is not unrealistic and may, under certain circumstances, be too lenient.

MIL-H-8501A restricts transient attitude changes to 10 degrees within the first 2 seconds while yaw attitude excursions of 20 degrees are allowed for speeds less than the speed for best climbs. The requirement of RTM-37 is similar. The V/STOL Specification is less restrictive and allows attitude changes up to 20 degrees from hover to V_{con}.

Touchdown speeds have not been specified since they are directly related to the use of wheels or skids and wind conditions. In addition, the procuring activity should specify the landing zone environment (i.e., wind conditions, level paved surface, carrier deck, sea state, etc), since they are operational considerations. The handling quality requirement is essentially that the aircraft flying qualities shall never be worse than Level 3 for autorotative (power off) landing in the operational environments envisioned by the procuring activity.

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3.8.11 VIBRATION CHARACTERISTICS

REQUIREMENT

3.8.11 Vibration characteristics. Throughout the Operational Flight Envelope, the aircraft shall be free of objectionable shake, vibration, or roughness. In addition, throughout the Operational Flight Envelope the aircraft shall not exhibit mechanical or aeroelastic instabilities (i.e., ground resonance, flutter, etc.) that degrade the flying qualities.

DISCUSSION

This requirement is intended to draw attention to the adverse effects of mechanical and aeroelastic vibration on flying qualities. MIL-H-8501A (Reference 15) places quantitative limits on vibratory acceleration and displacement both at the cockpit controls and at pilot, crew, passenger and litter stations. There appears to be little data to substantiate these requirements. Considering the great variety of possible V/STOL concepts, it is felt that a quantitative vibration requirement is not feasible at the present time.

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4. QUALITY ASSURANCE PROVISIONS

REQUIREMENT

4.1 Determination. Quality assurance shall be determined through:

Analysis
Simulation
Ground test
Flight test.

The contract end item specification for each procurement will delineate, for each requirement of section 3, which of these methods shall be used. In order to restrict the number of design and test conditions, representative flight conditions, configurations, external store complements, loadings, etc., shall be determined for detailed investigation. The selected design points must be sufficient to allow extrapolation to the other conditions at which the requirements apply. The required failure analyses shall be thorough, excepting only approved Special Failure States (3.1.6.2.1).

4.2 Interpretation of qualitative requirements. Requirements which are not stated in terms of quantitative values of a particular stability or control parameter are to be interpreted with due regard to the intent of the Level definitions of 1.5. Final determination of compliance with such qualitative requirements will be made by the procuring activity through flight test or other suitable means.

DISCUSSION

The philosophy underlying the V/STOL Specification is that the requirements should apply under those conditions in which the aircraft operates. The requirements therefore apply in those flight regimes, with the loadings, external store combinations, and geometric configurations required by the aircraft's missions; plus failure considerations. It is recognized, however, that the number of design or flight test points that can be examined in detail is generally severely limited by both time and money; so guidance should be provided to limit the magnitude of the design task or flight test program.

In MIL-F-8785B (Reference 10), a significant amount of guidance is provided in choosing the flight conditions which should be tested. This is amplified by the discussion in the BIUG (Reference 84), page 463.

To establish guidelines it is necessary to know:

- which factors, such as speed, inertia and aerodynamic characteristics, influence the statics, dynamics and response
- how the important factors change throughout the flight envelope.

For conventional aircraft there is a good deal of experience on

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which to base prognostication; also the aerodynamic characteristics vary in reasonably predictable ways, e.g., nondimensional stability and control derivatives are constant (except for Mach number variation) as speed and altitude are changed (there is little or no power effect).

On this basis, for conventional aircraft it is possible to reason that for example (Reference 84, page 471):

$\omega_{n_{SP}}$ will be proportional to $V_T\sqrt{\sigma} = V_e$

ζ_{SP} will be proportional to $\sqrt{\sigma}$

$1/\tau_R$ will be proportional to $V_T\sigma$

n/α will be proportional to $V_T^2\sigma = V_e^2$

The effect of load factor on the aerodynamic characteristics is deduced to be primarily that caused by the increased angle of attack. This, in turn, is inversely proportional to the changes which occur with increased speed.

Such assumptions as those outlined above could be quite misleading if applied to a V/STOL:

- Speed changes.
  If a V/STOL increases speed it may be necessary to change configuration (e.g., using tilt angle). This will obviously cause significant nondimensional derivative changes. If the speed is increased at a fixed configuration there will almost certainly be large power effects which will modify the nondimensional derivatives.

- Altitude changes.
  As altitude is increased at a given speed and fixed configuration the trends in stability and control characteristics are likely to be similar to those for conventional aircraft.

- Effect of normal load factor.
  This is likely to result in the most complex changes since normal load factor will be produced by combinations of power change and angle of attack change. Changes in power can influence all the longitudinal and lateral-directional stability and control derivatives. Changes in angle of attack could be complex if, as is likely, they exceed ranges of linearity. Certainly large perturbations should be considered in this regard.

As a result of these complications, which make it appear that generalized trends will not be predictable for all possible V/STOL configurations, it is

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up to the designer to choose the representative conditions for flight testing. He will very likely have to make use of computer simulations to determine the critical conditions which should be investigated by flight test, and also to show that interpolation between flight test points is valid.

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5. PREPARATION FOR DELIVERY

5.1 General. Section 5 is not applicable to this specification.

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6. NOTES

DISCUSSION

This section provides definitions of the symbols used in the specification and also gives some notes of clarification on various concepts such as application of Levels.

Because of the large number of definitions, paragraph 6.2 has been divided into subsections for clarity. Most of the definitions should be self-explanatory; but some of the more complex parameters, such as the roll-sideslip coupling parameters, are explained more thoroughly in the discussions of the requirements to which they apply.

For completeness, Section 6 of Reference 1 is given below. No further discussion seems necessary because the notes themselves are explanatory in nature.

6.1 Intended use. This specification contains the flying qualities requirements for military piloted V/STOL aircraft operating at speeds up to V_{con} and shall form one of the bases for determination, by the procuring activity, of aircraft acceptability. The specification shall serve as design requirements and as criteria for use in stability and control calculations, analysis of wind-tunnel test results, flying qualities simulation tests, and flight testing and evaluation. To the extent possible, this specification should be met by providing an inherently good basic airframe. Where that is not feasible, or where inordinate penalties would result, a mechanism is provided herein to assure that the flight safety, flying qualities and reliability aspects of dependence on stability augmentation and other forms of system complication will be considered fully.

6.2 Definitions. Terms and symbols used throughout this specification are defined in the following subparagraphs.

6.2.1 General

s	-	Laplace transform variable
MSL	-	mean sea level
Aircraft Normal States	-	the nomenclature and format of table XIV shall be used in defining the Aircraft Normal States (3.1.6.1)
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

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h_{max} - maximum service altitude (defined in 3.1.8.4)

ho_{max} - maximum operational altitude (3.1.7)

ho_{min} - minimum operational altitude (3.1.7)

c.g. - aircraft center of gravity

hover - to remain stationary relative to either the air mass or a point on the ground as specified in the applicable requirement

6.2.2 Speeds

refusal speed - the maximum speed to which the aircraft can accelerate and then stop in the available runway length

TAS - true airspeed

V - airspeed along the flight path

$V_{max}(X)$, $V_{min}(X)$ - short-hand notation for the speeds V_{max}, V_{min} for a given configuration, weight, center-of-gravity position, and external store combination associated with Flight Phase X

V_{end} - speed for maximum endurance

V_{range} - speed for maximum range in zero wind conditions

V_{MAT} - high speed, level flight, maximum augmented thrust

V_{max} - maximum service speed (defined in 3.1.8.1)

V_{min} - minimum service speed (defined in 3.1.8.2)

V_{omax} - maximum operational speed (3.1.7)

V_{omin} - minimum operational speed (3.1.7)

V_{con} - the speed which establishes the upper limit of applicability of the requirements of this specification and the lower limit of applicability of the requirements of MIL-F-8785B. No more precise definition of V_{con} will be attempted as it is assumed that V_{con} will be chosen by the contractor subject to approval by the procuring activity. Factors to be considered in the selection of V_{con} are discussed in the Background Information and User's Guide (BIUG); see 6.7.

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6.2.3 Thrust and power

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

T/W - the ratio formed by dividing the thrust available by the aircraft's weight

6.2.4 Control parameters

Pitch, Roll, Yaw Controls - the stick or wheel and rudder pedals manipulated in the cockpit by the pilot to produce pitching moments, rolling moments and yawing moments, respectively

Thrust Magnitude Control - the lever which is manipulated in the cockpit by the pilot to produce changes in the magnitude of the thrust vector

Thrust Angle Control - the lever or switch manipulated by the pilot to produce changes in the thrust angle, for example, wing-tilt angle control

Pitch Control Force - component of applied force, exerted by the pilot on 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 or wheel and the stick or control column pivot

Roll Control Force - 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 Force - difference of push-force components of forces exerted by the pilot on the rudder 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 rudder pedals

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Thrust Magnitude Control Force	-	component of applied force, exerted by the pilot on the cockpit control, in or parallel to the plane of symmetry acting at the center of the lever grip in a direction perpendicular to a line between the center of the lever grip and the lever column pivot
Control Surface	-	a surface or device which is positioned by a cockpit control or by stability augmentation, and which produces aerodynamic or jet-reaction type forces in such a manner as to control the forces, moments, or both, on the aircraft. As used in this specification, the pitch control surface, roll control surface, and yaw control surface are the control surfaces or devices which are controlled by the pitch, roll and yaw controls respectively
Nominal Control Moment	-	one-half of the total control moment change available to the pilot using only the pitch, roll or yaw control at the given flight condition
Control Power	-	the angular or linear acceleration available to the pilot with full cockpit control displacement from the given trim condition

6.2.5 Longitudinal parameters

n	-	normal load factor
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 (defined in 3.1.8.5)
$n(+), n(-)$	-	for a given altitude, the upper and lower boundaries of n in the V-n diagrams depicting the Service Flight Envelope
$n_{o_{max}}, n_{o_{min}}$	-	maximum and minimum operational load factors (3.1.7)
$n_o(+), n_o(-)$	-	for a given altitude, the upper and lower boundaries of n in the V-n diagrams depicting the Operational Flight Envelope (see Figures 6 and 7)
n/α	-	the steady-state normal acceleration change per unit change in angle of attack for an incremental pitch control deflection at constant speed

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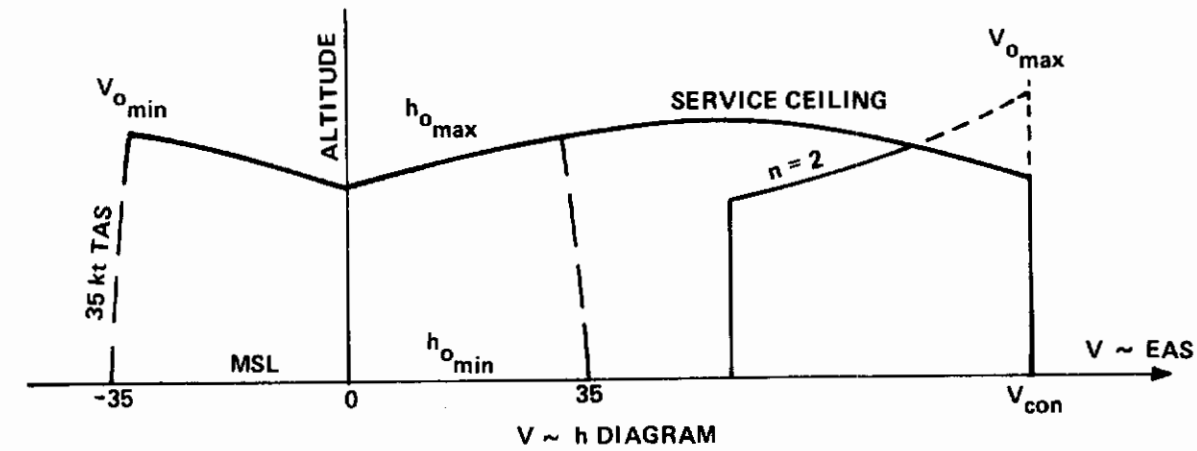
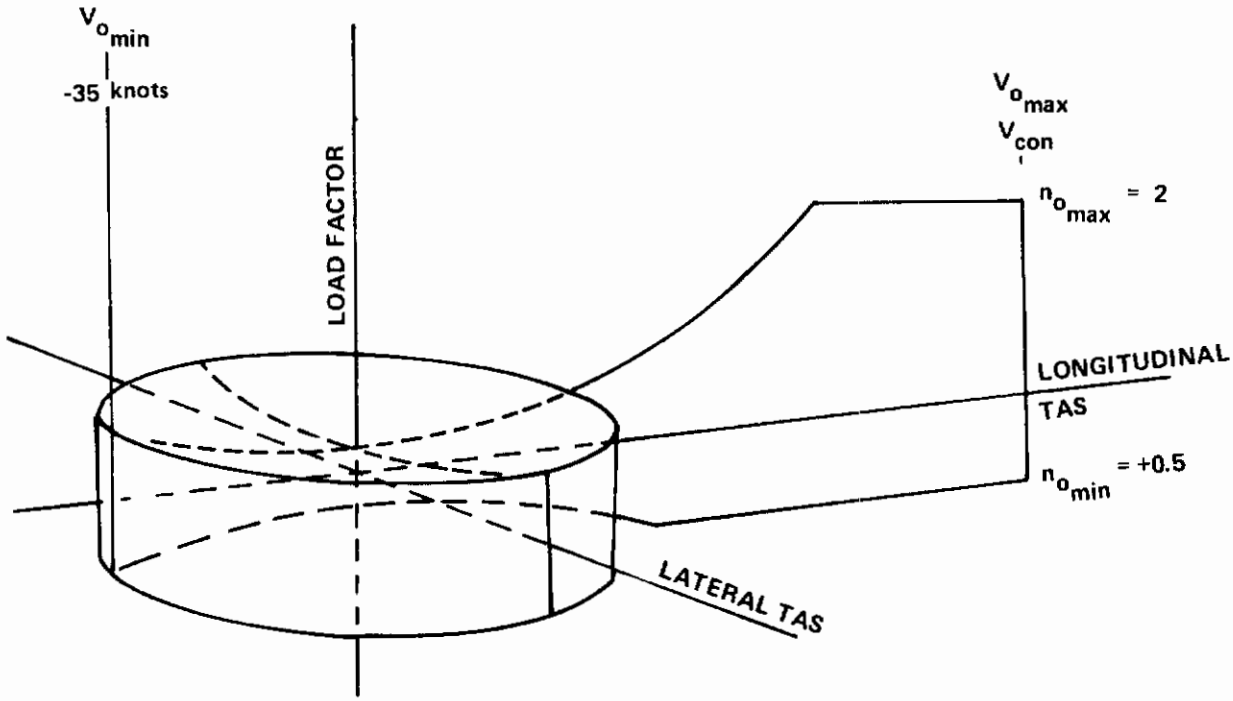


V ~ h DIAGRAM



V ~ n DIAGRAM (CONSTANT ALTITUDE)

Figure 6 TYPICAL OPERATIONAL FLIGHT ENVELOPE FOR FLIGHT PHASE CATEGORY B, NON-TERMINAL TRANSITION (NT), BASED ON LIMITS OF TABLE I

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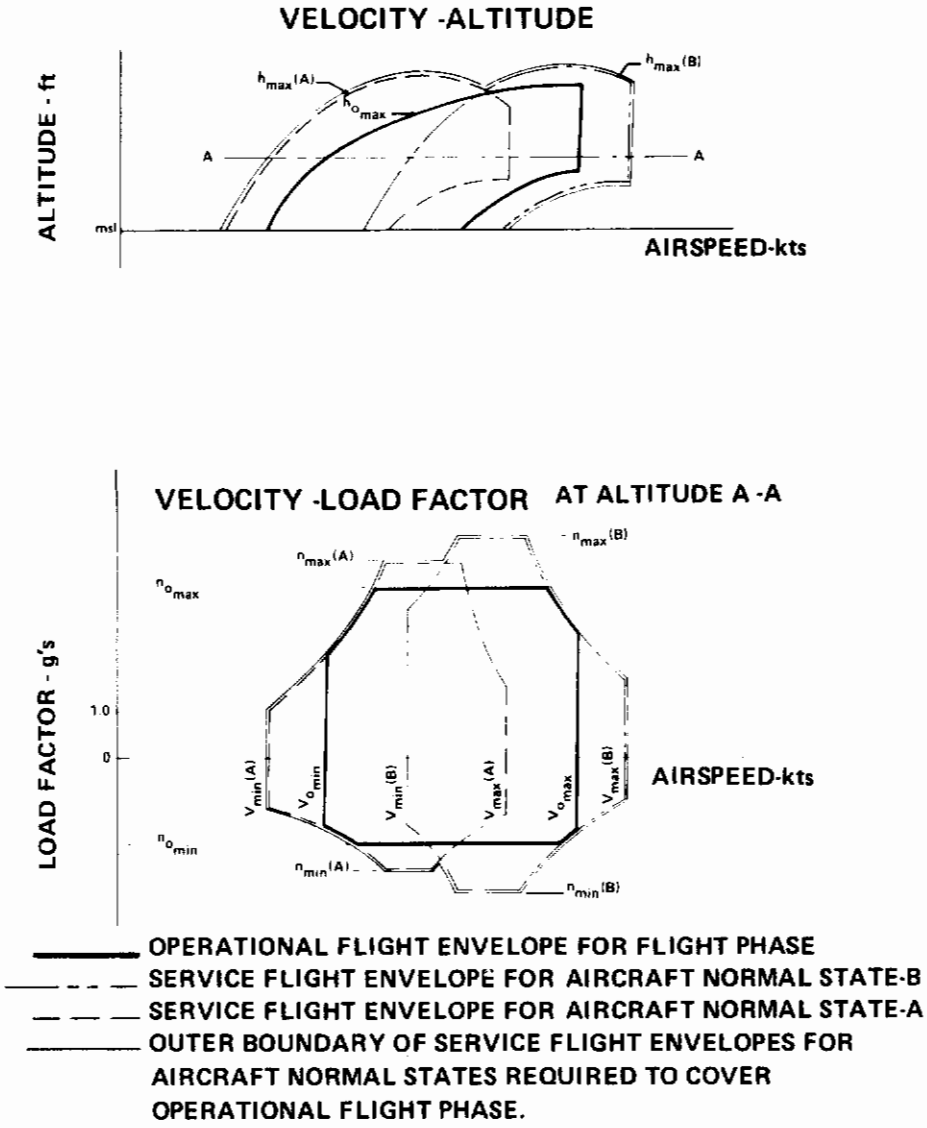

Figure 7 TYPICAL RELATIONSHIP BETWEEN OPERATIONAL ENVELOPE AND SERVICE FLIGHT ENVELOPE FOR A GIVEN FLIGHT PHASE REQUIRING TWO NORMAL STATES

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6.2.6 Lateral-directional parameters

δ_{AS} - roll control displacement

τ_R - first-order roll mode time constant

δ_{RP} - yaw control displacement

ω_{n_d} - undamped natural frequency of the Dutch roll oscillation; $\omega_{n_d}^2$ greater than zero is indicative of positive weathercock stability (3.3.7.1)

ζ_d - damping ratio of the Dutch roll oscillation

T_d - damped period of the Dutch roll, $T_d = \frac{2\pi}{\omega_{n_d}\sqrt{1-\zeta_d^2}}$

ϕ - bank angle

ϕ_1, ϕ_2, ϕ_3 - bank angles at the first, second and third peaks, respectively (figures 8 and 9)

p - roll rate

$\frac{\phi_{osc}}{\phi_{AV}}$ - a measure of the ratio of the oscillatory component of bank angle to the average component of bank angle following an impulse roll control command with yaw control free:

$$\zeta_d \leq 0.2 : \frac{\phi_{osc}}{\phi_{AV}} = \frac{\phi_1 + \phi_3 - 2\phi_2}{\phi_1 + \phi_3 + 2\phi_2}$$

$$\zeta_d > 0.2 : \frac{\phi_{osc}}{\phi_{AV}} = \frac{\phi_1 - \phi_2}{\phi_1 + \phi_2}$$

ϕ_{AV} - $\zeta_d \leq 0.2 : \phi_{AV} = \frac{1}{4}(\phi_1 + \phi_3 + 2\phi_2)$

$\zeta_d > 0.2 : \phi_{AV} = \frac{1}{2}(\phi_1 + \phi_2)$

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.

$\Delta\beta$ - the maximum change in sideslip following an abrupt roll control pulse command within time $t_{\Delta\beta}$; where $t_{\Delta\beta}$ is the lesser of 6 seconds or one-half the Dutch roll period, and is measured from a point halfway through the duration of the pulse command (figures 8 and 9)

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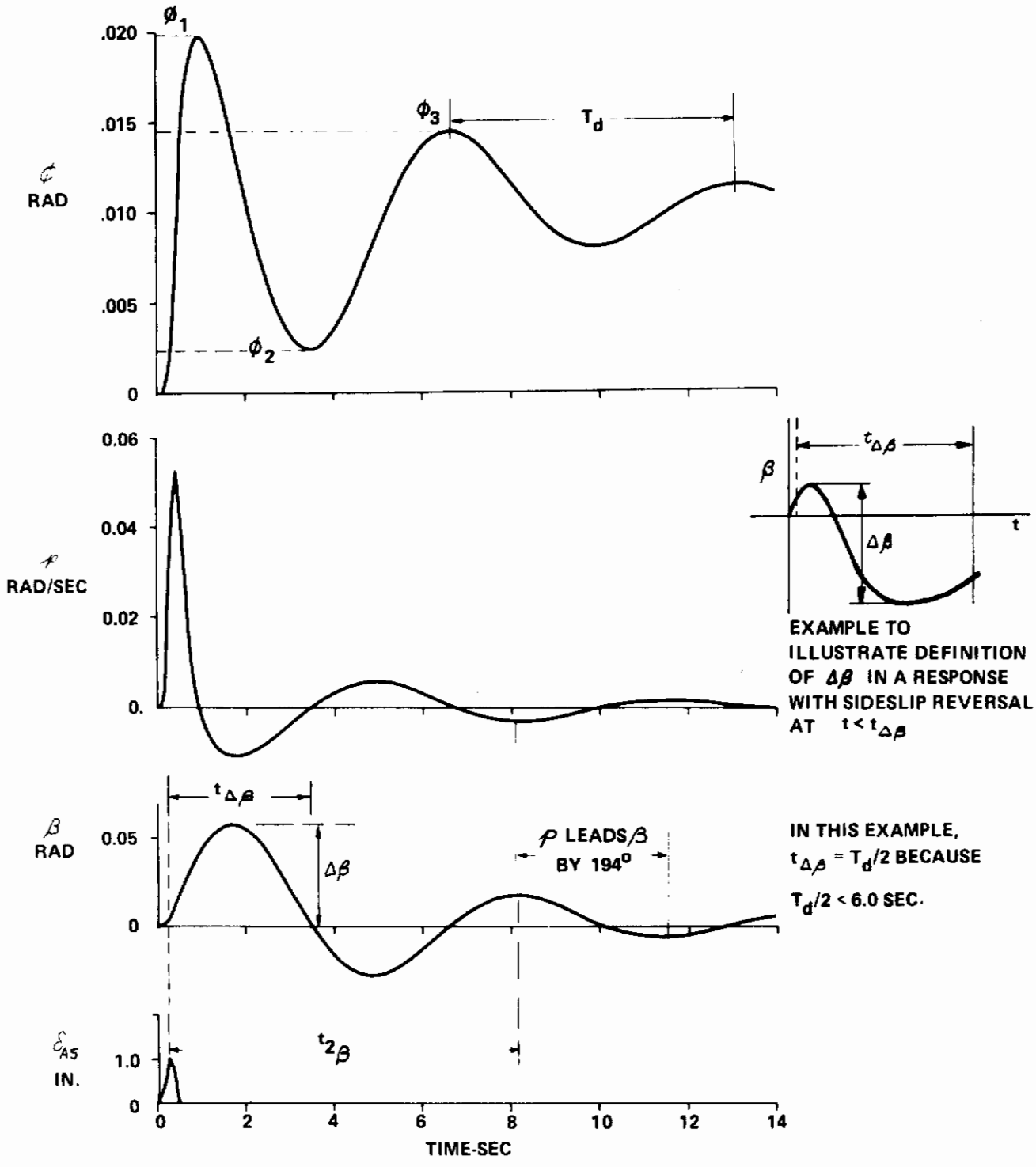

Figure 8 ROLL-SIDESLIP COUPLING PARAMETERS
RIGHT ROLLS

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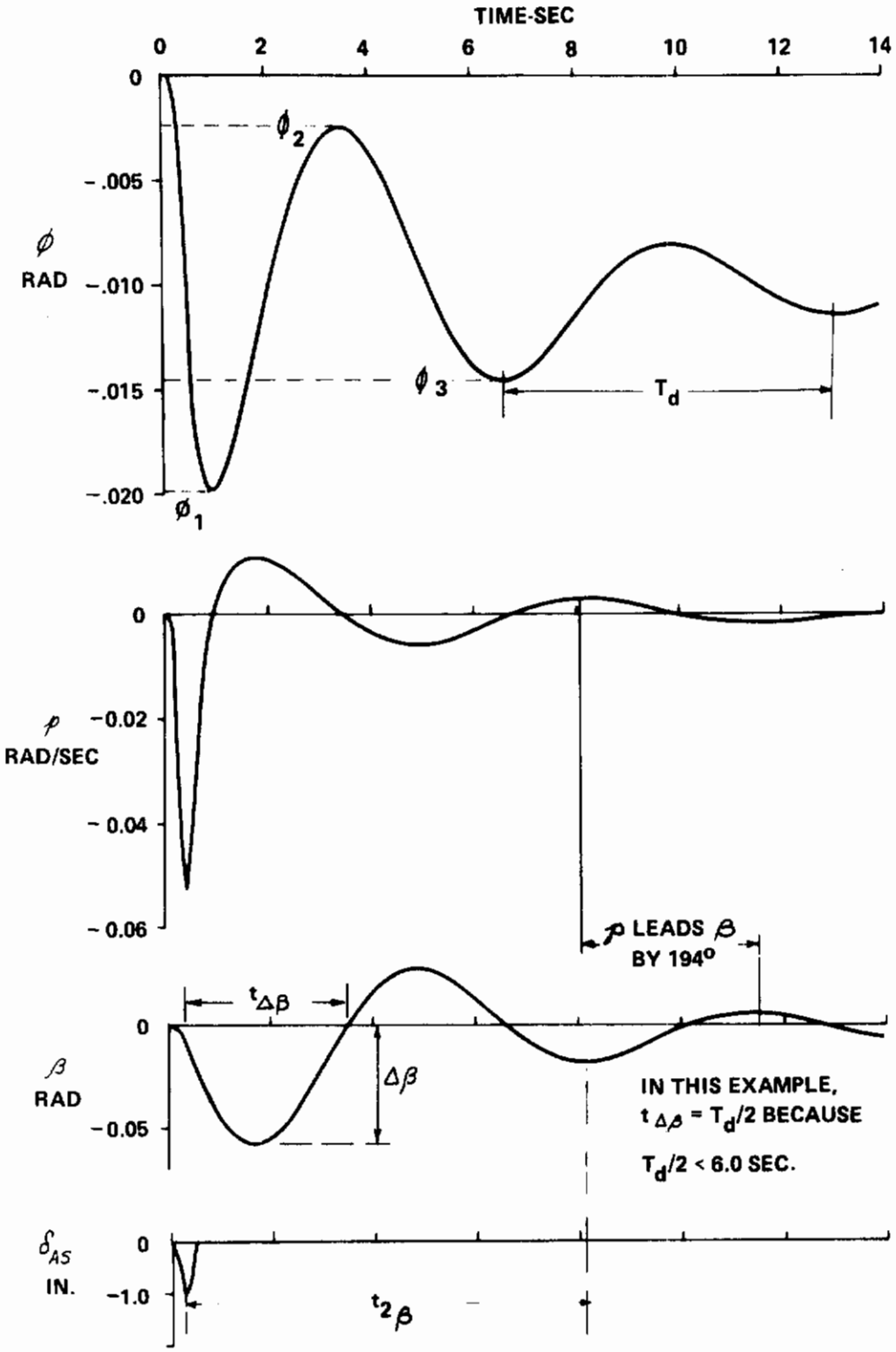

Figure 9 ROLL-SIDESLIP COUPLING PARAMETERS LEFT ROLLS

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t_{n_β} - time for the Dutch roll oscillation in the sideslip response to reach the n^{th} local maximum for a right pulse roll control command, or the n^{th} local minimum for a left command (figures 8 and 9). The control shall be moved as abruptly as practical and, for purposes of this definition, 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

$$\psi_\beta = \frac{-360}{T_d} t_{n_\beta} + (n-1)360 \quad \text{degrees}$$

with n as in t_{n_β} above

$|\phi/\beta|_d$ - 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 given in figure 8 for right rolls and figure 9 for left rolls. It should be noted that since ψ_β is the phase angle of the Dutch roll component of sideslip, care must be taken to select a peak far enough downstream that the position of the peak is not influenced by the roll mode. In practice, peaks occurring one or two roll mode time constants after the roll control 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.

6.3 Gain scheduling. Changes of mechanical gearings and stability augmentation gains in the flight control system are sometimes accomplished by scheduling the changes as a function of the settings of thrust lift angle or 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 aircraft in trim.

6.4 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.

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6.5 Application of Levels. Part of the intent of 3.1.10 is to ensure that the probability of encountering significantly degraded flying qualities because of component or subsystem failures is small. For example, the probability of encountering very degraded flying qualities (Level 3) must be less than specified values per flight.

6.5.1 Theoretical compliance. To determine theoretical compliance with the requirements of 3.1.10.2, the following steps must be performed:

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

b. Define the longest flight duration to be encountered during operational missions (3.1.1).

c. Determine the probability of encountering various Aircraft Failure States, per flight, based on the above flight duration (3.1.10.2).

d. Determine the degree of flying qualities degradation associated with each Aircraft Failure State in terms of Levels as defined in the specific requirements.

e. Determine the most critical Aircraft Failure States (assuming the failures are present at whichever point in the Flight Envelope being considered is most critical in a flying qualities sense), and compute the total probability of encountering Level 2 flying qualities in the Operational Flight Envelope due to equipment failures. Likewise, compute the probability of encountering Level 3 flying qualities in the Operational Flight Envelope, etc.

f. Compare the computed values above with the requirements in 3.1.10.2 and 3.1.10.3.

If the requirements are not met, the designer must consider alternate courses such as:

(a) Improve the aircraft flying qualities associated with the more probable Failure States, or

(b) Reduce the probability of encountering the more probable Failure States through equipment redesign, redundancy, etc.

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

6.5.2 Level definitions. To determine the degradation in flying qualities parameters for a given Aircraft Failure State the following definitions are provided:

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a. Level 1 is better than or equal to the Level 1 boundary, or number, given in section 3.

b. Level 2 is worse than Level 1, but no worse than the Level 2 boundary, or number.

c. Level 3 is worse than Level 2, but no worse than the Level 3 boundary, or number.

When a given boundary, or number, is identified as Level 1 and Level 2, this means that flying qualities outside the boundary conditions shown, or worse than the number given, are at best Level 3 flying qualities . Also, since Level 1 and Level 2 requirements are the same, flying qualities must be within this common boundary, or number, in both the Operational and Service Flight Envelopes for Aircraft Normal States (3.1.10.1). Aircraft Failure States that do not degrade flying qualities beyond this common boundary are not considered in meeting the requirements of 3.1.10.2. Aircraft Failure States that represent degradations to Level 3, must, however, be included in the computation of the probability of encountering Level 3 degradations in both the Operational and Service Flight Envelopes. Again, degradation beyond the Level 3 boundary is not permitted, except for Special Failure States.

6.5.3 Computational assumptions. Assumptions a and b of 3.1.10.2 are somewhat conservative, but they simplify the required computations in 3.1.10.2 and provide a set of workable ground rules for theoretical predictions. The reasons for these assumptions are:

a. "...components and systems are....operating for a time period per flight equal to the longest operational mission time...." Since most component failure data are in terms of failures per flight hour, even though continuous operation may not be typical (e.g., yaw damper on during supersonic flight only), failure probabilities must be predicted on a per flight basis using a "typical" total flight time. The "longest operational mission time" as "typical" is a natural result. If acceptance cycles-to-failure reliability data are available (MIL-STD-756), these data may be used for prediction purposes based on maximum cycles per operational mission, subject to procuring activity approval. Also, finite wearout life components, such as engines at maximum take-off thrust, may be considered as exceptions and failure calculations shall be based on maximum normal operating time per flight in these cases, again subject to procuring activity approval. In any event, compliance with the requirements of 3.1.10.2, as determined in accordance with section 4, is based on the probability of encounter per flight.

b. "....failure is assumed to be present at whichever point....is most critical...." This assumption is in keeping with the requirements of 3.1.6.2 regarding Flight Phases subsequent to the actual failure in question. In cases that are unrealistic from the operational standpoint, the specific Aircraft Failure States might fall in the Aircraft Special Failure State classification (3.1.6.2.1).

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6.6 Superseding data. This specification supersedes Military Specification MIL-H-8501A for U. S. Air Force use.

6.7 Related documents. The documents listed below, while they do not form a part of this specification, are so closely related to it that their contents should be taken into account in any application of this specification.

SPECIFICATIONS

Military

MIL-C-5011	Charts; Standard Aircraft Characteristics and Performance, Piloted Aircraft
MIL-S-5711	Structural Criteria, Piloted Airplanes, Structural Tests, --Flight
MIL-M-7700	Manual, Flight
MIL-G-38478	General Requirements for Angle of Attack Based Systems
MIL-S-25015	Spinning Requirements for Airplanes

STANDARDS

MIL-STD-882	System Safety Program for Systems and Associated Subsystems, and Equipment; Requirements for.

PUBLICATIONS

AFSC Design Handbook, Series 1-0 and 2-0

Background Information and User Guide for a Military Specification for V/STOL Flying Qualities. AFFDL-TR-70-88 (March 1971)

Custodians:

Army -

Navy -

Air Force - 11

Preparing Activity:

Air Force - 11

Project No. 1500-0086

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13. ABSTRACT

This document is published in support of Military Specification MIL-F-83300, "Flying Qualities of Piloted V/STOL Aircraft".

The Specification was compiled after an extensive literature review and many meetings and discussions with personnel from essentially all concerned civilian and governmental organizations. This report attempts to explain the concept and philosophy underlying the V/STOL Specification and to present some of the data and arguments upon which the requirements were based.

The document should also serve as a summary of the state of the V/STOL flying qualities art as determined from flight test, simulation, analysis, and theory.

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14 KEY WORDS	LINK A		LINK B		LINK C	
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