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BACKGROUND INFORMATION AND USER GUIDE FOR MIL-F-8785B(ASG), "MILITARY SPECIFICATION – FLYING QUALITIES OF PILOTED AIRPLANES"

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TECHNICAL REPORT AFFDL-TR-69-72

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BACKGROUND INFORMATION AND USER GUIDE FOR MIL-F-8785B(ASG), "MILITARY SPECIFICATION - FLYING QUALITIES OF PILOTED AIRPLANES"

C. R. Chalk, T. P. Neal, T. M. Harris, F. E. Pritchard
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FOREWORD

This document was drafted for the United States Air Force by the Cornell Aeronautical Laboratory, Inc., Buffalo, New York, in partial fulfillment of Contract AF 33(615)-3294, Part II, Paragraph A,5 (Item IV), b, (1), (2) under Project 8219, Task 821905-015. Also prepared under this contract was Recommendations for Revision of MIL-F-8785, dated May 1968, which was modified by the Air Force and Navy to form MIL-F-8785B (ASG), published in 1969. The contracted work was performed by CAL's Flight Research Department under the sponsorship of the Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio. The Air Force Systems Command's Aeronautical Systems Division was heavily involved throughout the time span of the work. The Naval Air Systems Command, the Air Force Flight Test Center, and the Aerospace Research Pilot School were also active in the project.

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This document is published to aid in interpreting specification MIL-F-8785B. The Background Information and User Guide is not a contractual document.

The authors released this document for publication in June 1969. Revisions to this document will be issued as MIL-F-8785B is amended in the future. Comments, suggestions and requests for copies of the Background Information and User Guide should be addressed to AFFDL (FDCC), Wright-Patterson AFB, Ohio 45433.

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ABSTRACT

This document is published in support of Military Specification MIL-F-8785B, "Flying Qualities of Piloted Airplanes." It was compiled after an extensive literature review and many meetings and discussions with personnel from essentially all concerned civilian and governmental organizations. The primary purpose is to explain the concept and philosophy underlying MIL-F-8785B and to present some of the data and arguments upon which the requirements were based.

A secondary purpose is to present what are believed to be the important governing variables in the field of flying qualities and to define their significance and relationship to each other. The significance of such mission-oriented factors as airplane class, flight phase, flight condition, loading and configuration is discussed, as is the treatment of failure states. The document should also, to a degree, serve as a summary of the state of the flying qualities art as determined from operational experience, flight test, experiment, analysis and theory.

TABLE OF CONTENTS
(Asterisks Indicate General Discussions)

Section Page

I INTRODUCTION 1

II HISTORICAL DEVELOPMENT 2

III STATEMENT AND DISCUSSION OF REQUIREMENTS 7

*1. SCOPE AND CLASSIFICATION 7

1.1 Scope 8

1.2 Application 8

1.3 Classification of airplanes 10

1.3.1 Land- or carrier-based designation 10

1.4 Flight Phase Categories 14

1.5 Levels of flying qualities 17

*2. APPLICABLE DOCUMENTS 21

2.1 21

*3. REQUIREMENTS 23

*3.1 General requirements 23

3.1.1 Operational missions 23
3.1.2 Loadings 25
3.1.3 Moments of inertia 25
3.1.4 External stores 25
3.1.5 Configurations 27
3.1.6 State of the airplane 28
3.1.6.1 Airplane Normal States 28
3.1.6.2 Airplane Failure States 28
3.1.6.2.1 Airplane Special Failure States 28
*3.1.7, 3.1.8, 3.1.9 General Discussion 32
3.1.7 Operational Flight Envelopes 35
3.1.8 Service Flight Envelopes 35
3.1.8.1 Maximum service speed 38
3.1.8.2 Minimum service speed 38
3.1.8.3 Maximum service altitude 38
3.1.8.4 Service load factors 39
3.1.9 Permissible Flight Envelopes 40

TABLE OF CONTENTS (Cont.)
(Asterisks Indicate General Discussions)

Section | Page

III (continued)

3.1.9.1 Maximum permissible speed 40
3.1.9.2 Minimum permissible speed 40
3.1.9.2.1 Minimum permissible speed other than stall speed 40
3.1.10 Application of Levels 42
3.1.10.1 Requirements for Airplane Normal States . 42
3.1.10.2 Requirements for Airplane Failure States. 42
3.1.10.2.1 Requirements for specific failures 43
3.1.10.3 Exceptions 43
3.1.10.3.1 Ground operations and terminal Flight Phases 43
3.1.10.3.2 When Levels are not specified. 43
3.1.10.3.3 Flight outside the Service Flight Envelope 43
*3.2 Longitudinal flying qualities 53
*3.2.1 Longitudinal stability with respect to speed . . . 54
3.2.1.1 Longitudinal static stability 55
3.2.1.1.1 Relaxation in transonic flight 65
3.2.1.1.2 Elevator control force variations during rapid speed changes 67
3.2.1.2 Phugoid stability 68
3.2.1.3 Flight path stability 80
*3.2.2 Longitudinal maneuvering characteristics 89
3.2.2.1 Short-period response 90
3.2.2.1.1 Short-period frequency and acceleration sensitivity . . . 92
3.2.2.1.2 Short-period damping 116
3.2.2.1.3 Residual oscillations 118
3.2.2.2 Control feel and stability in maneuvering flight 119
3.2.2.2.1 Control forces in maneuvering flight 120
3.2.2.2.2 Control motions in maneuvering flight 133
3.2.2.3 Longitudinal pilot-induced oscillations .135
3.2.2.3.1 Transient control forces . . .157
*3.2.3 Longitudinal control 164
3.2.3.1 Longitudinal control in unaccelerated flight 165

TABLE OF CONTENTS (Cont.)
(Asterisks Indicate General Discussions)

Section | Page

III (continued)

3.2.3.2 Longitudinal control in maneuvering flight 166
3.2.3.3 Longitudinal control in takeoff 167
3.2.3.3.1 Longitudinal control in catapult takeoff 169
3.2.3.3.2 Longitudinal control force and travel in takeoff 170
3.2.3.4 Longitudinal control in landing 172
3.2.3.4.1 Longitudinal control forces in landing 173
3.2.3.5 Longitudinal control forces in dives - Service Flight Envelope 174
3.2.3.6 Longitudinal control forces in dives - Permissible Flight Envelope 175
3.2.3.7 Longitudinal control in sideslips . . . 176

*3.3 Lateral-directional flying qualities 177

*3.3.1 Lateral-directional mode characteristics 181
3.3.1.1 Lateral-directional oscillations (Dutch roll) 182
3.3.1.2 Roll mode 206
3.3.1.3 Spiral stability 220
3.3.1.4 Coupled roll-spiral oscillation 224
3.3.2 Lateral-directional dynamic response characteristics 227
3.3.2.1 Lateral-directional response to atmospheric disturbances 235
3.3.2.2 Roll rate oscillations 236
3.3.2.2.1 Additional roll rate requirement for small inputs. 236
3.3.2.3 Bank angle oscillations 271
3.3.2.4 Sideslip excursions 272
3.3.2.4.1 Additional sideslip requirement for small inputs 272
3.3.2.5 Control of sideslip in rolls 289
3.3.2.6 Turn coordination 290
3.3.3 Pilot-induced oscillations 291
3.3.4 Roll control effectiveness 292
3.3.4.1 Roll performance for Class IV airplanes. 292
3.3.4.1.1 Air-to-air combat 292
3.3.4.1.2 Ground attack with external stores 292

TABLE OF CONTENTS (Cont.)
(Asterisks Indicate General Discussions)

Section | Page

III (continued)

3.3.4.1.3 Roll rate characteristics for ground attack 293
3.3.4.1.4 Roll response 293
3.3.4.2 Aileron control forces 323
3.3.4.3 Linearity of roll response 342
3.3.4.4 Wheel control throw 343
3.3.4.5 Rudder-pedal-induced rolls 344
3.3.5 Directional control characteristics 345
3.3.5.1 Directional control with speed change. . 346
3.3.5.1.1 Directional control with asymmetric loading 346
3.3.5.2 Directional control in wave-off (go-around) 347
3.3.6 Lateral-directional characteristics in steady sideslips 348
3.3.6.1 Yawing moments in steady sideslips . . . 350
3.3.6.2 Side forces in steady sideslips 351
3.3.6.3 Rolling moments in steady sideslips . . 352
3.3.6.3.1 Exception for wave-off (go-around) 354
3.3.6.3.2 Positive effective dihedral limit 355
3.3.7 Lateral-directional control in cross winds . . . 356
3.3.7.1 Final approach in cross winds 358
3.3.7.2 Takeoff run and landing rollout in cross winds 359
3.3.7.2.1 Cold- and wet-weather operation 360
3.3.7.2.2 Carrier-based airplanes . . . 361
3.3.7.3 Taxiing wind speed limits 362
3.3.8 Lateral-directional control in dives 363
3.3.9 Lateral-directional control with asymmetric thrust 364
3.3.9.1 Thrust loss during takeoff run 365
3.3.9.2 Thrust loss after takeoff 366
3.3.9.3 Transient effects 367
3.3.9.4 Asymmetric thrust-rudder pedals free . . 368
3.3.9.5 Two engines inoperative 369

*3.4 Miscellaneous flying qualities 371

3.4.1 Approach to dangerous flight conditions 372
3.4.1.1 Warning and indication 372

TABLE OF CONTENTS (Cont.)
(Asterisks Indicate General Discussions)

Section — Page

III (continued)

3.4.1.2 Prevention .372
3.4.2 Stalls .373
3.4.2.1 Required conditions .374
3.4.2.2 Stall warning requirements .375
3.4.2.2.1 Warning speed for stalls at 1g normal to the flight path .375
3.4.2.2.2 Warning range for accelerated stalls .375
3.4.2.3 Stall characteristics .377
3.4.2.4 Stall recovery and prevention .378
3.4.2.4.1 One-engine-out stalls .378
3.4.3 Spin recovery .379
3.4.4 Roll-pitch-yaw coupling .380
3.4.5 Control harmony .381
3.4.5.1 Control force coordination .381
3.4.6 Buffet .382
3.4.7 Release of stores .383
3.4.8 Effects of armament delivery and special equipment .383
3.4.9 Transients following failures .384
3.4.10 Failures .384

3.5 Characteristics of the primary flight control system .385

3.5.1 General characteristics .385
3.5.2 Mechanical characteristics .386
3.5.2.1 Control centering and breakout forces .387
3.5.2.2 Cockpit control free play .389
3.5.2.3 Rate of control displacement .390
3.5.2.4 Adjustable controls .391
3.5.3 Dynamic characteristics .392
3.5.3.1 Control feel .397
3.5.3.2 Damping .398
3.5.4 Augmentation systems .399
3.5.4.1 Performance of augmentation systems .402
3.5.4.2 Saturation of augmentation systems .403
3.5.5 Failures .404
3.5.5.1 Failure transients .404
3.5.5.2 Trim changes due to failures .404
3.5.6 Transfer to alternate control modes .406
3.5.6.1 Transients .406
3.5.6.2 Trim changes .406

TABLE OF CONTENTS (Cont.)
(Asterisks Indicate General Discussions)

Section | Page

III (continued)

*3.6 Characteristics of secondary control systems 407

3.6.1 Trim system 408
3.6.1.1 Trim for asymmetric thrust 408
3.6.1.2 Rate of trim operation 408
3.6.1.3 Stalling of trim systems 408
3.6.1.4 Trim system irreversibility 408
3.6.2 Speed and flight-path control devices 410
3.6.3 Transients and trim changes 411
3.6.3.1 Pitch trim changes 412
3.6.4 Auxiliary dive recovery devices 415
3.6.5 Direct normal-force control 416

3.7 Atmospheric disturbances 417

3.7.1 Use of turbulence models 417
3.7.2 Turbulence models 419
3.7.2.1 Continuous random model (von Karman form). 424
3.7.2.2 Continuous random model (Dryden form) . . 424
3.7.2.3 Discrete model 428
3.7.3 Scales and intensities (clear air turbulence) . . . 435
3.7.3.1 Clear air turbulence (von Karman scales) . 444
3.7.3.2 Clear air turbulence (Dryden scales) . . . 444
3.7.4 Scales and intensities (thunderstorm turbulence). . 447
3.7.4.1 Thunderstorm turbulence (von Karman scales)448
3.7.4.2 Thunderstorm turbulence (Dryden scales). . 448
3.7.5 Application of the turbulence models in analyses. . 449

4. QUALITY ASSURANCE 463

4.1 Compliance demonstration 463

4.2 Airplane States 463

4.2.1 Weights and moments of inertia 463
4.2.2 Center-of-gravity positions 463
4.2.3 Thrust settings 463

4.3 Design and test conditions 463

4.3.1 Altitudes 463
4.3.2 Special conditions 464

4.4 Interpretation of qualitative requirements 464

TABLE OF CONTENTS (Cont.)
(Asterisks Indicate General Discussions)

Section Page

5. PREPARATION FOR DELIVERY 479
5.1 General 479

*6. NOTES 481
6.1 Intended use 481
6.2 Definitions 481
6.2.1 General 482
6.2.2 Speeds 482
6.2.3 Thrust and power 485
6.2.4 Control parameters 486
6.2.5 Longitudinal parameters 487
6.2.6 Lateral-directional parameters 489
6.2.7 Atmospheric disturbance parameters 495
6.3 Interpretation of the F_s/n limits of table III 496
6.4 Gain scheduling 497
6.5 Engine considerations 497
6.6 Effects of aeroelasticity, control equipment, and structural dynamics 497
6.7 Application of Levels 497
6.7.1 Theoretical compliance 497
6.7.2 Level definitions 498
6.7.3 Computational assumptions 498
6.8 Related documents 499
6.9 Marginal indicia 499

BIBLIOGRAPHY 501

APPENDIX I: MIL-F-8785(ASG) 585

APPENDIX II: Wind Speed Information for MIL-F-8785B 615

APPENDIX III: Measurement of ω_n and ζ of Second-Order Responses 621

TABLE OF CONTENTS (Cont.)
(Asterisks Indicate General Discussions)

Section			Page
APPENDIX IV:	Measurement Techniques for Selected Longitudinal Parameters		629
	IVA	Longitudinal Static Stability	629
	IVB	Flight-Path Stability	630
	IVC	Short-Period Response Measurement	631
	IVD	Analog Matching Techniques for Short-Period Data Reduction	634
	IVE	Control Forces and Motions in Pullups and Steady Turns	637
	IVF	Transient Control Forces and Control System Dynamics	638
APPENDIX V:	Measurement Techniques and Discussion of Selected Lateral-Directional Parameters		639
	VA	Roll Response to Side Gusts	639
	VB	Measurement of Lateral-Directional Modal Parameters	645
	VC	Significance of the Phase Angles ψ_β and $\psi_{p/\beta}$	668
	VD	Notes on Roll Performance of Current Airplanes and Adequacy of MIL-F-8785 and Reference A1 Requirements	684

LIST OF SYMBOLS AND ABBREVIATIONS*

Symbols

b	Wing span, ft
c	Mean aerodynamic chord, ft
C_d	Residue of Dutch roll mode for sideslip response to step aileron input
C_R	Residue of roll mode for sideslip response to step aileron input
C_s	Residue of spiral mode for sideslip response to step aileron input
C_o	Constant term in time solution for sideslip response to step aileron input
$\frac{1}{C_{1/2}}$	Reciprocal of cycles to damp to half amplitude
d_m	Generalized discrete gust length (always positive), $m = x, y, z$ (ft)
D	Aerodynamic drag, parallel to flight path, lb
F_s	Elevator control force, applied by pilot, lb
F_s/n	Gradient of steady-state elevator control force versus n at constant speed, lb/g
F_{AS}	Aileron stick force, lb
F_{AW}	Aileron wheel force, lb
F_{cs}	Net elevator control force (lb), $F_{cs} = F_s - (F_s)_b$
$(F_s)_b$	That portion of F_s which is needed to balance control system moments caused by feedback of airplane responses to the stick e.g., K_{α_e}, K_n, $K_{\dot{g}}$. $(F_s)_b = 0$ for airplanes without K_{α_e} or control system mass unbalance
$(F_s/n)_b$	That portion of F_s/n contributed by H_{α_e} or control system mass unbalance
$(F_s/n)_{fs}$	That portion F_s/n contributed by feel springs, H_{δ_e}, or other devices generating forces which are proportional to δ_s only: $(F_s/n)_{fs} = (F_s/n) - (F_s/n)_b$

*Additional and more detailed definitions are given in Section 6.2 of MIL-F-8785B

Symbols

$(F_s/n)_{min}$ The minimum value of the amplitude ratio $|F_s/n|$, measured from an inverted frequency response of $\frac{n(s)}{F_s(s)}$.

g Acceleration of gravity, ft/sec^2

h Height above ground level (AGL) or above mean sea level (MSL), ft

h_{max} Maximum service altitude, ft

$h_{o_{max}}$ Maximum operational altitude, ft

$h_{o_{min}}$ Minimum operational altitude, ft

H Elevator hinge moment, ft-lb

H_{α_e} Gradient of H with α_e

H_{δ_e} Gradient of H with δ_e

$H_{\dot{\delta}_e}$ Gradient of H with $\dot{\delta}_e$

I_x, I_y, I_z Moments of inertia about x, y, and z axes, respectively, slug-ft^2

I_{xz} Product of inertia, slug-ft^2

j $\sqrt{-1}$

k Ratio of "commanded roll performance" to "applicable roll performance requirement" of 3.3.4 or 3.3.4.1, where:

(a) "Applicable roll performance requirement," $(\phi_t)_{requirement}$, is determined from 3.3.4 and 3.3.4.1 for the Class, Flight Phase Category and Level under consideration.

(b) "Commanded roll performance," $(\phi_t)_{command}$, is the bank angle attained in the stated time for a given step aileron command with rudder pedals employed as specified in 3.3.4 and 3.3.4.1.

$$k = \frac{(\phi_t)_{command}}{(\phi_t)_{requirement}}$$

K_d Residue of Dutch roll mode for roll rate response to step aileron input

Symbols

K_f	Feedback gain for bobweight dynamics
K_n	Feedback gain of normal acceleration to the stick, caused by mass unbalance in the control system, $K_n = (F_s/n)_b$
K_R	Residue of roll mode for roll rate response to step aileron input
K_S	Residue of spiral model for roll rate response to step aileron input
K_{α_e}	Feedback gain of α_e to the stick, caused by H_{α_e}
$K_{\ddot{\theta}}$	Feedback gain of $\ddot{\theta}$ to the stick, caused by control system mass distribution
K_ϕ	Gain constant in roll-aileron transfer function
ℓ_b	Equivalent bobweight length, ft (distance ahead of c.g.), $\ell_b = g(K_{\ddot{\theta}}/K_n)$
ℓ_{cp}	Distance of airframe center-of-percussion ahead of c.g., ft, $\ell_{cp} = Z_{\delta_e}/M_{\delta_e}$
L	Aerodynamic lift plus thrust component, normal to the flight path, lb
L	Rolling moment about the x-axis, including thrust effects, ft-lb
L_i	$= \frac{1}{I_x} \frac{\partial L}{\partial i}$, $i = \beta, \dot{\beta}, \delta_{AS}, \delta_{RP}, p, r$
L_i'	$= \left[1 - \frac{I_{xz}^2}{I_x I_z}\right]^{-1} \left[L_i - \frac{I_{xz}}{I_x} N_i\right]$, $i = \beta, \dot{\beta}, \delta_{AS}, \delta_{RP}, p, r$
L_α	$= \frac{1}{mV} \frac{\partial L}{\partial \alpha}$
L_u	Scale for u_g, ft
L_v	Scale for v_g, ft
L_w	Scale for w_g, ft
m	Mass of airplane, slugs
M	Mach number
M	Pitching moment about the y-axis, including thrust effects, ft-lb

Symbols

M_i $= \frac{1}{I_y} \frac{\partial M}{\partial i}$, $i = \alpha, \dot{\alpha}, u, \theta, q, \delta_e$

M_{F_s} $= \frac{\delta_e}{F_s} M_{\delta_e}$

n Normal acceleration or normal load factor, measured at the c.g., g's

n/α The steady-state normal acceleration change per unit change in angle of attack for an incremental elevator deflection at constant speed (airspeed and Mach number), g's/rad

n_f Load factor normal to the flight path, measured at the c. g.,g's

n_L Symmetrical flight limit load factor for a given Airplane Normal State, based on structural considerations

n_{max}, n_{min} Maximum and minimum Service load factors

$n(+), n(-)$ For a given altitude, the upper and lower boundaries of n in the V-n diagrams depicting the Service Flight Envelope

$n_{o_{max}}, n_{o_{min}}$ Maximum and minimum Operational load factors

$n_o(+), n_o(-)$ For a given altitude, the upper and lower boundaries of n in the V-n diagrams depicting the Operational Flight Envelope

N Yawing moment about the z-axis, including thrust effects, ft-lb

N_i $= \frac{1}{I_z} \frac{\partial N}{\partial i}$, $i = \beta, \dot{\beta}, \delta_{AS}, \delta_{RP}, p, r$

N_i' $= \left[1 - \frac{I_{xz}^2}{I_x I_z}\right]^{-1} \left[N_i - \frac{I_{xz}}{I_z} L_i\right]$, $i = \beta, \dot{\beta}, \delta_{AS}, \delta_{RP}, p, r$

p Roll rate about the x-axis

p_n Amplitude of roll rate response at Dutch roll peaks for step aileron input

$\frac{p_{osc}}{p_{AV}}$ A measure of the ratio of the oscillatory component of roll rate to the average component of roll rate following a rudder-pedals-free step aileron control command

$$\zeta_d \leq 0.2 : \frac{p_{osc}}{p_{AV}} = \frac{p_1 + p_3 - 2p_2}{p_1 + p_3 + 2p_2}$$

$$\zeta_d > 0.2 : \frac{p_{osc}}{p_{AV}} = \frac{p_1 - p_2}{p_1 + p_2}$$

where p_1 , p_2 and p_3 are roll rates at the first, second and third peaks, respectively

Symbols

$\frac{pb}{2V}$	Wing tip helix angle, rad
$\measuredangle \frac{p}{\beta}$	Phase angle between roll rate and sideslip in the free Dutch roll oscillation. Angle is positive when p leads β
$p(\eta)$	Probability density of random variable η
$p(\eta \mid \varepsilon)$	Probability density of η conditioned on ε
$P(\eta)$	Cumulative probability that η equals or exceeds a given value
P	Proportion of time spent in turbulence at a given altitude
q	Dynamic pressure, lb/ft^2
q	Pitch rate
r	Yaw rate
s	Laplace operator, sec^{-1}
S	Wing area, ft^2
t	Time, sec
t_{n_β}	Time for the Dutch roll component of the sideslip response to reach the n^{th} local maximum for a right step or pulse aileron-control command, or the n^{th} local minimum for a left command. In the event a step control input cannot be accomplished, the control shall be moved as abruptly as practical and, for purposes of this definition, time shall be measured from the instant the cockpit control deflection passes through half the amplitude of the commanded value. For pulse inputs, time shall be measured from a point halfway through the duration of the pulse.
$1/T_{cs}$	Inverse time constant of the first-order control-system mode when the feel system dynamics are third order, sec^{-1}: $$\frac{\delta_s(s)}{F_{cs}(s)} = \frac{(\delta_s/F_{cs})(\omega_{n_{cs}}^2 \frac{1}{T_{cs}})}{(s^2 + 2\zeta_{cs}\omega_{n_{cs}} s + \omega_{n_{cs}}^2)\ s + \frac{1}{T_{cs}}}$$
T_d	Damped period of the Dutch roll, $T_d = \frac{2\pi}{\omega_{n_d}\sqrt{1-\zeta_d^2}}$, sec

Symbols

T_2	Time to double amplitude, $T_2 = \frac{-0.693}{\zeta\omega_n}$ for an oscillation, $T_2 = -0.693\tau$ for a first-order divergence, sec
$\frac{1}{T_{1/2}}$	Reciprocal of time to damp to half amplitude, $\frac{1}{T_{1/2}} = \frac{\zeta\omega_n}{0.693}$ for an oscillation, $\frac{1}{T_{1/2}} = 0.693\tau$ for a first-order convergence, sec^{-1}
$\frac{1}{T_{es}}$	Inverse time constant of first-order representation of elevator-servo dynamics, sec^{-1}
$\frac{1}{T_{f_1}}, \frac{1}{T_{f_2}}$	Inverse time constants of feedback zero caused by bobweights or H_{α_e}, when the zero is not complex, sec^{-1}
$\frac{1}{T_{h_1}}$	Lowest-frequency zero of the altitude-elevator transfer function
$\frac{1}{T_{n_1}}, \frac{1}{T_{n_2}}$	Inverse time constants of the constant-speed n-to-δ_e transfer-function numerator, sec^{-1}
$\frac{d(T/W)}{dV}$	$(1/W)$ times the slope of the thrust required (for trimmed flight at constant γ) vs. airspeed plot
$\frac{1}{T_\alpha}$	Inverse time constant of the constant-speed α-to-δ_e transfer function numerator, sec^{-1}
$\frac{1}{T_{\theta_2}}$	The first-order zero of the constant-speed attitude-elevator transfer function, sec^{-1}
u	Incremental velocity along the x reference axis, ft/sec
u_g	Random gust velocity along the x body axis, ft/sec
v	Incremental velocity along the y reference axis, ft/sec
v_m	Generalized discrete gust velocity, positive along the positive airplane body axes, $m = x, y, z$, ft/sec
v_g	Random gust velocity along the y body axis, ft/sec
V	Airspeed
$V_{L/D}$	Speed for maximum lift to drag ratio
V_{MAT}	High speed, level flight, maximum augmented thrust
V_{max}	Maximum service speed
V_{end}	Speed for maximum endurance
V_{range}	Speed for maximum range in zero wind conditions

Symbols

V_{min}	Minimum service speed
V_{MRT}	High speed, level flight, military rated thrust
V_{NRT}	High speed, level flight, normal rated thrust
$V_{R/C}$	Speed for maximum rate of climb
V_S	Stall speed (equivalent airspeed), at 1 g normal to the flight path, defined as the highest of: - speed for steady straight flight at $C_{L_{max}}$, the first local maximum of the curve of lift coefficient (L/qS) vs. angle of attack which occurs as C_L is increased from zero - speed at which abrupt controllable pitching, rolling or yawing occurs; i.e., loss of control about a single axis - speed at which intolerable buffet or structural vibration is encountered (Note that 3.1.9.2.1 allows an alternative definition of V_S in some cases.)
V_{trim}	Trim speed
$V_{o_{max}}$	Maximum operational speed
$V_{o_{min}}$	Minimum operational speed
W	Weight of the airplane, lb
w	Incremental velocity along the z reference axis, ft/sec
w_g	Random gust velocity along the z body axis, ft/sec
x	Body-fixed axis of the airplane, along the projection of the undisturbed (trim or operating-point) velocity onto the plane of symmetry, with its origin at the c.g.
X	force along the x-axis, aerodynamic plus thrust, lb
X_i	$= \frac{1}{m}\frac{\partial X}{\partial i}$, where $i = \alpha, \dot{\alpha}, u, \theta, q, \delta_e$
y	Body-fixed axis of the airplane perpendicular to the plane of symmetry directed out the right wing, with its origin at the c.g.
Y	Side force along the y-axis, aerodynamic plus thrust component, lb

Symbols

Y_i $= \frac{1}{mV}\frac{\partial Y}{\partial i}$, $i = \beta, \dot{\beta}, \delta_{AS}, \delta_{RP}, p, r$

z Body-fixed axis of the airplane, directed downward perpendicular to the x and y axes, with its origin at the c.g.

Z Force along z-axis, lb

Z_i $= \frac{1}{m}\frac{\partial Z}{\partial i}$, $i = \alpha, \dot{\alpha}, u, \theta, q, \delta_e$

α Angle of attack, the angle in the plane of symmetry between the fuselage reference line and the tangent to the flight path at the airplane center of gravity

α_e Local angle-of-attack of the horizontal tail (angle between the horizontal tail reference line and the relative wind)

α_e/α Gradient of α_e with α

$\alpha_e/\dot{\alpha}$ Gradient of α_e with $\dot{\alpha}$

α_e/q Gradient of α_e with q $(\dot{\theta})$

α_s The stall angle of attack at constant speed for the configuration, weight, center-of-gravity position and external-store combination associated with a given Airplane Normal State; defined as the highest of the following:

- Angle of attack for the highest steady load factor, normal to the flight path, that can be attained at a given speed or Mach number
- Angle of attack, for a given speed or Mach number, at which abrupt uncontrollable pitching, rolling or yawing occurs, i.e., loss of control about a single axis
- Angle of attack, for a given speed or Mach number, at which intolerable buffeting is encountered

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

$\Delta\beta_{max}$ Maximum sideslip excursion at the c.g., occurring within two seconds or one half-period of the Dutch roll, whichever is greater, for a step aileron-control command

γ Climb angle, $= \sin^{-1} \frac{\text{vertical speed}}{\text{true airspeed}}$, positive for climb

Symbols

Δ	Used in combination with other parameters to denote a change from the initial value
δ_a	Aileron surface deflection
δ_{AS}	Displacement of the aileron stick along its path
δ_{AW}	Displacement of the aileron wheel along its path
δ_e	Elevator surface deflection
δ_e/F_{cs}	Gradient of steady-state δ_e with F_{cs} at constant speed
δ_e/F_s	Gradient of steady-state δ_e with F_s at constant speed
δ_e/δ_s	Gradient of steady-state δ_e with δ_s at constant speed
δ_r	Rudder surface deflection
δ_{RP}	Rudder pedal deflection, in.
δ_s	Elevator stick deflection, in.
δ_s/F_{cs}	Gradient of steady-state δ_s with F_{cs} at constant speed
δ_s/F_s	Gradient of steady-state δ_s with F_s at constant speed
ζ_{cs}	Damping ratio of the elevator feel system
ζ_d	Damping ratio of the Dutch roll oscillation
ζ_f	Damping ratio of the feedback zero caused by H_{α_e} or control-system mass unbalance
ζ_p	Damping ratio of the phugoid oscillation
ζ_{RS}	Damping ratio of the roll-spiral 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
λ	Turbulence wavelength, ft

Symbols

λ_d Dutch roll mode root, where $\lambda_d = -\zeta_d \omega_{n_d} \pm j\omega_{n_d}\sqrt{1-\zeta_d^2}$

λ_R $= -1/\tau_R$

λ_S $= -1/\tau_S$

ρ Air density, slug/ft^3

σ Real part of a complex dynamic root, sec^{-1}

σ Root-mean-square gust intensity, where $\sigma^2 = \int_0^\infty \Phi(\Omega)\, d\Omega = \int_0^\infty \phi(\omega)\, d\omega$

$\sigma_u, \sigma_v, \sigma_w$ Root-mean-square intensities of u_g, v_g, w_g, respectively

τ_R First-order roll mode time constant, positive for a stable mode, sec

τ_S First-order spiral mode time constant, positive for a stable mode, sec

$1/\tau_{\beta_{1,2}}$ Zeros of sideslip-aileron transfer function

ϕ Bank angle measured in the y-z plane, between the y-axis and the horizontal

ϕ_t Bank angle change in time t, in response to control deflection of the form given in 3.3.4

$\dfrac{\phi_{osc}}{\phi_{AV}}$ A measure of the [illegible] of the oscillatory component of bank angle to the average component of bank angle following a rudder-pedals-free impulse aileron control command

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

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

where ϕ_1, ϕ_2 and ϕ_3 are bank angles at the first, second and third peaks, respectively

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

$\Phi_{u_g}(\Omega)$ Spectrum for u_g, where $\Phi_{u_g}(\Omega) = V\phi_{u_g}(\omega)$, (ft/sec)^2/(rad/ft)

$\Phi_{v_g}(\Omega)$ Spectrum for v_g, where $\Phi_{v_g}(\Omega) = V\phi_{v_g}(\omega)$, (ft/sec)^2/(rad/ft)

$\Phi_{w_g}(\Omega)$ Spectrum for w_g, where $\Phi_{w_g}(\Omega) = V\phi_{w_g}(\omega)$, (ft/sec)^2/(rad/ft)

Symbols

ψ_β — Phase angle in a cosine representation of the Dutch roll component of sideslip - negative for a lag

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

with n as in t_{n_β} above

ψ_n — Angular coordinate of vector n in s plane

ψ_p — Phase angle of Dutch roll oscillation in roll rate response to step aileron input, deg

ω — Temporal frequency, rad/sec, where $\omega = \Omega V$

ω — Imaginary part of a complex dynamic root, sec^{-1}

ω_f — Undamped natural frequency of the feedback zero caused by H_{α_e} or control-system mass unbalance, rad/sec

$\omega_{n_{cs}}$ — Undamped natural frequency of the elevator feel system, rad/sec

ω_{n_d}, ω_d — Undamped natural frequency of the Dutch roll oscillation, rad/sec

ω_{n_p} — Phugoid undamped natural frequency, rad/sec

$\omega_{n_{sp}}$ — Undamped natural frequency of the short-period oscillation, rad/sec

ω_{RS} — Roll-spiral undamped natural frequency, rad/sec

ω_ϕ — Undamped natural frequency of numerator quadratic of ϕ/δ_{AS} transfer function, rad/sec

Ω — Longitudinal spatial reduced frequency

$\left(\Omega = \frac{2\pi}{\lambda} = \frac{\omega}{V}\right)$, rad/ft

Abbreviations

AGL	Above ground level
c.g.	Center of gravity
exp ( )	The Napierian logarithmic base (e = 2.718...) raised to the power indicated
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
R/C	Rate of climb
SAS	Stability augmentation system
TLF	Thrust for level flight
PR	Pilot rating
PIO	Pilot-induced oscillation
$(\dot{\ })$	A dot above a symbol signifies the time derivative, e.g. $\dot{\alpha} = \frac{d\alpha}{dt}$
( )'	A prime used in conjunction with $\omega_{n_{sp}}$, ζ_{sp}, $\omega_{n_{cs}}$, ζ_{cs}, $1/T_{cs}$, or $1/T_{es}$ denotes stick-free values of the parameters when the stick-free and stick-fixed values are not the same (e.g. $\omega'_{n_{sp}}$ or ζ'_{sp}). In particular, this notation is used when bobweights or H_{α_e} caused the airplane response to feed back to the stick, unprimed parameters denoting values with the stick-fixed or the bobweight feedback loop open, and primed parameters denoting stick-free values with the feedback loop closed.

Section I

INTRODUCTION

This document is published in support of Military Specification MIL-F-8785B(ASG) "Flying Qualities of Piloted Airplanes" as part of a three-year effort of the Air Force Flight Dynamics Laboratory, with the contracted help of Cornell Aeronautical Laboratory (CAL), to conduct a coordinated theoretical and experimental investigation of airplane flying qualities. The intent of this document is to explain the concept and philosophy underlying MIL-F-8785B 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.

Section II outlines the historical development of the project.

The order in which the material is presented in Section III parallels that of MIL-F-8785B(ASG). The main subject headings are:

1 Scope and Classification
2 Applicable Documents
3.1 Requirements - General
3.2 Longitudinal Flying Qualities
3.3 Lateral-Directional Flying Qualities
3.4 Miscellaneous Flying Qualities
3.5 Characteristics of the Primary Flight Control System
3.6 Characteristics of Secondary Control Systems
3.7 Atmospheric Disturbances
4 Quality Assurance Provisions
6 Notes

There is a general discussion of each of these main subjects. Where appropriate, further general discussion precedes a related smaller group of requirements. Each paragraph of the specification is discussed in sequence, individually or together with a few closely related paragraphs, under the following subheadings:

Requirement
Related MIL-F-8785 paragraphs
Discussion

A bibliography of reports which were collected is also presented. It represents a fairly complete listing of reports pertaining to flying qualities and related topics. References cited in this report are listed in the bibliography.

Appendices are included which contain the previous specification, MIL-F-8785, and pertinent supplementary information such as flight test and measurement techniques.

Section II

HISTORICAL DEVELOPMENT

The effort to revise MIL-F-8785B(ASG) began on 10 January 1966 and the first round of meetings with industry took place during the weeks of 17 and 31 January 1966. The objective of the meetings was to obtain inputs from a wide range of users of MIL-F-8785 with regard to its adequacy as a design and evaluation specification, the consequences of specification deficiencies in terms of system design problems and adverse effects on mission capabilities, and recommendation on revisions needed to alleviate these problems. The following organizations and companies were represented:

United States Air Force
Cornell Aeronautical Laboratory, Inc.
Federal Aviation Administration
The Boeing Company/Renton
The Boeing Company/Seattle
The Boeing Company/Wichita
Douglas Aircraft Company, Inc.
General Dynamics/Convair Division
General Dynamics/Fort Worth Division
Grumman Aircraft Engineering Corp.
LTV/Vought Aeronautics Division
Lockheed California Company
Lockheed Georgia Company
Martin Company
McDonnell Aircraft Corporation
North American Aviation, Inc./Columbus Division
North American Aviation, Inc./Los Angeles Division
Northrop Corporation/Norair Division
Republic Aviation Corporation

A library of reports pertaining to flying qualities was also begun in January 1966. Over six hundred reports were collected by Cornell Aeronautical Laboratory.

During April and May 1966, the following data gathering activities took place:

- Air Force Flight Dynamics Laboratory-Aeronautical Systems Division flying qualities symposium on 5 and 6 April (Reference A3).

- Visits to NASA Langley and the Naval Air Test Center at Patuxent River, Maryland on 19 and 20 April.

- Visits to NASA Ames and FRC, AFFTC at Edwards AFB, and Systems Technology, Incorporated on 25-29 April.

* Meetings with the following control equipment manufacturers on 9, 10, 12 May:

  Bendix Corporation
  General Electric Company
  Honeywell, Inc.
  Lear Siegler, Inc.
  Sperry Phoenix Company

The Air Force also acquired and worked up a great deal of stability and control data on current airplanes in that time period.

The data thus obtained, in conjunction with considerable analytical work, formed the basis for the first CAL draft of the Recommendations for Revision of MIL-F-8785(ASG), which was submitted to the Air Force in March 1967. Substantiation information for selected longitudinal and lateral-directional requirements of the draft revision was submitted to the Air Force in May 1967. These reports, the MIL-F-8785 revision and the backup documents, were then distributed for comments to all concerned governmental agencies and all companies that had participated in the initial meetings.

Written comments were received during 1967 and through April 1968 from many organizations within the USAF and USN, from NASA, FAA and AIA, and from the following companies:

Autonetics
Bendix Corporation/Navigation and Control Division
Boeing Company
Douglas Aircraft Company/Aircraft Division
Fairchild Hiller/Aircraft Division
Fairchild Hiller/Republic Aviation Division
General Electric Company/Defense Electronics Division
General Dynamics/Convair Division
General Dynamics/Fort Worth Division
Grumman Aircraft Engineering Corporation
Honeywell, Inc./Aerospace Division
Lear Siegler, Inc./Astronics Division
Lockheed Georgia Company
LTV Aerospace Corporation/Vought Aeronautics Division
Martin-Marietta Corporation
McDonnell Company
North American Rockwell Corporation/Columbus Division
North American Rockwell Corporation/Los Angeles Division
Northrop Corporation/Norair Division
Sperry Phoenix Company
Systems Technology, Inc.

Also, approximately ten review meetings were held with the Navy, Systems Technology, Incorporated and several Air Force organizations during 1967 and through April 1968. In addition, data gathering trips were made to the Air Force Flight Test Center and Nellis Air Force Base.

CAL tabulated the written comments and notes from meetings. This information was systematically reviewed. Each comment was categorized as either:

(a) valid and can be incorporated into specification
(b) probably valid but more work needed
(c) considered to be invalid.

Many comments were considered valid, and CAL incorporated them in one form or another into a revised draft of the Recommendations for Revision of MIL-F-8785, which was submitted to the Air Force in May 1968. CAL also included a considerable amount of new analytical work and data in this draft. Those areas in which there is believed to be a deficiency in the requirements, but which require more work, are discussed in this report, a preliminary version of which was submitted in June 1968. Those comments which were considered to be invalid often resulted from lack of understanding or misinterpretation of the requirement. In those cases, an attempt was made to clarify and expand upon the requirement in this report.

From May 1968 through July 1968, the Navy and the Air Force reviewed the May 1968 specification draft in detail. Detailed changes to the May 1968 specification draft were made at joint Air Force-Navy meetings during the week of July 22, 1968 with CAL acting in an advisory capacity. As a result of these meetings, a preliminary version of MIL-F-8785B, dated July 1968, was distributed to industry for comment in early August.

Comments were received from the following organizations:

Autonetics
Boeing Company
Fairchild Hiller/Republic Aviation Division
General Dynamics/Convair Division
General Dynamics/Fort Worth Division
Lockheed California Company
Lockheed Georgia Company
Martin-Marietta Corporation
McDonnell Douglas Corporation
McDonnell Douglas Corporation/Douglas Aircraft Division
NASA/Langley Research Center
North American Rockwell Corporation/Los Angeles Division
Northrop Corporation/Norair Division
Sperry Rand Corporation/Flight Systems Division
Systems Technology, Inc.

CAL also recommended some changes during this time period. The comments on the July 1968 draft were relatively minor compared to those received on the May 1967 draft.

The significant comments from industry and CAL were incorporated into the final version of MIL-F-8785B at a joint Air Force-Navy meeting during the week of October 7, 1968, with CAL again acting in an advisory capacity. The final draft was finished on October 22, 1968. An Air Force interim version, MIL-F-008785A, to be used in lieu of MIL-F-8785, was published October 31, 1968; and the draft was sent to the Aeronautical Standards Group by the Air Force on December 19, 1968.

Following finalization of the specification, CAL prepared the complete draft of this document, including significant contributions from the Air Force and Navy, and submitted it for Air Force and Navy review in January 1969. Again, extensive changes resulted. CAL prepared copy for printing in June 1969.

SECTION III · DISCUSSION OF REQUIREMENTS

Section III

STATEMENT AND DISCUSSION OF REQUIREMENTS

1. SCOPE AND CLASSIFICATIONS

DISCUSSION

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 well the job must be done (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. Seldom will such a fine breakdown be required, nor will there be sufficient information available to make such fine discriminations. Thus in most cases, the 36 possible requirements are combined to some extent, but not necessarily in the same pattern for all requirements. In other cases, different or additional breakdowns are required: land- or carried-based airplanes, or specific Flight Phases.

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			

The framework then is comprised of those MIL-F-8785B paragraphs:

1.3 Classification of airplanes
1.4 Flight Phase Categories
1.5 Levels of flying qualities

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

1.1 SCOPE

REQUIREMENT

1.1 Scope. This specification contains the requirements for the flying qualities of U.S. military piloted airplanes.

RELATED MIL-F-8785 PARAGRAPHS

1.1

DISCUSSION

The scope is unchanged from that of MIL-F-8785. The requirements are not particularly aimed at such unconventional aircraft as helicopters, V/STOL or re-entry vehicles, but many of the requirements may be found to apply reasonably well to those aircraft in specific instances. Separate flying qualities specifications are being prepared, also with the contracted help of Cornell Aeronautical Laboratory, for V/STOL aircraft and for re-entry vehicles.

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 airplanes proposed or contracted for shall be in accordance with the provisions of this specification unless specific deviations are authorized by the procuring activity. Additional or alternate special requirements may be specified by the procuring activity.

RELATED MIL-F-8785 PARAGRAPHS

1.2

DISCUSSION

To the material in the MIL-F-8785 paragraph has been added a statement of purpose. Additional insight on the rationale is given in an expanded Note 6.1:

"Intended use. This specification contains the flying qualities requirements for piloted airplanes and forms one of the bases for determination by the procuring activity of airplane acceptability. The specification serves 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. The requirements are intended to assure adequate flying qualities regardless of design implementation or flight control system mechanization. To the extent possible, this specification should be met by providing an inherently good basic airframe. Where that is not entirely 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."

1.3 CLASSIFICATION OF AIRPLANES

REQUIREMENT

1.3 Classification of airplanes. For the purpose of this specification, an airplane shall be placed in one of the following Classes:

Class I — Small, light airplanes such as
- Light utility
- Primary trainer
- Light observation

Class II — Medium weight, low-to-medium maneuverability airplanes such as
- Heavy utility/search and rescue
- Light or medium transport/cargo/tanker
- Early warning/electronic countermeasures/airborne command, control, or communications relay
- Antisubmarine
- Assault transport
- Reconnaissance
- Tactical bomber
- Heavy attack
- Trainer for Class II

Class III — Large, heavy, low-to-medium maneuverability airplanes such as
- Heavy transport/cargo/tanker
- Heavy bomber
- Patrol/early warning/electronic countermeasures/airborne command, control, or communications relay
- Trainer for Class III

Class IV — High-maneuverability airplanes such as
- Fighter/interceptor
- Attack
- Tactical reconnaissance
- Observation
- Trainer for Class IV

The procuring activity will assign an airplane 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 airplane of one Class may be required by the procuring activity to meet selected requirements ordinarily specified for airplanes of another Class.

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

RELATED MIL-F-8785 PARAGRAPHS

1.3, 1.3.1

DISCUSSION

These paragraphs replace paragraphs 1.3 and 1.3.1 in MIL-F-8785. The aspects of intended use seem implicit in this classification scheme and, although it is not specifically stated, one can see some correlation with weight and limit load factor. Because a classification scheme that consists only of groupings of a list of mission titles will generally become obsolete as new missions are devised, it seems desirable to try to define Classes on a more general basis. Reference A1 has four Classes and groups airplanes on the basis of maneuverability, size, weight, and intended use. Within each group, examples are given which use the basic mission title nomenclature defined in DOD Directive Number 4505.6 dated 14 August 1967. The four Classes are also related qualitatively to maximum design gross weight and symmetrical flight limit load factor at the basic flight design gross weight.

In the proposed revision of MIL-A-8861, Airplane Strength and Rigidity Flight Loads, dated October 1967, symmetrical flight limit load factor design values are specified for the aircraft designations of the DOD Directive. The specified limit load factors, together with weight data for U.S. military airplanes listed in the March 18, 1968 issue of Aviation Week & Space Technology, were used to form Figure 1. The presentation of Figure 1 makes it obvious that highly maneuverable airplanes such as fighter and attack types, together with certain trainer and observation craft should be designed for high limit load factor. These vehicles tend to group in the weight range from 5000-100,000 lb. There are a few small, light-weight trainers and observation airplanes which are also designed for fairly high load factors, which could be in either Class I or Class IV. Classification of these airplanes should be done on the basis of more detailed information about the intended use; or alternatively, the detail specification should be composed of requirements selected from those stated for both of these Classes in Reference A1. Figure 1 also illustrates that all other airplanes are required to be designed for a limit load factor of less than 4 g's, and that current airplanes span the weight range from 1,000 to almost 1,000,000 lb.

Historically, flying qualities specifications have recognized the need to specify different values of parameters for vehicles of different size and different operational missions. It is intuitive to expect the handling qualities of sports cars to be different from those of trucks, speed boats to handle differently than ocean liners, and small utility airplanes to fly differently than large transports. In each of these examples there is a difference in size, but also in operational use or intended mission. In addition, there may be significant differences in the way each vehicle responds to external disturbances such as road roughness, sea state and atmospheric turbulence or wind. Another factor of possible significance is the location of the driver or pilot in the vehicle relative to the center of gravity and the extremities of the

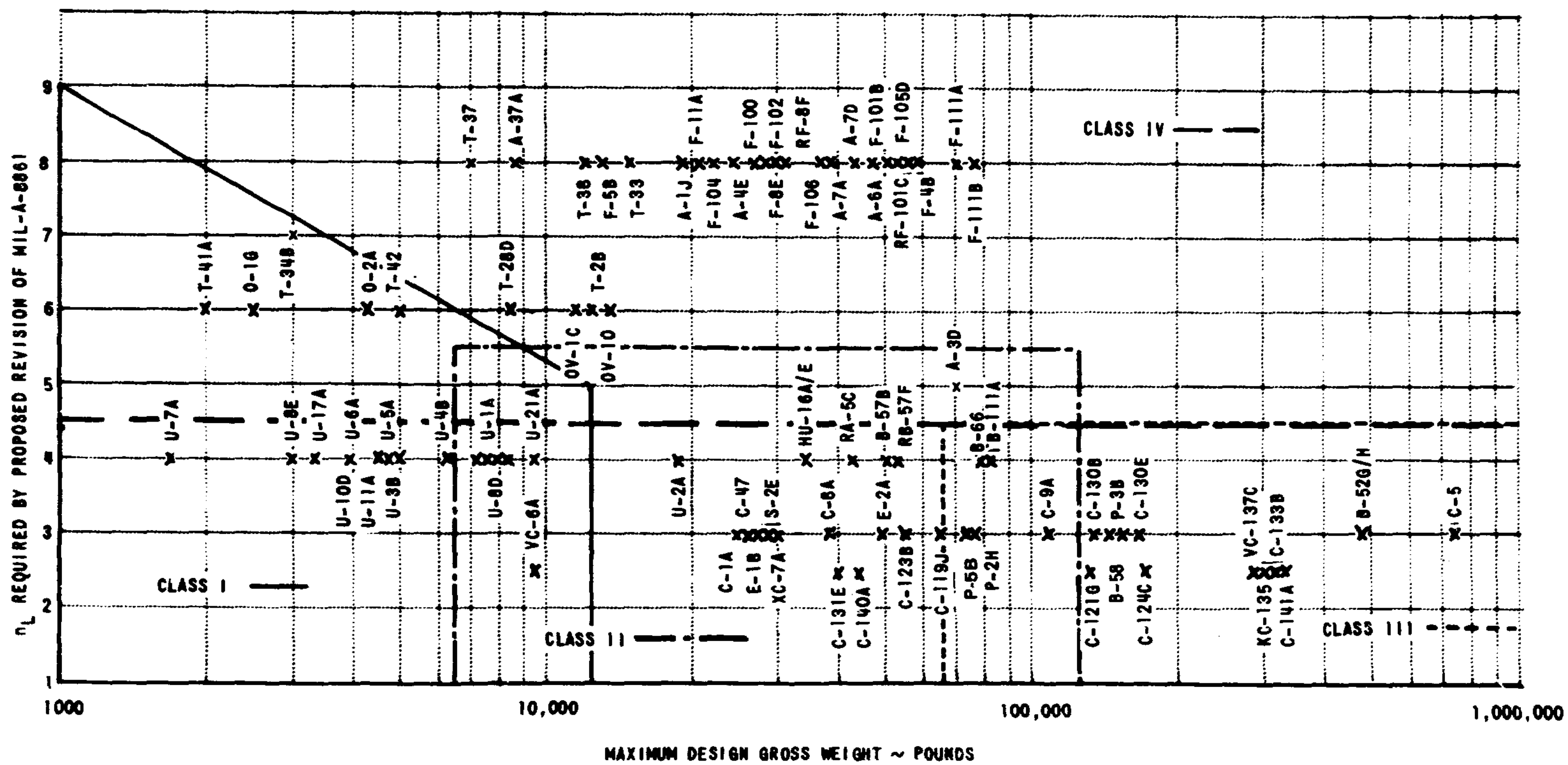


Figure 1(1.3) CLASSIFICATION OF AIRPLANES vehicle. The location of the pilot in the vehicle affects the motions and riding characteristics he experiences. If the effect of each of these factors on handling or flying qualities were fully understood and a sufficient data base existed, then the quantitative requirements could be stated as mathematical or empirical functions of the significant factors, and there would be no need for any classification breakdown to accommodate these effects in the specification requirements.

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

How best to handle the factors discussed above is not completely clear at this time. Ideally the requirements should be expressed as mathematical functions of the significant factors. The current state of knowledge and the experimental data available do not permit this, so it is necessary to make the relatively arbitrary Class definitions of Reference A1. Further research into possible scaling parameters, simulation study and operational experience are required in this area.

As in MIL-F-8785, the -C requirements reflect both the unique and the more critical aspects of carrier operation in terminal Flight Phases.

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 and so that transition will be smooth. When no Flight Phase or Category is stated in a requirement, that requirement shall apply to all three Categories. In certain cases, requirements are directed at specific Flight Phases identified in the requirement. Flight Phases descriptive of most military airplane 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)

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

a. Climb (CL)
b. Cruise (CR)
c. Loiter (LO)
d. In-flight refueling (tanker) (RT)
e. Descent (D)
f. Emergency descent (ED)
g. Emergency deceleration (DE)
h. Aerial delivery (AD)

Terminal Flight Phases:

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

a. Takeoff (TO)

b. Catapult takeoff (CT)

c. Approach (PA)

d. Wave-off/go-around (WO)

e. Landing (L)

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

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

Experience with airplane operations indicates that certain Flight Phases require more stringent values of flying qualities parameters than do others (e.g., air-to-air combat requires more Dutch roll damping than does cruising flight). Also, a given mission Flight Phase will generally have an Airplane Normal State associated with i (e.g., flaps and gear down for landing approach and up for cruising flight). In many instances, therefore, the flying qualities specification should state requirements as a function of mission Flight Phase. This degree of breakdown gives the designer additional gudiance in optimizing his design so that each Airplane State has adequate flying qualities for the tasks to be performed while the airplane is in that State.

In flight and simulator evaluations, pilots generally have rated a set of flying qualities on suitability for a given mission segment like one of these Flight Phases. The pilots assign an overall rating, based on ability, or effort required, to perform certain appropriate tasks such as precision tracking of a target or a glide slope, trimming and making heading changes at constant altitude; and of flight in turbulence. As they extrapolate to other flight situations (tasks, environment, etc.), their uncertainty in their ratings grows. These considerations led naturally to statement of flying qualities requirements in terms of the Flight Phases of 1.4.

For the most part, the Flight Phase titles are descriptive enough to facilitate picking those applicable to a given design. The formation flying (FF) Flight Phase is intended to be used, if desired, where there is no other requirement for rapid maneuvering, precision tracking or precise flight-path control in up-and-away flight. An example might be a Class 1 trainer for which the procuring activity desires Category A flying qualities (note the current use of the T-37, T-28, etc.).

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

Not all of these Flight Phases apply to a given airplane. 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 the stated primary and alternate missions), 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 transition between phases of each flight will be smooth) will be accomplished. It is the procuring activity's responsibility either to agree with the contractor's suggestions or to recategorize the Flight Phases.

In certain cases, both flying qualities requirements and airplane capabilities may be less than one would ordinarily expect. An example is a zoom climb--a dynamic maneuver in which qualities such as speed stability and natural frequency cannot be measured in flight, and the effectiveness of aerodynamic controls is necessarily low at low dynamic pressure. Lacking enough data to formulate general requirements for these cases, it has been left for the procuring activity to provide specific requirements as specific mission needs dictate.

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 airplane 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 airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate, or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed.

RELATED MIL-F-8785 PARAGRAPHS

None

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 6.7.2: "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...."

According to 4.4, 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 B5). 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 B113). 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 B5. The revised rating scale from Reference B113 is reproduced in Figure 1.

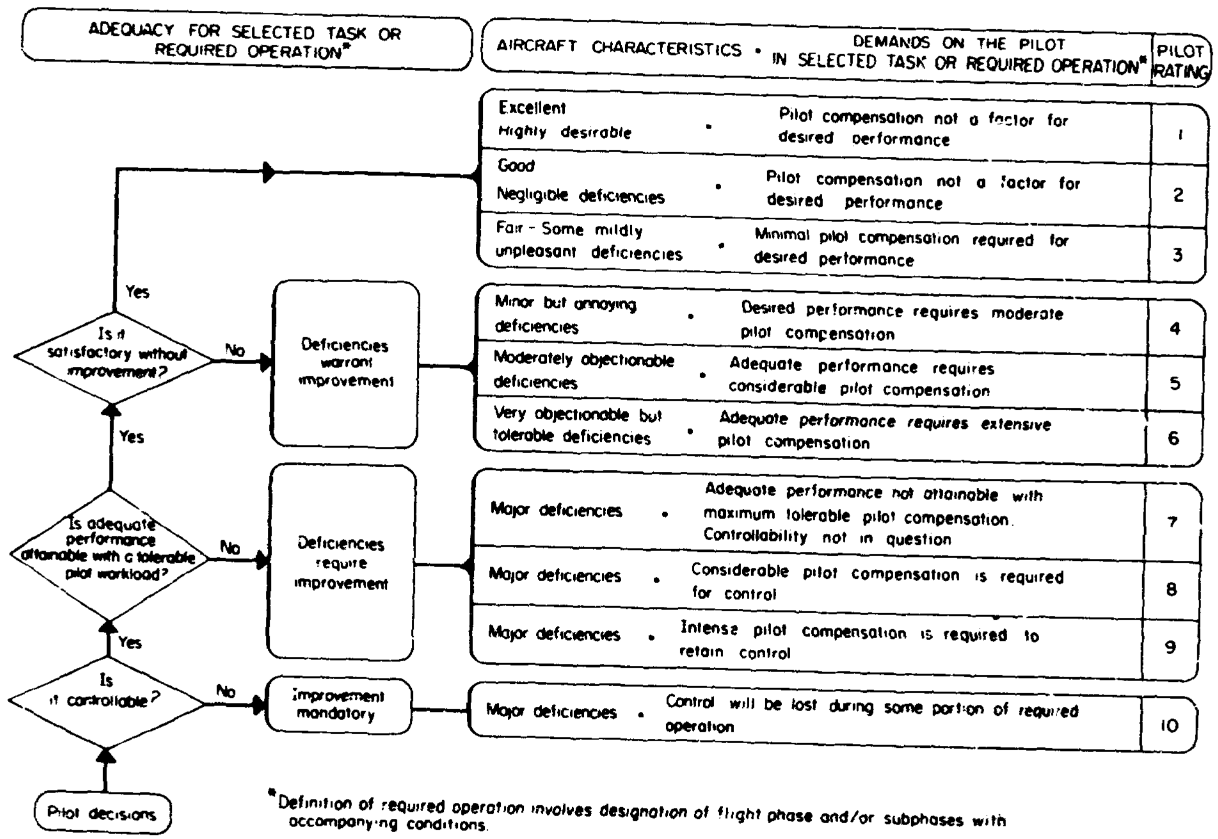
Although a direct association is intended between the Levels of Reference A1 and the revised rating scale in Figure 1, the association with previous rating scales is not as direct. Since the majority of the experimental flying qualities data available at this time was produced using either the original 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	Standard CAL Scale	Interim Revision-Cooper-Harper Scale (Ref. B5)	Final Revision-Cooper-Harper Scale (Ref. B113)
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+

The particular association between Levels and rating scales other than those shown above is presented, along with the particular set of data under consideration, in the following sections.

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.



*Definition of required operation involves designation of flight phase and/or subphases with accompanying conditions.

Figure 1 (1.5) REVISED RATING SCALE (REFERENCE B113)

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 airplane may be unflyable. Some Level 3 requirements have been stiffened arbitrarily, partly to account for this possibility. There is too little data to treat the problem more accurately. Our solace is that with low probabilities of single failures, the joint probabilities of most multiple degradations are very small. As any flying quality approaches or passes the Level 3 limit, it will become of interest to people concerned with flight safety.

Inclusion of further-degraded Levels (below Level 3) was discussed: get-home capability or stabilization for ejection. But by definition, Level 3 gives ability to recover, return and land safely following a failure at the most adverse point in the flight. It seems best to demand no less than that.

2. APPLICABLE DOCUMENTS

REQUIREMENTS

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-D-8708	Demonstration Requirements for Airplanes
MIL-F-9490	Flight Control Systems - Design, Installation and Test of, Piloted Aircraft, General Specification for
MIL-C-18244	Control and Stabilization Systems, Automatic, Piloted Aircraft, General Specification for
MIL-F-18372	Flight Control Systems, Design, Installation and Test of, Aircraft (General Specification for)
MIL-S-25015	Spinning Requirements for Airplanes
MIL-W-25140	Weight and Balance Control Data (for Airplanes and Rotorcraft)

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.)

RELATED MIL-F-8785 PARAGRAPHS

2.1

DISCUSSION

At places it was felt necessary to refer to other documents in the specific requirements of -8785B. All such referenced documents are listed in 2.1. Another group, closely related but not referenced, appears in 6.8:

"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-A-8860	Airplane Strength and Rigidity - General Specification for
MIL-A-8861	Airplane Strength and Rigidity - Flight Loads
MIL-S-38130	Safety Engineering of Systems and Associated Subsystems, and Equipment, General Requirements for
MIL-G-38478	General Requirements for Angle of Attack Based Systems

PUBLICATION

USAF

HIAD-Handbook of Instructions for Airplane Designers"

3.1 · GENERAL REQUIREMENTS

3.1 · GENERAL REQUIREMENTS

3. REQUIREMENTS

3.1 GENERAL REQUIREMENTS

DISCUSSION

This section of Reference A1 specifies the conditions under which the requirements of this specification apply. The main factors are determined by the operational missions for which the airplane is to be designed. The airplane, described by its Airplane 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.

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 airplane to meet the flying qualities requirements of this specification. These missions will include the entire spectrum of intended operational usage.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

The word "missions" 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 airplane 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 vehicle 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 performan e 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 airplane must be known before the required configurations, loadings, and the Operational Flight Envelopes can be defined and the design of the airplane to meet the requirements of this specification can be undertaken. Should the using command decide to use an airplane for an operational mission other than those for which it was designed, the responsibility must be assumed by the using command since the airplane designer can only be held responsible for the requirements specified in the contract covering procurement of the airplane. 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 airplane and the impact this has on the configurations, loadings, and Operational Flight Envelopes for which the airplane is to be designed. Once the intended uses or operational missions are defined, a Flight Phase analysis of each mission must be conducted. With the Flight Phases established, the configurations and loading states which will exist during each Phase can be defined. After the configuration and loading states have been defined for a given Flight Phase, Service and Permissible Flight Envelopes can be determined and Operational Flight Envelopes more fully defined.

loadings is presented in Section 4 of Reference A1 in terms of weight, center-of-gravity position or moments of inertia. Only permissible center-of-gravity positions need be considered for Airplane 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 Airplane Failure States. The worst possible cases that are not approved Special Failure States (3.1.6.2.1) must be examined.

Since the requirements apply over the full range of service loadings, effects of fuel slosh and shifting should be taken into account in design. Balance, controllability, and airframe and structure dynamic characteristics may be affected. For example, take-off acceleration has been known to shift the c.g. embarrassingly far aft. Airplane 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 airplane 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.

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 airplane 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.

RELATED MIL-F-8785 PARAGRAPHS

3.1.1, 3.1.5

DISCUSSION

The loading of an airplane is determined by what is in (internal loading), and attached to (external loading) the airplane. 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 airplane'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. Guidance on such critical

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 to the crew except for rudder, aileron, elevator, throttle and trim controls. Examples are: the flap control setting and the yaw damper ON or OFF. The selected configurations to be examined must consist of those required for performance and mission accomplishment. Additional configurations to be investigated may be defined by the procuring activity.

RELATED MIL-F-8785 PARAGRAPHS

3.1.9

DISCUSSION

The settings of such controls as flaps, speed brakes, landing gear, wing sweep, high lift devices, and wing incidence are related uniquely to each aircraft design. Reference A1 requires that the configurations to be examined shall be those required for performance and mission accomplishment. The position of rudder, aileron, elevator, trim controls and the thrust setting are not included in the definition of configuration since the positions of these controls are usually either specified in the individual requirements or determined by the specified flight conditions.

The requirements of Reference A1 are stated for Flight Phases, rather than for airplane configurations as was done in MIL-F-8785. The flying qualities should be a function of the job to be done rather than of the configuration of the airplane. However, the designer must define the configuration or configurations which his airplane will have during each Flight Phase.

3.1.6 STATE OF THE AIRPLANE

REQUIREMENTS

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

3.1.6.1 Airplane Normal States. The contractor shall define and tabulate all pertinent items to describe the Airplane 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, wing sweep, or thrust setting may vary continuously over a range of values during a Flight Phase. The contractor shall replace this continuous variation by a limited number of values of the parameter in question which will be treated as specific states, and which include the most critical values and the extremes encountered during the Flight Phase in question.

3.1.6.2 Airplane Failure States. The contractor shall define and tabulate all Airplane Failure States, which consist of Airplane Normal States modified by one or more malfunctions in airplane 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 Airplane 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.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

These paragraphs introduce the Airplane State terminology for use in the requirements. The contractor is required to define the Airplane Normal States for each applicable Flight Phase, in the format of Table XVI. A

Table XVI

AIRPLANE NORMAL STATES

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

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 XVI, 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 Airplane 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: 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 applications.

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

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

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

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

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

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

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

3.1.7 OPERATIONAL FLIGHT ENVELOPES
3.1.8 SERVICE FLIGHT ENVELOPES
3.1.9 PERMISSIBLE FLIGHT ENVELOPES

GENERAL DISCUSSION

The increased emphasis in Reference A1 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 airplane 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 only exception is control power, which may set some boundaries if the requirements on the Operational Flight Envelope are still met. The rationale for each type of Envelope is presented later, in the discussion of each paragraph; but here it is in order to discuss procedures in constructing and using the Envelopes.

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 those and other requirements, a contractor will design an airplane. For that design the contractor can relate the Flight Phases to Airplane Normal States, then:

- Further define the Operational Flight Envelope for each Flight Phase, based on the associated Airplane Normal States,
- Construct the larger Service Flight Envelope for the Airplane 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.

The requirements apply at all points within the volume of the pertinent Flight Envelope. These Flight Envelopes, which necessarily are drawn as two-dimensional figures, form skeletons which depict three-dimensional (speed, altitude, normal load factor) volumes of conditions where requirements apply. In picking the altitudes at which to define speed - load-factor envelopes, consideration should be given to critical flight conditions and to how the airplane will be flight tested.

Some Flight Phases of the same Category will involve the same, or very similar, Airplane Normal States; so one set of Flight Envelopes may represent several Flight Phases. Each Flight Phase will involve a range of loadings. Generally it will be convenient to represent this variation by superimposing boundaries for the discrete loadings of Table XVI, 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.

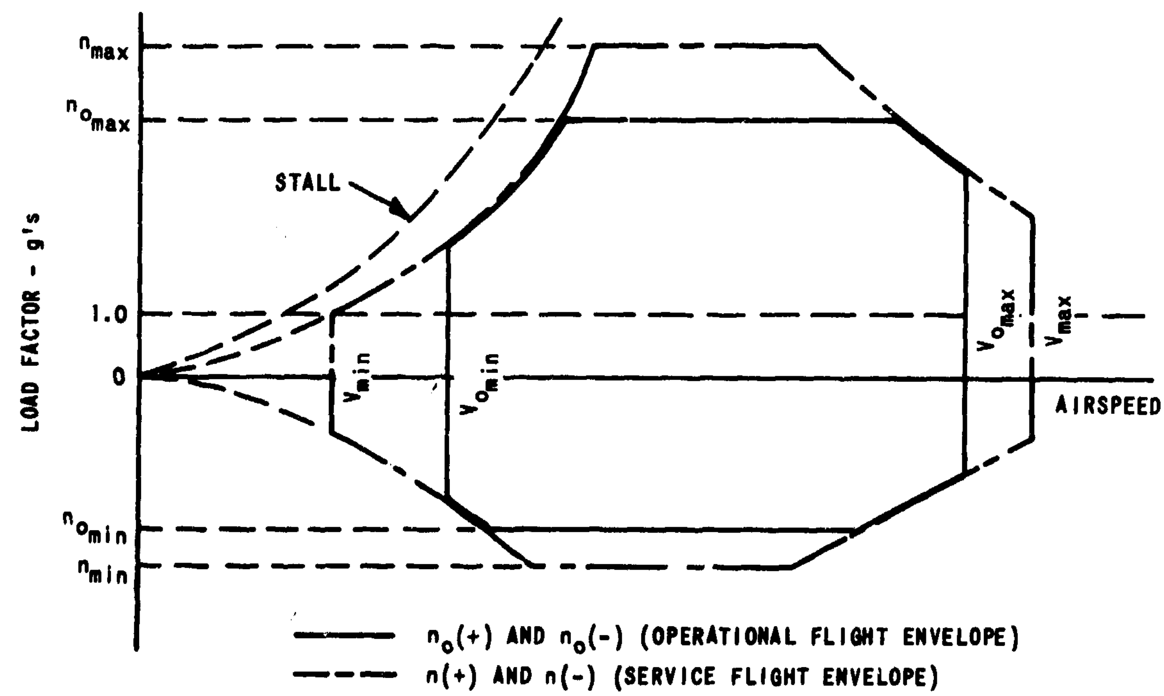
Flight tests will be conducted to evaluate the airplane against requirements in known (a priori) Flight Envelopes. Generally, flight tests will cover the Service Flight Envelope, with specific tests (stalls, dives, etc.) to the Permissible limits. The same test procedures usually apply in both Service and Operational envelopes; only the numerical requirements and qualitative levels differ. If, for example, speed and altitude are within the Operational Flight Envelope but normal load factor is between the Operational and Service Flight Envelope Boundaries, the requirements for the Service Flight Envelope apply. Ideally, the flight test program should also lead to definition of Flight Envelopes depicting Level 1 and Level 2 boundaries (paragraph 1.5). These Level boundaries should aid the using commands in tactical employment, even long after the procurement contract has been closed out.

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

A sketch in Section 6.2.5 of Reference A1 illustrates the specification nomenclature for the Service and Operational Flight Envelopes.

In all the Flight Envelopes, n denotes maneuverability aside from the influence of thrust available. The flying qualities specification places no requirements on the magnitude of load-factor capability in constant-speed, level flight.

Sketch from Section 6.2.5



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 airplane must be capable of operating in order to accomplish the missions of 3.1.1. 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.

RELATED MIL-F-8785 PARAGRAPHS

3.1.2, 3.1.3, 3.1.3.1, Table II

DISCUSSION

This paragraph is essentially new, incorporating a somewhat different concept of "operational flight envelopes" than that of MIL-F-8785, paragraph 3.1.3. Table II of MIL-F-8785 was an attempt to define the most important flight regions arbitrarily for application of some of the requirements.

Operational Flight Envelopes are now regions in speed-altitude-load factor space where it is necessary for an airplane, in the configuration and loading associated with a given Flight Phase, to have very good flying qualities, as opposed, for example, to regions where it is only necessary to ensure that the airplane 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 airplane 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 airplane should, to the extent possible, be given in the detail specification for the airplane, 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 airplane 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.

Table I

OPERATIONAL FLIGHT ENVELOPE

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

*Appropriate to the operational mission.

Guidance for establishing Operational Flight Envelopes for various Flight Phases is contained in Table I of Reference A1. The detail specification should be as specific as possible about the speed and altitude ranges over which stated load-factor capabilities are required. Obviously limit load factor cannot be attained at a lift-limited combat ceiling; but normally it would be insufficient at a lower altitude to have n_L capability at only one speed. The procuring activity should, further, assure that the Operational Flight Envelopes encompass the flight conditions at which all appropriate performance guarantees will be demonstrated.

In setting the minimum approach speed, $V_{0_{min}}(PA)$, care should be taken to allow sufficient stall margin. Commonly 1.2 V_S has been used for military land-based airplanes and 1.15 V_S for carrier-based airplanes. FAR Part 25 (Reference A6) specifies 1.3 V_S for landing-distance calculations; while Part 23 (Reference A19) specifies approach at 1.5 V_S for these calculations when required.

3.1.8 SERVICE FLIGHT ENVELOPES

REQUIREMENT

3.1.8 Service Flight Envelopes. For each Airplane Normal State, the contractor shall establish, subject to the approval of the procuring activity, Service Flight Envelopes showing combinations of speed, altitude, and normal acceleration derived from airplane limits as distinguished from mission requirements. For each applicable Flight Phase and Airplane Normal State, the boundaries of the Service Flight Envelopes can be coincident with or lie outside the corresponding Operational Flight Envelopes, but in no case shall they fall inside those Operational boundaries. The 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, and 3.1.8.4.

3.1.8.1 Maximum service speed. The maximum service speed, V_{max} or M_{max}, for each altitude is the lowest of:

a. The maximum permissible speed

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

c. The maximum airspeed at MAT, for each altitude, for dives (at all angles) from V_{MAT} at all altitudes, from which recovery can be made at 2000 feet above MSL or higher without penetrating a safe margin from loss of control, other dangerous behavior, or intolerable buffet, and without exceeding structural limits.

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

a. 1.1 V_s

b. V_s + 10 knots equivalent airspeed

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

d. The lowest speed at which level flight can be maintained with MRT and, for Category C Flight Phases:

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

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

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

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

(b) The steady load factor corresponding to the minimum allowable stall warning angle of attack (3.4.2.2.2)

(c) The steady load factor at which the elevator control is in the full airplane-nose-up [nose-down] position

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

RELATED MIL-F-8785 PARAGRAPHS

3.1.3, 3.1.4

DISCUSSION

The Service Flight Envelope encompasses the Operational Flight Envelope for the same Flight Phase and Airplane Normal State. Its larger volume denotes the extent of flight conditions that can be encountered without fear of exceeding airplane limitations (safe margins should be determined by simulation and flight test). Requirements are less severe than in the Operational Flight Envelope, but still stringent enough that a pilot can accomplish the mission Flight Phase associated with the Airplane Normal State. 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 progress 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 airplane, and it also allows for possible inadvertent flight outside the Operational Flight Envelope.

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

For engine failure during take-off, Reference A1 requires control at speeds down to V_{min} (T`; but requirements for engine-out climb capability are left to performance specifications.

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 airplane is both allowable and possible. These are the boundaries of flight conditions outside the Service Flight Envelope which the airplane is capable of safely encountering. Stalls, spins, zooms, and some dives may be representative of such conditions. The Permissible Flight Envelopes define the boundaries of these areas in terms of speed, altitude, and load factor.

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

(a) Limit speed based on structural considerations

(b) Limit speed based on engine considerations

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

(d) Maximum dive speed at MAT for each altitude, for dives (at all angles) from V_{MAT} at all altitudes, from which dive recovery at 2000 feet above MSL or higher is possible without encountering loss of control or other dangerous behavior, intolerable buffet or structural vibration, and without exceeding structural limits.

3.1.9.2 Minimum permissible speed. The minimum permissible speed in 1g flight is V_S as defined in 6.2.2 or 3.1.9.2.1.

3.1.9.2.1 Minimum permissible speed other than stall speed. For some airplanes, considerations other than maximum lift determine the minimum permissible speed in 1g flight (e.g., ability to perform altitude corrections, excessive sinking speed, ability to execute a wave-off (go-around), etc.). In such cases, an arbitrary angle-of-attack limit, or similar minimum speed and maximum load factor limits, shall be established for the Permissible Flight Envelope, subject to the approval of the procuring activity. This defined minimum permissible speed shall be used as V_S in all applicable requirements.

RELATED MIL-F-8785 PARAGRAPHS

3.1.2, 3.1.4, 3.1.4.1

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, are cut short of these limits.

The maximum permissible speed in dives or level flight, and the minimum permissible speed in level flight, can and must be defined for pilots' information. Additionally, some minimum airspeed may need to be defined for zooms, to assure recoverability. For maneuvers such as spins, no minimum permissible speed is normally stated; one accepts the low airspeed attained in the maneuver, satisfactory recovery being the only criterion. The contractor must define at least the boundaries of 3.1.9.1 and 3.1.9.2 or 3.1.9.2.1.

Paragraph 3.1.9.2.1 allows an increase in the minimum permissible speed to correspond to a minimum usable speed, subject to procuring activity approval, so that the requirements need not apply in regions where the airplane would not be flown.

To allow for upsets, phugoid oscillations and other inadvertent excursions beyond placard speed, some margin is often needed between the maximum permissible speed and the high-speed boundaries of the Operational and Service Flight Envelopes. The airplane should meet the minimal requirements of 3.1.10.3.3 in that margin. Such a margin is not specified because no satisfactory general requirement could be formulated at this time. For specific designs, the procuring activity should consider setting additional requirements, for example 1.1 V_H (commonly used for structural specification) or the upset requirements of FAR Part 25 (Reference A6) and Advisory Circular AC 25.253-1 (Reference A20).

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 airplane 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 airplane 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 Airplane Normal States. The minimum required flying qualities for Airplane Normal States (3.1.6.1) are as shown in table II.

TABLE II. Levels for Airplane Normal States

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

3.1.10.2 Requirements for Airplane Failure States. When Airplane 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. At intervals established by the procuring activity, the contractor shall determine, based on the most accurate available data, the probability of occurrence of each Airplane Failure State per flight and the effect of that Failure State on the flying qualities within the Operational and Service Flight Envelopes. These determinations shall be based on MIL-STD-756 except that (a) all airplane components and systems are assumed to be operating for a time period, per flight, equal to the longest operational mission time to be considered by the contractor in designing the airplane, and (b) each specific failure is assumed to be present at whichever point in the Flight Envelope being considered is most critical (in the flying qualities sense). From these Failure State probabilities and effects, the contractor shall determine the overall probability, per flight, that one or more flying qualities are degraded to Level 2 because of one or more failures. The contractor shall also determine the probability that one or more flying qualities are degraded to Level 3. These probabilities shall be less than the values shown in table III.

TABLE III. Levels for Airplane 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

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 Requirements for specific failures. The requirements on the effects of specific types of failures, e.g., propulsion or flight control system, 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 and terminal Flight Phases. Some requirements pertaining to takeoff, landing, and taxiing involve operation outside the Operational, Service and Permissible Flight Envelopes, as at V_S or on the ground. When requirements are stated at conditions such as these, the Levels shall be applied as if the conditions were in the Operational Flight Envelope.

3.1.10.3.2 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 Airplane 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 Envelopes, 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 on stall, spin, and dive characteristics, on dive recovery devices, and on approach to dangerous flight conditions shall also apply.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

Concept

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, 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 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 Reference A1 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 also imposed for flight within the Service Flight Envelope, for both Normal and Failure States. Outside the Service Flight Envelope, most of the requirements of MIL-F-8785B do not apply. There is a qualitative requirement in 3.1.10.3.3, and 3.1.10.3.1 and 3.1.10.3.3 refer 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 aircraft. To illustrate this, the following table presents actual control system failure information for several piloted aircraft:

Source	System	Mean Time Between Malfunctions (MTBM)
Ref J65	F-101B	86 hours
Ref J65	F-104	300 hours
Ref J65	F-105D (Flight Control plus Electronics)	14 hours
Ref J65	E-1B	185 hours
Ref J66	B-58	20 hours

Unfortunately the flying qualities effects of the reported failures are not given along with the above data. Reference J67 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(Level 2) = 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-105D	.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 A13) and Anglo-French (Reference A11) 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 B108 indicates the following aircraft accident loss rates during 1967. Also shown is the probability of aircraft loss, per 4-hour flight, for an assumed exponential loss distribution.

Aircraft	1967 Loss Rate (Losses/100,000 hrs)	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 B109 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-S-38130A (safety engineering). These related specifications are, in turn, mandatory for use by all Departments and Agencies of the Department of Defense. Implementation of the flying qualities specification is, for the most part, a union of the work required by these related specifications with normal stability and control analysis.

Failure States influence the 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 sweep forward during descent would require consideration of a wings-aft landing that otherwise would never be encountered. There are failures that would always result in an aborted mission, even in a war emergency. The pertinent Flight Phases after such failures would be those required to complete the aborted (rather than the planned) mission. For example, failure of the flaps to retract after takeoff might mean a landing with flaps at the take-off setting, with certain unexpended external stores; but supersonic cruise would be impossible. If the mission might be either continued or aborted, both contingencies need to be examined.

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

Note that specific Special Failure States (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.

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: 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.

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 effectiveness. The net result can be system improvement with a minimum expenditure of effort. Examples, using a similar approach, are presented in References J68, J69, J70 and J71.

Notes

Additional insight is given in paragraph 6.7 of Reference A1:

"6.7 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.

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

a. Identify those Airplane 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 Airplane 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 Airplane Failure State in terms of Levels as defined in the specific requirements

e. Determine the most critical Airplane 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 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. An example which illustrates an approximate estimate of the probabilities of encounter follows: if the failures are all statistically independent, determine the sum of the probabilities of encountering all Airplane Failure States which degrade flying qualities to Level 2 in the Operational Envelope. This sum must be less than 10^{-2} per flight.

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

a. Improve the airplane 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 Airplane Failure States (with the exception of Special Failure States) the flying qualities shall not degrade below Level 3.

"6.7.2 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. 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 Airplane Normal States (3.1.10.1). Airplane Failure States that do not degrade flying qualities beyond this common boundary are not considered in meeting the requirements of 3.1.10.2. Airplane 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.

"6.7.3 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. 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 Airplane Failure States might fall in the Airplane Special Failure State classification (3.1.6.2.1)."

For predicting failure, no account is taken of the likelihood of the different possible flight conditions. A given flight may be entirely within the Operational Flight Envelope or largely outside it, as with practice stalls. The flight may involve many Phases or only a few, as with practice approaches. In view of these factors and normal changes in operational use, it seems impractical in a design specification to apportion time among Flight Phases or other flight conditions for this purpose.

Special Applications

Paragraphs 3.1.10.2.1 through 3.1.10.3.3 enumerate special applications of the Level concept, tailoring it to fit several different instances. Pertinent paragraphs of Reference A1 are listed below:

3.1.10.2.1 Requirements for specific failures.

3.3.9-3.3.9.5, 3.4.9, 3.4.10, 3.5.5-3.5.5.2, 3.4.2.4.1

3.1.10.3.1 Ground operation and terminal Flight Phases.

3.2.3.3-3.2.3.3.2, 3.2.3.4, 3.2.3.4.1, 3.3.7-3.3.7.3,
3.3.9-3.3.9.5, 3.4.1.2

3.1.10.3.2 When Levels are not specified.

Paragraphs too numerous to mention.

3.1.10.3.3 Flight outside the Service Flight Envelope.

3.2.3.6, 3.3.8, 3.4.1-3.4.3

3.2 · LONGITUDINAL FLYING QUALITIES

3.2 LONGITUDINAL FLYING QUALITIES

DISCUSSION

Section 3.2 deals with basically the same subjects treated in Section 3.3 of MIL-F-8785. The major changes are in the areas of static stability, phugoid stability, short-period response, stick force per g, and stick forces in sudden pull-ups. A new requirement has been added concerning operation on the backside of the power-required curve.

In addition to extensive changes to individual requirements, the entire longitudinal flying qualities section has been reorganized in an attempt to group paragraphs more logically. All the subjects having to do with long-term stability and trim changes with speed have been grouped under 3.2.1, longitudinal stability with respect to speed. All the subjects dealing with short-term response to rapid control inputs at essentially constant speed are grouped under 3.2.2, longitudinal maneuvering characteristics. Miscellaneous control effectiveness and force requirements then follow under 3.2.3, longitudinal control.

The MIL-F-8785 paragraphs dealing with longitudinal trim changes caused by actuation of drag devices, gear, flaps, etc. have been moved to Section 3.6, characteristics of secondary control systems. This was done because these particular trim changes are primarily characteristic of these secondary control devices, rather than the basic airframe.

3.2.1 LONGITUDINAL STABILITY WITH RESPECT TO SPEED

DISCUSSION

The requirements of the subparagraphs under 3.2.1 deal with long-term stability with respect to speed. The major topics discussed are static stability, phugoid damping, and flight-path stability.

Static stability and phugoid damping describe the airspeed response characteristics when the stick is either free or fixed. Static stability means that restoring pitching moments are generated when the airspeed is disturbed from trim. This airspeed stiffness gives rise to a second-order phugoid mode, which is usually oscillatory with slightly positive or negative damping, but may also be overdamped (two negative real roots). Static instability, on the other hand, means that the phugoid mode has degenerated into two first-order modes, one of which is an aperiodic divergence. For ease of analysis and test, requirements on control gradients with speed were selected as the means to prevent aperiodically divergent modes.

The relationships between static and phugoid stability are discussed more fully under 3.2.1.1. The static stability requirement may be somewhat conservative at high speed. Air density variation during the altitude oscillations tends to shorten the phugoid period; but the corresponding change in Mach number as temperature varies with altitude, or the change in thrust with altitude, could either increase or decrease stability. These effects (Reference B11) will not be apparent in static stability tests at constant altitude.

Phugoid period and damping have meaning only in nominally level or near-level flight. At appreciable flight-path angles the characteristics may change significantly during even one period of the motion. Nevertheless, meeting the static stability requirement at all altitudes in level flight with a climb thrust setting helps assure that the long-period motion in a climb is bounded. All the discussions of the static stability and phugoid paragraphs are restricted to near-level flight.

The discussions of dynamic theory are in terms of the stability-axes equations of motion as presented in Reference B73 for a straight-and-level-flight operating point:

$$(s - X_u)u \quad - X_w w + g\theta = X_\delta \delta$$

$$- Z_u u + (s - Z_w) w - V s\theta = Z_\delta \delta$$

$$- M_u u - (M_{\dot{w}} s + M_w) w + (s - M_q) s\theta = M_\delta \delta$$

These equations neglect atmospheric gradients with altitude, but they are still adequate for the present purpose. The air-density gradient introduces an additional, but exceedingly small, characteristic root which is neglected here and in Reference A1.

3.2.1.1 LONGITUDINAL STATIC STABILITY

REQUIREMENT

3.2.1.1 Longitudinal static stability. There shall be no tendency for the airspeed to diverge aperiodically when the airplane is disturbed from trim with the cockpit controls fixed and with them free. This requirement will be considered satisfied if the variations of elevator control force and elevator control position with airspeed are smooth and the local gradients stable, with:

Trimmer and throttle controls not moved from the trim settings by the crew, and

1g acceleration normal to the flight path, and

Constant altitude

over a range about the trim speed of ±15 percent or ±50 knots equivalent airspeed, whichever is less (except where limited by the boundaries of the Service Flight Envelope). Stable gradients mean increasing pull forces and aft motion of the elevator control to maintain slower airspeeds and the opposite to maintain faster airspeeds. 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 range.

RELATED MIL-F-8785 PARAGRAPHS

3.3.1, 3.3.1.1, 3.3.2, 3.3.2.1

DISCUSSION

Introduction

Paragraphs 3.3.1, 3.3.1.1, 3.3.2, and 3.3.2.1 of MIL-F-8785 are written in terms of neutral points, control force variations with speed, and static stability with respect to angle of attack at constant speed. According to the definitions in MIL-F-8785, neutral points are related to control force and position gradients with speed. The static stability with respect to angle of attack at constant speed, however, is only partially related to the neutral points as defined in MIL-F-8785 (principally because of the effect of M_u) and is very difficult to measure in flight test. Since there are many different interpretations of the terms "static stability" and "neutral point", and since MIL-F-8785 in this area is not entirely clear, it was decided to completely rewrite the static stability paragraphs.

The primary purpose of the static stability paragraphs of MIL-F-8785 is to prevent divergences in airspeed and angle of attack which might remain undetected by a busy pilot so that, at the worst, the airplane would end up in an unsafe flight condition or run out of control available for recovery. A statement banning such divergences was therefore made the primary requirement of 3.2.1.1.

Airplanes having certain types of SAS, such as maneuver-command systems, have zero gradients of control and position with speed yet can be quite stable with respect to external disturbances. If such systems meet the primary intent of 3.2.1.1, i.e., positive stability with respect to speed, paragraph 3.2.1.1 should not be interpreted as disallowing these systems.

FAR Part 25 (Reference A6) has no controls-fixed stability requirement, relying on stable force gradients and certification flight tests to assure adequate flight safety. In a procurement specification, however, it is reasonable to require more than in a regulatory specification. Reference A1 is more lenient than Reference A6 on force gradients.

Several techniques for measuring static longitudinal stability are discussed in Appendix IVA. Although the requirement is stated in terms of constant-altitude flight, one generally acceptable flight test method involves small altitude changes.

The requirement is applied over a limited range (±15% or ±50 kt) about the trim speed so that the designer will not be forced to provide stability where a pilot could be expected to retrim, or where the pilot force is excessive. Note that, in general, the curve of the pilot force versus airspeed is nonlinear.

Control-Surface-Fixed Instability

The Level 3 requirements generally apply in the worst possible Failure States. Except for approved Special Failure States, then, 3.2.1.1 does not permit basic-airframe speed instability (elevator surface fixed). Cases will arise, however, in which the procuring activity is asked to consider allowing basic-airframe instability as a Special Failure State. Even if the reliability of stability augmentation should be judged sufficiently high, or if the degree of instability seems acceptable in itself, a number of aspects of combined airframe-flight control system behavior in normal operation need to be examined before accepting appreciable instability in a Special Failure State.

Obviously, extremes of either stability or instability require more control to balance the airplane throughout an angle-of-attack range. In the stable case, at the control limit the airplane at least has a restoring tendency. But when an airplane has an unstable variation of elevator-surface position with airspeed, the surface position required to maintain off-trim airspeeds is in a direction which reduces the control available to initiate recovery to the trim speed. If the unstable gradient is large enough, the pilot could fly far enough off the trim speed that there would be no elevator control available for recovery. With the elevator against the stops, the airspeed would continue to diverge and the pilot would be powerless to prevent it from doing so. Examples of this behavior can be found in Mach tuck for subsonic airplanes and during wave-offs for some propeller-driven airplanes.

For Airplane Normal States, then, over the entire permissible range of speed and altitude, safety comparable to that of a stable basic airframe would require pilot-control and control-surface authority to balance the airplane at positive and negative ultimate load factors, with some margin of control power remaining, wherever the basic airframe is unstable. (In flight test, of course, limit load factor would not intentionally be exceeded.) For a given configuration, the elevator surface and control positions for balance determine the amount of control authority left for stabilization and control. The relative authority and interactions of command, augmentation and trim controls are important considerations. Authority and rate saturation may be particularly important consideration for dual-purpose controls such as elevons.

In both Normal and Failure States, the augmentation must maintain appropriate levels of stability in responses to both control and disturbance inputs. For a basically unstable airframe, the sizes of these inputs should be stated specifically, rather than taking the primarily qualitative approach of Reference A1. Some margin above structural design gusts and turbulence might be suitable. Hard-over failures should be made impossible in the flight control system; engine-failure transients conceivably could be critical. Large control inputs of various forms and phasing should be considered. The response to disturbances during commanded maneuvers must be considered. The effect of flight at off-trim conditions on all these factors must be examined.

In addition, the stall requirements under 3.4.2 and, if applicable, the spin recovery requirements of 3.4.3 must be met. Survivability after damage to the flight control system may be an additional consideration.

Dynamic Theory

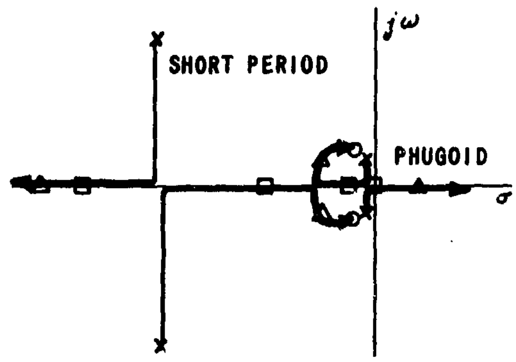
The two most common causes of static instability are the center of gravity being too far aft (M_w positive) and M_u being negative. In both these cases, a single real root will generally become unstable as the critical loading or flight condition is approached, as shown in the following sketches (the scale of the sketches is exaggerated in the vicinity of the origin). These sketches were made by treating M_ω and M_u, respectively, in the longitudinal characteristic equation (written in terms of the literal stability derivatives) as gains in a root locus analysis.

× LOCATION OF ROOTS WITH M_w WELL NEGATIVE AND $M_u = 0$

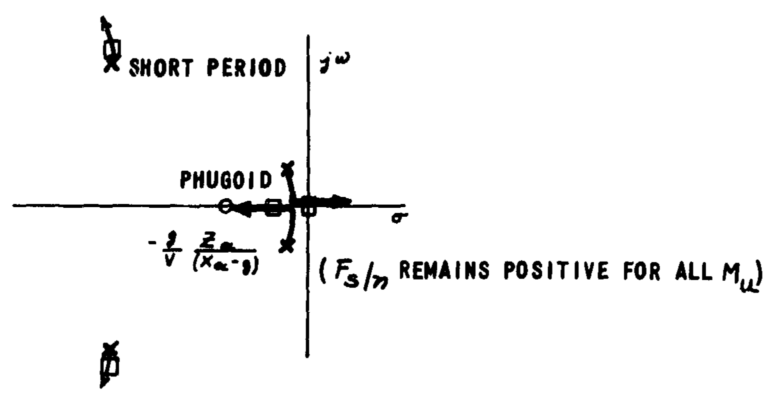
□ LOCATION OF ROOTS WITH NEUTRAL STATIC STABILITY

△ LOCATION OF ROOTS WITH $F_s/\eta = 0$

○ ZEROS: ROOTS OF $s^2 - X_u s - \frac{g}{V} Z_u$ [$\Delta(s)$ TERMS INVOLVING M_w: C.G. OR M_w EFFECT] OR $s + \frac{g}{V}\frac{Z_\alpha}{(X_\alpha - g)}$ [$\Delta(s)$ TERMS INVOLVING M_u: M_u EFFECT]



EFFECT OF M_w BECOMING POSITIVE ($M_u = 0$)



EFFECT OF M_u BECOMING NEGATIVE (NORMAL VALUES OF M_w)

When a single real root does become unstable, the presence of the unstable root can be detected 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 the constant term becomes negative:

$$g\,(Z_u\,M_w - Z_w\,M_u)$$

where the vertical force and pitching moment derivatives include the effects of engine thrust as well as airframe aerodynamics.

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 divergence 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, however; 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.

When determining whether or not there is a divergent mode with the stick free, the same simple method can be used if the stability derivatives are modified to account for elevator surface movements caused by hinge moments, downsprings, bobweights, q-bellows, auto-trim, SAS, etc. (see, for example, Reference B11). Series stability augmentation is sometimes used to modify long-term as well as short-period stability. In that case the equations can be recast in terms of u, w, q and δ_e response to control-stick inputs.

Relation of Dynamic and Static Stability

The foregoing discussion is concerned with the primary static stability requirement of 3.2.1.1 and is useful for design purposes. It is, however, extremely difficult to test directly for slightly divergent modes. The result might be confounded by being slightly out of trim, or by an aerodynamic non-linearity. Especially at high speed, it is also hard to assure a long enough patch of smooth, homogeneous air so that additional disturbances will not mask the effect of stability. Also, evaluation of slow motions is time-consuming.

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 is a speed stiffness, indicating the presence of a phugoid mode (stick-free). For example, if a steady push force is required to hold a speed above the 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 with speed are therefore a restatement of the requirement banning airspeed divergences, but in a form more directly useful for flight test purposes.

The expression for the control position gradient with speed, for example, is:

$$\frac{d\delta_s}{du} = \frac{d\delta_s}{d\delta_e} \quad \frac{Z_u M_w - M_u Z_w}{Z_w M_{\delta_e} - M_w Z_{\delta_e}}$$

A negative value of $Z_w M_{\delta_e} - M_w Z_{\delta_e}$ would result in a negative value of $(d\delta_s/dn)$. 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. But this behavior would be unacceptable according to 3.2.2.1.1 and 3.2.2.2. For all practical purposes, therefore, $Z_w M_{\delta_e} - M_w Z_{\delta_e}$ will always be positive; and stable control gradients with speed will always ensure that there are no divergent aperiodic modes present. Appendix IVA discusses flight test methods in more detail.

Data Interpretation

A certain amount of static instability might have been allowable for Level 3, as shown in References B59, D31, D34, E5, and E12. The data of References B59, D31, and D34 are presented in Figures 1 through 4 in the form of ω_n^2 versus $2\zeta\omega_n$. These terms are the stiffness and damping terms of a second-order representation of the airplane's dynamics; and as such, they describe the location of only two of the airplane's four roots. The second-order representation has the form:

$$s^2 + (2\zeta\omega_n)s + \omega_n^2 = 0$$

When ω_n^2 is positive and ζ is greater than 1.0, the two roots are real rather than complex. If the two real roots are $s = \lambda_1$ and $s = \lambda_2$, the characteristic equation has the following form:

$$(s - \lambda_1)(s - \lambda_2) = s^2 - (\lambda_1 + \lambda_2)s + \lambda_1\lambda_2 = 0$$

The damping $(2\zeta\omega_n)$ is now represented by the sum of the two real roots, and the stiffness (ω_n^2) by their product. When stiffness is negative, the two roots are real, but one is positive and the other is negative. The question now is: what are the other two roots doing in the meantime?

The statically unstable roots in Figures 1 through 4 were obtained by making M_w positive. The two roots are the short-period (oscillatory or aperiodic) roots for the statically stable cases, and the most positive and negative real roots for the statically unstable cases (see the preceding sketches). The $(2\zeta\omega_n)$ axis of Figures 1 through 4 is therefore the sum of the most stable and most unstable roots, for the statically unstable cases. Since the other two roots are always close to the origin (see the preceding sketch), $(2\zeta\omega_n)$ is a fairly good measure of the total damping.

Figures 1 through 4 all indicate that little or no static instability can be tolerated when the total damping is low, while quite a large amount can be tolerated when the damping is large. For the statically unstable case, the magnitude of the unstable root can be expressed in terms of the ordinate and abscissa of Figures 1 through 4 as follows:

$$\sigma = \frac{2\zeta\omega_n - \sqrt{(2\zeta\omega_n)^2 - 4\omega_n^2}}{2}$$

One statically unstable configuration was evaluated in the landing approach experiment of Reference E12. This configuration had s = +.20 sec^{-1} and s = -1.6 sec^{-1} and was rated 8.0. (The total damping is therefore approximately 1.4 sec^{-1}.)

Two statically unstable configurations were evaluated in the landing approach experiment of Reference E5. These configurations had s = +.19, s = -.19 and s = +.26, s = -.26, and were rated 8.0 and 8.5 respectively. The total damping of each of these pairs of roots is zero, but the damping of the λ_1, λ_2 pair is not representative of the airplane's total damping in this experiment because the unstable root was caused by making M_u negative rather than making M_w positive (see the preceding sketches). In this case it is seen that the total damping is all in the short-period pair of roots, and is actually quite good.

After studying the available data, it is obvious that many factors influence the amount of instability which can be handled. Because even a small instability can be quite dangerous under some circumstances (e.g., low total damping), it was decided to require the airplane to be statically stable, even for Level 3. Aside from the data, there is great reluctance to allow airplanes to be designed with any instabilities, because of design and requirement uncertainties, and because of the possibility of experiencing several Level 3 flying qualities simultaneously (See the discussion of 1.5, Levels of flying qualities).

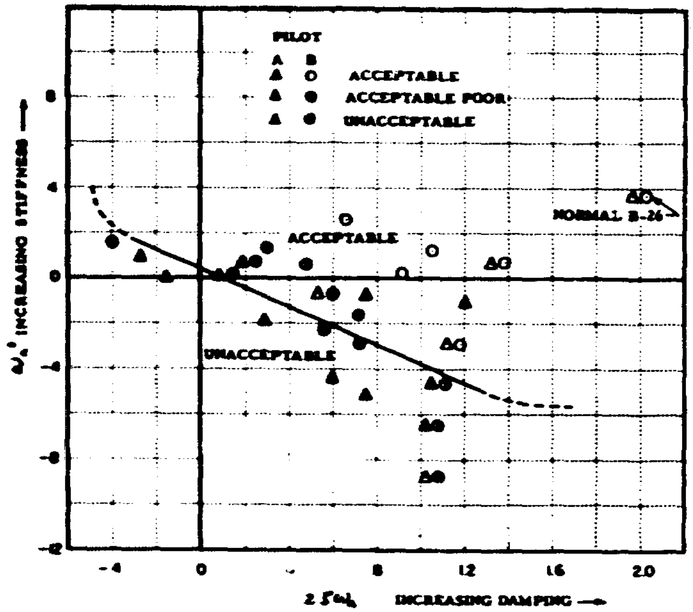


Figure 1 (3.2.1.1)
MIRROR LANDING APPROACHES - SMOOTH AIR
(B-26, REFERENCE D31)

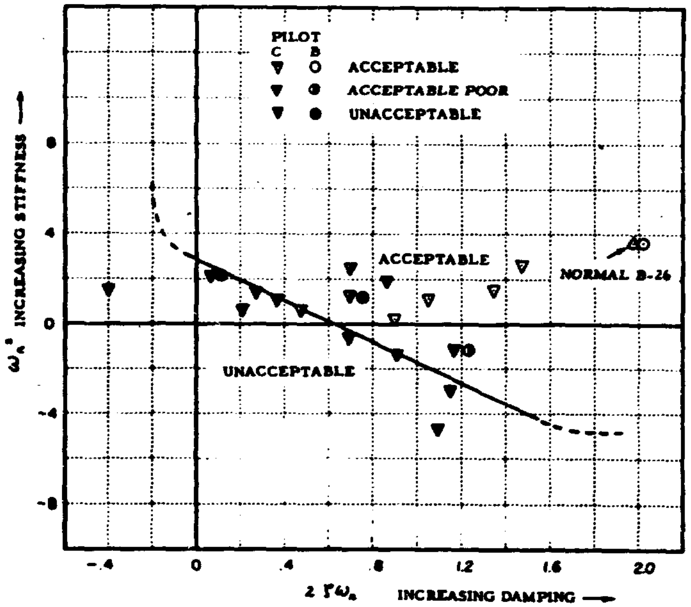


Figure 2 (3.2.1.1)
LANDING APPROACHES - ROUGH AIR
(B-26, REFERENCE D31)

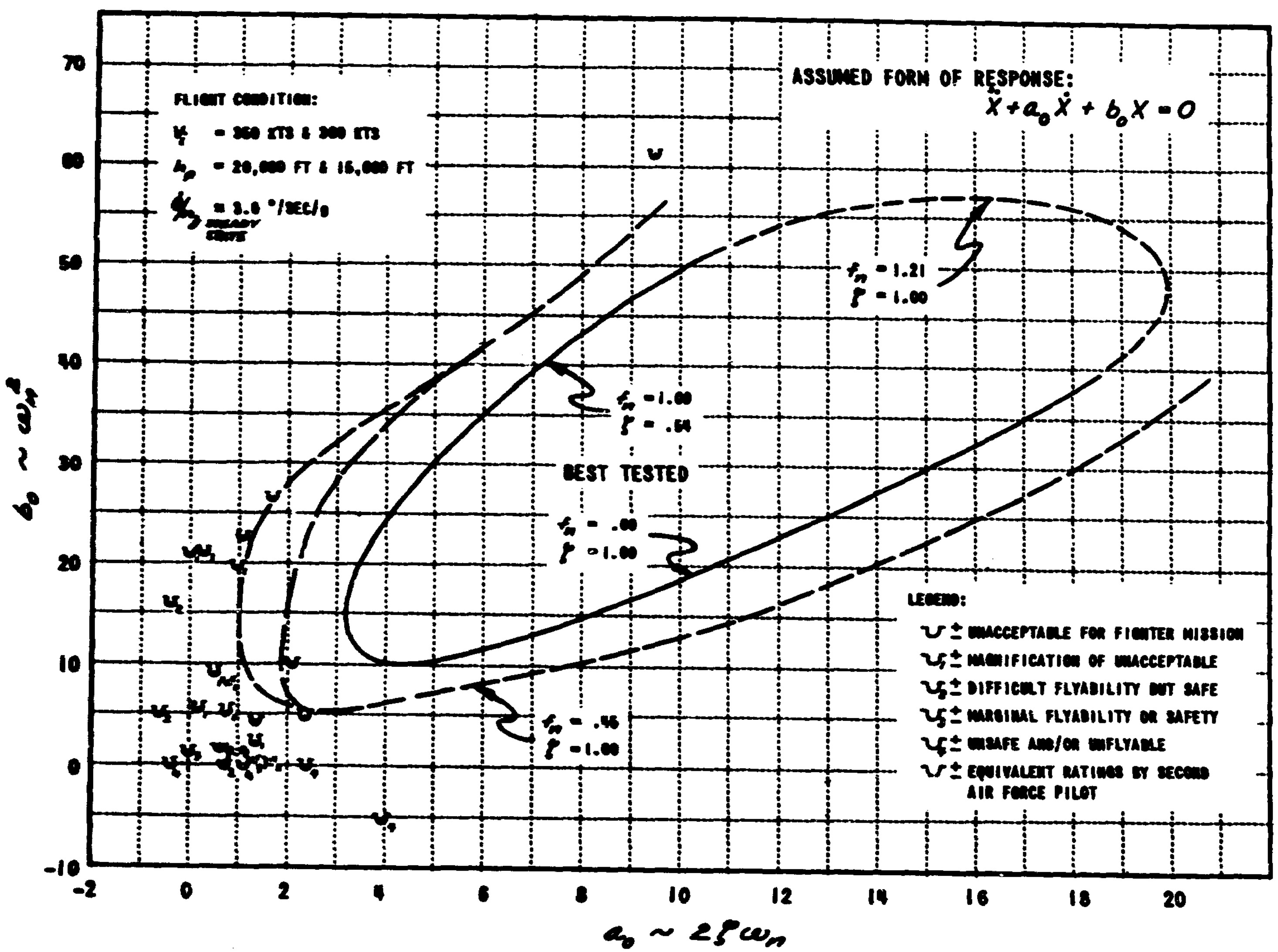


Figure 3 (3.2.1.1)
CATEGORY A FLIGHT PHASES (F-94, REFERENCE D34)

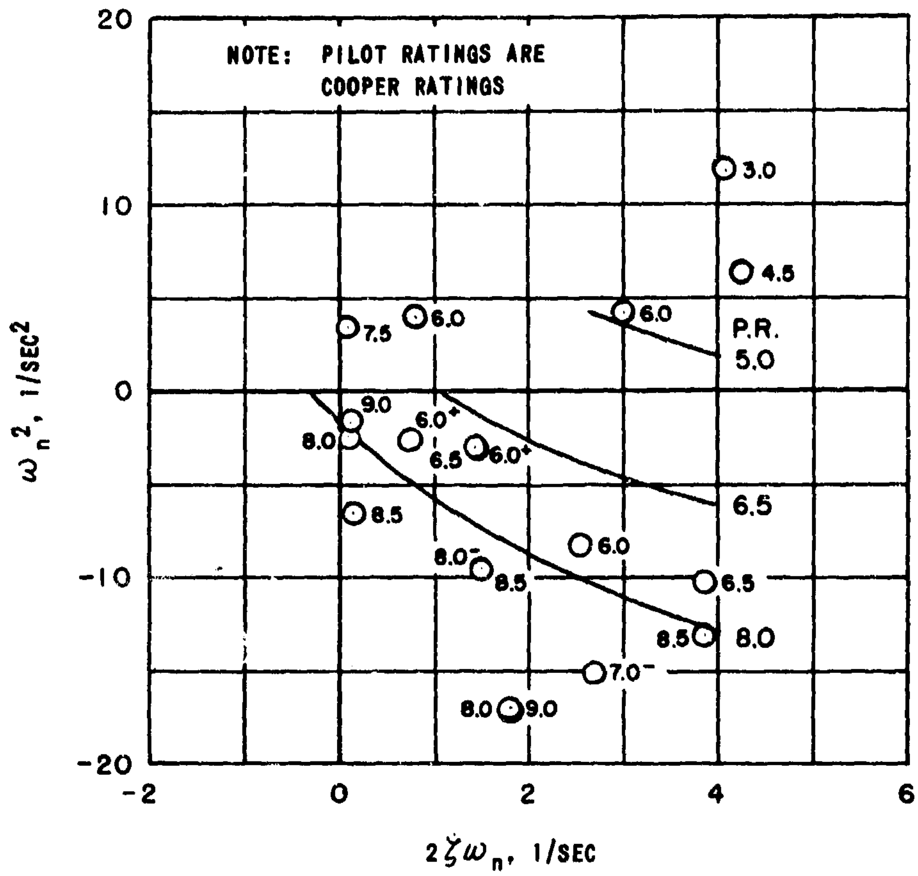


Figure 4 (3.2.1.1)
CATEGORY A FLIGHT PHASES (F-86, REFERENCE B59)

3.2.1.1.1 RELAXATION IN TRANSONIC FLIGHT

REQUIREMENT

3.2.1.1.1 Relaxation in transonic flight. The requirements of 3.2.1.1 may be relaxed in the transonic speed range provided any divergent airplane motions or reversals in slope of elevator control force and elevator control position with speed are gradual and not objectionable to the pilot. In no case, however, shall the requirements of 3.2.1.1 be relaxed more than the following:

a. Levels 1 and 2 - For center-stick controllers, no local force gradient shall be more unstable than 3 pounds per 0.01 M nor shall the force change exceed 10 pounds in the unstable direction. The corresponding limits for wheel controllers are 5 pounds per 0.01 M and 15 pounds, respectively.

b. Level 3 - For center-stick controllers, no local force gradient shall be more unstable than 6 pounds per 0.01 M nor shall the force ever exceed 20 pounds in the unstable direction. The corresponding limits for wheel controllers are 10 pounds per 0.01 M and 30 pounds, respectively.

This relaxation does not apply to Level 1 for any Flight Phase which requires prolonged transonic operation.

RELATED MIL-F-8785 PARAGRAPHS

3.3.3

DISCUSSION

Since airplanes naturally exhibit local static instabilities in the transonic region, the requirements of 3.3.3 of MIL-F-8785 were retained and mapped into the new format. Requiring elevator force and position stability for prolonged transonic operation is consistent with the description (1.5) of Level 1 flying qualities. The MIL-F-8785 relaxation seems aimed at short-time Level 1 and 2 operation. The Level 3 limits were rather arbitrarily determined by doubling the limits for Levels 1 and 2, based on limited flight test experience.

The extent of the region that may be considered transonic has been left unspecified because of the difficulty in stating a definition that can be applied with generality. It is not the intent to define the transonic region as that where a relaxation is necessary; such a definition would leave essentially no requirement for stability. For airplanes that do not have supercritical wings, the lower end of the transonic region might be taken as the drag-rise Mach number. The upper bound might be the Mach number at which the lift and drag approach the classical $(M^2\cos^2\Lambda - 1)^{-1/2}$ variation with free-stream Mach number, where Λ is the sweepback angle.

For subsonic airplanes particularly, note that there are also dive requirements (3.2.3.5, 3.2.3.6) to be met.

Since phugoid oscillations involve speed changes, all speeds at which operational missions (3.1.1) might require prolonged flight should be reasonably far removed from the region of transonic trim changes. Otherwise, normally encountered disturbances would cause divergence.

Note that, while the requirement might be met through use of stability augmentation, the Level 1 requirement for cases of prolonged transonic operation will not be satisfied just by an input to the feel system. Feel can affect the control-force gradient with speed, but not the control-position gradient.

A statement should be included in the detail specification for each procurement delineating if the relaxation is to be applied and for which Flight Phases.

3.2.1.1.2 ELEVATOR CONTROL FORCE VARIATIONS DURING RAPID SPEED CHANGES

REQUIREMENT

3.2.1.1.2 Elevator control force variations during rapid speed changes. When the airplane is accelerated and decelerated rapidly through the operational speed range and through the transonic speed range by the most critical combination of changes in power, actuation of deceleration devices, steep turns and pullups, the magnitude and rate of the associated trim change shall not be so great as to cause difficulty in maintaining the desired load factor by normal pilot techniques.

RELATED MIL-F-8785 PARAGRAPHS

3.3.3.1

DISCUSSION

This paragraph is essentially a rewording of 3.3.3.1 of MIL-F-8785, in an attempt to make it more general. There are two kinds of problems for which this requirement is primarily intended. First, airplanes can have stick force and position gradients with speed which are so stable that considerable pilot effort is required during rapid speed-change maneuvers. Second, in the transonic region the local gradients may change so rapidly with Mach number that it is difficult for the pilot to maintain the desired pitch attitude or normal acceleration during rapid speed changes.

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

3.2.1.2 PHUGOID STABILITY

REQUIREMENT

3.2.1.2 Phugoid stability. The long-period airspeed oscillations which occur when the airplane seeks a stabilized airspeed following a disturbance shall meet the following requirements:

a. Level 1 ------ ζ_p at least 0.04

b. Level 2 ------ ζ_p at least 0

c. Level 3 ------ T_2 at least 55 seconds.

These requirements apply with the elevator control free and also with it fixed. They need not be met transonically in cases where 3.2.1.1.1 permits relaxation of the static stability requirement.

RELATED MIL-F-8785 PARAGRAPHS

3.3.6

DISCUSSION

Assuming that the airplane meets the static stability requirements of 3.2.1.1, an identifiable second-order phugoid mode will probably exist. Paragraph 3.3.6 of MIL-F-8785 specifies that phugoid damping must be at least neutrally stable for phugoid periods of less than 15 seconds, and has no quantitative requirements for longer periods. These requirements are considered inadequate. Although pilots can handle airplanes having poor phugoid damping, they will make such comments as: the airplane "requires constant attention," "is frustrating to fly," and "is difficult to trim." The requirements of 3.2.1.2 were developed to prevent such problems.

For purposes of analysis, it should be understood that altitude changes during the phugoid motion become large compared to the airspeed changes during high speed flight; and the air density gradient with altitude will tend to make the phugoid period shorter than indicated by the equations discussed under paragraph 3.2.1. These density-gradient effects should be accounted for in any analytical calculations concerning the phugoid mode during high speed flight (see Reference B11).

The phugoid damping limits of 3.2.1.2 are partially based on the data and pilot comments of References D38 and E5. The data of Reference E5 were taken during IFR and VFR landing approaches, and show the importance of the "backside" parameter, $1/T_{h_1}$, on the acceptability of a given phugoid damping ratio. The data of Reference D38 were taken during cross-country IFR flights and IFR landing approaches. Some consideration was given to the idea of having different limits for different Flight Phases, e.g., Category B and C Flight Phases may require more phugoid damping than Category A Flight Phases. There are not sufficient data to do this, however.

The data of Reference E5 are presented in Figures 1 through 11. Figures 1 through 4 are for the basic T-33 phugoid frequency of .15 rad/sec. Figures 5 through 9 are for ω_{n_p} = .32 rad/sec, and Figures 10 and 11 are for ω_{n_p} = .45 rad/sec. By comparing the data for the different values of ω_{n_p}, it is quite apparent that more positive values of ζ_p are necessary as ω_{n_p} increases. For this reason, time-to-double-amplitude seems to be the appropriate parameter for specifying phugoid damping when the damping is negative. Time-to-double-amplitude can be expressed as:

$$T_2 = -\left(\frac{0.693}{\zeta_p \, \omega_{n_p}}\right)$$

When the damping is positive, the parameter ζ_p is specified because a limit on time-to-half-amplitude would allow ζ_p to decrease as ω_{n_p} increases - a trend which is opposite to that indicated by the data. Time-to-half-amplitude can be expressed as:

$$T_{1/2} = +\left(\frac{0.693}{\zeta_p \, \omega_{n_p}}\right)$$

The data of Reference D38 are presented in Figure 12.

Several of the Figures 1 through 11 do not contain enough data to determine the independent effects of ζ_p and $(1/T_{h_1})$. From those figures where the effects can be separated, however, the following Level 1 values were obtained, for $(1/T_{h_1}) = 0$:

Figure No.	ζ_p
1	0 to +0.10
2	(The basic configuration is worse than Level 1)
7	0 to +0.10
12	+0.07

In summary, the Level 1 limit on ζ_p seems to lie between 0 and +0.10. After studying typical values of ζ_p for several existing airplanes, it was decided to use ζ_p = 0.04 as the Level 1 limit.

For the Level 2 limit, the following values were obtained from the data:

Figure No.	ω_{n_p}	ζ_p	$\zeta_p \omega_{n_p}$
1	.15	-.28	-.042
2	.15	-.30	-.045
7	.32	-.17	-.054
8	.32	-.21	-.067
11	.45	-.14	-.063
12	.126	-.14	-.018

It should be noted that for the data of Figures 1, 2, and 12, negative phugoid damping was obtained by introducing positive values of $M_{\dot{u}}$. As pointed out in Reference E5, $M_{\dot{u}}$ increases the pitching response of the airplane to horizontal gusts at frequencies considerably above ω_{n_p} . This effect is illustrated in Figures 13 and 14. (The high-frequency gust response of the actual airplane was not quite as bad as indicated in these figures because low-pass filters were used in u and $\dot{u}$ feedback channels for both experiments.) The pilot comments of Reference E5 indicate that this pitching response was troublesome to the pilot. Since $M_{\dot{u}}$ is a derivative which conventional airplanes are not likely to possess, the pilot ratings associated with the configurations having $M_{\dot{u}}$ can be viewed as being somewhat pessimistic for conventional airplanes. In view of this problem, the data of Figures 7, 8, and 11 should be weighted more heavily than those of Figures 1, 2, and 12. The data then indicate that the Level 2 limit should be a time-to-double-amplitude between 10 and 13 seconds. In view of the uncertainties associated with the rather limited amount of data, it was decided that no instability would be allowed for Level 2. The Level 2 limit was therefore set at $\zeta_p = 0$.

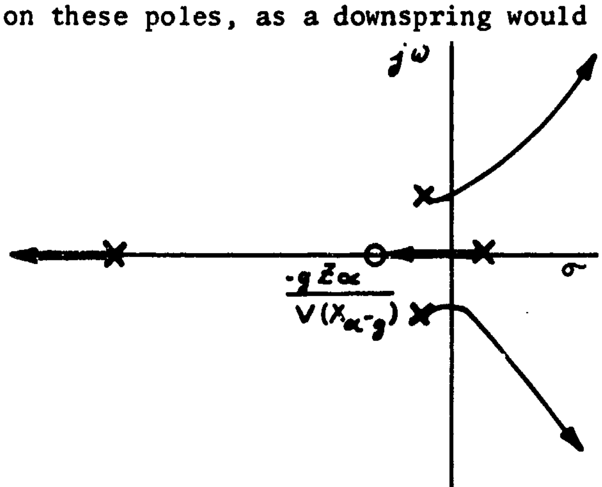
For the Level 3 limit, the following values were obtained from the data:

Figure No.	ω_{n_p}	ζ_p	$\zeta_p \omega_{n_p}$
1	.15	-.47	-.070
7	.32	-.22	-.070
8	.32	-.27	-.086
11	.45	-.18	-.081
12	.126	(The rating scale used does not extend far enough to determine a Level 3 boundary)	

These data indicate a range of time-to-double-amplitude of 8 to 10 seconds. Again, there is only a small amount of data on which to base the requirement. It was decided that it would be unwise to allow phugoid modes as unstable as the data allow, even for Level 3. A time-to-double amplitude of 55 seconds was therefore selected as a conservative limit for Level 3.

The data of Figures 5 through 11 and the associated pilot comments clearly show that there are piloting difficulties due to high values of ω_{n_p}. In fact, pilot ratings of 6.5 (CAL) are obtained for ω_{n_p} = .45 rad/sec even when ζ_p, $1/T_{h_1}$, and $\omega_{n_{sp}}$ are all good. From this fact, it might seem logical to establish upper limits on ω_{n_p}. The data were not used in this manner, however, because of the specific way in which the variable stability T-33 was augmented to obtain these high values of ω_{n_p}. The phugoid frequency was increased by using positive values of M_u, which causes a strong increase in the high-frequency pitching response of the airplane to horizontal gusts. This effect is also illustrated in Figures 13 and 14. It is very evident from the pilot comments that the configurations having high values of ω_{n_p} were given poor ratings because of this effect. The high-frequency pitching response to horizontal gusts will certainly increase with increasing ω_{n_p} regardless of how ω_{n_p} is altered, but the presence of M_u makes this effect more pronounced. What is really needed is a criterion for the pitch response to horizontal gusts, but such a criterion does not presently exist.

It should be noted that excessive amounts of positive M_u cannot only increase ω_{n_p} to undesirable values, but can also cause ζ_p to become quite unstable. For example, downsprings are often employed to stabilize statically unstable (stick-free) airplanes at aft center-of-gravity positions. If too large a spring is used, the designer will simply trade a statically unstable airplane for a dynamically unstable one, as shown in the following s-plane plot. The X's represent the unstable M_w case with zero M_u, of the first sketch in the discussion of 3.2.1.1. The root locus here shows the effects of positive M_u on these poles, as a downspring would produce stick-free.



EFFECT OF INCREASING M_u (POSITIVE)

Examples of this behavior are documented in References J51 and B96.

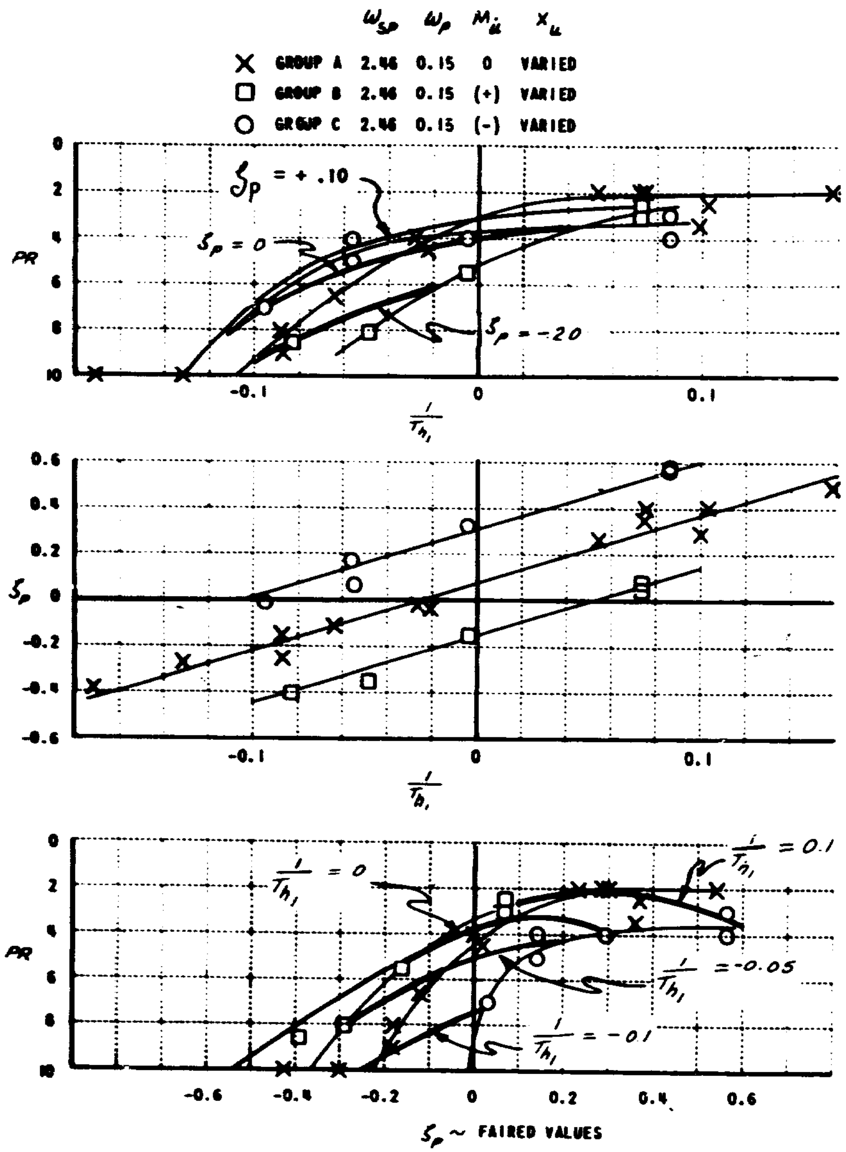


Figure 1 (3.2.1.2)
LANDING APPROACH (T-33, REFERENCE E5)

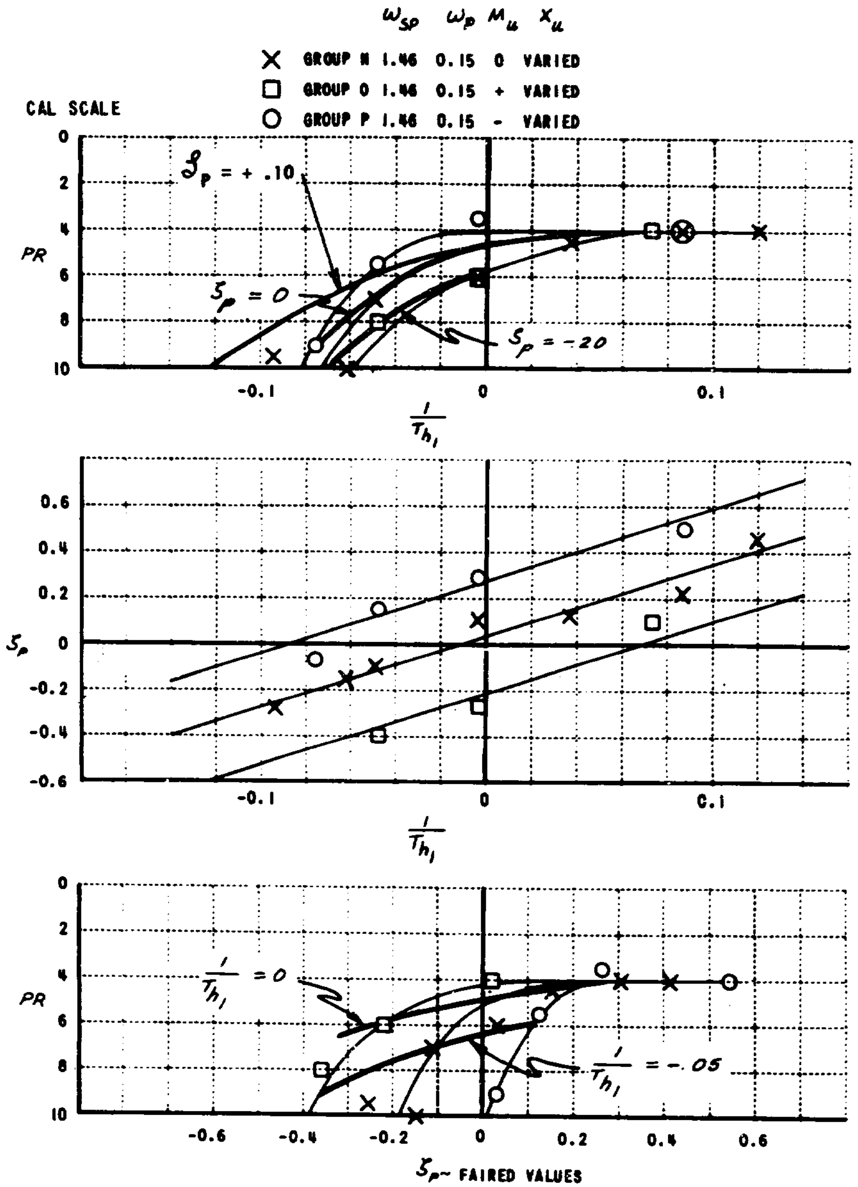


Figure 2 (3.2.1.2)
LANDING APPROACH (T-33, REFERENCE E5)

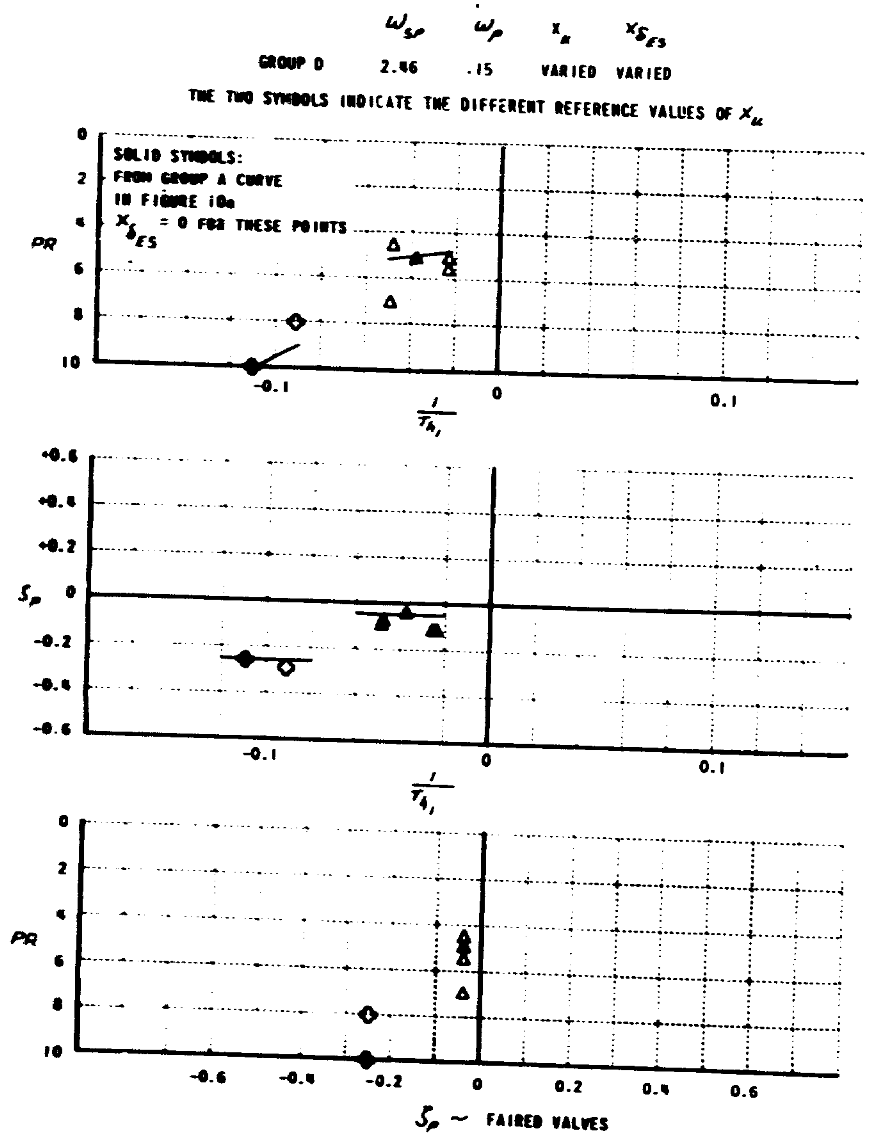


Figure 3 (3.2.1.2)
LANDING APPROACH (T-33, REFERENCE E5)

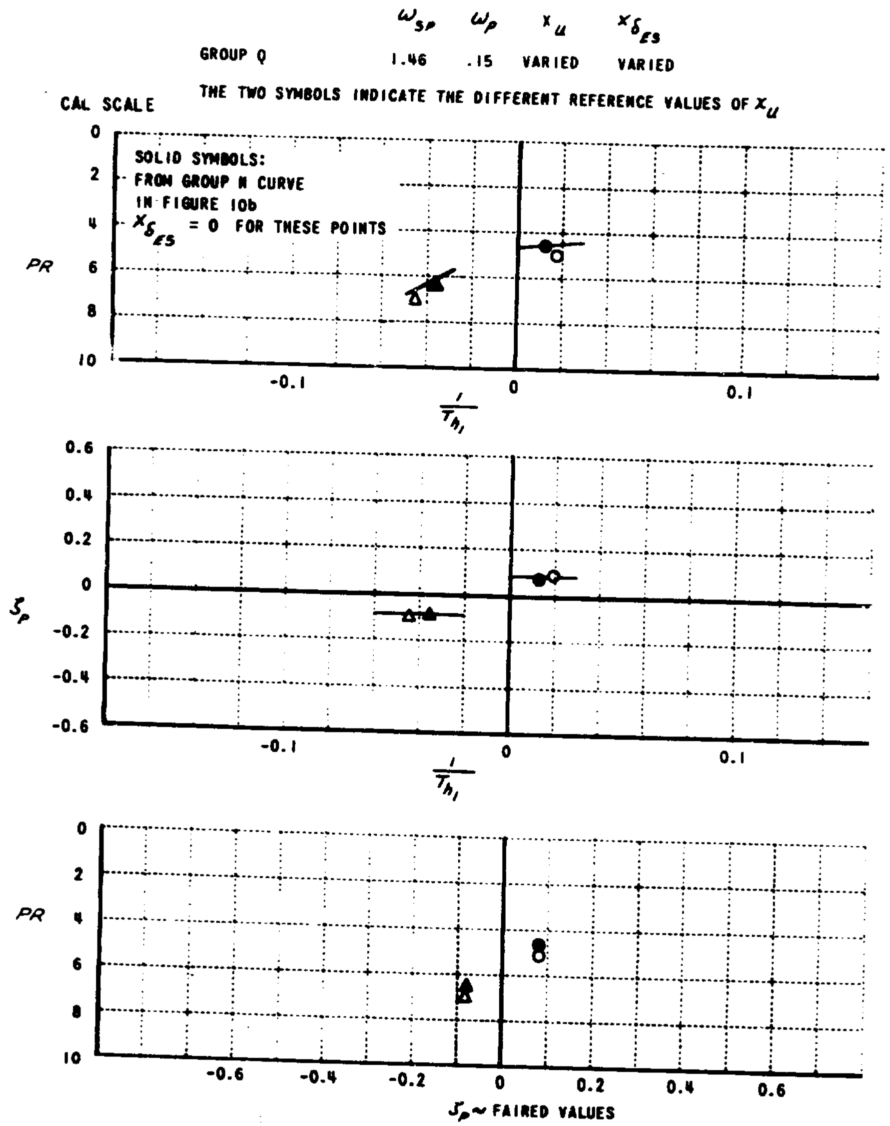


Figure 4 (3.2.1.2)
LANDING APPROACH (T-33, REFERENCE E5)

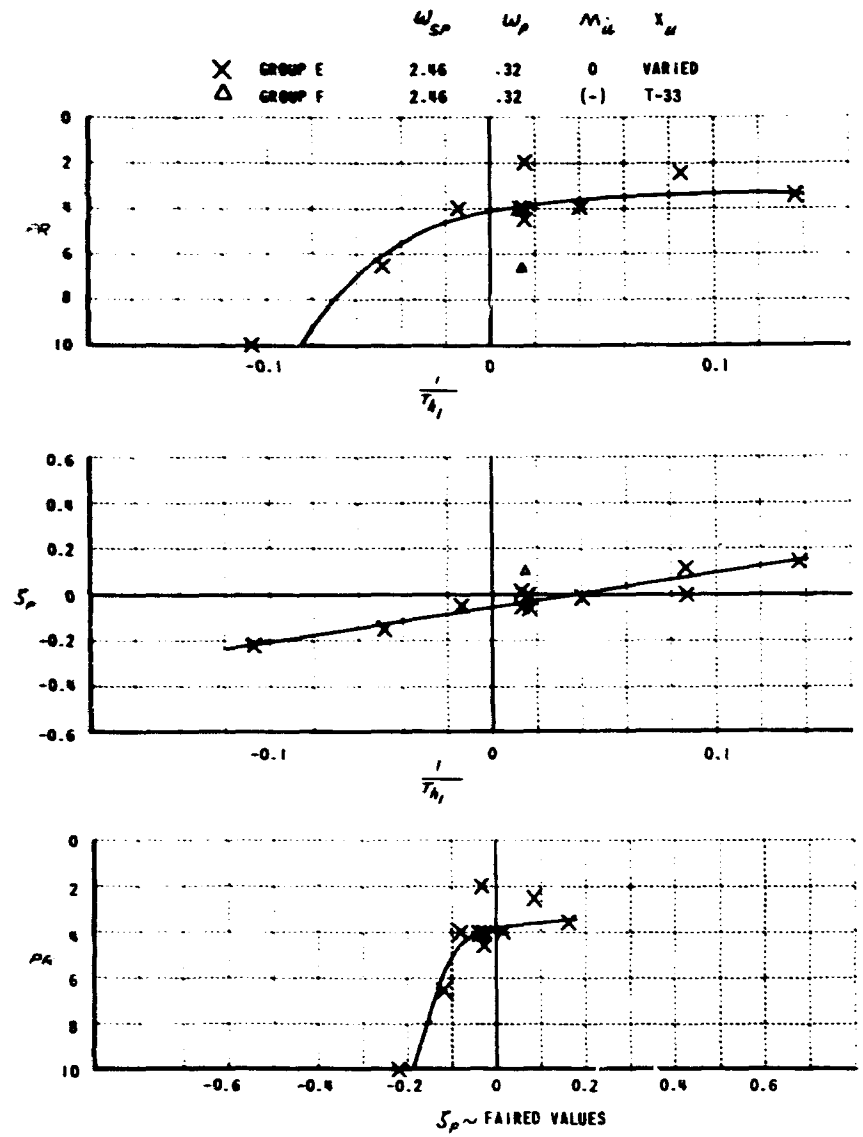


Figure 5 (3.2.1.2)
LANDING APPROACH (T-33, REFERENCE E5)

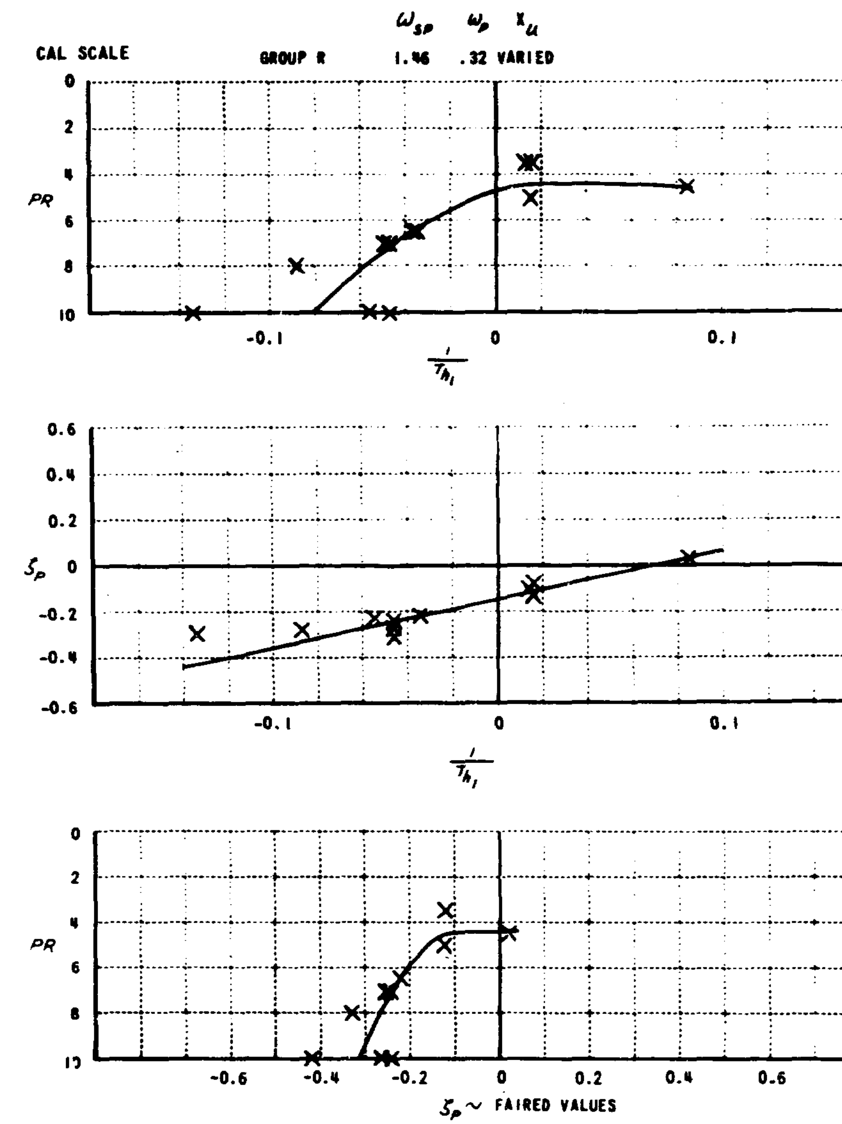


Figure 6 (3.2.1.2)
LANDING APPROACH (T-33, REFERENCE E5)

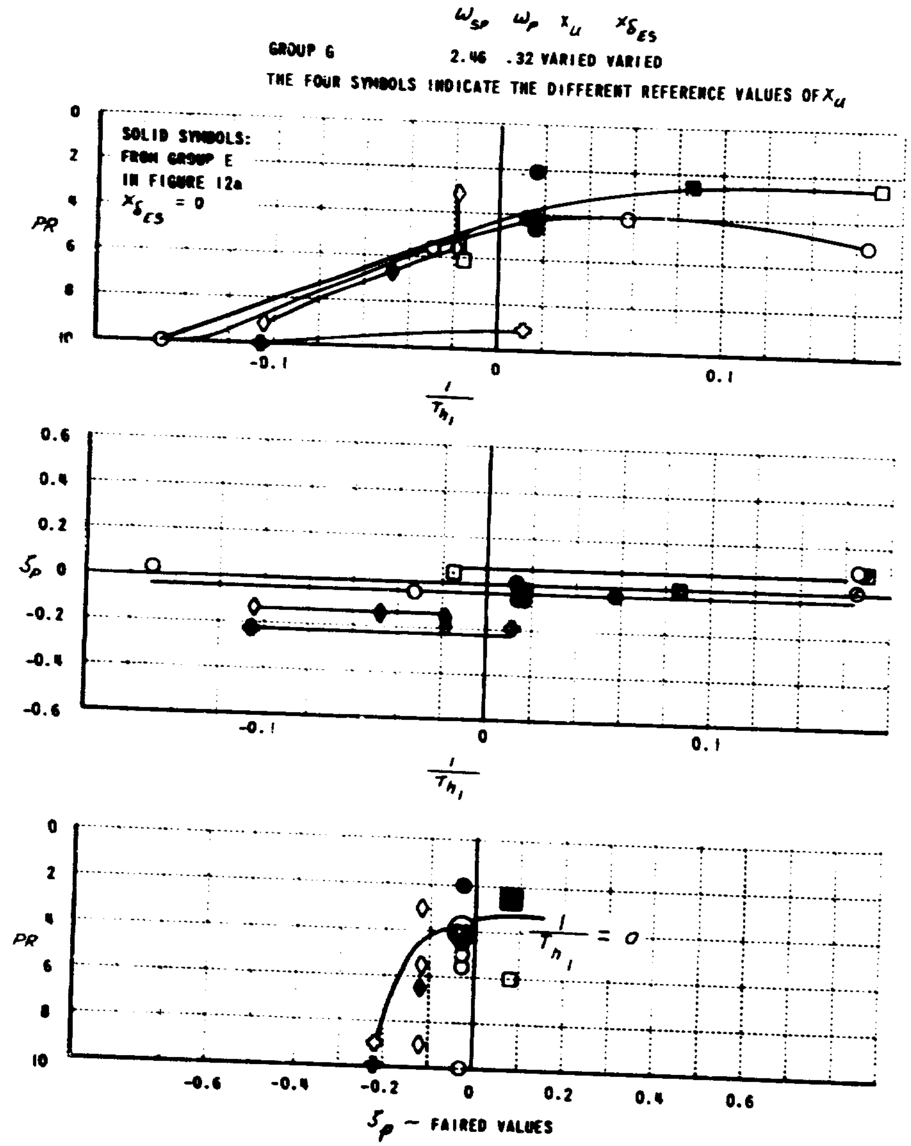


Figure 7 (3.2.1.2)
LANDING APPROACH (T-33, REFERENCE E5)

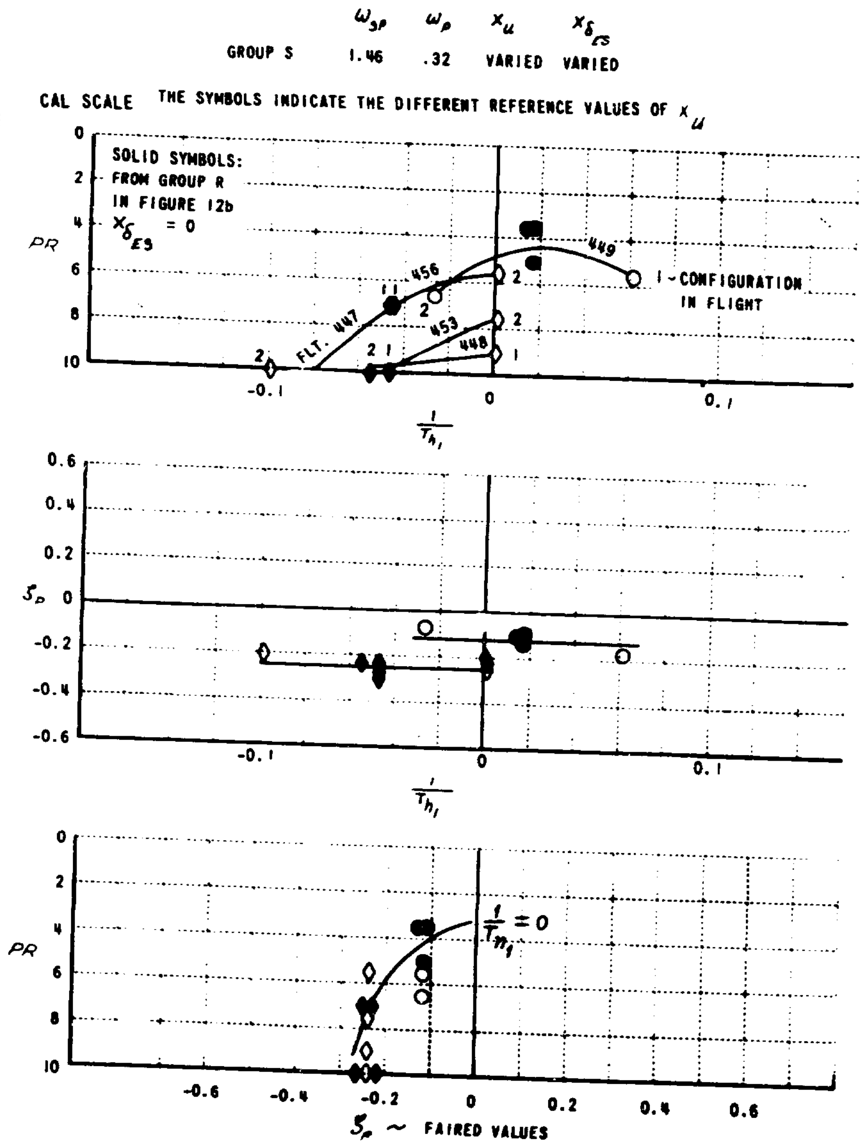


Figure 8 (3.2.1.2)
LANDING APPROACH (T-33, REFERENCE E5)

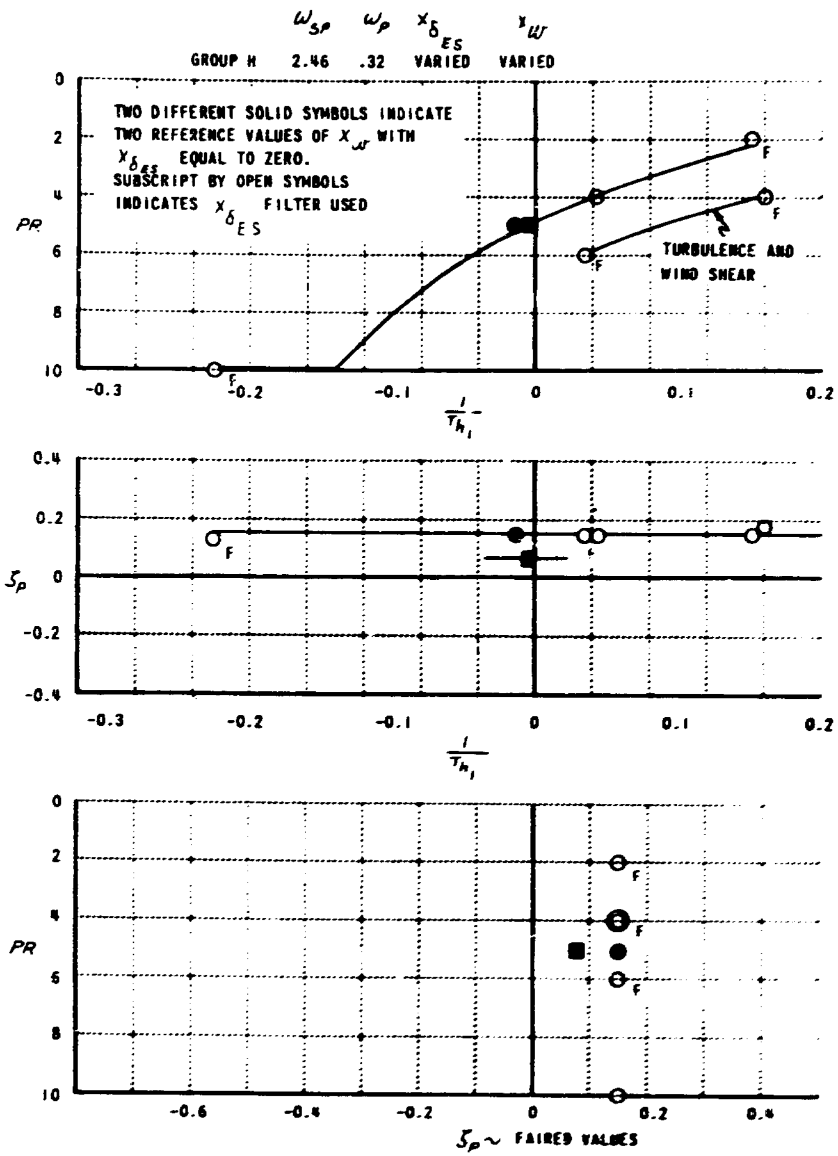


Figure 9 (3.2.1.2)
LANDING APPROACH (T-33, REFERENCE E5)

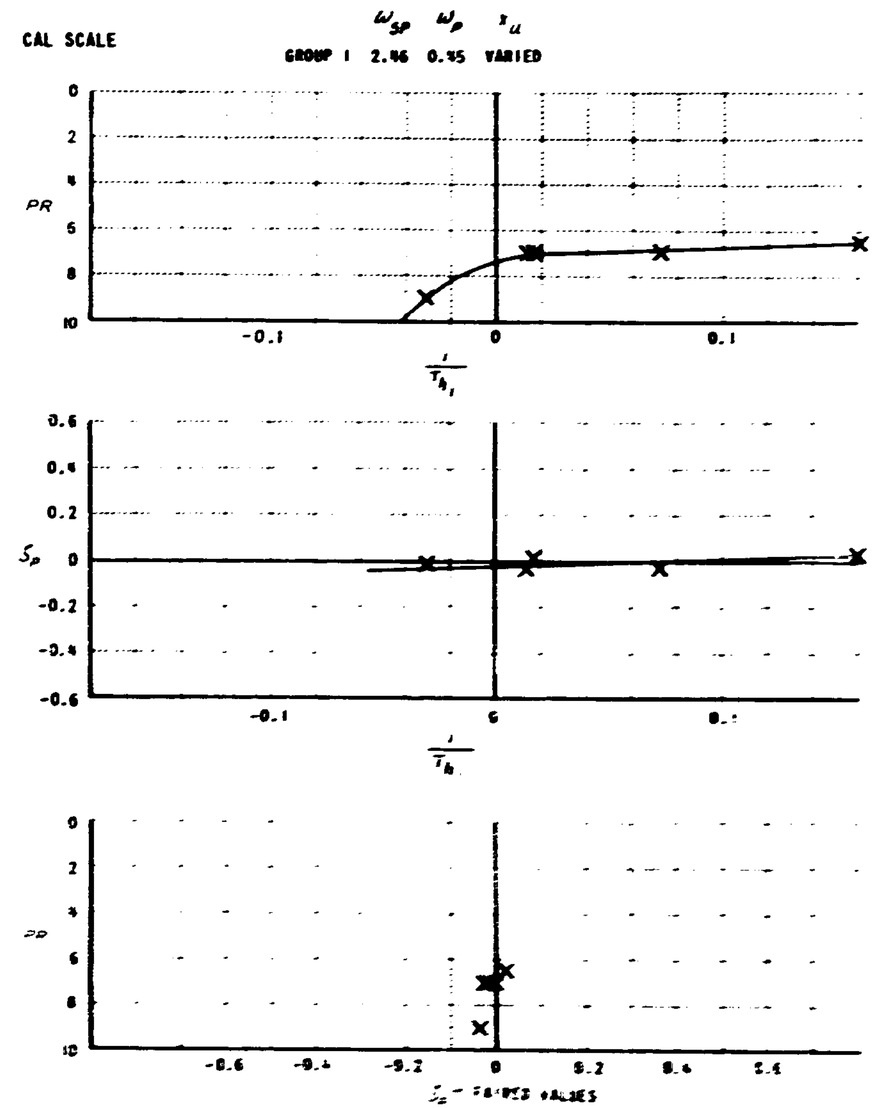


Figure 10 (3.2.1.2)
LANDING APPROACH (T-33, REFERENCE E5)

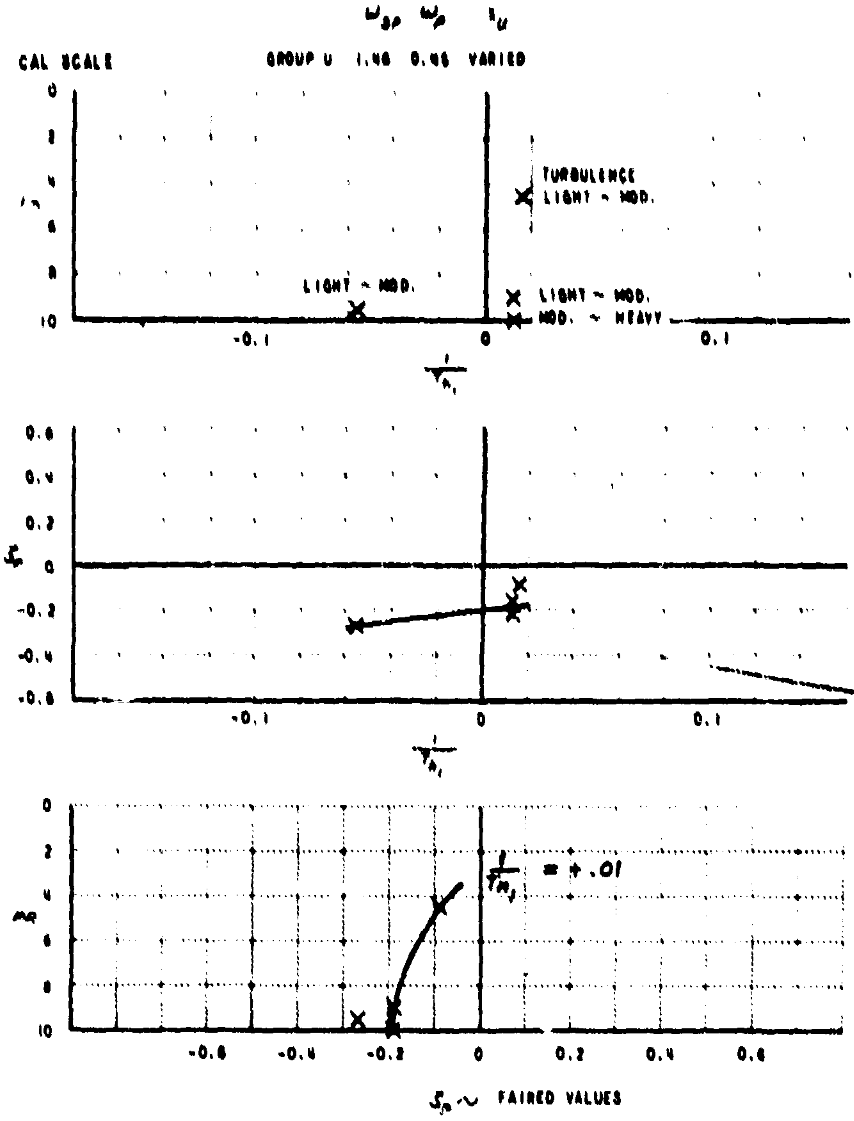


Figure 11 (3.2.1.2)
LANDING APPROACH (T-33, REFERENCE E5)

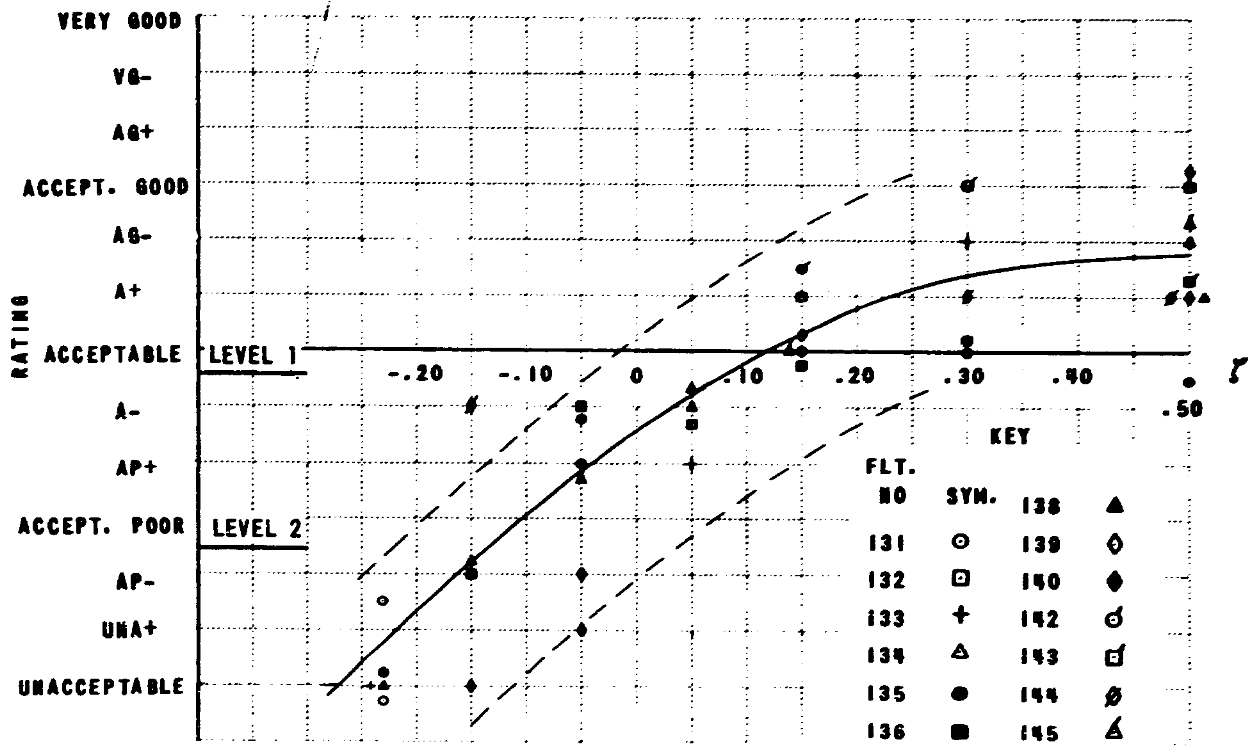


Figure 12 (3.2.1.2)
CRUISE AND LANDING APPROACH (B-26, REFERENCE D38)

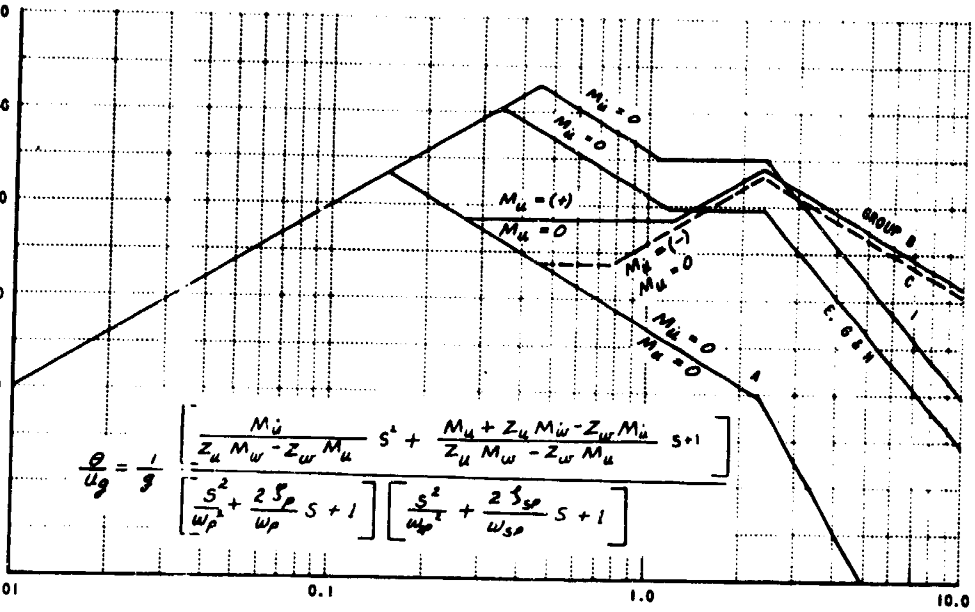


Figure 13 (3.2.1.2)
PITCH RESPONSE TO HORIZONTAL GUSTS FOR $\omega_{n_{sp}} = 2.46$
(T-33, REFERENCE E5)

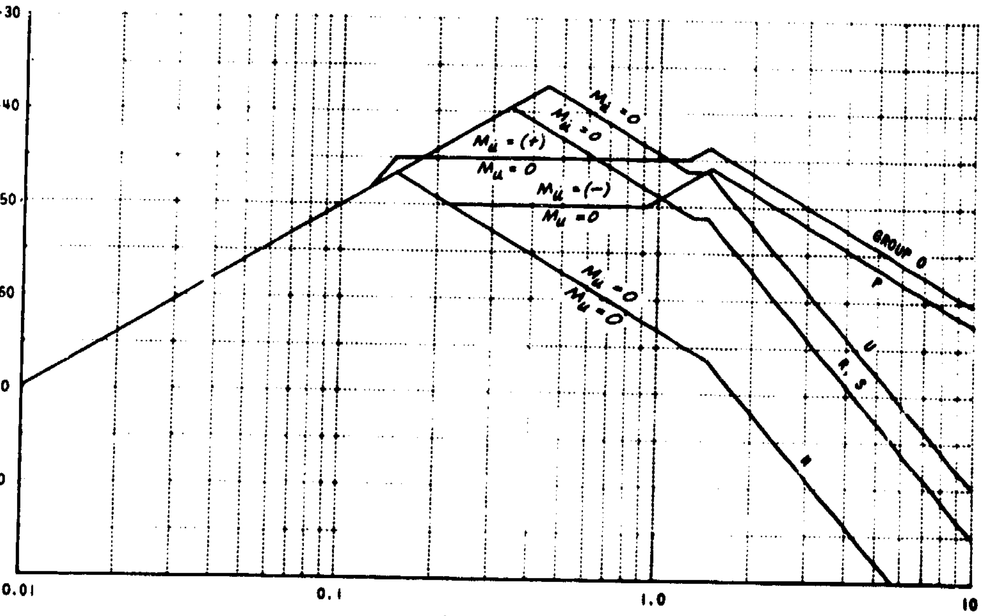


Figure 14 (3.2.1.2)
PITCH RESPONSE TO HORIZONTAL GUSTS FOR $\omega_{n_{sp}} = 1.46$
(T-33, REFERENCE E5)

3.2.1.3 FLIGHT PATH STABILITY

REQUIREMENT

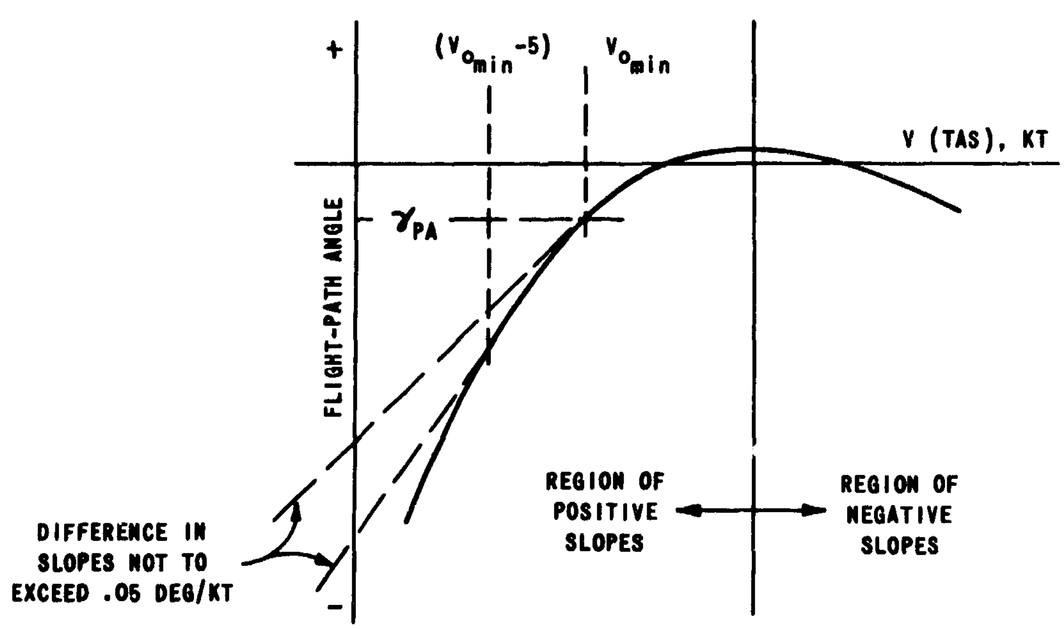
3.2.1.3 Flight-path stability. Flight-path stability is defined in terms of flight-path-angle change where the airspeed is changed by the use of the elevator control only (throttle setting not changed by the crew). For the landing approach Flight Phase, the flight-path-angle versus true-airspeed curve shall have a local slope at $V_{o_{min}}$ which is negative or less positive than:

a. Level 1 ------ 0.06 degrees/knot

b. Level 2 ------ 0.15 degrees/knot

c. Level 3 ------ 0.24 degrees/knot.

The thrust setting shall be that required for the normal approach glide path at $V_{o_{min}}$. The slope of the flight-path angle versus airspeed curve at 5 knots slower than $V_{o_{min}}$ shall not be more than 0.05 degrees per knot more positive than the slope at $V_{o_{min}}$, as illustrated by:



RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

Background

Operation on the "backside" of the drag curve in the landing approach leads to problems in airspeed and flight-path control. References E24, E30, E32, C39 show that airspeed behavior, when elevator is used to control attitude and altitude, is characterized by a first-order root which becomes unstable at speeds below minimum drag speed. This closed-loop instability, even when the open-loop (unattended-airplane) phugoid motion is stable, is caused by an "unstable" zero in the h/δ_e airplane transfer function. Specifically, Reference E24 uses closed-loop analyses to show the importance of the factor $1/T_{h_1}$ as an indicator of closed-loop system stability and throttle activity required. A useful measure of the quantity $1/T_{h_1}$ is needed.

Working from the altitude-to-elevator transfer function, Reference E12 shows that $(1/T_{h_1})$ is closely approximated by the ratio (D/C), where D and C are defined implicitly as follows:

$$\frac{\dot{h}(s)}{\delta_e(s)} = \frac{As^3 + Bs^2 + Cs + D}{(s^2 + 2\zeta_p\omega_{n_p}s + \omega_{n_p}^2)(s^2 + 2\zeta_{sp}\omega_{n_{sp}}s + \omega_{n_{sp}}^2)}$$

The additional assumption that C is approximately equal to $\left[V(Z_{\delta_e}M_w - M_{\delta_e}Z_w)\right]$ is generally valid, so that:

$$\frac{1}{T_{h_1}} = \frac{D}{V(Z_{\delta_e}M_w - M_{\delta_e}Z_w)}$$

The climb-angle-to-elevator transfer function is as follows:

$$\frac{\gamma(s)}{\delta_e(s)} = \frac{1}{V}\frac{\dot{h}(s)}{\delta_e(s)} = \frac{1}{V}\;\frac{As^3 + Bs^2 + Cs + D}{(s^2 + 2\zeta_p\omega_{n_p}s + \omega_{n_p}^2)(s^2 + 2\zeta_{sp}\omega_{n_{sp}}s + \omega_{n_{sp}}^2)}$$

Applying the limit value theorem, for a step $\delta_e\,(\delta_e/s)$ the slope of the steady-state γ versus δ_e curve is equal to the value of this transfer function when s approaches zero, so that:

$$\frac{d\gamma}{d\delta_e} = \left.\frac{\gamma(s)}{\delta_e(s)}\right|_{ss} = \frac{1}{V}\;\frac{D}{\omega_{n_p}^2\,\omega_{n_{sp}}^2}$$

In a similar manner, the slope of the steady-state u versus δ_e curve is obtained:

$$\frac{du}{d\delta_e} = \left.\frac{u(s)}{\delta_e(s)}\right|_{ss} = -\,\frac{g(Z_{\delta_e}M_w - M_{\delta_e}Z_w)}{\omega_{n_p}^2\,\omega_{n_{sp}}^2}$$

Therefore, the slope of the steady-state γ versus u curve for elevator inputs is

$$\frac{d\gamma}{du} = \frac{d\gamma/d\delta_e}{du/d\delta_e} = -\frac{1}{g}\,\frac{D}{V(Z_{\delta_e} M_w - M_{\delta_e} Z_w)}$$

Using the expression for $1/T_{h_1}$, we finally obtain

$$\frac{d\gamma}{du} = -\frac{1}{g}\,\frac{1}{T_{h_1}}$$

The $(d\gamma/du)$ limits of 3.2.1.3, therefore, set limits on $(1/T_{h_1})$.

The limit on $(d\gamma/du)$ at 5 knots slower than $V_{o_{min}}$ was added to assure that the airplane remains tractable at commonly encountered off-nominal speeds.

For design purposes, $(d\gamma/du)$ can be estimated from the dimensional stability derivatives as follows:

$$\frac{d\gamma}{du} = \frac{1}{g}\left[X_u - \left(X_w - \frac{g}{V}\right)\left(\frac{Z_u - \frac{Z_{\delta_e}}{M_{\delta_e}} M_u}{Z_w - \frac{Z_{\delta_e}}{M_{\delta_e}} M_w}\right) - \frac{X_{\delta_e}}{M_{\delta_e}}\left(\frac{M_w Z_u - M_u Z_w}{-Z_w + M_w \frac{Z_{\delta_e}}{M_{\delta_e}}}\right)\right]$$

or

$$\frac{d\gamma}{du} = \frac{1}{g}\left[X_u - \left(X_w - \frac{g}{V}\right)\left(\frac{Z_u - \frac{Z_{\delta_e}}{M_{\delta_e}} M_u}{\frac{1}{T_{\theta_2}}}\right) - \frac{X_{\delta_e}}{M_{\delta_e}}\left(\frac{\omega_{n_p}^2\,\omega_{n_{sp}}^2}{g\left(\frac{1}{T_{\theta_2}}\right)}\right)\right]$$

For M_u and X_{δ_e} small, the following approximation is valid except for very-short-tailed airplanes:

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

Flight test techniques are discussed in Appendix IVB.

<u>Data for Landing Approach</u>

The $(1/T_{h_1})$ data used to set numerical limits on $(d\gamma/du)$ for the approach flight phase are given in References E5, E16, E20, E22, and C14. The experiment of Reference E5 is discussed under paragraph 3.2.1.2. Only Figures 1, 2, 7 and 9 of that discussion contained data from which the effects of $1/T_{h_1}$ could be separated from the effects of phugoid damping. These four figures are reproduced as Figures 1 through 4 of this discussion.

It is apparent from Figures 1-3 that pilot ratings of $1/T_{h_1}$ are dependent on the value of ζ_p. For Level 1, 3.2.1.2 requires $\zeta_p \geq 0.04$; greater damping might result from autothrottle or similar augmentation. Therefore the positive ζ_p data of Figure 1 were used to establish the Level 1 requirement for $1/T_{h_1}$ or $d\gamma/dV$ (The data from Figures 2-4 are obviously too conservative for Level 1. The configurations for Figure 2 had $\omega_{n_{sp}}$ marginally close to the lower Level 1 boundary; while those for Figure 4 were downrated because of the pitch response to horizontal gusts caused by M_u, as discussed under 3.2.1.2). For Levels 2 and 3, the zero-ζ_p data seem appropriate:

Figure	Level 2	Level 3
1	$1/T_{h_1} \geq -.08$	$1/T_{h_1} \geq -.12+$
2	$1/T_{h_1} \geq -.05$	$1/T_{h_1} \geq -.08$

From Figure 3, with near-zero ζ_p:

Level 2	Level 3
$1/T_{h_1} \geq -.05$	$1/T_{h_1} \geq -.12$

From Figure 4, with high ζ_p but in turbulence:

Level 2	Level 3
$1/T_{h_1} \geq -.05$	$1/T_{h_1} \geq -.12$

Combinations of Level -2 or -3 values of $1/T_{h_1}$ with low ζ_p, ω_{sp} or both appear worse than cases with high ζ_p and ω_{sp}. With these considerations in mind, $1/T_{h_1} = -.02$ was chosen for the Level 1 boundary, -.05 for Level 2, and -.08 for Level 3. These values of $1/T_{h_1}$ correspond to the $d\gamma/dV$ values specified in 3.2.1.3: multiply $1/T_{h_1}$ by -(57.3)(1.689)/(32.2) = -3. The chosen criteria will be re-examined as more data become available.

The ground simulator experiment of Reference E20 altered $1/T_{h_1}$ by changing X_w and X_{δ_e}, and also considered the influences of thrust-line inclination and thrust-line offset on the flying qualities. There are very limited data for thrust-line offset, and the decision was made to assume that designers will take reasonable steps to keep the offset as small as possible. The data for zero thrust-line offset are presented in Figure 5 for different values of thrust-line inclination. The data do seem to indicate that some thrust-line inclination is desirable, but the variations in rating due to inclination are well within the scatter of the data considered as a whole.

The data from ground simulator experiments of References C14 and E16 are presented in Figure 6. It should be mentioned that only the data for the highest static margin in Reference E16 are presented because the lower static margins result in values of $\omega_{n_{sp}}$ which are too low for Level 1.

The data from the in-flight experiment of Reference E22 are presented in Figure 7. There are several factors which influence interpretation of this data. First, the pilot rating scale used is a modified version of the Cooper scale and is rather difficult to interpret. Second, the "speed stability" was changed by altering $\partial T/\partial V$ as well as $\partial T/\partial \alpha$, which means that unstable values of "speed stability" were accompanied by negative values of phugoid damping. Since the "speed stability" was altered in this experiment by using engine thrust, the pilot could use the engine noise as an airspeed cue. The final (and probably most significant) factor is that most of the approaches were flown VFR, with a ground controller supplying continuous flight-path information by radio using a theodolite. Reference E22 states that this type of technique resulted in very tight control of flight path. A few approaches were made using precision-approach radar, which were much more difficult for the pilot to successfully accomplish. The relationship between the "speed stability" parameter $1/T_2$ of Figure 7 and $1/T_{h_1}$ is as follows:

$$\frac{1}{T_{h_1}} = 0.693 \frac{1}{T_2}$$

A comparison of the requirements derived from Figures 1 through 4 and the data from Figures 5 through 7 are presented below.

	$1/T_{h_1}$ for Level 1	$1/T_{h_1}$ for Level 2	$1/T_{h_1}$ for Level 3
Requirement of 3.2.1.3	-.02	-.05	-.08
Figure 5 (Reference E20)	-.035	-.084	-.107
Figure 6 (Reference C14)	-.020 to -.035	-.095	-.121
Figure 6 (Reference E16)	-.010	-	-
Figure 7 (Ref. E22) - no thrust lag	+.010	-.190	-.360
Figure 7 (Ref. E22) - thrust lag	+.017	-.060	-.125

The primary problem with Figure 7 seems to be that the majority of the data is for VFR approaches with unusually good flight-path information available to the pilot (see References E22 and E35).

Some qualitative data are available in Reference C13, although the basic "longitudinal stability" (probably short-period dynamics) was poor.

	$\frac{d(T/W)}{dV}$ ~ knot^{-1}	$1/T_{h_1}$ ~ sec^{-1}
"Minimum comfortable" range	-.001 to -.002	-.019 to -.038
"Unsatisfactory" range	-.0025 to -.004	-.048 to -.076

The in-flight SST evaluation of Reference C11 shows a rating degradation from 3.5 to 4.0 when $1/T_{h_1}$ is changed from +.030 to -.034.

Other Flight Phases

"Backside" operation is also troublesome for takeoff, cruise, and high-altitude maneuvering, but it will probably not be as critical as for the landing approach, and there are virtually no data to define numerical limits for these flight phases.

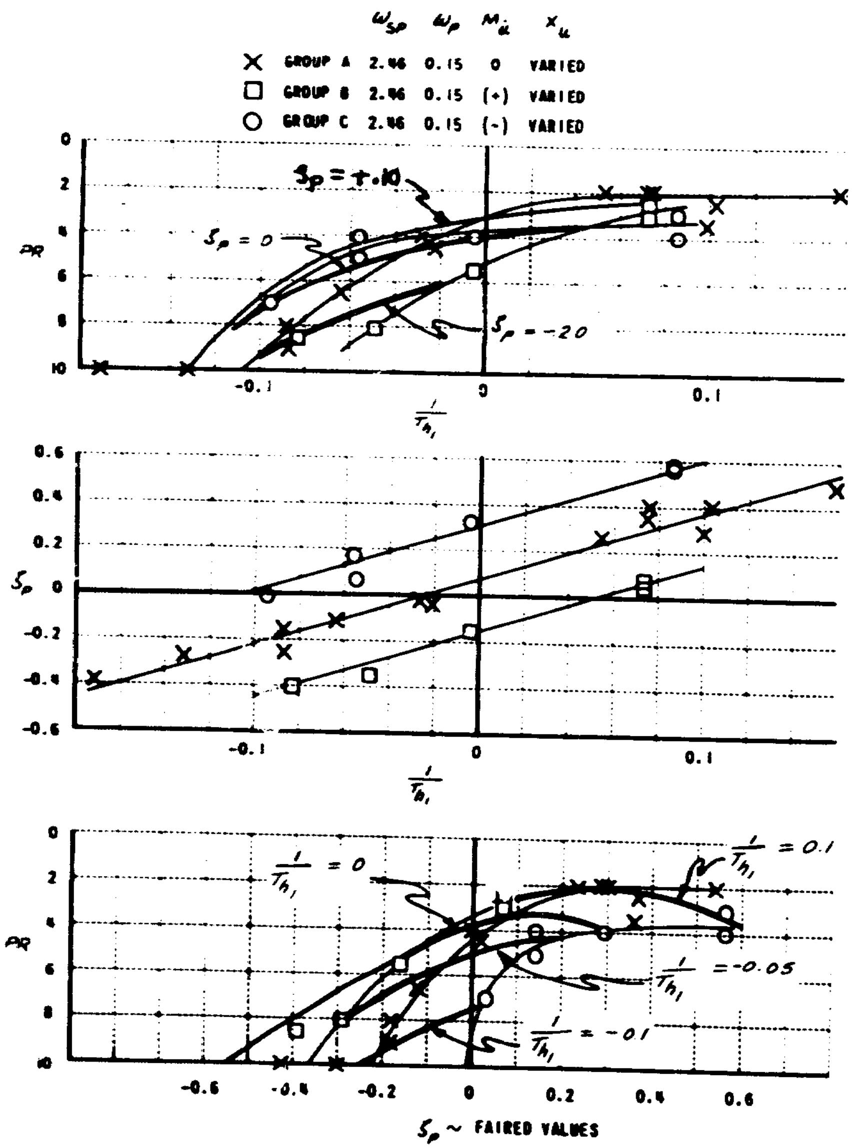


Figure 1 (3.2.1.3)
LANDING APPROACH (T-33, REFERENCE E5)

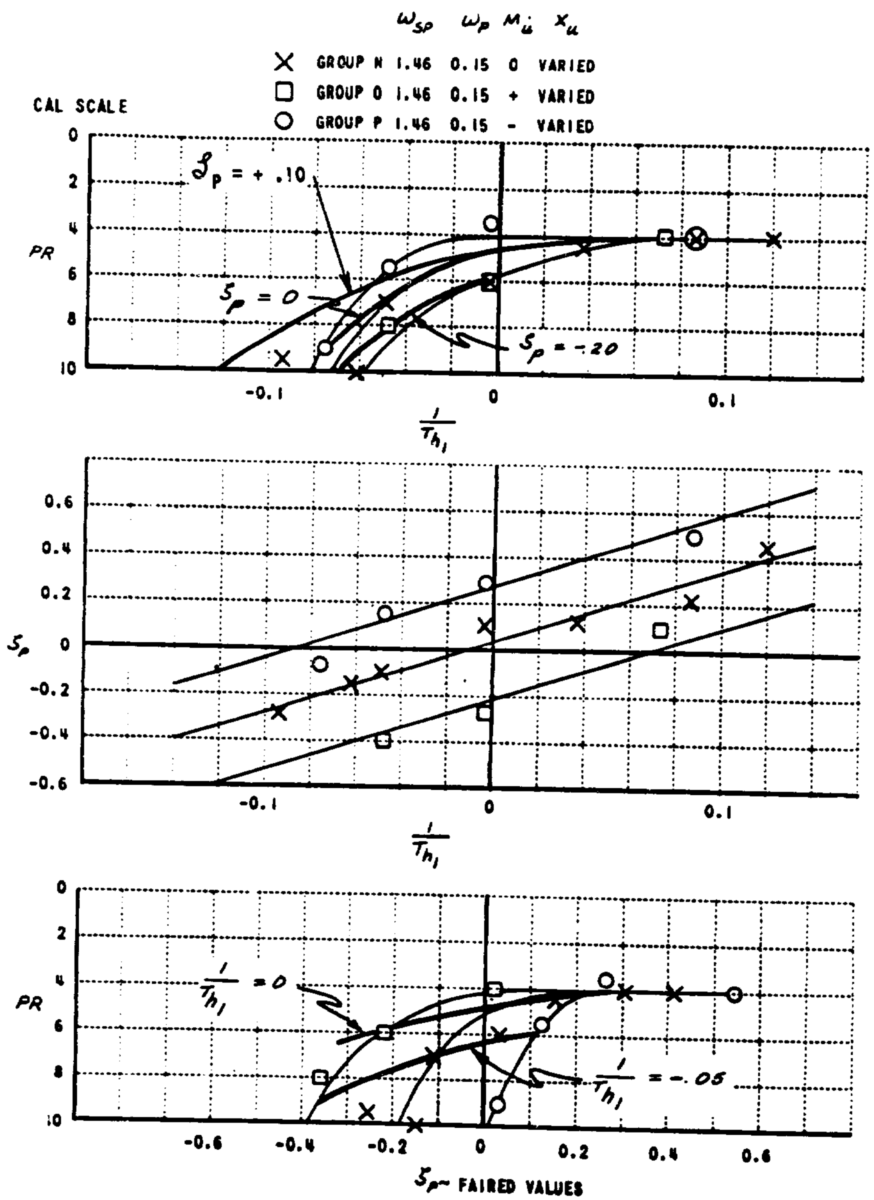


Figure 2 (3.2.1.3)
LANDING APPROACH (T-33, REFERENCE E5)

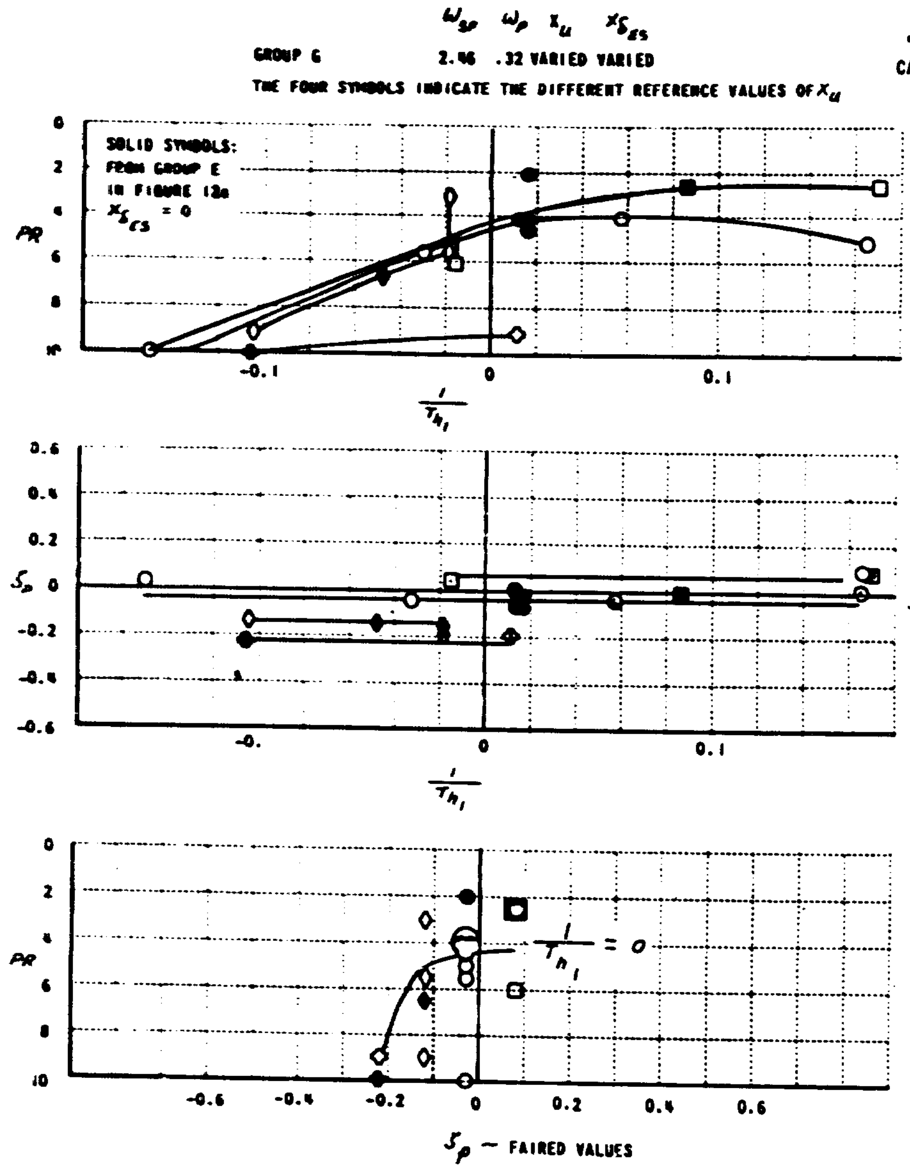


Figure 3 (3.2.1.3)
LANDING APPROACH (T-33, REFERENCE E5)

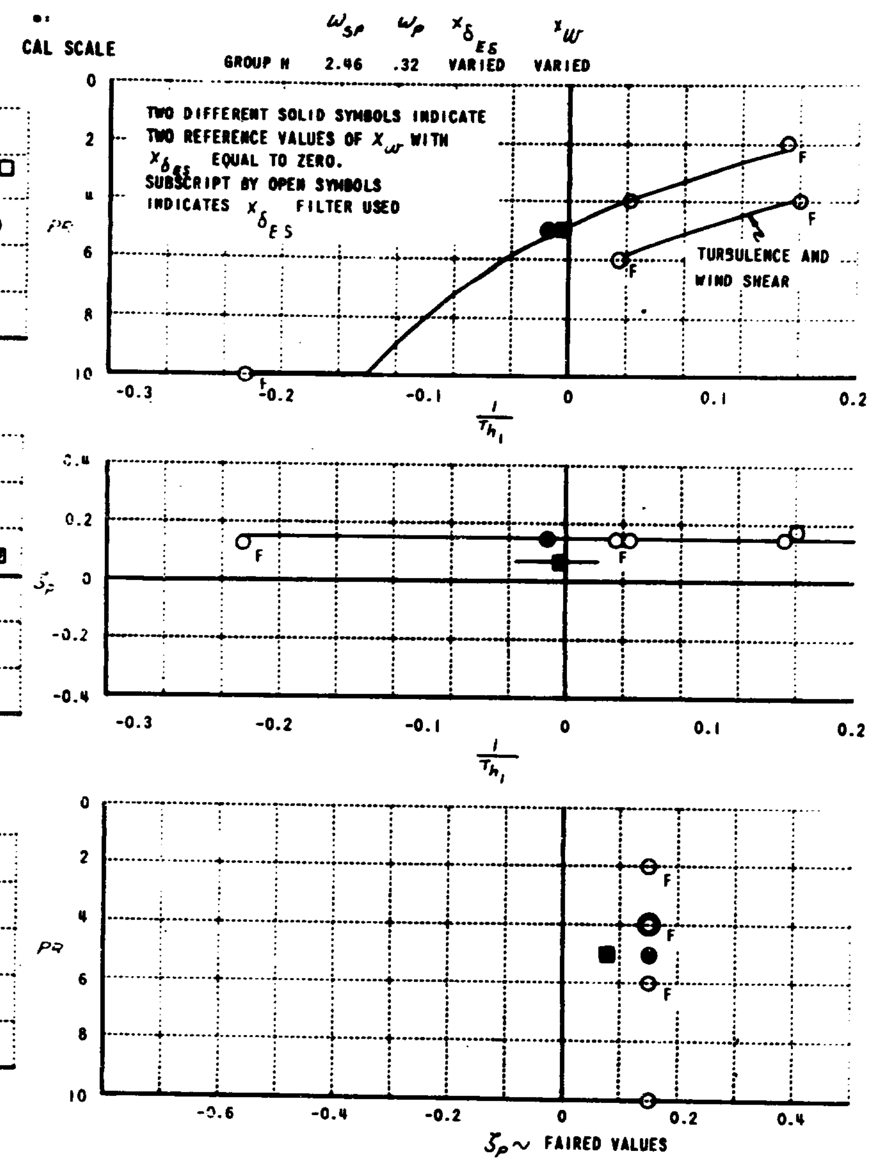


Figure 4 (3.2.1.3)
LANDING APPROACH (T-33, REFERENCE E5)

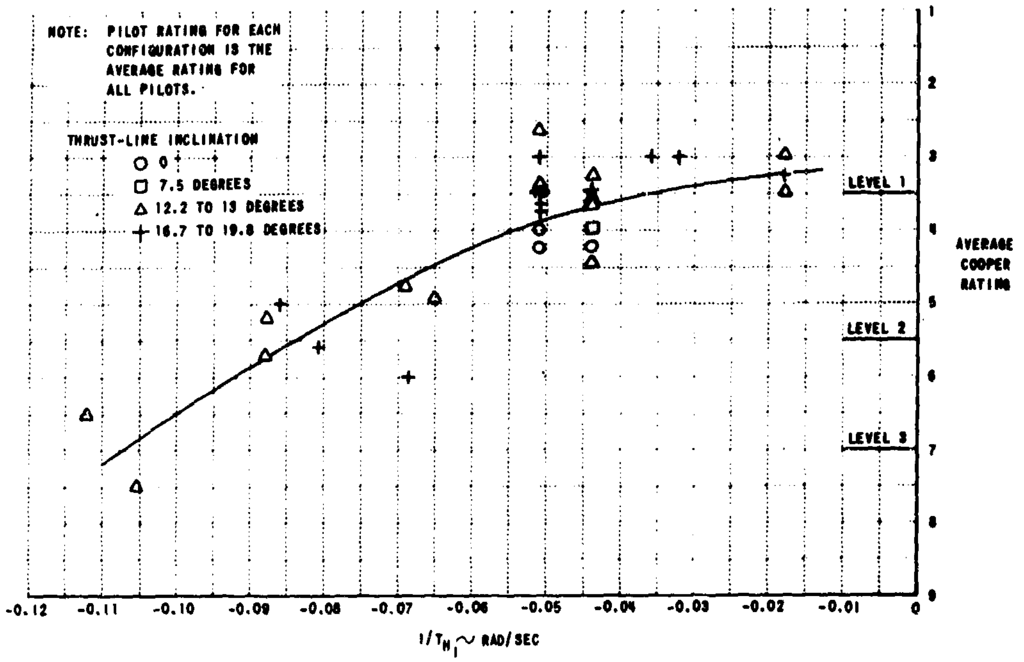


Figure 5 (3.2.1.3)
CARRIER APPROACH (GROUND SIMULATOR EXPERIMENT, REFERENCE E20)

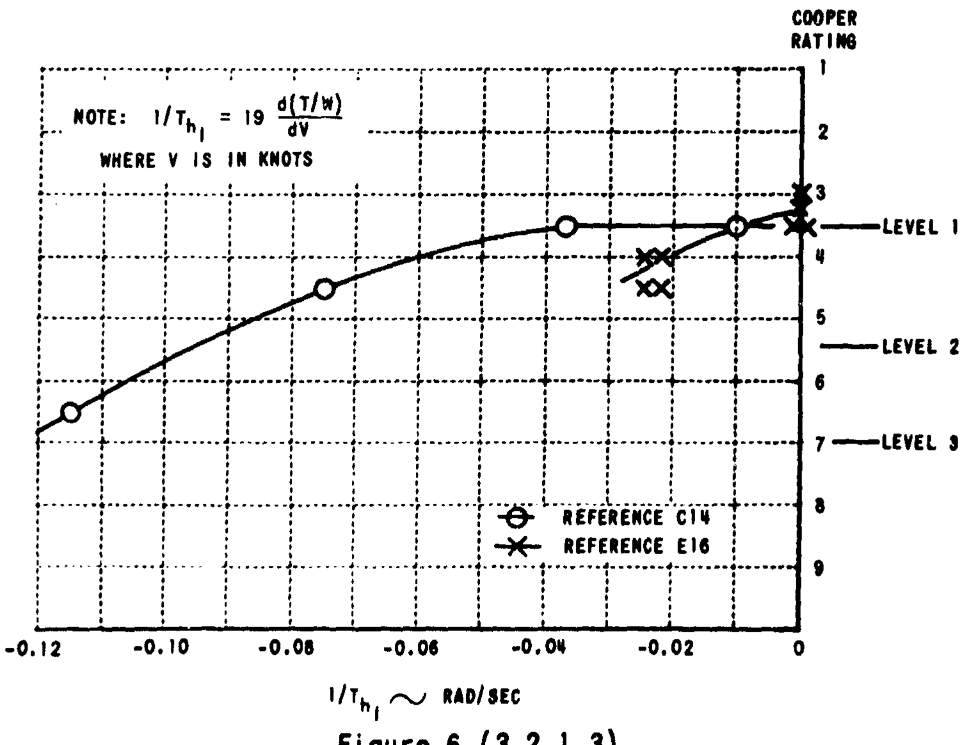


Figure 6 (3.2.1.3)
SST LANDING APPROACH (GROUND SIMULATOR EXPERIMENTS, REFERENCES C14 AND E16)

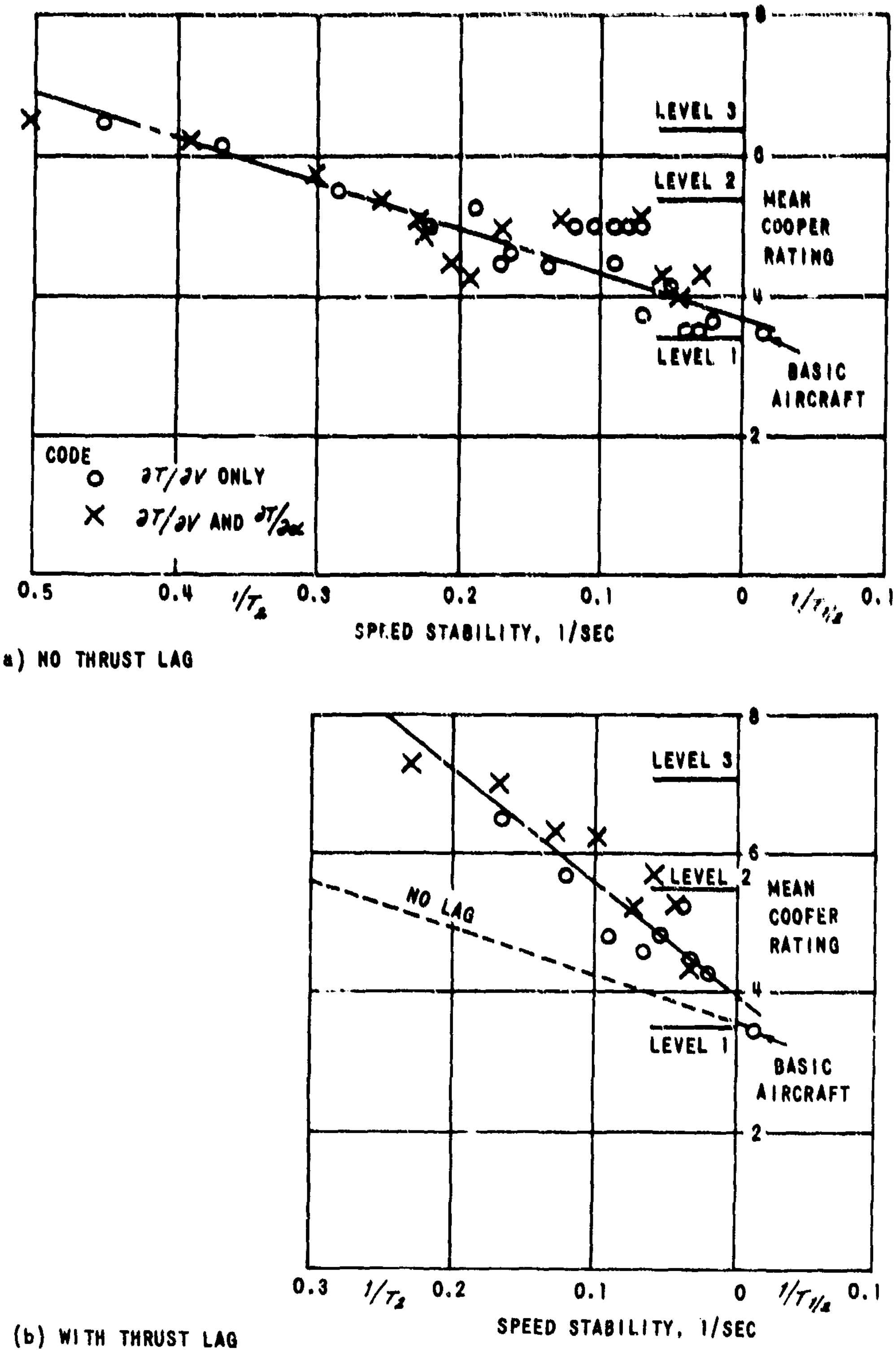


Figure 7 (3.2.1.3)
LANDING APPROACH (AVRO 707, REF. E22)

3.2.2 LONGITUDINAL MANEUVERING CHARACTERISTICS

DISCUSSION

The requirements of the subparagraphs under 3.2.2 deal with maneuvering characteristics, i.e., the airplane's dynamic response to pilot inputs at essentially constant speed. The major topics discussed are short-period response, control feel during maneuvers, and pilot-induced oscillations (PIO).

Short-period damping ratio is a measure of the amount of overshooting present in the short-period motion of the airplane, and low values of ζ_{sp} can contribute to pilot-induced oscillations.

Short-period natural frequency and stick force per g are the primary parameters which affect the acceptability of the airplane's maneuvering characteristics (for a given value of ζ_{sp}). Also important in this regard is the stick motion gradient during maneuvers.

The relationship between ζ_{sp} and F_s/n as a factor in pilot-induced oscillations is expressed in the requirement on transient stick forces in maneuvers.

3.2.2.1 SHORT-PERIOD RESPONSE

REQUIREMENT

3.2.2.1 Short-period response. The short-period response of angle of attack which occurs at approximately constant speed, and which may be produced by abrupt elevator control inputs, shall meet the requirements of 3.2.2.1.1 and 3.2.2.1.2. These requirements apply, with the cockpit control free and with it fixed, for responses of any magnitude that might be experienced in service use. If oscillations are nonlinear with amplitude, the requirements shall apply to each cycle of the oscillation. In addition to meeting the numerical requirements of 3.2.2.1.1 and 3.2.2.1.2, the contractor shall show that the airplane has acceptable response characteristics in atmospheric disturbances.

RELATED MIL-F-8785 PARAGRAPHS

3.3.5

DISCUSSION

The proposed short-period requirements are a complete rewrite of 3.3.5 of MIL-F-8785.

In specifying longitudinal dynamics, it is desirable to use criteria which are precise descriptions of the airplane responses directly important to the pilot. This would allow the designer the greatest freedom in the use of various combinations of airframe and control system dynamics to achieve the desired overall responses. Unfortunately, specification of short-period dynamics in this form is not possible at the present time, due to lack of systematic flying qualities data obtained for various control-system and airframe dynamics in combination with various types of feel system dynamics. Also, the response of the airplane to pilot inputs is only a partial description of longitudinal dynamics, since short-period response to turbulence and feel system dynamics are also important by themselves. For these reasons, it was decided to use conventional short-period modal parameters as criteria for the present revision of MIL-F-8785, treating control-system dynamics separately.

Stability augmentation systems (SAS) are obviously here to stay, and many people argue that a second-order description of short-period dynamics is not possible for some airplanes employing complex SAS. These airplanes may have SAS natural frequencies very close to the short-period natural frequency, which make the overall airplane response to pilot inputs higher than second-order. However, the results of in-flight evaluations of various types of higher-order control systems indicate that such systems can introduce lags large enough to cause very serious pilot-induced oscillations (see Reference J59). On the basis of this experiment, the requirements of 3.5.3 were formulated. In order to meet these requirements, the control system natural frequencies will normally have to be appreciably higher than the short-period natural frequency. In this situation, the airplane's response to pilot inputs can be approximated quite well by a second-order response plus a small time delay. For practical control systems, then, it will usually be possible to identify the short-period mode.

Airplane responses are sometimes fed back through the feel system in such a way that the short-period dynamics are altered. In this situation, the stick-fixed and stick-free short-period dynamics can be different. Since pilot control inputs are a combination of force and position commands, it was considered necessary to specify the same limits on short-period dynamics for both the stick-fixed and stick-free cases. This reasoning is especially applicable where ζ_{sp} is concerned. The sudden pull-up criterion of MIL-F-8785, discussed under 3.2.2.3, was the only requirement in that document related to ζ_{sp} (stick-free), and was much too stringent. The new stick-free short-period damping requirements provide part of a reasonable replacement for this criterion.

The amplitudes at which the short-period requirements apply are indicated only qualitatively here and in 3.5.4.2, Saturation of Augmentation Systems. While more definitive guidance would be desirable, it is not possible at this time to be more explicit in a general specification. The intent is to avoid part-time stability augmentation and other objectionable non-linearities. With improving technology, the penalties associated with augmentation redundancy are becoming smaller, so that large authority can be provided with safety.

The discussions that follow are all in terms of airplane response at constant speed. The "steady-state" characteristics meant are therefore related to the two-degree-of-freedom approximate equations of motion,

$$(s - Z_w)\, w \qquad - Vq = Z_\delta\, \delta$$

$$-(M_{\dot w}\, s + M_w)\, w + (s - M_q)\, q = M_\delta\, \delta$$

plus the effects of the flight control system. These are the equations given in the discussion of 3.2.1, with the speed degree of freedom suppressed; so, strictly speaking, they apply only in or near level flight.

3.2.2.1.1 SHORT-PERIOD FREQUENCY AND ACCELERATION SENSITIVITY

REQUIREMENT

3.2.2.1.1 Short-period frequency and acceleration sensitivity. The short-period undamped natural frequency, $\omega_{n_{sp}}$, shall be within the limits shown in figures 1, 2 and 3. If suitable means of directly controlling normal force are provided, the lower bounds on $\omega_{n_{sp}}$ and n/α of Figure 3 may be relaxed if approved by the procuring activity.

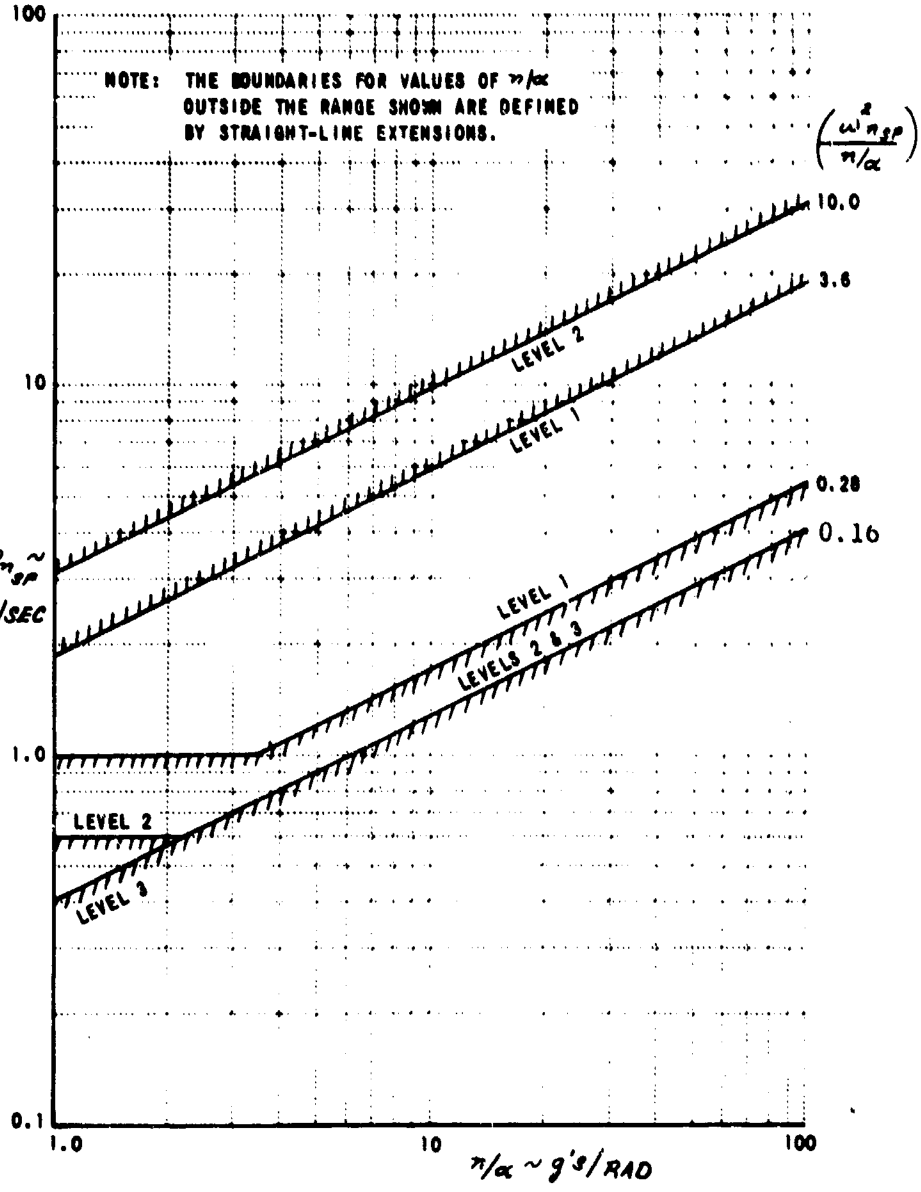


FIGURE 1

SHORT-PERIOD FREQUENCY REQUIREMENTS - CATEGORY A FLIGHT PHASES

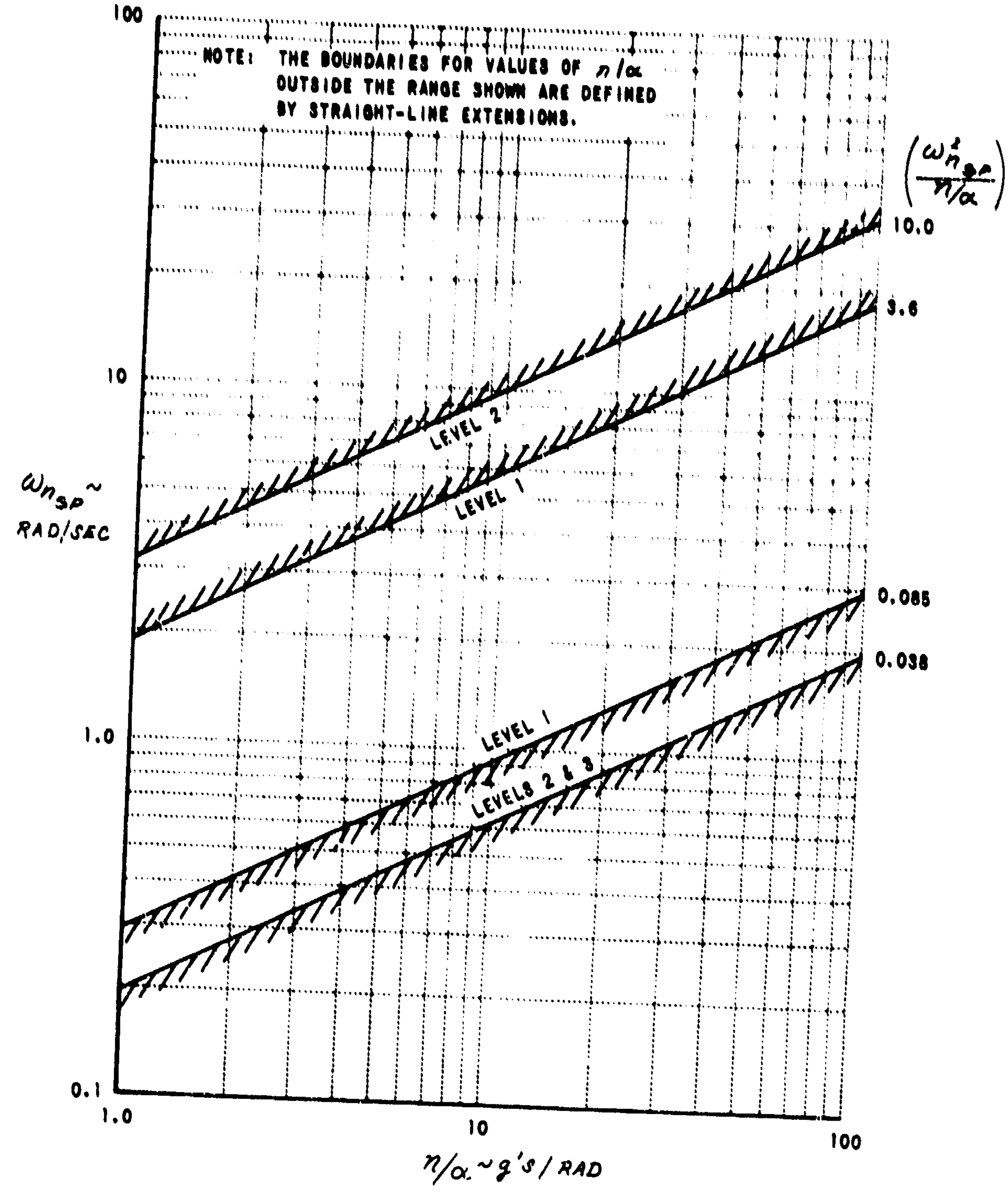


FIGURE 2

SHORT-PERIOD FREQUENCY REQUIREMENTS-CATEGORY B FLIGHT PHASES

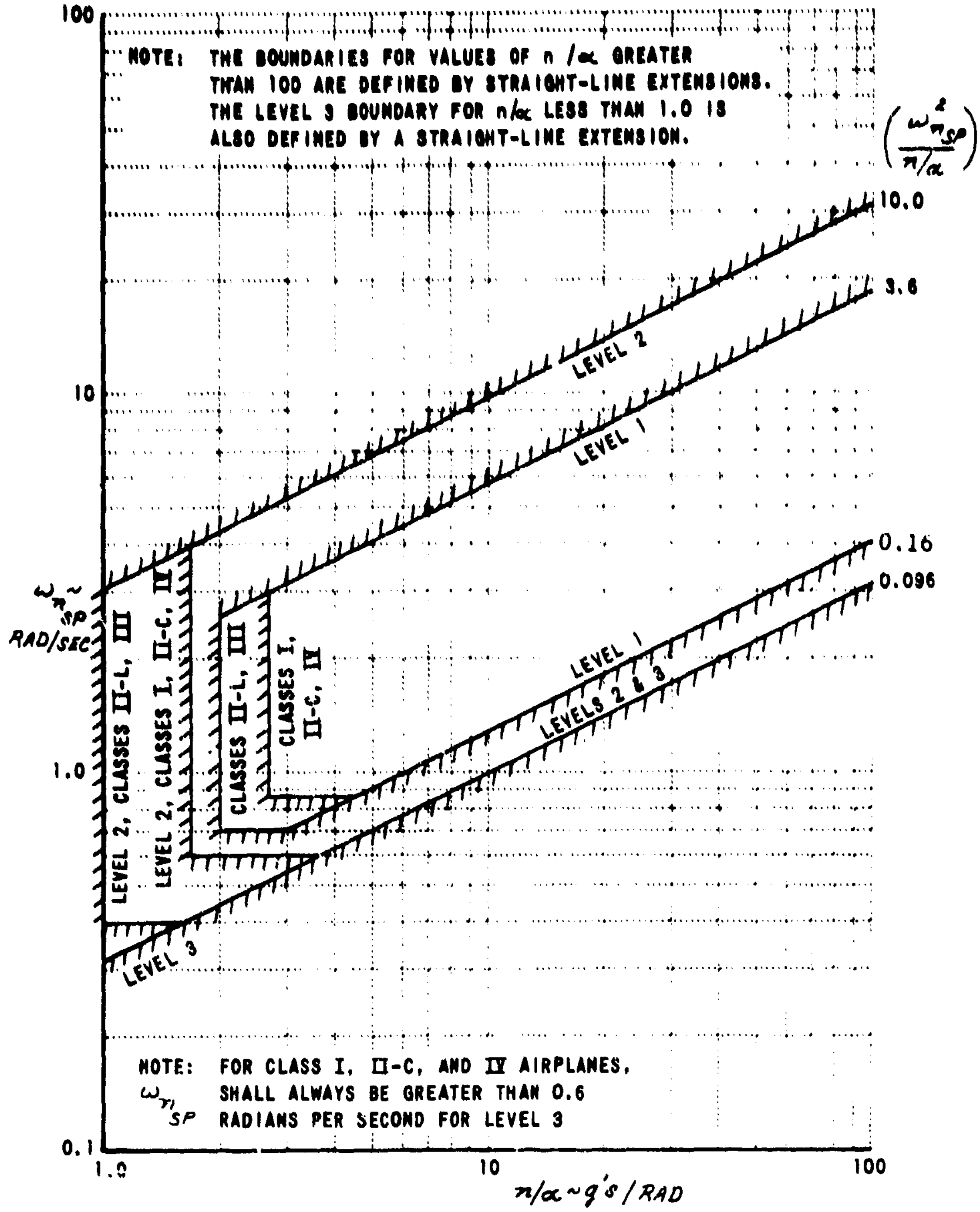


FIGURE 3

SHORT - PERIOD FREQUENCY REQUIREMENTS- CATEGORY C FLIGHT PHASES

RELATED MIL-F-8785 PARAGRAPHS

3.3.1, 3.3.1.1, 3.3.2, 3.3.4

DISCUSSION

Related Factors

It is generally agreed that short-period frequency and damping alone are not sufficient to describe the acceptability of airplane longitudinal dynamics. In Reference D19, the effects on flying qualities of the parameters L_α, V and n/α were explored. Reference D21 expresses the idea that $1/T_{\theta_2}$ is of primary importance because it appears in the numerator of the pitch-attitude-to-elevator transfer function. References E13, D10 and D11 are based on the premise that pitch rate response is of primary importance at low speed and normal acceleration response is of primary importance at high speed. From this premise, References E13 and D10 conclude that the short-period frequency should be a function of $1/T_{\theta_2}$ for low values of n/α and that it should be a function of n/α when n/α is large. Reference D11 recommends envelopes on the weighted sum of the pitch rate and normal acceleration responses to a step stick-force command. In References D7 and D3 the relationship between initial pitch acceleration and steady-state normal acceleration is discussed and related to the short-period frequency and n/α. The theories discussed in References D7 and D3 best explain pilot objections to excessively high and low short-period frequencies, and have therefore been used as a framework to mold the available pilot rating data into short-period frequency requirements.

A detailed analysis of the pilot comments from the flight programs of References D3 and E12 yields a very good interpretation of the pilot's objections when $\omega_{n_{sp}}$ is either too high or too low. In these flight programs, the pilots were required to vary the control gearing and select an optimum or best compromise value for use in the evaluation of each short-period configuration. By increasing the control gain, it was always possible, regardless of the flight condition or short-period configuration, to make the sensitivity too high with the result that the response was abrupt and gross for small control inputs. Conversely, it was always possible to make the gain so low that large stick motions and forces were necessary in turns and pullups. The pilot would vary the control gain between these extreme situations and search for the optimum value or, as was often the case, the least objectionable compromise value. From these experiments, it was observed that for each flight condition there was a range of short-period frequencies for which the pilots could select rather well-defined optimum control gains, but at lower and at higher short-period frequencies they would encounter conflicting requirements which imposed unsatisfactory compromises in the selection of the control gain. To better understand this problem, it is useful to derive an algebraic relationship between sensitivity and steady forces. In the steady state, the constant-speed equations from the discussion of 3.2.2.1 yield

$$\frac{\Delta n}{\delta_e} = \frac{V}{g}\,\frac{q}{\delta_e} = \frac{V}{g}\left(\frac{Z_{\delta_e}M_w - M_{\delta_e}Z_w}{Z_w M_q - VM_w}\right) = \frac{V}{g}\;\frac{(Z_{\delta_e}M_w - M_{\delta_e}Z_w)}{\omega_{n_{SP}}^2}$$

so that

$$\frac{F_S}{n} = \frac{1}{\left(\frac{\Delta n}{\delta_e}\right)\left(\frac{\delta_e}{F_S}\right)} = \frac{Z_w M_q - VM_w}{\frac{\delta_e}{F_S}\cdot\frac{V}{g}\left(Z_\delta M_w - M_\delta Z_w\right)} = \frac{\omega_{n_{SP}}^2}{\frac{\delta_e}{F_S}\cdot\frac{V}{g}\left(Z_{\delta_e}M_w - M_{\delta_e}Z_w\right)}$$

With

$$\frac{\Delta\alpha}{\delta_e} = \frac{1}{V}\;\frac{w}{\delta_e} = \frac{M_{\delta_e}V - Z_{\delta_e}M_q}{V(Z_w M_q - VM_w)} = \frac{M_{\delta_e} - \frac{Z_{\delta_e}M_q}{V}}{\omega_{n_{SP}}^2} \approx \frac{M_{\delta_e}}{\omega_{n_{SP}}^2}$$

(Normally $|M_{\delta_e}| \gg |Z_{\delta_e}M_q/V|$ even for short-tailed airplanes.)

Therefore, for elevator inputs,

$$\frac{n}{\alpha} = \frac{\Delta n}{\Delta\alpha} = \frac{\left(\frac{\Delta n}{\delta_e}\right)}{\left(\frac{\Delta\alpha}{\delta_e}\right)} = \frac{V}{g}\left(\frac{Z_{\delta_e}M_w - M_{\delta_e}Z_w}{M_{\delta_e} - \frac{Z_{\delta_e}M_q}{V}}\right) \approx \frac{V}{gM_{\delta_e}}\left(Z_{\delta_e}M_w - M_{\delta_e}Z_w\right)$$

(When $|M_{\delta_e}Z_w| \gg |Z_{\delta_e}M_w|$, this expression reduces to $n/\alpha = C_{L_\alpha}/C_L$ for 1g flight.) Substituting the expression for n/α into the expression for F_S/n, the following very good approximation results:

$$\frac{F_S}{n} = \frac{\omega_{n_{SP}}^2}{\frac{\delta_e}{F_S}M_{\delta_e}(n/\alpha)} = \frac{\omega_{n_{SP}}^2}{M_{F_S}(n/\alpha)}$$

where M_{F_S} is the initial pitch acceleration per pound of stick force or sensitivity, assuming negligible control-system dynamics and transport lag. Finally, the relationship between F_S/n and sensitivity can be expressed as

$$\frac{F_S}{n}M_{F_S} = \frac{\omega_{n_{SP}}^2}{n/\alpha}$$

Thus it can be seen that when $\left(\omega_{n_{SP}}^2/\frac{n}{\alpha}\right)$ is small, either the stick force per g must be reduced to maintain high enough sensitivity or the sensitivity must be reduced to maintain satisfactorily high stick force per g in maneuvers. If $\left(\omega_{n_{SP}}^2/\frac{n}{\alpha}\right)$ is too small, the pilot will not be able to achieve a satisfactory compromise between sensitivity and steady forces and, depending on the individual, may select one or the other extreme. For example, one pilot in Reference D3 selected a low control gain in this situation so that the control forces were high, thus tending to guard against overcontrol, while the other pilot in Reference D3 selected the control gain on the basis of sensitivity and accepted the light steady forces that resulted. Both pilots, however, considered these configurations unacceptable and gave them pilot ratings of 7-10.

From the same equation, when $\left(\omega_{n_{sp}}^2 / \frac{n}{\alpha}\right)$ is large, either the stick force per g must be increased to maintain satisfactorily low sensitivity or the sensitivity must be increased to maintain comfortable stick forces in maneuvers. If $\left(\omega_{n_{sp}}^2 / \frac{n}{\alpha}\right)$ is too large, the pilot again will not be able to achieve an acceptable compromise and will downrate the configuration. When presented this situation, both pilots in Reference D3 and the pilot in Reference E12 selected the control gain to keep the sensitivity from being too high, thus reducing the abruptness and tendency to bobble for small inputs. This, of course, caused heavy steady forces during sustained maneuvers and turns and resulted in poorer pilot ratings. Because of this, paragraph 3.2.2.1 contains a recommendation that $\omega_{n_{sp}}$ near the upper limits be accompanied by high F_s/n.

A somewhat different theory, which also indicates the importance of the parameter $\left(\omega_{n_{sp}}^2 / \frac{n}{\alpha}\right)$, is presented in Reference D7. In this analysis, it is assumed that the initial pitch acceleration is the response that is of concern to the pilot when he initiates a maneuver, and that the steady-state response of concern is the normal acceleration experienced in a pull-up. By assuming constant-speed equations of motion and by applying the initial value theorem to the $\ddot{\theta}/\delta_e$ transfer function and the final value theorem to the n/δ_e transfer function, the following expression can be written for the ratio of initial pitch acceleration to steady-state normal acceleration following a step input.

$$\frac{\left.\frac{\ddot{\theta}}{\delta_e}\right|_{t=0^+}}{\left.\frac{n}{\delta_e}\right|_{ss}} = \frac{\omega_{n_{sp}}^2}{\frac{V}{g}\frac{1}{T_{\theta_2}}}$$

where $(1/T_{\theta_2})$ is the numerator lead factor in the constant-speed θ/δ_e transfer function. This lead factor can be expressed as follows:

$$\frac{1}{T_{\theta_2}} = \frac{Z_{\delta_e} M_w - M_{\delta_e} Z_w}{M_{\delta_e} + Z_{\delta_e} M_{\dot{w}}}$$

It is usually valid to assume $|M_{\delta_e}| >> |Z_{\delta_e} M_{\dot{w}}|$ (comparable to the previous assumption that $|M_{\delta_e}| >> |Z_{\delta_e} M_q|$), and therefore $(1/T_{\theta_2})$ reduces to

$$\frac{1}{T_{\theta_2}} \approx \frac{Z_{\delta_e} M_w - M_{\delta_e} Z_w}{M_{\delta_e}} \approx \frac{g}{V}\left(\frac{n}{\alpha}\right)$$

Finally, then, we have:

$$\frac{\left.\frac{\ddot{\theta}}{\delta_e}\right|_{t=0^+}}{\left.\frac{n}{\delta_e}\right|_{ss}} \approx \frac{\omega_{n_{sp}}^2}{\frac{n}{\alpha}}$$

On the basis of the above considerations and the trends indicated by data from variable stability airplane experiments, the upper and lower limits on short-period frequency have been defined as functions of n/α (the ratio of the change in n to the change in α, in the steady-state response to a control input), so as to bound the parameter $(\omega_{n_{sp}}^2/\frac{n}{\alpha})$. Specification of short-period frequency requirements in this form has implications for the design of augmentation systems in that it should be more permissive for fixed-gain systems than single $\omega_{n_{sp}}$ vs. ζ_{sp} bull's-eye requirements, and it takes the emphasis off invariant models for self-adaptive systems.

Flight-test and data-reduction techniques for getting $\omega_{n_{sp}}$ and n/α are discussed in Appendices III and IVC, D and E.

Intuition leads most people to suspect that limits on $(\omega_{n_{sp}}^2/\frac{n}{\alpha})$ alone are not sufficient, and that there are likely to be absolute lower limits on both $\omega_{n_{sp}}$ and n/α (or $1/T_{\theta_2}$). Using closed-loop analyses of the pilot-vehicle combinatio⁻, Reference B1 concludes there are lower limits on $\omega_{n_{sp}}$ for tasks requiring precise control of pitch attitude, and lower limits on $\omega_{n_{sp}}$ and $1/T_{\theta_2}$ for tasks requiring precise control of both pitch attitude and flight path. In these studies, major assumptions were made concerning the form of the pilot model, pilot loop closures, and desired closed-loop characteristics. In addition, the analyses were performed on very specific tasks, with the assumption of continuous closed-loop control. In spite of their oversimplified nature, the analyses do provide some insight into problems that might occur if $\omega_{n_{sp}}$ or $1/T_{\theta_2}$ were to become very low.

For Category A and C Flight Phases, the pilot must introduce phase lead into his control inputs in order to maintain precise control of pitch attitude as $\omega_{n_{sp}}$ decreases. It therefore seems likely that the required phase lead will become excessive as $\omega_{n_{sp}}$ is reduced below some lower limit, regardless of the magnitude of n/α.

For normal values of n/α and $1/T_{\theta_2}$, flight path control is achieved by controlling pitch attitude, since flight path angle will quickly follow attitude changes. As $1/T_{\theta_2}$ decreases, however, the response of flight path angle to pitch attitude changes becomes slower. Decreases in n/α, which normally accompany decreases in $1/T_{\theta_2}$, increase the angle-of-attack change required to control the flight path. The pilot comments from Reference D19 indicate that when $1/T_{\theta_2}$ and n/α become small, the pilot must overdrive the attitude of the airplane in order to maintain rapid control of flight path. In other words, he must make an initial attitude change which is quite large, and then ease off on the attitude as the flight path angle approaches the desired value. For Category C Flight Phases, therefore, the attitude changes required for precise control of flight path will probably become unacceptably large when n/α or $1/T_{\theta_2}$ fall below some lower limit.

On the basis of the above arguments, absolute limits on $\omega_{n_{sp}}$ and n/α were established, in addition to the $(\omega_{n_{sp}}^2 / \frac{n}{\alpha})$ limits.

Direct Normal-Force Control

The requirements of 3.2.2.1.1 state that the limits on $\omega_{n_{sp}}$ and n/α may be relaxed, with the approval of the procuring activity, if a suitable type of direct-lift control (DLC) is provided. In other words, the absolute limits on $\omega_{n_{sp}}$ and n/α indicate when it is necessary to change the mode of control of flight path from rotation of the whole airplane through use of the elevator to direct control of lift through thrust vectoring or circulation control by flap actuation, etc.

It should not be assumed, however, that direct-lift control can be used to cure all the problems associated with low $\omega_{n_{sp}}$ and low n/α. The limited experience with DLC to date indicates that DLC can improve control of flight path in certain situations. This experience has also demonstrated, however, that the way in which the DLC is mechanized can either improve or degrade flying qualities. Some important considerations in the design of a direct-lift control system are:

1) Should a separate controller be provided to the pilot, or should the DLC be hooked to the throttle or elevator?

2) If a separate controller is used, should it command attitude, rate of climb, or normal acceleration?

3) Are the drag changes associated with the DLC acceptable?

4) Are the pitching moments caused by the DLC acceptable?

Also note that some control of pitch attitude must be provided, even if DLC is used. In the approach Flight Phase, for instance, there are constraints on the attitude during the actual touchdown; and these attitudes must be achieved in the presence of turbulence, as well as in smooth air. In addition, certain flight phases such as air-to-air gunnery require precise attitude tracking. For these flight phases, the use of DLC will probably not solve the problems associated with low $\omega_{n_{sp}}$.

Data Interpretation

The pilot rating data used to set the Level 1 and 2 limits on $\omega_{n_{sp}}$ as a function of n/α were taken only from in-flight programs. This was done to insure that the motion cues and tasks were realistic. There is a surprisingly large amount of such data available (References B59, B102, C18, C57,

D3, D23, D29, D34, D41, D43, D52, H4, H12, and J60), but only References D3, D52, and J60 contain data to define the upper limits of $\omega_{n_{sp}}$. The data of Reference D29 were not used at all because F_s/n was not held within the Level 1 limits and because the report did not contain enough information to properly identify the short-period mode. The data are shown in Figures 1 through 21 for programs involving Category A Flight Phases and in Figures 22 through 25 for programs involving Category C Flight Phases. Figure 26 shows the XB-70 data used to define the boundaries for Category B Flight Phases. All these figures show the $\omega_{n_{sp}}$ boundaries specified in 3.2.2.1.1 to facilitate comparison with the data. The maximum and minimum boundaries specified in 3.2.2.1.2 are also shown on Figures 1 through 25. The data points have been divided into three groups by pilot rating, each group having a different symbol. The groups are: Level 1 ratings, Level 2 ratings, and ratings worse than Level 2. The specific rating for each data point is omitted for clarity of presentation, except where needed in areas having little data. Unless otherwise stated, the rating data use the numerical CAL rating scale.

The $\omega_{n_{sp}}$ boundaries used in the specification do not fit the data from all the programs perfectly; this is the nature of flying qualities data. If the data are viewed as a whole, however, it can be seen that the trend of changing $\omega_{n_{sp}}$ limits with n/α is clearly indicated. Many details of each program were taken into account in fairing the boundaries and are too numerous to fully discuss here, but a few of the most important considerations can be mentioned. Two data points in the middle of the Level 1 region of Figure 6 are rated 7.0; these ratings are in complete disagreement with the pilot comments. The data of Figures 8 and 9 were taken at a flight condition where the maneuvering of the airplane was limited to a total load factor of 2.0 because of buffet; this bothered the CAL pilot especially. The data of Figures 12 and 13 were taken at heavy fuel loads which lowered the wing structural natural frequencies enough to couple with the short-period mode when $\omega_{n_{sp}}$ was high; again, this especially bothered the CAL pilot. The data of Figure 14 do not fit the criteria very well, but the data of Figures 11 through 13 were obtained for essentially the same value of n/α and they do fit the criteria. The pilot whose data are shown in Figure 16 had his own definition of the rating scale and the task, and his rating data exhibit an unusual amount of scatter. Figures 14 through 17 all indicate a possible upper limit on $\omega_{n_{sp}}$ which is considerably lower than the boundaries used in the specification; however, these pilot ratings were not confirmed by the experiment of Reference D3 which evaluated configurations with considerably higher values of $\omega_{n_{sp}}$. Notice that many points on Figures 6, 7, 9, 12, 13, 18, 19 and 20 are labeled " F_s/n high"; this means that the pilot rating was influenced by the fact that F_s/n was near or outside the F_s/n limits of paragraph 3.2.2.2.1.

The data of Figures 1 through 26 define lower limits on $\omega_{n_{sp}}$ for Category A, B, and C Flight Phases and upper limits for Category A Flight Phases (although the Level 2 upper limit is not well defined). The upper limits for Categories B and C were arbitrarily made the same as for Category A, due to the complete absence of definitive data from in-flight evaluations.

Figures 1 through 26 contain only 4 data points rated 9.0 (CAL) or worse. In addition, data presented in the discussion of 3.2.1.1 show that unstable short-period modes (i.e., aperiodically divergent angle-of-attack response, even at constant speed) are controllable under some conditions. The data would seem to indicate that the Level 3 requirements of 3.2.2.2 and 3.2.2.2.1 indirectly provide sufficient Level 3 limits on $\omega_{n_{sp}}$, since positive values of F_s/n and δ_s/n are indicative of short-period stiffness. However, pilot comments indicate that the safety aspects of the situation deteriorate rapidly as zero short-period stiffness is approached. This is especially true when the pilot is busy with other cockpit duties, and the problem is compounded during flight in turbulence. Because of the uncertainties associated with low short-period stiffness, the lower $\omega_{n_{sp}}$ boundary for Level 3 was made coincident with the Level 2 boundary (there are insufficient data to establish a Level 3 upper limit on $\omega_{n_{sp}}$). As a result, in the absence of approved Special Failure States, this Level 2 requirement must generally be met even in the event of total stability augmentation failures.

After the $\left(\omega_{n_{sp}}^2 / \frac{n}{\alpha}\right)$ limits were established, concern was expressed that no absolute lower bounds on either $\omega_{n_{sp}}$ or n/α had been established, due to lack of data. In September of 1968, preliminary data for the carrier approach Flight Phase were made available. These data were generated during a Navy-sponsored moving-base simulator experiment conducted by Grumman aircraft. The results of this experiment were used to establish Category C Level 1 and 2 lower limits on $\omega_{n_{sp}}$ and n/α for Class I, II-C, and IV airplanes (see Figures 27 and 28). Even though the data were obtained for the carrier approach, it seems logical that Class I and IV airplanes should meet the same requirements. Carrier airplanes need more precise control of flight path and attitude, but Class I and IV land-based airplanes do more maneuvering in the landing pattern and execute a flare to land.

The lower Category C limits on n/α for Class II-L and III airplanes are somewhat arbitrary, although NASA experience with the Boeing 367-80 indicates that pilot ratings of 3.0 to 4.5 can be expected for values of n/α in the vicinity of 1.4 (see References C58 and G12). The absolute lower $\omega_{n_{sp}}$ limits for Category C Flight Phases (Classes II-L and III) and Category A (all classes) were established rather arbitrarily, since all the available data on low values of $\omega_{n_{sp}}^2$ for these Flight Phases are adequately handled by the $\left(\omega_{n_{sp}}^2 / \frac{n}{\alpha}\right)$ criteria.

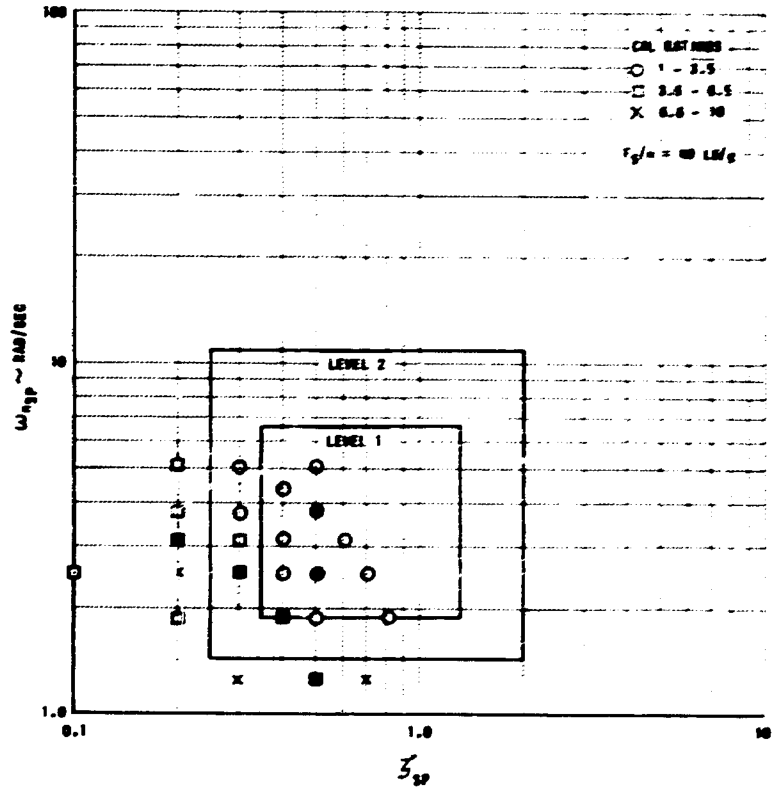


Figure 1 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, $n/\alpha = 12.3$
(B-26, PILOT A, LONG-LOOK, REFERENCE D23)

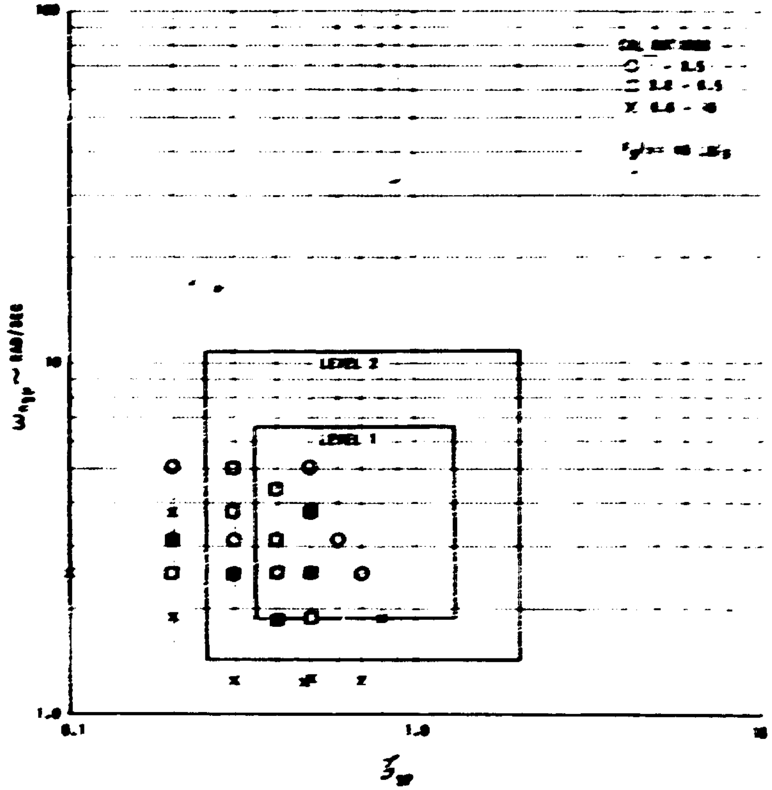


Figure 2 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, $n/\alpha = 12.3$
(B-26, PILOT B, LONG-LOOK, REFERENCE D23)

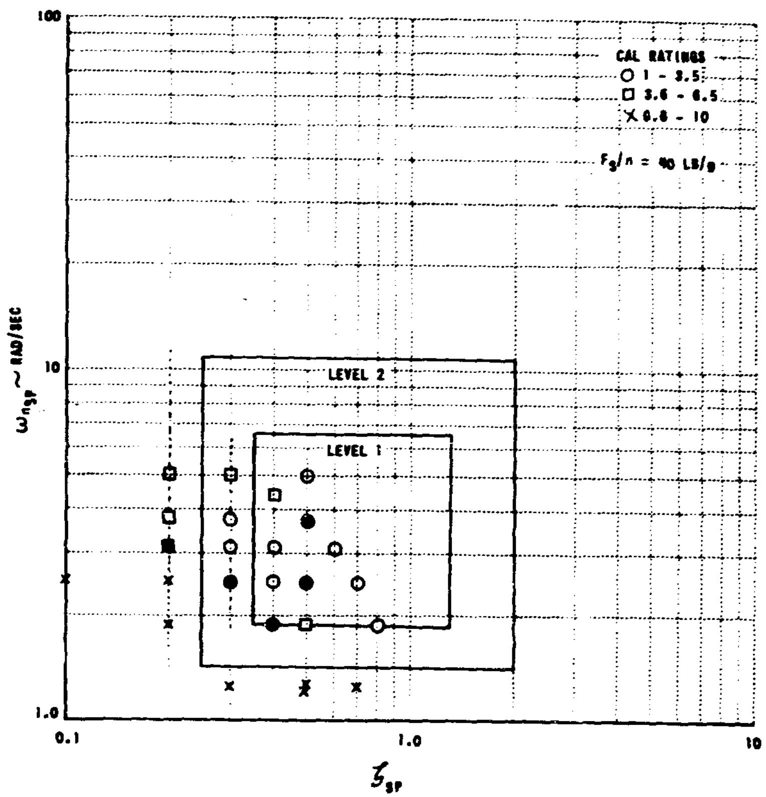


Figure 3 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, $n/\alpha = 12.3$
(B-26, PILOT C, LONG-LOOK, REFERENCE D23)

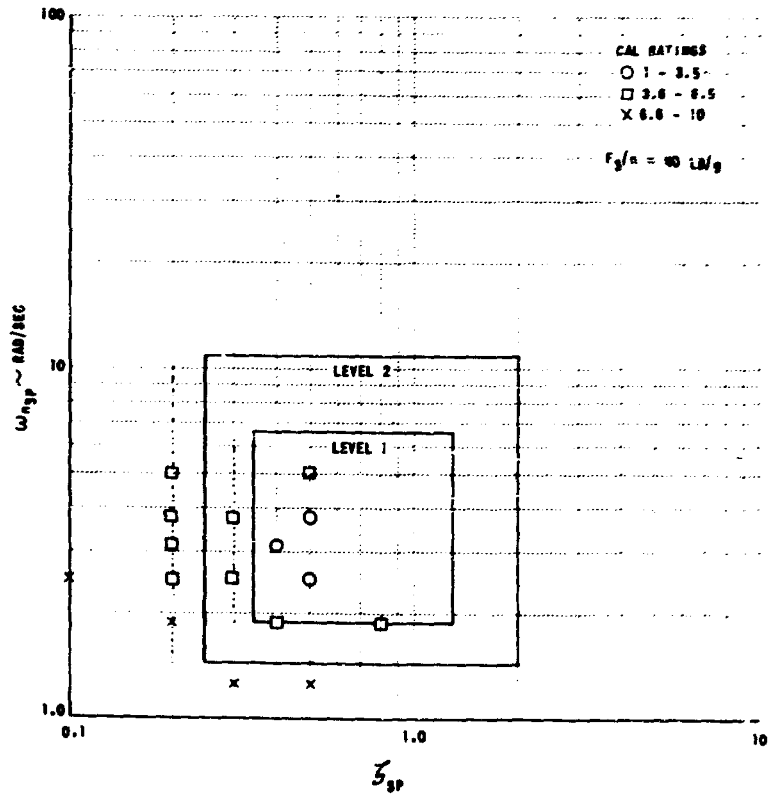


Figure 4 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, $n/\alpha = 12.3$
(B-26, AVG. OF ALL PILOTS, SHORT-LOOK, REFERENCE D23)

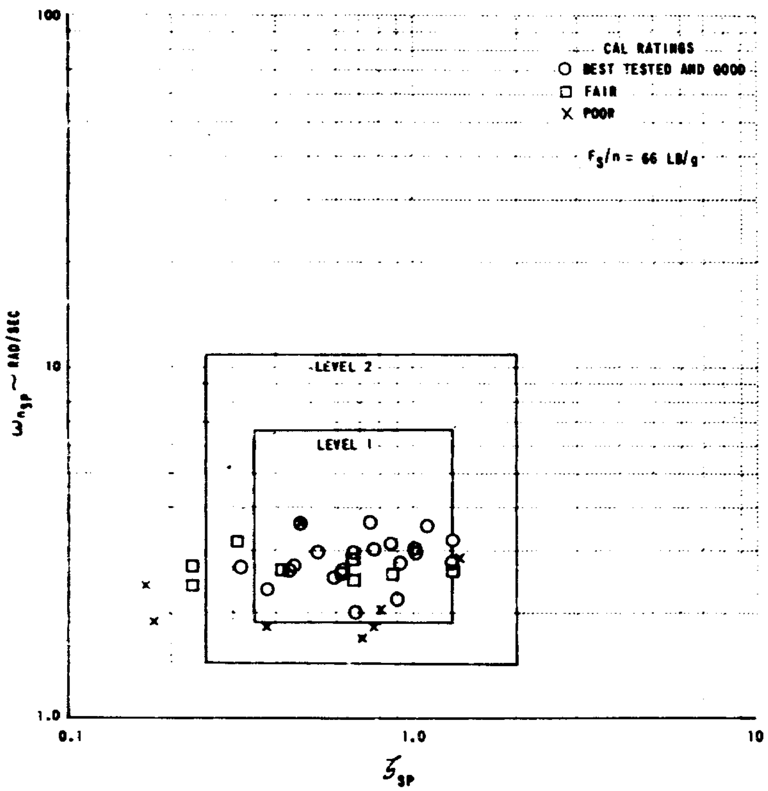


Figure 5 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, n/α = 12.3
(B-26, REFERENCE D43)

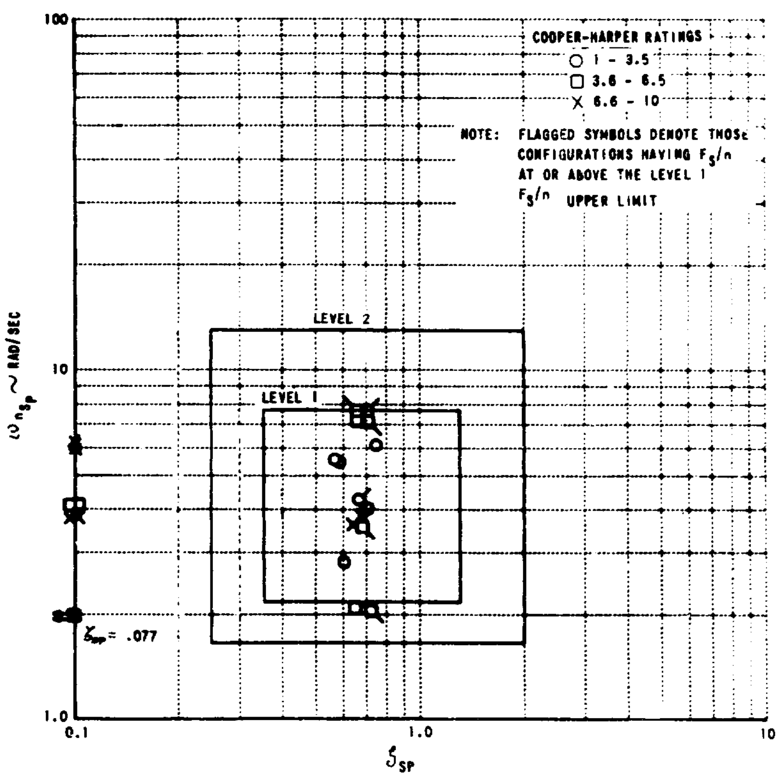


Figure 6 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, n/α = 16.5
(T-33, PILOT A, REFERENCE D52)

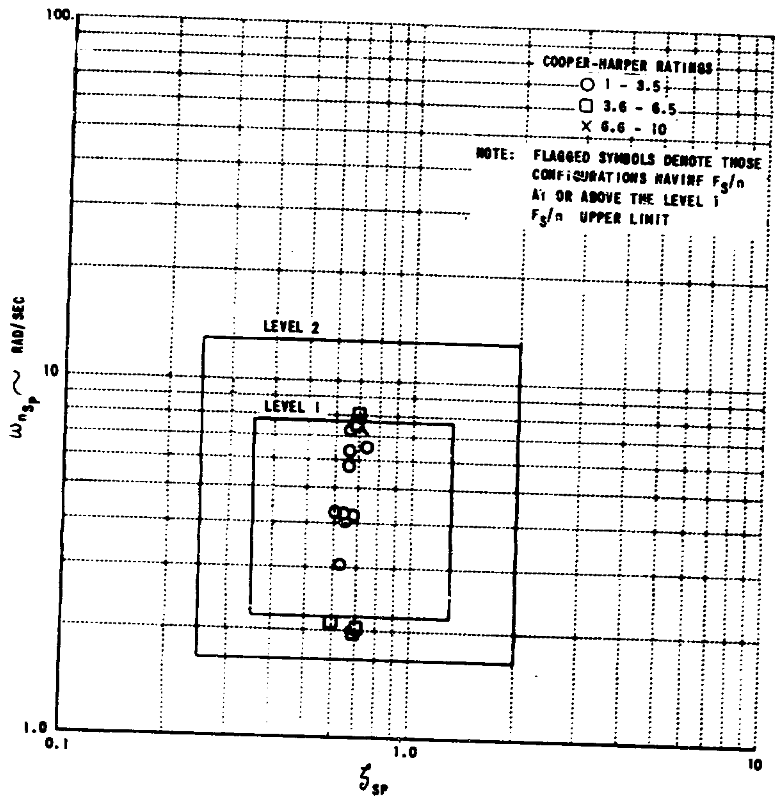


Figure 7 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, n/α = 16.5
(T-33, PILOT B, REFERENCE D52)

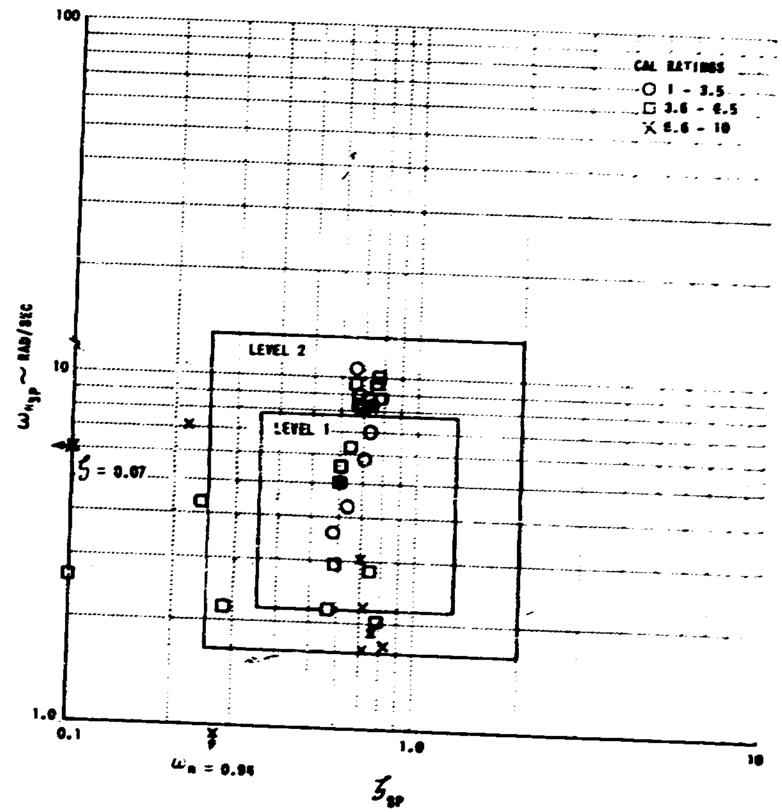


Figure 8 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, n/α = 16.9
(T-33, CAL PILOT, REFERENCE D3)

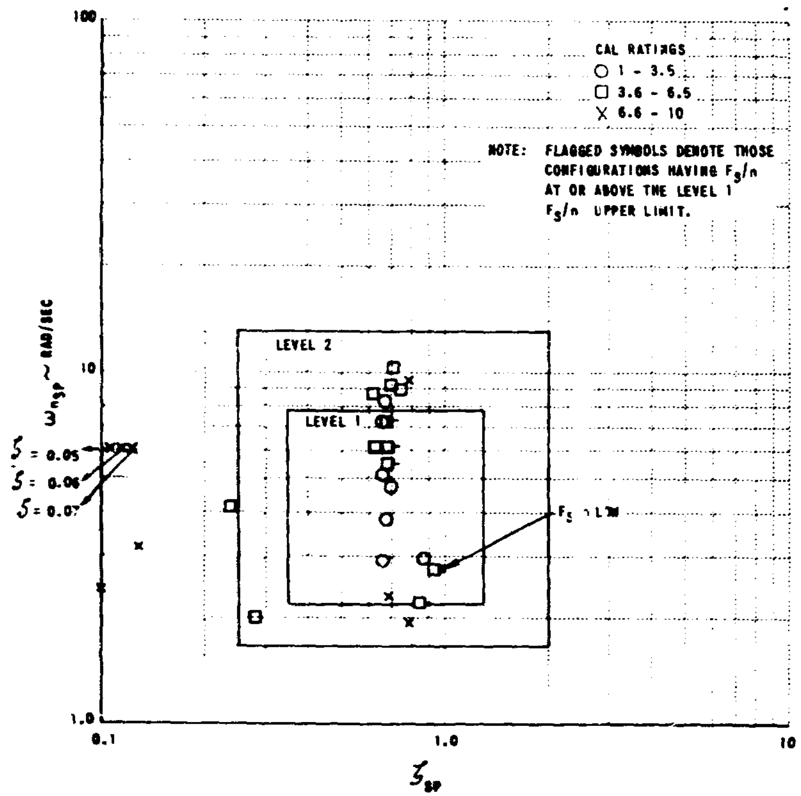


Figure 9 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, n/α = 16.9
(T-33, USAF PILOT,, REFERENCE D3)

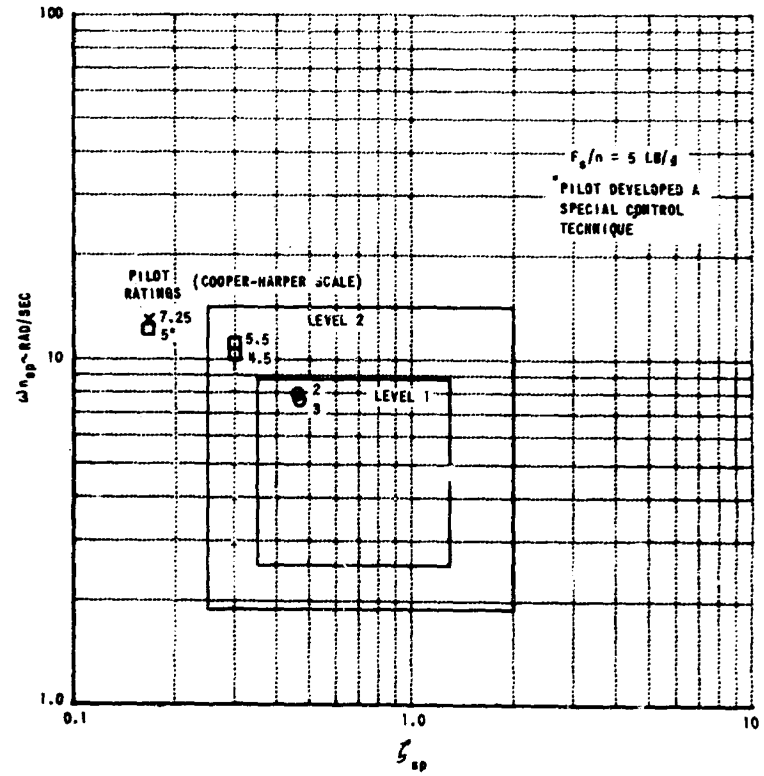


Figure 10 (3.2.2.2.1)
CATEGORY A FLIGHT PHASES, n/α = 22
(T-33, REFERENCE J60)

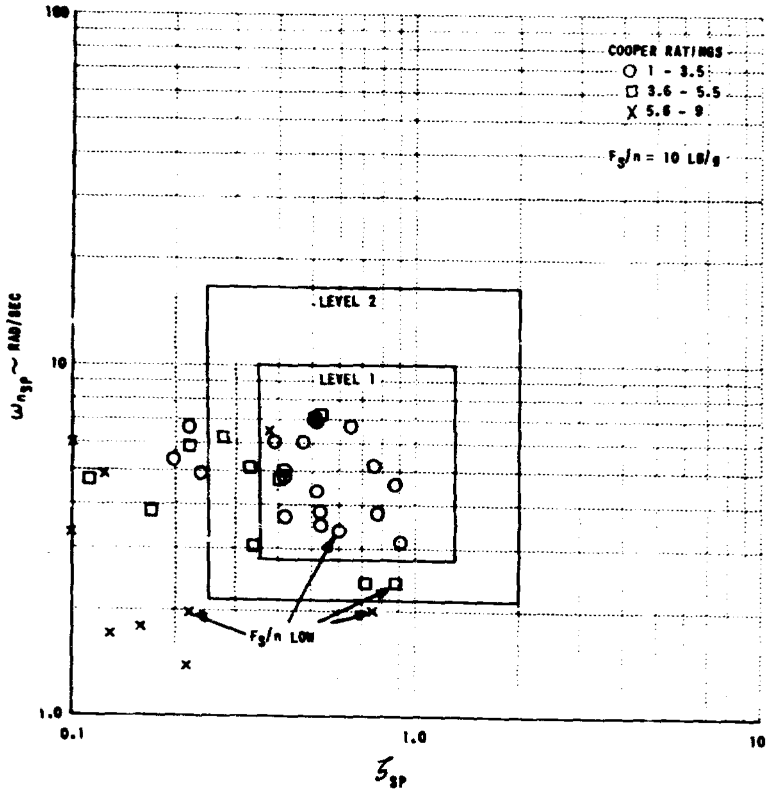


Figure 11 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, n/α = 27.2
(F-86, FAST CONTROL SYSTEM, REFERENCE B59)

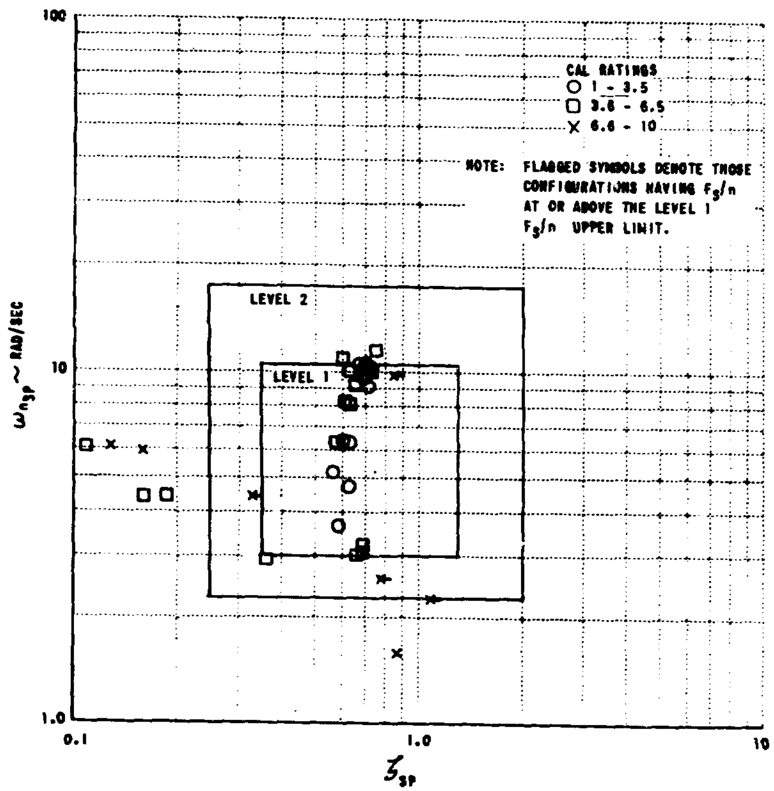


Figure 12 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, n/α = 30.1
(T-33, CAL PILOT, REFERENCE D3)

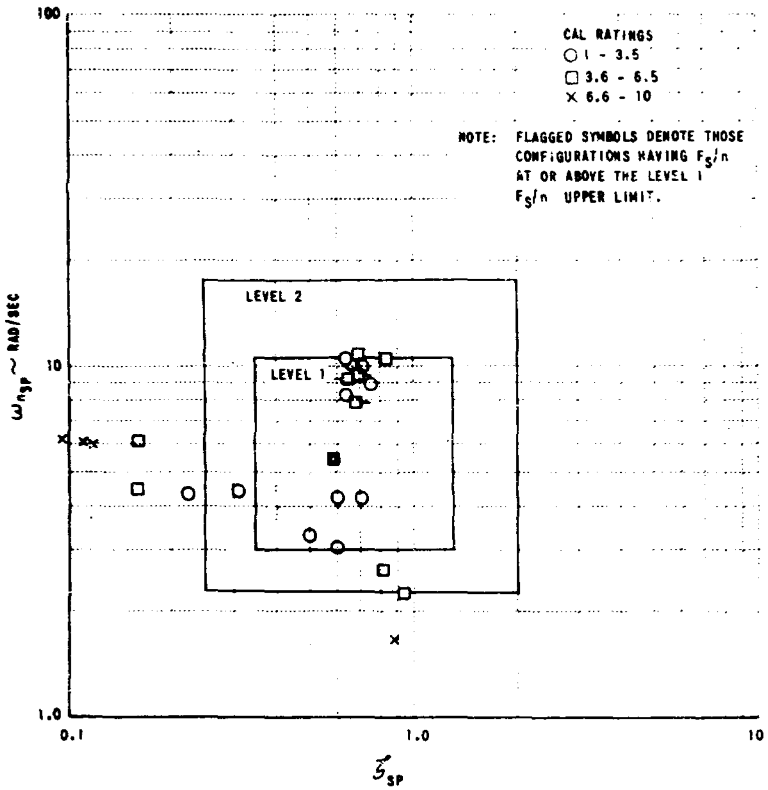


Figure 13 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, n/α = 30.1
(T-33, USAF PILOT, REFERENCE D3)

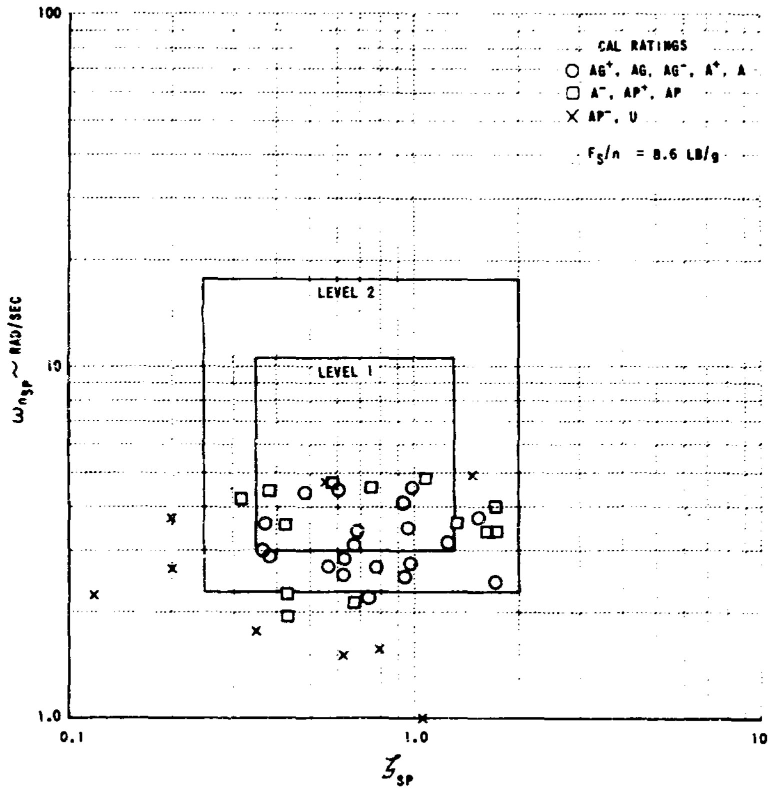


Figure 14 (3.2.2.1.1)
CATEGURY A FLIGHT PHASES, n/α = 31
(F-94, REFERENCE D41)

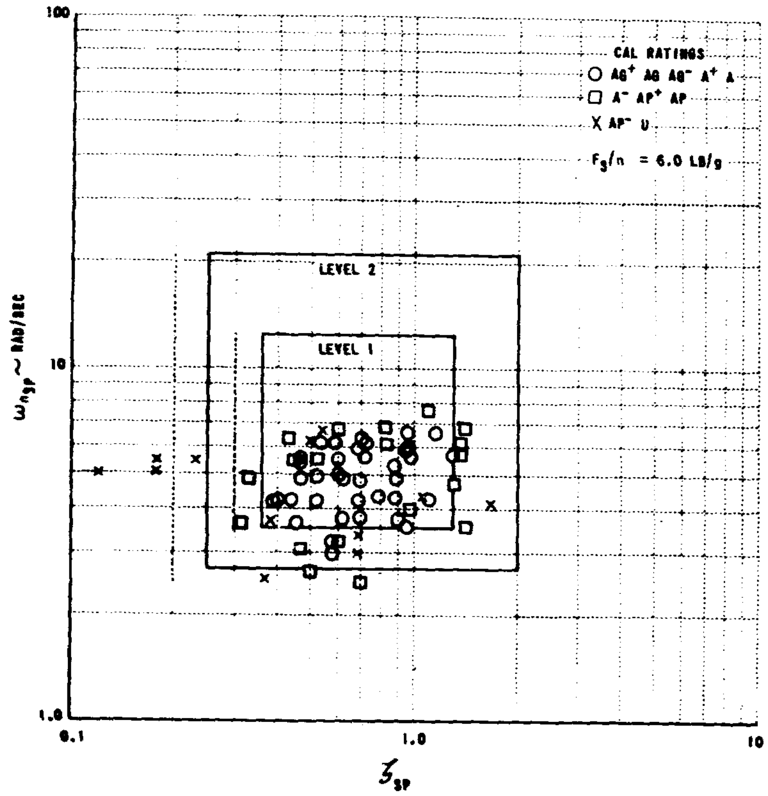


Figure 15 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, n/α = 43
(F-94, CAL PILOT, REFERENCE D34)

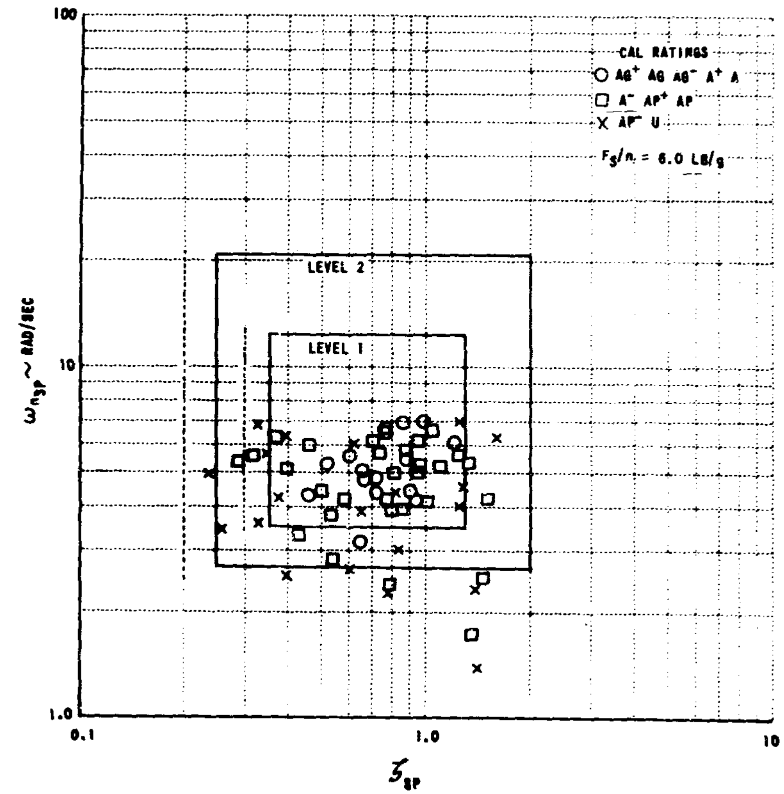


Figure 16 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, n/α = 43
(F-94, 1st USAF PILOT, REFERENCE D34)

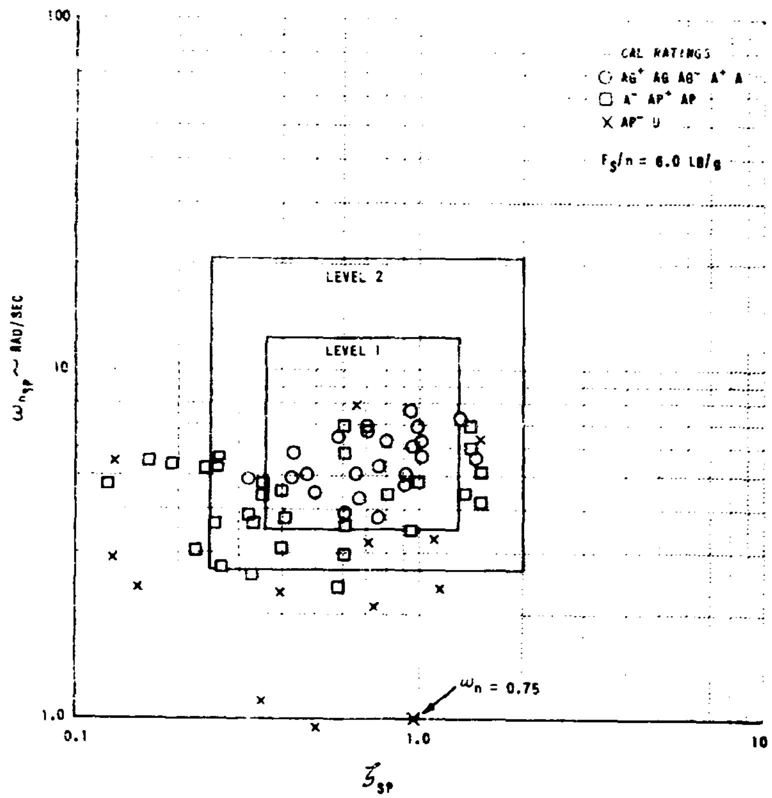


Figure 17 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, n/α = 43
(F-94, 2nd USAF PILOT, REFERENCE D34)

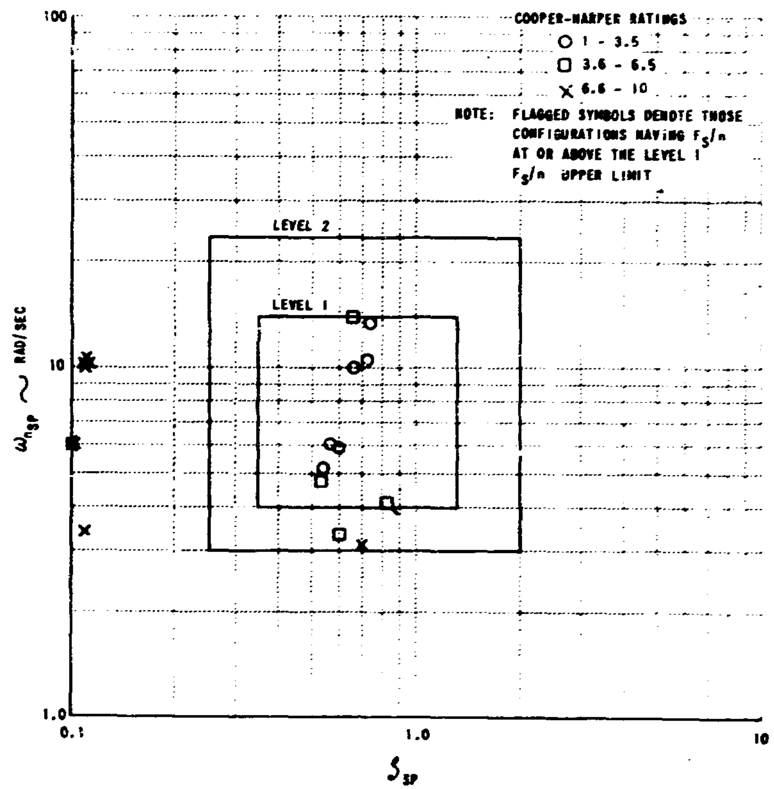


Figure 18 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, n/α = 56.2
(T-33, PILOT A, REFERENCE D52)

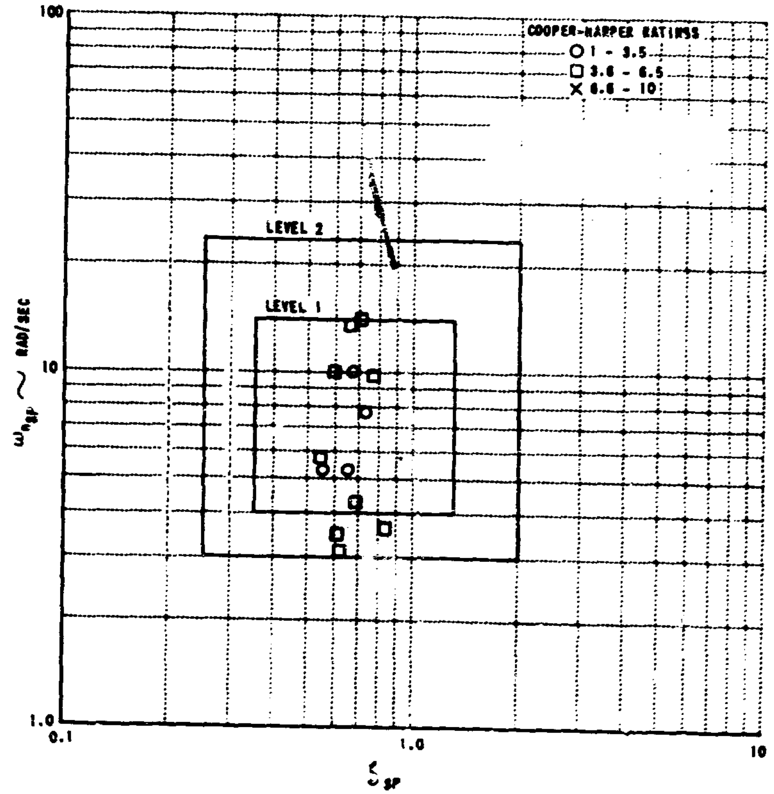


Figure 19 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, $n/\alpha = 56.2$
(T-33, PILOT B, REFERENCE D52)

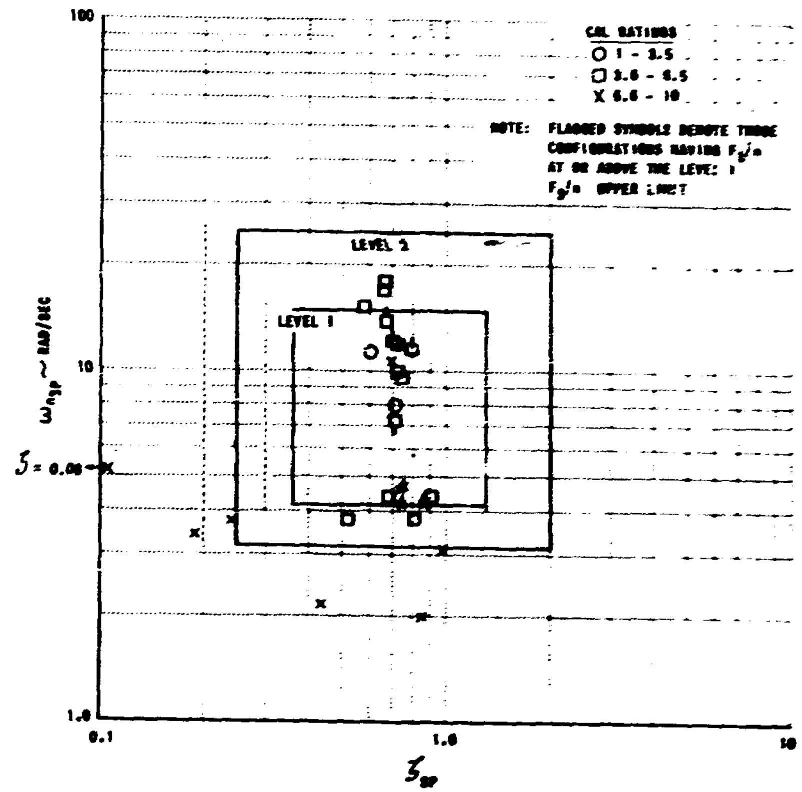


Figure 20 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, $n/\alpha = 61.5$
(T-33, CAL PILOT, REFERENCE D3)

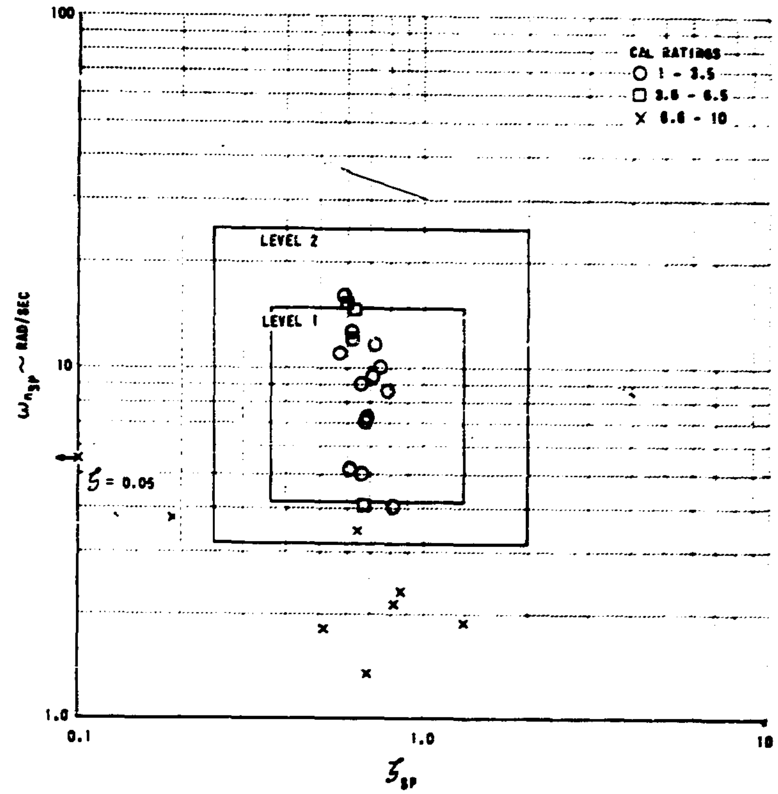


Figure 21 (3.2.2.1.1)
CATEGORY A FLIGHT PHASES, $n/\alpha = 61.5$
(T-33, USAF PILOT, REFERENCE D3)

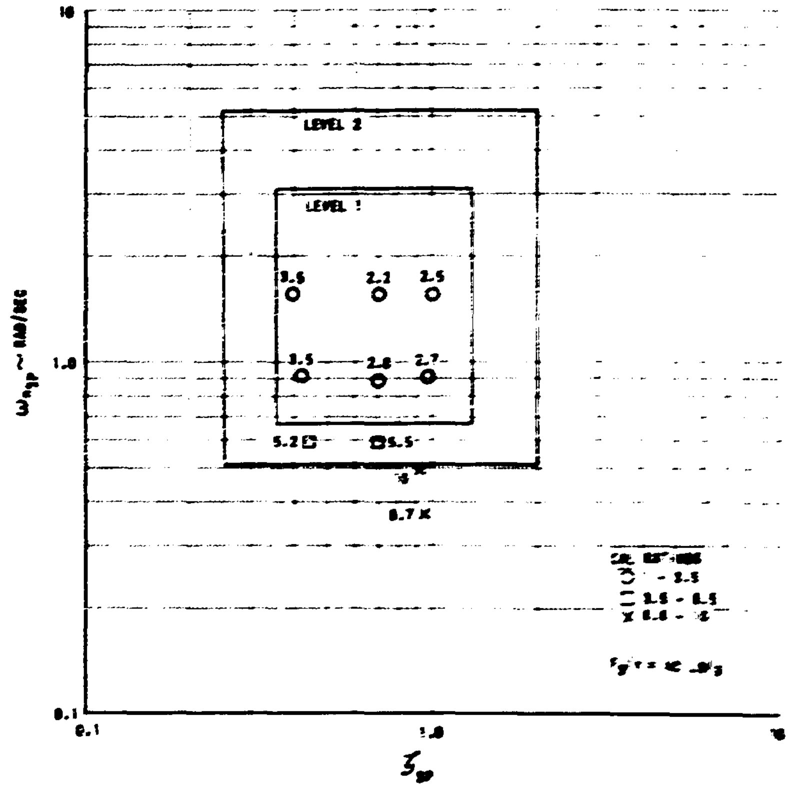


Figure 22 (3.2.2.1.1)
LANDING APPROACH, $n/\alpha = 2.76$
(BOEING 367-80, REFERENCE C18)

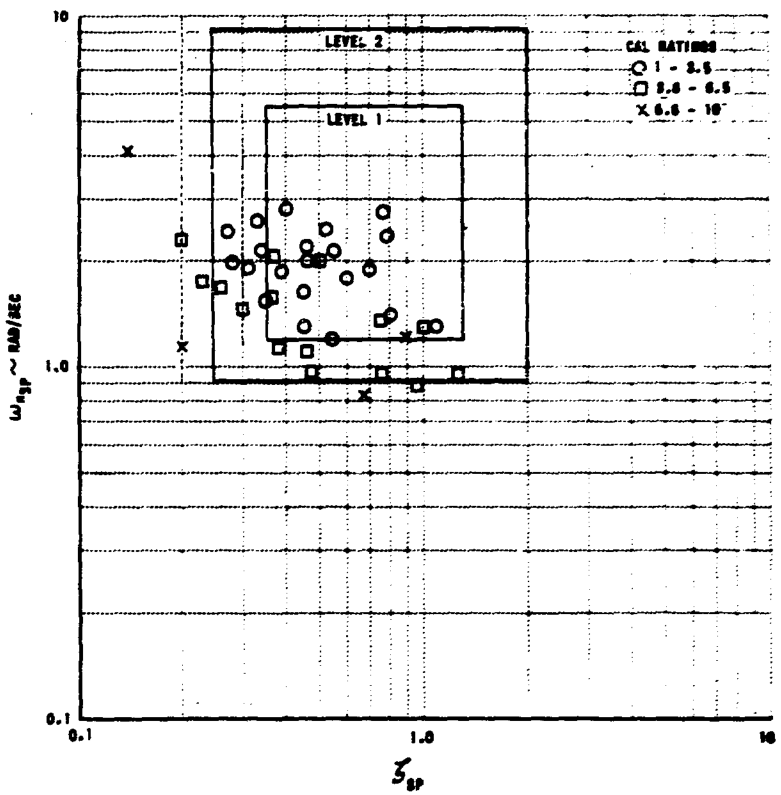


Figure 23 (3.2.2.1.1)
LANDING APPROACH, $n/\alpha = 8.5$
(T-33, "FRONT-SIDE" DATA ONLY, REFERENCE E12)

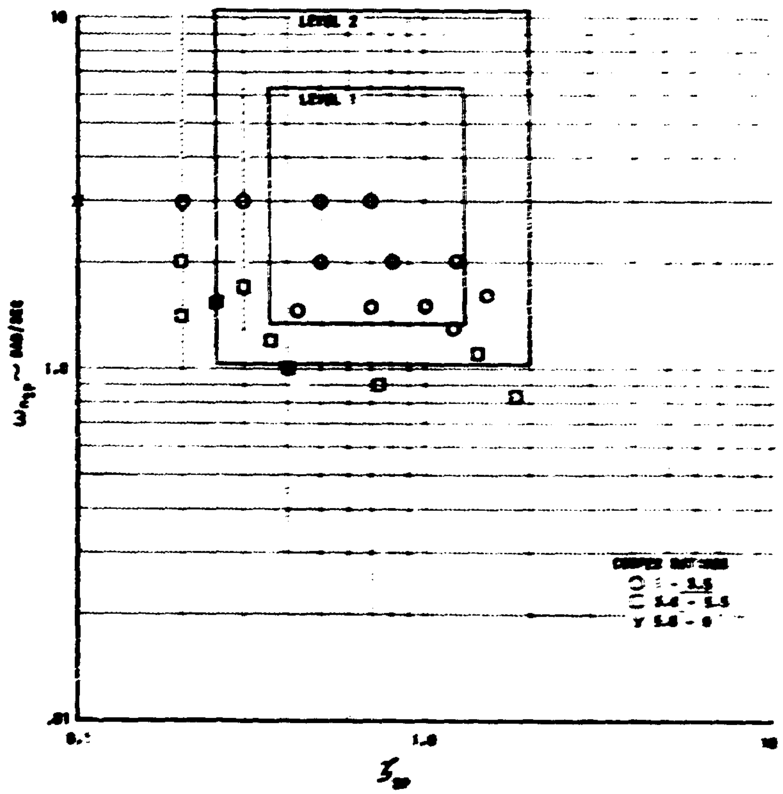


Figure 24 (3.2.2.1.1)
LANDING APPROACH, $n/\alpha = 11.0$
(NAVION, OPTIMUM F_s/n, REFERENCE E4)

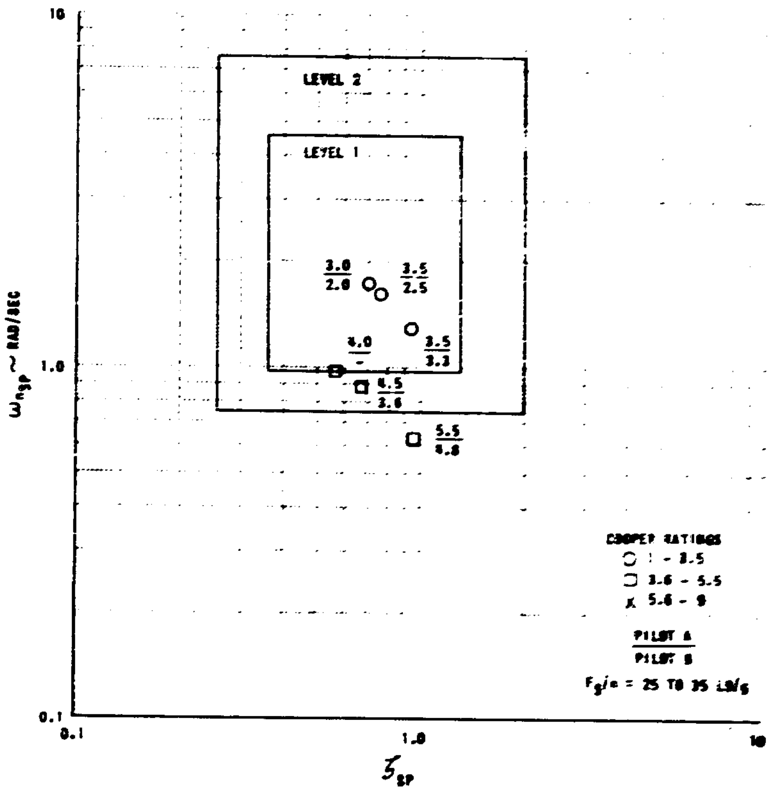


Figure 25 (3.2.2.1.1)
LANDING APPROACH, n/α = 5.5 (BOEING 367-80, "FRONT-SIDE" DATA ONLY, REFERENCE C57)

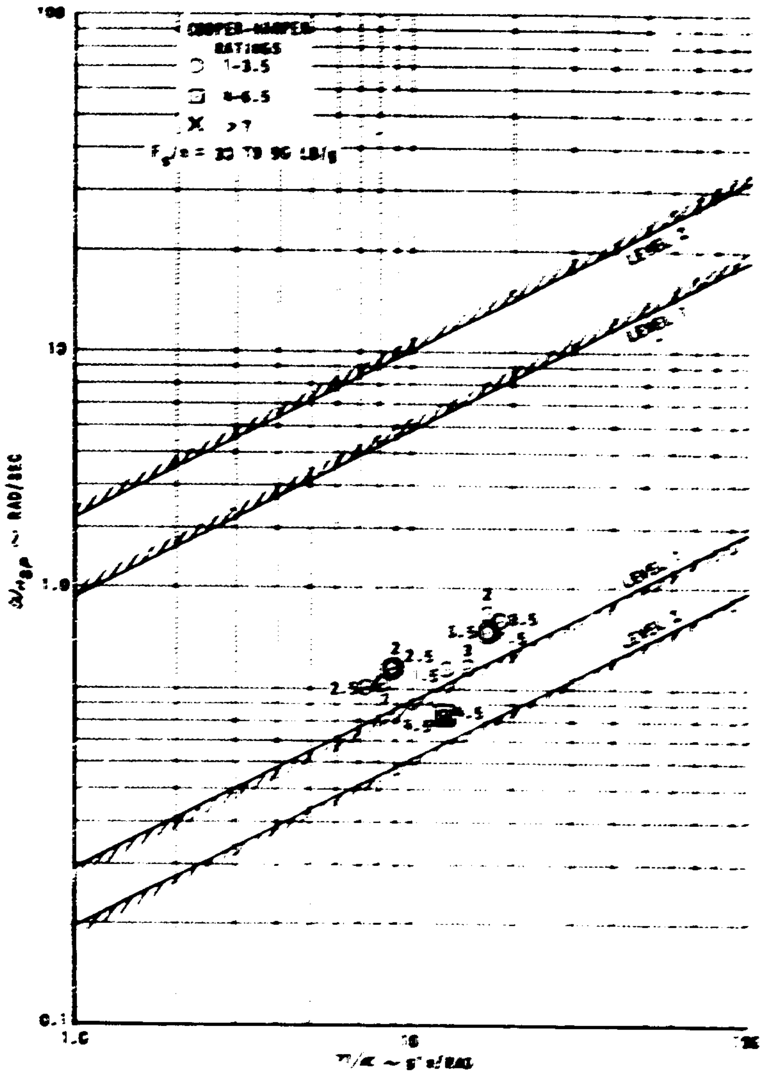


Figure 26 (3.2.2.1.1)
CATEGORY B FLIGHT PHASES, $\zeta_{sp} \geq 0.20$
(XB-70, REFERENCE B102)

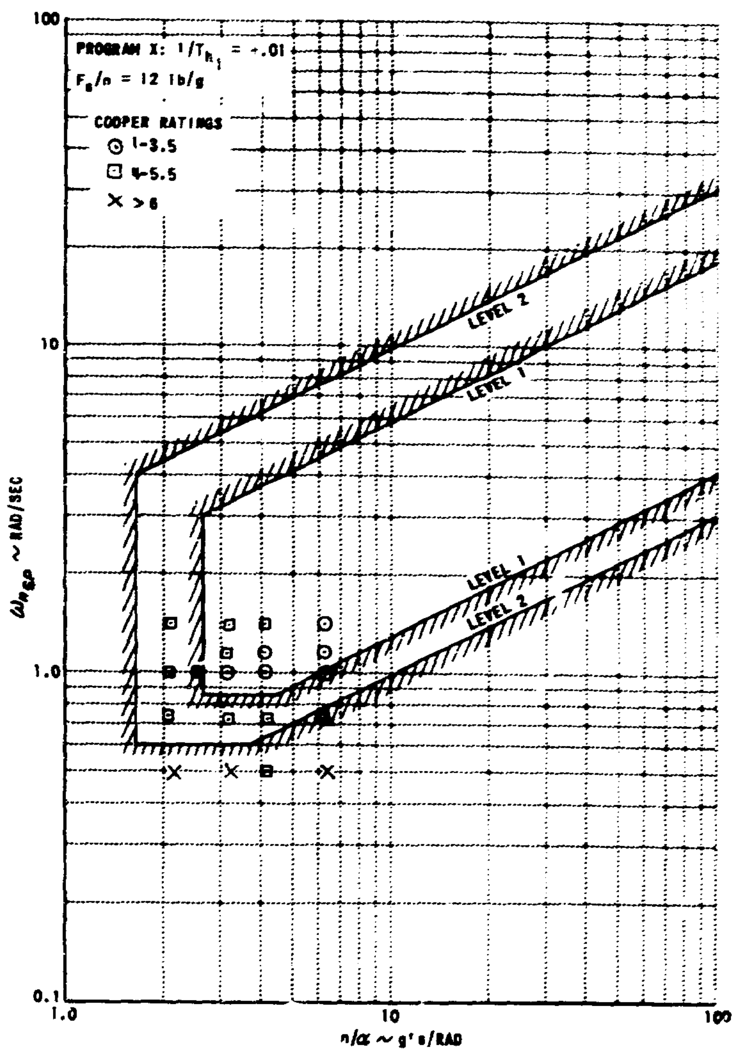


Figure 27 (3.2.2.1.1)
CARRIER APPROACH, ζ_{SP} = .55
(MOVING-BASE SIMULATOR, NAVY/GRUMMAN)

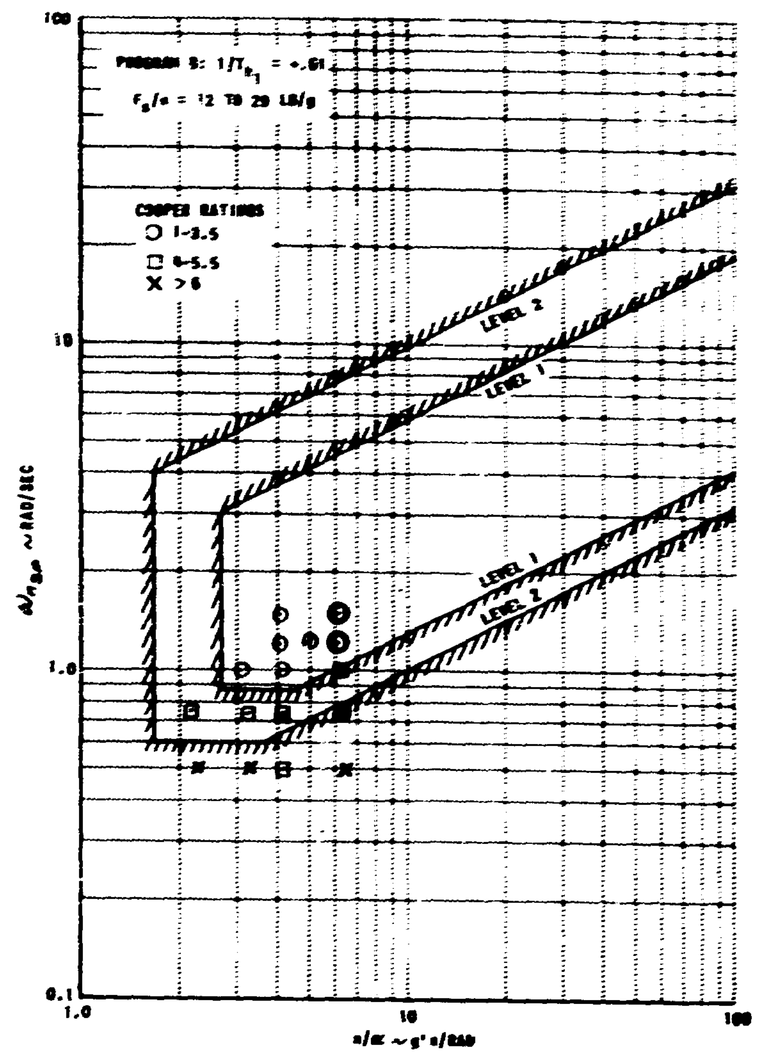


Figure 28 (3.2.2.1.1)
CARRIER APPROACH, $\zeta_{SP} \geq 0.30$
(MOVING-BASE SIMULATOR, NAVY/GRUMMAN)

3.2.2.1.2 SHORT-PERIOD DAMPING

REQUIREMENT

3.2.2.1.2 Short-period damping. The short-period damping ratio, ζ_{SP}, shall ,e within the limits of table IV.

TABLE IV. Short-period Damping Ratio Limits

Level	Category A and C Flight Phases		Category B Flight Phases	
	Minimum	Maximum	Minimum	Maximum
1	0.35	1.30	0.30	2.00
2	0.25	2.00	0.20	2.00
3	0.15*	—	0.15*	—

*May be reduced at altitudes above 20,000 feet if approved by the procuring activity.

RELATED MIL-F-8785 PARAGRAPHS

3.3.5

DISCUSSION

The discussion of 3.2.2.1.1 pertains to what is important to the pilot when the short-period frequency is satisfactory. However, everyone agrees that damping is also important. When the damping is too low, the airplane short-period response overshoots and oscillates. When the damping is too high, the response is sluggish. Therefore, upper and lower limits have been placed on short-period damping ratio. From the constant-speed equations of motion, the expression for short-period damping ratio is

$$\zeta_{sp} = -\frac{Z_w + M_q + VM_{\dot{w}}}{2\omega_{n_{sp}}}$$

The data used to define the Level 1 and 2 damping ratio limits for Category A and C Flight Phases are shown in Figures 1 through 24 of the short-period frequency section, and details of the data are discussed in that section. It is often argued that the damping ratio boundaries should be curved to cut off the corners of the short-period boxes. In particular, the lower left-hand corner could be cut off by a lower limit on $(2\zeta_{sp}\omega_{n_{sp}})$, the upper left-hand corner might be cut off to avoid combining low damping with sensitivity, and the lower right-hand corner might be restricted to avoid making an already sluggish airplane even more sluggish. However, no such limits were used because the additional complication seemed unjustified in view of the fact that constant damping ratio lines bound the available data quite adequately.

The same damping ratio lines seem to bound the Category A and Category C data equally well; and therefore, no attempt was made to separate the requirements for those two Flight Phase Categories.

The XB-70 data presented in Figure 1 might be used to establish the Level 1 and 2 lower limits for Category B Flight Phases. The upper limits were set equal to 2.00, based on engineering judgement.

There are very little data available to establish Level 3 limits on ζ_{SP}, but the trends indicate that the lower limit is probably below 0.05.

Summarizing, the data indicate these limits on ζ_{sp} :

Level	Category A and C Flight Phases		Category B Flight Phases	
	Minimum	Maximum	Minimum	Maximum
1	0.35	1.30	0.18	2.00
2	0.20	2.00	0.07	2.00
3	<0.05	—	<0.03	—

Although the data used to establish the ζ_{sp} limits included the effects of turbulence to some degree, there was some concern that the lower limits indicated by the data are not adequate for flight in turbulence. Therefore, the lower limits were increased to the values specified in 3.2.2.1.2.

Measurement and data reduction techniques are discussed in Appendices III and IVC and D.

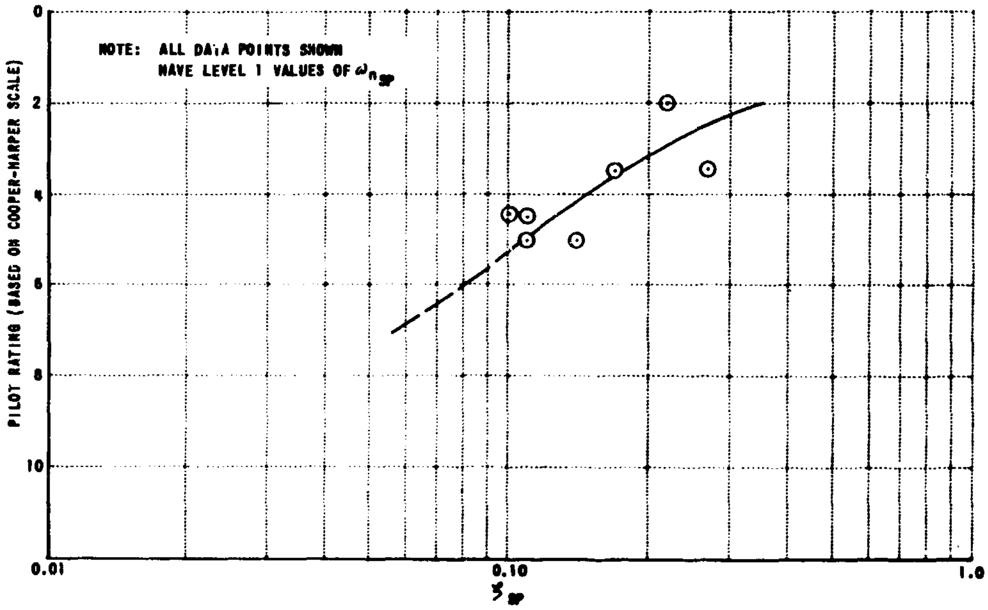


Figure 1 (3.2.2.1.2)
CATEGORY B FLIGHT PHASES (XB-70, REFERENCE B102)

3.2.2.1.3 RESIDUAL OSCILLATIONS

REQUIREMENT

3.2.2.1.3 Residual oscillations. Any sustained residual oscillations shall not interfere with the pilot's ability to perform the tasks required in service use of the airplane. 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, as will pitch attitude oscillations greater than ±3 mils for Category A Flight Phases requiring precision control of attitude. These requirements shall apply with the elevator control fixed and with it free.

RELATED MIL-F-8785 PARAGRAPHS

3.3.5

DISCUSSION

This paragraph is essentially a rewrite of part of 3.3.5 of MIL-F-8785. The primary purpose of the requirement is to prevent limit cycles in the control system or structural oscillations which might compromise tactical effectiveness, cause pilot discomfort, etc. The limit of ±5 mils for pitch attitude oscillations was reduced to ±3 mils, on the recommendation of several USAF pilots.

3.2.2.2 CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

REQUIREMENT

3.2.2.2 Control feel and stability in maneuvering flight. In steady turning flight and in pullups at constant speed, increasing pull forces and aft motion of the elevator control and airplane-nose-up deflection of the elevator surface are required to maintain increases in normal acceleration throughout the range of service load factors defined in 3.1.8.4. Increases in push force, forward control motion, and airplane-nose-down deflection of the elevator surface are required to maintain reductions of normal acceleration in pushovers.

RELATED MIL-F-8785 PARAGRAPHS

3.3.4, 3.3.9

DISCUSSION

This paragraph is essentially a restatement of part of 3.3.9 of MIL-F-8785. Stable control force and position variations with normal acceleration at constant speed ensure that the airplane has a stick-free and stick-fixed short-period mode; that is, there is a restoring tendency which tries to rapidly return the airplane to 1 g flight following a disturbance. This relationship is analogous to that between the control force and position variations with speed and the phugoid mode.

The requirement for a stable variation of elevator surface position with normal acceleration is simply a restatement of 3.3.4 of MIL-F-8785. This requirement was retained to ensure that the basic airframe has a restoring tendency, at least in the short term. This is an absolute limit on the aft c.g. location.

Here is another place where some data show that slight instability (negative $\omega^2_{n_{sp}}$) might be tolerable for Level 3 in some instances. But concern for requirement and design uncertainties and the possibility of having several Level 3 flying qualities at the same time, has resulted in keeping a conservative requirement.

Flight test methods are discussed in Appendix IVE.

3.2.2.2.1 CONTROL FORCES IN MANEUVERING FLIGHT

REQUIREMENT

3.2.2.2.1 Control forces in maneuvering flight. At constant speed in steady turning flight, pullups, and pushovers, the variations in elevator-control force with steady-state normal acceleration 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. All local force gradients shall be within the limits of table V. In addition, whenever the short-period frequency is near the upper boundaries of figure 1, F_s/n should be near the Level 1 upper boundaries of table V. This may be necessary to avoid abrupt response, sensitivity, or tendencies toward pilot-induced oscillations. The term gradient does not include that portion of the force versus n curve within the preloaded breakout force or friction band.

TABLE V. Elevator Maneuvering Force Gradient Limits

Center Stick Controllers		
Level	Maximum Gradient, $(F_s/n)_{max}$, pounds per g	Minimum Gradient, $(F_s/n)_{min}$, pounds per g
1	$\frac{240}{n/\alpha}$ but not more than 28.0 nor less than $\frac{56}{n_L-1}$ *	The higher of $\frac{21}{n_L-1}$ and 3.0
2	$\frac{360}{n/\alpha}$ but not more than 42.5 nor less than $\frac{85}{n_L-1}$ *	The higher of $\frac{18}{n_L-1}$ and 3.0
3	56.0	3.0
*For $n_L < 3$, $(F_s/n)_{max}$ is 28.0 for Level 1, 42.5 for Level 2.		

Table V (Cont.)

Wheel Controllers		
Level	Maximum Gradient, $(F_s/n)_{max}$, pounds per g	Minimum Gradient, $(F_s/n)_{min}$, pounds per g
1	$\frac{500}{n/\alpha}$ but not more than 120.0 nor less than $\frac{120}{n_L-1}$	The higher of $\frac{45}{n_L-1}$ and 6.0
2	$\frac{775}{n/\alpha}$ but not more than 182.0 nor less than $\frac{182}{n_L-1}$	The higher of $\frac{38}{n_L-1}$ and 6.0
3	240.0	6.0

RELATED MIL-F-8785 PARAGRAPHS

3.3.9, 3.3.9.1, 3.3.9.3, 3.3.9.5, 3.7.4

DISCUSSION

This paragraph is a complete rework of the (F_s/n) requirements of MIL-F-8785.

It was decided that the major differences in the desired maneuvering forces between fighter airplanes and transports are due to the type of controller, in addition to airplane class. The effects of airplane class (a grouping of types of missions really) seem to be adequately described by limit load factor, through the (K/n_L-1) formulae of MIL-F-8785. In addition, however, there are several arguments for having different maneuvering forces for center-stick and wheel controllers. For example, the lower limits on maneuvering forces must be higher with a wheel control because the pilot's arm is usually unsupported; whereas the pilot has very good vernier control with a center stick even with light forces because his forearm is partially supported on his thigh. In any case, pilots seem to agree that they cannot maintain the precision of control with a wheel that they can with a stick, and that the maneuvering control forces should be higher.

There is evidence from many sources that F_s/n at very low n/α can or should be higher than at high n/α. This idea is given impetus, for instance, by the results of the ground simulator program of Reference D19 and the in-flight evaluation of Reference C2. These results indicate that the pilot tends to select a constant value of F_s/n at high n/α, but a constant value of F_s/α at low n/α. This trend is possibly due to a gradual change from concern with n and structural protection at high n/α to concern with control of pitch attitude alone at low n/α. Specification of forces in the form of limits on F_s/α at low n/α can be accomplished by making the F_s/n limits vary inversely with n/α, as c i be seen from the following constant-speed relation:

$$\frac{F_s}{\alpha} = \left(\frac{F_s}{n}\right)\left(\frac{n}{\alpha}\right)$$

On the basis of these considerations, the limits on F_s/n were expressed in the form $\left(\frac{K}{n/\alpha}\right)$ at low n/α and $\left(\frac{K}{n_L-1}\right)$ at high n/α, with separate requirements for stick and wheel controllers.

Center-Stick Controllers

To provide data in this format for center-stick controllers, results of the in-flight T-33 programs of References D3 and E12 were studied, since these were the only in-flight short-period experiments found in which the pilots were allowed to select the "optimum" value of longitudinal control gain for each configuration tested. The data from these two programs are presented in Figures 1 through 7 as plots of sensitivity, M_{F_s}, versus F_s/n. The slanted boundaries are simply the constant $(\omega^2_{n_{sp}}/\frac{n}{\alpha})$ limits of paragraph 3.2.2.1.1, as can be seen from the following constant-speed approximation:

$$\frac{\omega^2_{n_{sp}}}{n/\alpha} \approx \frac{F_s}{n} M_{F_s}$$

(See the discussion of paragraph 3.2.2.1.1, which also justifies the recommended high F_s/n at high $\omega_{n_{sp}}$.) The pilots were reasonably consistent in selecting the "optimum" gain. But, because of limited time available for gain selection, the pilots sometimes made mistakes in selecting the control gain, with the result that the configuration was given a poor pilot rating upon completion of the overall evaluation. The pilot ratings associated with configurations having misselected values of F_s/n were very useful in determining where the F_s/n boundaries should be drawn. In areas where there are no rating data available, a detailed analysis of pilot comment data for each program was made, since the pilot did conduct partial evaluations at several values of F_s/n in order to select the "optimum" value for each configuration.

For high values of n/α, the fighter values of $\left(\frac{21}{n_L-1}\right)$ and $\left(\frac{56}{n_L-1}\right)$ from MIL-F-8785 fit the pilot rating data of Figures 1 through 4 and the associated pilot comments quite well for Level 1 (assuming $n_L = 7$). Using the same rating data and comments, values of $(18/n_L-1)$ and $(85/(n_L-1))$ were chosen for the Level 2 F_s/n boundaries.

For low values of n/α, the pilot rating data of Figures 5 through 7 and the associated pilot comments indicate that the lines $F_s/n = 90/(n/\alpha)$ and $F_s/n = 240/(n/\alpha)$ would serve well as Level 1 boundaries.

A summary of the Level 1 and 2 center-stick boundaries which are reasonably well supported by experiment is presented in Figure 8. The low n/α portion of the lower Level 1 boundary is dashed to indicate that it is not very strongly supported by data. The following reasoning was used to establish the boundaries where rating data are lacking and pilot comments are sparse. It seemed logical to expect the Level 2 upper boundary to follow the same trend as the Level 1 upper boundary. The lower boundaries at low n/α are very difficult to set because the data in this area are very sparse. In fact, airplanes for which stick forces are due entirely to aerodynamic hinge moments from the elevator tend to have values of F_s/n which are the same at low values of n/α as they are at high values. Many successful airplanes have been built with this characteristic, which does not follow the trend indicated in Figure 8. Of course, it is possible that the F_s/n gradients of these airplanes were objectionable at low n/α . It is more likely, however, that these characteristics were not seriously objectionable because they were accompanied by an increasing stick motion per pound of stick force as n/α decreased. The experiments of References D3 and E12, on the other hand, maintained a fixed value of stick motion per pound of stick force. It appears that pilots will accept somewhat lower force gradients if they can regain some feel from stick position. Because of the general scarcity of F_s/n data at low n/α and the apparently significant but poorly understood effects of stick motion gradients in this flight regime, the lower limits on F_s/n were made constant for all values of n/α .

It is not likely that the upper limits of Figure 8 continue to increase indefinitely as n/α is decreased. Since the upper boundaries of Figure 8 are fairly well supported by data for values of n/α down to 8.5, any cutoffs that might exist must occur below n/α = 8.5. The rationale used to establish cutoffs was that the maximum stick force allowed to obtain an incremental 2.0 g at low n/α should be the same as the maximum force allowed to obtain n_L at high n/α .

The lower limit for Level 3 was set at 3.0 pounds per g, which is the absolute minimum specified in MIL-F-8785. The upper boundary for Level 3 was set by the criterion that an incremental 1.0 g could be obtained with the same stick force required to obtain n_L for the upper Level 1 limit at high n/α . The final boundaries for n_L = 7.0 with a center-stick controller are presented graphically in Figure 9. A similar plot can be constructed for any other combination of limit load factor and controller.

Wheel Controllers

Since there are few F_s/n data available for wheel controllers, the Level 1 limits for high n/α were initially set equal to the transport values of MIL-F-8785, which are $(45/(n_L-1))$ and $(120/(n_L-1))$. Most of the remaining limits for wheel controllers were established by multiplying the boundaries for stick controllers by the ratio of the Level 1 limits at high (n/α) for wheel and center-stick controllers. This ratio is $\frac{120/(n_L-1)}{56/(n_L-1)}$ or $\frac{45/(n_L-1)}{21/(n_L-1)}$ and is equal to 2.15. The upper Level 1 and 2 cutoffs at low n/α were established by the criterion that the maximum wheel force allowed to obtain an incremental 1.0 g at low n/α should be the same as the maximum force allowed to obtain n_L at high n/α. The upper Level 3 limit was set using the rationale that an incremental 0.5 g could be obtained with the same wheel force required to obtain n_L for the upper Level 1 limit at high n/α. The final boundaries for n_L = 3.0 with a wheel controller are presented graphically in Figure 10.

To confirm whether or not the wheel-controller boundaries obtained in this manner are of the right magnitude, they were compared against the limited data available. The primary source of such data is Reference D44. There is considerable disagreement among the pilots used in this program as to which values of F_s/n are desirable, which indicates that the evaluation tasks were not explained to the pilots carefully enough in preflight briefings, that is, each pilot was probably conducting his evaluation somewhat differently. The major differences between pilots, however, can be accounted for by dividing the 12 pilots into two groups - one group consisting of those pilots having primarily fighter backgrounds, and the other consisting of pilots having primarily multi-engine experience. Each pilot determined an "optimum" and an "acceptable" range of F_s/n in the evaluation. Since it is difficult to determine what "optimum" means in terms of pilot rating, only the "acceptable" range of data was utilized here. Figure 11 shows the ranges of F_s/n values which were "acceptable" to 50% of the fighter pilots and the ranges "acceptable" to 50% of the multi-engine pilots. (The lower limit of the range for n/α = 6.9 and the upper limit for n/α = 19 have not been shown because the text of Reference D44 states that these limits were arrived at somewhat hastily.) Shown in the same figure are the proposed requirements for wheel controllers for n_L = 2.7, which is the limit load factor for typical B-26 combat loadings, obtained from a B-26 pilot handbook. The rather large difference between the fighter and multi-engine pilots is largely due to a learning process which continued through much of the program for many of the pilots. That is, several of the pilots with multi-engine experience were used to flying large transports and bombers having extremely large values of F_s/n. These pilots showed a definite preference for very heavy gradients early in the program. They came to prefer lighter gradients as the program progressed, but the "acceptable" range of F_s/n includes configurations evaluated early in the program. The opposite sort of trend is true of the fighter pilots. They started the program with a preference for lighter gradients until they learned that heavier gradients were required to prevent inadvertent overstressing of the airplane. The multi-engine pilots gave up their preference for heavy gradients more slowly than the fighter pilots gave up the lighter gradients, however, so that the ranges of F_s/n values shown in Figure 11 for the fighter pilots are probably more valid than the ranges shown for multi-engine pilots. It is difficult to say whether the "acceptable" range should correspond to the Level 1 or Level 2 boundaries. With the above background comments in mind, however, Figure 11 shows that the requirements are the correct order of magnitude.

Also shown in Figure 11 are two values of F_s/n used in the Boeing 367-80 landing approach study of Reference C18. A value of 40 pounds per g is the value used throughout the short-period evaluations, but several pilots expressed a preference for 60 pounds per g.

Another source of data is experience gained from CAL's variable stability B-26 demonstration programs for the Air Force and Navy Test Pilot Schools. The demonstrations are conducted at an n/α of 12.3, and the pilot is told that he is evaluating a moderately maneuverable airplane with a limit load factor of 3.5. However, he is also told that the variable stability system has a load factor limit of 2.5 g. Assuming that the pilot is effectively evaluating a 3.0 g airplane, the requirements are 23 pounds per g and 60 pounds per g for Level 1, and 19 pounds per g and 91 pounds per g for Level 2. The CAL pilots running these programs have discovered that 40 pounds per g is a value that "makes most pilots happy," while 85 pounds per g is "too high" and 20 pounds per g is "too low." These values are in agreement with the proposed requirements; and while they were determined in a very qualitative manner, they are based on the evaluations of hundreds of pilots.

The pilot comments and ratings for the few data points of Reference D52 having F_s/n about 60 pounds per g indicate that this force is an appropriate upper value for Level 1 when $n_L = 3.0$.

Reference J53 describes a flight evaluation of a booster added to the control system of a B-29 airplane. The F_s/n from this experiment is shown in Figure 12, along with the pilot's evaluation comments. Also shown are the proposed F_s/n requirements for $n_L = 2.7$, which is the appropriate value for the B-29 at normal operating weights. Unfortunately, the pilots' assessments were overall assessments of the acceptability of a given boost ratio throughout the speed range. Since F_s/n changed with speed for a given boost ratio, the pilots' comments apply to a range of F_s/n , rather than a specific value. The ranges are small enough, however, that the data shown in Figure 12 clearly indicate the validity of the requirements. The boost ratio called best by the pilots results in values of F_s/n all well within the Level 1 boundaries. The boost ratio of 8.2 which is undesirably light has values of F_s/n which all lie below the Level 2 limits. The pilot comments for no boost and for a boost ratio of 4.6 are not very specific, but seem to generally fit the proposed boundaries.

Other Considerations

There was no attempt made to break down the requirements further according to Flight Phase Category because of the lack of definitive data indicating that such a breakdown is required. It was also reasoned that a number of Category A Flight Phases to be performed by an airplane are at least crudely represented by the value of n_L designed into the structure.

The control-force gradient requirements are written in terms of a linear system. Actual systems, however, exhibit friction, backlash, and preload, scattered throughout the flight control system, that contribute to control feel and breakout force. In assessing both local and average gradients, then, such nonlinearities should be removed from consideration as best possible. The requirement on control-force nonlinearity is directed at spring, gearing and aerodynamic nonlinearities.

Because quantitative limits on the amount of nonlinearity allowable are difficult to determine, the qualitative guidance provided by 3.3.9 of MIL-F-8785 was retained. Also, it was decided that limits established by Figures 2a and 2b should apply to all local gradients, in addition to the average gradient. (Physically the average gradient is within the range of local gradients.) This was done because the pilot opinion data used to establish these limits were based on considerations of small-amplitude precision maneuvers, as well as gross maneuvers. For precision maneuvers, the local gradient, not the average gradient over a large range of load factors, is the important gradient.

Lack of standardization prevents specification of gradients for side-stick controllers. Data can be found in the literature for some specific controller configurations.

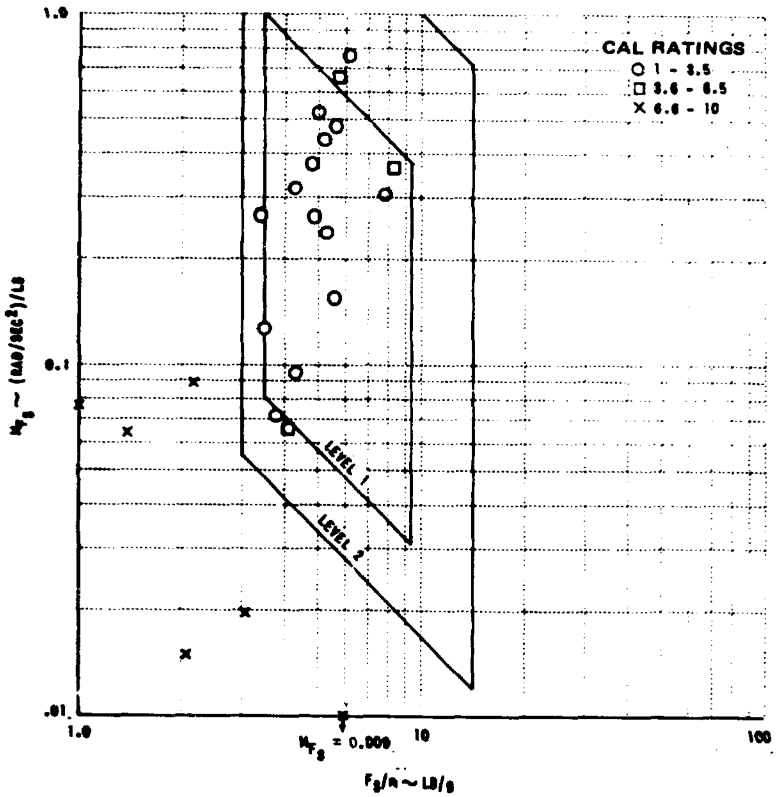


Figure 1 (3.2.2.2.1)
CENTER-STICK CONTROLLER, $n/\alpha = 61.5$
(T-33, USAF PILOT, REFERENCE D3)

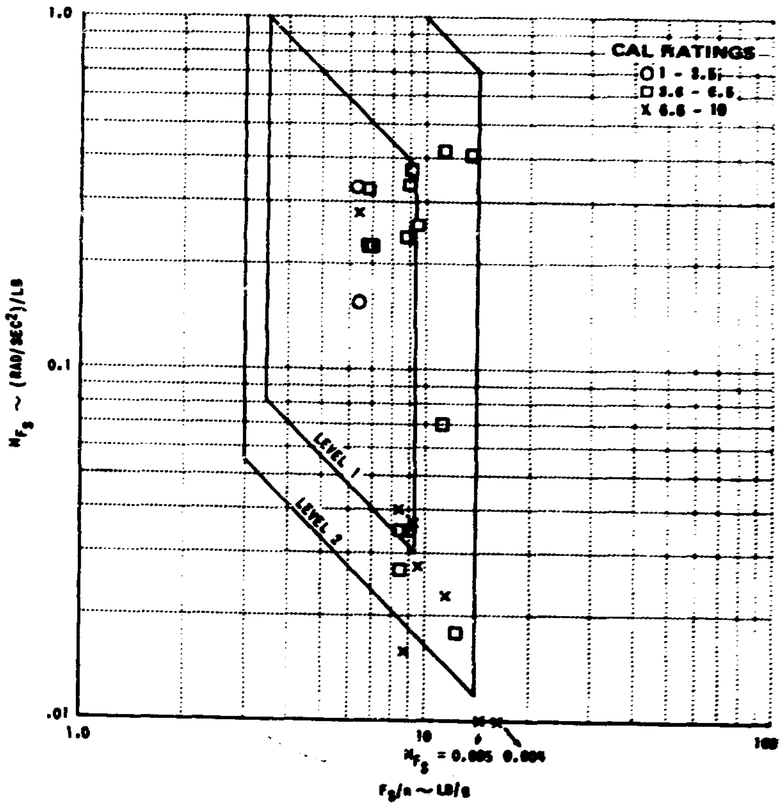


Figure 2 (3.2.2.2.1)
CENTER-STICK CONTROLLER, $n/\alpha = 61.5$
(T-33, CAL PILOT, REFERENCE D3)

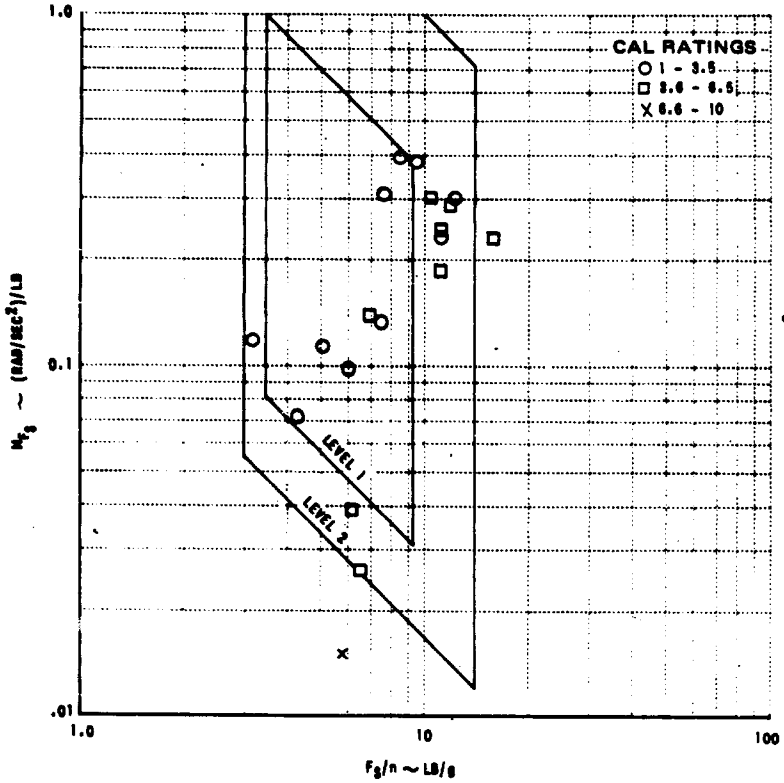


Figure 3 (3.2.2.2.1)
CENTER-STICK CONTROLLER, n/α = 30.1
(T-33, USAF PILOT, REFERENCE D3)

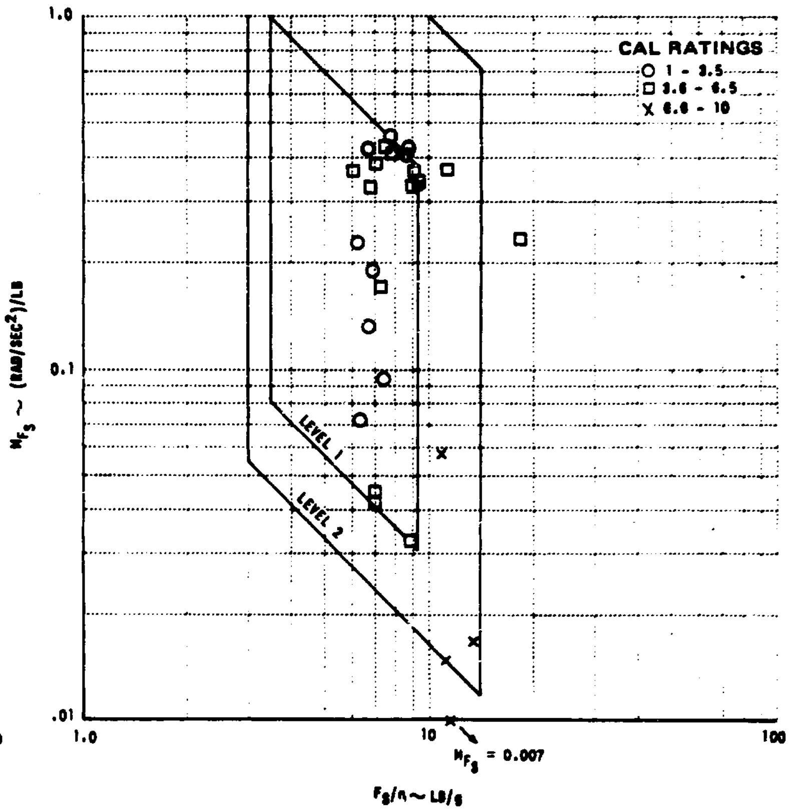


Figure 4 (3.2.2.2.1)
CENTER-STICK CONTROLLER, n/α = 30.1
(T-33, CAL PILOT, REFERENCE D3)

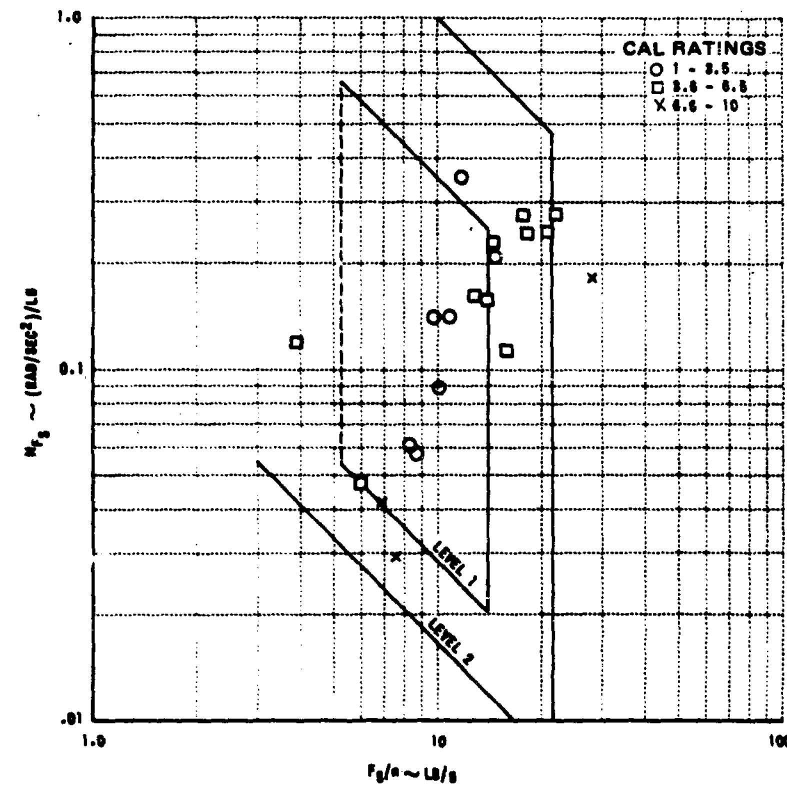


Figure 5 (3.2.2.2.1)
CENTER-STICK CONTROLLER, n/α = 16.9
(T-33, USAF PILOT, REFERENCE D3)

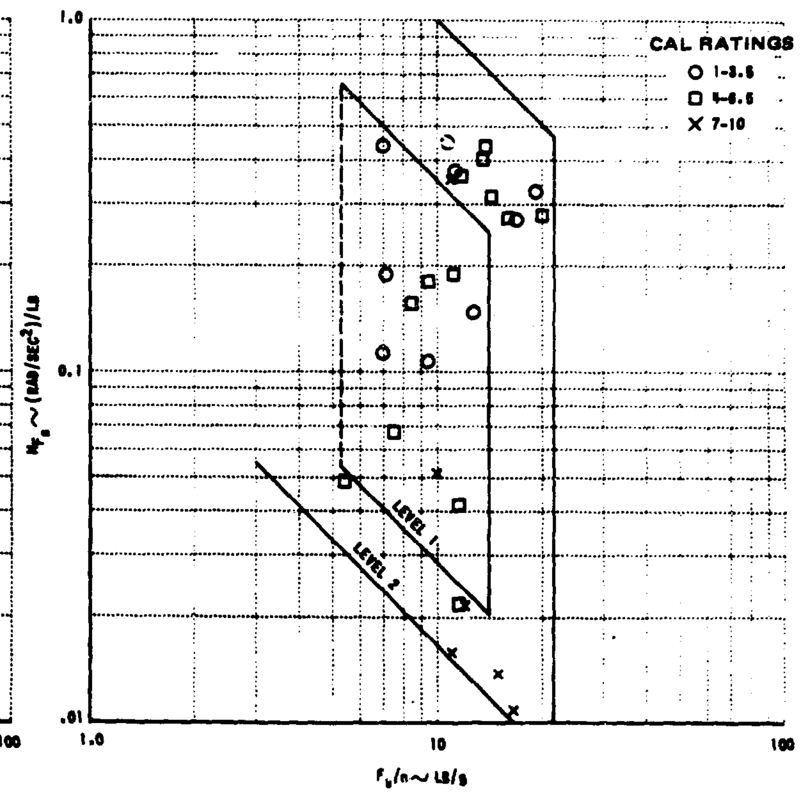


Figure 6 (3.2.2.2.1)
CENTER-STICK CONTROLLER, n/α = 16.9
(T-33, CAL PILOT, REFERENCE D3)

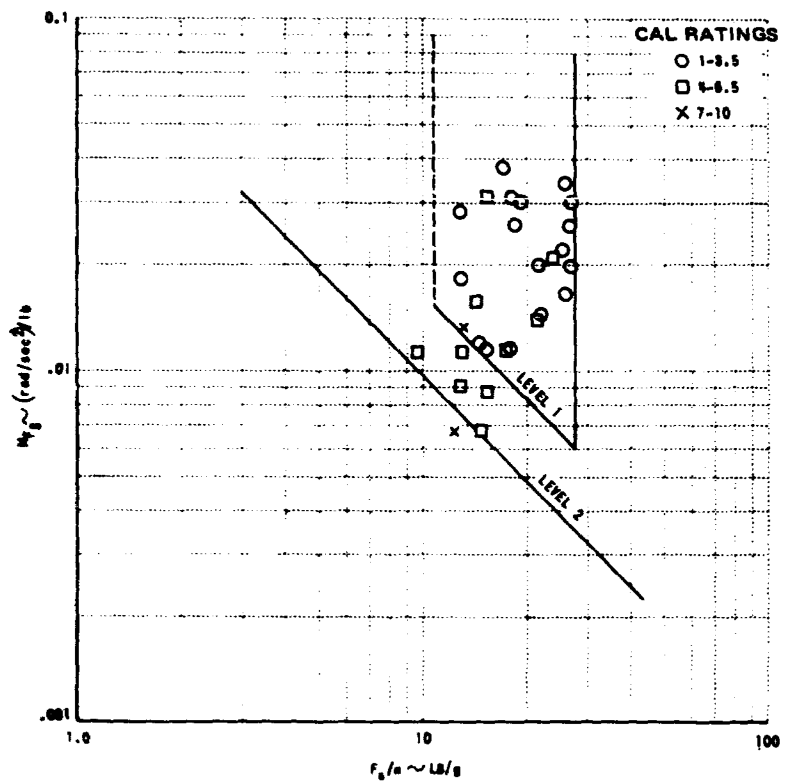


Figure 7 (3.2.2.2.1)

T-33 CENTER-STICK CONTROLLER, $n/\alpha = 8.5$
(T-33, "FRONT-SIDE" AND BOTTOM DATA ONLY,
REFERENCE E12)

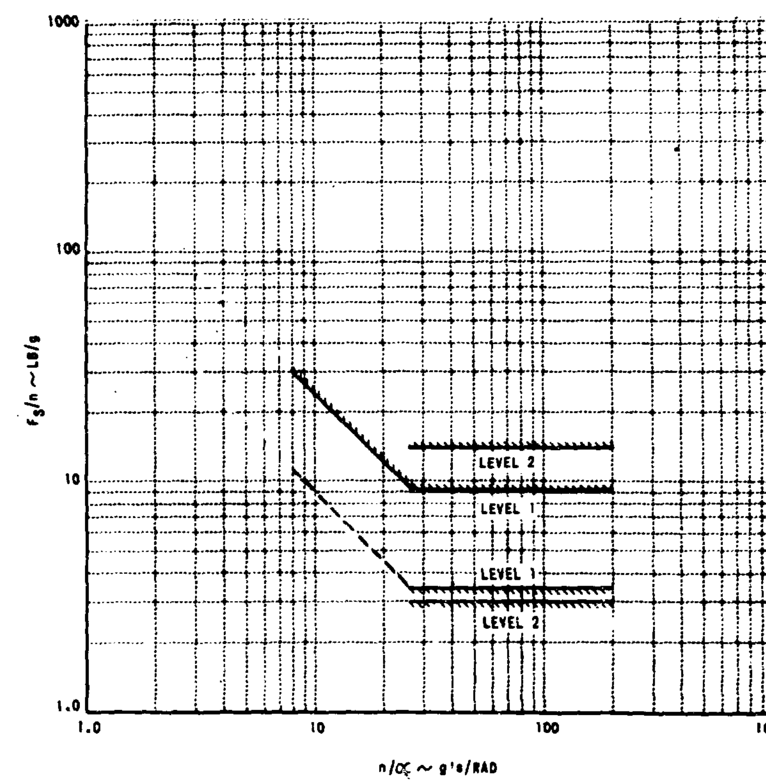


Figure 8 (3.2.2.2.1)

MANEUVERING FORCE GRADIENT DATA SUMMARY
-CENTER -STICK CONTROLLER

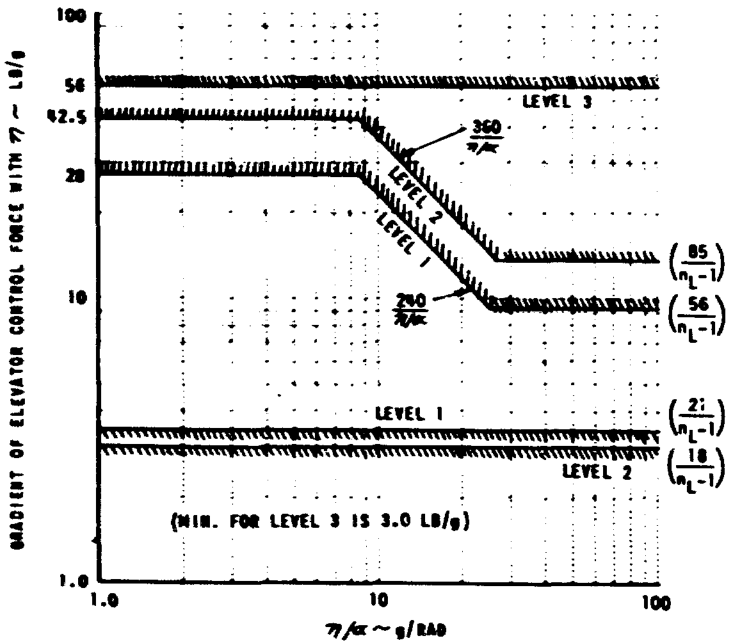


Figure 9 (3.2.2.2.1)
ELEVATOR MANEUVERING FORCE GRADIENT LIMITS:
CENTER-STICK CONTROLLER, n_L = 7.0

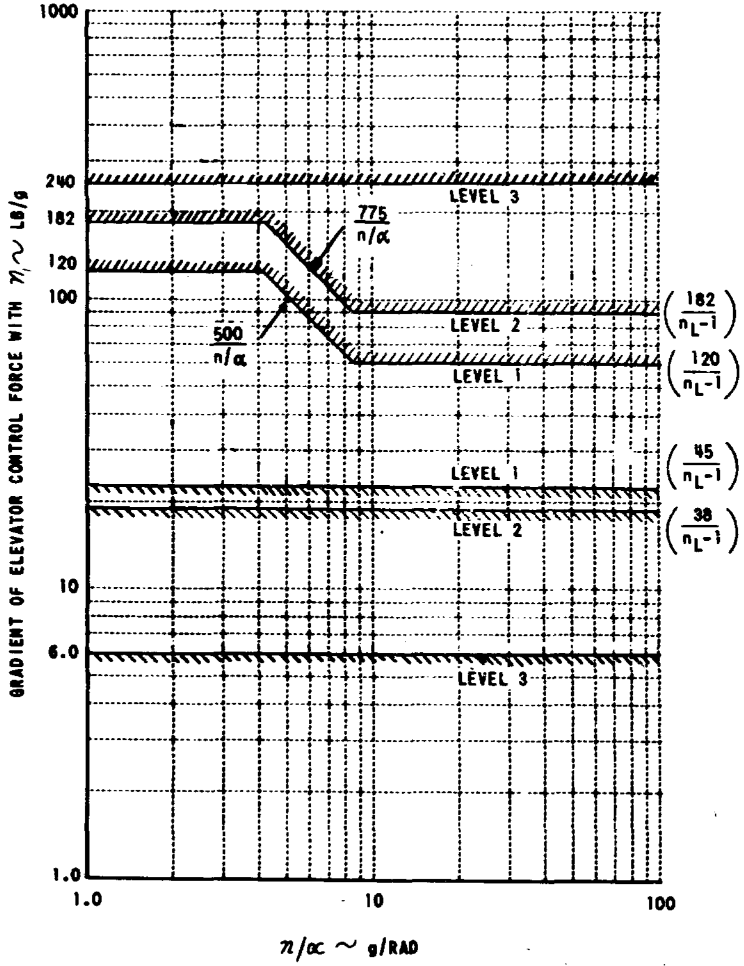


Figure 10 (3.2.2.2.1)
ELEVATOR MANEUVERING FORCE GRADIENT LIMITS:
WHEEL CONTROLLER, n_L = 3.0

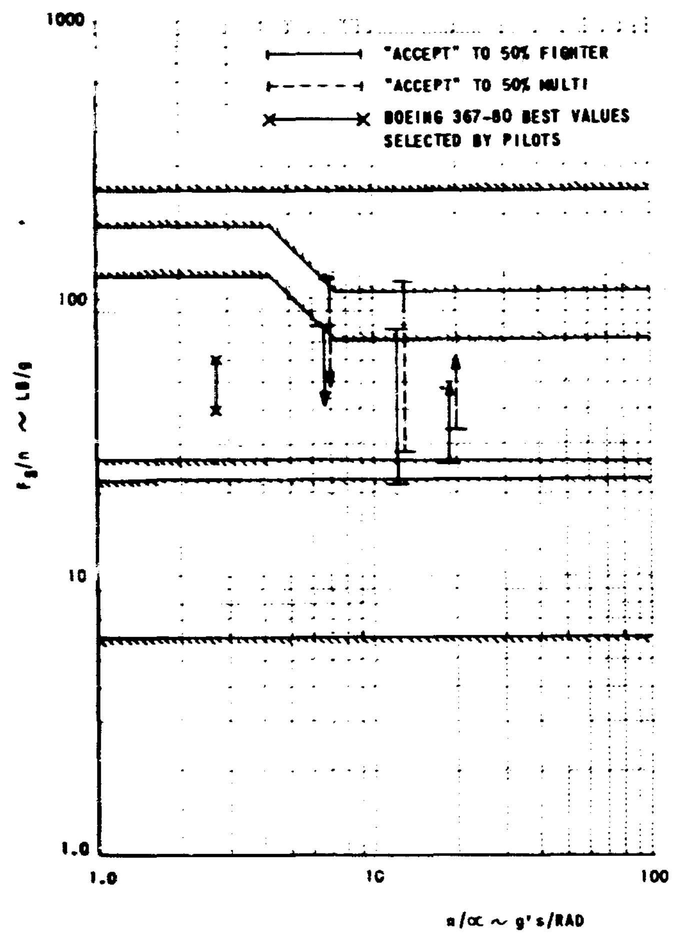


Figure 11 (3.2.2.2.1)
WHEEL CONTROLLER, n_L = 2.7 (B-26, REFERENCE D44)

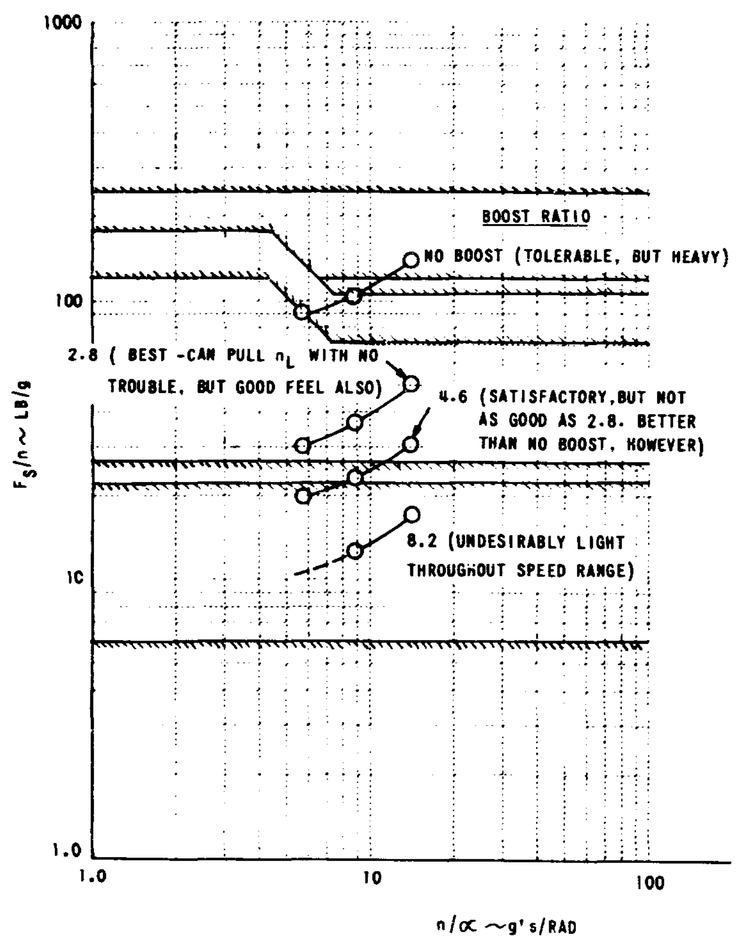


Figure 12 (3.2.2.2.1)
WHEEL CONTROLLER, n_L = 2.7 (B-29, REFERENCE J53)

3.2.2.2.2 CONTROL MOTIONS IN MANEUVERING FLIGHT

REQUIREMENT

3.2.2.2.2 Control motions in maneuvering flight. The elevator-control motions in maneuvering flight shall not be so large or so small as to be objectionable. For Category A Flight Phases, the average gradient of elevator-control force per inch of elevator-control deflection at constant speed shall be not less than 5 pounds per inch for Levels 1 and 2.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

The flying qualities investigations of References D34, D41, D37, D52, C12, and C2 all included variations of control position per g as well as control force per g. References C2 and C12 deal with the landing approach Flight Phase, while all the others are for Category A Flight Phases.

Both References C2 and C12 indicate unfavorable pilot comments when the control motions required to maneuver the airplane become too large. Since these investigations were specific simulations of some C-5A configurations, the short-period natural frequency was below the minimum Level 1 limit for Category C Flight Phases. When the short-period frequency is low, the pilots tend to overdrive the airplane with large pulse-like inputs to speed up the response. Therefore the pilots might not have disliked the control motion gradients as much if the short-period response had been faster. Because of the uncertainties caused by the low short-period frequencies, and because of the limited amount of data, no attempt was made to place quantitative limits on control motion gradients for Category C Flight Phases.

For Category A Flight Phases, however, large control motions appear to be more critical because these flight phases involve rapid maneuvering in which large control motions cause delays in making rapid inputs. For these Flight Phases, References D34, D52, and D37 provide a fair amount of data. Although the data exhibit inconsistencies, the following general conclusions can be made:

1) Since Reference D41 contains data showing that zero control motion is completely acceptable for good values of F_s/n, it does not appear that serious problems result from too little control motion by itself.

2) Since pilot comments from all programs indicate serious problems when the motions become too large, upper limits on the motions appear necessary.

3) The general trend of the data indicates that lines of constant control force per control motion provide the most consistent upper limits on control motions (see Figure 1).

Working under the assumption that there are lower limits on F_s/δ_s (upper limits on δ_s/F_s), the Level 1 and Level 2 boundaries were initially drawn as a best fit to the data of Figure 1. There are not sufficient data to define a Level 3 limit. Although the only data plotted were those having Level 1 values of F_s/n, there are poorly rated configurations from References D34 and D41 which lie inside the Level 1 F_s/δ_s boundary. A likely explanation for this is that the CAL evaluation pilot had a preference for higher values of F_s/n than do most fighter pilots.

Because of the limited data available and because of strong objections from the manufacturers, the Level 1 and 2 limits shown in Figure 1 were reduced to 5 pounds per inch. Examples of "good" operational airplanes were produced that indicate gradients lower than the original lower limit can be acceptable, even desirable.

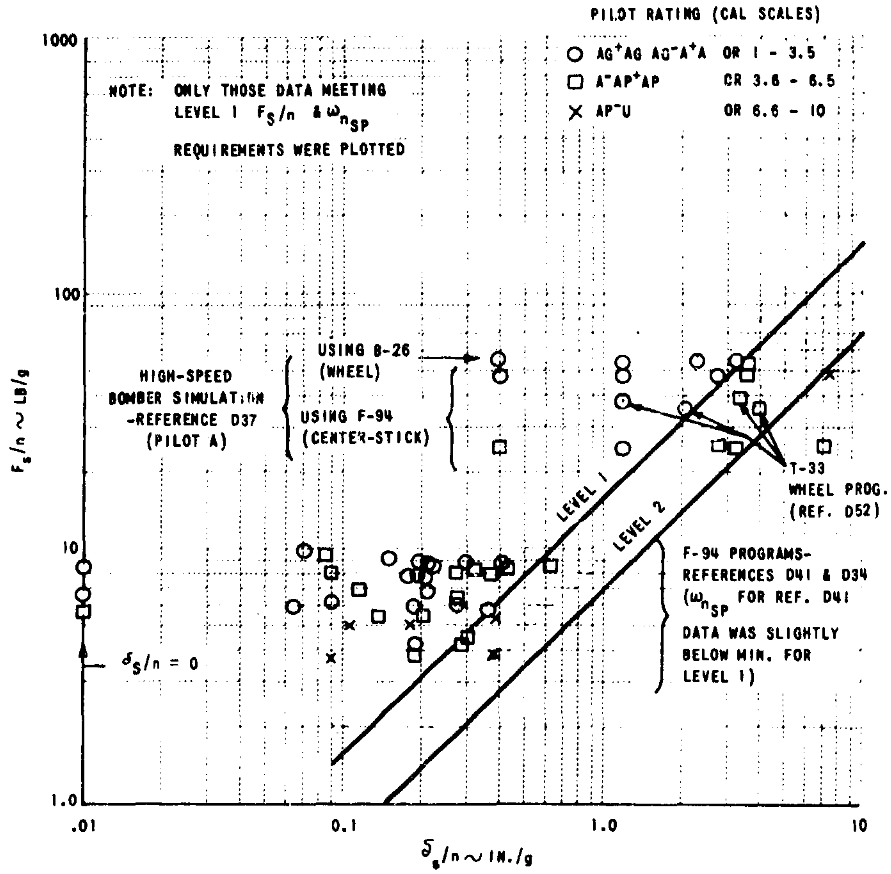


Figure 1 (3.2.2.2.2)
CONTROL FORCE PER CONTROL DISPLACEMENT-CATEGORY A FLIGHT PHASES

3.2.2.3 LONGITUDINAL PILOT-INDUCED OSCILLATIONS

REQUIREMENT

3.2.2.3 Longitudinal 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 airplane.

RELATED MIL-F-8785 PARAGRAPHS

3.3.5.2

DISCUSSION

Introduction

This requirement is an expansion of 3.3.5.2 of MIL-F-8785. It was decided to retain this qualitative requirement because there are many factors determining the susceptibility of a given airplane to longitudinal pilot-induced oscillations (PIO). Some of the known factors are short-period dynamics, control system dynamics, feel system phasing, control force and motion gradients, and control system friction and lost motion. Although it is hoped that most PIO's can be prevented by the requirements in these areas, the problem is not well enough understood at the present time to make that hope a certainty. Therefore, 3.2.2.3 serves as a check list item and establishes the responsibility for correction of PIO problems with the contractor.

Summary of MIL-F-8785B PIO-Related Requirements

Prior to the 1940's, the flight and loading envelopes of most airplanes were limited enough that F_s/n could easily be kept within reasonable limits; and the relatively long tail lengths resulted in good short-period damping throughout most of the flight envelope. In addition, the control systems were low-inertia mechanical systems, deriving restoring and damping forces from aerodynamic hinge moments. The dynamics of the control system were therefore quite fast, and there was little dynamic coupling of the airframe and control system modes. Friction and free play could cause appreciable control system lag for small control inputs, however, resulting in control difficulties during precision maneuvers. Considerable early research effort by the NACA and others was therefore directed at mechanical characteristics such as these. Such factors can still cause difficulties and are therefore limited in Paragraphs 3.5.2.1, 3.5.2.2, and the parts of 3.5.3 and 3.5.3.1 dealing with very small force amplitudes.

As airplane performance increased, it became increasingly difficult to keep maneuvering control forces (F_s/n) within reasonable limits as Mach numbers, altitude and loading changed. In order to minimize these variations in F_s/n, normal acceleration bobweights were often employed - a practice which is still common today. The introduction of bobweights led to some serious controllability problems, primarily attributable to two factors. The first is the tendency of many bobweight designs to cause the stick motion during rapid maneuvers to lead the buildup of stick force. This problem is addressed in Paragraph 3.5.3.1. The second factor is the reduction in stick-free short-period damping ratio which usually accompanies the introduction of bobweights. Lower limits on ζ_{sp} (stick-free) are contained in Paragraph 3.2.2.1.2. These limits are not sufficient, however, when F_s/n is also low. In order to limit problems due to the combined effects of low ζ_{sp} (stick-free) and low F_s/n, the requirements of 3.2.2.3.1 were devised. These two considerations will be discussed in more detail later.

It is important to note that controllability problems due to low ζ_{sp} (stick-free) and low F_s/n are not limited to airplanes employing bobweights. The current trend in airplane design is toward highly augmented airplanes with rather poor short-period damping in the basic airframe, especially at high altitudes. It is therefore possible that future airplanes will have difficulty in meeting the requirements of 3.2.2.1.2 and 3.2.2.3.1 in the event of total SAS failure, even if bobweights are not used.

Another important possible source of PIO problems is a value of $\omega_{n_{sp}}$ which is too high. This effect is limited by the upper boundaries of Paragraph 3.2.2.1.1. The designer should be aware, however, that an airplane which has values of $\omega_{n_{sp}}$ near (but within) the upper limits of 3.2.2.1.1 can still exhibit PIO tendencies if F_s/n is low. He should therefore attempt to keep F_s/n well above the lower limits of 3.2.2.2.1 whenever $\omega_{n_{sp}}$ is high.

Again referring to the trend toward highly augmented airplanes, another source of controllability problems is likely in the future. Complex SAS systems have a tendency to increase the number of dynamic modes involved in the maneuvering response of the airplane, and the in-flight experiments of Reference J59 show that the higher-order dynamics of such systems can cause serious PIO's. The requirements of 3.5.3 are designed to prevent such problems (see the discussion for Paragraph 3.5.3).

The above discussion is by no means a complete list of all the factors which can contribute to PIO's. For additional details, the reader is referred to Sections H and J of the bibliography.

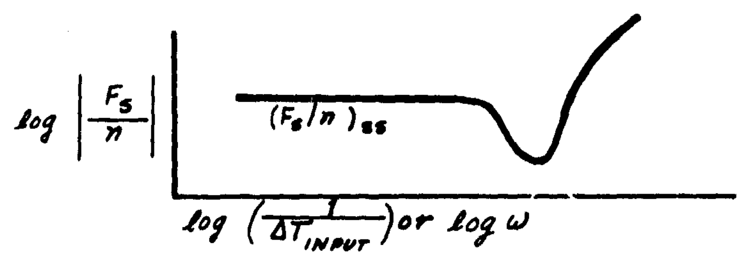
A study of the history of PIO problems shows that, of the possible causes listed above, the use of bobweights has been a major contributor to PIO's. Since bobweights are also likely to be used in the future, it is important that their effects on the dynamics of the airframe and control system be well understood. To this end, the following discussion applies simplified linear descriptions of these dynamics to specific airplanes having bobweight-related PIO problems, from the early 1940's to the present. This discussion is intended to illustrate the major problems which are amenable to analysis at the design stage, and is <u>not</u> intended to be a complete description of any particular PIO problem.

Following these analyses of specific PIO cases, an attempt is made to generalize the results and provide guidance in the design of control systems employing bobweights.

Historical Development of Bobweight-Related PIO Problems

By the late 1930's, airplane performance had increased to the point that it became increasingly difficult to keep control forces within reasonable limits. In particular, F_s/n began to exhibit large changes with altitude and loading. The obvious way to deal with this problem was to replace control-force gradients due to H_{δ_e}, which are sensitive to altitude and loading, with gradients which are insensitive to these factors. This was often accomplished by adding a bobweight and reducing the elevator aerodynamic balance to near zero. This type of control system has caused some rather serious controllability problems. In the early 1940's, the NACA performed analytical and experimental studies aimed at this problem (References J54, J55, J56, J73, J74). In particular, References J54 and J55 studied specific airplanes with control problems. The pilots described their difficulties as resulting from unduly light control forces during rapid control movements and the uncertainty of control in rough air. Obviously, the control characteristics in rough air can be somewhat peculiar when forces proportional to the airplane responses are fed back directly to the stick. However, the control problem seemed to be related primarily to the light forces in rapid maneuvers.

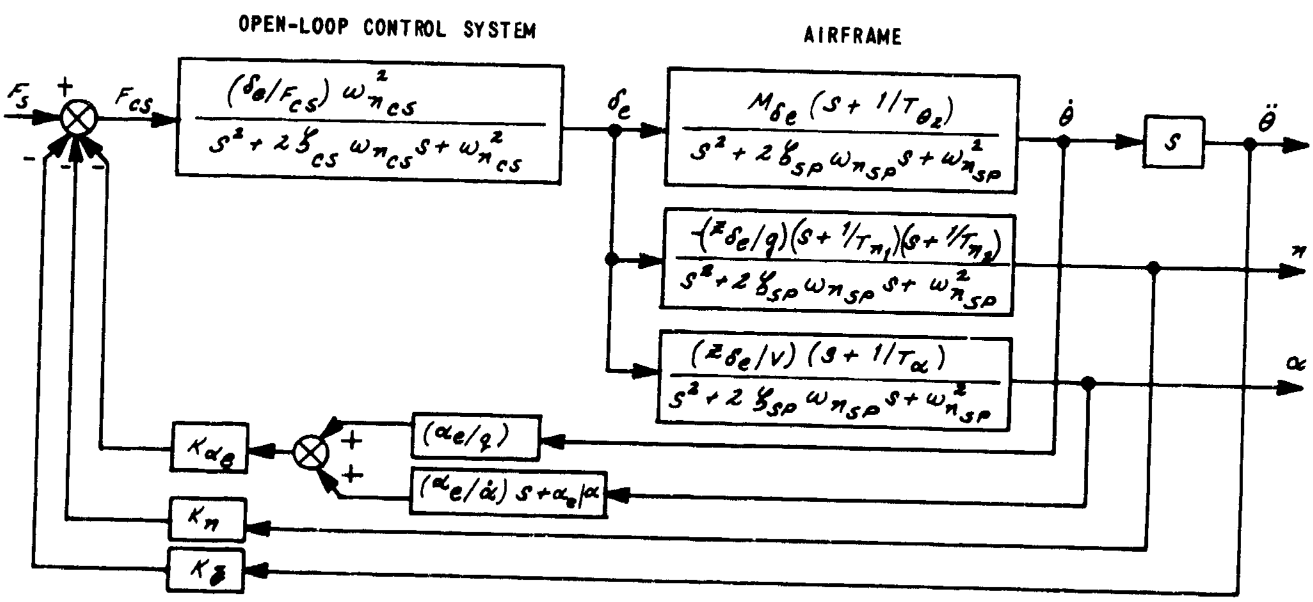
As a result of its studies, the NACA proposed a criterion to deal with the control problem, which eventually became Paragraph 3.3.10 of MIL-F-8785. The criterion required measurement of the ratio of peak stick force to peak normal acceleration during rapid pull-and-return stick movements of various time durations. The requirement was that these ratios never be less than the steady-state value of F_s/n. If these ratios are plotted versus the reciprocal of the input duration time, the upside-down version of a crude n/F_s Bode plot is obtained:



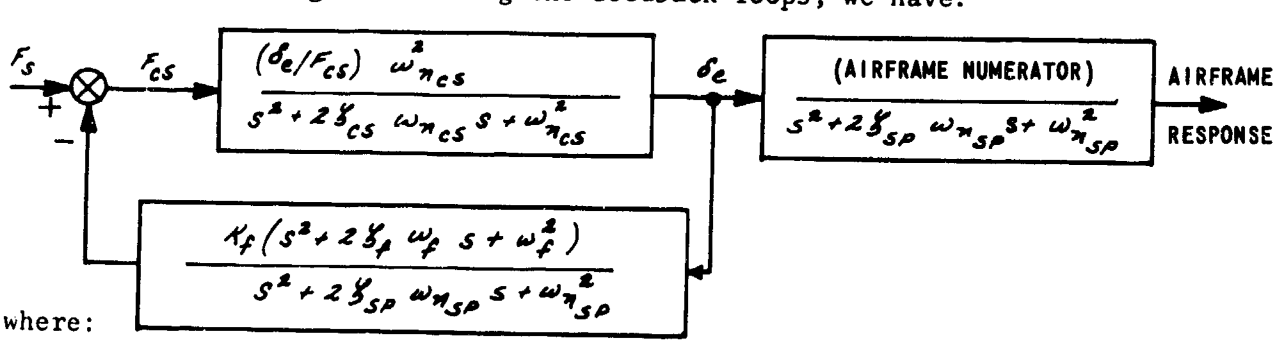
Since the control system natural frequency of these airplanes was significantly greater than $\omega_{n_{sp}}$ (stick-free), the dip in the curve corresponds to the resonance at the stick-free short-period natural frequency; and the ratio of the minimum value to the steady-state value is a measure of the stick-free short-period damping ratio. From all this, it is apparent that the NACA sudden-pullup criterion required ζ_{sp} (stick-free) to be on the order of 0.7, which is unnecessarily high. The data used in the requirements of 3.2.2.1.2, for example, show that it is possible to obtain a very good airplane with $\zeta_{sp} = 0.35$ (stick fixed and free). Thus, it is quite normal and satisfactory to have $|F_s/n|$ at the resonant dip only 65% of steady-state F_s/n. It is obvious, therefore, that the sudden pullup criterion is unduly restrictive and that the effects of ζ_{sp} (stick-free) are more properly treated by the stick-free requirements of Paragraph 3.2.2.1.2. Since most of the undesirable configurations evaluated by the NACA had ζ_{sp} (stick-free) on the order of 0.30-0.35, it would appear that there was much more to the problem than ζ_{sp} (stick-free) alone.

The parameter F_s/n is known to have a strong influence on whether a given value of ζ_{sp} (stick-free) will cause PIO's. A new PIO criterion, which takes into account the influence of F_s/n, is discussed under Paragraph 3.2.2.3.1. This criterion requires that $(F_s/n)_{min}$ (which is $|F_s/n|$ at the resonant dip) be greater than a specified value (see the sketch above). Since several of the undesirable configurations evaluated by the NACA were marginally close to this limit, this criterion reflects part, though not all, of the problem. In order to understand the entire problem, it is first necessary to describe mathematically the dynamics of the airframe and control system.

The dynamic characteristics of airplanes having unpowered control systems can be described by starting with a block diagram representation of the linearized constant-speed dynamics:



The open-loop control system characteristics are determined by the mechanical gearing, the control system inertia, and the hinge moment parameters H_{δ_e} and $H_{\dot{\delta}_e}$. The airframe characteristics are represented by standard constant-speed transfer functions, referenced to the airplane's center of gravity. The K_n and $K_{\ddot{\theta}}$ feedback gains represent the feedbacks caused by bobweights or mass unbalance in the control system. The K_{α_e} feedback gain is caused by the elevator hinge moment parameter H_{α_e}. The parameters (α_e/q), (α_e/α), and $(\alpha_e/\dot{\alpha})$ are included to account for the changes in the local flow angle at the horizontal tail contributed by the pitching motions, the downwash angle, and the downwash lag. Combining the feedback loops, we have:



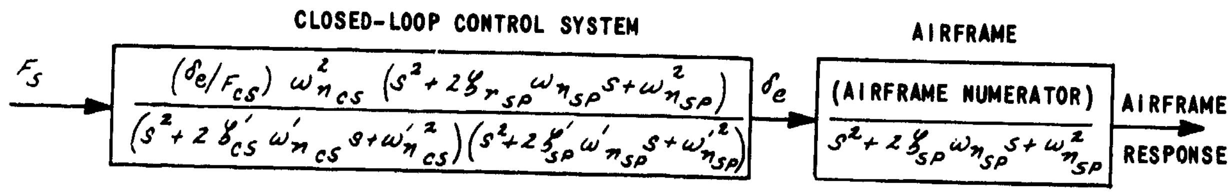
where:

$$K_f = K_{\alpha_e}\left(\frac{Z_{\delta_e}}{V}\;\frac{\alpha_e}{\dot{\alpha}}\right) - K_n\;\frac{Z_{\delta_e}}{g} + K_{\ddot{\theta}}\,M_{\delta_e}$$

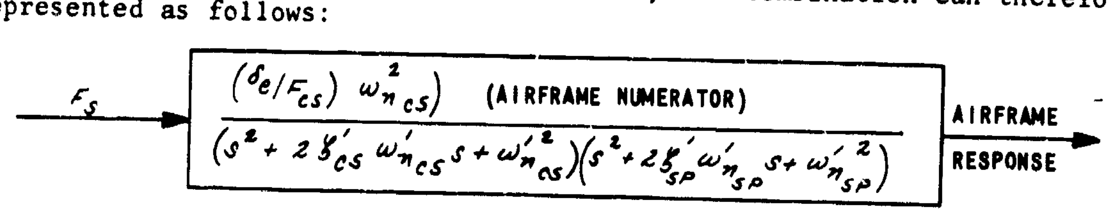
$$(2\zeta_f\omega_f) = \frac{1}{K_f}\left[K_{\alpha_e}\left(M_{\delta_e}\,\frac{\alpha_e}{q} + \frac{Z_{\delta_e}}{V}\,\frac{1}{T_\alpha}\,\frac{\alpha_e}{\dot{\alpha}} + \frac{Z_{\delta_e}}{V}\frac{\alpha_e}{\alpha}\right) - K_n\,\frac{Z_{\delta_e}}{g}\left(\frac{1}{T_{n_1}}+\frac{1}{T_{n_2}}\right) + K_{\ddot{\theta}}\left(M_{\delta_e}\;\frac{1}{T_{\theta_2}}\right)\right]$$

$$\omega_f^2 = \frac{1}{K_f}\left[K_{\alpha_e}\left(\frac{Z_{\delta_e}}{V}\frac{1}{T_\alpha}\frac{\alpha_e}{\alpha} + M_{\delta_e}\frac{1}{T_{\theta_2}}\;\frac{\alpha_e}{q}\right) - K_n\left(\frac{Z_{\delta_e}}{g}\;\frac{1}{T_{n_1}}\;\frac{1}{T_{n_2}}\right)\right]$$

Closing the feedback loop of this block diagram will result in a fourth-order characteristic equation, which normally factors into two complex pairs. The closed-loop dynamics can then be expressed as follows:



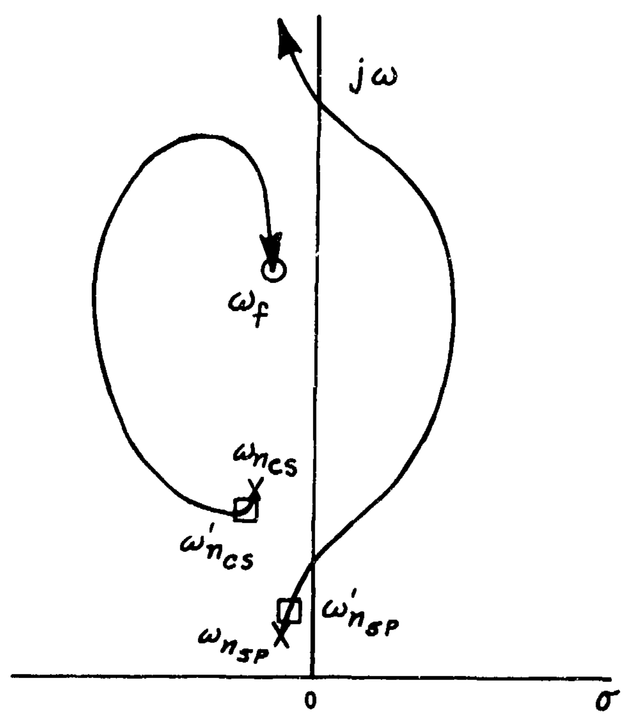
The overall dynamics of the airframe/control system combination can therefore be represented as follows:



Note that the primed modal parameters denote stick-free characteristics and unprimed parameters denote stick-fixed values. Since the responses of both the control system and the airplane to stick force inputs are composed of two second-order modes, it is rather arbitrary which closed-loop mode is called the stick-free short-period mode. Since none of the airframe zeros are likely to be near either mode, however, the mode having the lower natural frequency will probably dominate the airplane's response to stick force inputs.

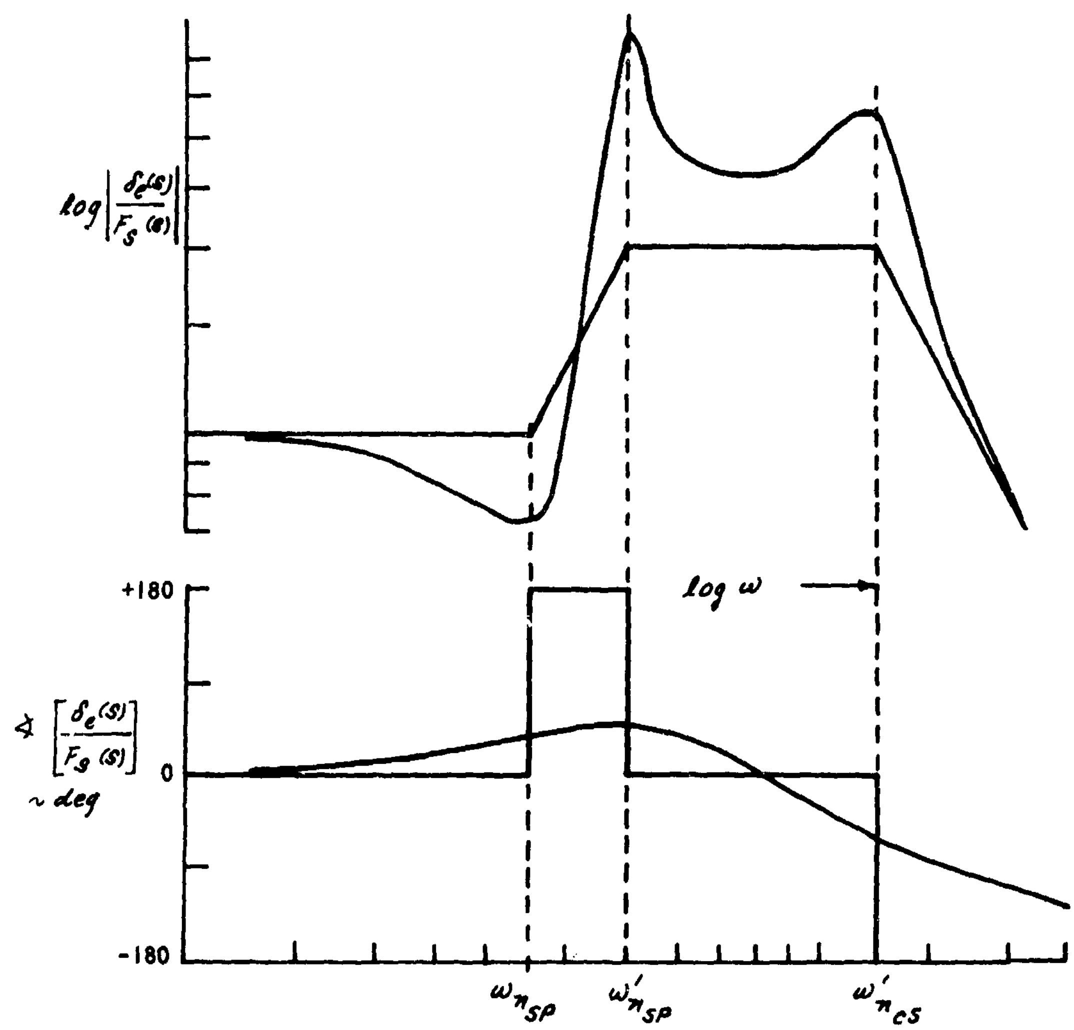
By applying these mathematical descriptions to a specific airplane which the NACA studied, such as the P-63A-1 described in Reference J54, further insight into the control-system problems of the 1940's can be gained.

The basic control system of the P-63A-1 described in Reference J54 consisted of a 3.7 lb/g bobweight in conjunction with closely balanced experimental elevators. The airplane showed relatively small variations in steady-state F_s/n with altitude and loading, but pilots complained about the light forces associated with rapid control movements. There are not sufficient data in Reference J54 to determine accurately the numerical values of the dynamic parameters discussed above. However, by carefully studying what data are available and making some approximations, a rough picture of the dynamics can be obtained. The following root-locus plot shows the relative locations of the various poles and zeros of the $\delta_e(S)/F_s(s)$ transfer function as the root-locus gain $(K_f \frac{\delta_e}{F_{cs}} \omega^2_{n_{cs}})$ is varied. (Note that the inverse of $\delta_e/F_{cs}(\omega^2_{n_{cs}})$ is equivalent to the control system inertia; so that for a fixed inertia, the root-locus gain is a constant times K_f, which is primarily composed of contributions from K_n and $K_{\ddot{\theta}}$:)



Thus it is seen that the location of the feedback zeros causes $\omega'_{n_{sp}}$ to increase and ζ'_{sp} to decrease, as the root-locus gain is increased. (In practice, changes in K_{η} and $K_{\ddot{\theta}}$ which alter the root-locus gain will also change the control system inertia, so that $\omega_{n_{cs}}$ will vary somewhat with the root-locus gain. For purposes of the following discussions, however, it is acceptable to assume $\omega_{n_{cs}}$ remains fixed as the loop gain is varied.) The influence of reduced ζ'_{sp} has already been discussed. However, some interesting consequences result from the fact that $\omega'_{n_{sp}}$ and $\omega'_{n_{cs}}$ are both appreciably greater than $\omega_{n_{sp}}$. Referring to the transfer function of the closed-loop control system, $\frac{\delta_e(S)}{F_S(S)}$, the following Bode plot is obtained:

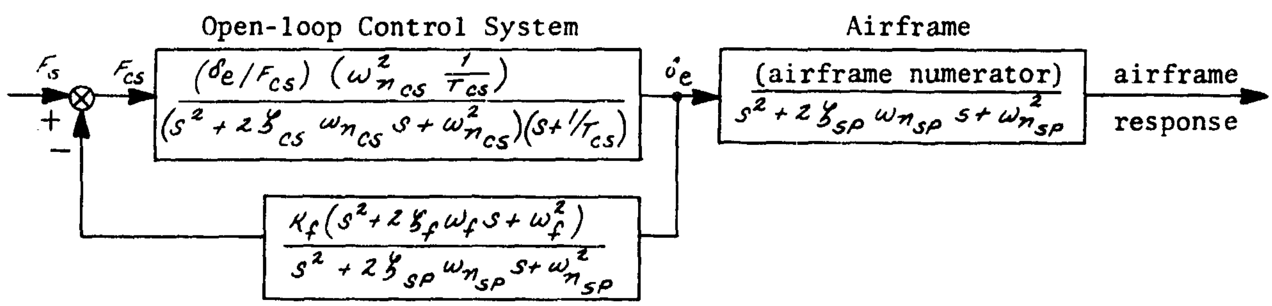
$$\frac{\delta_e(s)}{F_s(s)} = \left(\frac{\delta_e}{F_{cs}}\,\omega'^{2}_{n_{cs}}\right)\frac{\left(s^2 + 2\zeta_{sp}\,\omega_{n_{sp}}\,s + \omega_{n_{sp}}^2\right)}{\left(s^2 + 2\zeta'_{cs}\,\omega'_{n_{cs}}s + \omega'^{2}_{n_{cs}}\right)\left(s^2 + 2\zeta'_{sp}\,\omega'_{n_{sp}}s + \omega'^{2}_{n_{sp}}\right)}$$



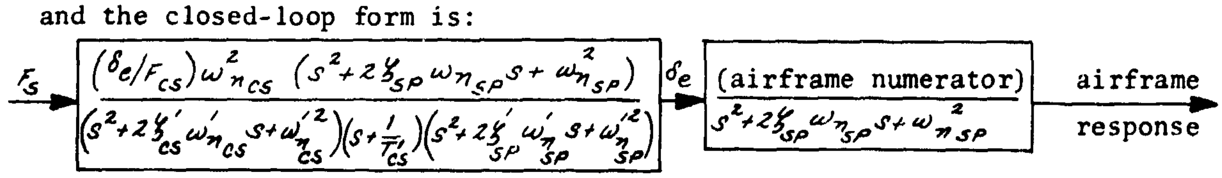
The fact that $\left|\frac{\delta_e(s)}{F_s(s)}\right|$ is higher at high frequencies than at low frequencies is believed to be a major source of the pilots' comments concerning light stick forces during rapid stick movements. Some of this increased amplitude is due to the resonance caused by low ζ'_{sp}, but this aspect is limited by 3.2.2.3.1. However, the effects of low ζ'_{sp} are further aggravated by the fact that the asymptotic Bode plot shows increased $\left|\frac{\delta_e(s)}{F_s(s)}\right|$ whenever $\omega'_{n_{sp}}$ and $\omega'_{n_{cs}}$ are appreciably greater than $\omega_{n_{sp}}$. If this last effect is signifiant, $\frac{\delta_e(s)}{F_s(s)}$ will exhibit phase lead at intermediate frequencies, as shown above. By limiting this phase lead in Paragraph 3.5.3.1, it is thought that problems due to light forces during rapid control movements can be minimized.

In summary then, the controllability problem of the P-63A-1 was partially due to the low $(F_s/n)_{min}$ caused by low ζ'_{sp} and low F_s/n, but appeared to be mostly due to peculiar feel-system characteristics. When the pilot attempted a rapid maneuver, the initial stick movement was accompanied by very light forces. As the airplane response developed, the pilot had to increase his applied force in order to hold the initial elevator input. With normal feel-system phasing (i.e., position in phase with, or lagging the force), the pilot is probably not too concerned with stick position, at least in up-and-away flight. But the peculiar phasing of the P-63A-1 feel system (i.e., position leading force) apparently made it difficult for the pilot to determine the proper elevator inputs needed to control the airplane.

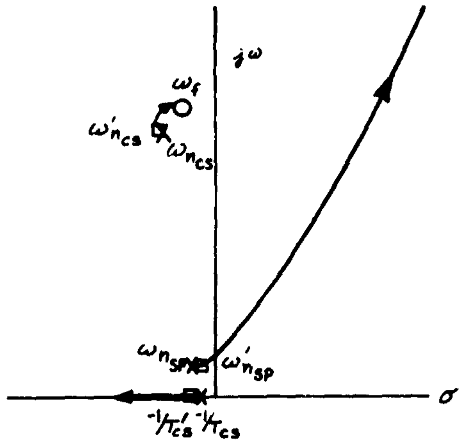
It is interesting to note that the controllability of the P-63A-1 was significantly improved by introducing a mechanical feel device into the control system. The effect of this device was to increase $\omega_{n_{cs}}$ and introduce a new low-frequency first-order mode. The block diagram of this system is as follows:



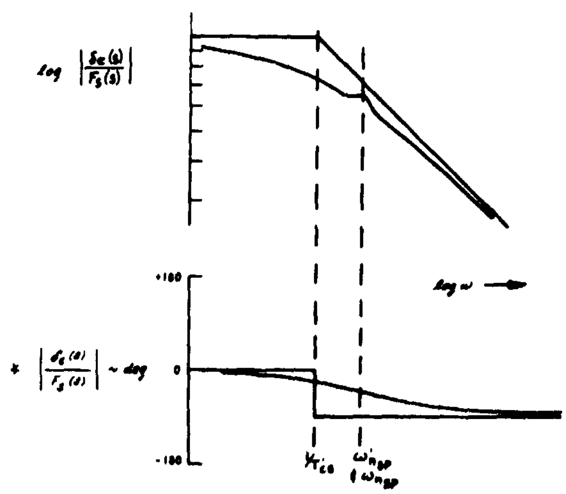
and the closed-loop form is:



From the data available in Reference J54, it is possible to determine that the root locus of $\delta_e(s)/F_s(s)$ has the following form:



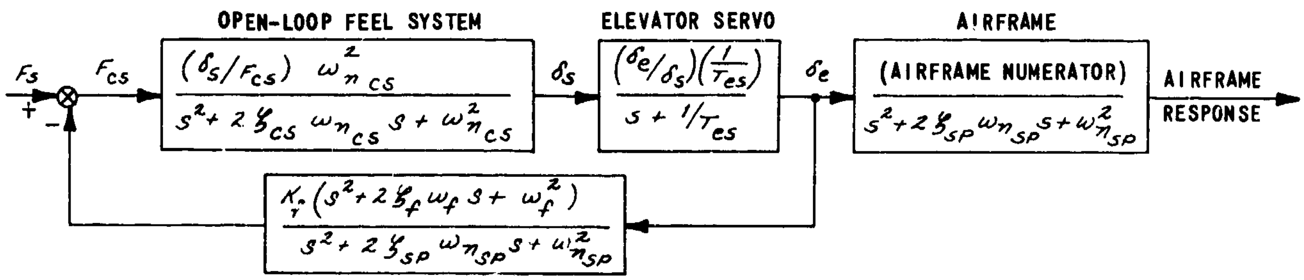
The Bode plot of the closed-loop $\frac{\delta_e(S)}{F_s(S)}$ transfer function is as follows:



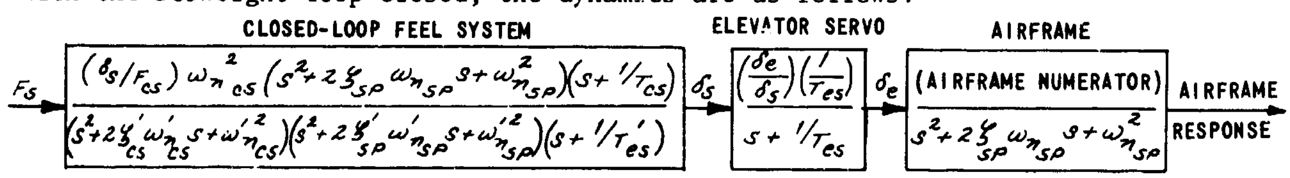
Introduction of the feel device did not significantly improve ζ'_{sp}, yet the pilots reported that the controllability of the airplane was greatly improved. The major difference between the original control system and the modified system is that the modified system exhibits no amplitude magnification or phase lead in $\frac{\delta_e(S)}{F_s(S)}$ at frequencies above $\omega_{n_{sp}}$.

By the early 1950's, fully powered (irreversible) control systems were coming into use on production airplanes. Typically, these systems employ bobweights to keep F_s/n within reasonable limits with changing loading and flight condition. For the unpowered control systems of the 1940's, F_s/n was fairly constant with speed. For powered systems, however, F_s/n tends to decrease rapidly with increasing equivalent airspeed because of the contribution of the feel springs. Thus, the low ζ'_{sp} due to the bobweight, combined with low F_s/n , makes the high dynamic-pressure flight regimes particularly PIO-prone. For most high-performance aircraft, the highest dynamic pressures are encountered at high speeds and low altitudes. Several high-performance aircraft of the 1950's have experienced catastrophic PIO's in this flight regime.

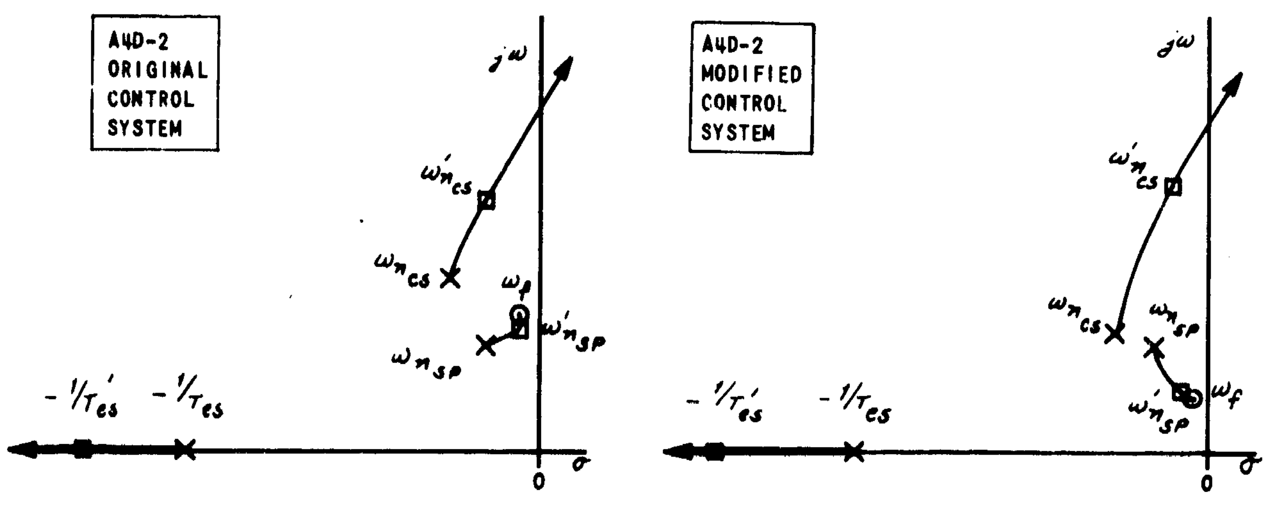
The dynamic characteristics of an airplane with a fully powered control system, bobweight, and feel spring, are quite similar to those of an unpowered system, except for the lack of any α_e feedback and the addition of the elevator servo dynamics (introduction of the servo means that the stick and elevator no longer have the same dynamics).



With the bobweight loop closed, the dynamics are as follows:

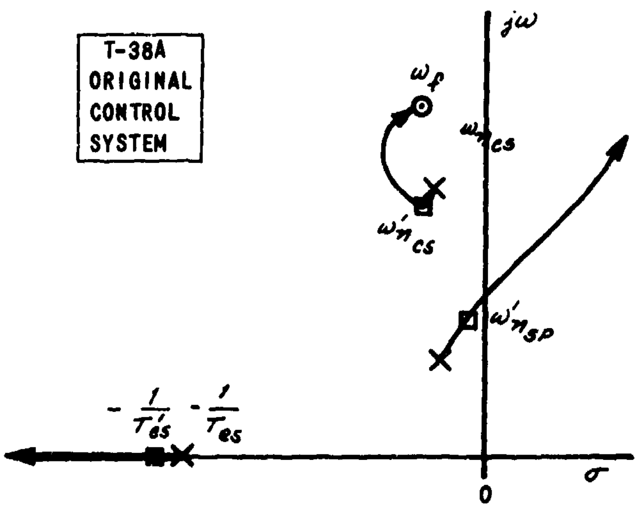


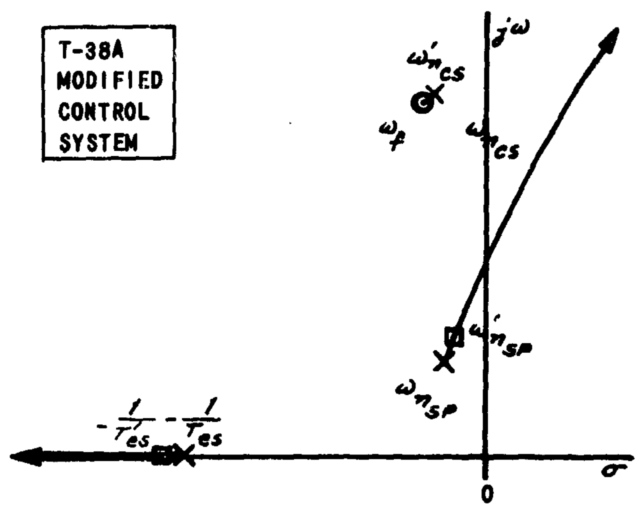
One rather well-documented example of a control system of this type which caused PIO's was the A4D-2 airplane (Reference H11). In the high-speed, low-altitude regime, the dynamic characteristics were of the following form:



The original configuration has very low ζ'_{sp} and low $(F_s/n)_{min}$. Because $\omega_{n_{sp}}$ is less than $\omega'_{n_{sp}}$, there is also appreciable phase lead in $\delta_s(s)/F_s(s)$. Note that because the total damping of all the roots must remain constant, the presence of the elevator servo causes the damping of the control system and short-period roots to decrease much faster than was possible with the unpowered system. The PIO problem was virtually eliminated by modifying the control system to increase $K_{\ddot{\theta}}$ and decrease K_n. This resulted in a smaller decrease in ζ'_{sp} and in $\omega'_{n_{sp}}$ being less than $\omega_{n_{sp}}$, so that there was no phase lead in $\delta_s(s)/F_s(s)$.

In the early 1960's, the T-38A airplane was introduced into service. This airplane also had PIO problems in the low-altitude, high-speed regime with pitch damper inoperative and a concentrated campaign was conducted to modify the control system (References H5 and H7). The dynamic characteristics of the airplane (damper off) in the critical flight regime are as shown below:





As with the A4D-2, the original T-38A control system had very low ζ'_{sp} and $(F_s/n)_{min}$, as well as considerable phase lead in $\delta_s(s)/F_s(s)$. In the case of the T-38A, however, the size of the bobweight was simply reduced ($K_{\ddot{\theta}}$ was reduced in proportion to K_n). Thus the poles and zeros of the modified system are the same as those of the original system, except that $\omega_{n_{cs}}$ is increased due to the reduced control system inertia and increased feel spring size. Thus, no fundamental changes in the nature of the control system dynamics were made, but the severity of the problem was considerably reduced by reducing the loop gain.

In the late 1950's, the F-4 airplane was introduced into service. Like the A4D-2 and T-38A this airplane had serious PIO problems at high speeds and low altitudes, with the pitch damper off. Until recently, the problem has been avoided by warning pilots to avoid the high-speed low-altitude regime with dampers off. Recently the Air Force evaluated a proposal modification to the control system (Reference P2), which reduced the PIO tendencies. The dynamics of the airplane with the original control system are estimated to be similar to those of the A4D-2 with its original system, except that $\omega_{n_{c.s.}}$ is higher for the F-4. Unlike the A4D-2, however, the system of the F-4 was modified by simply reducing the size of the bobweight, without moving the bobweight zeros.

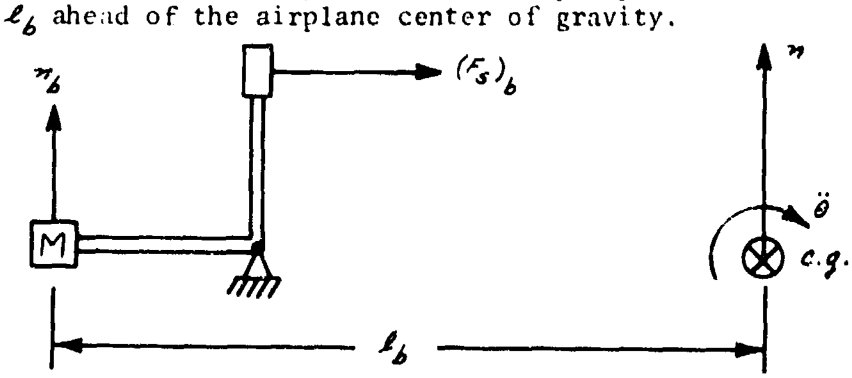
Design Options (Fully Powered Control Systems)

From the above historical development, it would appear that the use of bobweights is bound to create problems. The most obvious way to avoid these problems is to avoid using bobweights entirely, and minimize variations in F_s/n by using such devices as dynamic-pressure-sensitive feel springs or gain scheduling. Of course, there are practical problems associated with the use of these devices also, so that bobweights will probably continue to be employed in many cases. While it is true that there are potential problems associated with the use of any device which causes the stick-free dynamics to differ appreciably from the stick-fixed dynamics, careful design can minimize the problems. The following paragraphs give some guidance in the design of control systems employing bobweights.

The above examples of airplanes having PIO problems due to bobweight have made it obvious that the locations of the feedback zeros are of paramount importance. Therefore, it will be useful to examine the previously desired expression for these zeros. Assuming a fully powered control system ($K_{\alpha_e} = 0$), the expression for $(s^2 + 2\zeta_f \omega_f s + \omega_f^2)$ reduces to:

$$s^2 + \left[\frac{K_{\ddot{\theta}}\left(M_{\delta_e}\,{}^{1}\!/\!_{T_{\theta_2}}\right) - K_n \frac{Z_{\delta_e}}{g}\left(\frac{1}{T_{n_1}} + \frac{1}{T_{n_2}}\right)}{K_{\ddot{\theta}}\, M_{\delta_e} - K_n\left(\frac{Z_{\delta_e}}{g}\right)}\right] s - \left[\frac{K_n\left(\frac{Z_{\delta_e}}{g}\,\frac{1}{T_{n_1}}\,\frac{1}{T_{n_2}}\right)}{K_{\ddot{\theta}}\, M_{\delta_e} - K_n\left(\frac{Z_{\delta_e}}{g}\right)}\right]$$

(Note that this expression is similar to, but more general than, the expression developed in Appendix I of Reference J60.) The effects of $K_{\ddot{\theta}}$ can be accounted for by defining a length $\ell_b = g(K_{\ddot{\theta}}/K_n)$. Physically, this is equivalent to replacing the entire control system with a simple point-mass bobweight located a distance ℓ_b ahead of the airplane center of gravity.



$$(F_s)_b = K_n n_b$$

$$n_b = n + \frac{\ell_b \ddot{\theta}}{g}$$

$$(F_s)_b = K_n n + \underbrace{\left(K_n \frac{\ell_b}{g}\right)}_{K_{\ddot{\theta}}} \ddot{\theta}$$

In addition, a distance ℓ_{cp} can be defined as the distance of the center of percussion ahead of the c.g. The center of percussion is the location in the airplane where the initial normal-acceleration response to a step control input is zero. The initial normal-accleration response a distance ℓ ahead of the c.g. can be expressed as follows for a step input:

$$n_{initial} = (n_{c.g})_{initial} + \frac{\ell \ddot{\theta}_{initial}}{g}$$

$$= \lim_{s \to \infty} \left[s \frac{\delta_e}{s} \left(\frac{n_{cg}(s)}{\delta_e(s)} + \frac{\ell}{g} \frac{\ddot{\theta}(s)}{\delta_e(s)} \right) \right]$$

$$= \delta_e \left(- \frac{z_{\delta_e}}{g} + \frac{\ell}{g} M_{\delta_e} \right)$$

Now, from the above definition of center of percussion, n initial is zero when $\ell = \ell_{cp}$:

$$n_{initial} = 0 = \delta_e \left(- \frac{z_{\delta_e}}{g} + \frac{\ell_{cp}}{g} M_{\delta_e} \right)$$

$$\ell_{cp} = \frac{z_{\delta_e}}{M_{\delta_e}} \quad \text{(positive for aft-tailed airplanes)}$$

In addition to the above expressions for ℓ_b and ℓ_{cp}, it is known that $\left| \frac{1}{T_{n_1}} + \frac{1}{T_{n_2}} \right| << \left| \frac{1}{T_{n_1}} \frac{1}{T_{n_2}} \right|$, so that the following assumption can be made:

$$\frac{z_{\delta_e}}{g} \left(\frac{1}{T_{n_1}} + \frac{1}{T_{n_2}} \right) \approx 0$$

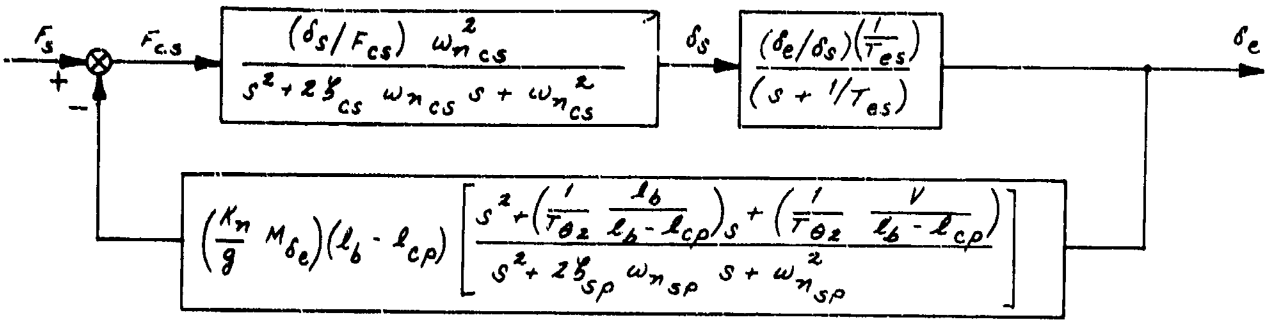
Also it can be shown readily that

$$\frac{z_{\delta_e}}{g} \frac{1}{T_{n_1}} \frac{1}{T_{n_2}} = - \frac{V}{g} \frac{1}{T_{\theta_2}} M_{\delta_e}$$

Finally, the expression to be solved for the feedback zeros reduces to:

$$s^2 + \left[\frac{1}{T_{\theta_2}} \left(\frac{\ell_b}{\ell_b - \ell_{cp}} \right) \right] s + \left[\frac{1}{T_{\theta_2}} \left(\frac{V}{\ell_b - \ell_{cp}} \right) \right] = 0$$

The previously derived block-diagram representation of a fully powered control system can now be simplified to the following form:



Thus for a given airframe, flight condition, set of control system dynamics, and ℓ_b , the closed-loop (i.e., stick-free) roots describe a locus on the complex plane which is determined by the open-loop poles and zeros shown in the above diagram. The locations of the roots along the locus are determined by the root-locus gain, which is:

$$\left(\frac{\delta_s}{F_{cs}}\right)\;\omega_{n_{cs}}^2\;\left(\frac{\delta_e}{\delta_s}\right)\;\left(\frac{1}{T_{es}}\right)\left(\frac{K_n}{g}M_{\delta_e}\right)(\ell_b-\ell_{cp})$$

The steady-state stick force per g can be expressed as the sum of feel-spring (f_s) and bobweight (b) components:

$$\frac{F_s}{n}=\left(\frac{F_s}{n}\right)_{f_s}+\left(\frac{F_s}{n}\right)_b=\frac{1}{(\delta_s/F_{cs})(\delta_e/\delta_s)\left(\frac{n}{\delta_e}\right)_{ss}}+K_n$$

Since

$$\left(\frac{n}{\delta_e}\right)_{ss}=-\frac{Z_{\delta_e}}{g\,\omega_{n_{sp}}^2}\;\frac{1}{T_{n_1}}\;\frac{1}{T_{n_2}}=\frac{V}{g}\;\frac{1}{T_{\theta_2}}\;\frac{M_{\delta_e}}{\omega_{n_{sp}}^2}$$

the expression for the feel system component of F_s/n can be written

$$\left(\frac{F_s}{n}\right)_{f_s}=\frac{\omega_{n_{sp}}^2}{(\delta_s/F_{cs})(\delta_e/\delta_s)\,M_{\delta_e}\,\frac{V}{g}\,\frac{1}{T_{\theta_2}}}$$

from which

$$\left(\frac{\delta_s}{F_{cs}}\right)\left(\frac{\delta_e}{\delta_s}\right)M_{\delta_e}=\frac{\omega_{n_{sp}}^2}{\frac{V}{g}\;\frac{1}{T_{\theta_2}}(F_s/n)_{f_s}}$$

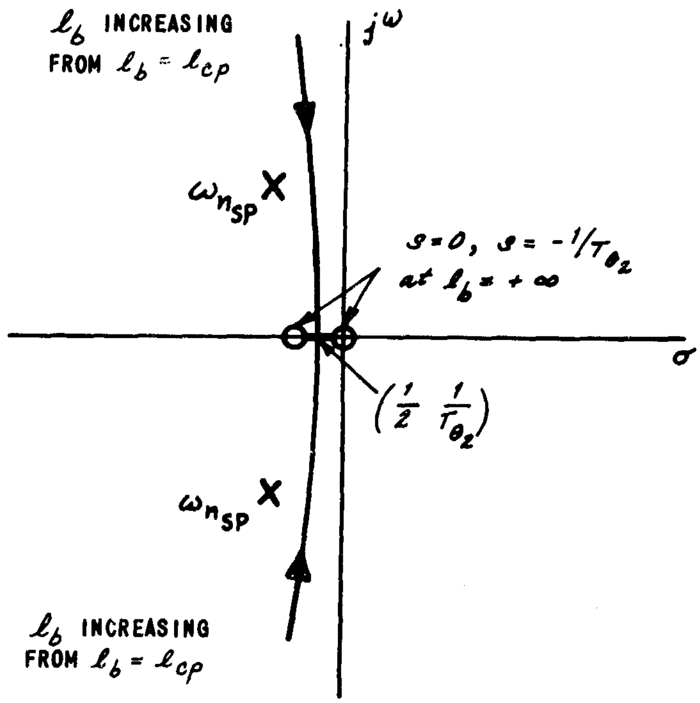
Finally the root-locus gain can be expressed as

$$\frac{(F_s/n)_b}{(F_s/n)_{f_s}}\;\left(T_{\theta_2}\;\frac{\ell_b-\ell_{cp}}{V}\right)\left(\omega_{n_{cs}}^2\;\omega_{n_{sp}}^2\;\frac{1}{T_{es}}\right)$$

Thus the locations of the stick-free roots along the locus are determined by the ratio of the contributions of the bobweight and the feel spring to F_s/n .

In order to minimize the variations of F_s/n with flight condition and loading, it is desirable to make $(F_s/n)_b/(F_s/n)_{fs}$ fairly high. Therefore, it is desirable to arrange the open-loop control-system poles and zeros to obtain desirable closed-loop root locations over as much of the locus as possible. To this end, it is interesting to sketch the locations of the feedback zeros for various values of ℓ_b :

$\ell_b > \ell_{cp}$ (Equivalent point-mass bobweight ahead of center of percussion)



It can be seen that the zeros are normally complex for $\ell_b > \ell_{cp}$. From the previously derived expressions for the feedback numerator quadratic,

$$\left(s^2 + 2\zeta_f\,\omega_f\,s + \omega_f^2\right)$$

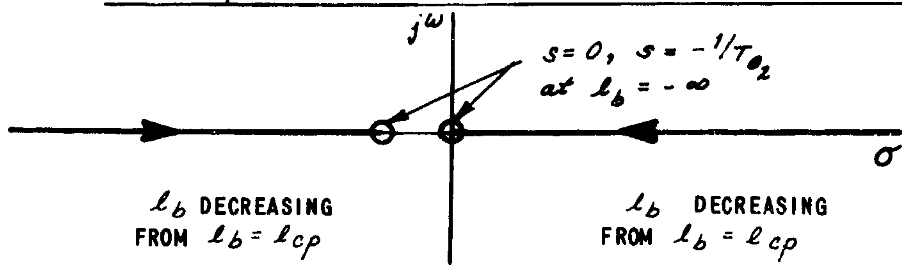
there results

$$\omega_f = \sqrt{\frac{1}{T_{\theta_2}}\ \frac{V}{\ell_b - \ell_{cp}}}$$

$$\zeta_f\,\omega_f = \frac{1}{2}\left(\frac{1}{T_{\theta_2}}\right)\left(\frac{\ell_b}{\ell_b - \ell_{cp}}\right)$$

The shape of the path which the zeros follow as ℓ_b varies is a function of V and ℓ_{cp} . If the airplane has a very large tail length $(\ell_{cp} \approx 0)$, the path of the zeros is almost a vertical line $\left(\zeta_f\,\omega_f \approx constant = \frac{1}{2}\frac{1}{T_{\theta_2}}\right)$. If V is low and the tail length is short (ℓ_{cp} large), the path curves more rapidly to the left. Unless ℓ_{cp} is very large, however, the path of the zeros will nearly always pass to the right of the short-period poles.

$\ell_b < \ell_{cp}$ (Equivalent bobweight aft of center of percussion)



The zeros are seen here to become real when $\ell_b < \ell_{cp}$, and can be described as:

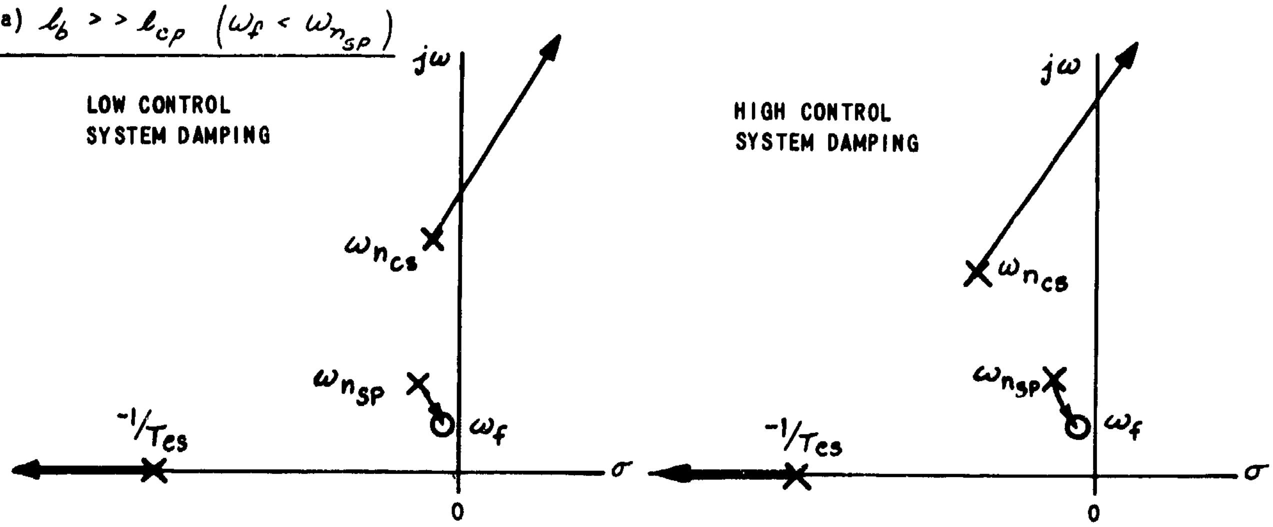
$$\left[\left(s + 1/T_{f_1}\right)\left(s + 1/T_{f_2}\right)\right]$$

where $1/T_{f_1}$ is positive and $1/T_{f_2}$ is negative.

With the above background in mind, it is now possible to examine some of the options available to the designer in controlling the relative positions of the poles and zeros. Obviously, the short-period and elevator servo poles are the least practical poles to alter. The control system poles can be altered fairly readily by changing the size of the feel spring $(\omega_{n_{cs}})$ and by introducing a viscous damper $(\zeta_{cs}\,\omega_{n_{cs}})$. The feedback zeros can also be altered easily, but since V , ℓ_{cp} , and $1/T_{\theta_2}$ are normally set by other aspects of the airplane design, the control system designer can affect the location of the zeros only through changes in ℓ_b . Thus, the designer can alter the control system poles and the feedback zeros quite easily, but the feedback zeros are constrained to move along fixed paths defined by V , ℓ_{cp} , and $1/T_{\theta_2}$.

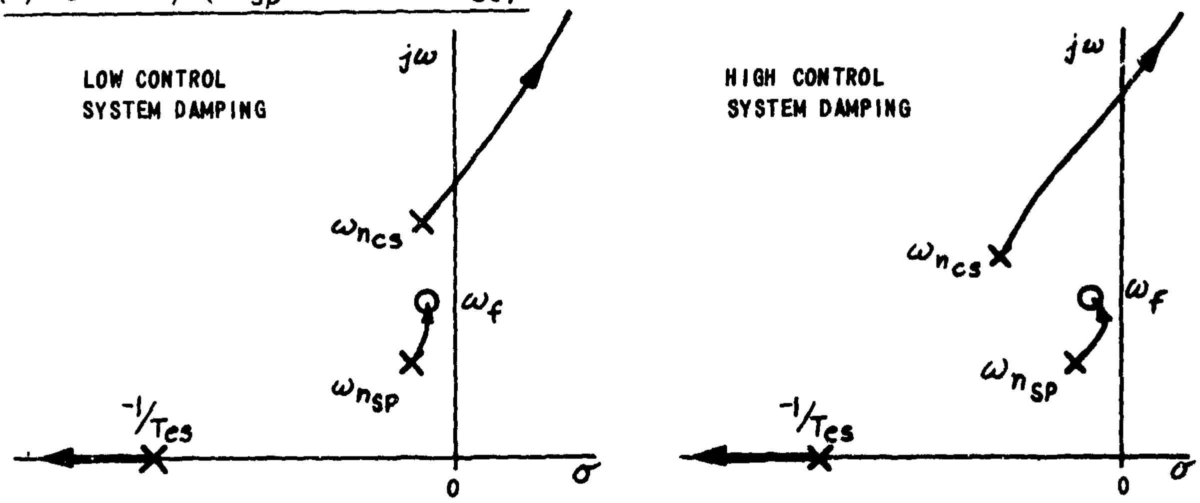
At this point, it is useful to sketch the loci of the stick-free roots for various locations of the control system poles and feedback zeros:

(a) $\ell_b >> \ell_{cp}$ $(\omega_f < \omega_{n_{sp}})$



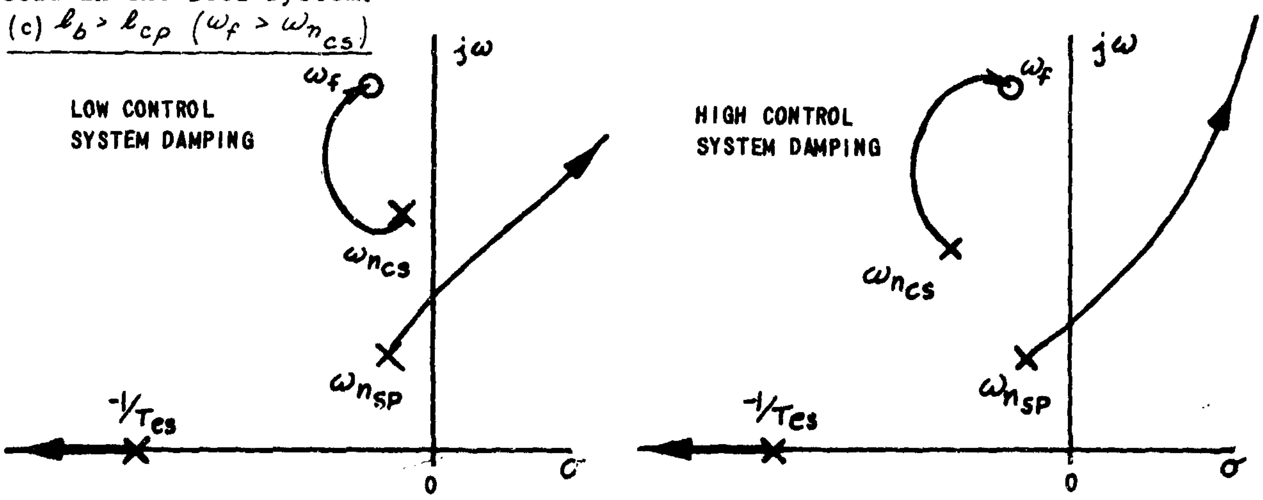
Configuration (a) is similar to the A4D-2 with the modified control system, and is generally fairly good. As $(F_s/n)_b/(F_s/n)_{fs}$ is increased, $\omega'_{n_{sp}}$ is reduced; but this should be no problem if ω_f is greater than the minimum $\omega'_{n_{sp}}$ limits of 3.2.2.1.1. Also, ζ'_{sp} is reduced; but this effect can be minimized by keeping ω_f significantly lower than $\omega_{n_{sp}}$. In addition, ζ'_{cs} decreases; but this problem can be minimized by keeping $(\zeta_{cs}\omega_{n_{cs}})$ high. High values of $(\zeta_{cs}\omega_{n_{cs}})$ also have the effect of slowing the reduction of ζ'_{sp}. Because $\omega'_{n_{sp}}$ is less than $\omega_{n_{sp}}$, there will be no phase lead in the feel system. This is a desirable feature, as explained in the historical development.

(b) $\ell_b > \ell_{cp}$ $(\omega_{n_{sp}} < \omega_f < \omega_{n_{cs}})$



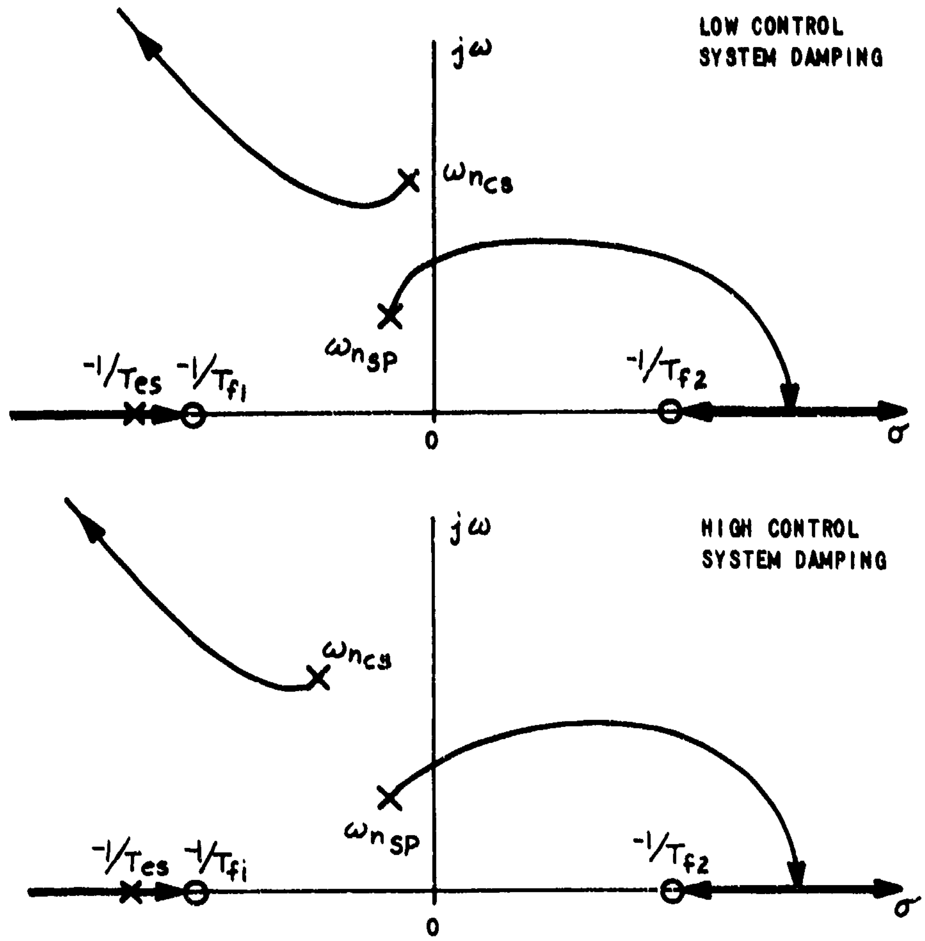
Configuration (b) is similar to the A4D-2 with the original control system, and the effects of normal-acceleration feedback are generally degrading. Both ζ'_{cs} and ζ'_{sp} decrease rapidly with increasing $(F_s/n)_b/(F_s/n)_{fs}$. Increasing the control system damping will improve ζ'_{cs}, but will cause ζ'_{sp} to decrease more rapidly. Because $\omega'_{n_{sp}}$ will be greater than $\omega_{n_{sp}}$, there will also be phase lead in the feel system.

(c) $\ell_b > \ell_{cp}$ $(\omega_f > \omega_{n_{cs}})$



Configuration (c) is similar to the T-38A, and is even less desirable than configuration (b). The short-period damping decreases very rapidly with increasing $(F_s/n)_b/(F_s/n)_{fs}$ and there is phase lead in the feel system because $\omega'_{n_{sp}}$ is greater than $\omega_{n_{sp}}$. Increasing the control system damping will reduce the feel system phase lead slightly, but will have little effect on ζ'_{sp}.

(d) $\ell_b < \ell_{cp}$



Notice that the root locus presented for configuration (d) is a reverse locus, i.e., the root-locus gain is negative. In fact, the root-locus gain is negative whenever $\ell_b < \ell_{cp}$, if the bobweight is to make a positive contribution to F_s/n, as can be seen from the following previously derived expression:

$$\text{loop gain} = \left[\frac{(F_s/n)_b}{(F_s/n)_{fs}}\left(T_{\theta_2}\ \frac{\ell_b - \ell_{cp}}{V}\right)\left(\omega^2_{n_{cs}}\ \omega^2_{n_{sp}}\ \frac{1}{T_{es}}\right)\right]$$

The configurations shown above have problems very similar to those of configuration (c).

From the above examples, it is fairly obvious that the best way to ensure maximum design flexibility in the choice of $(F_s/n)_b/(F_s/n)_{fs}$ is to make ℓ_b quite large and positive, so that ω_f is less than $\omega_{n_{sp}}$. At the same time, ω_f should not be so low that $\omega'_{n_{sp}}$ becomes too close to the lower limits of

3.2.2.1.1. Since $\omega'_{n_{sp}}$ will never reach ω_f for usable values of $(F_s/n)_b/(F_s/n)_{fs}$, it is probably reasonable to design ω_f equal to the appropriate minimum value of $\omega_{n_{sp}}$ from 3.2.2.1.1. Thus, a useful design criterion for selection of ℓ_b is:

$$\left[\frac{1}{T_{\theta_2}}\left(\frac{V}{\ell_b - \ell_{cp}}\right)\right] = (\omega_{n_{sp}})^2_{min}$$

Except for flight conditions having low dynamic pressure (low n/α), the $\omega_{n_{sp}}$ limits of 3.2.2.1.1 are actually lines of constant $\omega^2_{n_{sp}}/(n/\alpha)$, so that the above criterion can be expressed as:

$$\left[\frac{1}{T_{\theta_2}}\left(\frac{V}{\ell_b - \ell_{cp}}\right)\right] = \left(\frac{\omega^2_{n_{sp}}}{n/\alpha}\right)_{min}\left(\frac{n}{\alpha}\right) \approx \left(\frac{\omega^2_{n_{sp}}}{n/\alpha}\right)_{min}\left(\frac{V}{g}\frac{1}{T_{\theta_2}}\right)$$

so that the design criterion finally reduces to:

$$\frac{\ell_b}{g} = \left(\frac{\omega^2_{n_{sp}}}{n/\alpha}\right)^{-1}_{min} + \frac{\ell_{cp}}{g}$$

where

$$\frac{\ell_b}{g} = \frac{K_{\ddot\theta}}{K_n}$$

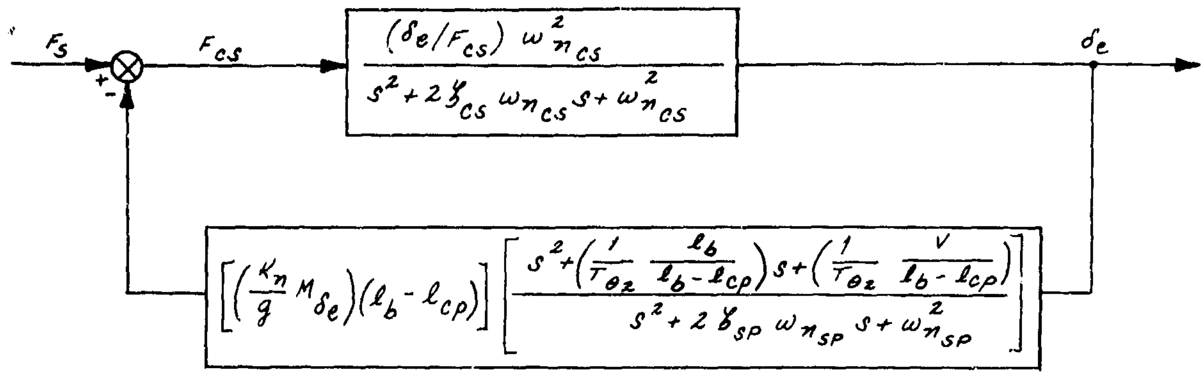
$$\ell_{cp} = \frac{Z_{\delta_e}}{M_{\delta_e}}$$

With ℓ_b determined by this criterion and with good control system damping, $(F_s/n)_b/(F_s/n)_{fs}$ can be increased to reasonably large values, the maximum value being limited primarily by ζ'_{sp} (3.2.2.1.2), $(F_s/n)_{min}$ (3.2.2.3.1), and ζ'_{cs} (3.5.3.2). Those configurations which have low enough ℓ_b so that $\omega_f > \omega_{n_{sp}}$ should also be checked for feel system phase lead (3.5.3.1). Note that by blending forward- and aft-mounted bobweights, ℓ_b can be made as large as desired, regardless of the airplane length.

Design Options (Unpowered Control Systems)

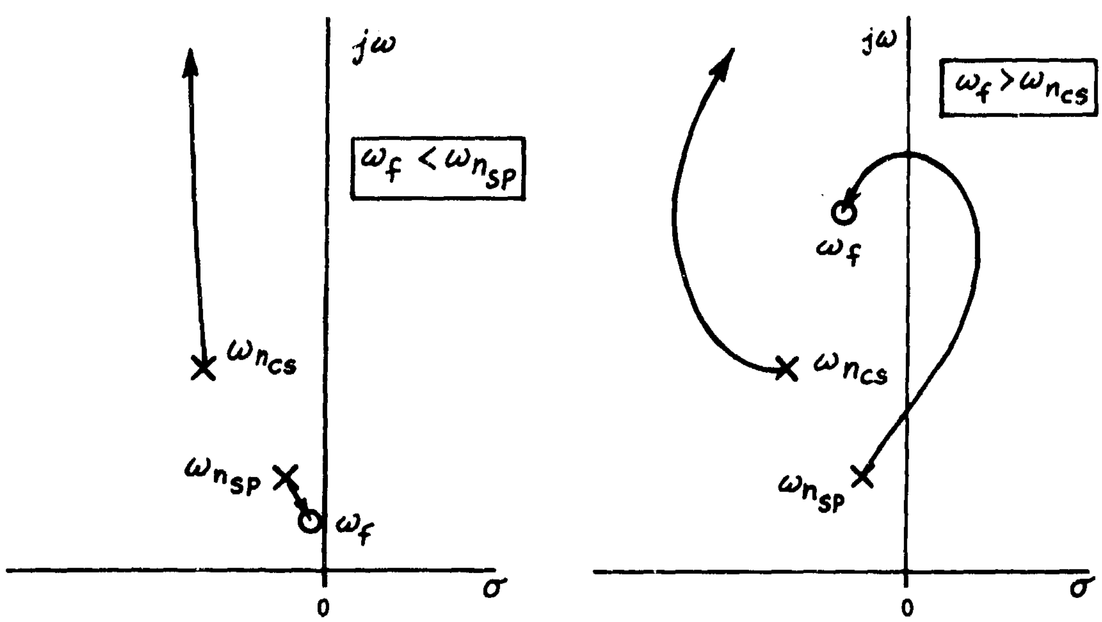
In the general case, the dynamics of airplanes having unpowered control systems are more difficult to analyze than those of airplanes having fully powered control systems. The dynamics are somewhat similar in nature, but the feedback zeros for the unpowered case are more complicated because of the influence of the elevator hinge moment parameter H_{α_e}. Normally, however, it is reasonable to divide such control systems into two types: those having a bobweight with $H_{\alpha_e} \approx 0$, and those having negligible mass unbalance. The former type will be considered first because it is the type most likely to cause problems.

For those cases where $H_{\alpha_e} \approx 0$, the feedback zeros can be described in the same form as for the powered control system. The block diagram representation for the unpowered system is therefore:



Root-locus plots for various values of ℓ_b and control system damping are similar to those given previously for powered control systems. Some root-locus plots for $\ell_b > \ell_{cp}$, for example, are as follows:

UNPOWERED CONTROL SYSTEMS



By comparison of these loci with those of the powered control system, it is apparent that there is less tendency for the loci of the unpowered system to cross the imaginary axis. This improvement in the combined damping of the control system and short-period modes is due to the absence of the elevator servo root. In fact, the roots are now constrained to move in such a manner that $(\zeta_{cs}\,\omega_{n_{cs}} + \zeta_{sp}\,\omega_{n_{sp}})$ is held constant. Thus, it is likely that the design of a suitable bobweight for an unpowered control system will be somewhat less critical than for a powered system.

In view of the above examples, the comments made in the previous section concerning the root loci for various values of ℓ_b apply also to the unpowered control system, with the exception that ζ'_{sp} and ζ'_{cs} decrease less rapidly for the unpowered system as $(F_s/n)_b/(F_s/n)_{ps}$ is increased. Thus, the unpowered system can also have serious problems, and should therefore be designed with the same care as a powered system.

In the case of an unpowered control system which is mass balanced but has H_{α_e}, the feedback zeros are quite different from those for a bobweight system. Referring back to the general expression for the feedback numerator quadratic given in the historical development, the expression reduces to the following form with $K_{\ddot{\theta}} = K_n = 0$:

$$s^2 + \left[\frac{M_{\delta_e}\frac{\alpha_e}{q} + \frac{Z_{\delta_e}}{V} + \frac{1}{T_\alpha}\frac{\alpha_e}{\dot{\alpha}} + \frac{Z_{\delta_e}}{V}\frac{\alpha_e}{\alpha}}{\frac{Z_{\delta_e}}{V}\ \frac{\alpha_e}{\dot{\alpha}}}\right] s + \left[\frac{\frac{Z_{\delta_e}}{V}\frac{1}{T_\alpha}\frac{\alpha_e}{\alpha} + M_{\delta_e}\frac{1}{T_{\theta_2}}\frac{\alpha_e}{q}}{\frac{Z_{\delta_e}}{V}\ \frac{\alpha_e}{\dot{\alpha}}}\right]$$

Using the approximate relationship $(1/T_\alpha) = \left(V\ \frac{M_{\delta_e}}{Z_{\delta_e}}\right)$, valid when $V\left|M_{\delta_e}\right| >> \left|Z_{\delta_e} M_q\right|$, the zeros can be expressed as the roots of:

$$s^2 + \left[V\ \frac{M_{\delta_e}}{Z_{\delta_e}}\left(\frac{\alpha_e/q}{\alpha_e/\dot{\alpha}} + 1\right) + \frac{\alpha_e/\alpha}{\alpha_e/\dot{\alpha}}\right] s + \left[V\ \frac{M_{\delta_e}}{Z_{\delta_e}}\ \frac{\frac{\alpha_e}{\alpha} + \frac{1}{T_{\theta_2}}\frac{\alpha_e}{q}}{\alpha_e/\dot{\alpha}}\right]$$

Since this polynomial normally factors into two widely-separated zeros on the negative real axis, the expression factors as follows:

$$\left(s + \frac{1}{T_{f_1}}\right)\left(s + \frac{1}{T_{f_2}}\right)$$

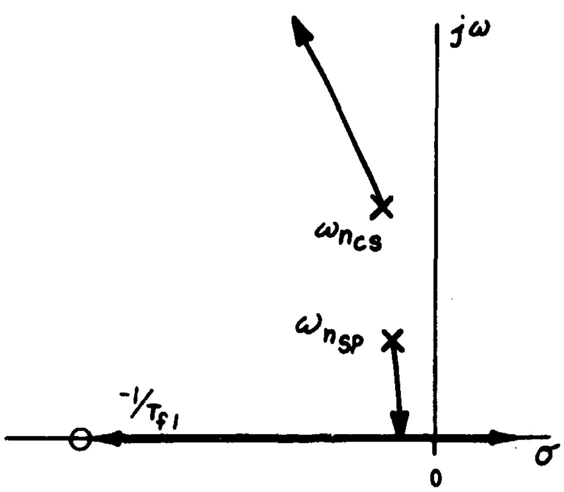
where

$$\frac{1}{T_{f_1}} = \left[V\ \frac{M_{\delta_e}}{Z_{\delta_e}}\left(\frac{\alpha_e/q}{\alpha_e/\dot{\alpha}} + 1\right) + \frac{\alpha_e/\alpha}{\alpha_e/\dot{\alpha}}\right] \qquad \text{(extremely large)}$$

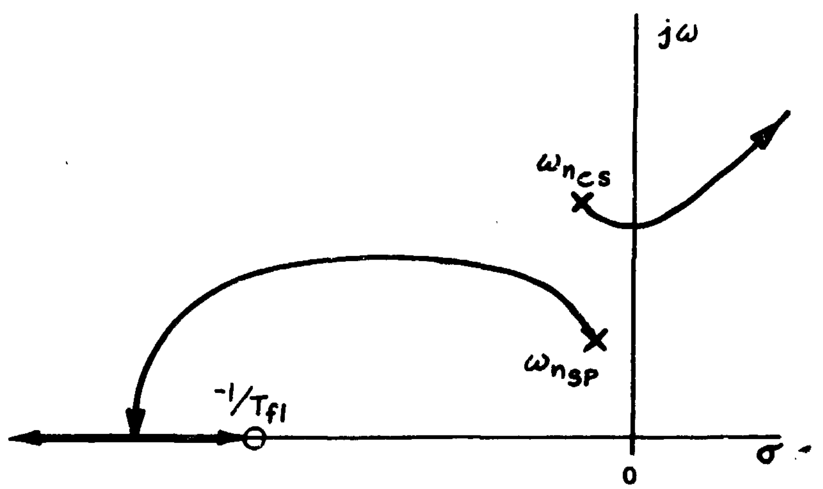
$$\frac{1}{T_{f_2}} = \left(\frac{\frac{\alpha_e}{\alpha} + \frac{1}{T_{\theta_2}}\alpha_e/q}{\frac{1}{V}\ \frac{Z_{\delta_e}}{M_{\delta_e}} + \alpha_e/q + \alpha_e/\alpha}\right)$$

The resulting root-loci are as follows:

Normal Elevator Float Tendency (H_{α_e} negative)



Elevator Float Against Relative Wind (H_{α_e} positive)



If the elevator is aerodynamically balanced $(H_{\alpha_e} = 0)$, the stick-free roots are identical to the stick-fixed roots. As H_{α_e} becomes negative (approaching a simple elevator with the hinge-line near its leading edge), $\omega'_{n_{sp}}$ is reduced and ζ'_{sp} is increased. As H_{α_e} becomes more negative, the stick-free short-period becomes overdamped. Finally, a real root reaches the origin $(F_s/n = 0)$, and then goes unstable (of course, the picture is actually somewhat more complicated than this because of the interaction of the low-frequency real root with the phugoid mode). This is the classic situation associated with the design of unpowered control systems, and the main problem is to keep $\omega'_{n_{sp}}$ above the minimum limits of 3.2.2.1.1. It is not likely, however, that negative H_{α_e} will cause PIO problems.

As H_{α_e} becomes positive, the above sketch shows that ζ'_{cs} decreases and is likely to become unstable. If ζ_{cs} is increased, the stick-free control system roots will move toward $\frac{1}{T_{f_1}}$, but the stick-free short-period roots will move in the unstable direction! In either case, there will be phase lead in the feel system because $\omega'_{n_{sp}} > \omega_{n_{sp}}$. The configuration is somewhat similar to a bobweight configuration with $\ell_b \approx \ell_{cp}$, and is rather poor. Therefore, a properly designed bobweight is definitely preferable to use of positive H_{α_e}, as a means for reducing variability in F_s/n .

3.2.2.3.1 TRANSIENT CONTROL FORCES

REQUIREMENT

3.2.2.3.1 Transient control forces. The peak elevator-control forces developed during abrupt maneuvers shall not be objectionably light, and the buildup of control force during the maneuver entry shall lead the buildup of normal acceleration. Specifically, the following requirement shall be met when the elevator control is pumped sinusoidally. For all input frequencies, the ratio of the peak force amplitude to the peak normal load factor amplitude at the c.g., measured from the steady oscillation, shall be greater than:

Center-Stick Controllers ---------- 3.0 pounds per g

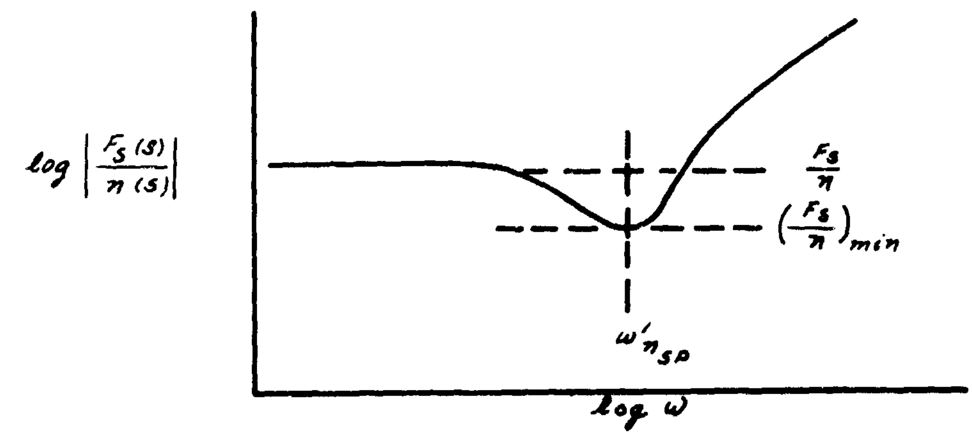
Wheel Controllers ---------------- 6.0 pounds per g

RELATED MIL-F-8785 PARAGRAPHS

3.3.10

DISCUSSION

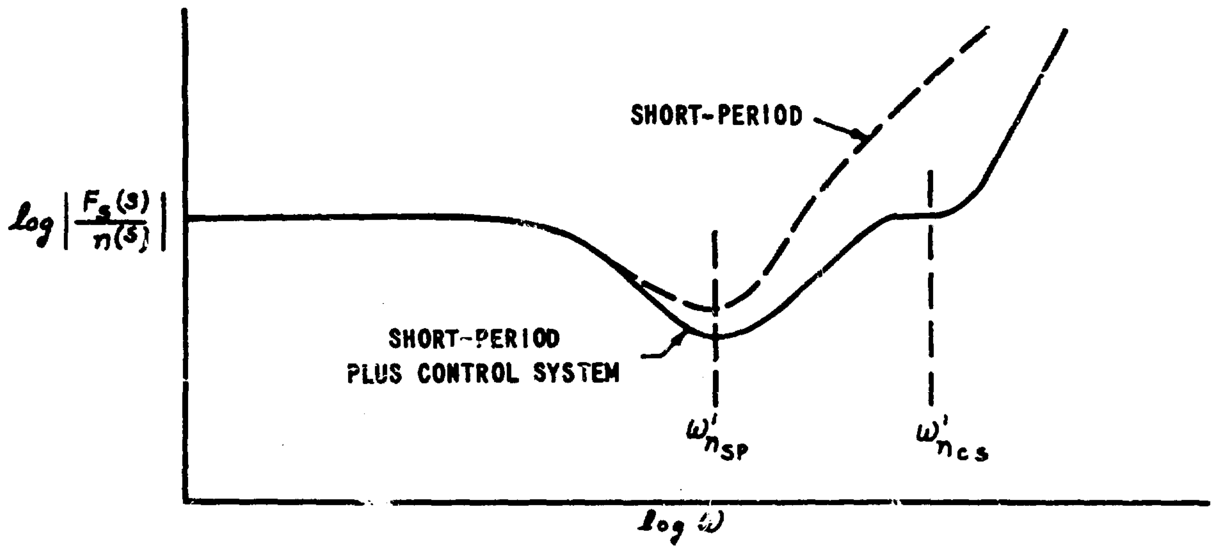
The requirements of MIL-F-8785B concerning $\omega_{n_{sp}}$, ζ_{sp}, and F_s/n will normally be sufficient to ensure adequate maneuvering characteristics. In certain situations, however, these requirements alone will not ensure against pilot-induced oscillations. Consider, for example, an airplane which meets the Level 2 requirements on $\omega_{n_{sp}}$, ζ_{sp}, and F_s/n for Category A Flight Phases. If ζ_{sp} and F_s/n are near the lower limits, the airplane can have PIO tendencies serious enough to make the airplane unacceptable. Paragraph 3.2.2.3.1 is designed to prevent this situation, by setting an upper limit on the frequency-response amplitude of $\frac{n(s)}{F_s(s)}$ (expressed as a lower limit on $\frac{F_s(s)}{n(s)}$). Flight-test techniques are discussed briefly in Appendix IVF.



This has the effect of increasing the minimum F_s/n requirements of 3.2.2.2.1 for low values of ζ_{sp} (stick-free), as can be seen by examination of the above sketch.

The dip in amplitude corresponds to the short-period resonance; and the size of the dip, expressed as the ratio $(F_s/n)/(F_s/n)_{min}$, is a unique function of ζ_{sp} (stick-free) (assuming that the control-system natural frequency is appreciably higher than $\omega_{n_{sp}}$). This functional relationship is shown in Figure 1. From this figure, it can be seen that F_s/n must increase rapidly with decreasing values of stick-free ζ_{sp}, in order to maintain a given value of $(F_s/n)_{min}$.

It should be understood that if the control system natural frequency is not appreciably higher than $\omega_{n_{sp}}$ (stick-free), the frequency response $\frac{F_s(s)}{n(s)}$ will not be entirely second-order in the region of $(F_s/n)_{min}$. If the control system damping is not very high, as is usually the case, the resonant dip can be accentuated by the control system mode, as can be seen from the following sketch:



In this situation, an equivalent ζ_{sp} (stick-free) can be obtained from Figure 1 by measurement of $(F_s/n)/(F_s/n)_{min}$.

Very little systematic data exist to support the criterion of 3.2.2.3.1, and the available data are all for center-stick controllers. The bulk of the systematic data was obtained from the flight program of Reference D3, and is presented in Figure 2. Note that ζ_{sp} (stick-fixed) and ζ_{sp} (stick-free) were identical for this program. Each configuration was rated by the pilot using a special PIO rating scale, shown in Table I, as well as by a conventional pilot rating scale. Figure 2 presents only those data having Level 1 values of $\omega_{n_{sp}}$ (Category A Flight Phases). Only part of the data is shown for $\zeta_{sp} \approx 0.6$ because of the tremendous amount available in this region. Because the pilot's opinion of the PIO tendencies of a given configuration is so highly dependent on how tightly he flies the airplane, there is considerable scatter in the data. In spite of the scatter, however, certain conclusions can be drawn. First, since the pilots were allowed to select the control system gearing in this program, it is apparent from Figure 1 that the pilots generally selected appreciably higher values of F_s/n at low ω_{sp} than at high ω_{sp}. Also, lines of constant $(F_s/n)_{min}$, obtained from Figure 1, seem to serve fairly well as iso-PIO rating lines. From the rating scale of Table I, it was decided that configurations having PIO ratings worse than 2.5 - 3.0 should not be allowed. From Figure 2, this dividing point is a line of $(F_s/n)_{min}$ equal to 3.0 lb/g. Notice that this value corresponds to the lowest value of F_s/n ever permitted by Table V of MIL-F-8785B, for each Level of flying qualities.

Flight test data are presented in Reference H5 for the T-38A. The pilots rated the airplane for one flight condition, using a PIO rating scale similar to that of Table I. These data are also presented in Figure 2. The original control system was rated 4, and the modified system was rated 2. Notice that these data fit in with the T-33 data very nicely.

The results of a limited in-flight investigation of bobweight effects (at constant F_s/n) are contained in Reference J60. Data from this program are presented in Figure 3. Although $\omega_{n_{sp}}$ was a bit high in this experiment, the rating trends are similar to those of Figure 2. Using average PIO ratings, Figure 2 indicates that $(F_s/n)_{min}$ is 2.4 lb/g for a PIO rating of 2.5, and 1.4 lb/g for a PIO rating of 3.5. These numbers compare reasonably well with the boundaries of Figure 2 for the same PIO ratings.

Data from three specific airplanes were analyzed, and the results are presented in Figure 4. In each case, the airplane with the original control system exhibited strong PIO tendencies in the high-speed, low-altitude flight regime. A modified control system was tried in each airplane, which significantly improved the situation. The majority of the points in Figure 4 are computed for the A4D-2 (Reference H11). The T-38A and F-4C data are from flight test (References H5 and P2, respectively). With the exception of the T-38A, there are no pilot ratings or detailed pilot comments available. It is only known that the shaded points of Figure 4 are associated with strong PIO tendencies. A line of $(F_s/n)_{min}$ = 1.4 lb/g divides the data very nicely. Comparing this value of $(F_s/n)_{min}$ with Figure 2, it would appear to correspond to a PIO rating of 3.5-4.0. Table I indicates that a PIO rating of 4 is indicative of fairly strong PIO tendencies, which is compatible with the very qualitative descriptions of the problems described in References H11, H5, and P2.

Some additional data on PIO tendencies are presented in Reference H2. Some data for the A4D-2 and T-38A are also included in this report but a more complete treatment of these airplanes has already been presented in Figure 4. The remaining data of Reference H2 are presented in Figure 5. The points are again rather crudely divided into those cases which exhibited PIO tendencies, and those which did not. Since little is known about the severity of the PIO problems associated with these airplanes, Figure 5 is used only to establish trends. As can be seen from the figure, a line of constant $(F_s/n)_{min}$ fits the data very well.

The results discussed above are for airplanes having center-stick controllers. For wheel controllers, the $(F_s/n)_{min}$ limit was doubled to reflect the difference in the F_s/n requirements for the two types of controllers (3.2.2.2.1).

Table I (3.2.2.3.1)

PIO TENDENCY RATING SCALE OF REFERENCE D3

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

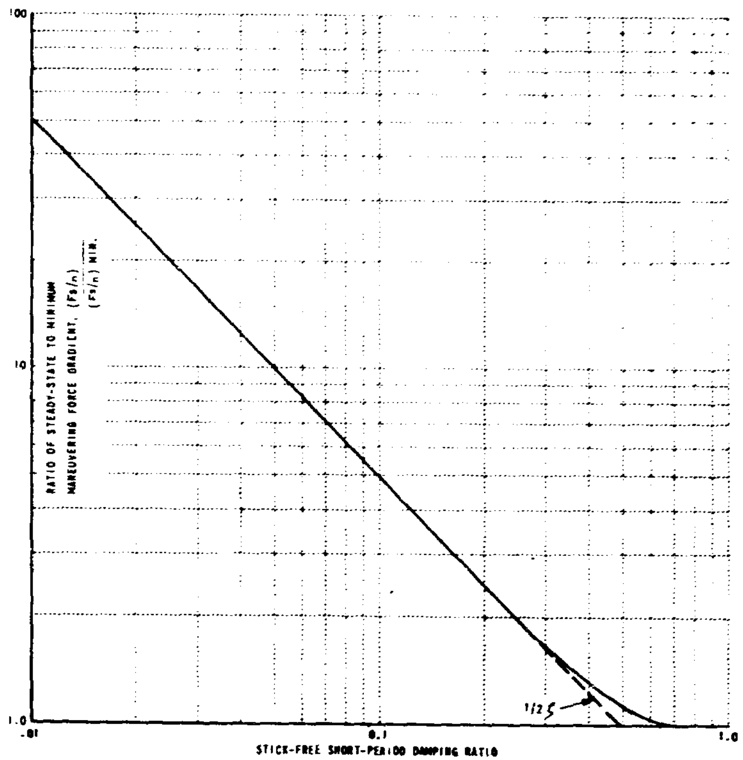


Figure 1(3.2.2.3.1)
THE RATIO $\left(F_s/n\right)/\left(F_s/n\right)_{min}$ VS. ζ_{SP}

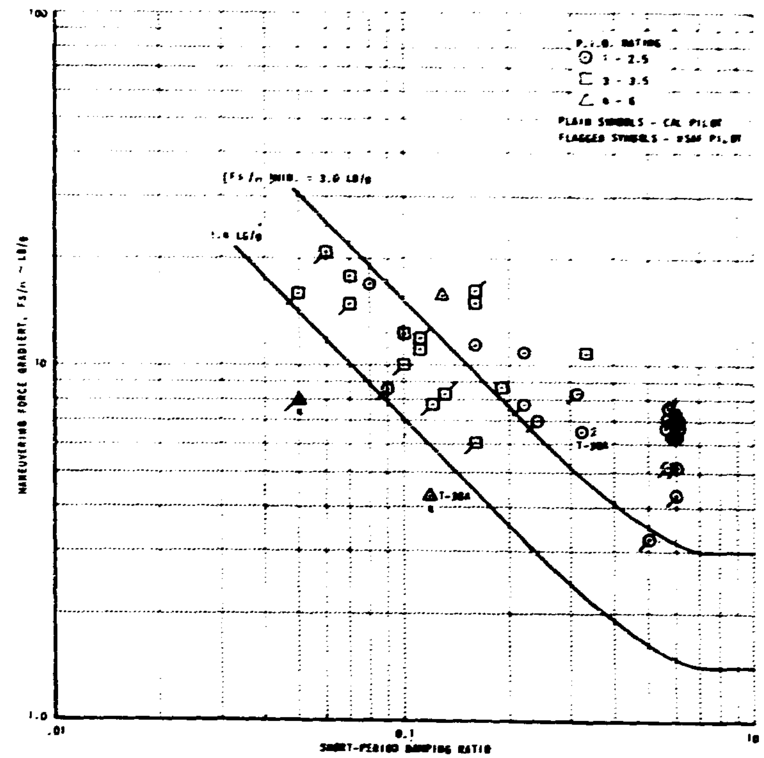


Figure 2(3.2.2.3.1)
T-33 FLIGHT PROGRAM OF REFERENCE D3

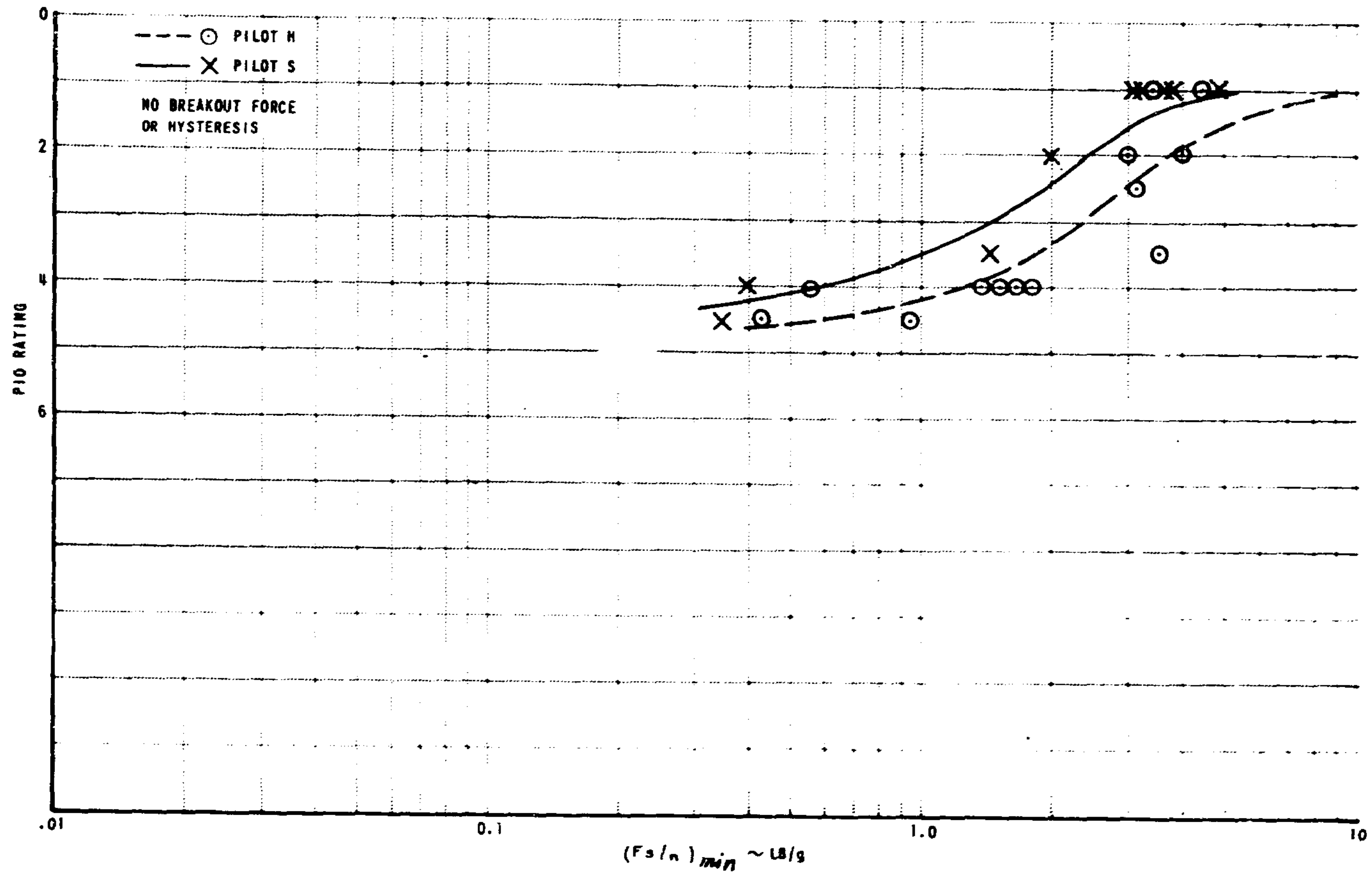


Figure 3(3.2.2.3.1) T-33 FLIGHT PROGRAM OF REFERENCE J60

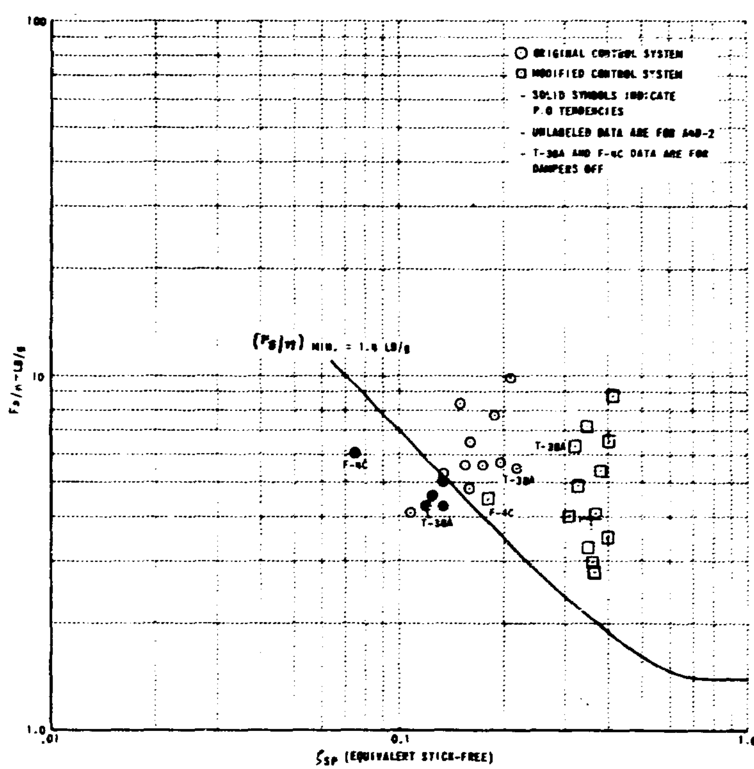


Figure 4(3.2.2.3.1)
PIO CHARACTERISTICS OF A4D-2, T-38A, F-4C
(REFERENCES H11, H5, P2)

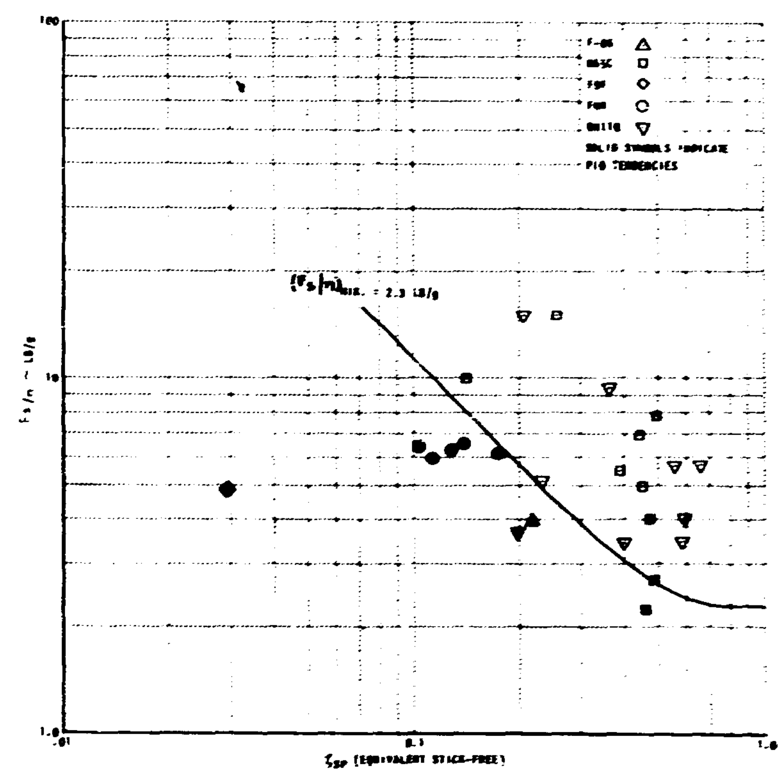


Figure 5(3.2.2.3.1)
PIO CHARACTERISTICS OF AIRPLANES
DESCRIBED IN REFERENCE H2

3.2.3 LONGITUDINAL CONTROL

GENERAL DISCUSSION

The requirements of the subparagraphs under 3.2.3 deal primarily with two aspects of longitudinal control. The first is to ensure that the airplane has adequate control effectiveness, and the second is to ensure that the elevator control forces required to perform certain specific maneuvers are within the pilot's capability to generate such forces.

As a minimum, the control effectiveness must be adequate to attain any speed and altitude within the permissible envelope, and to attain certain load factors. The control effectiveness must also be adequate to perform certain specific maneuvers associated with takeoffs, landings, dives, and sideslips.

For these same four types of specific maneuvers, the maximum forces required must be within certain limits.

Both the control-capability requirements and the control-force limitations of MIL-F-8785 were examined. In most cases, these requirements and limitations were considered necessary and reasonable, and have been incorporated into Reference A1 with little change.

3.2.3.1 LONGITUDINAL CONTROL IN UNACCELERATED FLIGHT

REQUIREMENT

3.2.3.1 Longitudinal control in unaccelerated flight. In erect unaccelerated flight at all service altitudes, the attainment of all speeds between V_S and V_{max} shall not be limited by the effectiveness of the longitudinal control, or controls.

RELATED MIL-F-8785 PARAGRAPHS

3.3.7

DISCUSSION

This requirement is essentially a rewording of 3.3.7 of MIL-F-8785 in the language of the new format. The need for such a requirement seemed obvious, and the paragraph was therefore retained.

3.2.3.2 LONGITUDINAL CONTROL IN MANEUVERING FLIGHT

REQUIREMENT

3.2.3.2 Longitudinal control in maneuvering flight. Within the Operational Flight Envelope, it shall be possible to develop, by use of the elevator control alone, the following range of load factors:

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

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

a) $n_o(+)$

b) $n = \begin{cases} 2.0 \text{ for } n_o(+) \le 3g \\ 0.5\,[n_o(+) + 1] \text{ for } n_o(+) > 3g. \end{cases}$

This maneuvering capability is required at the 1g trim speed and, with trim and throttle settings not changed by the crew, over a range about the trim speed the lesser of ±15 percent or ±50 knots equivalent airspeed (except where limited by the boundaries of the Operational Flight Envelope). Within the Service and Permissible Flight Envelopes, the dive-recovery requirements of 3.2.3.5 and 3.2.3.6, respectively, shall be met.

RELATED MIL-F-8785 PARAGRAPHS

3.3.8, 3.7.4

DISCUSSION

This paragraph is a rewrite of 3.3.8 of MIL-F-8785 in an attempt to make the requirement more reasonable, or at least more understandable.

Paragraph 3.3.8 of MIL-F-8785 applies to all permissible speeds, and requires that the airplane have enough elevator effectiveness to develop either n_L or the maximum operational load factor (a function of speed and altitude), except where limited by stall. This requirement has now been restricted in application to the Operational Flight Envelopes of Reference A1, with relaxed requirements for infrequent Failure States. Outside the Operational Flight Envelope, whatever falls out of the design is now acceptable, as long as the other control requirements are met.

The requirements for control effectiveness over a ±15 percent range about the trim speed were added to assure that excessive amounts of elevator-surface-fixed static stability (3.2.1.1) or instability will not limit maneuver capability unduly, for any possible mechanization of the trim system. Where elevator control authority limits normal-acceleration capability, the requirement at off-trim speeds often will be the designing consideration for elevator control effectiveness.

3.2.3.3 LONGITUDINAL CONTROL IN TAKEOFF

REQUIREMENT

3.2.3.3 Longitudinal control in takeoff. The effectiveness of the elevator control shall not restrict the takeoff performance of the airplane and shall be sufficient to prevent over-rotation to undesirable attitudes during takeoffs. Satisfactory takeoffs shall not be dependent upon use of the trimmer control during takeoff or on complicated control manipulation by the pilot. For nose-wheel airplanes it shall be possible to obtain, at 0.9 V_{min}, the pitch attitude which will result in takeoff at V_{min}. For tail-wheel airplanes, it shall be possible to maintain any pitch attitude up to that for a level thrust-line at 0.5 V_S for Class I airplanes and at V_S for Class II, III, and IV airplanes. These requirements shall be met on hard-surfaced runways. In the event that an airplane has a mission requirement for operation from unprepared fields, these requirements shall be met on such fields.

RELATED MIL-F-8785 PARAGRAPHS

3.3.11

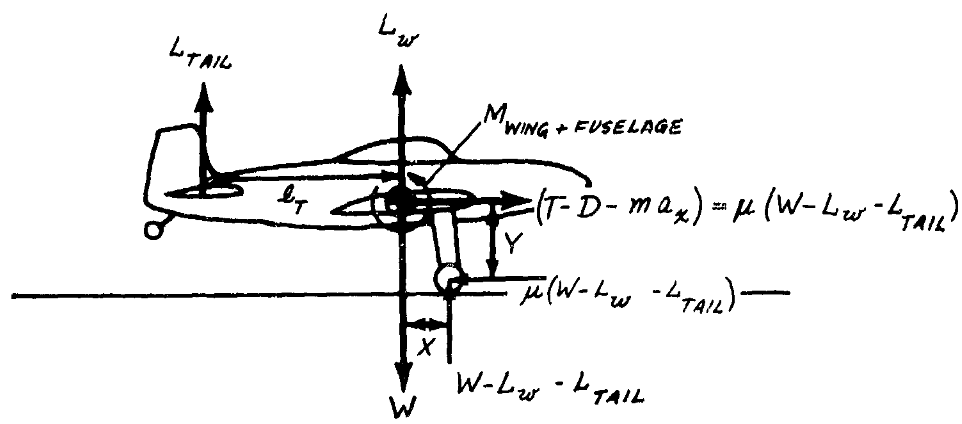
DISCUSSION

The control effectiveness requirements of 3.3.11 of MIL-F-8785 have been retained, with a few changes.

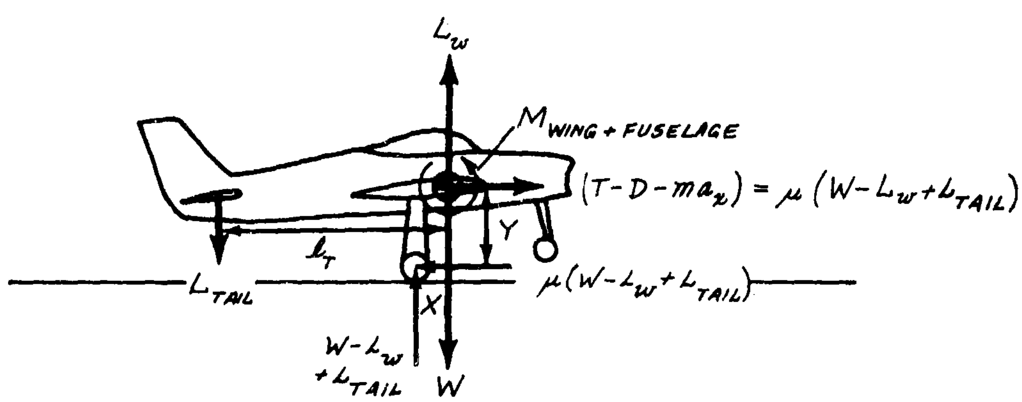
As a first step, all references to takeoff guarantees were deleted since there were several comments from industry and the military which indicated that meeting the takeoff guarantees does not necessarily guarantee enough elevator effectiveness for safe takeoffs.

For nose-wheel airplanes, it seemed more logical to specify the speeds in terms of V_{min} rather than V_S, and the term "takeoff attitude" was clarified by specifying the "attitude which will result in takeoff at V_{min}."

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



From ΣM_{CG}, $\quad L_{TAIL} = \dfrac{M_{W+F} + (X - \mu Y)(W - L_W)}{\ell_T + X - \mu Y}$



$$L_{TAIL} = \frac{-M_{W+F} + (X + \mu Y)(W + L_W)}{\ell_t - X - \mu Y}$$

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

3.2.3.3.1 LONGITUDINAL CONTROL IN CATAPULT TAKEOFF

REQUIREMENT

3.2.3.3.1 Longitudinal control in catapult takeoff. On airplanes designed for catapult takeoff, the effectiveness of the elevator control shall be sufficient to prevent the airplane from pitching up or down to undersirable attitudes in catapult takeoffs at speeds ranging from the minimum safe launching speed to a launching speed 30 knots higher than the minimum. Satisfactory catapult takeoffs shall not depend upon complicated control manipulation by the pilot.

RELATED MIL-F-8785 PARAGRAPHS

3.3.12

DISCUSSION

This requirement is essentially the same as 3.3.12 of MIL-F-8785, with an updated maximum launching speed.

3.2.3.3.2 LONGITUDINAL CONTROL FORCE AND TRAVEL IN TAKEOFF

REQUIREMENT

3.2.3.3.2 Longitudinal control force and travel in takeoff. With the trim setting optional but fixed, the elevator-control forces required during all types of takeoffs for which the airplane is designed, including short-field takeoffs and assisted takeoffs such as catapult or rocket-augmented, shall be within the following limits:

Nose-wheel and bicycle-gear airplanes

Classes I, IV-C ---------- 20 pounds pull to 10 pounds push

Classes II-C, IV-L ------- 30 pounds pull to 10 pounds push

Classes II-L, III -------- 50 pounds pull to 20 pounds push

Tail-wheel airplanes

Classes I, II-C, IV ------ 20 pounds push to 10 pounds pull

Classes II-L, III -------- 35 pounds push to 15 pounds pull

The elevator-control travel during these takeoffs shall not exceed 75 percent of the total travel, stop-to-stop. For purposes of this requirement, the term takeoff includes the ground run, rotation and lift-off, the ensuing acceleration to V_{max} (TO), and the transient caused by assist cessation. Takeoff power shall be maintained until V_{max} (TO) is reached, with the landing gear and high-lift devices retracted in the normal manner at speeds from V_{omin} (TO) to V_{max}(TO).

RELATED MIL-F-8785 PARAGRAPHS

3.3.13

DISCUSSION

The essential elements of 3.3.13 of MIL-F-8785 have been retained, but the tests have been reworded to cover the forces obtained in normal operation, rather than in some artificial situation.

The test described in 3.3.13 of MIL-F-8785 requires rotation at V_{STO}, with the ensuing takeoff occurring at a very low speed. Since there is no guarantee that such takeoffs are even critical in terms of forces, the forces during normal takeoffs are not effectively limited in MIL-F-8785. For this reason, the requirement was restated to include all types of takeoffs used in normal operation. This way of stating things is not as specific as MIL-F-8785, but is certainly more realistic. It is not intended to penalize an airplane in the event that a possible technique requires high pilot force or large control travel, but another easily learned and repeatable technique can be found that involves satisfactory control force and travel at no sacrifice in performance. Any technique having all these latter qualities is acceptable.

Another change to make the test more realistic was the requirement for acceleration to V_{max} (TO) rather than 1.3 $V_{S_{TO}}$ while allowing gear and flaps to be retracted normally. As explained in 6.2, V_{max} (TO) is related specifically to the configuration in the takeoff Flight Phase, though the airplane may no longer be in that configuration when V_{max} (TO) is reached.

3.2.3.4 LONGITUDINAL CONTROL IN LANDING

REQUIREMENT

3.2.3.4 Longitudinal control in landing. The elevator control shall be sufficiently effective in the landing Flight Phase in close proximity to the ground, that:

a) the geometry-limited touchdown attitude can be maintained in level flight, or

b) the lower of V_S (L) or the guaranteed landing speed can be obtained.

This requirement shall be met with the airplane trimmed for the approach Flight Phase at the recommended approach speed. The requirements of 3.2.3.4 and 3.2.3.4.1 define Levels 1 and 2. For Level 3, it shall be possible to execute safe approaches and landings in the presence of atmospheric disturbances.

RELATED MIL-F-8785 PARAGRAPHS

3.3.14

DISCUSSION

Paragraph 3.3.14 of MIL-F-8785 has been reworded slightly. One change is to trim at the recommended approach speed rather than 1.2 V_{S_L}. (V_{S_L} in MIL-F-8785 is equivalent to V_S(L) in MIL-F-8785B.) Even though this is not as well-defined a speed as 1.2 V_{S_L}, the change was considered necessary because 1.2 V_{S_L} may be quite unrealistic as an approach speed for many airplanes, especially STOL's.

Some manufacturers consider the requirement to fly near the ground at V_{S_L} unnecessarily strict. However, the requirement is necessary because of the imprecise nature of the landing flare maneuver. It is quite probable for a pilot, intentionally or unintentionally, to 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 elevator control to prevent the nose wheel from hitting the runway before the main gear. This same argument was used to change the references to guaranteed landing speed. V_S(L) is defined in 6.2 as being determined out of ground effect.

An additional requirement seems to be needed to assure that airplanes with large pitching inertia will have adequate landing flare capability. A neutrally stable airplane, or one with thrust below the c.g. or with pitch-up, could meet 3.2.3.4 and still not have enough control. Some minimum pitching acceleration capability is needed in approaches at speeds down to V_{min}. However, there was not enough information to allow a definitive general requirement to be written.

3.2.3.4.1 LONGITUDINAL CONTROL FORCES IN LANDING

REQUIREMENT

3.2.3.4.1 Longitudinal control forces in landing. The elevator-control forces required to meet the requirement of 3.2.3.4 shall be pull forces and shall not exceed:

Classes I, II-C, IV ------ 35 pounds

Classes II-L, III -------- 50 pounds

RELATED MIL-F-8785 PARAGRAPHS

3.3.15, 3.7.4.2

DISCUSSION

This requirement is essentially the same as 3.3.15 of MIL-F-8785, with the appropriate changes for the new Class definitions. The force levels still appear to be appropriate for landing.

3.2.3.5 LONGITUDINAL CONTROL FORCES IN DIVES - SERVICE FLIGHT ENVELOPE

REQUIREMENT

3.2.3.5 Longitudinal control forces in dives - Service Flight Envelope. With the airplane trimmed for level flight at speeds throughout the Service Flight Envelope, the elevator control forces in dives to all attainable speeds within the Service Flight Envelope shall not exceed 50 pounds push or 10 pounds pull for airplanes with center-stick controllers, nor 75 pounds push or 15 pounds pull for airplanes with wheel controllers. In similar dives, but with trim optional following the dive entry, it shall be possible with normal piloting techniques to maintain the forces within the limits of 10 pounds push or pull for airplanes with center-stick controllers, and 20 pounds push or pull for airplanes with wheel controllers. The forces required for recovery from these dives shall be in accordance with the gradients specified in 3.2.2.2.1 although speed may vary during the pullout.

RELATED MIL-F-8785 PARAGRAPHS

3.3.16, 3.7.4.1

DISCUSSION

This requirement is essentially the same as 3.3.16 of MIL-F-8785; the Service Flight Envelope of Reference A1 corresponds roughly to the operational envelope of MIL-F-8785. Also, the requirement is broken down more rationally, it is felt, according to type of controller, rather than class. The force levels were reviewed and found still appropriate.

Note that there are also lateral-directional requirements in these dives (3.3.8).

3.2.3.6 LONGITUDINAL CONTROL FORCES IN DIVES - PERMISSIBLE FLIGHT ENVELOPE

REQUIREMENT

3.2.3.6 Longitudinal control forces in dives - Permissible Flight Envelope. With the airplane trimmed for level flight at V_{MAT} but with trim optional in the dive, it shall be possible to maintain the elevator control force within the limits of 50 pounds push or 35 pounds pull in dives to all attainable speeds within the Permissible Flight Envelope. The force required for recovery from these dives shall not exceed 120 pounds. Trim and deceleration devices, etc., may be used to assist in recovery if no unusual pilot technique is required.

RELATED MIL-F-8785 PARAGRAPHS

3.3.16.1, 3.7.4.1

DISCUSSION

This requirement is essentially the same as 3.3.16.1 of MIL-F-8785. Higher, but reasonable, forces are allowed outside the Service Flight Envelope, and other means are allowed to assist recovery in these extreme conditions.

Note that there are also lateral-directional requirements in these dives (3.3.8).

3.2.3.7 LONGITUDINAL CONTROL IN SIDESLIPS

REQUIREMENT

3.2.3.7 Longitudinal control in sideslips. With the airplane trimmed for straight, level flight with zero sideslip, the elevator-control force required to maintain constant speed in steady sideslips with up to 50 pounds of rudder pedal force in either direction shall not exceed the elevator-control force that would result in a 1g change in normal acceleration. In no case, however, shall the elevator-control force exceed:

Center-stick controllers -------- 10 pounds pull to 3 pounds push

Wheel controllers --------------- 15 pounds pull to 10 pounds push

If a variation of elevator-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. These requirements define Levels 1 and 2. For Level 3, there shall be no uncontrollable pitching motions associated with the sideslips discussed above.

RELATED MIL-F-8785 PARAGRAPHS

3.3.20

DISCUSSION

There are two primary reasons for having requirements for maximum longitudinal forces in sideslips. The first is to ensure that small amounts of sideslip inadvertently developed during normal operations do not result in large or possibly dangerous angle-of-attack changes. This is the purpose of the requirement relating the maximum allowable forces in 3.3.20 of MIL-F-8785 to maneuvering control forces. The second reason is simply to limit the longitudinal corrections required when the pilot intentionally changes the sideslip angle, as in a crosswind landing for instance. The other force limits of 3.3.20 are designed for this purpose.

The requirements of 3.2.3.7 are therefore a straightforward rewording of 3.3.20 of MIL-F-8785 in the new format. The allowable elevator control force is limited to the lower of the stated value or the force (including breakout) required to pull up at 2g's at the same flight condition.

3.3 - LATERAL-DIRECTIONAL FLYING QUALITIES

3.3 - LATERAL-DIRECTIONAL FLYING QUALITIES

3.3 LATERAL-DIRECTIONAL FLYING QUALITIES

DISCUSSION

General

This section was difficult to organize since, primarily because of coupling between lateral and directional motions, each requirement has implications in many areas of flying qualities. Conversely, each flying qualities area is generally a function of many different parameters. For example, the rolling moment acting on an aircraft following an aileron input is a function of the directional stability, the yaw due-to-aileron, and the dihedral characteristics of the aircraft in addition to the aileron roll effectiveness characteristics. Directional stability, however, also influences the Dutch roll characteristics and the flying qualities in steady sideslips, with asymmetric thrust, and during takeoff and landing. Thus, to entitle a requirement dealing with only one of the above flying qualities areas as "Directional stability," as is done in MIL-F-8785, is misleading since all these areas pertain to directional stability. To compound the problem, the paragraphs in MIL-F-8785 that are entitled directional stability are based on control inputs required in steady sideslips. Thus, since control inputs introduce control derivatives and coupling derivatives, the paragraphs also have implications to areas other than directional stability.

To best avoid this problem, headings such as directional stability and dihedral effect have not generally been used; instead, headings that relate directly to the requirement or test in question are used. For example, the MIL-F-8785 paragraphs entitled "Static directional stability" are now entitled "Yawing moments in steady sideslips." Similarly, the MIL-F-8785 paragraphs entitled "Dihedral effect" are now entitled "Rolling moments in steady sideslips." Since these headings are a more accurate indication of the contents of the paragraphs, it is believed that use of the specification will be facilitated.

In MIL-F-8785, there are several aspects of dynamic lateral-directional characteristics that are not very well covered. These deficiencies have been clearly recognized and are discussed in some detail in References A4, A5 and A15. For example, at present, there is no requirement on roll damping or the coupled roll-spiral, even though many experimental programs and studies, for example, References F8 and F12, have confirmed their importance to flying qualities. There is also practically nothing concerning the very complex coupling between yawing motions and rolling motions and the response to control inputs and turbulence, although here also, there is a great deal of literature that shows the importance of these factors on flying qualities. To help overcome these deficiencies, requirements have been formulated for roll damping, for the coupled roll-spiral and for the amount of coupling that can exist between roll and yaw during turns and turn entries.

The requirements on the roll and the coupled roll-spiral modes were integrated with existing requirements on the Dutch roll and spiral modes into a section entitled "Lateral-directional mode characteristics." Roll-sideslip coupling involves more than modal characteristics, so the requirements on response to turbulence and roll-sideslip coupling during turns and turn entries have been placed in a separate section. This underlines the importance to flying qualities of coupling phenomena.

The equations of motion which form the basis for the discussions of theory, along with some of the more involved theoretical developments, are given in Appendices VA and VC.

Requirements Pertaining to Sideslips

One of the more difficult problems in the lateral-directional section was how best to specify the flying quality characteristics involving sideslips. Sideslips can be either steady or dynamic and can develop in many ways. They can be caused by control inputs and coupling effects, by thrust or aerodynamic asymmetries such as engines out, asymmetric store loadings, and uneven gear retraction or extension, or by atmospheric disturbances. Since the implication of sideslip to flying qualities depends upon the nature of the forcing function and the type of maneuvers to be performed, it was necessary to specify several different requirements to cover the most significant combinations of forcing functions and required maneuvers. This subject is discussed in a general way in the following paragraphs to familiarize the reader with some of the many considerations involved.

Rudder pedals are used for many different purposes. Although no list of rudder pedal usage would be complete, some of the more important uses are listed below.

a. To perform a crosswind landing - either employ a steady rudder-pedal-induced sideslip or else a decrab maneuver.

b. To augment roll rate anywhere within the flight envelope.

c. To raise a wing when the pilot is busy with his hands, such as when taking a clearance.

d. For tracking, for example, in air-to-ground gunnery in a crosswind or when acquiring targets.

e. For wing-overs or other tactical maneuvers to obtain a rapid change in heading or bank angle.

f. For close formation flying.

g. To lose altitude as in a "forward" sideslip or to improve visibility, for example, a pilot landing from the rear seat of a tandem-seat airplane.

h. To counter yawing moments from propeller torque, speed or Mach number change, asymmetric thrust or stores, etc.

i. To coordinate turn entries or steady turns.

j. To taxi

The requirements of 3.3.4.5, 3.3.6 and 3.3.7 are directed at those maneuvers, described above, that employ rudder-pedal-induced sideslips and attempt to insure that the airplane responds to specific control inputs in such a manner that the maneuvers can be readily performed.

The dynamic effects of coupling between sideslip and roll following an aileron input can be manifested in different ways. Thus, requirements were directed both at aircraft with sufficient roll-sideslip coupling that excitation of the Dutch roll results in roll control or bank angle tracking problems (paragraphs 3.3.2.2, 3.3.2.2.1, 3.3.2.3), and at aircraft with low enough roll-sideslip coupling that excitation of the Dutch roll results primarily in sideslip or heading control problems (paragraphs 3.3.2.4 and 3.3.2.4.1).

The requirements on rudder-pedal-induced sideslips are directed primarily at the static characteristics of the aircraft, whereas the roll sideslip coupling requirements are directed primarily at the dynamic controllability problem. As might be expected, different combinations of stability derivatives are important for the two cases. For example, for the rudder-pedal-induced steady sideslips, L_β , N_β , $L_{\delta_{AS}}$, $N_{\delta_{AS}}$, $L_{\delta_{RP}}$, $N_{\delta_{RP}}$ are important, whereas for dynamic controllability the primed rate derivatives L'_β , N'_β , L'_r , N'_r , L'_p , N'_p and the bank angle side force term, $\frac{g}{V}$, must also be considered (the primed derivatives account for I_{xz} effects).

Sideslips resulting from asymmetric thrust are covered in the paragraphs under 3.3.9. Paragraphs 3.3.9.1, 3.3.9.2 and 3.3.9.3 are directed at ensuring that a pilot, making full use of cockpit controls, can safely control an airplane when thrust is suddenly lost during takeoff. These requirements not only ensure that the pilot can maintain straight flight without reducing thrust on the remaining engines, so that the takeoff may be continued, but that he can handle the transient following thrust loss as well. Paragraph 3.3.9.4 is directed at ensuring a balance between upsetting moments due to asymmetric thrust and the restoring moment due to directional stability and rudder control. Paragraph 3.3.9.5 considers two or more engines out.

Organization of Section 3.3 "Lateral-directional flying qualities"

In order to incorporate the new requirements into the specification and to overcome some of the deficiencies in MIL-F-8785 previously discussed, a major reorganization of the lateral-directional section was required. In this reorganization the paragraphs were arranged as much as possible so that like things are grouped together, and headings were chosen to reflect this grouping.

The first three three-digit paragraphs (3.3.1 "Lateral-directional mode characteristics," 3.3.2 "Lateral-directional dynamic response characteristics," and 3.3.3 "Pilot-induced oscillations") deal with complex coupled lateral-directional motions in which the various motions cannot be treated independently. The next two three-digit paragraphs (3.3.4 "Roll control effectiveness," and 3.3.5 "Directional control characteristics") deal with, hopefully, relatively uncoupled motions or control considerations that can be identified as primarily dealing with either roll or directional control characteristics. The final four three-digit paragraphs (3.3.6 "Lateral-directional characteristics in steady sideslips," 3.3.7 "Lateral-directional control in cross winds," 3.3.8 "Lateral-directional control in dives," and 3.3.9 "Lateral-directional control with asymmetric thrust") all deal with control under specific operating conditions that have implications to both roll and directional control characteristics.

The term "roll control" is used in lieu of "lateral control" since lateral motions are literally motions along the y axis. Also, often "lateral control" is used to describe both rolling and yawing commands. On the other hand, the meaning of directional control is not ambiguous, is accepted practice, and so is used throughout. "Roll control" means simply control of roll. "Aileron control" refers to the cockpit aileron control and aileron surface of course refers to whatever flight control surfaces provide the roll control.

3.3.1 LATERAL-DIRECTIONAL MODE CHARACTERISTICS

DISCUSSION

This section combines new requirements on the roll mode and the coupled roll-spiral mode with modified existing requirements on the Dutch roll and spiral modes. The requirements are basically specified in terms of such conventional modal parameters as roll mode time constant, Dutch roll natural frequency, Dutch roll damping ratio, and time to double amplitude for the spiral, in order to most clearly and completely take into account the many complex phenomena that are associated with the dynamic situations. The form of the requirements is dictated by the extent of our knowledge of aircraft motions.

Although airplanes may exhibit nonlinear and higher-order motions that make requirements based on first- and second-order parameters not strictly applicable, airplane motions can normally be closely approximated by equivalent conventional motions in order to determine compliance with the requirements. Also, where possible, provisions have been made within the requirements to account for nonlinear and higher-order responses.

Requirements on minimum Dutch roll damping are specified to limit the oscillations of the Dutch roll after it has been excited. Requirements on minimum Dutch roll frequency are specified to limit the amount of sideslip generated by a given yawing disturbance and to ensure that the aircraft will naturally tend to point the way it is going.

Requirements on maximum roll mode time constant and the coupled roll-spiral are directed at precision of roll control.

Requirements on the spiral are directed at limiting the amount of attention required by the pilot to fly the aircraft and at restricting the amount of aileron that must be held during a turn.

3.3.1.1 LATERAL-DIRECTIONAL OSCILLATIONS (DUTCH ROLL)

REQUIREMENT

3.3.1.1 Lateral-directional oscillations (Dutch roll). The frequency, ω_{n_d}, and damping ratio, ζ_d, of the lateral-directional oscillations following a rudder disturbance input shall exceed the minimums in table VI. The requirements shall be met with cockpit controls fixed and with them free, in oscillations of any magnitude that might be experienced in operational use. If the oscillation is nonlinear with amplitude, the requirement shall apply to each cycle of the oscillation. Residual oscillations may be tolerated only if the amplitude is sufficiently small that the motions are not objectionable and do not impair mission performance. For Category A Flight Phases, angular deviations shall be less than ±3 mils. With the control surfaces fixed, ω_{n_d} shall always be greater than zero.

TABLE VI. Minimum Dutch Roll Frequency and Damping

Level	Flight Phase Category	Class	Min ζ_d *	Min $\zeta_d \omega_{n_d}$, rad/sec.*	Min ω_{n_d}, rad/sec.
1	A	I, IV	0.19	0.35	1.0
		II, III	0.19	0.35	0.4**
	B	All	0.08	0.15	0.4**
	C	I, II-C, IV	0.08	0.15	1.0
		II-L, III	0.08	0.15	0.4**
2	All	All	0.02	0.05	0.4**
3	All	All	0.02	-	0.4**

*The governing damping requirement is that yielding the larger value of ζ_d.

**Class III airplanes may be excepted from the minimum ω_{n_d} requirement, subject to approval by the procuring activity, if the requirements of 3.3.2 through 3.3.2.4.1, 3.3.5 and 3.3.9.4 are met.

When $\omega_{n_d}^2 |\phi/\beta|_d$ is greater than 20 (rad/sec)^2, the minimum $\zeta_d \omega_{n_d}$ shall be increased above the $\zeta_d \omega_{n_d}$ minimums listed above by:

Level 1 - $\Delta \zeta_d \omega_{n_d} = .014(\omega_{n_d}^2 |\phi/\beta|_d - 20)$

Level 2 - $\Delta \zeta_d \omega_{n_d} = .009(\omega_{n_d}^2 |\phi/\beta|_d - 20)$

Level 3 - $\Delta \zeta_d \omega_{n_d} = .005(\omega_{n_d}^2 |\phi/\beta|_d - 20)$

with ω_{n_d} in rad/sec.

RELATED MIL-F-8785 PARAGRAPHS

3.4.1, 3.4.1.1, 3.4.1.2

DISCUSSION

General

The MIL-F-8785 requirements on "Damping of the lateral-directional oscillations" have been altered and expanded significantly in MIL-F-8785B. Whereas the requirements in MIL-F-8785 were in terms of $1/C_{1/2}$ and $|\phi/v_e|_s$, the requirements in MIL-F-8785B are in terms of ζ_d, $\zeta_d\omega_{n_d}$, ω_{n_d} and $\omega_{n_d}^2|\phi/\beta|_d$. Frequency, as indicated in References A4 and A15 and as will be shown in the following paragraphs, is a significant lateral-directional flying qualities parameter. The requirement which specifies Dutch roll damping as a function of the parameter $|\frac{\phi}{v_e}|$ was deleted since, at best, it offered an inadequate solution to a complex situation.

As with longitudinal short-period characteristics, the amplitudes at which these requirements apply are indicated only qualitatively here and in 3.5.4.2. See the discussion of 3.2.2.1.

The numerical requirements given in Table VI can be considered as basic requirements. Additional direct requirements on $\zeta_d\omega_{n_d}$ as a function of the coupling parameter $\omega_{n_d}^2|\frac{\phi}{\beta}|_d$ are also specified in 3.3.1.1, and additional indirect requirements on Dutch roll frequency and damping ratio as functions of various disturbance parameters are contained elsewhere throughout the lateral-directional section. For example, requirements that are directed at coupling and interaction effects, such as roll-sideslip coupling (paragraphs 3.3.2.2 through 3.3.2.6), have strong implications to lateral-directional oscillatory characteristics.

The two primary references used in evaluating the merits of $|\frac{\phi}{v_e}|$ were References F9 and F51. Reference F9 discusses the parameter in detail and points to many cases where misplaced faith has been placed in $|\frac{\phi}{v_e}|$. Reference F51 was examined since it was a pioneering effort in this area and made tentative recommendations for Dutch roll damping in the form of $1/C_{1/2}$ and $|\frac{\phi}{v_e}|$ parameters (Figures 1 through 4). The data show that, although the trends can be explained in terms of $1/C_{1/2}$ and $|\frac{\phi}{v_e}|$, there are other effects besides the model parameter $|\frac{\phi}{v_e}|$ that would be expected to significantly affect the pilot ratings. Even with extreme $\left|\frac{\phi}{v_e}\right|$, the response to roll commands would not be oscillatory unless something like adverse yaw or a side gust is present to excite the Dutch roll. In particular, the configurations with large $\left|\frac{\phi}{v_e}\right|$ exhibited roll rate reversals. It is reasonable to assume that the pilot ratings were significantly affected, thus making the resultant requirements strictly applicable only to airplanes with excessive adverse yaw.

In order to determine basic lateral-directional oscillatory requirements and to investigate $\left|\frac{\phi}{v_e}\right|$ or $\left|\frac{\phi}{\beta}\right|$ type effects, applicable data for which roll control coupling effects were small were plotted as a function of frequency and damping ratio. The data are presented in Figures 5 through 11 with curves of ζ_d, $\zeta_d \omega_{n_d}$ and ω_{n_d} from the requirements of 3.3.1.1 superimposed.

Reference G7 and F21	Figure 5
Reference F55	Figure 6
Reference B44	Figure 7
References F5, F7 and F22	Figure 8
Reference F76	Figure 9
Reference F75	Figure 10
Reference A4	Figure 11
(Data on existing airplanes)	

Requirements on ζ_d

From examination of the data, it was apparent that over a wide range of frequencies and $\left|\frac{\phi}{\beta}\right|_d$ response ratios, lines of constant damping ratio fit the data quite well. Moreover, the values of $1/C_{1/2}$ in MIL-F-8785 for normal operation (equivalent $\zeta_d \approx .078$) and operation with an artificial stabilization device inoperative (equivalent $\zeta_d \approx .022$) tie in well with pilot rating boundaries of satisfactory (Level 1) and acceptable (Level 2) for B and C Category Flight Phases. Although correlation is far from perfect, there are insufficient data upon which to substantiate a change from existing requirements for frequencies above 2 radians/second.

For Flight Phase Category A, Level 1, the $1/C_{1/2}$ value in MIL-F-8785 for armed aircraft (equivalent $\zeta_d \approx .19$) has been applied. This value was retained, even though data from Reference F1 (discussed in paragraph 3.3.2.2) show that a damping ratio of $\zeta_d = 0.1$ can provide satisfactory handling qualities for Flight Phase Category A, Level 1. The reasoning behind this is that Reference F44 (Figures 12 through 15) shows that tracking errors in turbulence decrease as damping ratio increases, and at this time there is no direct quantitative requirement on the response to turbulence. It is anticipated that when such a requirement can be specified, the required $\zeta_d = 0.19$ could be relaxed to $\zeta_d = 0.1$, if both the roll-sideslip coupling requirements and the turbulence requirements are met.

Requirements on $\zeta_d \omega_{n_d}$

From the literature survey and from discussion with aircraft manufacturers, it became apparent that even for "basic" lateral-directional oscillatory requirements, there were differing views on the relative merits of the parameters ζ_d and $\zeta_d \omega_{n_d}$. The former is equivalent to cycles to damp, while $\zeta_d \omega_{n_d}$ is equivalent to time to damp. As was mentioned previously, examination of the raw data from several reports indicates that the existing requirements in MIL-F-8785, which are a direct function of ζ_d, fit the data well over quite a wide range of frequencies and $\left|\frac{\phi}{\beta}\right|_d$ response ratios. On the other hand, there is considerable opinion in the literature that under certain conditions, the pilot ratings correlate better with $\zeta_d \omega_{n_d}$ than with ζ_d. Some examples are given in the following paragraphs.

Reference G7 reports on an in-flight landing approach study using a variable stability F-86E. The data are presented on a $\zeta_d \sim \omega_{n_d}$ plot in Figure 5 and in terms of various other parameters in Figures 16 through 21. No firm conclusions can be drawn, however, since detailed pilot comments are not given which would permit identification of problems.

Reference F75 considered data of Reference F55. From these data (Figure 22) it was found that best correlation of the pilot ratings was obtained with the parameter $1/C_{1/2}$ for Dutch roll natural frequencies greater than 2.4 rad/sec, and with the parameter $1/T_{1/2}$ for Dutch roll frequencies less than 2.4 rad/sec. It should be noted, however, that at low frequencies the aircraft used in the Reference F55 program did not meet the military requirements on roll rate reversals and sideslip generated during uncoordinated rolls. This coupling influenced the low-frequency ratings sufficiently that the data cannot be used to determine basic lateral-directional oscillatory requirements.

Reference F9, using the data of References G7, F21, F22, F77 and F7, concluded that the "basic damping requirement appears to be best specified in terms of total damping $\zeta_d \omega_{n_d}$ rather than damping ratio, ζ_d." Although the data do correlate reasonably well with the parameter $\zeta_d \omega_{n_d}$, examination of the raw data from the referenced reports (presented in Figures 5, 8 and 9) indicates no clear superiority over the parameter ζ_d, particularly for moderate and high frequencies. Moreover, since there are practically no data in the low-frequency region ($\omega_{n_d} < 1$ rad/sec), the question of whether ζ_d or $\zeta_d \omega_{n_d}$ is the better is difficult to answer on the basis of available data. Since, however, $\zeta_d \omega_{n_d}$ is generally accepted in the literature, minimum $\zeta_d \omega_{n_d}$ requirements were specified. This is consistent with the approach taken by Reference A14 (Figure 23) in specification of criteria for low frequencies.

Requirements on ω_{n_d}

In determining the minimum frequency boundaries, it was found that the more closely the low-frequency data were examined, the more difficult it became to assess the importance of low Dutch-roll frequency per se. For example, the data from Reference G11 (Figure 24) were examined, but were considered inconclusive because of the degree of coupling present. The low-frequency point at $L_\beta = -16$ had considerable roll rate oscillation following a step aileron input, and the low-frequency point at $L_\beta = 0$ had considerable sideslip excitation following a step aileron input. This coupling would be expected to cause a significant degradation in pilot rating. Degradation of pilot rating could also have been caused by the zero dihedral in the latter case.

The low-frequency data of Reference F22 were also examined, but again poor pilot ratings were explainable through Dutch-roll excitation. For the $\omega_{n_d} = 1$ cases, when Dutch-roll excitation following step aileron inputs was small and damping was high, the pilot ratings were good. For the $\omega_{n_d} < 1$ cases, the ratings were all poor but are explainable by the amount of sideslip generated by aileron inputs.

In spite of these findings, since flying qualities can degrade in so many ways when directional stiffness becomes low, there was a strong conviction that a minimum frequency should be specified. For example, the amount of sideslip caused by yawing moments (say yaw due-to-aileron), is directly proportional to $1/\omega_{n_d}^2$. From Figure 25 it can be seen that for $\omega_{n_d} < 1$, $1/\omega_{n_d}^2$ increases very rapidly.

Since little experimental data could be found to determine minimum values of ω_{n_d}, these requirements were selected on the basis of characteristics of existing airplanes. From Figure 11 it can be seen that all current large aircraft for which data are presented have a minimum ω_{n_d} about or above 0.4 rad/sec, while all current small aircraft for which data are presented have a minimum ω_{n_d} above or about 1.0 rad/sec.

On this basis, a minimum basic requirement of $\omega_{n_d} = 0.4$ rad/sec has been specified. A more stringent requirement of $\omega_{n_d} = 1$ rad/sec has been placed on Class I and IV airplanes in Flight Phase Categories A and C since, when performing tasks such as weapons delivery or landing, it is clearly important that the aircraft be pointed the way it is going unless the pilot intentionally sideslips. Since the $\omega_{n_d} = 0.4$ rad/sec requirement could be unduly restrictive for large airplanes, Class III airplanes have been excepted from this requirement, subject to specific approval, if they meet several other requirements which would tend to become critical at low directional stiffness.

Since the data of Reference F44 (Figures 12 through 15) and Reference G11 (Figure 24) indicate a degradation in performance and pilot rating at high frequencies in the presence of turbulence, an indirect requirement on maximum ω_{n_d} has been specified through the parameter $\omega_{n_d}^2 |\phi/\beta|_d$. In addition, a qualitative requirement, which addresses the problem more comprehensively, has been specified in 3.3.2.1, Lateral-directional response to atmospheric disturbances.

The requirement that $\omega_{n_d} > 0$ with control surfaces fixed prescribes a short-term restoring tendency in yaw for the basic, unaugmented airframe, analogous to the control-surface-fixed maneuvering stability requirement of 3.2.2.2.

Effect of $\omega_{n_d}^2 \left|\frac{\phi}{\beta}\right|_d$

The criterion proposed by Reference F9, in which the value of $\zeta_d \omega_{n_d}$ required to maintain a given pilot rating is made a function of $\omega_{n_d}^2 \left|\frac{\phi}{\beta}\right|_d$ (Figure 26), has also been incorporated into the lateral-directional oscillatory requirements of Reference A1. It can be seen that $\omega_{n_d}^2 \left|\phi/\beta\right|_d$ is analogous to $\left|\ddot{\phi}/\beta\right|_d$. The data upon which the curve of Figure 26 is based are from Reference F76 and are presented in Figure 9. The primary limitations of these data are that few frequencies below 2 rad/sec and few damping ratios greater than 0.2 were examined, and that the pilot ratings were obtained from observation of the control-free response of the airplane following a rudder kick. In spite of these limitations, the curve of Figure 26 does fit the high $\omega_{n_d}^2 \left|\phi/\beta\right|_d$ data of References G7, F22 and F7 which are included in Figures 5 and 8. The parameter $\left|\phi/\beta\right|_d$ is explained in detail in Appendix VC.

The specific requirement of 3.3.1.1 that deals with $\omega_{n_d}^2 \left|\frac{\phi}{\beta}\right|_d$ is of the form $\Delta\zeta_d \omega_{n_d} = C\left(\omega_{n_d}^2 \left|\frac{\phi}{\beta}\right|_d - 20\right)$. The values of C are taken directly from Figure 26 and correspond to the $\Delta\zeta_d\omega_{n_d}$ that are required to maintain the various levels of acceptability. For the pilot rating used, Level 1 corresponds to a PR = 3.5, Level 2 corresponds to a PR $\approx$ 5, and Level 3 corresponds to a PR $\approx$ 7. The number 20 comes from analysis of the data, which reveals that adverse effects associated with large values of $\omega_{n_d}^2 \left|\frac{\phi}{\beta}\right|_d$ - such as high roll acceleration to side gusts - are not generally significant for values of $\omega_{n_d}^2 \left|\frac{\phi}{\beta}\right|_d < 20$.

For airplanes with large values of the parameters $\frac{\phi}{v_e}$ or $\omega_{n_d}^2 \left|\frac{\phi}{\beta}\right|_d$, response to atmospheric disturbances or to sideslip from any source becomes of major significance, and is influenced by many parameters other than Dutch roll damping. However, since Dutch roll damping is one of the significant parameters affecting controllability in atmospheric disturbances, and since heavy Dutch roll damping can often mask the effect of certain undesirable characteristics, it is believed that a ζ_d-(ϕ/v_e) type requirement is needed. Moreover, until the significance of, and interaction between, all significant variables is more fully understood and quantified, reliance will probably continue to be placed on heavy Dutch roll damping to improve flying qualities in the presence of turbulence.

Significance of $\measuredangle\, p/\beta$

Although the relative phasing of the Dutch roll in the p and β responses, $\measuredangle\, p/\beta$, has been used in the roll-sideslip coupling requirements as a measure of type of dihedral effect, no requirements have been placed on $\measuredangle\, p/\beta$. In reviewing the literature it was found that, although several researchers noted that the phasing of the Dutch roll may in some way have affected their results, no systematic study of the parameter was found.

Reference H8 discussed the parameter, but in connection with determining which "combinations of roll angle, sideslip angle, and sideslip resultant from an aileron deflection were prone to PIO's and ... which were not." This material is discussed in connection with the roll rate oscillation requirements, 3.3.2.2 and its subparagraphs.

Reference F76 also discussed the parameter in connection with an investigation of Dutch roll dynamics and found that:

> "About the only conclusion that could be drawn from these calculations, and of supplementary calculations in which various components of the actual oscillations were arbitrarily varied, was that phase angle has a strong influence on lateral acceleration/yaw angle, increasing it as much as four or five times as phasing of the Dutch roll goes from 0° to 180°. It may be that in addition to unnatural-feeling motions at 90° and 180°, the acceleration increase plays a part, but there are too few data to draw any conclusion."

This aspect of the data of Reference F76 (the significance of lateral acceleration) is discussed in the write-up on the sideslip limitation requirement.

A number of people recommend, as a design requirement, $C_{\ell_\beta} < 0$. It was felt that this is desirable, but not sufficiently substantiated to incorporate in Reference A1. Also it is not in the form of a performance requirement.

<u>Concluding Comments</u>

It is obvious that a great deal more data are needed to rigorously define the areas discussed. Until such data become available, it is believed that the lateral-directional oscillatory requirements, and other requirements of Reference A1 that have implications on lateral-directional oscillatory characteristics, will adequately address the problems associated with low directional stiffness.

Comparison of these requirements with the requirements proposed by Reference A14 in Figure 23 indicates that, although the available data were treated in a somewhat different manner in the two studies, the resulting proposed requirements are very similar in many respects.

The Level 3 boundaries are more stringent than indicated by the presented data, but since in Level 3 operation the airplane may have several serious flying qualities deficiencies, the airplane should be dynamically stable and have some minimum directional stiffness.

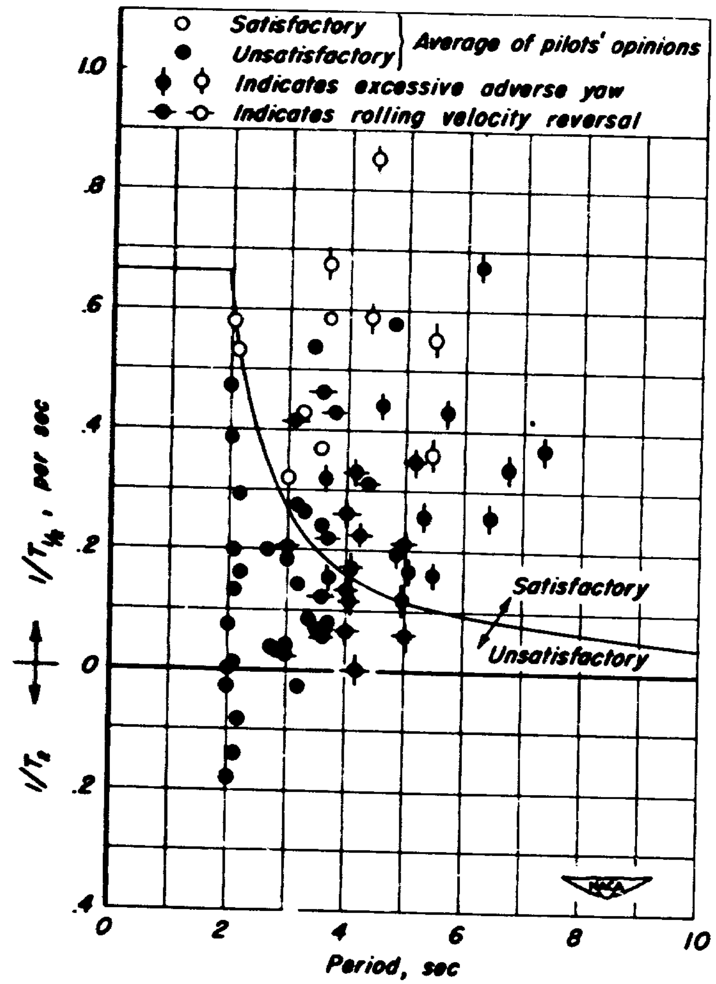


Figure 1 (3.3.1.1)
A COMPARISON OF PILOTS' OPINION WITH LATERAL OSCILLATORY REQUIREMENTS OF REFERENCE A18. (FROM REFERENCE F51)

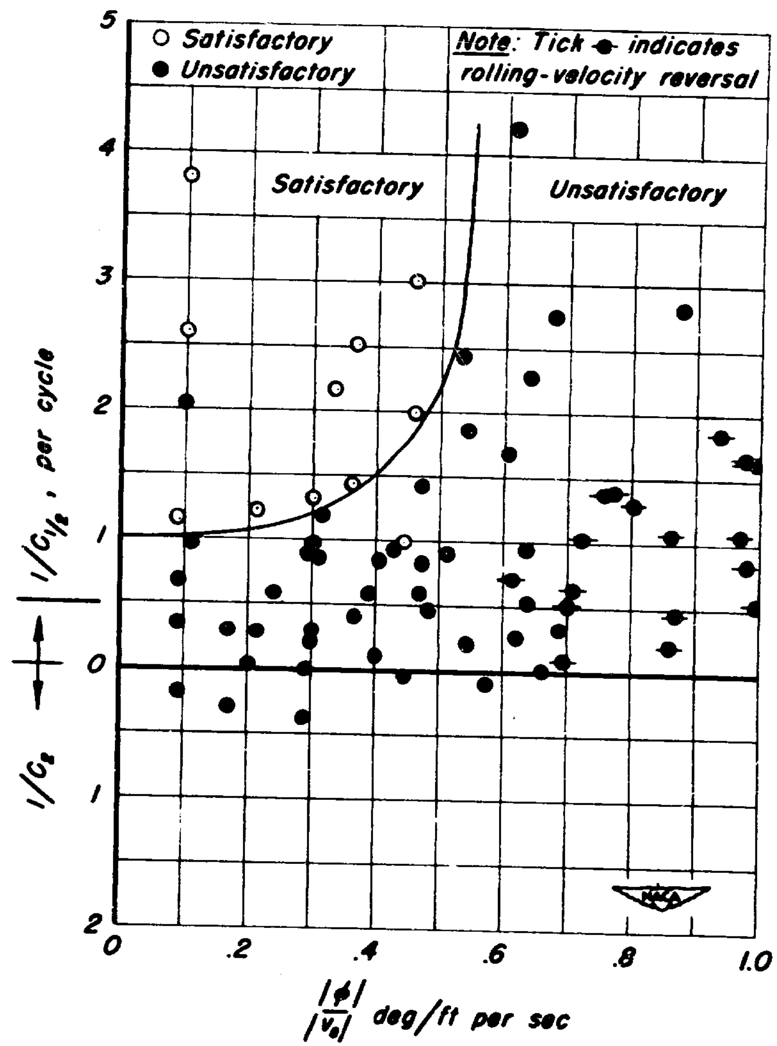


Figure 2 (3.3.1.1)
BOUNDARY BETWEEN SATISFACTORY AND UNSATISFACTORY LATERAL OSCILLATORY CHARACTERISTICS. (FROM REFERENCE F51)

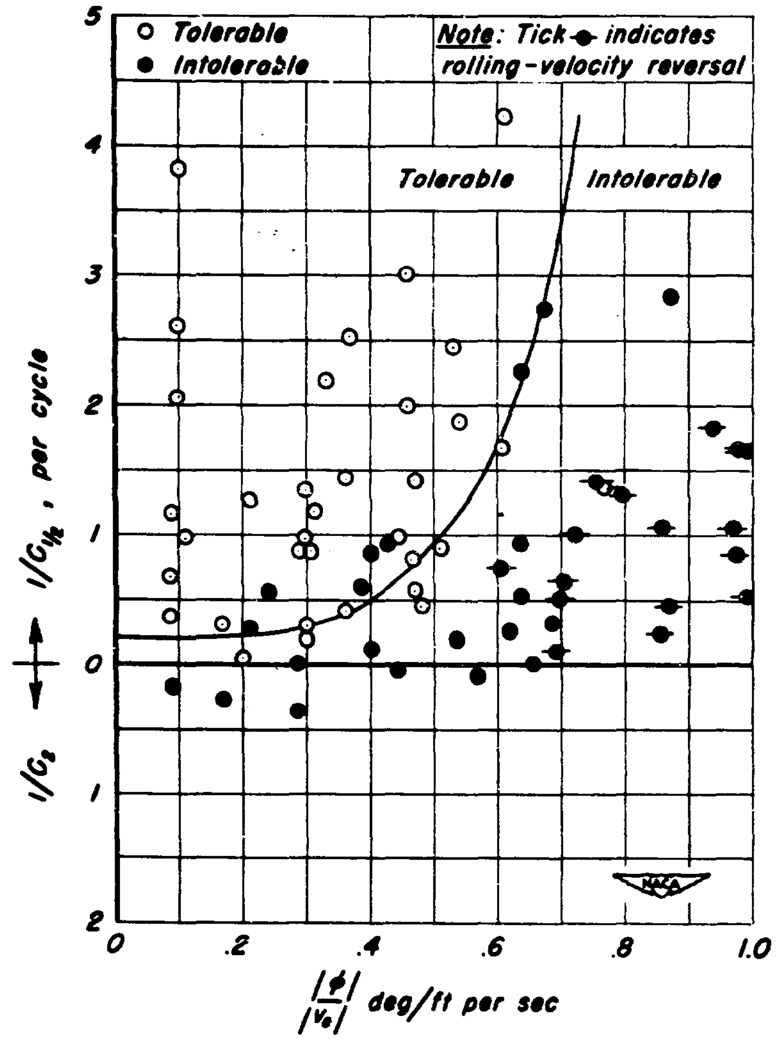


Figure 3 (3.3.1.1)
BOUNDARY BETWEEN TOLERABLE AND INTOLERABLE LATERAL OSCILLATORY CHARACTERISTICS. (FROM REFERENCE F51)

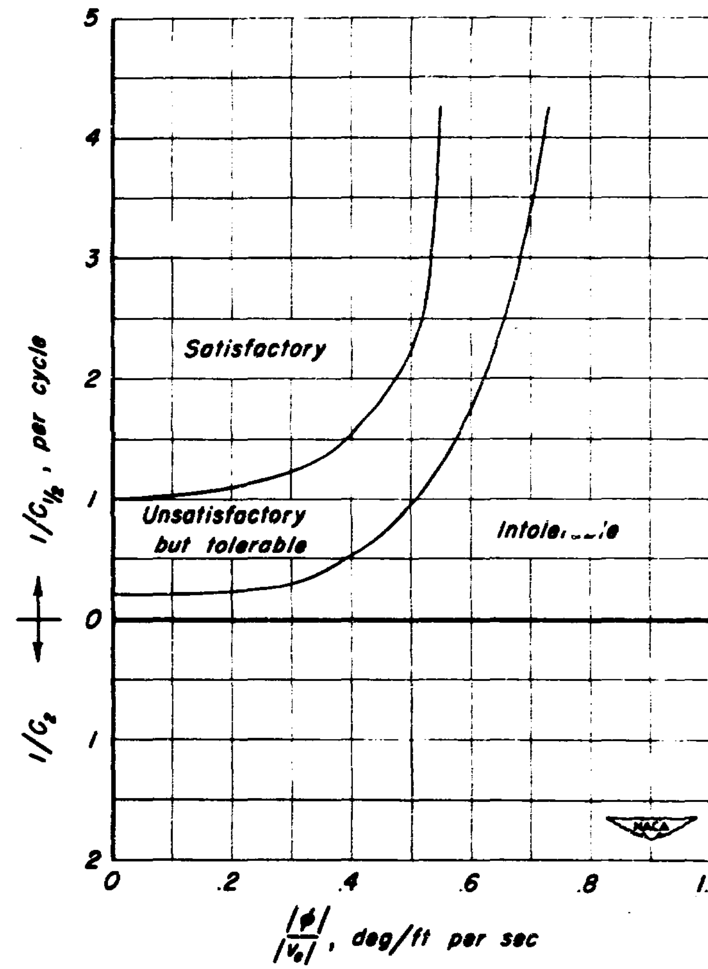


Figure 4 (3.3.1.1)
PROPOSED TENTATIVE BOUNDARY BETWEEN SATISFACTORY AND UNSATISFACTORY AND BETWEEN TOLERABLE AND INTOLERABLE LATERAL OSCILLATORY CHARACTERISTICS. (FROM REFERENCE F51)

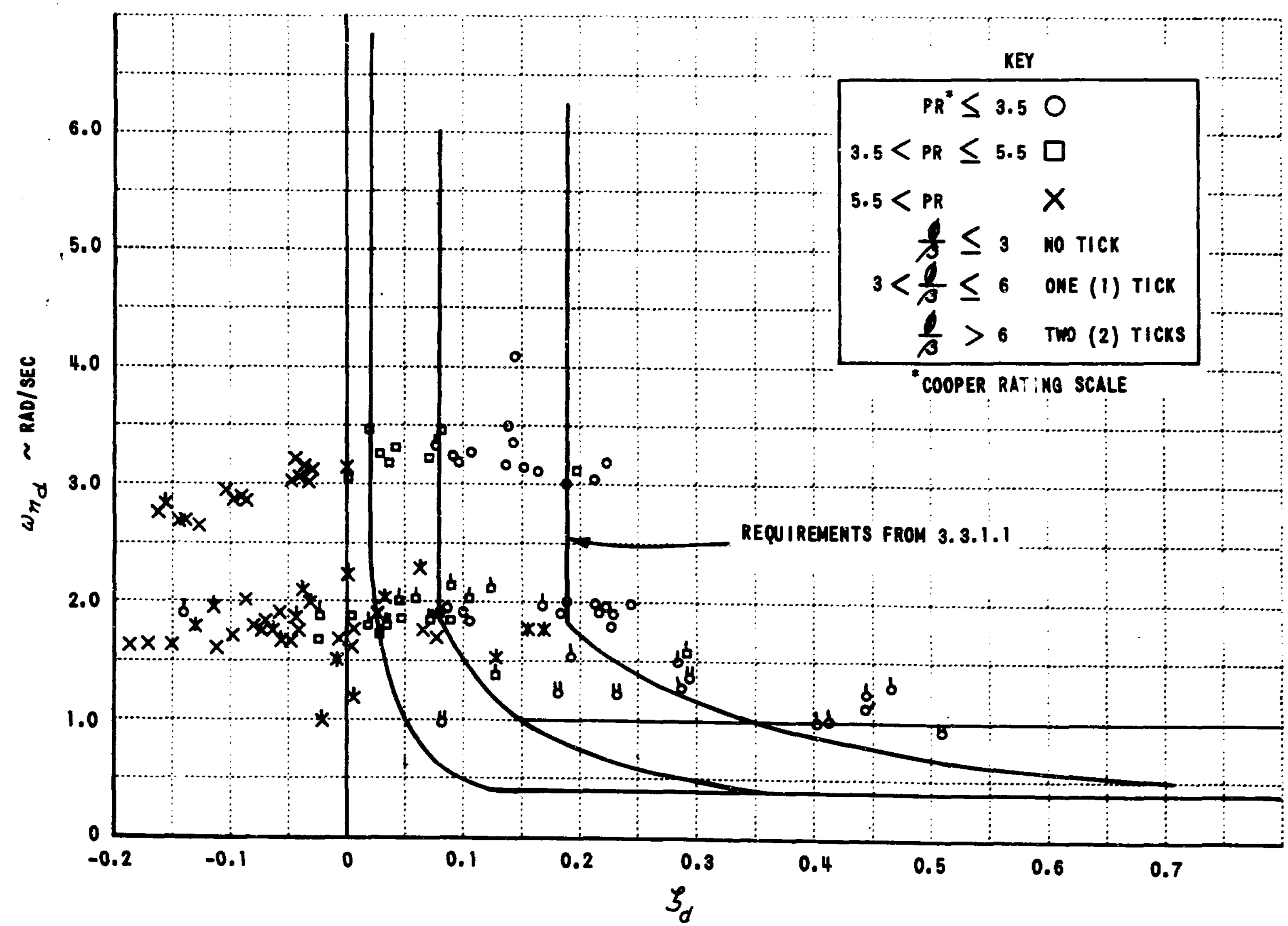


Figure 5 (3.3.1.1)
DUTCH ROLL DATA (FROM REFERENCES G7 AND F21)

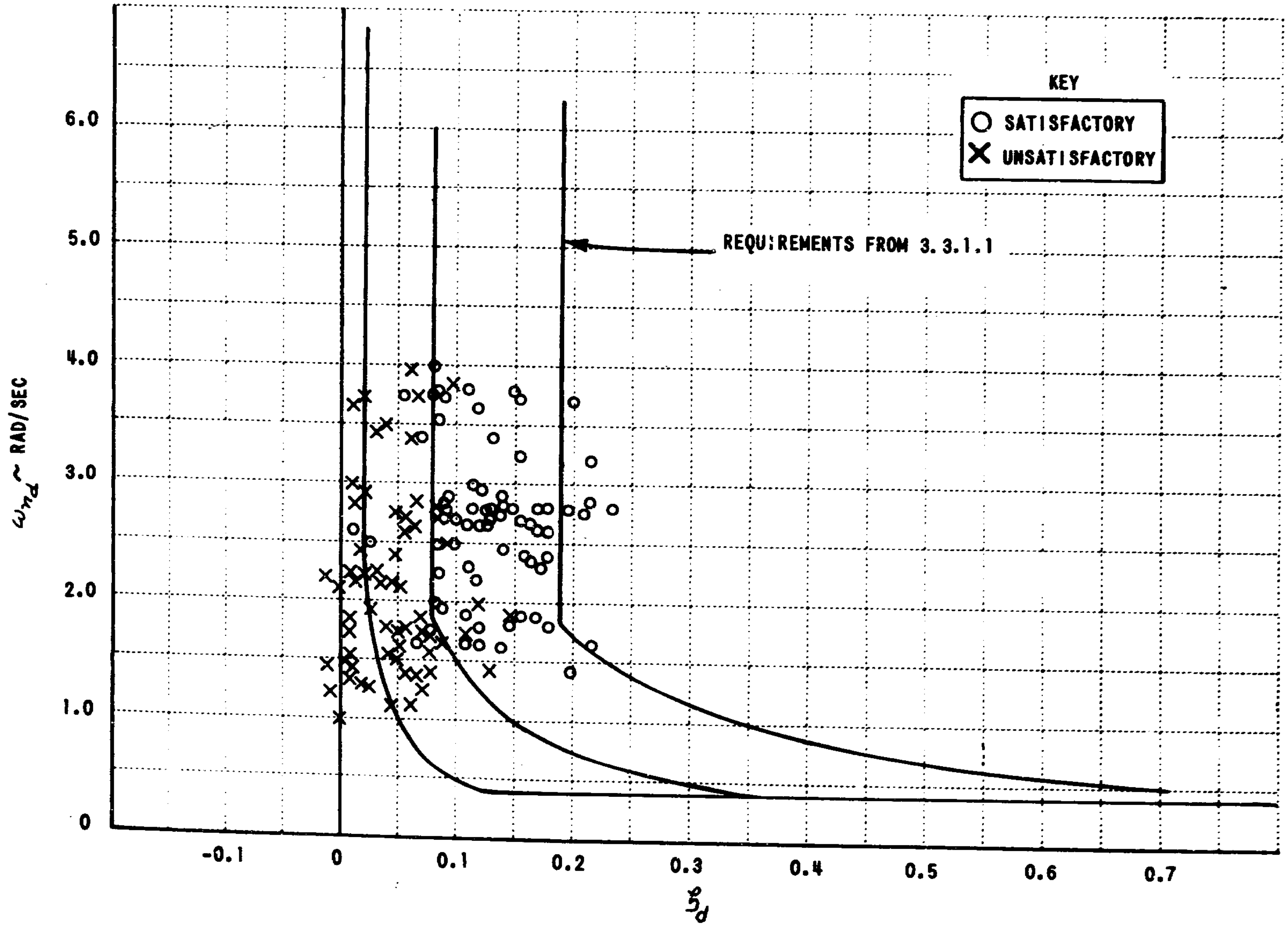


Figure 6 (3.3.1.1)
DUTCH ROLL DATA (FROM REFERENCE F55)

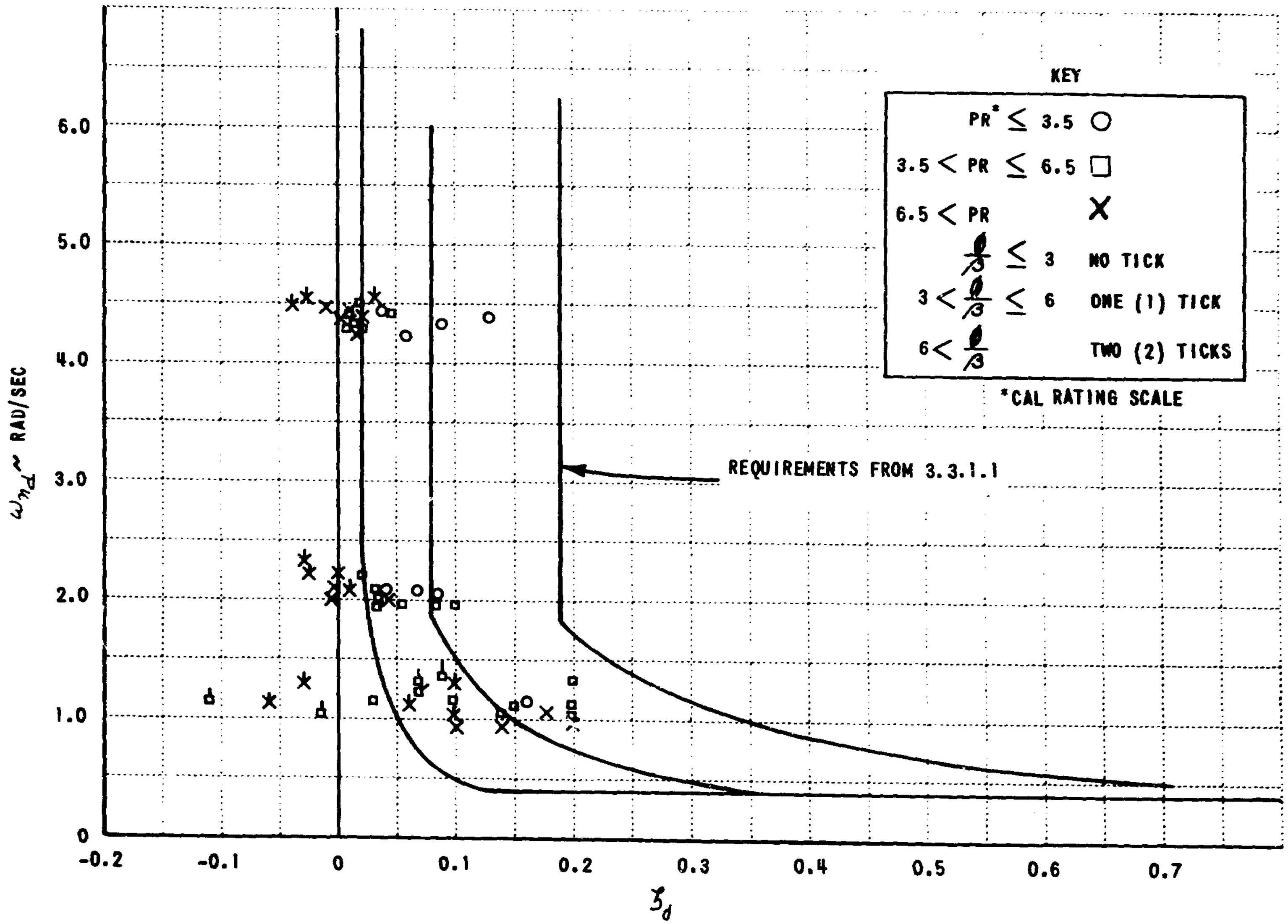


Figure 7 (3.3.1.1)
DUTCH ROLL DATA (FROM REFERENCE B44)

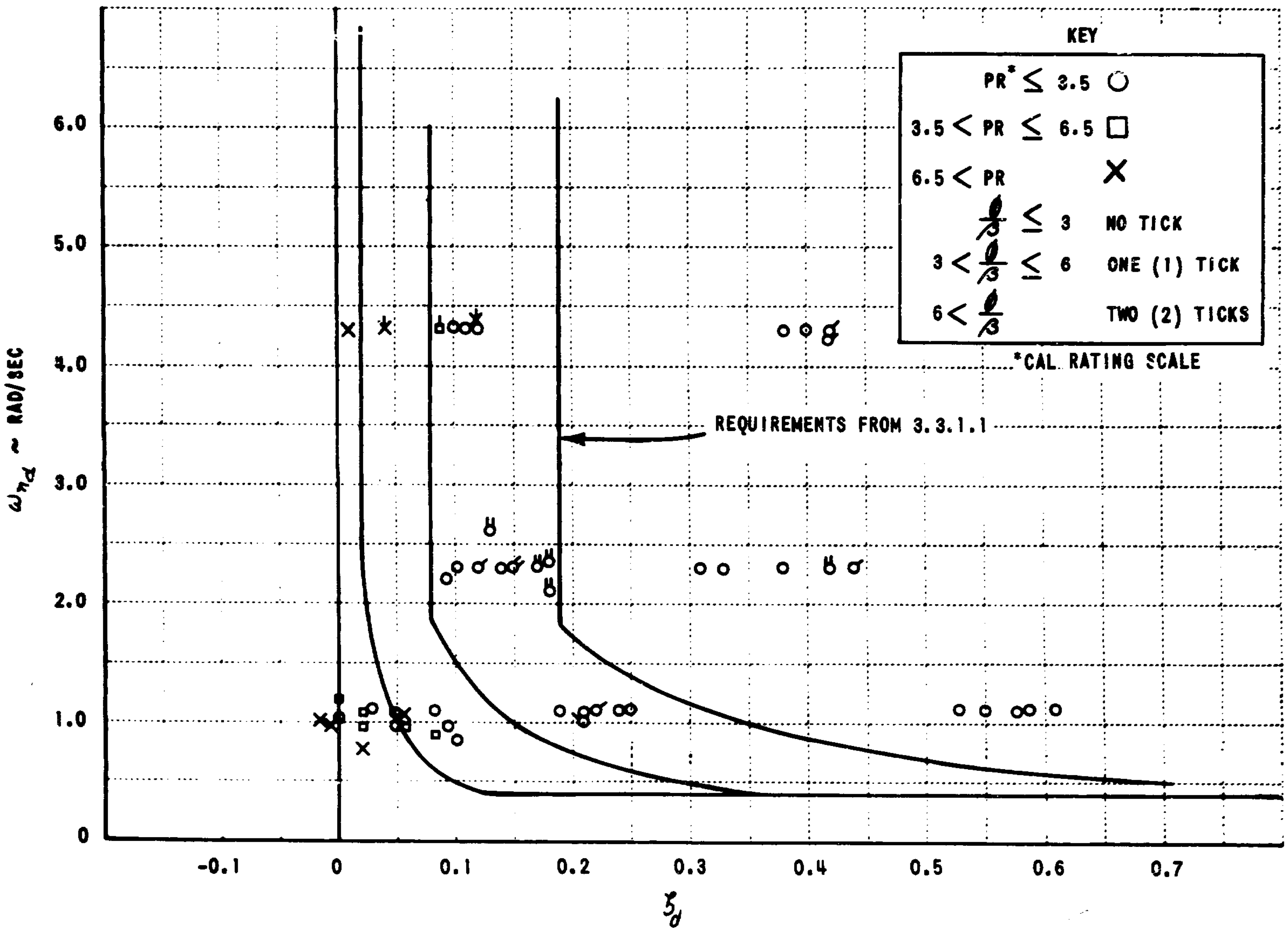


Figure 8 (3.3.1.1)
DUTCH ROLL DATA (FROM REFERENCES F5, F7 AND F22)

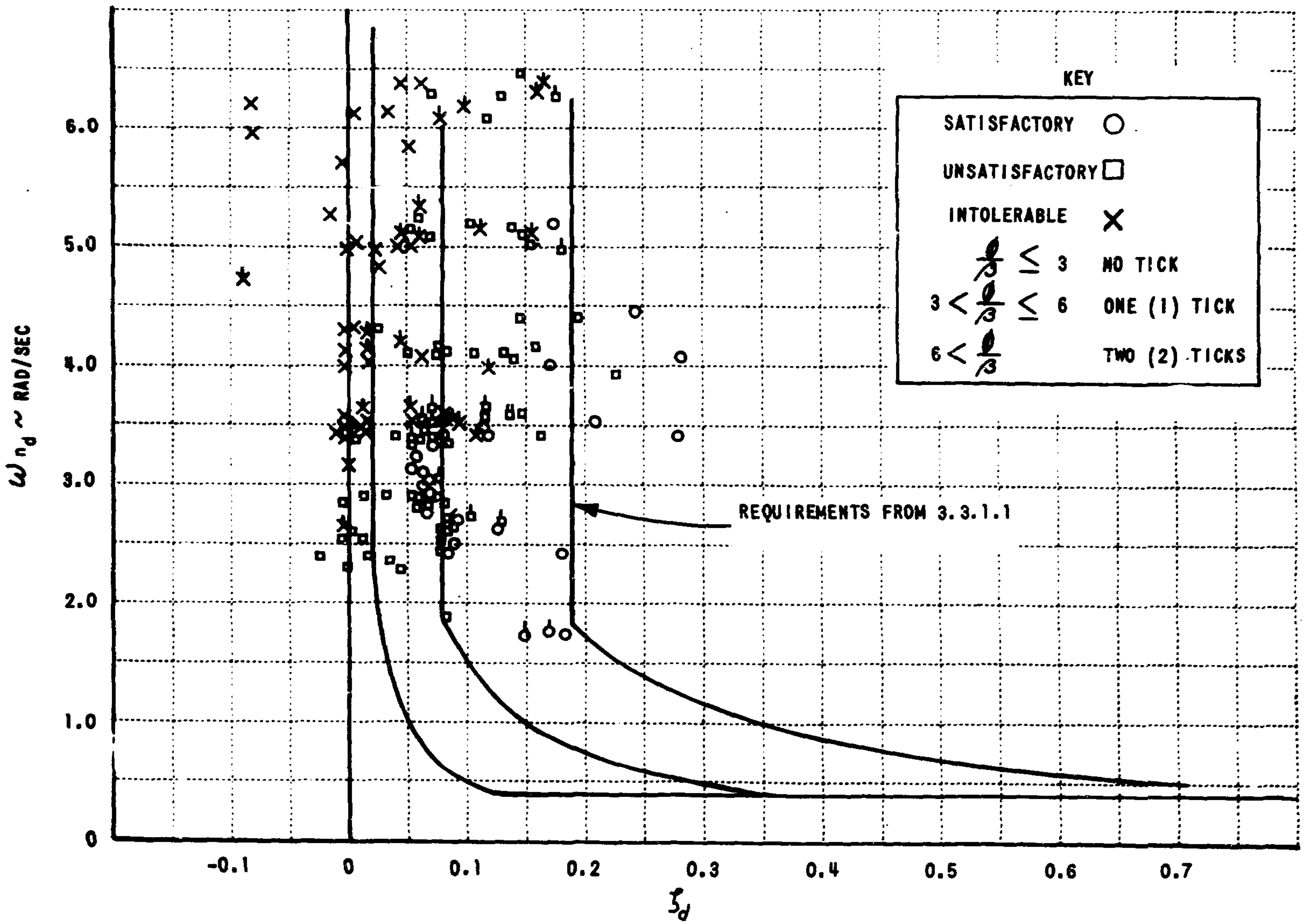


Figure 9 (3.3.1.1)
DUTCH ROLL DATA (FROM REFERENCE F76)

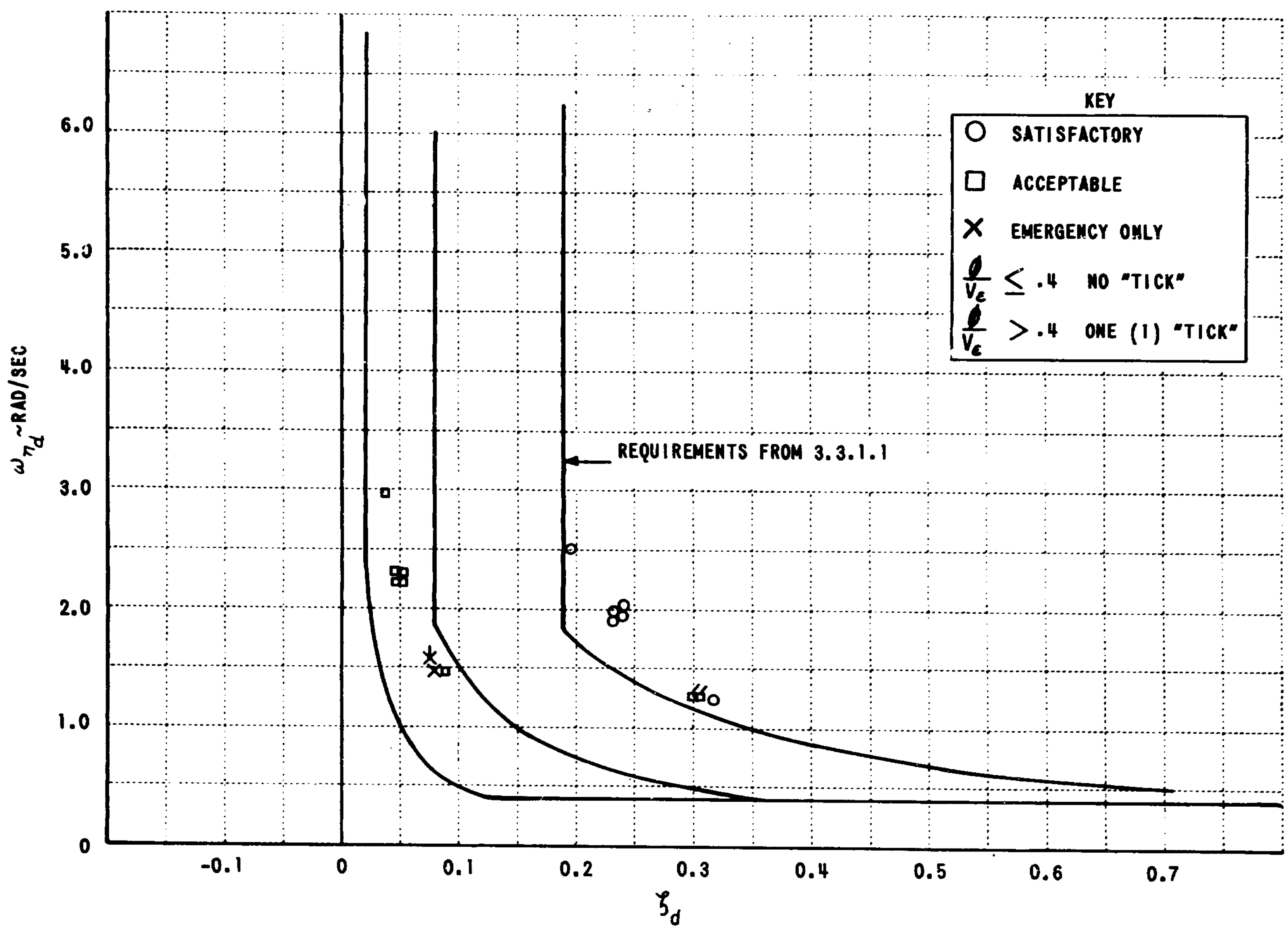


Figure 10 (3.3.1.1)
DUTCH ROLL DATA (FROM REFERENCE F75)

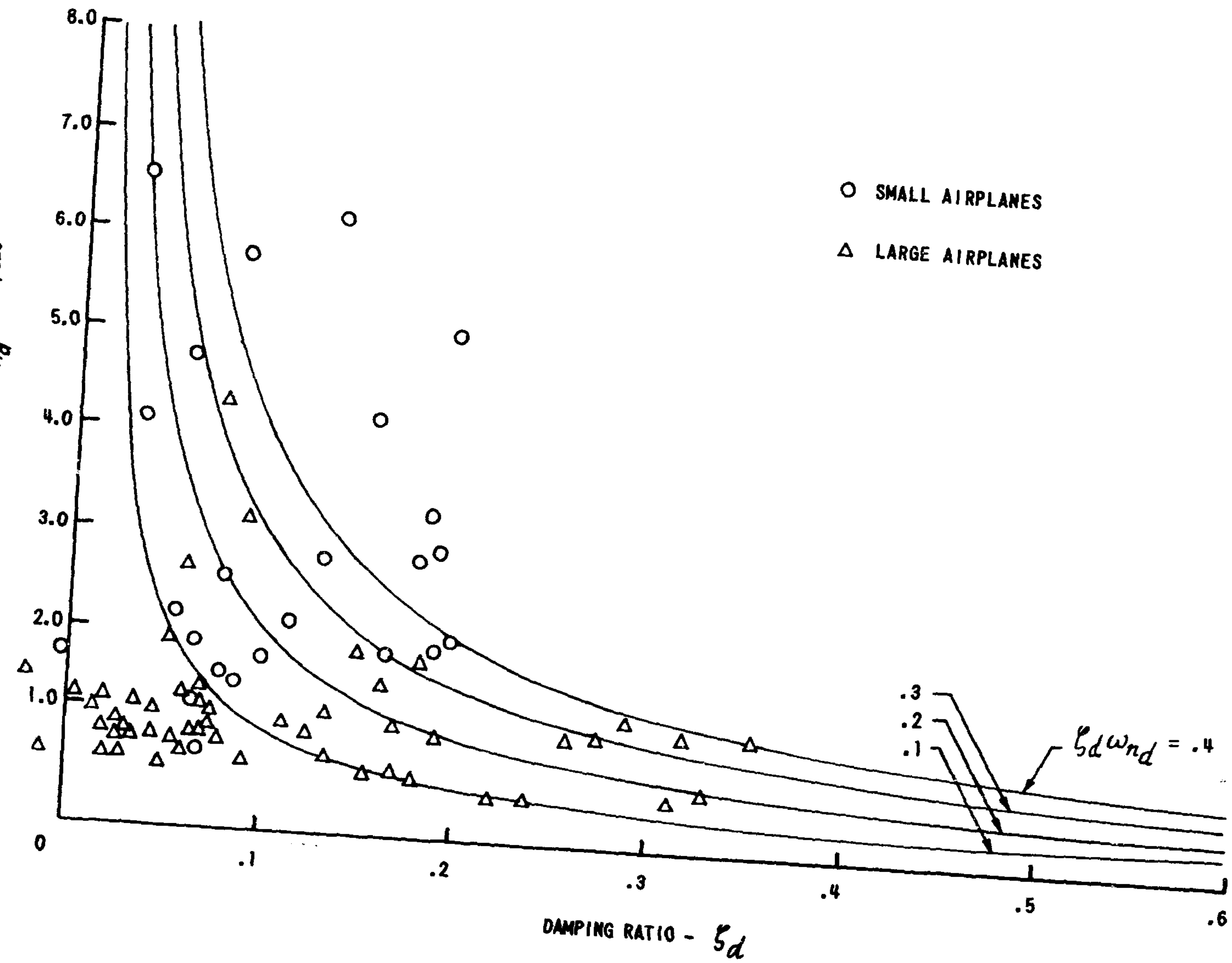


Figure 11 (3.3.1.1)
DUTCH ROLL DATA ON EXISTING AIRPLANES (FROM REFERENCE A4)

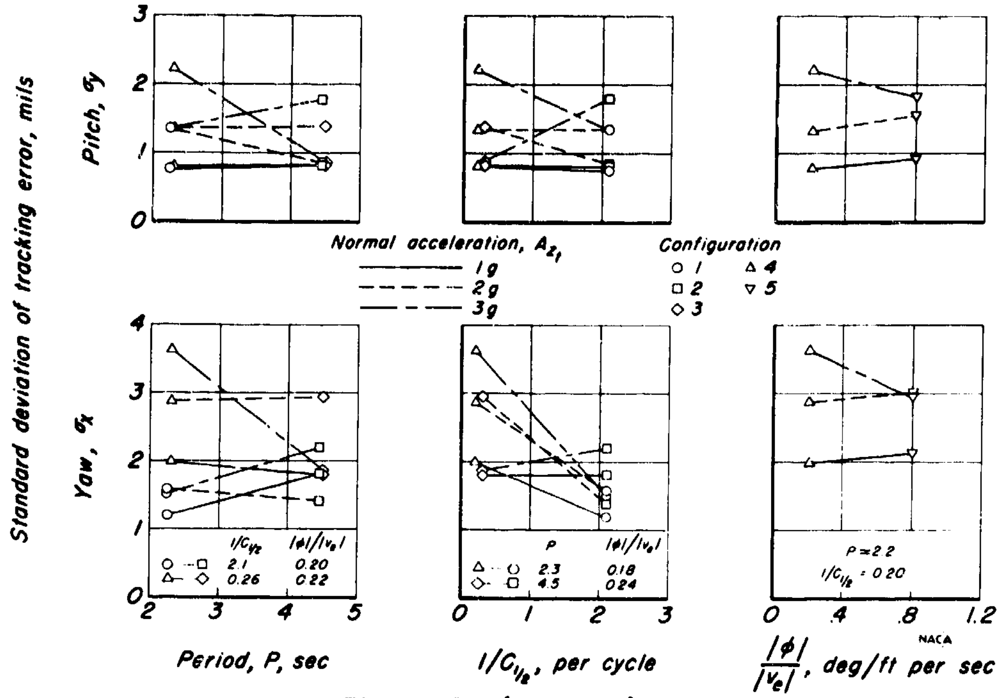


Figure 12 (3.3.1.1)
PILOT A

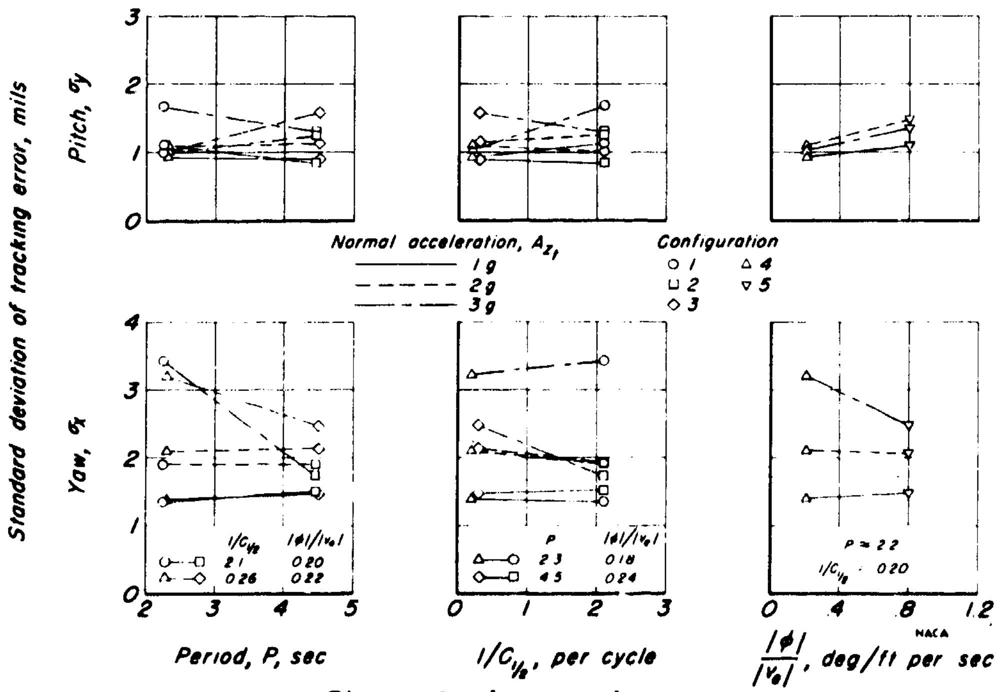


Figure 13 (3.3.1.1)
PILOT B

VARIATION OF STANDARD DEVIATION OF TRACKING ERROR WITH LATERAL OSCILLATORY CHARACTERISTICS IN STEADY-STRAIGHT AND STEADY-TURNING FLIGHT. SMOOTH AIR. (FROM REFERENCE F44)

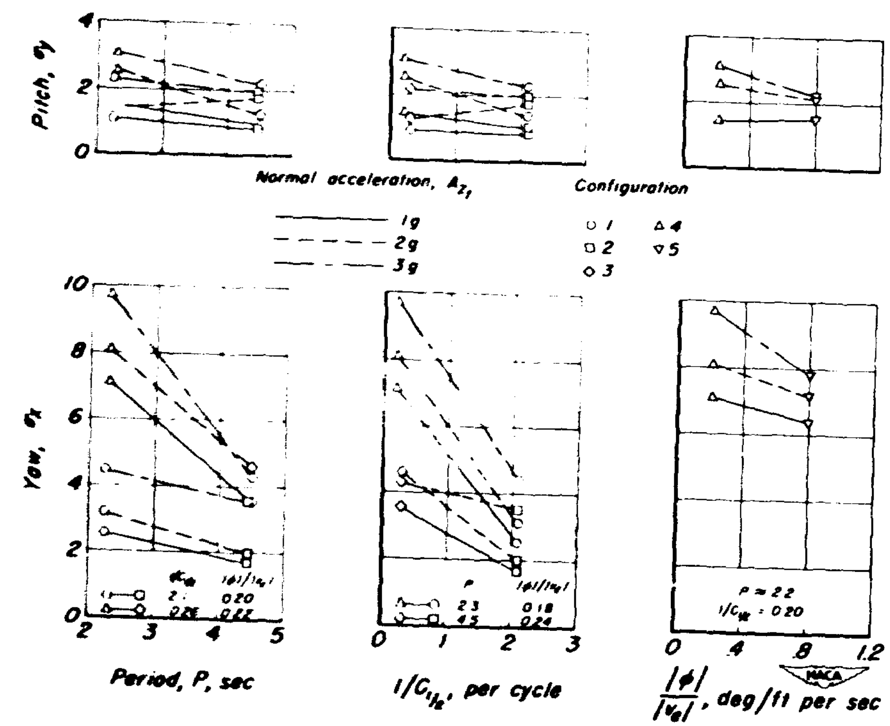


Figure 14 (3.3.1.1)

VARIATION OF STANDARD DEVIATION OF TRACKING ERROR WITH LATERAL-OSCILLATORY CHARACTERISTICS IN STEADY-STRAIGHT AND STEADY-TURNING FLIGHT. SIMULATED ROUGH AIR; PILOT A. (FROM REFERENCE F44)

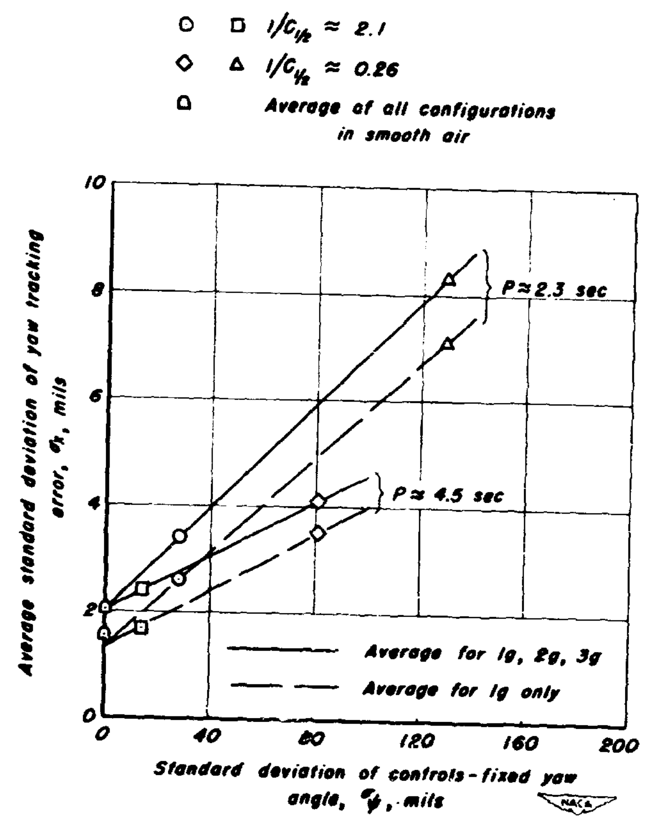


Figure 15 (3.3.1.1)

VARIATION OF AVERAGE STANDARD DEVIATION OF YAW TRACKING ERROR WITH STANDARD DEVIATION OF CONTROLS-FIXED YAW ANGLE IN SIMULATED ROUGH AIR. CONFIGURATIONS 1, 2, 3, AND 4; PILOT A. (FROM REFERENCE F44)

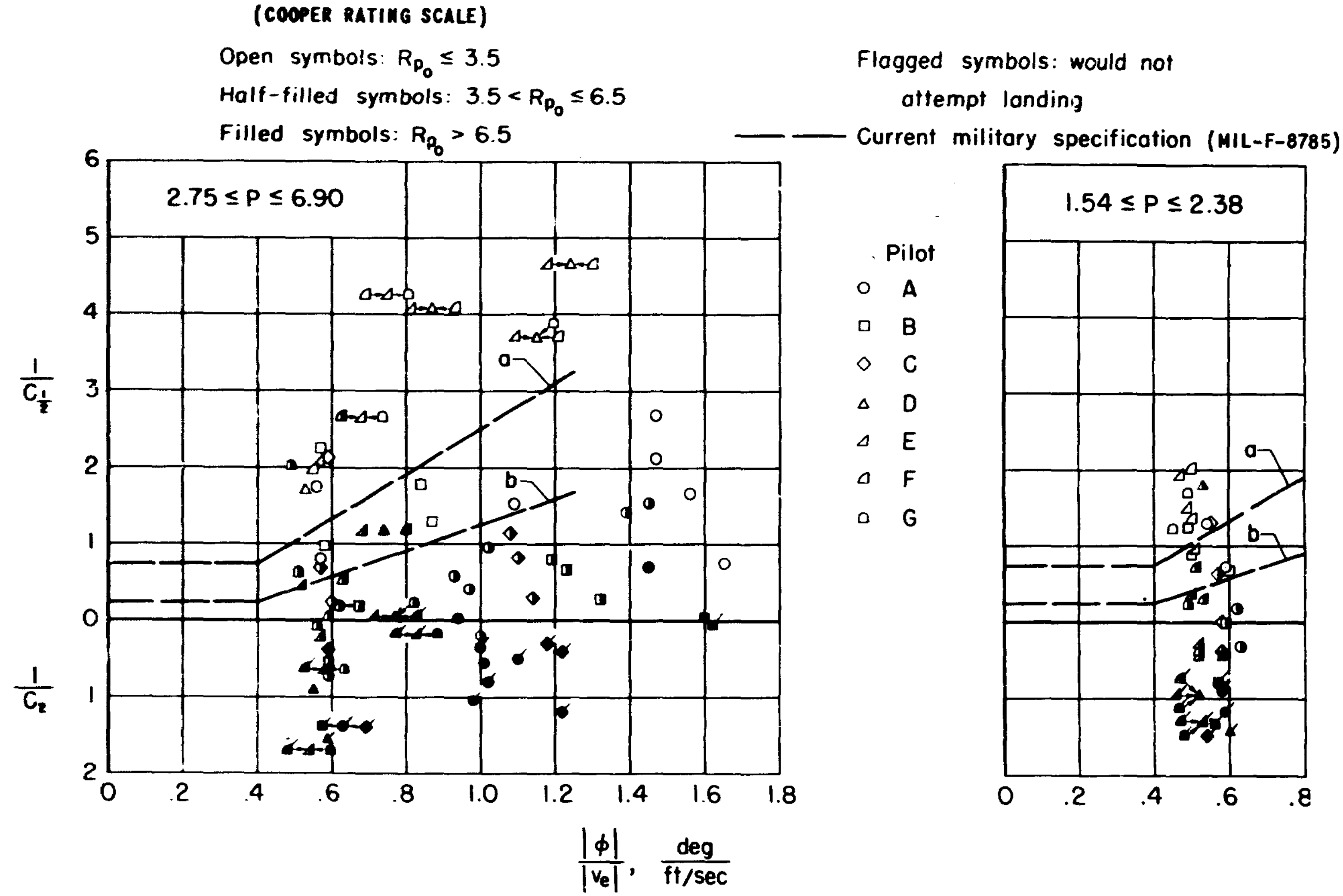


Figure 16 (3.3.1.1.)

LATERAL OSCILLATORY CHARACTERISTICS AND OVER-ALL PILOT RATINGS OF THE CONFIGURATIONS FLOWN. (FROM REFERENCE G7)

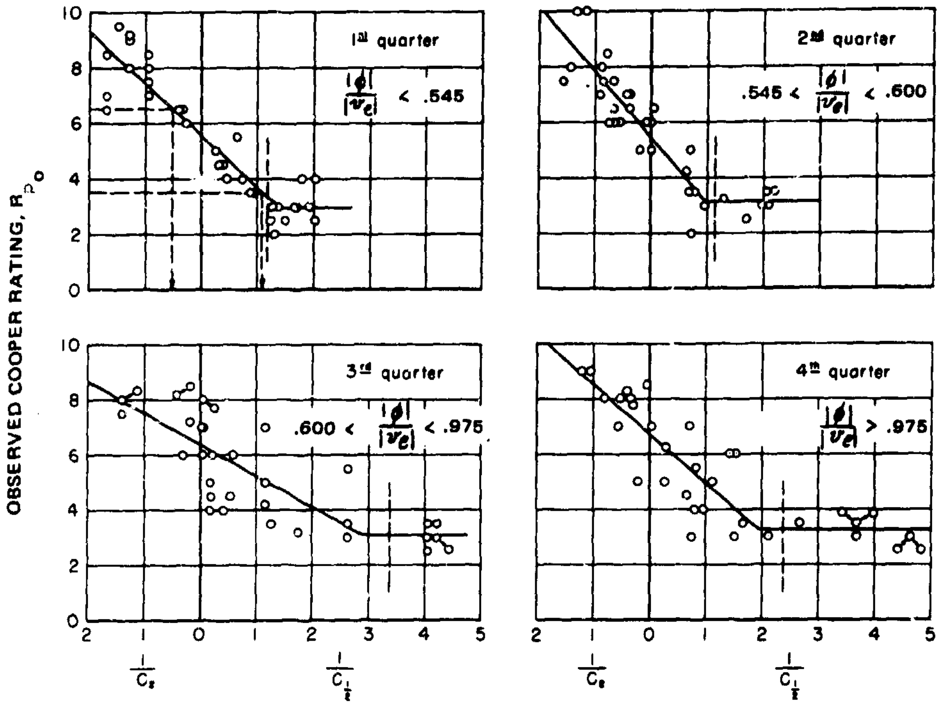


Figure 17 (3.3.1.1)

MODIFIED LINEAR LEAST-SQUARES FITS OF VARIATIONS OF PILOT RATING WITH DAMPING IN THE FOUR DATA GROUPS; DAMPING EXPRESSED AS $1/C_{1/2}$. (FROM REFERENCE G7)

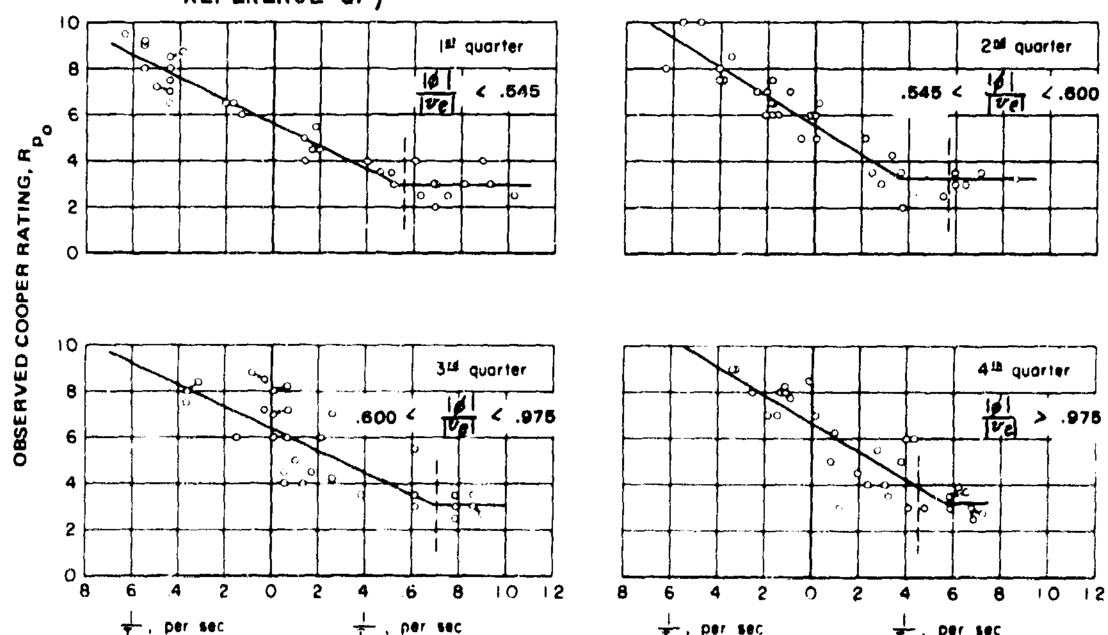


Figure 18 (3.3.1.1)

MODIFIED LINEAR LEAST SQUARES FITS OF VARIATIONS OF PILOT RATING WITH DAMPING IN THE FOUR DATA GROUPS; DAMPING EXPRESSED AS $1/T_{1/2}$. (FROM REFERENCE G7)

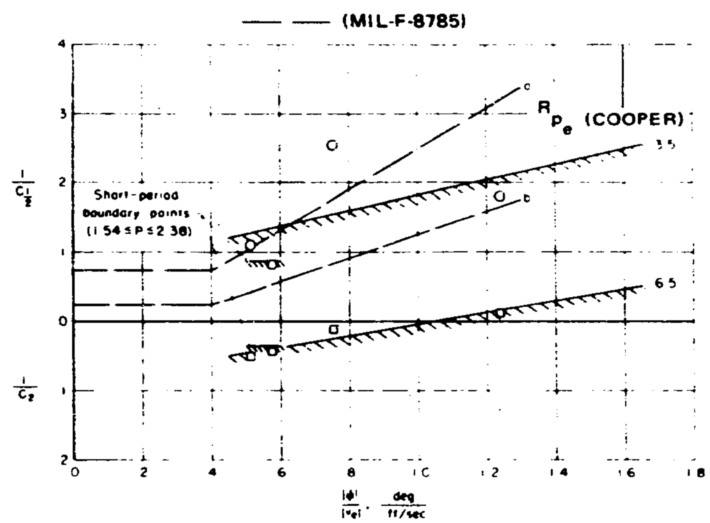


Figure 19 (3.3.1.1)

COMPARISON OF LATERAL OSCILLATORY DAMPING BOUNDARIES (IN TERMS OF $1/C_{1/2}$) DETERMINED FROM PILOT OPINIONS IN THE PRESENT STUDY WITH MIL-F-8785 (FROM REFERENCE G7)

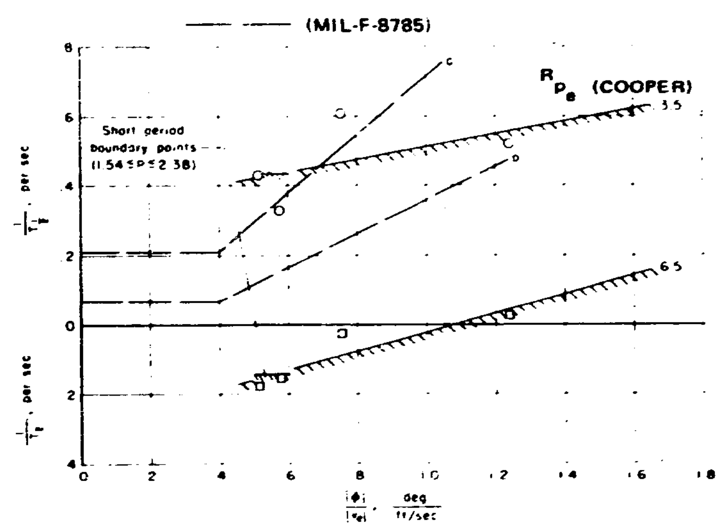


Figure 20 (3.3.1.1)

COMPARISON OF LATERAL OSCILLATORY DAMPING BOUNDARIES (IN THE TERMS OF $1/T_{1/2}$) DETERMINED FROM PILOT OPINIONS IN THE PRESENT STUDY WITH MIL-F-8785 (FROM REFERENCE G7)

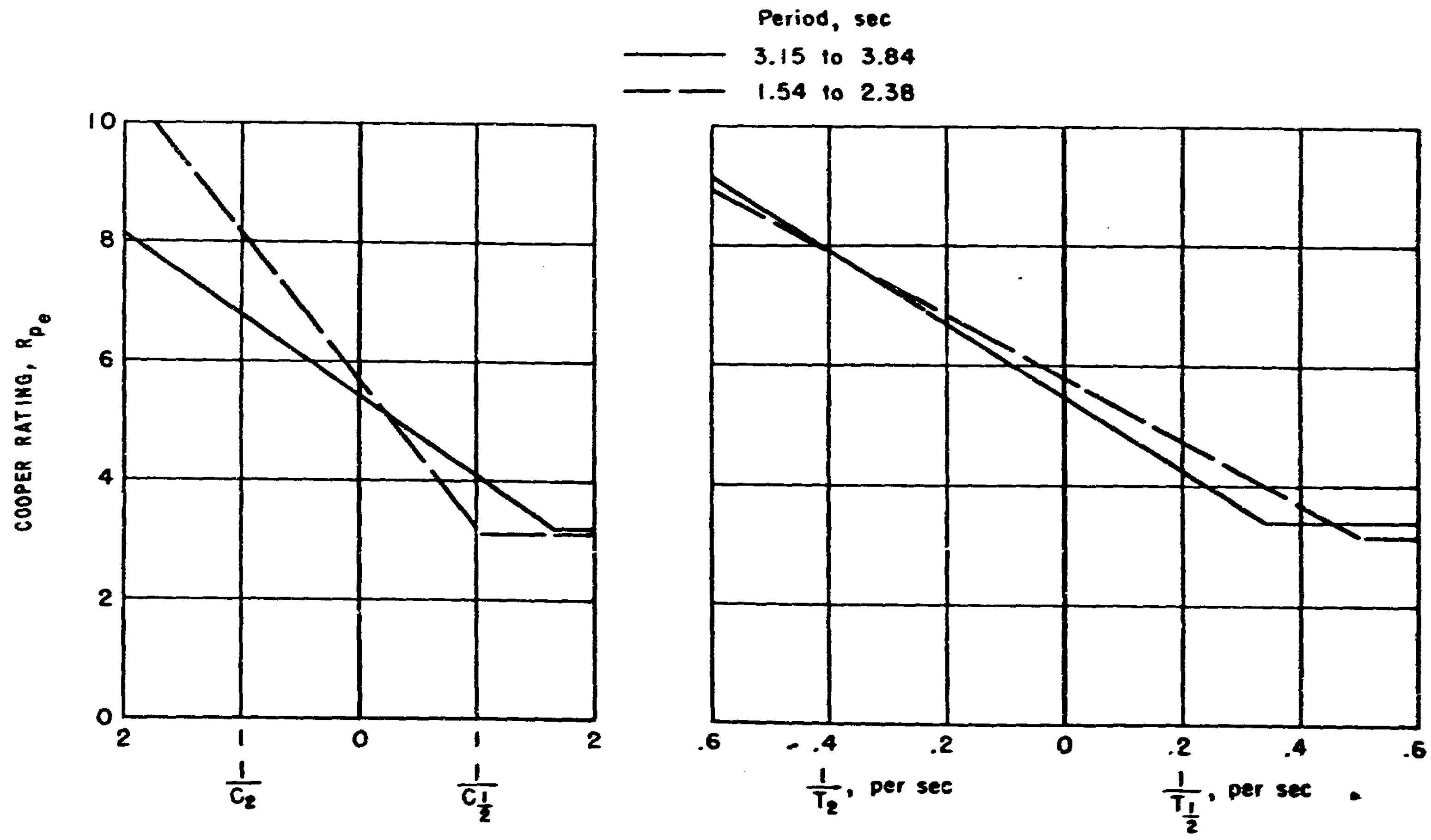


Figure 21 (3.3.1.1)
MODIFIED LINEAR LEAST-SQUARES VARIATIONS OF PILOT RATING WITH DAMPING EXPRESSED AS $1/C_{1/2}$ AND $1/T_{1/2}$; $0.45 \le |\phi_i| / |v_e| \le 0.60 \ \frac{\text{deg}}{\text{ft/sec}}$ (FROM REFERENCE G7)

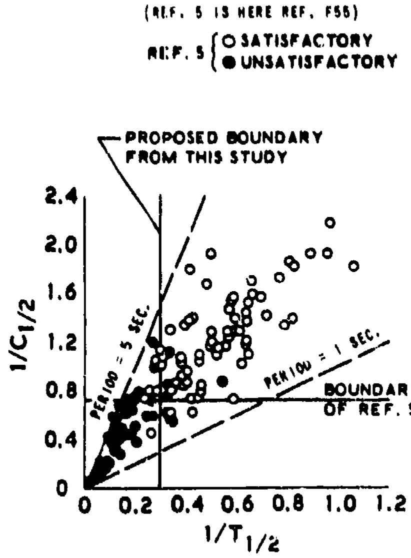


Figure 22 (3.3.1.1)
COMPARISON OF PILOT OPINION
WITH MIL-F-8785 LATERAL-DIRECTIONAL
OSCILLATORY REQUIREMENT.
(FROM REFERENCE F75)

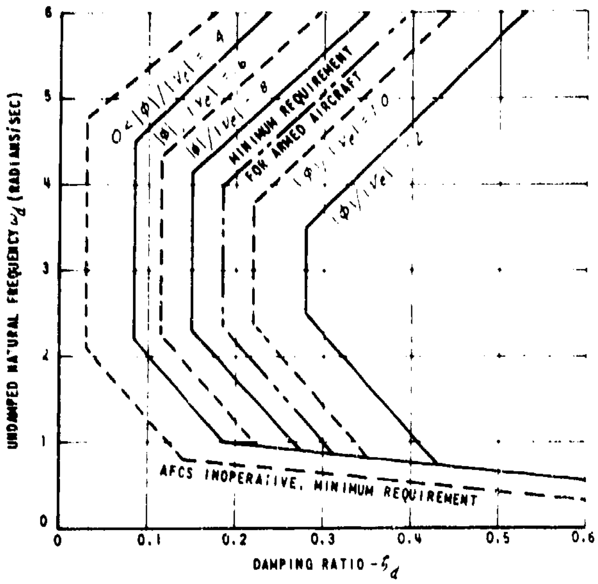


Figure 23 (3.3.1.1)
PROPOSED LATERAL-DIRECTIONAL
OSCILLATION REQUIREMENTS
(FROM REFERENCE A14)

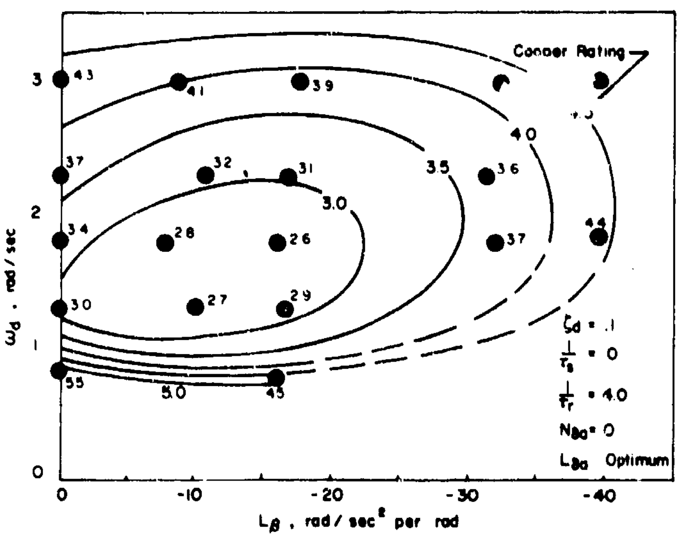


Figure 24 (3.3.1.1)
PILOT OPINION CONTOURS (ω_d vs L_β) LOW DUTCH ROLL
DAMPING (FROM REFERENCE G11)

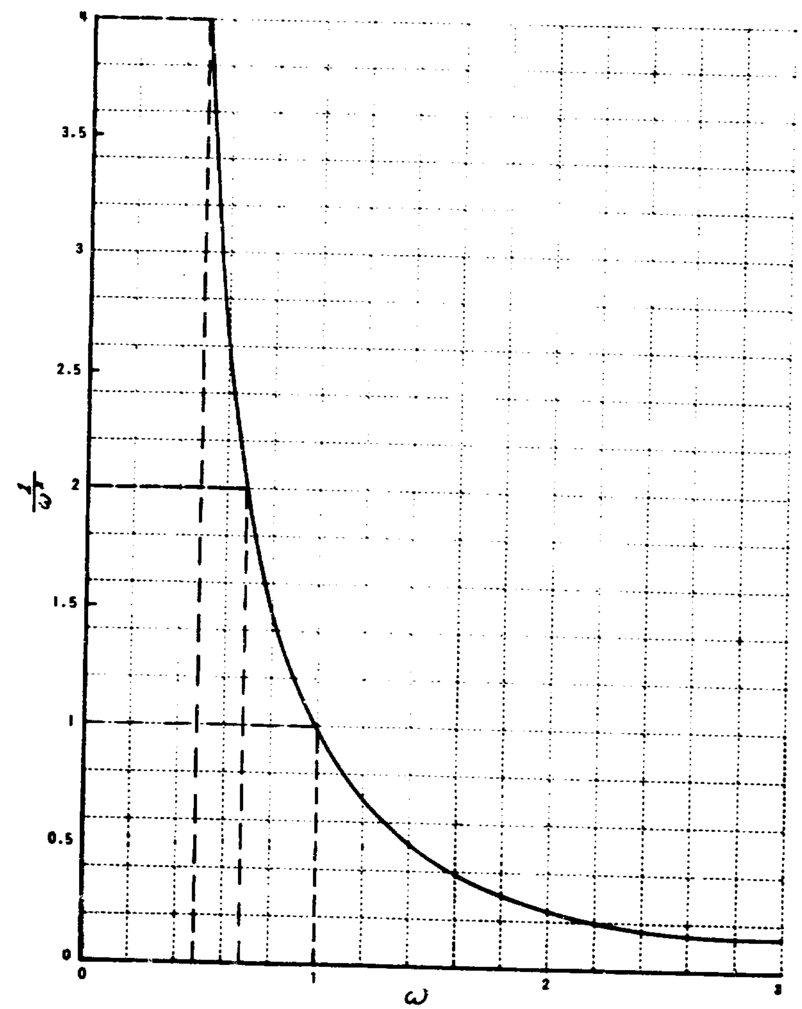


Figure 25 (3.3.1.1)

$1/\omega^2$ VS. ω

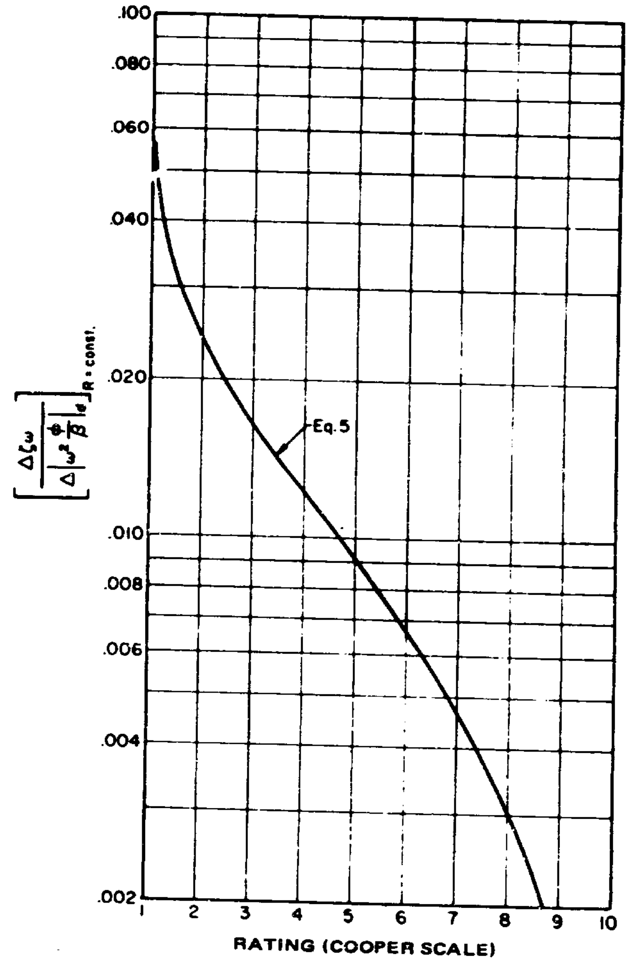


Figure 26 (3.3.1.1)

$\frac{\Delta\zeta\omega}{\Delta\left|\omega^2\frac{\phi}{\beta}\right|_d}$ REQUIRED TO MAINTAIN A GIVEN BASIC RATING (FROM REFERENCE F9)

3.3.1.2 ROLL MODE

REQUIREMENT

3.3.1.2 Roll mode. The roll-mode time constant, τ_R , shall be no greater than the appropriate value in table VII.

TABLE VII. Maximum Roll-Mode Time Constant

Flight Phase Category	Class	Level 1	Level 2	Level 3
A	I, IV	1.0	1.4	10
	II, III	1.4	3.0	
B	All	1.4	3.0	
C	I, II-C, IV	1.0	1.4	
	II-L, III	1.4	3.0	

RELATED MIL-F-8785 PARAGRAPHS

3.4.16.2

DISCUSSION

General

This requirement, which is directed at precision of control, fills a void in MIL-F-8785 since at present there is no direct reference to roll damping or the "shape" of the initial roll rate response. In the lateral control section of MIL-F-8785, paragraph 3.4.16.2, a requirement on roll acceleration is stated in terms of time to reach peak rate of roll; however, even this indirect requirement is only applied to Class II-L airplanes of MIL-F-8785.

There are considerable data to show that pilot rating is a function of roll damping, for example Figure 1 (from Reference F8). Roll damping is generally expressed in terms of the first-order roll mode time constant, τ_R, of the roll rate response following a step rolling moment input. Therefore, a direct requirement on τ_R has been specified. This is consistent with the approach taken by Reference A14, which also proposed direct requirements on τ_R .

The roll damping interacts with many other parameters, such as roll performance, roll sensitivity, Dutch roll characteristics and control system dynamics; so to determine the shape of the roll rate response and hence the flying qualities, interaction effects must also be considered. For example, although curves of constant level of flying qualities are often presented on plots of some measure of roll performance, such as $L_{\delta_a} \delta_{a_{max}}$ vs. τ_R , it is usually possible, by careful selection of the roll performance parameter, to isolate the effects of roll performance and roll damping on flying qualities. In this way, the individual effects can be identified and treated independently. Where interaction cannot be so isolated, as for example the coupling of the Dutch roll through aileron yaw, requirements covering known interaction effects are explicitly stated.

The Reference A1 criteria go a long way toward ensuring proper rolling response. But still they are based on a first-order roll-rate response, and so do not adequately cover all the many coupling and interaction effects. A great deal more work is required in this area.

For a step aileron input, ideally a roll damper will not affect initial rolling acceleration; but it will cause the steady roll rate to be reached sooner. As a result, the roll damper may reduce roll performance in the specified terms (3.3.4) of bank angle in a given time. One solution that avoids increasing roll control power is to wash out the roll damper as a function of roll control position. This has proven satisfactory on at least one current fighter airplane, but we cannot attest to its general suitability.

The numerical criteria, given in Table VII of Reference A1 for the three levels of flying qualities, came from consideration of data from References A14, F8, G10, F1, B96, F30, B48, F22, B39, F5, F12 and from discussions with aircraft manufacturers. Some measurement techniques for obtaining τ_R are given in Appendix VB.

Level 1 Requirements

The starting point for specification of the criteria was the recommendation pertaining to roll mode time constant given in References A14 and F8. Both references report on extensive surveys of roll flying qualities and so were directly applicable to this effort. Reference A14 proposes a maximum $\tau_R = 1.3$ seconds for Class IV airplanes and $\tau_R = 1.5$ seconds for all other classes (Figure 2). From theoretical considerations and from data analysis, Reference F8 concluded that, "The maximum value of τ_R considered satisfactory is about 1.3 to 1.5; and there is no strong evidence in existing data or theory for allowing this value to increase with airplane size or mission." While there is still no strong evidence to indicate that the requirements can be relaxed, several recent reports on in-flight evaluations (References G10, F1, and B96) and discussions with aircraft manufacturers indicate that for Class I and IV airplanes performing precision tasks, even lower values of τ_R are required to obtain satisfactory flying qualities.

Reference G10 (Figures 3, 4 and 5) shows that maximum satisfactory τ_R for fighter aircraft for a carrier approach is approximately 1 second.

Reference F1 (Figure 6) shows that with a τ_R = 1.3 seconds, the best pilot rating obtained was 5 and in conclusion stated, "Because of the roll control difficulties the pilot experienced with the long roll mode time constant configuration, it was concluded that a roll mode time constant of 1.3 seconds or greater is unsatisfactory for a fighter mission." Reference B96 indicates, from consideration of time required to reach maximum roll rate, that Class I and small Class II aircraft require reasonably short roll mode time constants as well.

The data of Reference F30 (Figures 7 through 11) have been widely referenced and interpreted, as for example in References A14 and F8. It should be noted, however, that the in-flight evaluations in Reference F30 were all for τ_R less than 0.8 seconds (Figure 10) and any conclusions about roll mode time constants longer than 0.8 seconds would be based on the ground simulation data only. In general, the in-flight ratings of Reference F30 were worse than for the single-degree-of-freedom ground simulation ratings (Figure 11). This indicates that the presented one-degree-of-freedom data (Figure 9) may be a little optimistic. Finally, one prominent manufacturer of fighter aircraft stated that fighter aircraft should have a τ_R = 0.6 to 0.8 seconds.

Since, in general, a knee occurs in most of the data at approximately τ_R = 1 second (Figure 1), and since τ_R = 1 second is at least consistent with all pertinent data, this value of τ_R has been selected for Class I, II-C, and IV airplanes for Flight Phase Category C, and for Class I and IV airplanes for Flight Phase Category A.

For Class I and IV airplanes performing Flight Phase Category B tasks, and for Class II-L and III airplanes performing all tasks, available data support a maximum value of τ_R = 1.3 to 1.5 seconds; an average value of τ_R = 1.4 seconds was selected. Ground simulator data in Reference B48 tend to support this value for large aircraft (cross-hatched curves in Figure 1), and in-flight data in Reference F22 support this value for small Class II airplanes, Flight Phase Category B (Figure 12).

No data were found to support two different values of τ_R as a function of Flight Phase Category for Class II and III airplanes, as was found for Class I and IV airplanes. Reference B39 (Figures 13 and 14) suggests a maximum satisfactory value of $\tau_R \approx 2.3$ seconds for large aircraft in the landing approach. However, careful examination of the rating terminology definitions indicates that this value of τ_R is probably more applicable to the Level 2 than the Level 1 requirements. Reference F5 shows (Figure 15) that for the reentry task, certain configurations received satisfactory ratings with a roll mode time constant as long as 5 seconds. Although the reentry task has many elements of Flight Phase Category B tasks, the duration differs, making the Reference F5 results not directly applicable. These data do show, however, that under some circumstances a satisfactory rating can be achieved with a long roll mode time constant.

An additional consideration that is demonstrated by much of the data, for example References F12 and G10, is that the required τ_R is, to a degree, determined by the value of $\angle_\beta$ or $|\phi/\beta|_d$. The in-flight data of Reference G10

(Figures 4 and 5) show this dependence directly. In the opinion of the author of Reference F12, the main reason for the differences between the Reference F12 and the data to which it is compared (see Figure 16), is that the Reference F12 ground simulator data were based on a much larger value of $|\phi/\beta|_d$ response ratio. The pilot ratings of both the Reference F12 and G10 data are degraded because of the response to atmospheric disturbances. This phenomenon is discussed in the substantiation for the paragraph covering the response to atmospheric disturbances.

Level 2 Requirements

References A14 and F8 do not make recommendations that pertain to Level 2 criteria as they did for Level 1. However, using available pilot rating data, it is possible to select values of τ_R which are consistent with available data.

Examination of Figure 1 (from Reference F8), which summarizes data from References F30 and B48, shows that for a change in pilot rating from 3 1/2 to 5 - 5 1/2, τ_R goes from approximately 1.3 to 3 seconds. Thus, even though the Reference F30 data are based on a fighter mission, the data do indicate the gradient of pilot rating with τ_R over the above noted ranges. Reference F1 indicates, from in-flight evaluations, that for fighter aircraft performing precision and maneuvering tasks, the pilot ratings degraded to marginally acceptable for τ_R values of 1.3 - 1.6 seconds. For large airplanes, Reference B39 (Figure 14) suggests τ_R values of 2.3 seconds for the satisfactory level, and 6 seconds for the acceptable levels; however, as noted earlier, these levels are probably associated with somewhat poorer flying qualities than are Levels 1 and 2 of Reference A1. From these considerations, the Level 2 requirements of Table VII of Reference A1 were selected.

Level 3 Requirement

The Level 3 value of τ_R = 10 seconds is relatively arbitrary but is based on data of Reference F5 (Figure 15) pertaining to lifting bodies and of Reference F12 (Figure 16) for fighter aircraft. While the selected value of τ_R cannot be vigorously defended, it does legislate against unstable roll modes while permitting effective acceleration-like responses to control inputs such as can be obtained on wingless vehicles.

REF. 12 IS HERE REF. F30
REF. 13 IS HERE REF. F22
REF. 14 IS HERE REF. F19
REF. 15 IS NASA TN D-1328
REF. 16 IS NASA TN D-792
REF. 57 IS HERE REF. B48

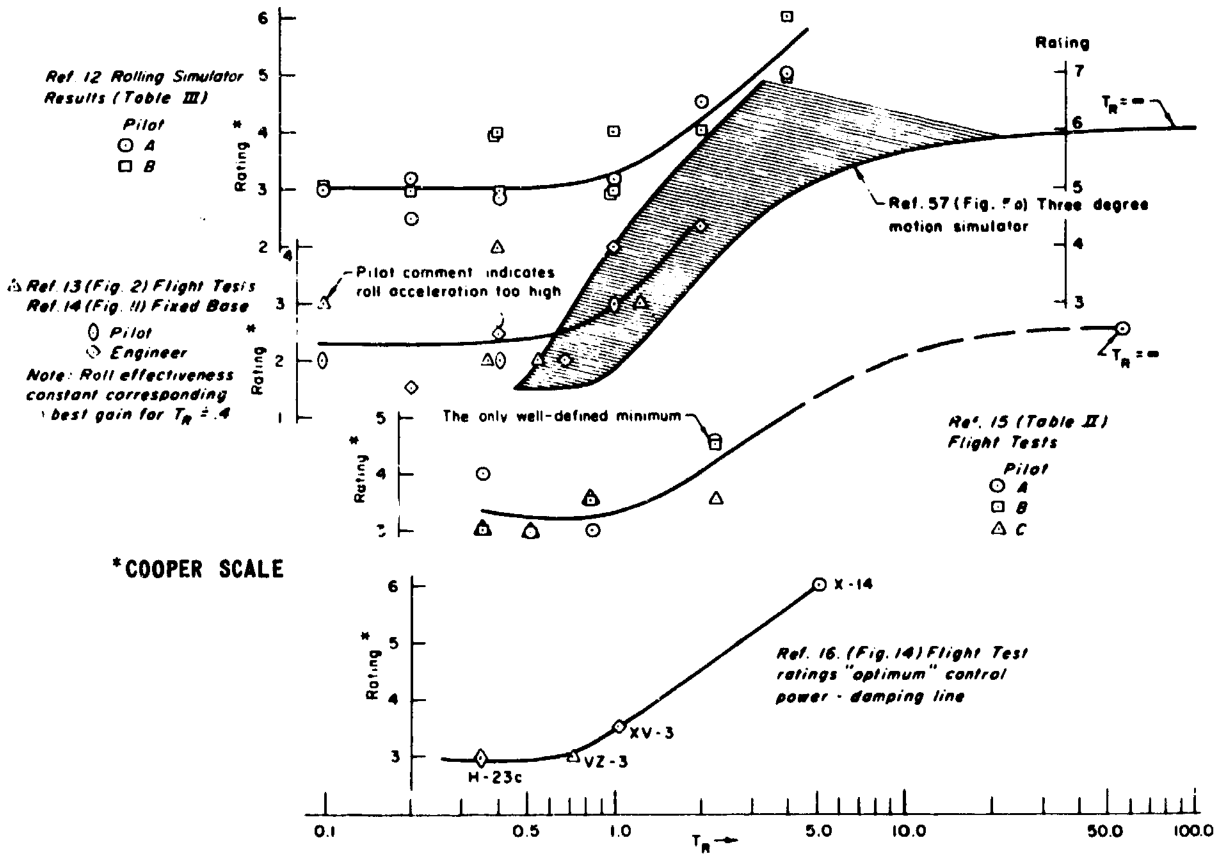


Figure 1 (3.3.1.2)

RATINGS VERSUS ROLL DAMPING - FLIGHT TEST, MOVING-BASE, FIXED-BASE WITH RANDOM INPUT (FROM REFERENCE F8)

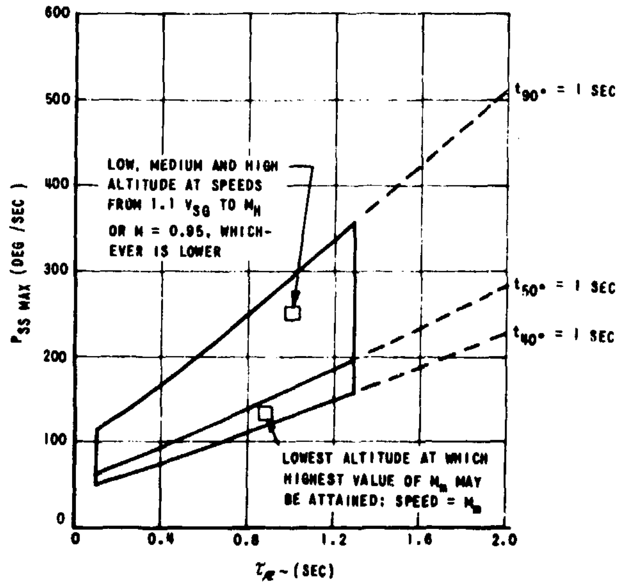


Figure 2 (3.3.1.2)
PROPOSED ROLL PERFORMANCE REQUIREMENTS FOR
(MIL-F-8785) CLASS III AIRCRAFT (FROM REFERENCE A14)

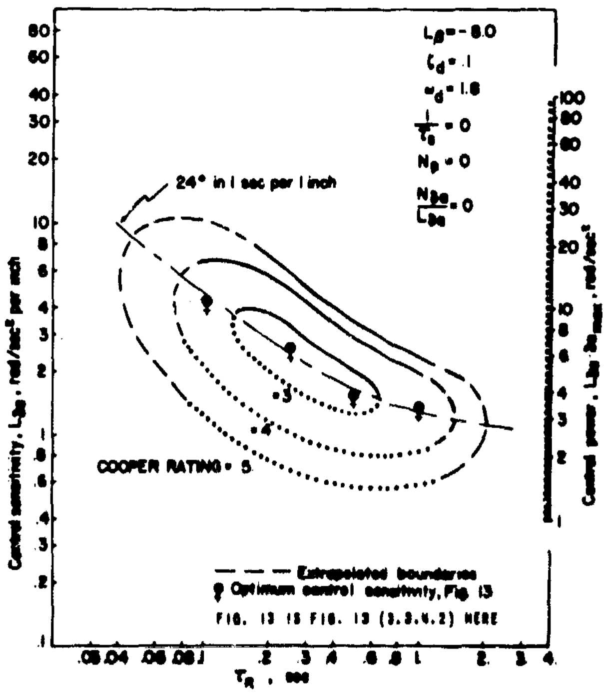


Figure 3 (3.3.1.2)
LATERAL CONTROL BOUNDARIES (FROM REFERENCE G10)

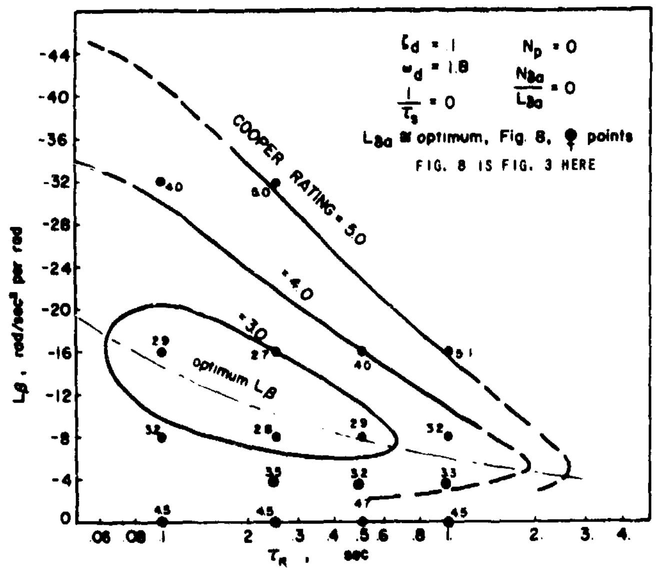


Figure 4 (3.3.1.2)

LATERAL FLYING QUALITIES BOUNDARIES

(L_β VS. τ_R, ζ_d=.1) (FROM REFERENCE G10)

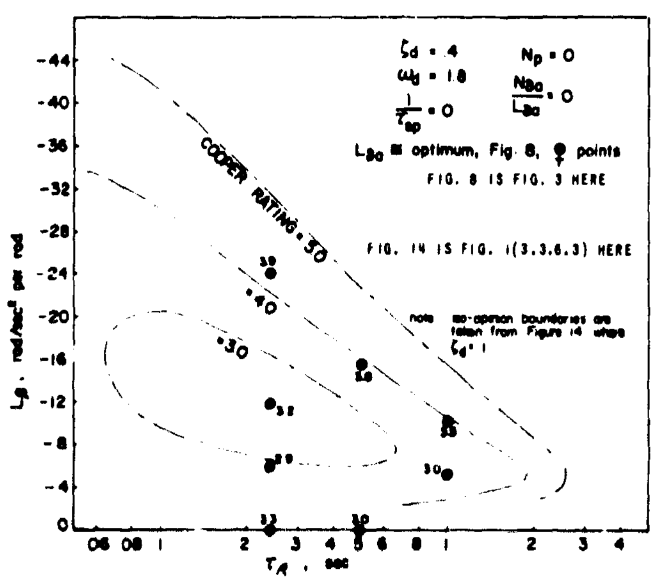


Figure 5 (3.3.1.2)

LATERAL FLYING QUALITIES BOUNDARIES

(L_β VS τ_R, ζ_d=.4) (FROM REFERENCE G10)

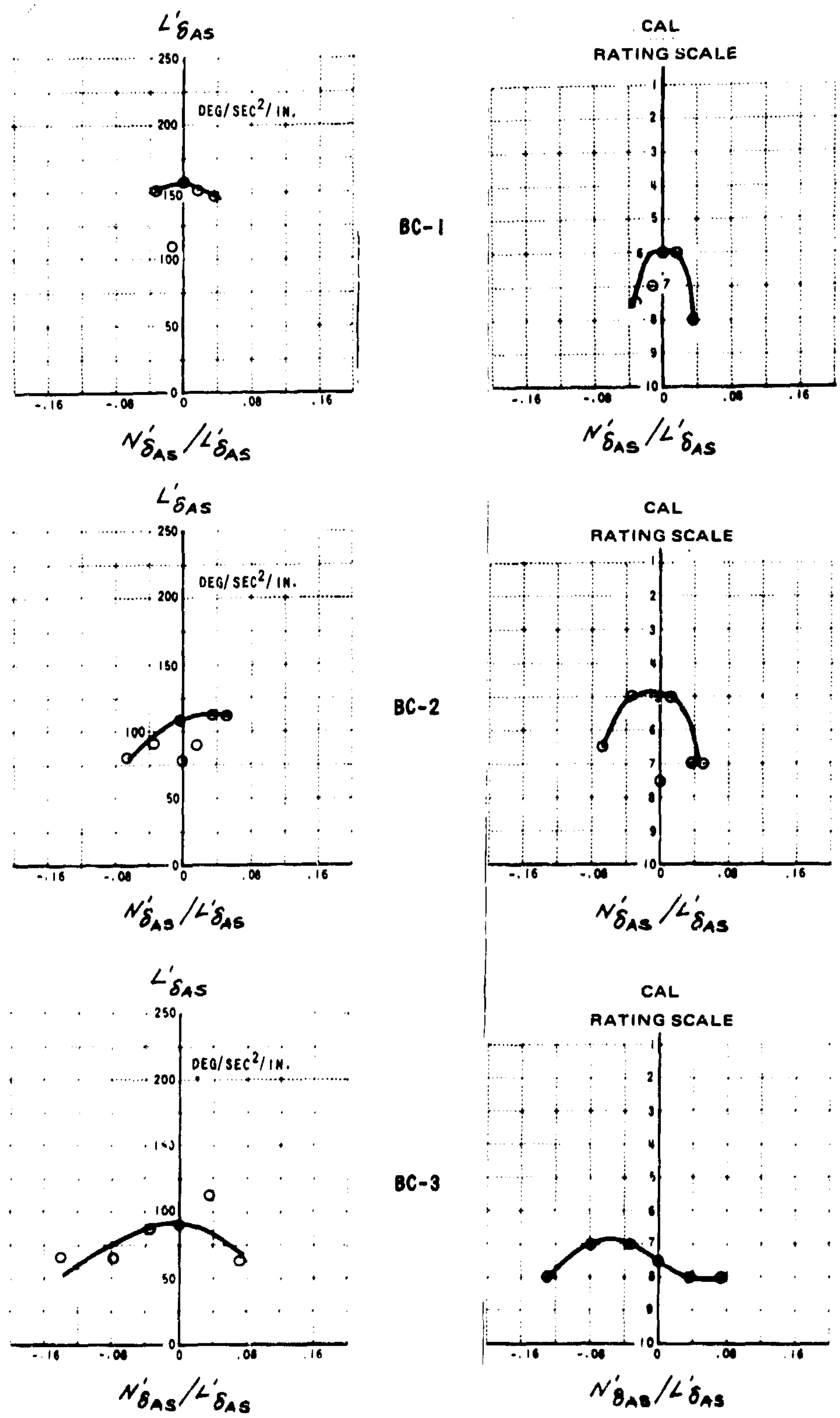
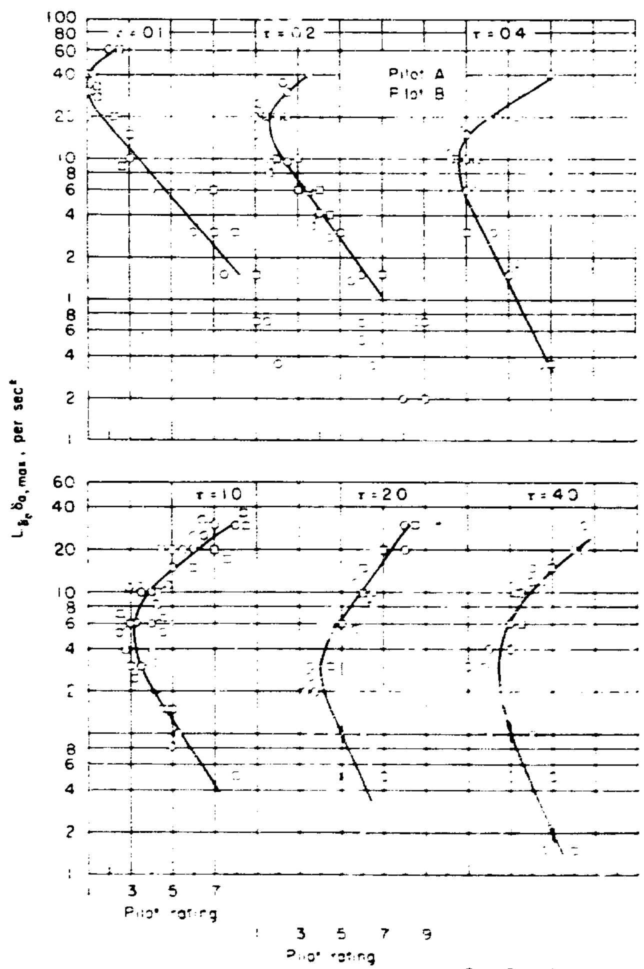


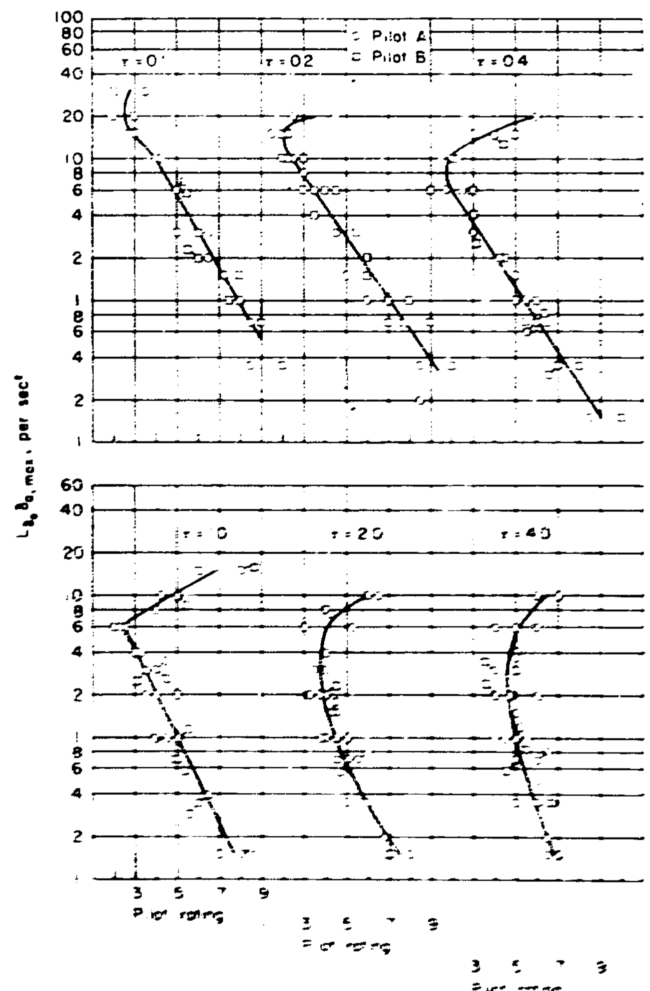
Figure 6 (3.3.1.2)
PILOT RATINGS AND OPTIMUM AILERON SENSITIVITY
(MEDIUM $\left|\frac{\phi}{\beta}\right|_d$, LONG τ_R) (FROM REF. F1)



(COOPER SCALE)

Figure 7 (3.3.i.2)

VARIATION OF PILOT OPINION WITH $L_{\delta_a}\delta_{a\,max}$, FOR CONSTANT VALUES OF τ AS OBTAINED FROM THE STATIONARY FLIGHT SIMULATOR (FROM REFERENCE F30)



(COOPER SCALE)

Figure 8 (3.3.i.2)

VARIATION OF PILOT OPINION WITH $L_{\delta_a}\delta_{a\,max}$. FOR CONSTANT VALUES OF τ AS OBTAINED FROM THE MOVING FLIGHT SIMULATOR. (FROM REFERENCE F30)

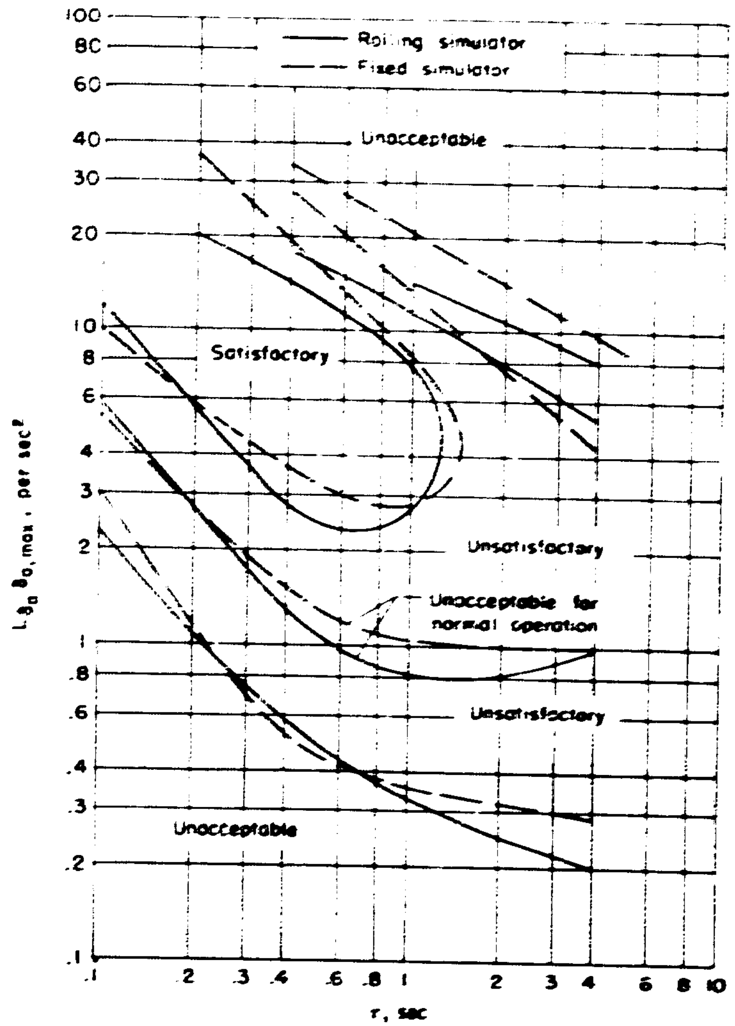


Figure 9 (3.3.1.2)
COMPARISON OF PILOT OPINION BOUNDARIES OBTAINED FROM THE FIXED AND MOVING FLIGHT SIMULATORS.
(FROM REFERENCE F30)

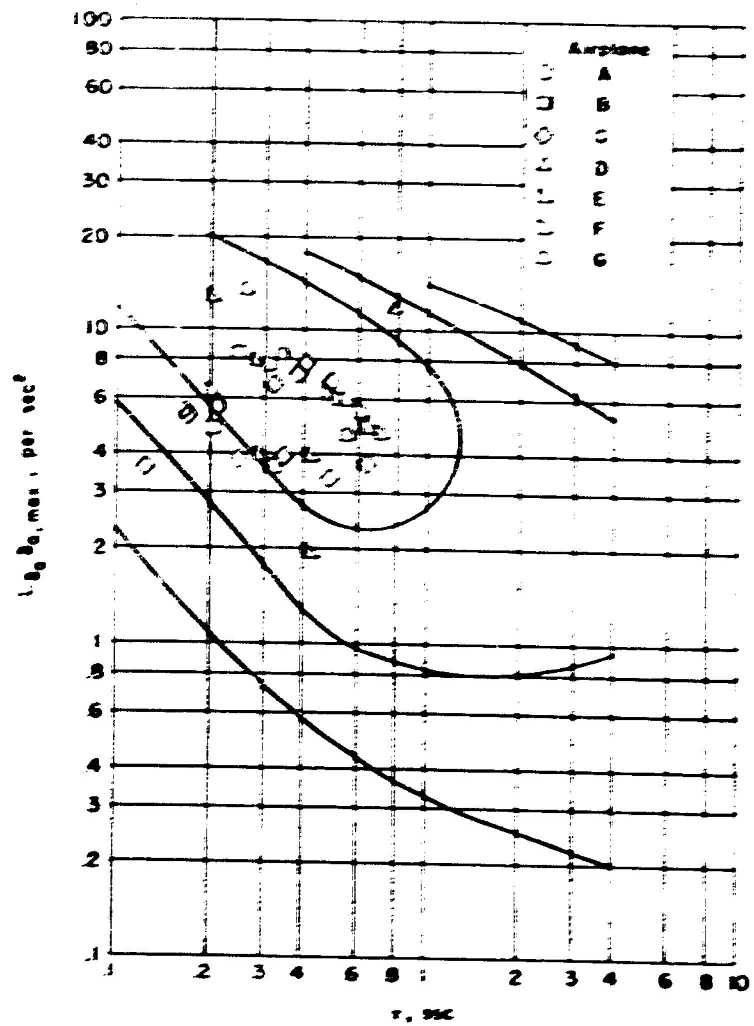


Figure 10 (3.3.1.2)
RANGE OF PARAMETERS $L_{\delta_a}\delta_{a\ max}$, AND τ, COVERED IN THE FLIGHT INVESTIGATION, SHOWN IN COMPARISON WITH THE MOTION SIMULATOR DERIVED BOUNDARIES.
(FROM REFERENCE F30)

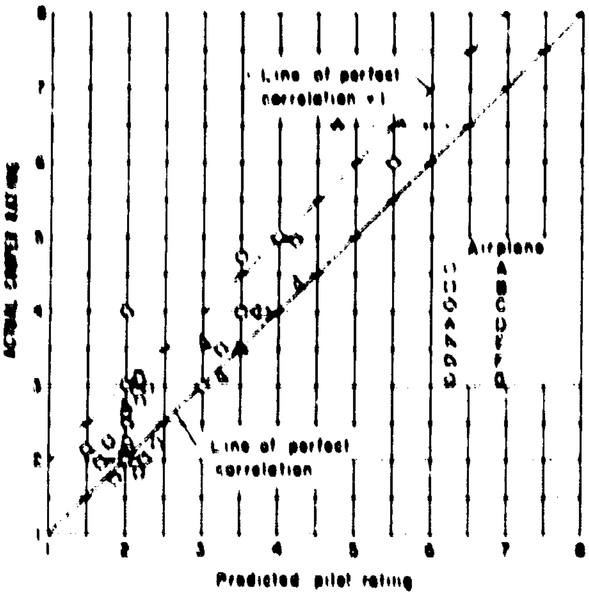


Figure 11 (3.3.1.2)

COMPARISON OF IN-FLIGHT PILOT-OPINION RATING WITH THOSE PREDICTED FROM FLIGHT SIMULATOR BOUNDARIES. (FROM REFERENCE F30)

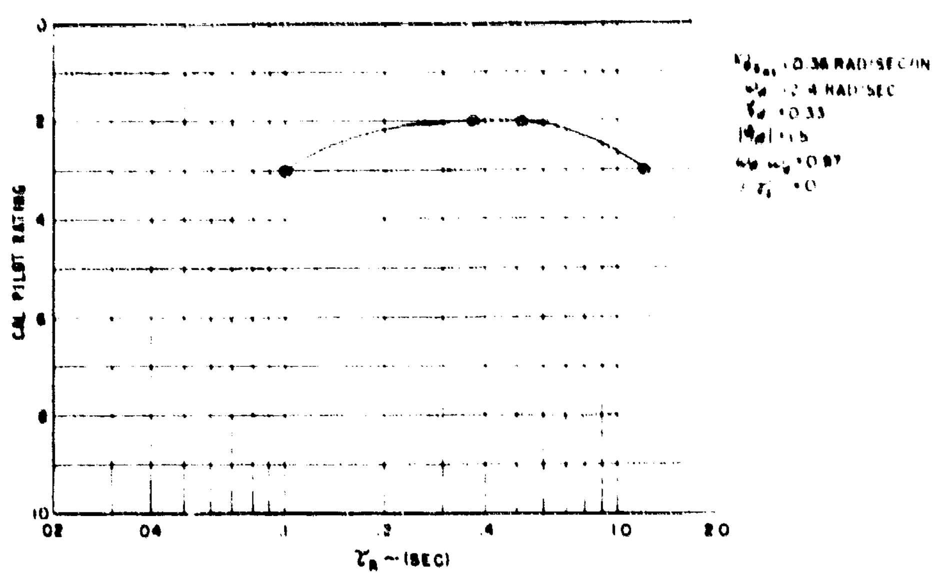


Figure 12 (3.3.1.2)

PILOT RATING VERSUS ROLL MODE TIME CONSTANT (FROM REFERENCE F22)

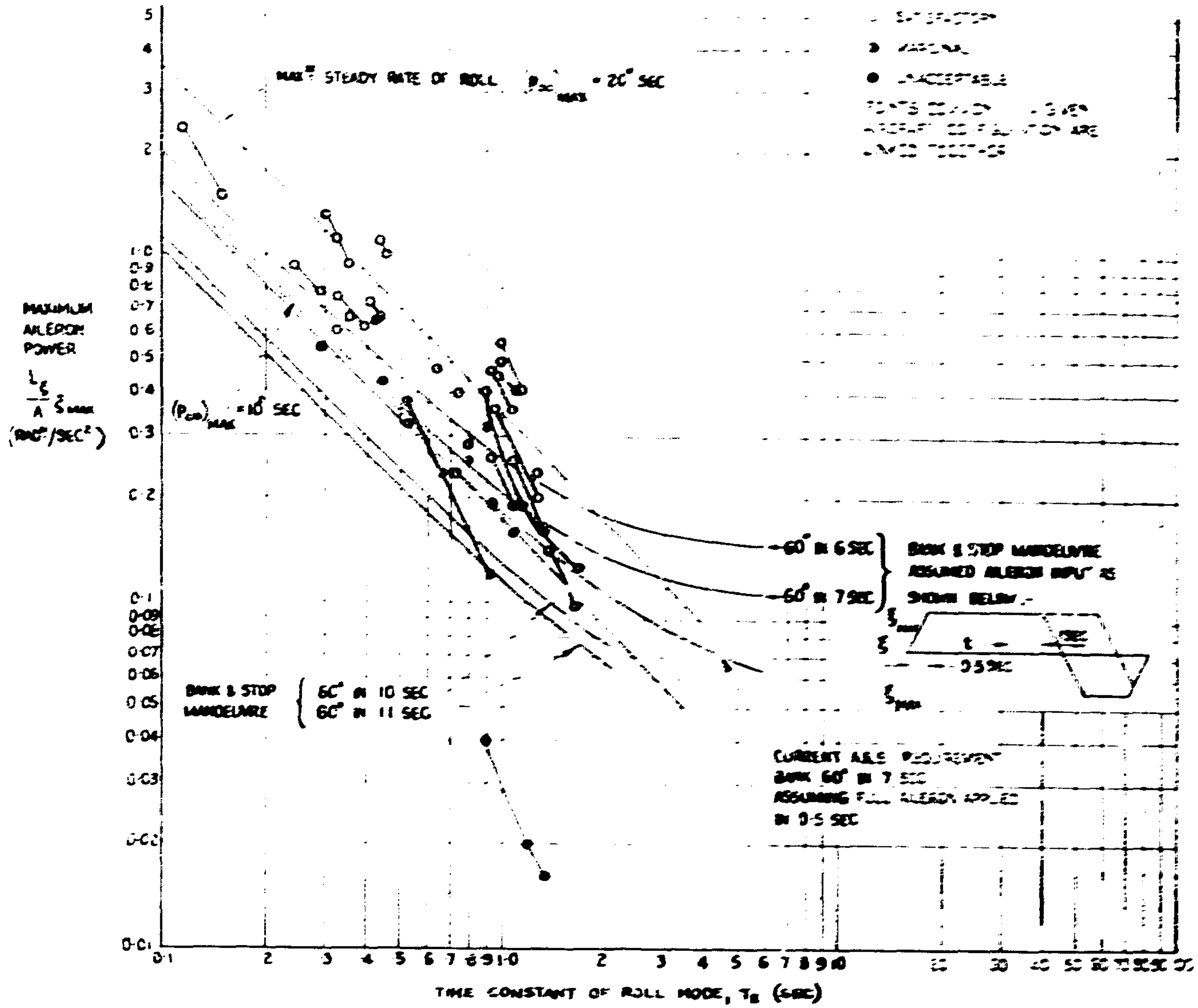


Figure 13 (3.3.1.2)

ROLL RESPONSE, LARGE AIRCRAFT IN APPROACH (FROM REFERENCE B39)

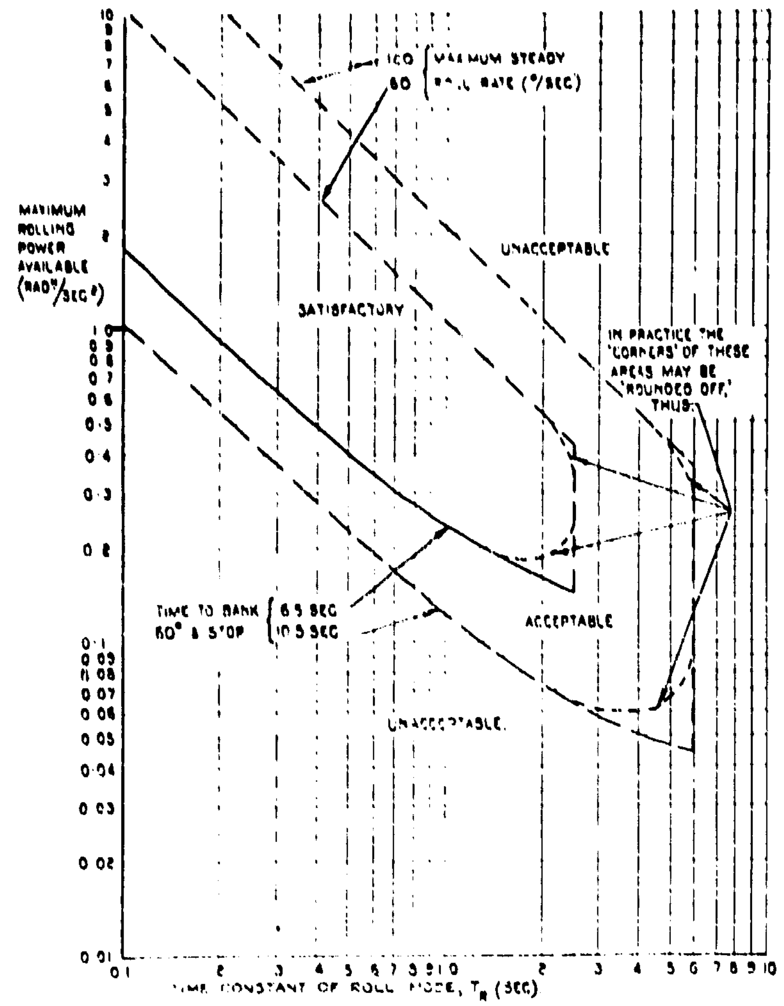


Figure 14 (3.3.1.2)

SUGGESTED ROLL-RESPONSE BOUNDARIES FOR LARGE AIRCRAFT. (APPROACH CONDITIONS). (FROM REFERENCE B39)

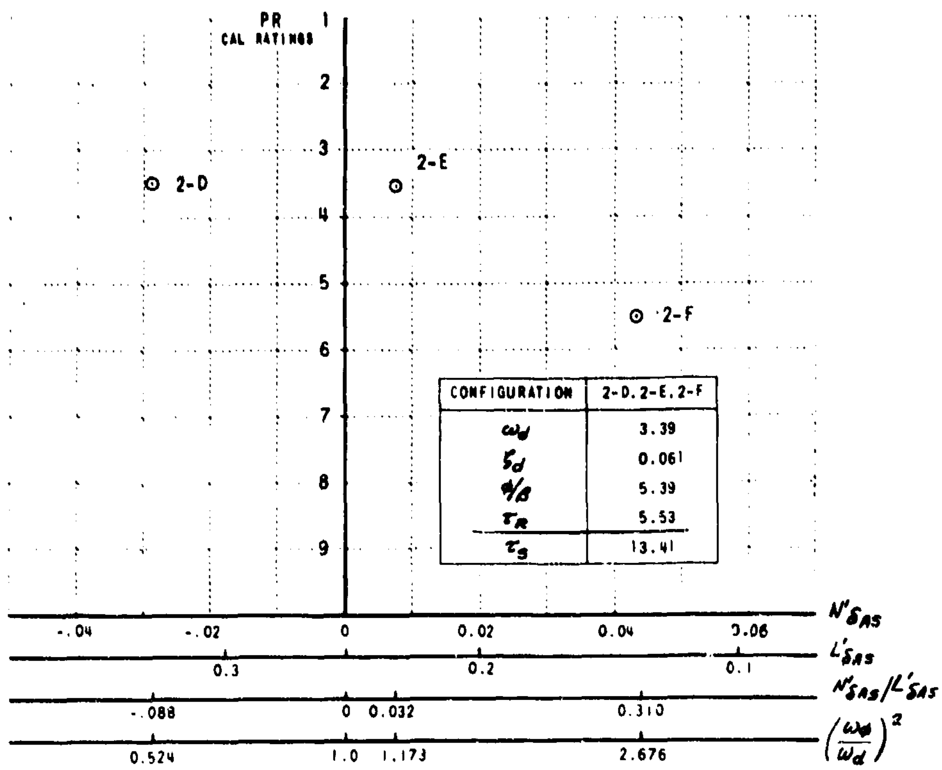


Figure 15 (3.3.1.2)

COMPOSITE PILOT RATINGS (FROM REFERENCE F5)

REF. 13 IS HERE REF. F22

REF. 14 IS HERE REF. F30

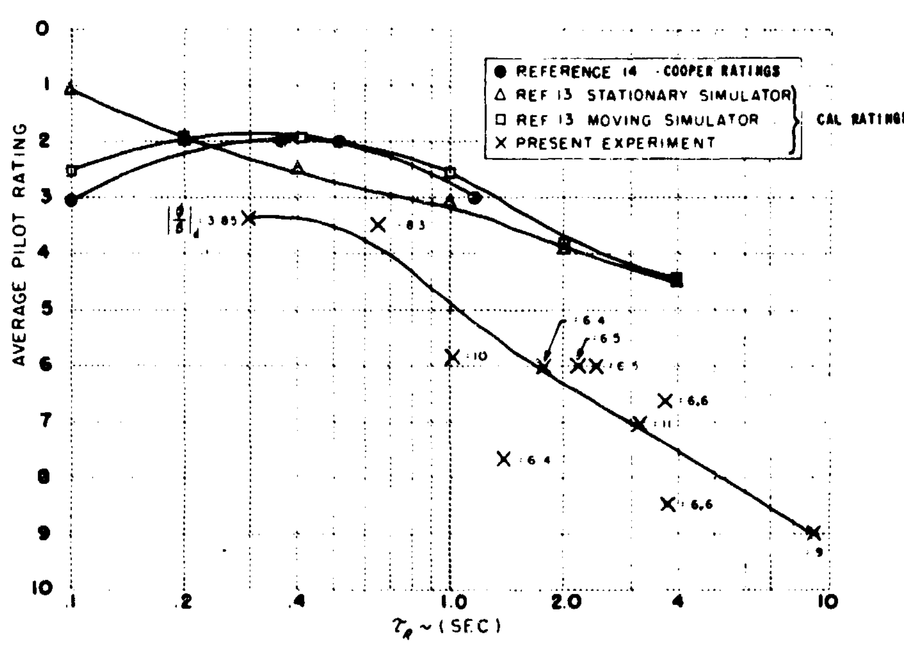


Figure 16 (3.3.1.2)

AVERAGE PILOT RATING OF ROLL MODE TIME CONSTANT (FROM REFERENCE F12)

3.3.1.3 SPIRAL STABILITY

REQUIREMENT

3.3.1.3 Spiral stability. The combined effects of spiral stability, flight-control-system characteristics, and trim change with speed shall be such that following a disturbance in bank of up to 20 degrees, the time for the bank angle to double will be greater than the values in table VIII. This requirement shall be met with the airplane trimmed for wings-level, zero-yaw-rate flight with the cockpit controls free.

TABLE VIII. Spiral Stability - Minimum Time to Double Amplitude

Class	Flight Phase Category	Level 1	Level 2	Level 3
I & IV	A	12 sec	12 sec	4 sec
	B & C	20 sec	12 sec	4 sec
II & III	All	20 sec	12 sec	4 sec

RELATED MIL-F-8785 PARAGRAPHS

3.4.2

DISCUSSION

The requirements on spiral divergence are aimed primarily at ensuring that the airplane will not diverge too rapidly from a wings-level condition during periods of pilot inattention.

This requirement of Reference A1 replaces paragraph 3.4.2 of MIL-F-8785. The data considered were almost identical to those considered by Reference A14, and the resulting conclusions are reasonably consistent with those of Reference A14.

From consideration of data in References F35, F49, and F29, Reference A14 recommended retention of the existing $T_2 \geq$ 20 seconds requirement in MIL-F-8785 and also proposed a requirement, $T_{1/2} \geq$ 10 seconds, on the degree of positive spiral stability permitted. The Reference A1 allowable instability is similar, in that for Flight Phase Categories B and C (analogous to the cruise and power approach configurations) the time to double amplitude is $T_2 \geq$ 20 seconds. But for Flight Phase Category A, where the pilot is generally closing a tight attitude loop, a less stringent value of $T_2 \geq$ 12 seconds was selected.

For Level 2, $\tau_2 \geq 12$ seconds was selected. This corresponds to the minimum tolerable boundary determined by Reference F49 (Figure 1).

For Level 3, a value of $\tau_2 \geq 4$ seconds was selected as a compromise between what is flyable and what is controllable if the pilot cannot devote full attention to flying the aircraft. This subject was discussed, as follows, in Reference F49:

> "The minimum tolerable time to double amplitude of the spiral divergence was very much longer than the minimum allowed by the existing handling qualities specifications (Reference A18). It is believed that the concept of the spiral divergence being unimportant to the pilot, because it is slow enough to be controlled, had led to considerable confusion on the subject. It is true that the pilot can control an airplane with a very rapid divergence (say, time to double amplitude of 2 or 4 seconds) when he has nothing to do but fly the airplane. Therefore, tests made with a rapid divergence where the pilot devoted full attention to flying, or made under conditions such as a landing approach, where the pilot necessarily devotes nearly all of his time to flying the airplane, will show that the minimum tolerable time to double amplitude is very low. However, there are many circumstances where the pilot does not, and indeed, cannot devote all of his attention to flying the airplane. He must read maps, work navigation problems, consult radio facilities handbooks, or route manuals, tune radios, and carry on various other activities. It is impossible for him to handle the : tasks effectively if, every time he diverts his attentio.ı, the airplane starts spiralling off. It is perfectly reasonable, then, for pilots to find an airplane with a rapid spiral divergence perfectly flyable yet absolutely intolerable."

In Reference A14, a limit of $\tau_{1/2} > 10$ seconds on the degree of spiral stability was recommended primarily from consideration of References F35 and F29. Reference F35 stated that "the maximum desired spiral stability appears to be a time to half amplitude of 10 seconds" and, based on closed-loop analysis, Reference F29 suggested that $\tau_{1/2}$ less than approximately 7 to 14 seconds would generally cause a degradation of pilot opinion. If the experimental in-flight data of Reference F49 (Figure 1) and Reference F35 (Figure 2) are examined, however, it can be seen that good pilot ratings are obtained for $\tau_{1/2} \approx 10$ seconds and that the flying qualities do not begin to degrade appreciably until $\tau_{1/2} \approx 5$ seconds.

Although there are some data that indicate there should be some limit on the degree of positive spiral stability, other data show that strong positive spiral stability can be beneficial. For example, in the program described in Reference F77, a wings-leveling device was installed in an aircraft which resulted in an effective highly convergent spiral. Although some pilots commented on the high forces required to hold the airplane in a turn, the flying qualities were considered to be quite acceptable and, in some respects, definitely preferable to neutral spiral stability.

For these reasons, it was decided not to place a requirement on τ_s or $T_{1/2}$, but rather to specify more direct requirements on other factors associated with convergent spirals, that is, aileron forces in turns and roll maneuverability. In 3.3.2.6, limits are placed on the amount of force required to hold an aircraft in a turn. In 3.3.4.1.1 and 3.3.4.1.3 requirements are placed on fighter aircraft performing large-bank-angle-change maneuvers.

It should be noted that the spiral requirements include the effect of lateral trim change with speed as well as the constant-speed spiral stability characteristics, since this is more representative of what the pilot sees than are constant-speed stability effects alone.

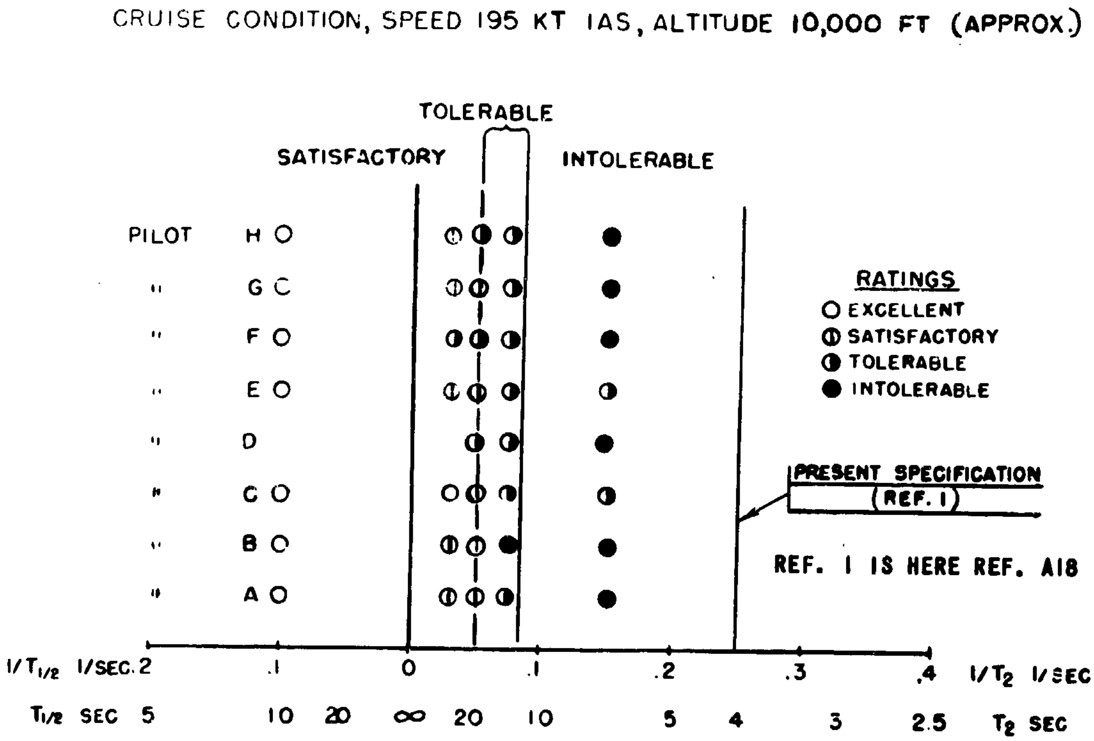


Figure 1 (3.3.1.3)

LIMITS OF SATISFACTORY AND TOLERABLE RATES OF SPIRAL DIVERGENCE (FROM REFERENCE F49)

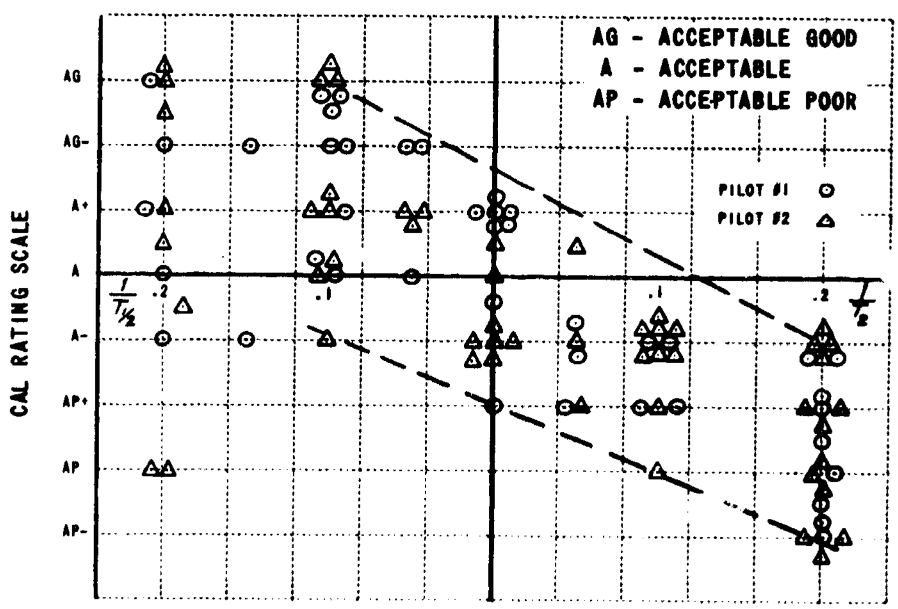


Figure 2 (3.3.1.3)

DATA FOR ALL TYPES OF FLYING - PILOT OPINION VS. SPIRAL DAMPING (FROM REFERENCE F35)

3.3.1.4 COUPLED ROLL-SPIRAL

REQUIREMENT

3.3.1.4 Coupled roll-spiral oscillation. A coupled roll-spiral mode will not be permitted.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

The coupled roll-spiral requirement is new and is based primarily on the data of Reference F12 (Figure 1), F5 (Figures 2, 3 and 4), and the analysis of Reference B1. These data show that the configurations investigated represented "poor to very bad" tactical airplanes. Since experience with lifting bodies, for example Reference F5, and the data of Reference F12 show that certain roll-spiral configurations are controllable (and even acceptable under certain conditions), consideration was given to permitting a coupled roll-spiral for Level 3 providing $(\zeta\omega_n)_{RS}$ was greater than 0.1. However, this was considered to be poor design practice since the coupled roll-spiral mode has not been investigated over the Level 1 or 2 ranges of all other related parameters. Also, under Level 3 conditions several flying qualities characteristics could be seriously degraded, so no coupled roll-spiral has been permitted.

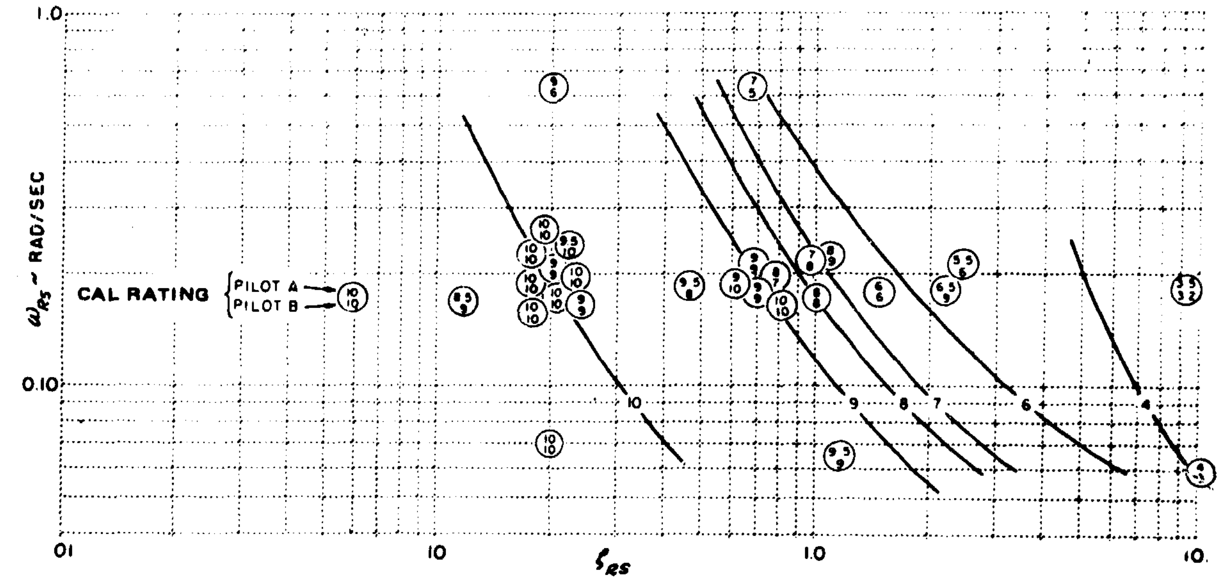


Figure 1 (3.3.1.4)

ω_{RS} vs. ζ_{RS} FOR CONVERGENT DUTCH ROLL.
(FROM REFERENCE F12)

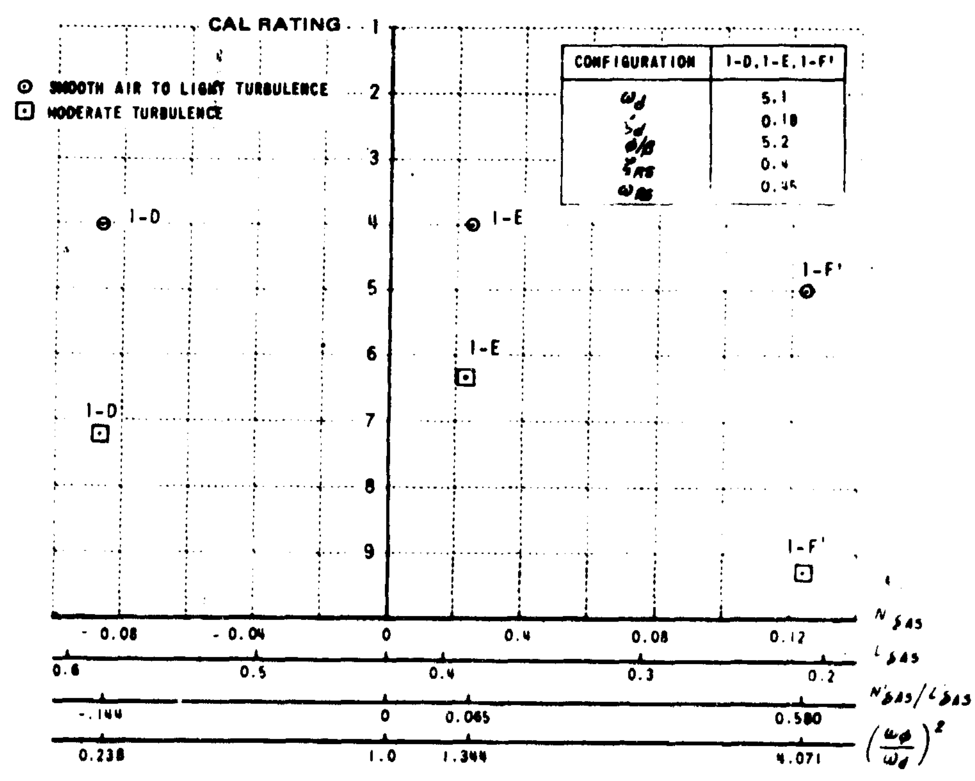


Figure 2 (3.3.1.4)

COMPOSITE PILOT RATINGS (FROM REFERENCE F5)

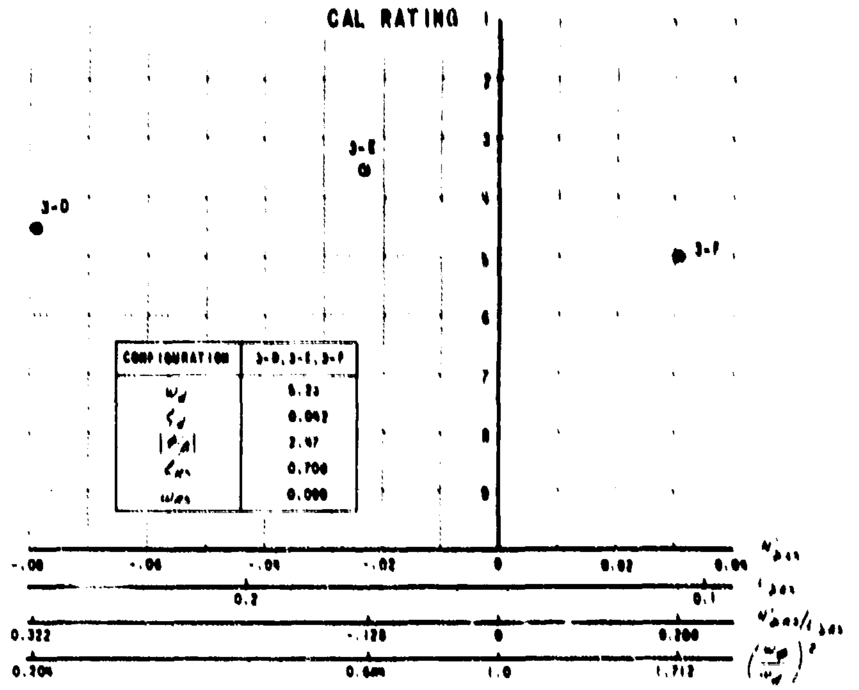


Figure 3 (3.3.1.4)
COMPOSITE PILOT RATINGS (FROM REFERENCE F5)

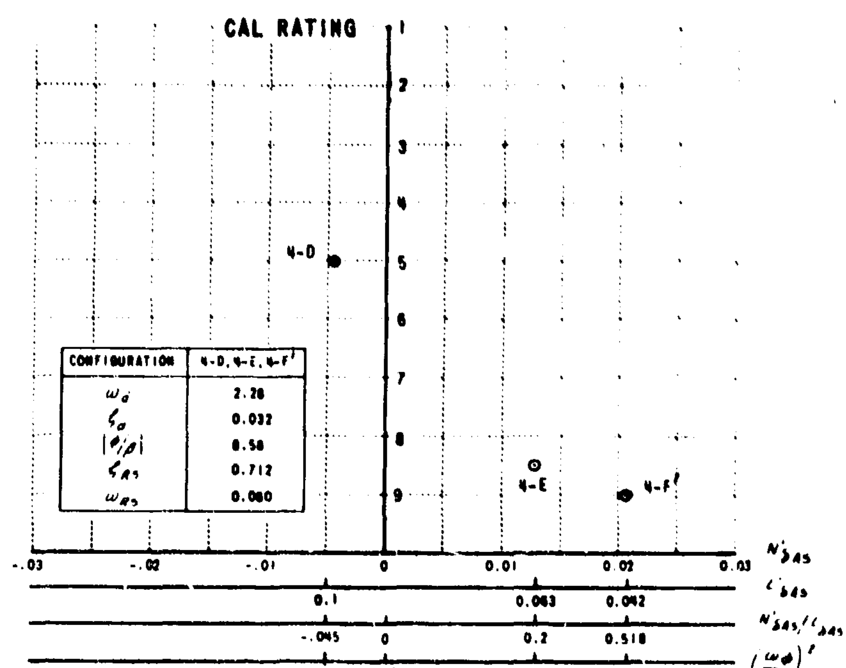


Figure 4 (3.3.1.4)
COMPOSITE PILOT RATINGS (FROM REFERENCE F5)

3.3.2 LATERAL-DIRECTIONAL DYNAMIC RESPONSE CHARACTERISTICS

REQUIREMENT

3.3.2 Lateral-directional dynamic response characteristics. Lateral-directional dynamic response characteristics are stated in terms of response to atmospheric disturbances and in terms of allowable roll rate and bank oscillations, sideslip excursions, aileron stick or wheel forces, and rudder pedal forces that occur during specified rolling and turning maneuvers. The requirements of 3.3.2.2, 3.3.2.3, and 3.3.2.4 apply for both right and left aileron commands of all magnitudes up to the magnitude required to meet the roll performance requirements of 3.3.4 and 3.3.4.1.

RELATED MIL-F-8785 PARAGRAPHS

3.4.6.3, 3.4.9

DISCUSSION

This section places both qualitative and quantitative requirements on the response of airplanes to control inputs and atmospheric disturbances. In contrast to most other requirements which specify desired response to control inputs, the requirements of this section place limits on unwanted responses resulting from control inputs and atmospheric disturbances. These unwanted responses detract from precision of control and can contribute to PIO tendencies. The requirements on the coupling that can exist between roll and sideslip are directed at precision of control for relatively small amplitude rolling maneuvers, whereas Paragraph 3.4.3 of Reference A1, "Pitch-roll-yaw coupling," considers much larger bank-angle changes and includes coupling between longitudinal and lateral-directional motions.

From a flying qualities viewpoint, roll-sideslip coupling is manifested in at least three ways depending on the $|\phi/\beta|_d$ response ratio. The next three paragraphs list the possible difficulties and may serve to guide evaluation pilots' ratings of flying qualities.

For low $|\phi/\beta|_d$ response ratios, sideslip per se is important to the pilot. For these cases, if roll rate or aileron control excite sideslip, the flying qualities can be degraded by such motions as an oscillation of the nose on the horizon during a turn or a lag or initial reversal in yaw rate during a turn entry, or by pilot difficulty in quickly and precisely taking up a given heading. In addition, the pilot cannot damp Dutch roll oscillations through the use of aileron control only.

For larger $|\phi/\beta|_d$ response ratios, the coupling of β with p and ϕ becomes important, causing oscillations in roll rate and ratcheting of bank angle. In this case, the pilot may have difficulty in precisely controlling roll rate or in acquiring a given bank angle.

For very large values of $|\phi/\beta|_d$ response ratios, sensitivity of roll to rudder pedals or response to atmospheric disturbances may be so great that the airplane is never considered to be very good.

Requirements on the amount of sideslip permitted during abrupt rudder-pedal-free turn entries, which may be most critical in airplanes with low $|\phi/\beta|_d$ response ratios, are specified in 3.3.2.4 and 3.3.2.4.1. It is to be noted that side acceleration or structural loads may well impose additional limitations which are more severe than the requirements of Paragraph 3.3.2.4, particularly at high speeds. Requirements on the amount of roll rate oscillation during abrupt turn entries, which may be most critical for airplanes with moderate $|\phi/\beta|_d$ response ratios, are specified in 3.3.2.2 and 3.3.2.2.1. Associated requirements on bank angle oscillations are given in 3.3.2.3. Qualitative requirements on the response and control in atmospheric disturbances, which may be most critical for airplanes with large $|\phi/\beta|_d$ response ratios, are specified in 3.3.2.1.

Another factor that has been considered in addressing the problem of roll-sideslip coupling is how well the pilot can control or prevent unwanted motions, that is, how well he can coordinate with rudder pedals. If the airplane is easy to coordinate during turn entries, then the pilot may tolerate relatively large unwanted motions during rudder-pedals-free turn entries since he can control these unwanted motions if desired. On the other hand, when coordination is difficult, the pilot will tolerate only small unwanted motions since he must either live with these motions or may even aggravate them if he tries to coordinate. The parameter " ψ_β " was introduced as the most precise measure of this very nebulous, but important, factor - difficulty of coordination.

It should be noted that measurement of $\frac{\phi_{osc}}{\phi_{AV}}$, $\frac{p_{osc}}{p_{AV}}$, ψ_β and $\Delta\beta_{MAX}$ requires some minimum length of time history. Although this should not pose a problem in design, in flight test when large step aileron inputs are used, large bank angles may be reached before the required length of time history can be obtained. When small inputs are used, however, sufficient time is normally obtainable. And these parameters are critical for closed-loop controllability aspects of flying qualities, which involve small inputs. Provision has been made to flight test for maximum sideslip and roll rate excitation following large aileron inputs, however, since these are also of concern. Such a sideslip requirement was contained in MIL-F-8785. The ϕ_{osc}/ϕ_{AV} impulse-response requirement also lends itself to flight testing with larger control inputs.

The roll oscillation and sideslip requirements were derived empirically from experimental in-flight and flight test data generated from aircraft possessing 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 correlation of the empirical data with the selected parameters.

Definitions of, and techniques for measuring, the parameters used in the roll sideslip coupling requirements are presented in Paragraph 6.[illegible].5 of Reference A1 and in Appendices VB and VC. The time histories presented in Figures 9 and 10 of Reference A1 are presented here in Figures 1 through 9 for greater clarity. The figures present p, ϕ and β time histories following an abrupt aileron stick input for four different $p/\delta_{AS}|_{STEP}$ transfer function zero locations. These figures present examples of measurement of ψ_β, ϕ/β, β_{osc}/β_{av} and $\Delta\beta_{max}$ for both right and left rolls. Figure 1 shows the four zero locations, Figures 2 through 5 show the time histories for right rolls, and Figures 6 through 9 show the time histories for left rolls.

Although it was necessary to tie these requirements very closely to a specific input and specific motion variables in order to specify them precisely, it is recognized that there may well be other techniques whereby the required data can be obtained.

Sideslip sensors are subject to sidewash and installation errors. Since generally they cannot be located near the airplane c.g., they also pick up components of yawing and rolling motion. For flight test, then, it may be more convenient to use a lateral accelerometer mounted near the c.g. than to calibrate and correct a β -sensor output. For some configurations Y_{δ_a}, Y_p, Y_r and other possible side-force derivatives have negligible effect compared to Y_β. In that case, from Reference B73 it can be shown that the sideslip and accelerometer-reading responses at the c.g. are related by a constant. In transfer-function form,

$$\frac{\beta(s)}{\delta_a(s)} = \frac{2m}{\rho V^2 S C_{y_\beta}} \quad \frac{a_y(s)}{\delta_a(s)}$$

In many cases both ψ_β and $\Delta\beta_{max}$ can be found accurately and easily this way. An example of a configuration for which a_y is not equivalent to β is a highly-swept wing with spoilers, which tends to have significant $Y(\delta_a, \beta)$.

The data base for the quantitative Level 1 and Level 2 boundaries is somewhat sparse. In view of the demonstrated inadequacy of the MIL-F-8785 requirements, however, we feel that Reference A1 is a tremendous improvement. Its requirements have a solid basis in theory and are substantiated by the data available.

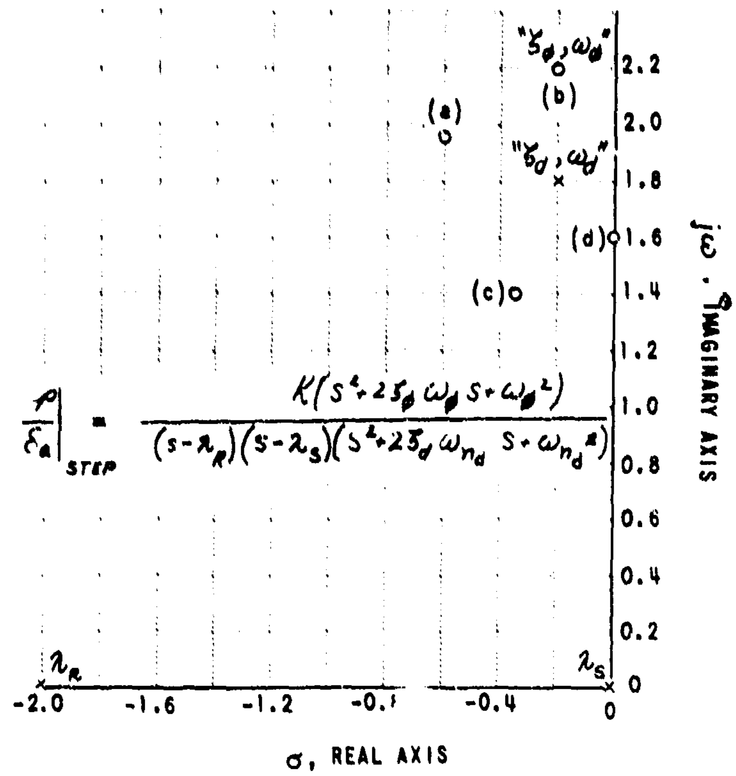


Figure 1 (3.3.2)
EXAMPLES OF POLE AND ZERO LOCATIONS
OF THE $\left.\frac{p}{\delta_a}\right|_{STEP}$ RESPONSE FUNCTION

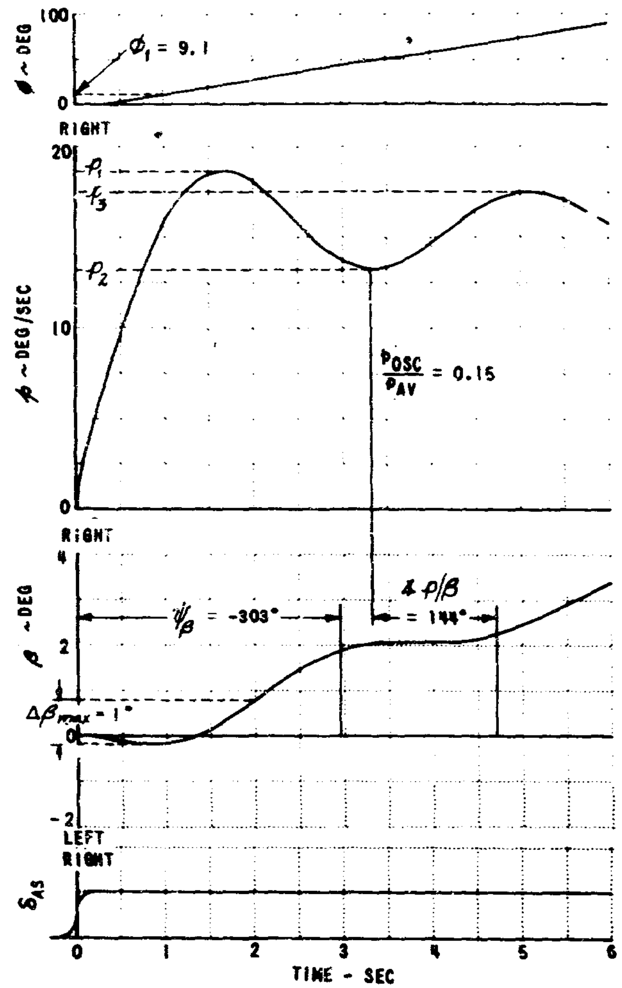


Figure 2 (3.3.2)
RIGHT ROLL - ZERO AT
POSITION (a) OF FIGURE 1

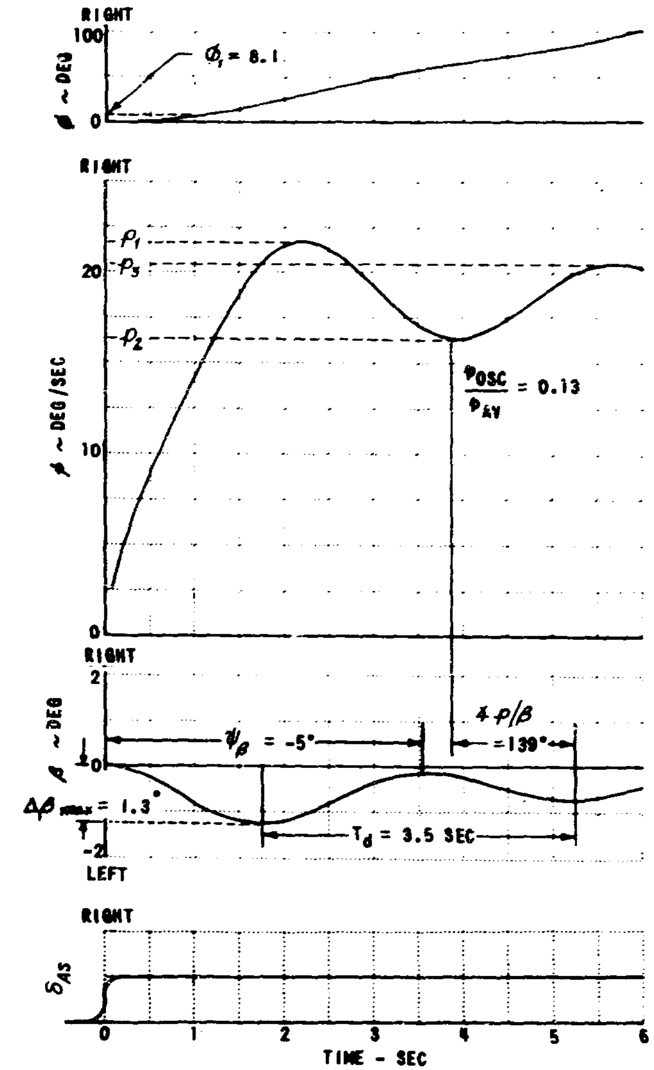


Figure 3 (3.3.2)
RIGHT ROLL - ZERO AT
POSITION (b) OF FIGURE 1

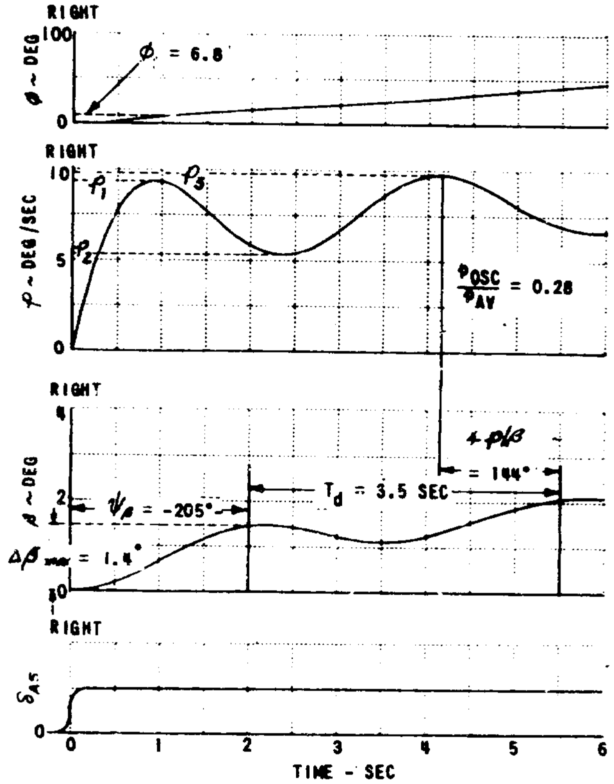


Figure 4 (3.3.2)
RIGHT ROLL - ZERO AT
POSITION (c) OF FIGURE 1

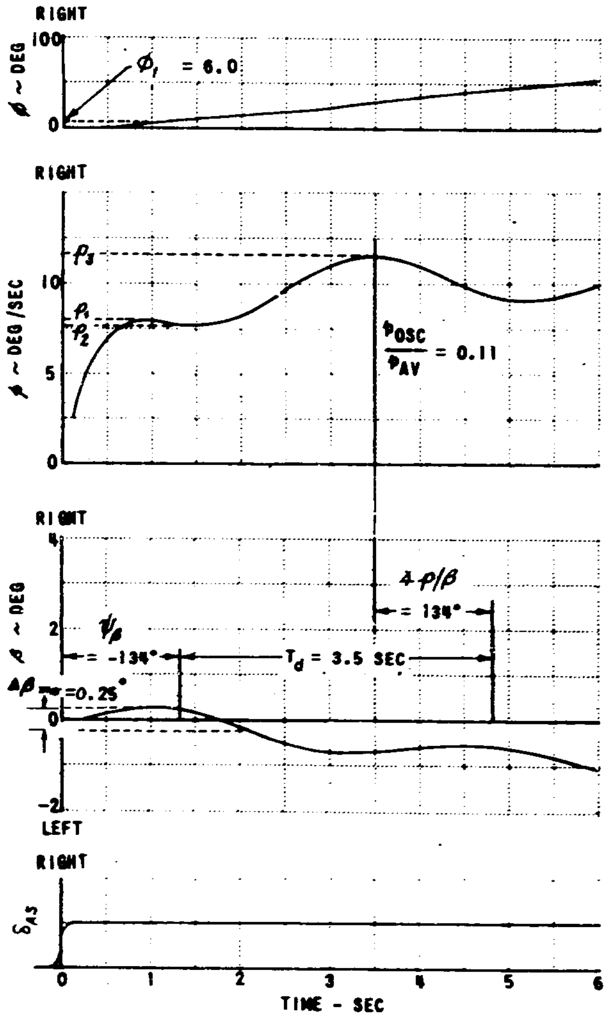


Figure 5 (3.3.2)
RIGHT ROLL - ZERO AT
POSITION (d) OF FIGURE 1

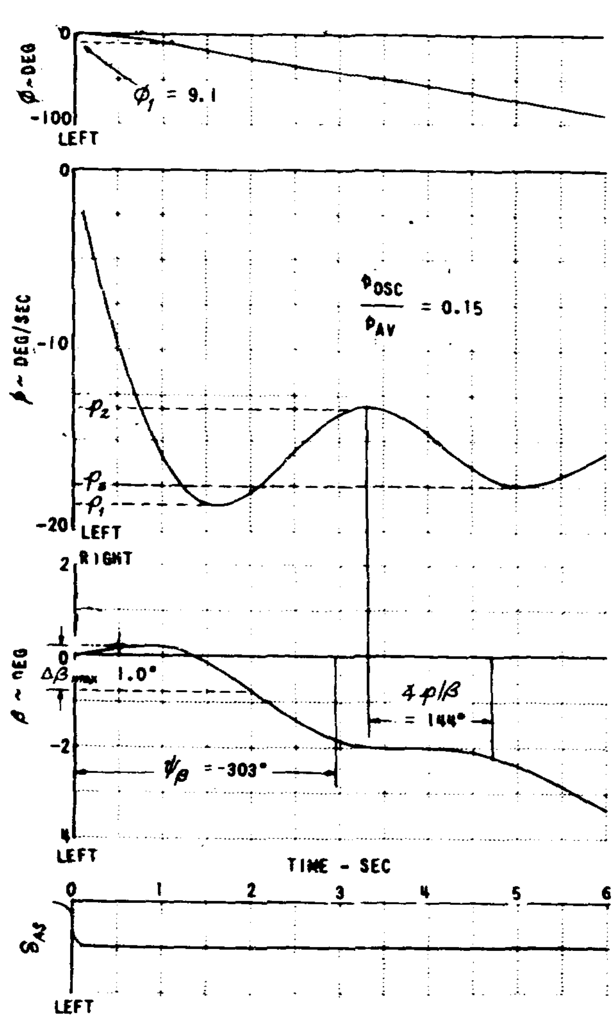


Figure 6 (3.3.2)
LEFT ROLL - ZERO AT
POSITION (a) OF FIGURE 1

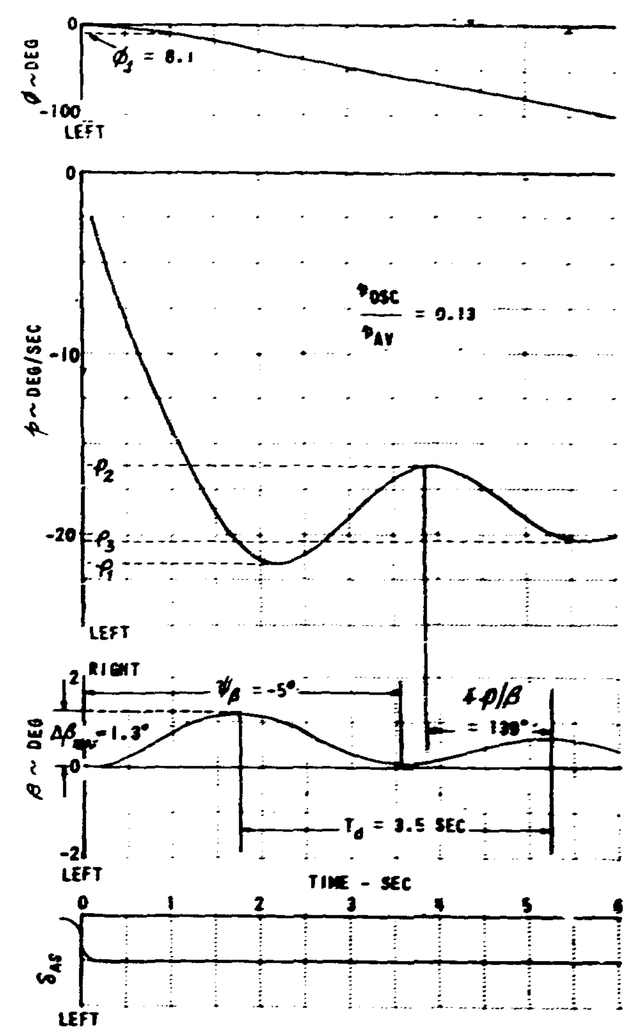


Figure 7 (3.3.2)
LEFT ROLL - ZERO AT
POSITION (b) OF FIGURE 1

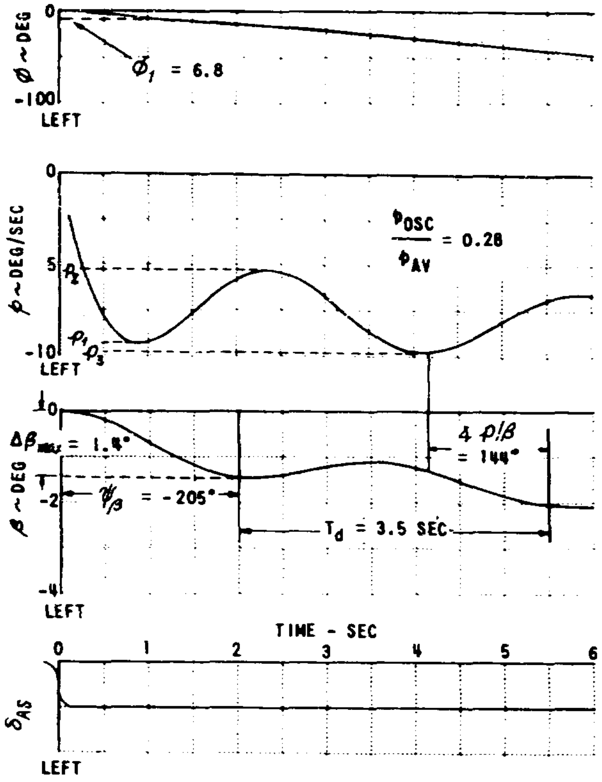


Figure 8 (3.3.2)
LEFT ROLL - ZERO AT
POSITION (c) OF FIGURE 1

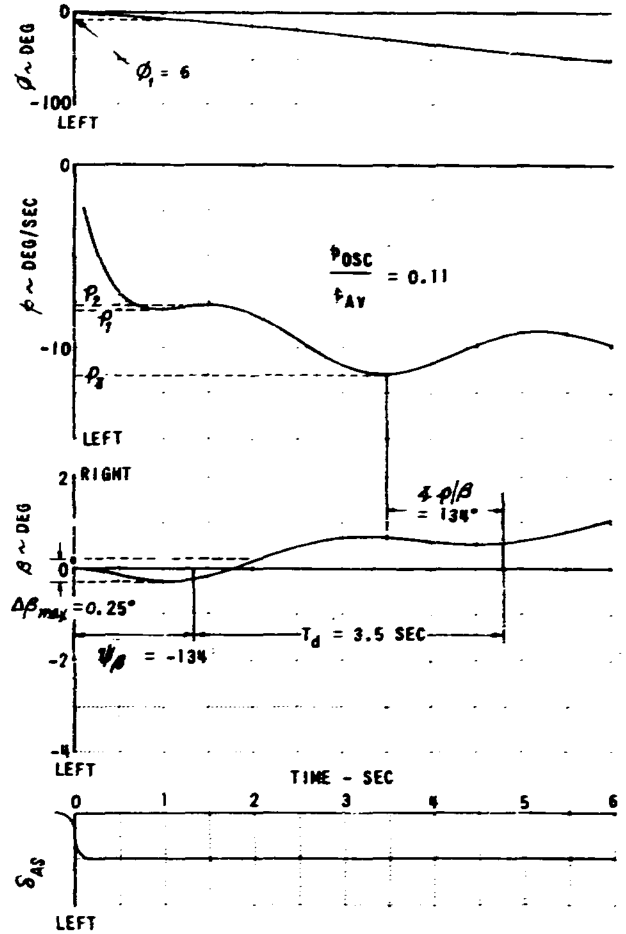


Figure 9 (3.3.2)
LEFT ROLL - ZERO AT
POSITION (d) OF FIGURE 1

3.3.2.1 LATERAL-DIRECTIONAL RESPONSE TO ATMOSPHERIC DISTURBANCES

REQUIREMENT

3.3.2.1 Lateral-directional response to atmospheric disturbances. Although no numerical requirements are specified, the combined effect of ω_{n_d}, ζ_d, τ_R, $\measuredangle\, p/\beta$, $|\phi/\beta|_d$, gust sensitivity, and flight-control-system nonlinearities shall be such that the airplane will have acceptable response and controllability characteristics in atmospheric disturbances. In particular, the roll acceleration, rate, and displacement responses to side gusts shall be investigated for airplanes with large rolling moment due to sideslip.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

To understand the complexity of the gust and turbulence response problem, and to see why it was necessary to specify a direct requirement, albeit qualitative, it is beneficial to examine the analysis given in Reference F5. Although this was a study of causes underlying the severe response to gusts of certain re-entry configurations, the results are of general applicability. From this study (presented in a slightly modified form in Appendix VA) it can be seen that parameters such as Dutch roll natural frequency, roll damping, and the stability derivative L_β, as well as Dutch roll damping ratio, all contribute significantly to the roll response due to side gusts. Pilots tend to downrate otherwise-acceptable configurations with high $\left|\frac{\phi}{\beta}\right|_d$ because of sensitivity to turbulence. ($\left|\frac{\phi}{\beta}\right|_d$ is discussed in detail in Appendix VC.)

This, however, does not tell the whole story. The ability of a pilot to control an aircraft in the presence of atmospheric disturbances depends not only on the response of the aircraft to atmospheric disturbances, but also on the closed-loop controllability characteristics of the pilot/aircraft combination. Since the latter is influenced by factors such as roll-sideslip coupling characteristics, the overall problem of controllability in the presence of atmospheric disturbances is extremely complex. In 3.3.1.1, an increment in Dutch roll damping, $\zeta_d\omega_{n_d}$, is specified when $\omega^2_{n_d}\left|\frac{\phi}{\beta}\right|$ exceeds 20/sec^2. This requirement is specifically directed at airplanes which have large roll acceleration response to side gusts, rudder control inputs, etc. Because the problem of controllability in turbulence is complex, simply increasing the Dutch roll damping may not be an adequate solution. For this reason, the qualitative requirement of 3.3.2.1 must be given serious attention.

3.3.2.2 ROLL RATE OSCILLATIONS

3.3.2.2.1 ADDITIONAL ROLL RATE REQUIREMENTS FOR SMALL INPUTS

REQUIREMENTS

3.3.2.2 Roll rate oscillations. Following a rudder-pedals-free step aileron control command, the roll rate at the first minimum following the first peak shall be of the same sign and not less than the following percentage of the roll rate at the first peak:

Level	Flight Phase Category	Percent
1	A & C	60
	B	25
2	A & C	25
	B	0

For all Levels, the change in bank angle shall always be in the direction of the aileron control command. The aileron command shall be held fixed until the bank angle has changed at least 90 degrees.

3.3.2.2.1 Additional roll rate requirement for small inputs. The value of the parameter p_{osc}/p_{AV} following a rudder-pedals-free step aileron command shall be within the limits shown on figure 4 for Levels 1 and 2. This requirement applies for step aileron control commands up to the magnitude which causes a 60 degree bank angle change in 1.7 T_d seconds.

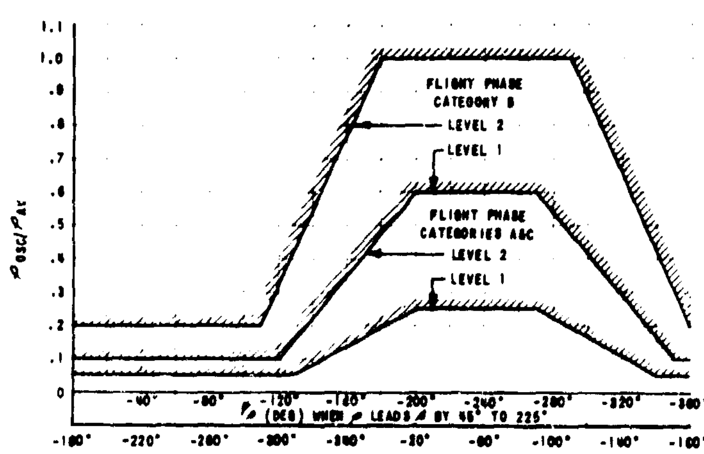


Figure 4. ROLL RATE OSCILLATION LIMITATIONS

RELATED MIL-F-8785 PARAGRAPHS

3.4.6.3

DISCUSSION

Background

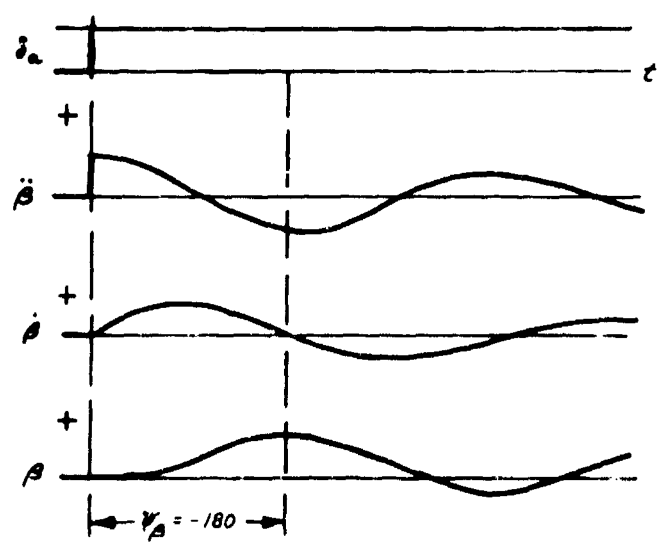
These requirements are new and are directed at precision of control of airplanes with moderate to high $\left|\frac{\phi}{\beta}\right|_d$ response ratios.

Until recently, most investigations of roll-sideslip coupling for moderate bank-angle-change maneuvers have been concerned with the effects of aileron yaw, $N'_{\delta_{AS}}$, which has been examined in relation to its effect in rolling or bank angle tracking maneuvers. The parameter which has been most widely used to describe the effects and specify handling qualities has been $\frac{\omega_\phi}{\omega_{n_d}}$ where

$$\left(\frac{\omega_\phi}{\omega_{n_d}}\right)^2 \approx 1 - \frac{N'_{\delta_{AS}}}{L'_{\delta_{AS}}} \frac{L'_\beta}{N'_\beta}$$

when $N'_\beta = \frac{g}{V}$, $Y_\beta = 0$, $Y_\delta = 0$ and $N'_r << L'_p$. Discussion of ω_ϕ/ω_{n_d} effects will serve as a lead-in to the more inclusive requirements of Reference A1.

Consider now the effects of "adverse yaw" ($N'_{\delta_{AS}}$ negative) with positive effective dihedral (L'_β negative) following a step aileron input. From Sketch 1 it can be seen that the resulting effect on sideslip is that β is initially small and then goes positive. (This and other time responses and transfer functions used in this discussion are derived in Appendix VC.)

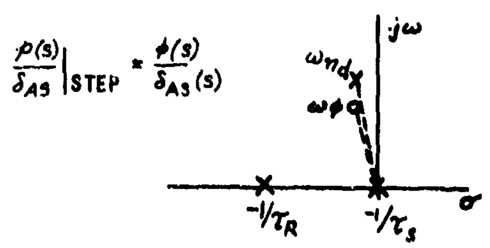


Sketch 1

For a cosine representation, the Dutch roll component of the sideslip response has a phase angle, ψ_β , of approximately -180°. This is the phase angle that appears in the time-history equation

$$\left|\frac{\beta}{\delta_{AS}}\right|_{STEP} = C_0 + C_S e^{\lambda_S t} + C_R e^{\lambda_R t} + C_d e^{-\zeta_d \omega_{n_d} t} \cos\left(\omega_{n_d}\sqrt{1-\zeta_d^2}\, t + \psi_\beta\right)$$

Sketch 2 is a plot of the roll-rate response-function poles and zeros on the complex plane for a step control input. For negative $N_{\delta_{AS}}$ and $N'_p - \frac{g}{V}$, $\frac{\omega_\phi}{\omega_{n_d}}$ is less than one; that is, the zero lies below the Dutch roll pole. Since $1/\tau_S$ must be very close to zero, the relative departure of the zero from the pole is a measure of the departure of the roll response from pure one-degree-of-freedom form. As more Dutch roll appears in the ϕ/δ_{AS} response, pilots object to its oscillatory nature and may experience difficulty in tracking (see e.g., References F29 and F9).



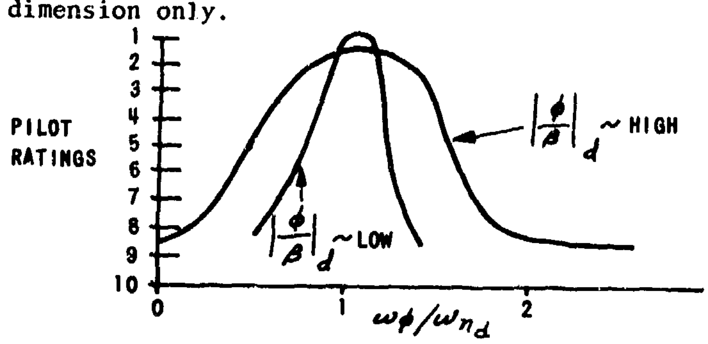
Sketch 2

It can be shown similarly that for "proverse yaw", β is initially small and then goes negative, and has a phase angle of approximately -360 degrees. The corresponding condition on the complex plane for the $\frac{\phi}{\delta_{AS}}$ transfer function is with the zero above the pole$\left(\frac{\omega_\phi}{\omega_{n_d}} > 1\right)$.

For small ζ_d and ζ_ϕ , it can be shown that the time response to a step input is approximately

$$\frac{p(t)}{\delta_{AS}} \approx L'_{\delta_{AS}} \tau_R \left\{\left(\frac{\omega_\phi}{\omega_{n_d}}\right)^2 - \left(\frac{1+\omega_\phi^2\tau_R^2}{1+\omega_{n_d}^2\tau_R^2}\right) e^{-t/\tau_R} + \frac{\left|\left(\frac{\omega_\phi}{\omega_{n_d}}\right)^2 - 1\right|}{\sqrt{1+\omega_{n_d}^2\tau_R^2}} e^{-\zeta_d\omega_{n_d} t} \cos\left(\omega_{n_d}\sqrt{1-\zeta_d^2} + \psi_p\right)\right\}$$

Because so much data have been presented in the form of $\frac{\omega_\phi}{\omega_{n_d}}$, as in Sketch 3, we have become accustomed to thinking in terms of either "adverse yaw" or "proverse yaw" and the ratio of the radial distances from the origin to the zero and to the Dutch roll pole. In other words, we have been thinking in terms of one dimension only.

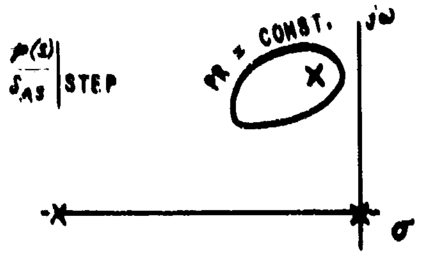


Sketch 3

Basis for the Requirement

When the effects of N_p are considered, however, the zero no longer only moves primarily in the radial direction: it moves within an area around the Dutch roll pole. Likewise, the phasing of the sideslip trace following a step aileron input is no longer only "adverse" ($\psi_\beta \approx -180°$) or "proverse" ($\psi_\beta \approx -360°$), but can have any phase angle between 0 and $-360°$.

Recently, experimental work has been conducted (References F1, F5, F22, F72, F73, F74 and G10) to extend our understanding of the effect of parameters other than $\psi'_{\beta_{AS}}$. Areas of acceptable zero locations have been mapped as shown in Sketch 4. This work has shown that the amount of roll rate oscillation that a pilot will tolerate in step aileron rolls is highly dependent upon the position of the zero with respect to the Dutch roll pole of the $\frac{p}{\delta_{AS}}$ transfer function. The data show that the optimum angular location of the zero is in the lower left quadrant with respect to the pole, with generally decreasing levels of desirability as the zero is moved around the pole or away from the optimum point.



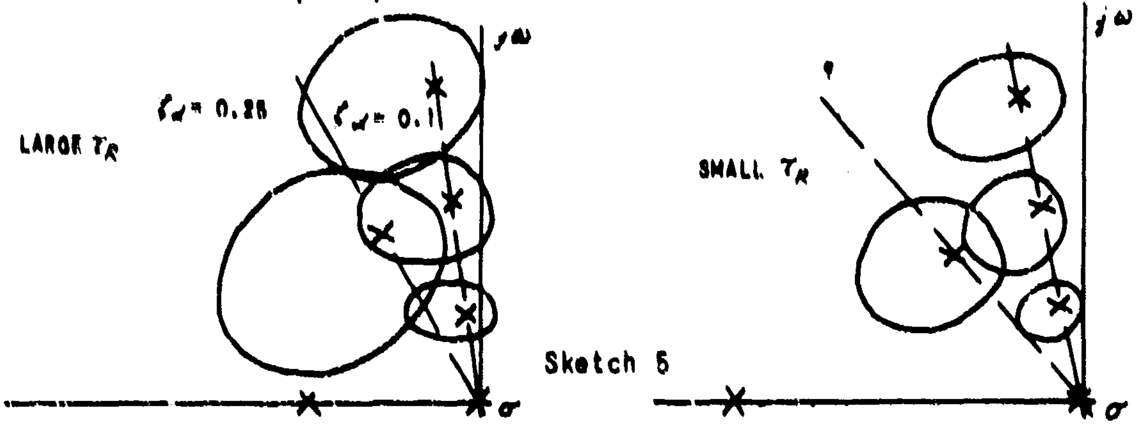
Sketch 4

Since the parameter $\frac{\omega_\phi}{\omega_{n_d}}$ only indicates the relative distance of the zero and the pole from the origin, $\frac{\omega_\phi}{\omega_{n_d}}$ does not adequately describe the physical situation. Therefore, the possibility of specifying a roll-sideslip coupling requirement by constructing an acceptable area of zero locations on the complex plane for the $\frac{p}{\delta_{AS}}$ transfer function was investigated.

One of the main shortcomings of this approach (and of using the parameter $\frac{\omega_\phi}{\omega_{n_d}}$) is that it requires knowledge of the location of the zero of the $\frac{p}{\delta_{AS}}$ transfer function. Industry was quite emphatic about their dislike of any requirement that was based on a preconceived transfer function format which involved very-difficult-to-measure parameters. Another shortcoming is that one would expect the acceptable area to be larger for high ζ_d than for low ζ_d, and larger for large τ_R than for small τ_R (assuming no degradation in flying qualities due to τ_R alone). Also, since the amount of Dutch roll excitation in roll is in some respects proportional to the ratio of:

$$\frac{\text{Distance from pole to zero}}{\text{Distance of pole from origin}}$$

the acceptable area would shrink in size as ω_{n_d} decreased. Thus, because of this dependence of size and shape of acceptable areas with the lateral-directional modal characteristics as indicated in Sketch 5, the requirement would be awkward to specify.



Sketch 5

Alternate Methods of Specification

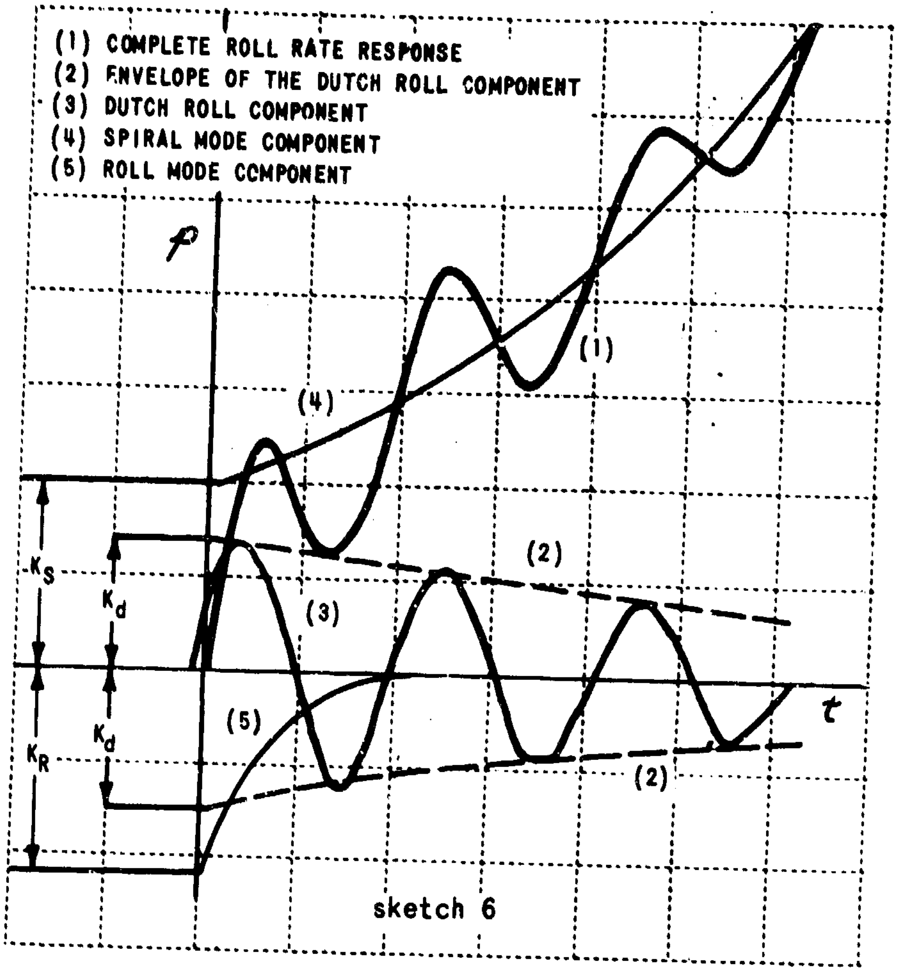
Analysis of time histories of experimental data shows that, for constant angular displacement of the "ω_ϕ" and "ω_{n_d}" roots, the degradation of flying qualities due to excitation of the Dutch roll in roll is roughly proportional to the ratio of:

$$\frac{\text{Dutch roll component of roll rate}}{\text{Steady state component of roll rate}}$$

that the pilot sees. In other words, the degradation in flying qualities is proportional to the amount of roll rate oscillation, p_{osc}, about some mean value of roll rate, p_{AV}. Thus the parameter p_{osc}/p_{AV} is a direct measure of the response which is degrading the flying qualities; and, because it is relatively easy to measure from flight test data and is relatively insensitive to sensor orientation, it is used as a parameter in specifying the requirement.

As was previously mentioned, in the $\frac{p}{\delta_{AS}}$ transfer function, the angular location of the zero with respect to the pole as well as the distance of the zero from the pole is of significance in determining or specifying aircraft flying qualities. In other words, the allowable $\frac{p_{osc}}{p_{AV}}$ for a given level of flying qualities is a function of the angular location of the associated zero with respect to the Dutch roll pole. Since the idea of specifying areas of acceptable zero locations was found to have shortcomings, another method of specifying the angular location of the zero with respect to the pole was investigated. This method consisted of using the phase angle, ψ_p, of the Dutch roll oscillation in roll rate following a step aileron input.

To see how the phase angle, ψ_p, relates to the other modal parameters, consider the roll rate response, $p(t)$, to a step aileron input as shown graphically in Sketch 6.



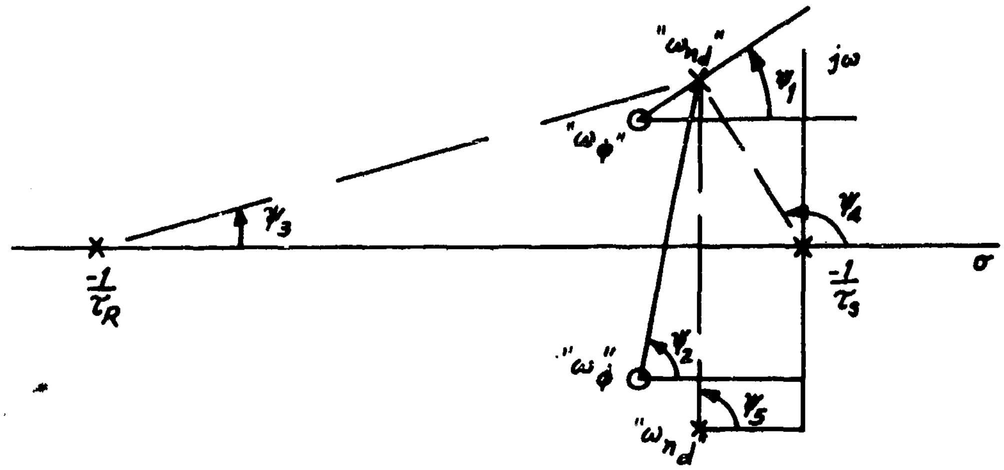
sketch 6

$$\left.\frac{p(t)}{L'_{\delta_{AS}}\delta_{AS}}\right|_{STEP} = K_S e^{-\frac{t}{\tau_s}} + K_R e^{-\frac{t}{\tau_R}} + K_d e^{-\zeta_d \omega_{n_d} t} \cos\left(\omega_{n_d}\sqrt{1-\zeta_d^2}\, t + \psi_p\right)$$

Transforming from the time domain to the frequency domain:

$$\left.\frac{p(s)}{\delta_{AS}}\right|_{STEP} = \frac{L'_{\delta_{AS}}\left(s^2 + 2\zeta_\phi \omega_\phi s + \omega_\phi^2\right)}{\left(s+\frac{1}{\tau_s}\right)\left(s+\frac{1}{\tau_R}\right)\left(s^2 + 2\zeta_d \omega_{n_d} s + \omega_{n_d}^2\right)}$$

The poles and zeros are shown in Sketch 7.



Sketch 7

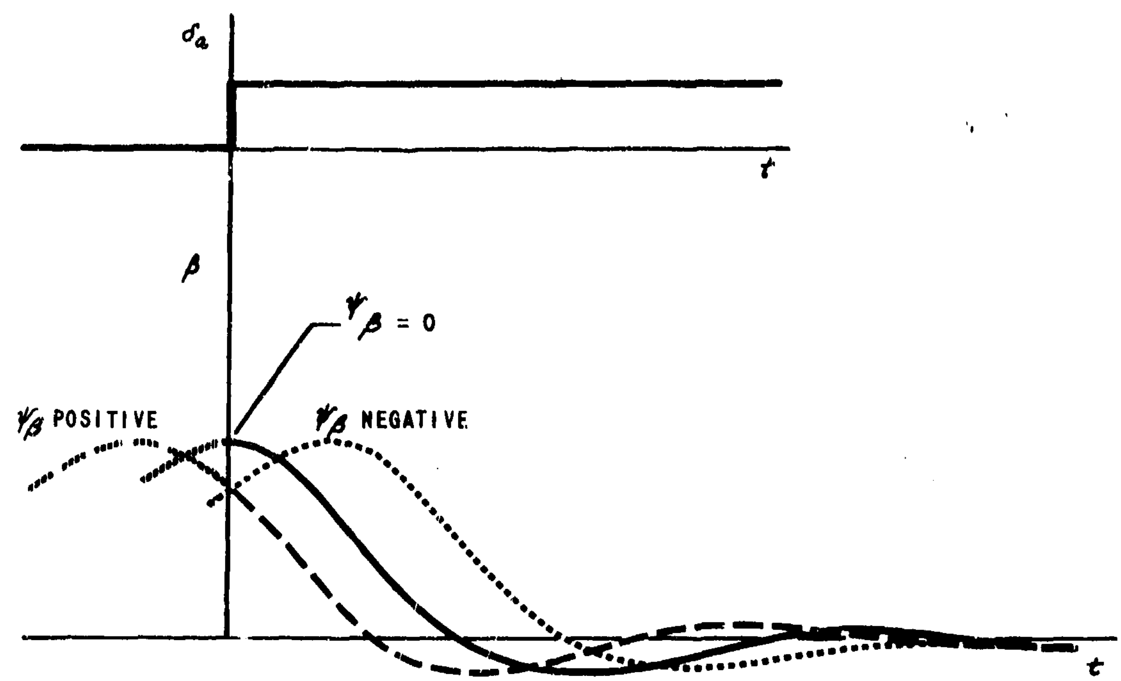
$$\psi_p = \psi_1 + \psi_2 - \psi_3 - \psi_4 - \psi_5$$

From the equation for ψ_p, it can be seen that for given Dutch roll, spiral and roll mode characteristics, and for the zero located relatively close to the pole $(\psi_2 \approx \psi_5)$, the phase angle ψ_p is directly related to ψ_1, the angular location of the zero with respect to the pole. Thus, it is possible to specify the angular zero location through the phase angle of the Dutch roll component of roll rate. A shortcoming of this method is that there is not a unique ψ_p for a given zero location, ψ_1, since ψ_p is directly related to $1/\tau_R$ and $1/\tau_s$. Thus a requirement employing the parameter ψ_p would be awkward to specify and would require knowledge of the roll mode and spiral mode roots.

Another, and more straightforward, way in which the angular location of the zero can be specified is through measurement of the phase angle of the Dutch roll component of sideslip following a step aileron input, where

$$\left.\frac{\beta_d}{\delta_{AS}}\right|_{STEP} = C_d \, e^{-\zeta_d \omega_{n_d} t} \cos\left(\omega_{n_d}\sqrt{1-\zeta_d^2}\; t + \psi_\beta\right)$$

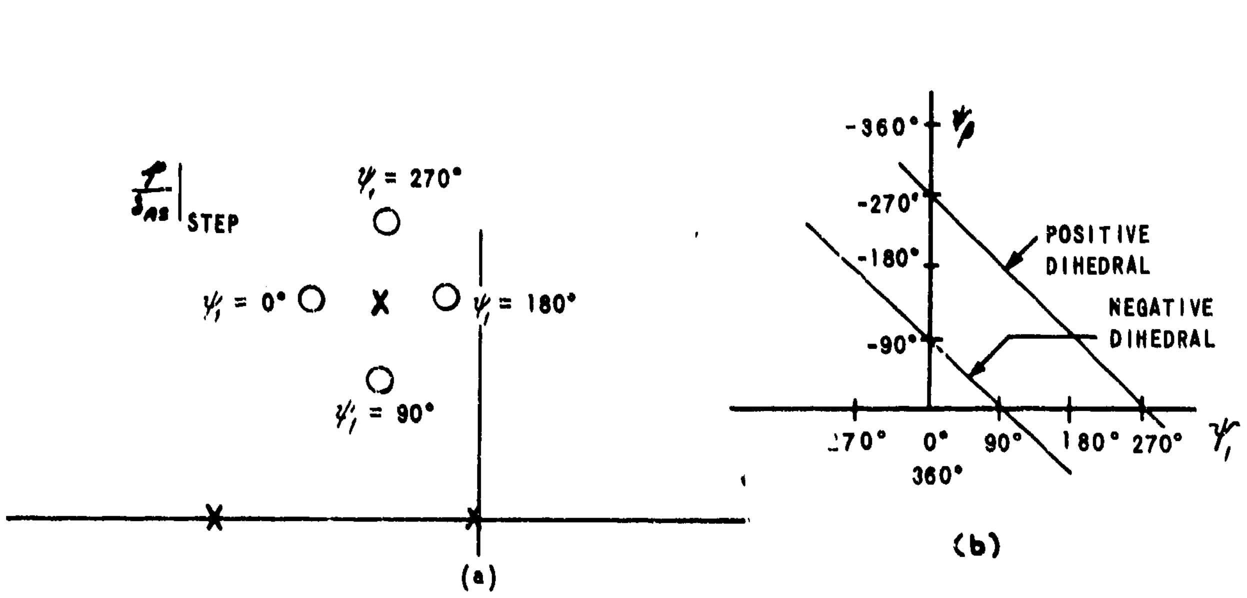
Sketch 8 portrays the Dutch roll component of sideslip response following a step aileron input for several values of ψ_β. Although the amplitude of the Dutch roll oscillation is the same for each of the responses shown, the phasing of the responses is different. This phasing of the Dutch roll oscillation in sideslip following a step aileron input is defined by ψ_β.



SKETCH 8

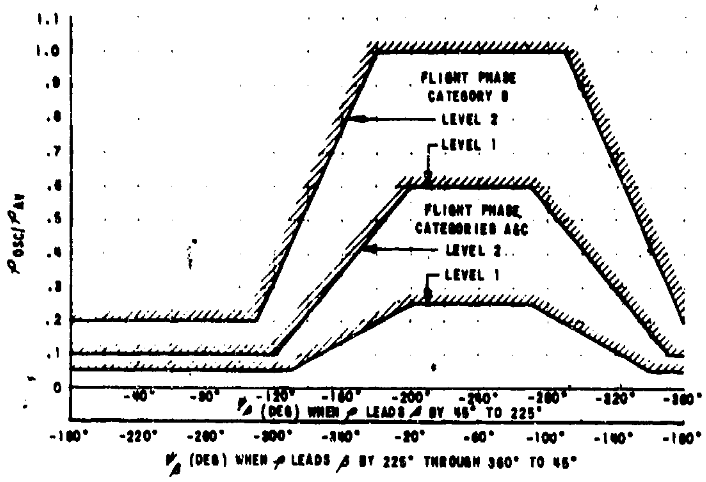
As will be shown, the angular position of the zero relative to the Dutch roll pole of the p/δ_{AS} transfer function is directly related to the phase angle of the Dutch roll oscillation in sideslip, ψ_β , relatively independent of roll mode and spiral mode characteristics for a wide range of stability derivatives. In other words, for each angular location of the zero with respect to the Dutch roll pole, there is a relatively unique value of ψ_β. That this is so is perhaps indicative of the fundamental importance of sideslip, not only as an important flying qualities parameter per se, but as a parameter that is basic to the coupling that exists in the lateral-directional motions during rolling maneuvers.

The substantiation of the relatively unique relationship between the $\frac{p}{\delta_{AS}}$ transfer function zero location and the phase angle of the Dutch roll in sideslip is given in Appendix VC. In simple terms, phasing in roll rate is related to phasing in sideslip through the $\left|\frac{p}{\beta}\right|_d$ response ratio. It develops that in ψ_β , angular contributions from the $\left|\frac{p}{\beta}\right|_d$ response ratio almost exactly cancel out the angular contributions of the roll and spiral mode roots to ψ_β without introducing any other appreciable angular contributions. This cancellation occurs for a wide range of lateral-directional stability derivatives. The relationship between the phase angle of the Dutch roll in sideslip and the p/δ_{AS} transfer function zero locations, developed in Appendix VC, is shown in Sketch 9.



Sketch 9

The parameters ϕ_{osc}/ϕ_{AV} and ψ_β have been used to specify criteria as a function of Flight Phase Category and Level as shown in Sketch 10 (Reference A1, Paragraph 3.3.2.2.1, Figure 4).



Sketch 10

It should be noted that this figure has two ψ_β scales, one for positive dihedral (ϕ leads β by 45° to 225°) and the other ~~for negative dihedral~~ (ϕ leads β by 225° through 360° to 45°). In Reference A1, dihedral is defined by the parameter $\measuredangle \phi/\beta$, since dihedral as currently used in flying qualities work seems to be an ambiguous and ill-defined parameter. The rationale behind the use of two scales and a discussion of the relationship between dihedral effect and $\measuredangle \phi/\beta$ is presented in Appendix VC.

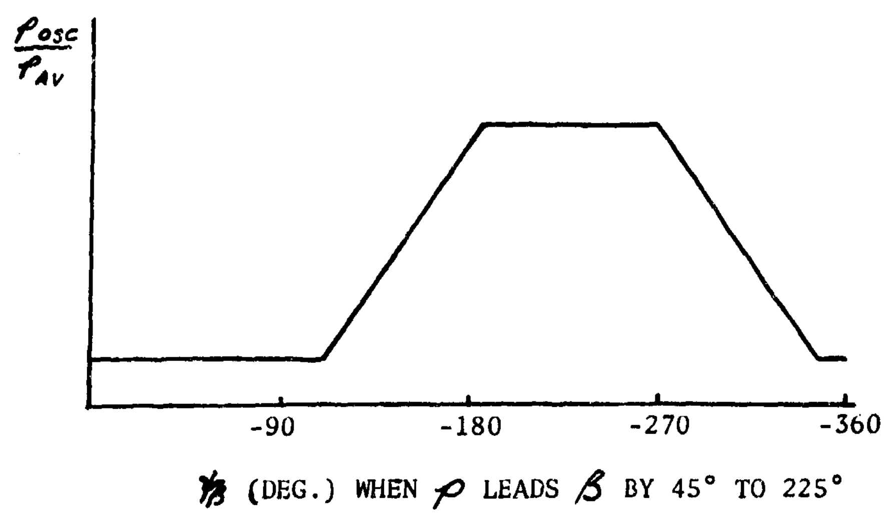
In this context, "positive dihedral" means negative $L'_\beta + Y_\beta L'_v \approx L'_\beta$. Note that

$$L'_\beta = \frac{L_\beta + \frac{I_{xz}}{I_x} N_\beta}{1 - \frac{I_{xz}^2}{I_x I_z}}$$

It should also be noted that the value, or even the sign, of L'_β cannot always be determined from steady rudder-pedal-induced sideslips. Not only are product of inertia effects absent in steady sideslips, but also the control deflections are affected by control cross-coupling derivatives. On the other hand, $\measuredangle p/\beta$ generally is a good discriminator of the sign of dihedral.

Effect of ψ_β on Flying Qualities

Since ψ_β (the phase angle in a cosine representation of the Dutch roll component of sideslip, negative for a lag) is a rather abstract parameter, it is well to consider its physical implications and significance to the piloting of an airplane. Very simply, ψ_β can be considered as an indicator of those closed-loop stability characteristics of an airplane that are related to the lateral-directional coupling derivatives; and of the difficulty a pilot will experience in coordinating a turn entry. Further clarification can be obtained by discussing the variation of the specified values of p_{osc}/p_{AV} with ψ_β for positive dihedral as shown in Sketch 10. (Additional discussions are presented in Appendix VC.)



Sketch 11

From Sketch 11 it can be seen that the ratio of roll rate oscillation to steady state roll rate can be much greater for some values of ψ_β than for others. Specifically, the specified values of p_{osc}/p_{AV} for $0° \geq \psi_\beta \geq -90°$ are far more stringent than for $-180° \geq \psi_\beta \geq -270°$. There are at least three reasons why this is so:

(a) differences in closed-loop stability

(b) differences in difficulty of rudder coordination

(c) differences in average roll rate.

From a root locus analysis, it can be shown that when the zero of the p/δ_{AS} transfer function lies in the lower left quadrant with respect to the Dutch roll pole, $(-180° \geq \psi_\beta \geq -270°)$, the closed-loop damping increases when the pilot closes a bank angle error to aileron loop. The reason for this in physical terms is that when the zero lies in the lower left quadrant, aileron inputs proportional to bank angle errors generate yawing accelerations that tend to damp the Dutch roll oscillations. Thus the Dutch roll damps out more quickly closed-loop than open-loop, so a pilot will tend to tolerate somewhat more p_{osc}/p_{AV}. Conversely, it can be shown that when the zero lies in the upper right quadrant with respect to the Dutch roll pole $(0° \geq \psi_\beta \geq -90°)$, the closed-loop damping decreases when the pilot applies aileron inputs proportional to bank angle error. The physical explanation for this is that aileron inputs generate yawing accelerations that tend to excite or sustain the Dutch roll oscillations. Thus the Dutch roll damps out less quickly closed loop than open loop, and can even go unstable closed loop; that is, pilot-induced oscillations can result. In this case a pilot's tolerance of p_{osc}/p_{AV} tends to reduce.

Significant differences in the p_{osc}/p_{AV} requirements also occur because of differences in difficulty of rudder coordination while performing coordinated turn entries or exits. For $-180° \geq \psi_\beta \geq -270°$, normal coordination may be effected, that is, right rudder pedal for right rolls. Thus, even if large roll rate oscillations occur in rudder-pedal-free rolls (the conditions under which the p_{osc}/p_{AV} tests are conducted), sideslip oscillations can be readily minimized by use of rudder pedals so the roll rate oscillations do not occur. On the other hand, for $0° \geq \psi_\beta \geq -90°$ it is necessary to cross control to effect coordination, that is, left rudder pedal with right aileron. Since pilots do not normally cross control (and if they must, have great difficulty in doing so) for $0° \geq \psi_\beta \geq -90°$, oscillations in sideslip, and hence oscillations in roll rate, either go unchecked or are amplified by the pilot's efforts to coordinate.

The third reason why the p_{osc}/p_{AV} requirements vary so significantly with ψ_β is that the average roll rate, p_{AV}, for a given aileron input, varies significantly with ψ_β. For positive dihedral, adverse yaw-due-to aileron $(\psi_\beta \approx -180°)$ tends to decrease average roll rate whereas proverse yaw-due-to aileron $(\psi_\beta \approx 0°)$ tends to increase average roll rate. As a matter of fact, proverse yaw-due-to aileron is sometimes referred to as "complementary yaw" because of this augmentation of roll effectiveness. Thus, for a given amplitude of p_{osc} , p_{osc}/p_{AV} will be greater at ψ_β = -180° than it will be at ψ_β = 0°.

In summary, the parameters that have been chosen in Paragraphs 3.3.2.2 and 3.3.2.2.1 to describe and specify the coupling that exists between sideslip and roll for moderate to high $|\phi/\beta|_d$ response ratios are p_{osc}/p_{AV} and ψ_β. These parameters were chosen as being measurable parameters which most simply, directly, and accurately reflect the important flying qualities considerations. The measurements are taken from the p and β traces which are relatively insensitive to sensor orientation and type of input.

Requirements for Large-Amplitude Rolls

The preceding discussion presents the rationale which supports the requirement of 3.3.2.2.1. The data were obtained for small aileron steps, and practical problems arise when large inputs are used, so 3.3.2.2.1 applies to small inputs only. However, a certain degree of precision is needed for large, possibly open-loop as well as small, closed-loop inputs; so an additional requirement (3.3.2.2) pertaining to large control inputs has been specified. The numerical values of the roll rates specified in 3.3.2.2 were transformed from the values of p_{osc}/p_{AV} for "adverse yaw" in 3.3.2.2.1. Thus, the requirements of 3.3.2.2 and 3.3.2.2.1 are essentially identical for airplanes with "adverse yaw". However, the requirement of 3.3.2.2 is far more lenient than the requirement of 3.3.2.2.1 for airplanes with "proverse yaw."

The 90° roll amplitude limit of 3.3.2.2 arbitrarily matches the 45°-to-45°-bank maneuver of MIL-F-8785 for a similar requirement. Obviously fighters roll through larger angles. In that respect it would have been rational to make the amplitude limit a function of Class or Flight Phase. However, the design implications could be rather drastic. For that reason, application is restricted to maneuvers of roll amplitude less than 90°.

Data Upon Which Requirements are Based

The data upon which these curves are based are presented in Figures 1 through 6 as follows:

Figure 1	- Flight Phase Category A (Reference F1)
Figure 2	- Flight Phase Category B (Reference F5)
Figure 3	- Flight Phase Category B (Reference F22)
Figure 4	- Flight Phase Category C (Reference G10)
Figure 5	- Data on some Class IV airplanes
Figure 6	- Data on some Class III airplanes

A sample of the raw data from Reference F1 from which the reduced data of Figure 1 were obtained is presented in Figures 7 through 15. These data are typical in that, although smooth curves can generally be drawn through most of the points, some ratings were off the expected value. Thus, no matter how most data are presented, there will always be some points that cannot be correlated. It should be noted from Figures 1 through 4 that, although the p_{osc}/p_{AV} criteria cannot be expected to fully account for all interaction effects, the correlation of the data, which encompass a wide range of lateral-directional modal characteristics and coupling effects, is in general quite good.

From Sketch 9 it can be seen that the Level 1 and 2 boundaries are the same for Category A Flight Phases as for Category C Flight Phases; yet comparison of Figure 1 (Category A Flight Phase data) with Figure 4 (Category C Flight Phase data) reveals that the Level 1 and Level 2 boundaries correspond to different pilot ratings on the two figures. The reason for this is that different pilot rating scales were used in the two programs and the degree of goodness of the base configurations was different. In Reference F1, the CAL rating scale was used and the base configurations were good; whereas in Reference G10, the Cooper rating scale was used and the base configurations were marginally satisfactory.

Although there are somewhat less data upon which to base the Category B Flight Phase boundaries, what data there are indicate that the Level 1 and 2 boundaries should be the same shape as for Category A and C Flight Phases and should be roughly twice as lenient. Since $\frac{p_{osc}}{p_{AV}} \approx 1$ approaches roll rate reversal, the Level 2 requirement for "adverse yaw due-to-aileron" has been made to correspond with the MIL-F-8785 requirement which prohibits roll rate reversals during rudder-free rolls.

Examination of Figure 3 reveals that several configurations have a poorer rating than is indicated by their location with respect to the Level 1 and Level 2 boundaries. This results from a limitation imposed by the definition of p_{osc}, which is a measure of the observed oscillation in roll rate. When there is a strong spiral, the roll rate time history may not contain even two peaks even though the Dutch roll has been excited. Also, for heavily damped, high-frequency Dutch rolls, the Dutch roll excitation can be essentially damped out by the time the roll mode has expired, so no p_{osc} will be measured. This problem is partially overcome by requiring that p_{osc} be measured over only the first two peaks when $\zeta_d > 0.2$. Reference F75 got around this problem by using as the measure of roll rate oscillation, the component of roll rate due to Dutch roll at the first peak of the Dutch roll, p_1. This parameter was not used in Reference A1 in an effort to simplify analysis and testing for the requirement, but has resulted in a requirement which can be too lenient for some configurations. Thus, although the requirement does not completely cover all aspects of roll-sideslip coupling, it is believed that it takes a significant step toward adequate specification of this area.

Since almost no data exist on strong roll-sideslip coupling with negative dihedral, it was necessary to specify the negative-dihedral requirement through analogy with the positive-dihedral requirements previously described. Reference H8 did provide some data, however, which are presented in Figure 16 for comparison with the roll rate oscillation requirement. The program of Reference H8 investigated lateral-directional instabilities relating to the X-15. In the course of this investigation, configurations were simulated either in flight or in a fixed-base simulator, or in both, that had:

(a) Positive dihedral, proverse yaw due-to-aileron
(b) Positive dihedral, adverse yaw due-to-aileron
(c) Negative dihedral, proverse yaw due-to-aileron
(d) Negative dihedral, adverse yaw due-to-aileron

These configurations, which all had very light Dutch roll damping and large $\left|\frac{\phi}{\beta}\right|_d$ response ratios, are plotted in Figure 16. The parameters $\frac{p_{osc}}{p_{av}}$, ψ_β and $\angle p/\beta$ were obtained from time histories of the responses to step aileron inputs. Configuration (a), which falls well outside the Level 2 boundary of Figure 16, was uncontrollable: "attempts by the pilot to control the oscillation resulted in excursions of increasing magnitude for both sideslip angle and the roll rate."

Configuration (b), which also falls well outside the Level 2 boundary of Figure 16, was unacceptable because of the oscillatory response. "However, it is significant that the pilot was able to control the aircraft, and, in fact, damp the oscillations when they occurred using only normal aileron control movements."

Configuration (c), which falls in the "good" area of Figure 16, was controllable and "it was found that attempts to control the roll angle in a normal manner also helped to reduce the excursions of the sideslip angle."

Configuration (d), which falls in an area of marginal acceptability on Figure 16, was uncontrollable because of pilot-induced oscillations.

Thus, with the possible exception of Configuration (d), the pilot comments pertaining to the configurations were compatible with those expected from their roll-sideslip coupling characteristics as indicated by Figure 16. Although Configuration (d) was rated worse than would be expected from the measured roll-sideslip coupling characteristics, the fact that the point fell in the region of Figure 16 where the amount of allowable roll rate oscillation changes rapidly with ψ_β would indicate that the flying qualities of the configuration are sensitive to small changes in ψ_β. For example, if ψ_β were only 30 degrees greater (or if the peaks on the time histories presented differed by only 0.2 seconds from those of the configuration flown), the roll-sideslip coupling characteristics as indicated by Figure 16 could be completely compatible with the pilot comments.

There is no quantitative requirement for Level 3 except a prohibition of roll reversal, for lack of sufficient data. While an extremely oscillatory roll response can become unmanageable, an airplane may still be flyable in the small, gentle maneuvers that might be involved in emergency termination of a Category A Flight Phase, return, and landing (1.5). The qualitative requirements of 3.3.3 and 3.3.2.1 do hold at Level 3, with appropriate interpretation of "acceptable."

Comparison with Other Data and Criteria

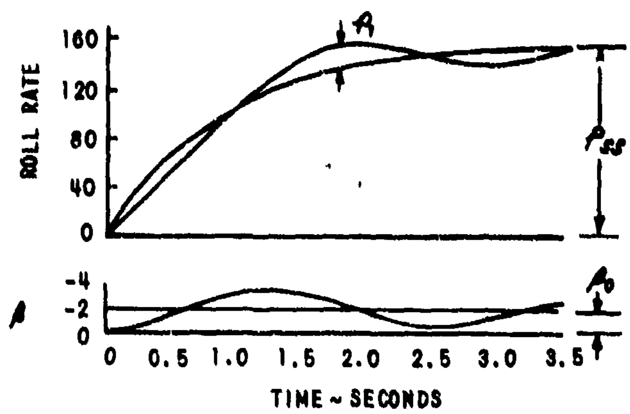
Although the Reference A1 criteria were generated directly from the correlation of roll rate and sideslip time history characteristics with pilot ratings, it is possible to transform the criteria for specific conditions into other more familiar - but less direct - formats. Thus, in order to look at the criteria in more familiar ways and in order to compare them with existing data, an example of how the lower two curves of Sketch 10 (Reference A1, Figure 4) transform into areas of satisfactory or acceptable zero locations on the complex plane for the p/δ_{AS} transfer function is given in Figures 17 through 20. From Figures 17 and 18 it can be seen that as ζ_d increases, the areas of satisfactory and acceptable p/δ_{AS} transfer function zero locations increase. From Figures 17, 19 and 20 it can also be seen that as ω_{n_d} increases, so do the areas of satisfactory and acceptable p/δ_{AS} transfer function zero locations. It should further be noted that the rate of increase of area with ω_{n_d} is such that $\frac{\omega_\phi}{\omega_{n_d}}$ at the Level 1 and Level 2 boundaries remains relatively constant over the frequency range investigated.

The Reference A1 requirements may be compared with criteria in the form of $\frac{\omega_\phi}{\omega_{n_d}}$ by considering the intercept of the Level 1 and Level 2 boundaries with a line drawn radially through the Dutch roll pole. For the cases with $\zeta_d = 0.1$, the Reference A1 requirements indicate that for Flight Phase Categories A and C, Level 1, $0.8 \approx \frac{\omega_\phi}{\omega_{n_d}} \approx 1.1$. For Level 2, $0.6 \approx \frac{\omega_\phi}{\omega_{n_d}} \approx 1.2$. By comparing Figures 17 and 18, it can be seen that the satisfactory and acceptable ranges of $\frac{\omega_\phi}{\omega_{n_d}}$ would be larger for larger damping ratios. These trends of $\frac{\omega_\phi}{\omega_{n_d}}$ with ω_{n_d} and ζ_d, and also with $\left|\frac{\phi}{\beta}\right|_d$, are consistent with the results of Reference F22 for moderate and large $\left|\frac{\phi}{\beta}\right|_d$ response ratios, as presented in terms of these parameters in Figures 21 through 26.

To illustrate the criteria in more physical terms, time histories from the Level 1 boundary of Figure 27 are presented in Figure 28. The traces 1 - 10 of Figure 28 correspond to $\frac{p}{\delta_{AS}}$ transfer-function zero locations as shown in Figure 27. From Figure 28 can be seen how the shape of the roll rate response changes, through different phasing and excitation of the Dutch roll mode, as the $\frac{p}{\delta_{AS}}$ transfer-function zero moves clockwise around the Dutch roll pole. Although the responses vary greatly in character, according to available data they are all approximately equally acceptable to the pilot.

The criteria were also compared with criteria proposed in Reference F75. This report recommended that $|p_1| / p_{ss} \leq 0.045$ for satisfactory flying qualities, $|p_1| / p_{ss} \leq 0.085$ for acceptable flying qualities, and $|p_1| / p_{ss} \leq 0.18$ for controllability, where $|p_1| / p_{ss}$ is the ratio of the amplitude of the roll oscillation at the first overshoot, designated as $|p_1|$, divided by the steady-state roll rate, p_{ss}. This parameter is very similar to the proposed parameter p_{osc}/p_{AV} for light Dutch roll damping, so a direct comparison can be made. The parameter p_1/p_{ss} was not used however, since data reduction problems were considered to be too severe.

Sketch 12 (Figure 8 of Reference F75) is reproduced to indicate the parameters p_1 and p_{ss}.



TYPICAL RESPONSE TO STEP INPUT OF LATERAL CONTROL

Sketch 12

From Sketch 12, a typical response in p and β is shown from which it can be determined that $\psi_\beta \approx 0°$. Comparing the Reference F75 criteria with the Reference A1 criteria at $\psi_\beta = 0°$ for precise and maneuvering tasks - the type of tasks performed in Reference F75 - for the Level 1 boundary (satisfactory boundary), $p_{osc}/p_{AV} = 0.05$, whereas $|p_1| / p_{ss} = 0.045$. For the Level 2 boundary (acceptable boundary), $p_{osc}/p_{AV} = 0.10$, whereas $|p_1|/p_{ss} = 0.085$. There is no Level 3 boundary (controllability limit); however, what little data there are for $\psi_\beta \approx 0$ are consistent with a $|p_1|/p_{ss}$ of 0.18. It can thus be seen that the correlation is excellent and, moreover, that the Reference A1 criteria go much farther than do the Reference F75 criteria.

The data of Reference F21, which are applicable to Flight Phase Category A, were also examined and are presented in Figure 29. The data for the $\zeta_d = 0.22$ and 0.1 configurations were analyzed and replotted in Figure 30 in the ψ_β vs p_{osc}/p_{AV} format. From Figure 30 it can be seen that the data correlate well with the Reference A1 criteria. Although as indicated in Figure 31, the data of Reference F21 do not correlate very well with the data of Reference F75 on the basis of the $|p_1| / |p_{ss}|$ parameter, when the phase angle of the Dutch roll is accounted for as in the Reference A1 criteria, the two sets of data correlate quite well.

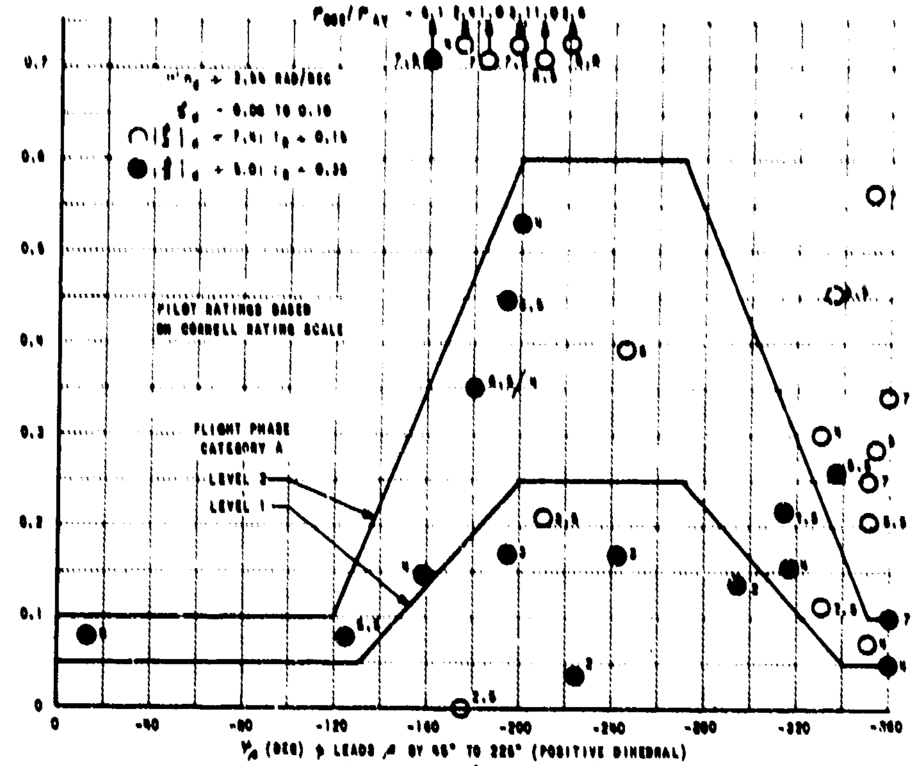


Figure 1 (3.3.2.2)
FLIGHT PHASE CATEGORY A DATA (FROM REFERENCE F1)

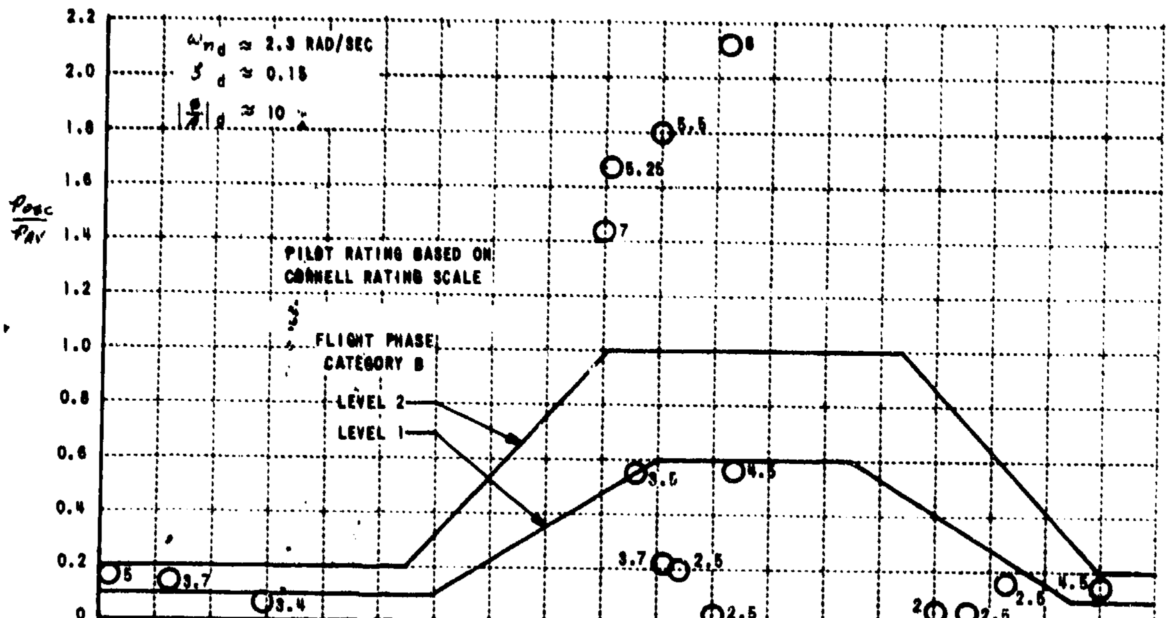


Figure 2 (3.3.2.2)
FLIGHT PHASE CATEGORY B DATA (FROM REFERENCE F5)

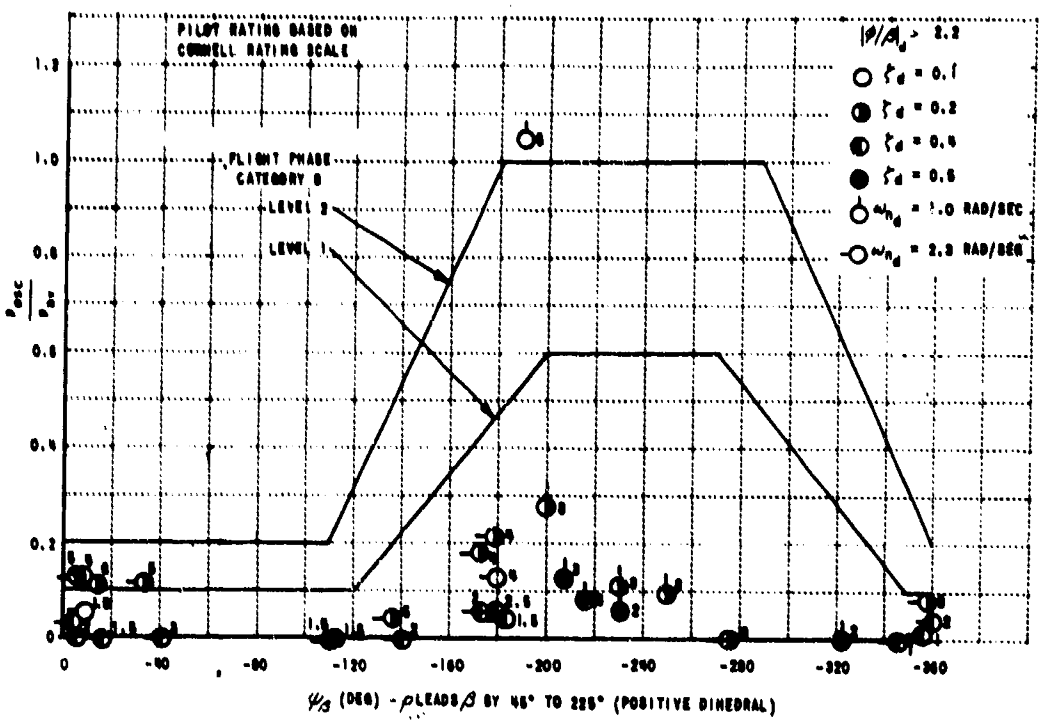


Figure 3 (3.3.2.2)
FLIGHT PHASE CATEGORY B DATA (FROM REFERENCE F22)

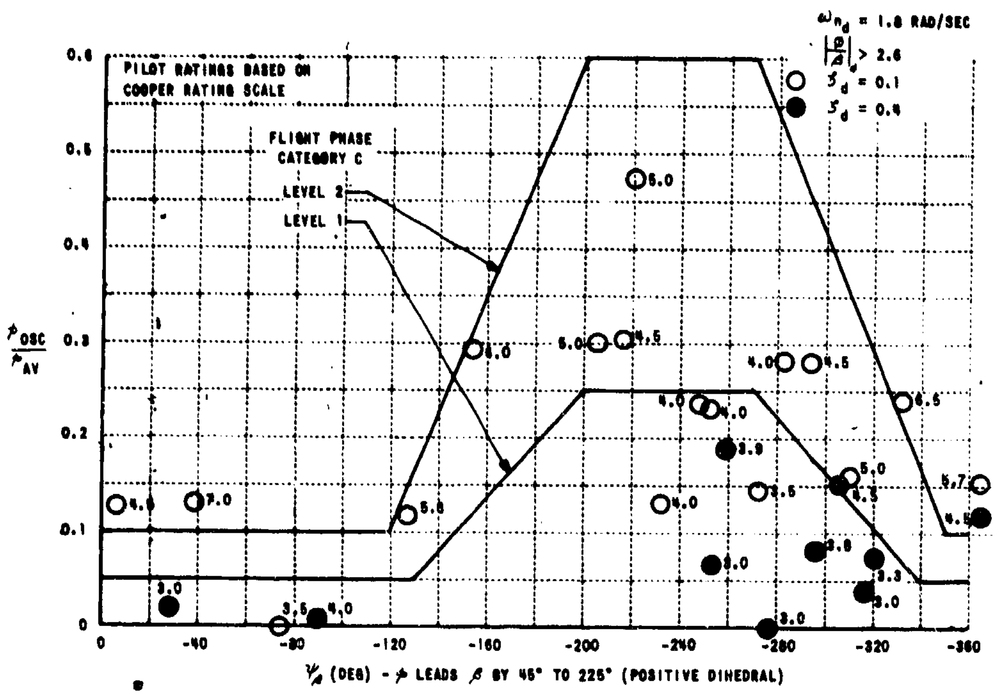


Figure 4 (3.3.2.2)
FLIGHT PHASE CATEGORY C DATA (FROM REFERENCE G10)

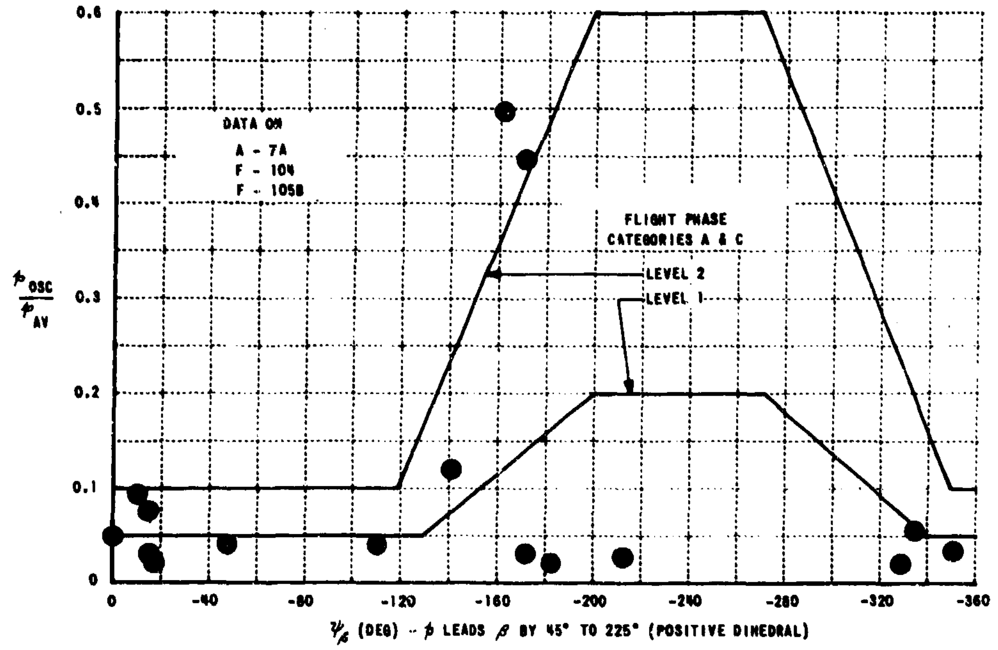


Figure 5 (3.3.2.2)
DATA ON SOME CLASS IV AIRPLANES

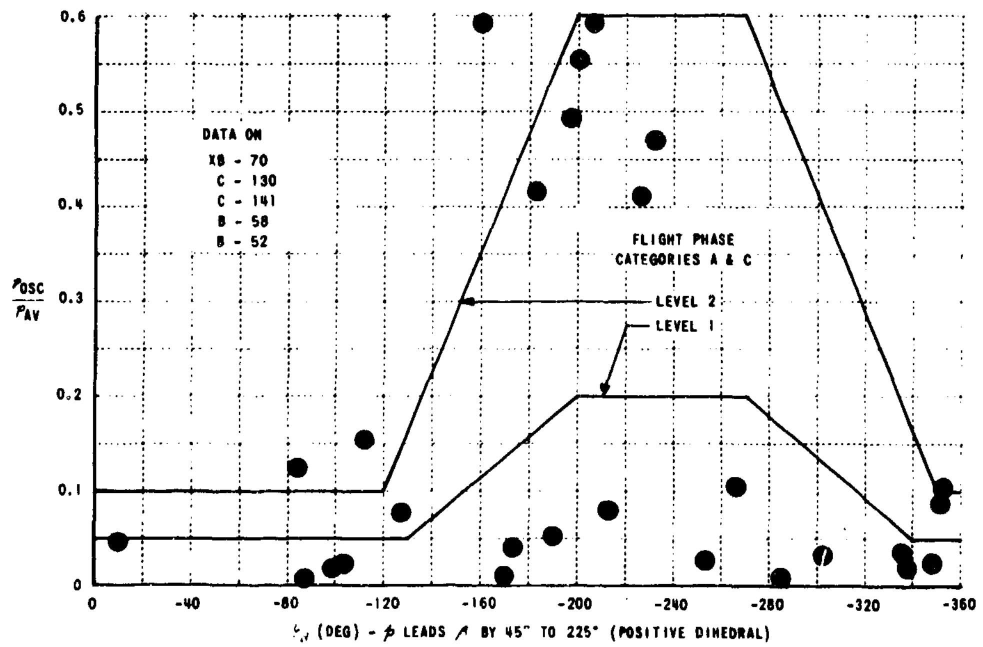


Figure 6 (3.3.2.2)
DATA ON SOME CLASS III AIRPLANES

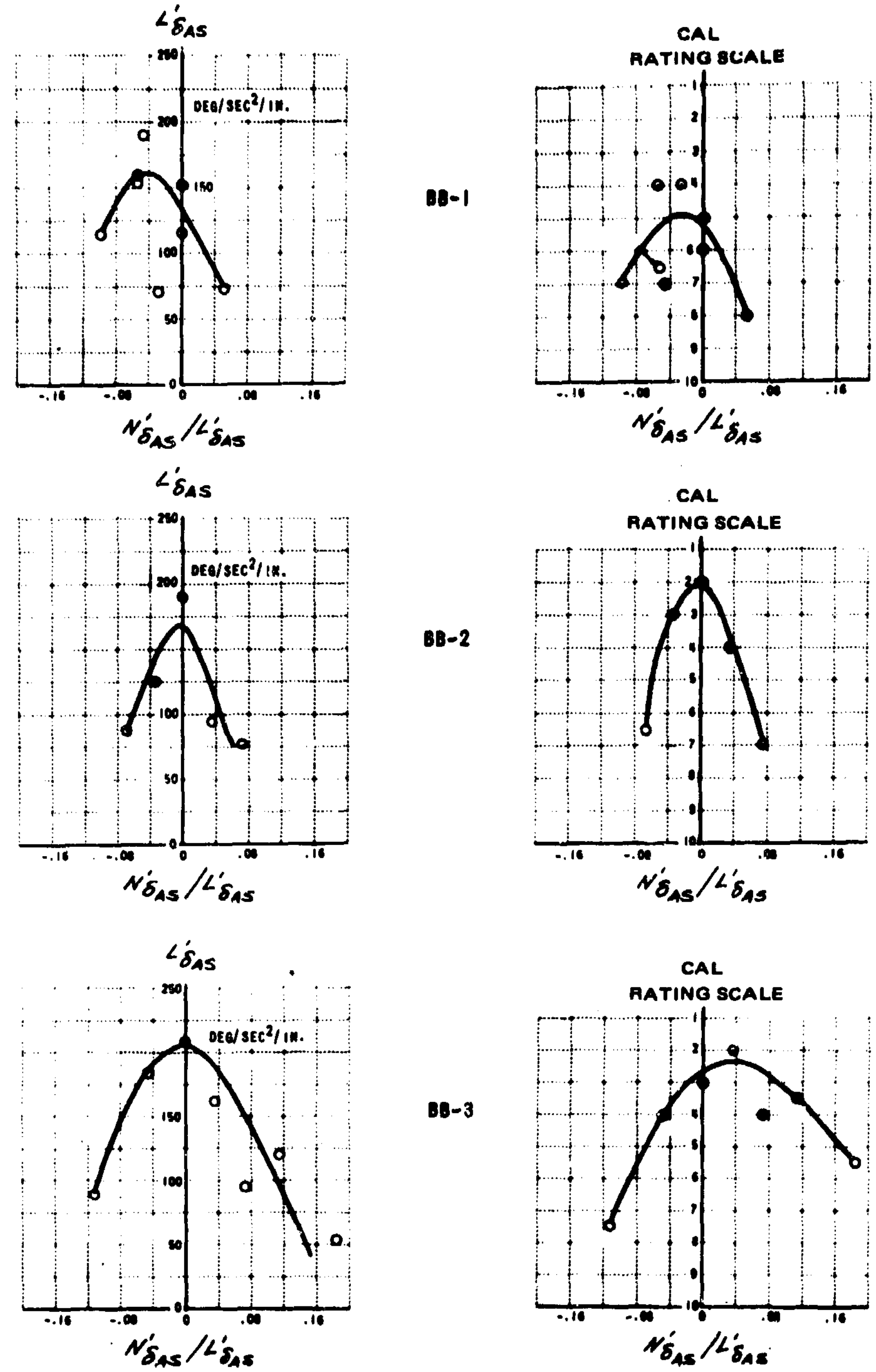


Figure 7 (3.3.2.2)

PILOT RATINGS AND OPTIMUM AILERON SENSITIVITY

(MEDIUM$|\phi/\beta|_d$, MEDIUMτ_R) (FROM REFERENCE F1)

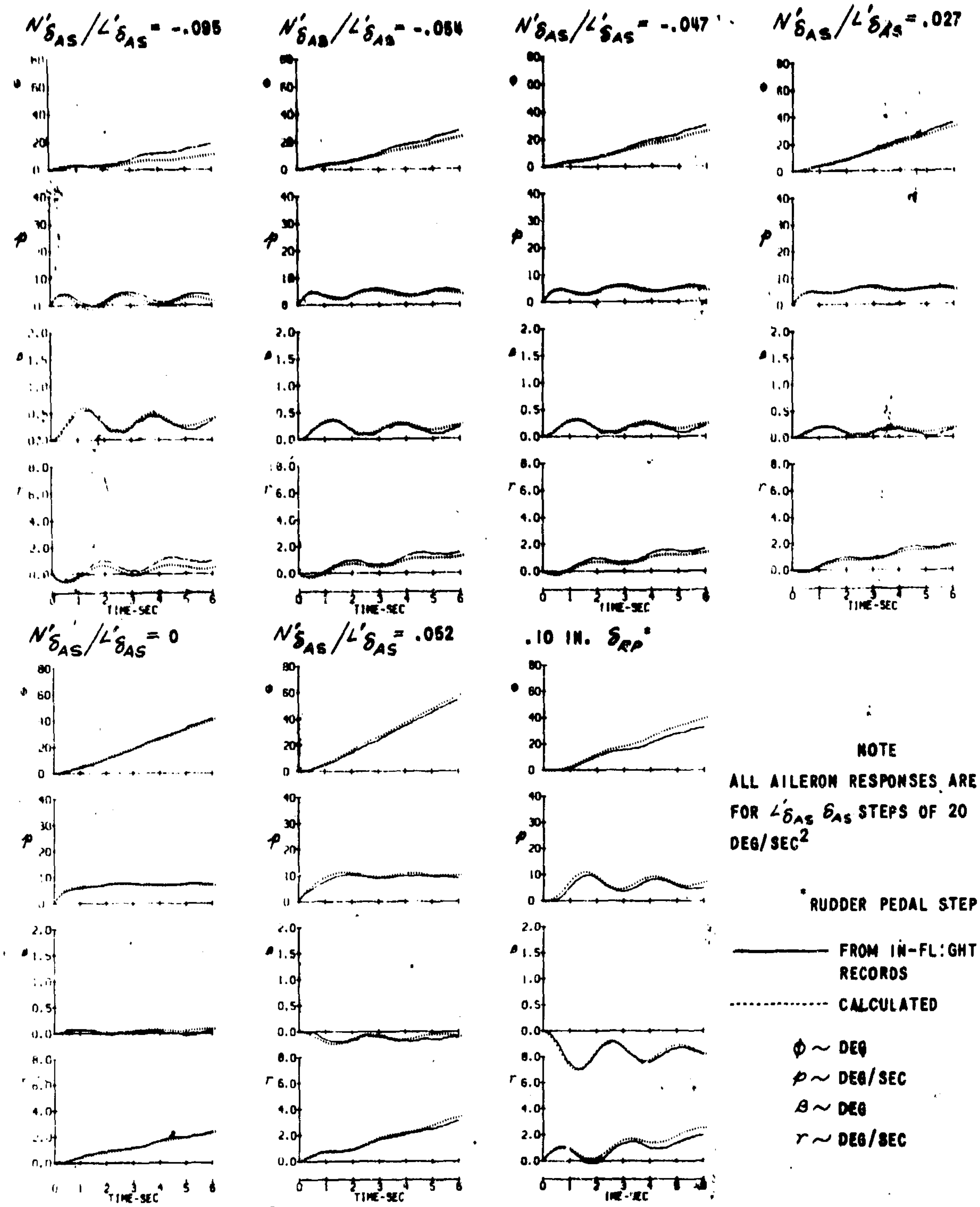


Figure 8 (3.3.2.2)
TRANSIENT RESPONSES FOR BB-1 CONFIGURATIONS (FROM REFERENCE F1)

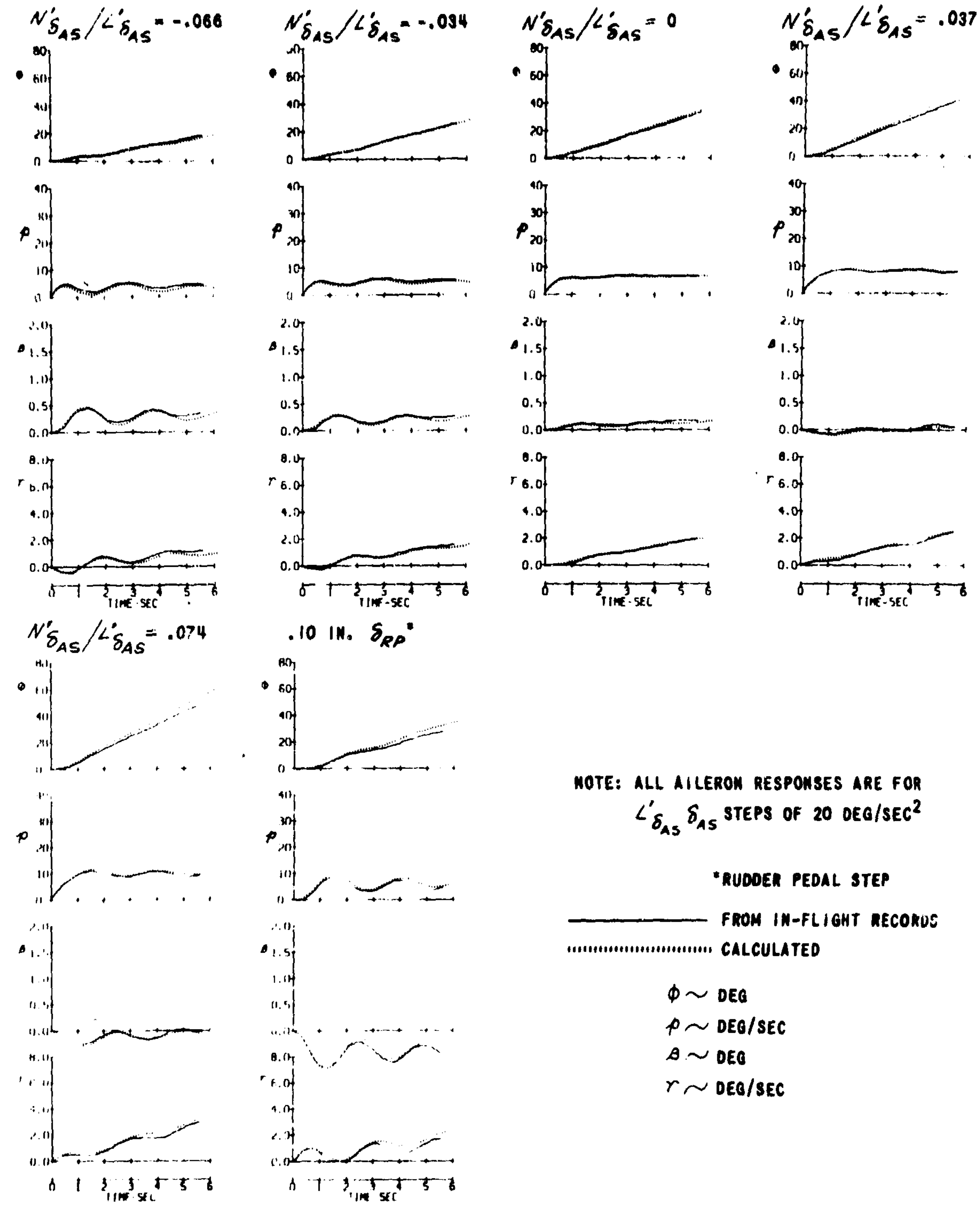


Figure 9 (3.3.2.2)
TRANSIENT RESPONSES FOR BB-2 CONFIGURATIONS (FROM REFERENCE F1)

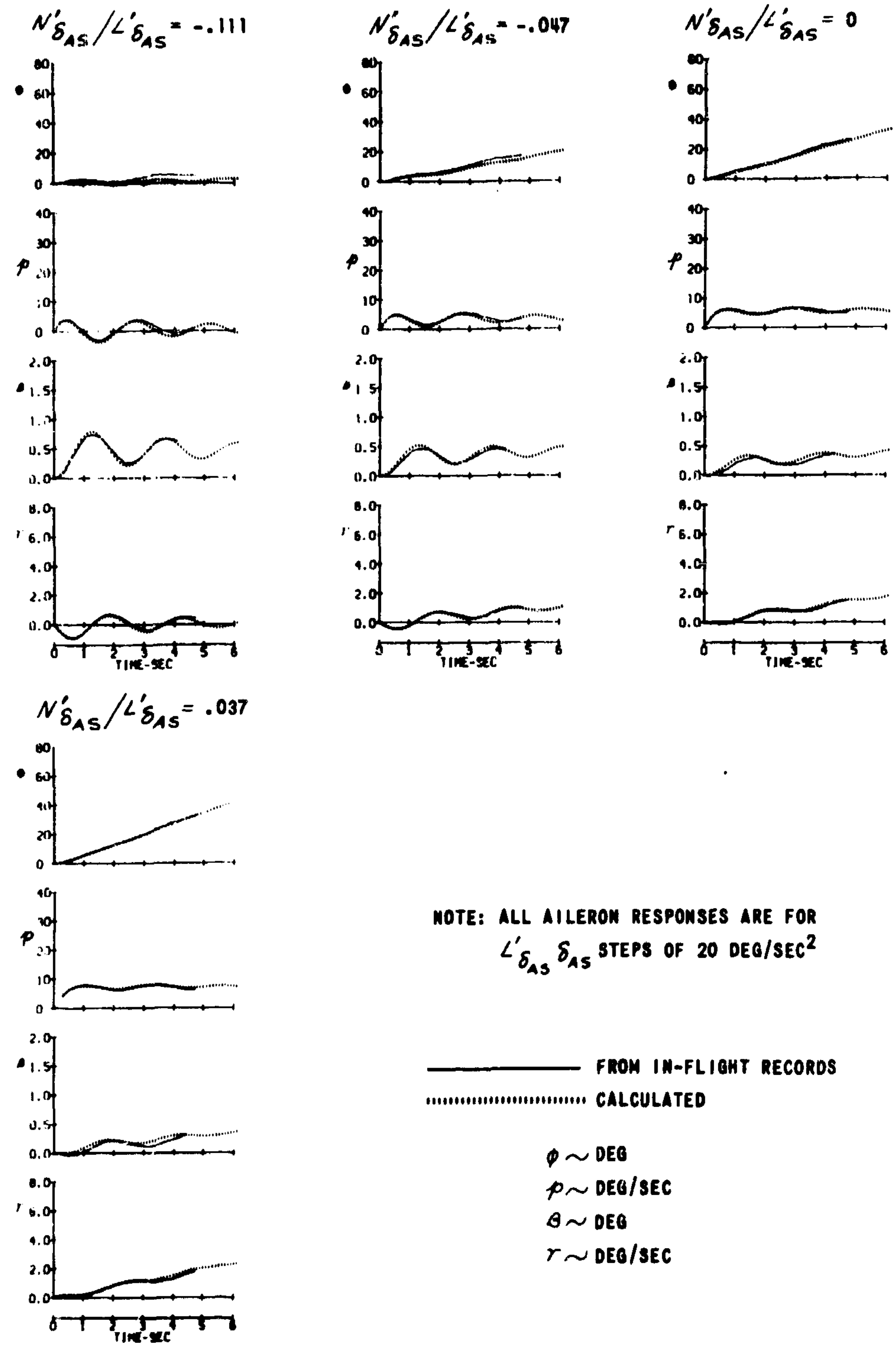


Figure 10 (3.3.2.2)
TRANSIENT RESPONSES FOR BB-3 CONFIGURATIONS (FROM REFERENCE F1)

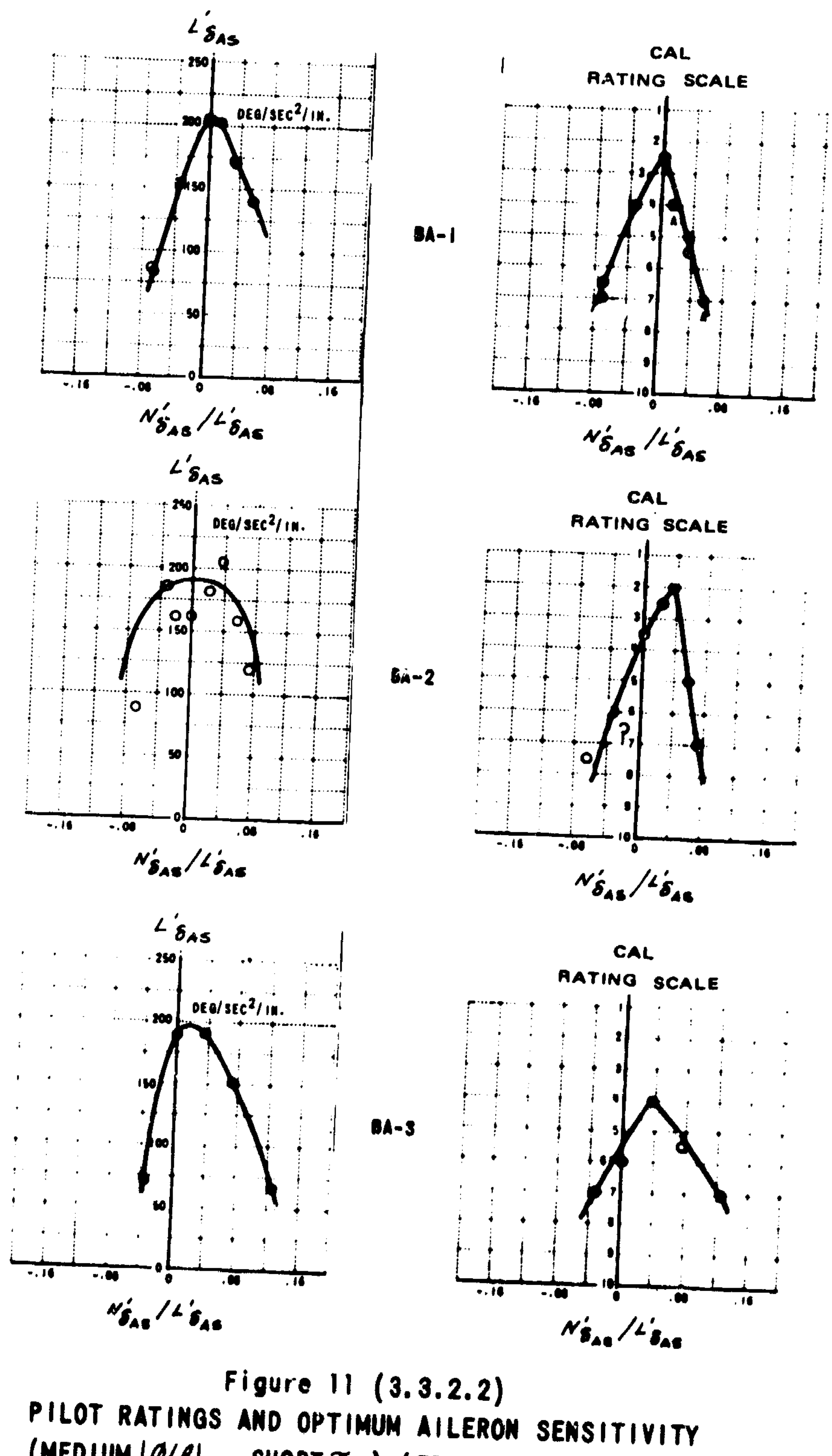


Figure 11 (3.3.2.2)
PILOT RATINGS AND OPTIMUM AILERON SENSITIVITY
(MEDIUM $|\phi/\beta|_d$, SHORT τ_R) (FROM REFERENCE F1)

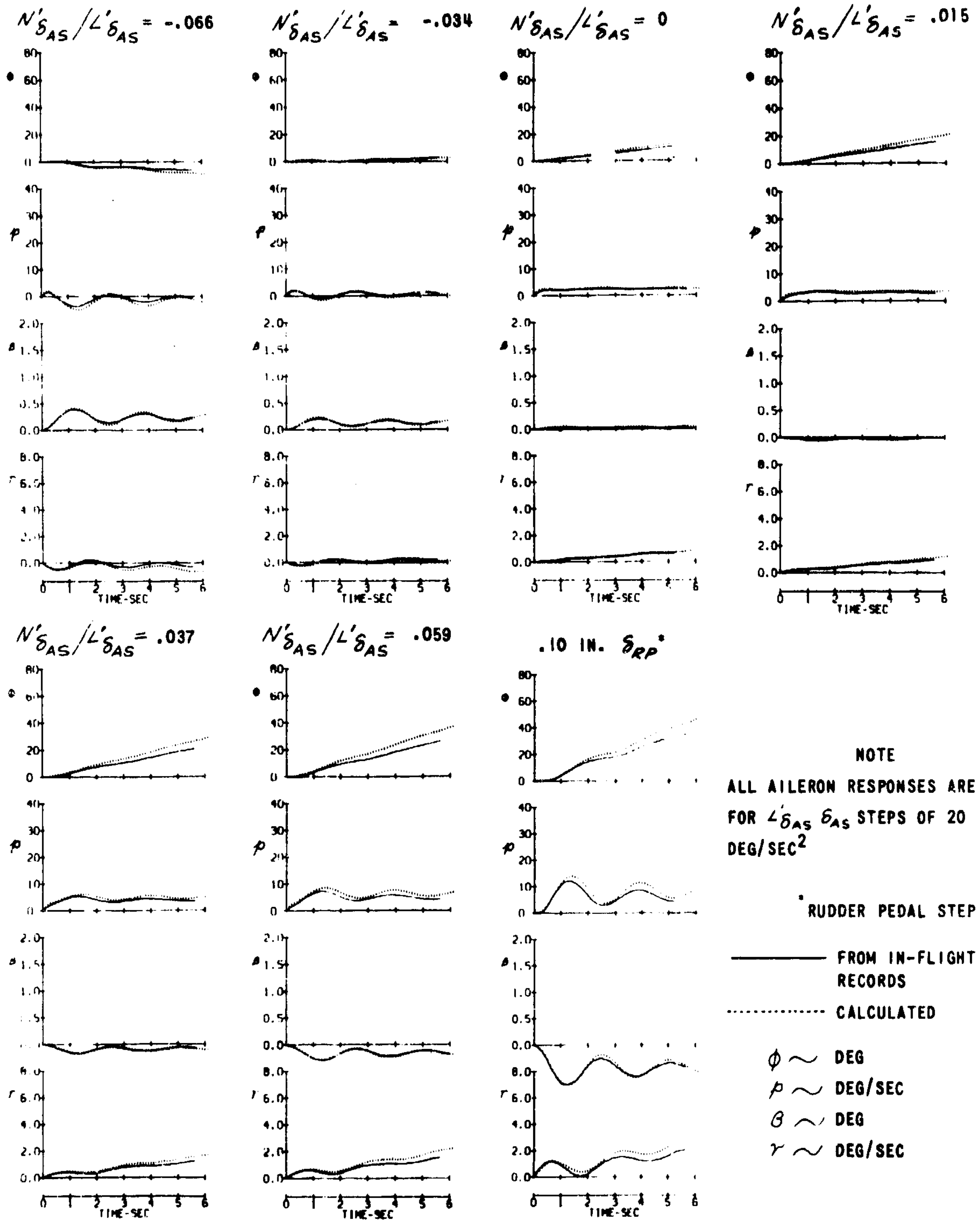


Figure 12 (3.3.2.2)

TRANSIENT RESPONSES FOR BA-1 CONFIGURATIONS (FROM REFERENCE F1)

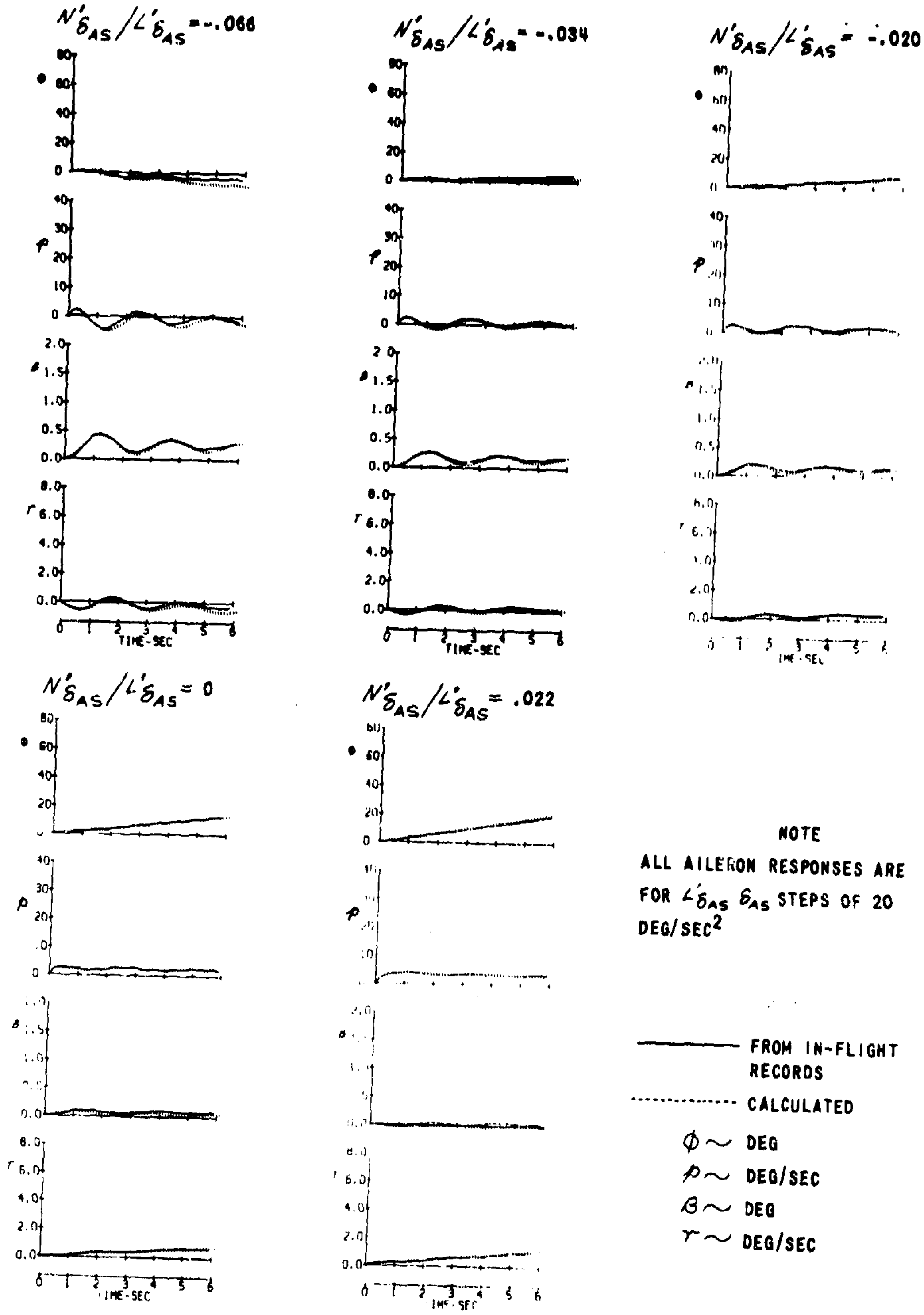


Figure 13 (3.3.2.2)
TRANSIENT RESPONSES FOR BA-2 CONFIGURATIONS (FROM REFERENCE F1)

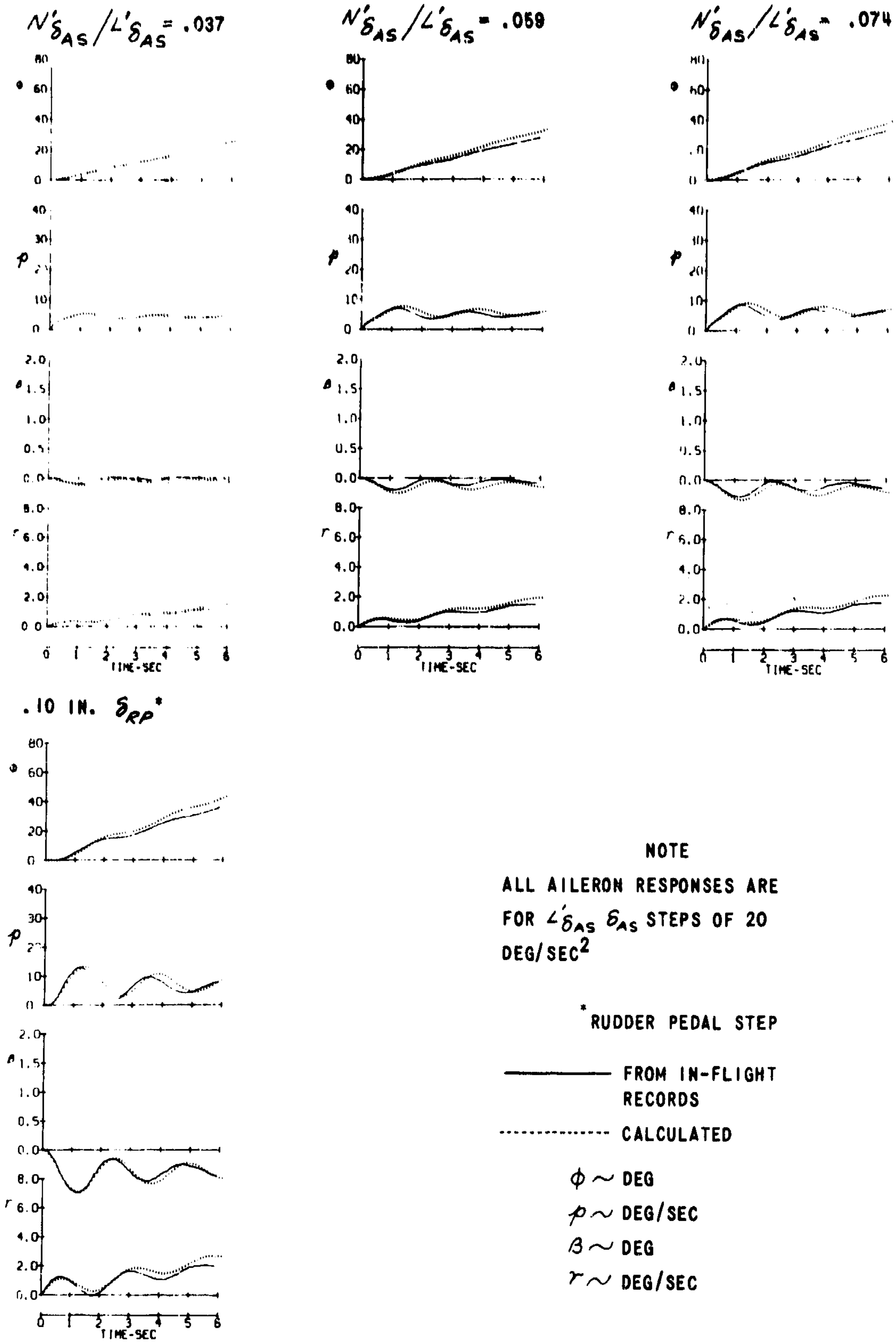


Figure 14 (3.3.2.2)
TRANSIENT RESPONSES FOR BA-2 CONFIGURATIONS (CONT.)
(FROM REFERENCE F1)

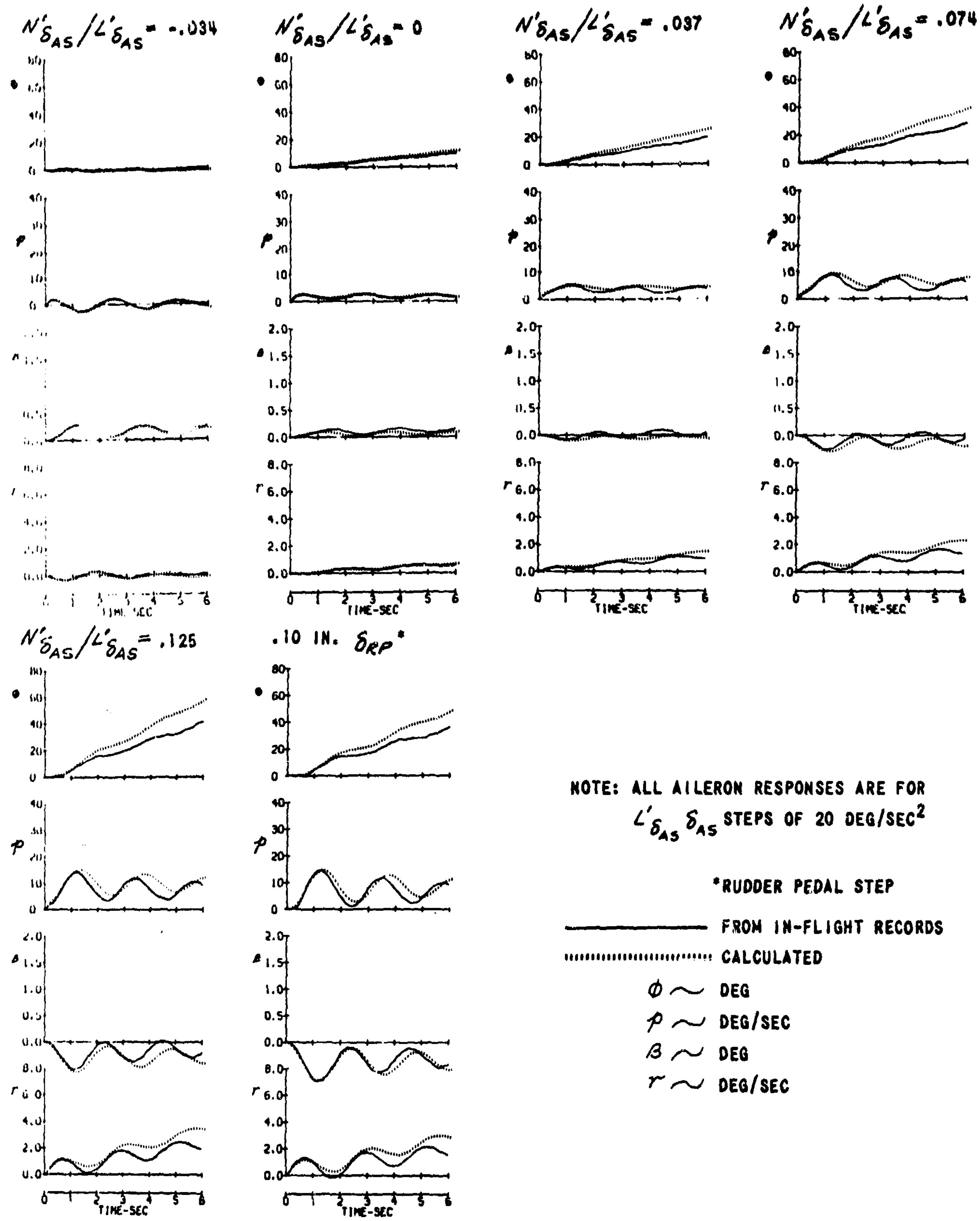


Figure 15 (3.3.2.2)

TRANSIENT RESPONSES FOR BA-3 CONFIGURATIONS (FROM REFERENCE F1)

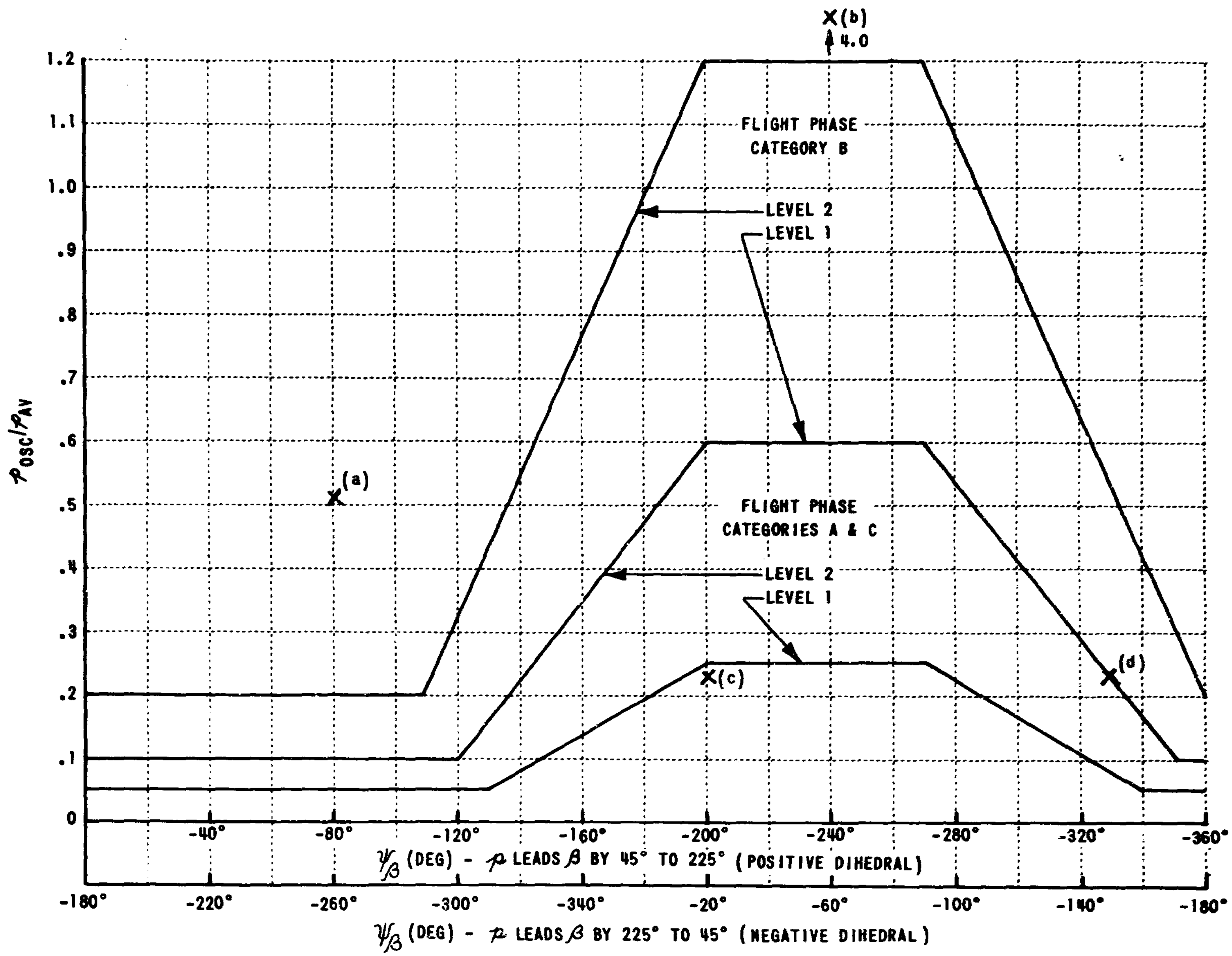


Figure 16 (3.3.2.2)

POSITIVE AND NEGATIVE DIHEDRAL DATA OF REFERENCE H8

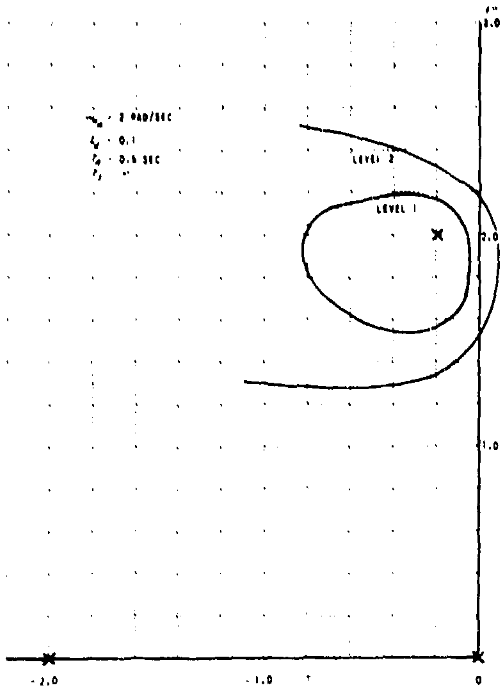


Figure 17 (3. 3.2.2)
AREAS OF p/δ_{AS} TRANSFER FUNCTION
ZERO LOCATIONS FOR FLIGHT PHASE
CATEGORIES A AND C

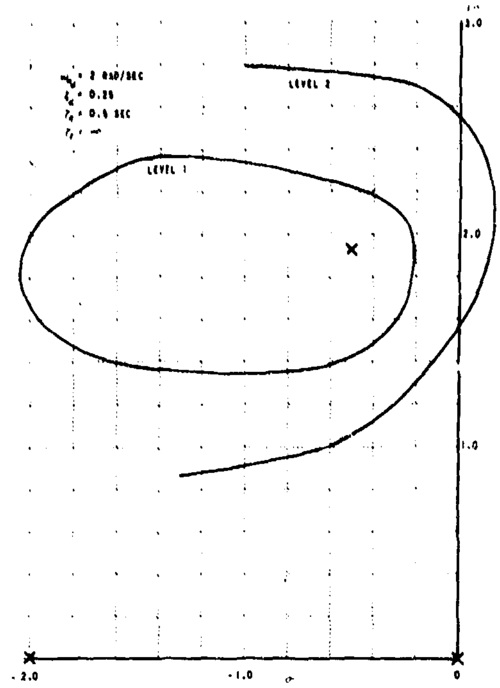


Figure 18 (3.3.2.2)
AREAS OF p/δ_{AS} TRANSFER FUNCTION
ZERO LOCATIONS FOR FLIGHT PHASE
CATEGORIES A AND C

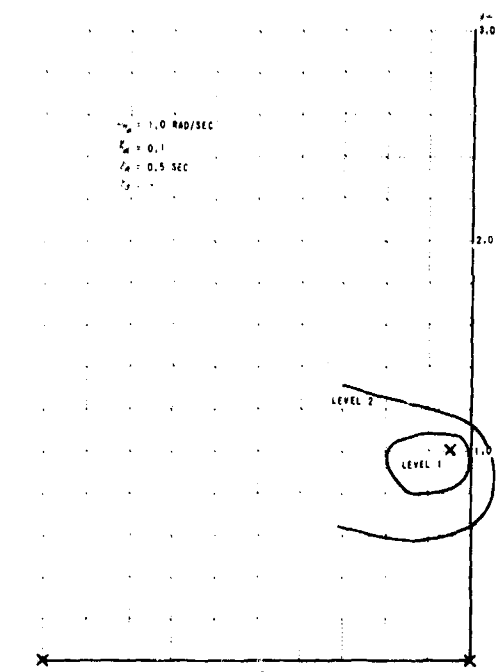


Figure 19 (3.3.2.2)
AREAS OF p/δ_{AS} TRANSFER FUNCTION
ZERO LOCATIONS FOR FLIGHT PHASE
CATEGORIES A AND C

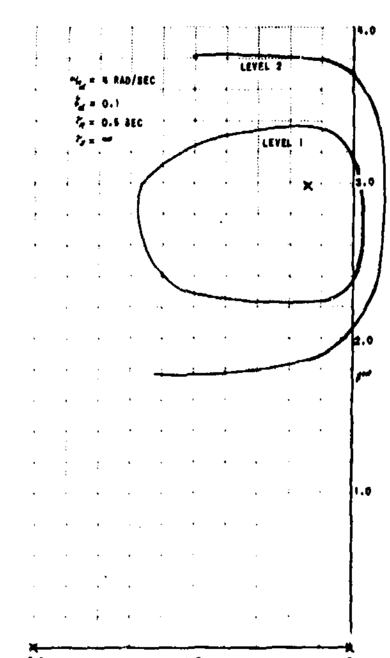


Figure 20 (3.3.2.2)
AREAS OF p/δ_{AS} TRANSFER FUNCTION
ZERO LOCATIONS FOR FLIGHT PHASE
CATEGORIES A AND C

NOMINAL BASE CONDITION	
DUTCH ROLL MODE	$\omega_d = 4.3$ RAD/SEC $\zeta_d = 0.1$
ROLL MODE	$\tau_R = 0.37$ SEC $K_{\dot{\phi}_{\delta_{AS}}} = 0.36$ RAD/SEC/IN
SPIRAL MODE	$\frac{1}{\tau_s} = 0$ SEC^{-1}

SYMBOL	$\lvert\phi/\beta\rvert$
—○—	1.07-1.20
- -△- -	2.42-3.19
—◇—	3.93-4.53
·····▷·····	7.06-7.40

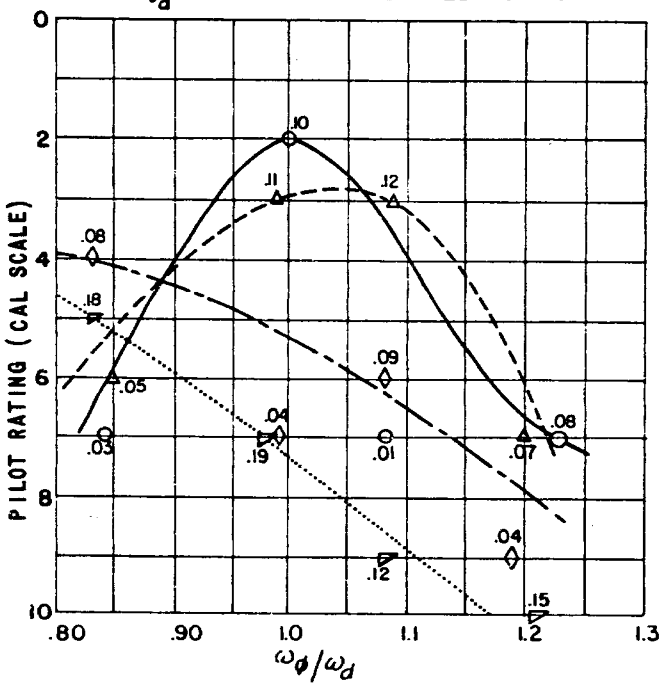


Figure 21 (3.3.2.2)
PILOT RATING VERSUS ω_ϕ/ω_d FOR HIGH FREQUENCY, LOW DAMPING DUTCH ROLL (FROM REFERENCE F22)

NOMINAL BASE CONDITION	
DUTCH ROLL MODE	$\omega_d = 4.3$ RAD/SEC $\zeta_d = .4$
ROLL MODE	$\tau_R = 0.37$ SEC $K_{\dot{\phi}_{\delta_{AS}}} = 0.36$ RAD/SEC/IN
SPIRAL MODE	$1/\tau_s = 0$ SEC^{-1}

SYMBOL	$\lvert\phi/\beta\rvert$
—○—	1.05 - 1.08
·····□·····	2.34 - 2.49
- -◇- -	3.42 - 3.61
-·-◇-·-	5.12 - 5.31

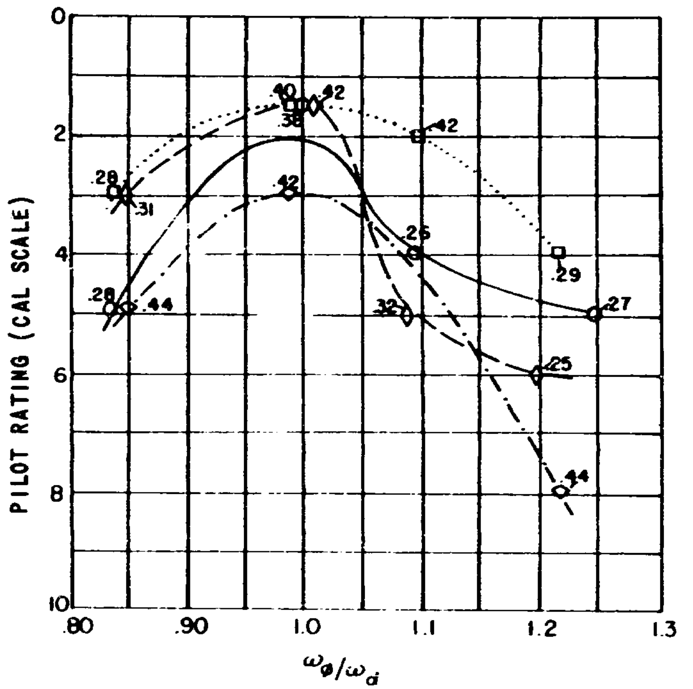


Figure 22 (3.3.2.2)
PILOT RATING VERSUS ω_ϕ/ω_d FOR HIGH FREQUENCY, MODERATE DAMPING DUTCH ROLL (FROM REFERENCE F22)

NOMINAL BASE CONDITION	
DUTCH ROLL MODE	ω_d = 2.3 RAD/SEC ζ_d = .12
ROLL MODE	τ_R = 0.37 SEC $K_{\dot{\phi}_{\delta_{AS}}}$ = 0.36 RAD/SEC/IN
SPIRAL MODE	$1/\tau_s$ = 0 SEC^{-1}

SYMBOL	$\lvert \phi/\beta \rvert$
—○—	.65 – .66
⌓	1.00
– –△– –	2.92 – 3.54
·····○·····	5.46 – 6.97
–·–x–·–	8.05 – 9.34

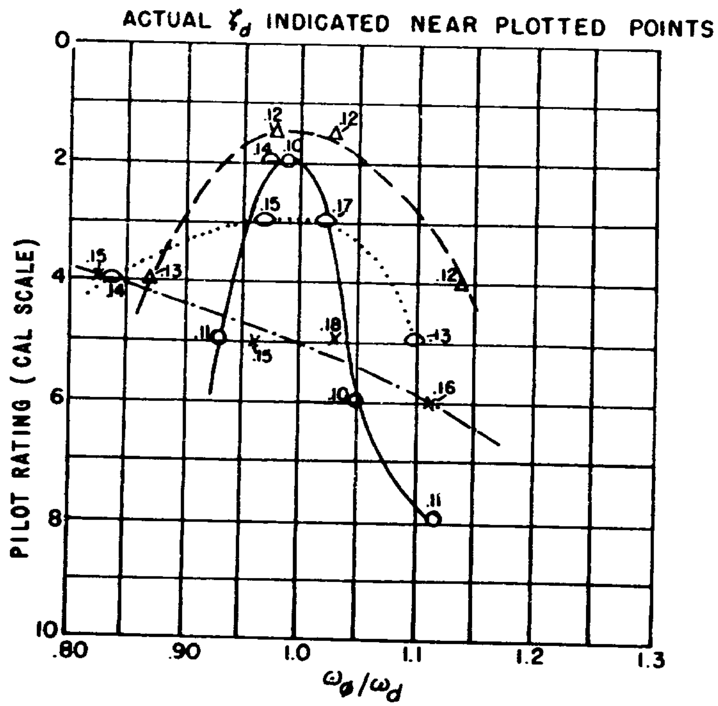


Figure 23 (3.3.2.2)
PILOT RATING VERSUS ω_ϕ/ω_d FOR MODERATE FREQUENCY, LOW DAMPING DUTCH ROLL (FROM REFERENCE F22)

NOMINAL BASE CONDITION	
DUTCH ROLL MODE	ω_d = 2.3 RAD/SEC ζ_d = 0.4
ROLL MODE	τ_R = 0.37 SEC $K_{\dot{\phi}_{\delta_{AS}}}$ = 0.36 RAD/SEC/IN.
SPIRAL MODE	$\frac{1}{\tau_s}$ = 0 SEC^{-1}

SYMBOL	$\lvert \phi/\beta \rvert$
—○—	.57 – .67
– –⌓– –	1.37 – 1.80
—△—	2.49 – 2.99
·····○·····	3.93 – 5.53
X	6.76 – 7.46

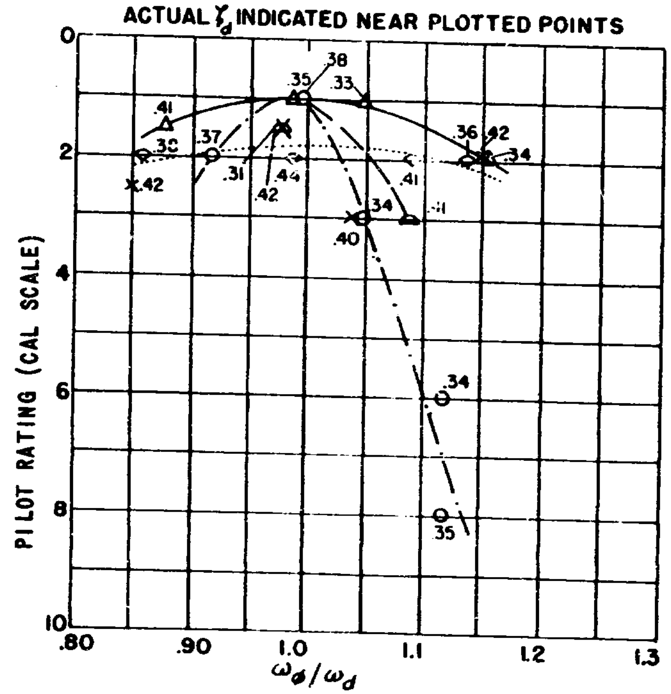


Figure 24 (3.3.2.2)
PILOT RATING VERSUS ω_ϕ/ω_d FOR MODERATE FREQUENCY, MODERATE DAMPING DUTCH ROLL (FROM REFERENCE F22)

NOMINAL BASE CONDITION	
DUTCH ROLL MODE	ω_d = 1.1 RAD/SEC ζ_d = 0.2
ROLL MODE	τ_R = 0.37 SEC $K_{\dot{\phi}_{\delta_{AS}}}$ = 0.36 RAD/SEC/IN
SPIRAL MODE	$\frac{1}{\tau_s}$ = 0 SEC^{-1}

SYMBOL	$\lvert\phi/\beta\rvert$
○	.71
—◒—	1.17-1.40
---□---	1.75-2.55
······◇······	2.90-4.56

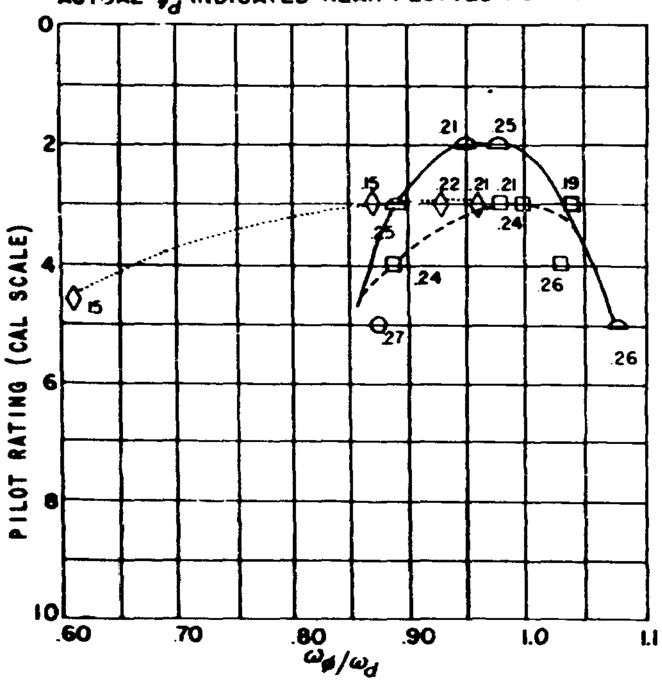


Figure 25 (3.3.2.2)

PILOT RATING VERSUS ω_ϕ/ω_d FOR LOW FREQUENCY, MODERATELY LOW DAMPING DUTCH ROLL (FROM REFERENCE F22)

NOMINAL BASE CONDITION	
DUTCH ROLL MODE	ω_d = 1.1 RAD/SEC ζ_d = .55
ROLL MODE	τ_R = 0.37 SEC $K_{\dot{\phi}_{\delta_{AS}}}$ = 0.36 RAD/SEC/IN
SPIRAL MODE	$\frac{1}{\tau_s}$ = 0 SEC^{-1}

SYMBOL	$\lvert\phi/\beta\rvert$
—○—	.64 - .80
◒	1.04-1.20
△	1.91-2.38
◇	4.07

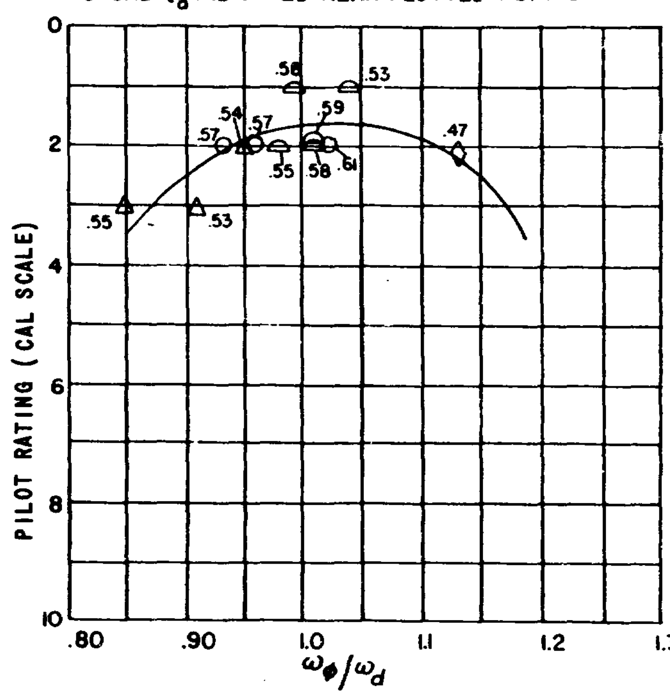


Figure 26 (3.3.2.2.)

PILOT RATING VERSUS ω_ϕ/ω_d FOR LOW FREQUENCY, HIGH DAMPING DUTCH ROLL (FROM REFERENCE F22)

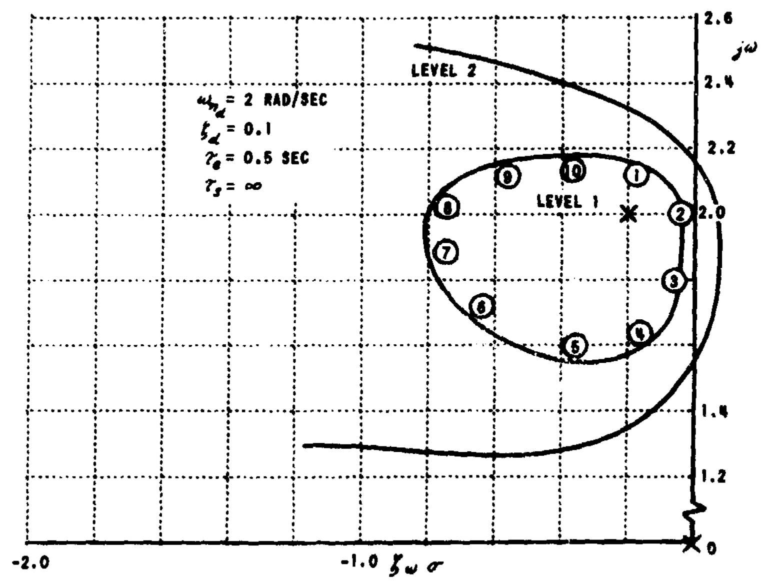


Figure 27 (3.3.2.2)
AREAS OF p/δ_{AS} TRANSFER FUNCTION ZERO LOCATIONS FOR LEVEL 1, FLIGHT PHASE CATEGORIES A AND C, SHOWING CONDITIONS FOR TIME HISTORIES OF FIGURE 28

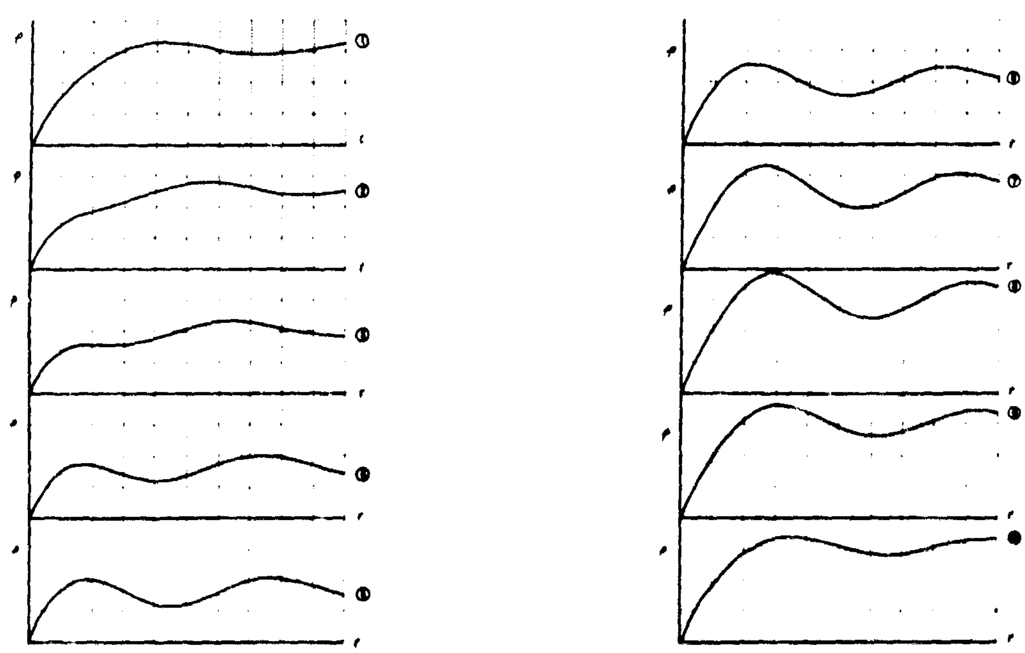


Figure 28 (3.3.2.2)
ROLL RATE TIME HISTORIES FOR ZERO LOCATIONS OF ISO-PILOT RATINGS IN FIGURE 27

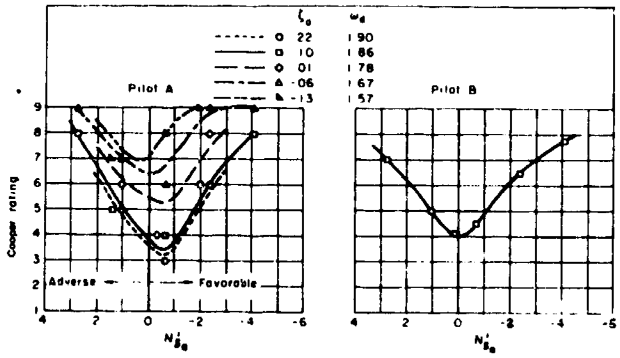


Figure 29 (3.3.2.2)

PILOT OPINION DATA OBTAINED IN THE FLIGHT TEST (CENTER STICK CONTROL): $|\phi|/|v_e| \approx 0.59$. (FROM REFERENCE F21)

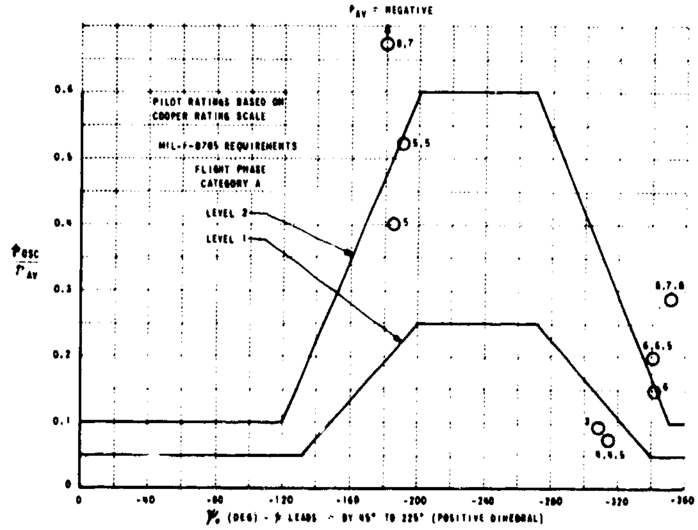


Figure 30 (3.3.2.2)

DATA FROM REFERENCE F21

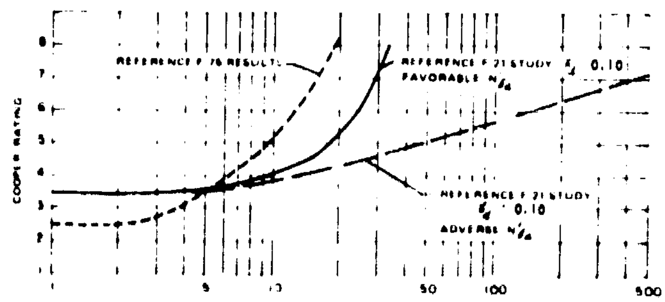


Figure 31 (3.3.2.2)

COMPARISON OF DATA OF THE REF. F21 STUDY WITH RESULTS OF REF. F75 (CENTER STICK CONTROL). (FROM REFERENCE F21)

3.3.2.3 BANK ANGLE OSCILLATIONS

REQUIREMENT

3.3.2.3 Bank angle oscillations. The value of the parameter ϕ_{OSC} / ϕ_{AV} following a rudder-pedals-free impulse aileron control command shall be within the limits in figure 5 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 aileron control system.

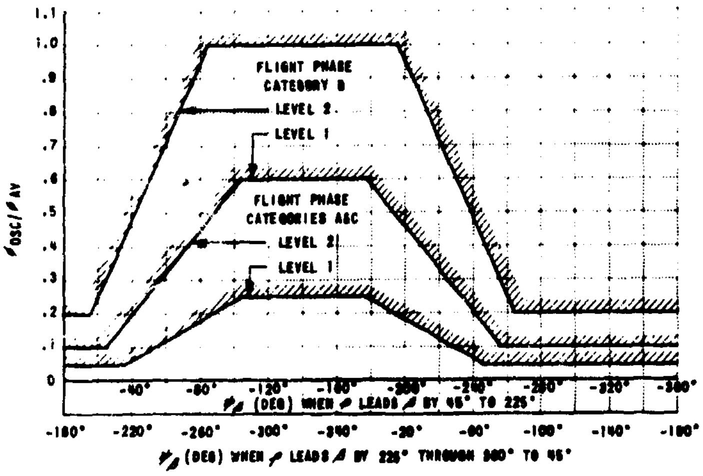


Figure 5 BANK ANGLE OSCILLATION LIMITATIONS

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

The requirements of 3.3.2.3 are essentially the same as those of 3.3.2.2.1 since, for linear motions, the bank angle time history following a pulse is the same as the roll rate time history following a step. The advantage of using a pulse, rather than a step, is that much larger airborne inputs may be used since large bank angles do not result, and also ψ_β may be easier to measure since the β trace will not tend to ramp.

Since ψ_β for a pulse is $(90 + \sin^{-1} \zeta_d)$ degrees more positive than ψ_β for a step, the curves of Figure 2 (3.3.2.2.1) have been shifted to the left 90 degrees plus an angle corresponding to $\zeta_d \approx 0.1$. The resulting curves of Figure 5 (3.3.2.3) should be within 5 degrees of the exact location for all damping ratios for which this requirement might be critical.

This paragraph, then, like 3.3.2.2 and 3.3.2.4, is an attempt to extend the roll-sideslip coupling requirements to larger control deflections. It may also help account for some flight control system nonlinearities such as stick-position-dependent roll damping augmentation.

3.3.2.4 SIDESLIP EXCURSIONS

3.3.2.4.1 ADDITIONAL SIDESLIP REQUIREMENT FOR SMALL INPUTS

REQUIREMENT

3.3.2.4 Sideslip excursions. Following a rudder-pedals-free step aileron control command, the ratio of the sideslip increment, $\Delta\beta$, to the parameter k (6.2.6) shall be less than the values specified herein. The aileron command shall be held fixed until the bank angle has changed at least 90 degrees.

Level	Flight Phase Catagory	Adverse Sideslip (Right roll command causes right sideslip)	Proverse Sideslip (Right roll command causes left sideslip)
1	A	6 degrees	2 degrees
	B & C	10 degrees	3 degrees
2	All	15 degrees	4 degrees

3.3.2.4.1 Additional sideslip requirement for small inputs. The amount of sideslip following a rudder-pedals-free step aileron control command shall be within the limits shown on figure 6 for Levels 1 and 2. This requirement shall apply for step aileron control commands up to the magnitude which causes a 60-degree bank angle change within T_d or 2 seconds, whichever is longer.

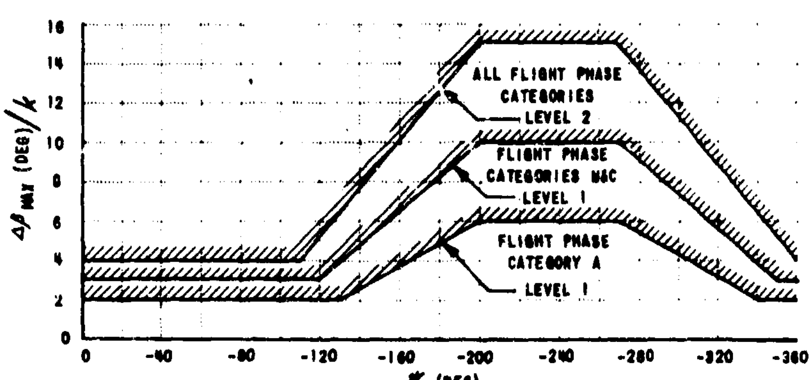


Figure 6 SIDESLIP EXCURSION LIMITATIONS

RELATED MIL-F-8785 PARAGRAPHS

3.4.9

DISCUSSION

Basis for Sideslip Requirements

This requirement, which is directed at precision of control, replaces Paragraph 3.4.9 in MIL-F-8785. The requirement in MIL-F-8785 states that it must be possible to roll from 45° of bank in one direction through 45° in the other direction - a bank angle change of 90° - without the sideslip exceeding 15 degrees. Rudder pedals are fixed and aileron control inputs are sufficient to meet the roll performance requirements. It also states "in similar rolls with partial aileron deflections, the angle of sideslip shall be proportional to the aileron cockpit control deflection." This test is to be performed in the cruise configuration at 1.4 $V_{S_{CR}}$ and in the power-approach configuration at 1.4 $V_{S_{PA}}$.

From the meetings with aircraft industry representatives, it became apparent that there was widespread dissatisfaction with this requirement. For example, one representative stated that "Magnitude of adverse yaw is not adequate to ensure good handling qualities. Task again is an important consideration. Phasing of sideslip angle during coordinated maneuvers should be investigated further." Another representative stated that "the requirement is not right - ... either 15° of sideslip is too much or else β is the wrong parameter." On the other hand, one manufacturer said that "10° of sideslip does not mean anything for large aircraft."

The primary source of data from which the sideslip requirement evolved is the low $|\phi/\beta|_d$ ($\approx$ 1.5) configurations of Reference F1 (Figures 1 through 4). The pilot comments associated with these configurations indicated that the pilots' difficulties were almost exclusively associated with sideslip, rather than with bank angle tracking as was the case for larger $|\phi/\beta|_d$ ($\approx$ 6) configurations.

Analysis of the data revealed that the amount of sideslip that a pilot will accept or tolerate is a strong function of the phase angle of the Dutch roll component of sideslip. When the phase angle is such that β is primarily adverse, the pilot can tolerate quite a bit of sideslip. On the other hand, when the phasing is such that β is primarily proverse, the pilot can only tolerate a small amount of sideslip because of difficulty of coordination.

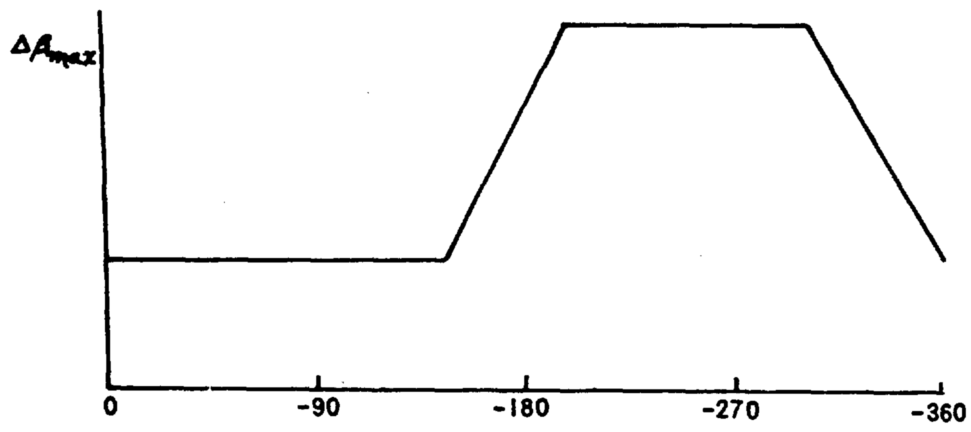
There is more to coordination, however, than whether the sideslip is adverse or proverse; the source and phasing of the disturbing yawing moment also significantly affect the coordination problem. If the yawing moment is caused by aileron and is in the adverse sense, then in order to coordinate the pilot must phase either right rudder with right aileron or left rudder with left aileron. Since pilots find this technique natural they can generally coordinate well even if the yawing moment is large. If on the other hand, the yawing moment is in the proverse sense or is caused by roll rate, coordination is far more difficult. For proverse yaw-due-to-aileron the pilot must cross control; and for either adverse or proverse yaw-due-to roll rate, required rudder inputs must be proportional to roll rate. Pilots find these techniques unnatural and difficult to perform. Since yawing moments may also be introduced by yaw rate, it can be seen that depending on the magnitude and sense of the various yawing moments, coordination may either be easy or extremely difficult. If coordination is sufficiently difficult that pilots cannot be expected to coordinate routinely, the flying qualities requirements must restrict rudder-pedals free unwanted motions to a size acceptable to pilots.

Analysis further revealed that it was not so much the absolute magnitude of the sideslip that bothered the pilot, but rather the maximum change occurring in sideslip. The latter was a better measure of the amount of coordination required. Thus, the data from this program were plotted in Figure 5 as the maximum change in sideslip occurring during a rudder-pedals-fixed rolling maneuver, $\Delta\beta_{max}$, versus the phase angle of the Dutch roll component of sideslip, ψ_β.

The phase angle, ψ_β, is a measure of the sense of the initial sideslip response, whether adverse or proverse, while $\Delta\beta_{max}$ is a measure of the amplitude of the sideslip generated. Both the sense and the amplitude affect the coordination problem.

It was observed from examination of the low $|\phi/\beta|_d$ data plotted on Figure 5 that the break points in curves of iso-pilot rating occurred at almost exactly the same values of ψ_β as for the moderate $|\phi/\beta|_d$ configurations (see the discussion of Paragraph 3.3.2.2), even though the degradation of flying qualities was due to sideslip problems with the low $|\phi/\beta|_d$ configurations and to bank angle problems with the moderate $|\phi/\beta|_d$ configurations. Since the break points were so close, and since the figures describe different manifestations of the same phenomena, the break points were made identical for both the low $|\phi/\beta|_d$ configurations ($\Delta\beta_{max}$ vs. ψ_β) and moderate $|\phi/\beta|_d$ configurations (p_{osc}/p_{AV} vs. ψ_β).

The sideslip excursion criteria were thus presented in the form shown in Sketch 1.



Sketch 1

As with the p_{osc}/p_{AV} requirement, it can be seen from Sketch 1 that the specified value of $\Delta\beta_{max}$ varies significantly with ψ_β . This difference is almost totally due to the differences in ability to coordinate during turn entries and exits. Since ψ_β is a direct indicator of the difficulty a pilot will experience in coordinating a turn entry, variation of $\Delta\beta_{max}$ with ψ_β is to be expected. For $-180° \gtrsim \psi_\beta \gtrsim -270°$, normal coordination may be effected, that is, right rudder pedal for right rolls. Thus, even if large sideslip excursions occur in rudder-pedal-free rolls (the conditions under which the $\Delta\beta_{max}$ tests are conducted), when coordinating in the normal manner sideslip oscillations can be readily minimized. As ψ_β varies from -270° to -360° coordination becomes increasingly difficult, and in the range $-360° \lesssim \psi_\beta \lesssim 90°$ cross controlling is required to effect coordination. Since pilots do not normally cross control and, if they must, have great difficulty in doing so, for $-360° \lesssim \psi_\beta \lesssim -90°$, oscillations in sideslip either go unchecked or are amplified by the pilot's efforts to coordinate with rudder pedals.

Although the Reference F1 data were orginally worked up in the form of $\Delta\beta_{max}/\phi_1$, a form similar to that used by References F72, F73, and F74, it was decided to retain the concept used in Paragraph 3.4.9 of MIL-F-8785 of relating the amount of allowable sideslips to the roll performance requirements. Through this tie to roll performance requirements, the effect of Class and some of the effects of Flight Phase and Level are taken into consideration. In the Reference A1 requirements, this concept is retained through the parameter "k".

In MIL-F-8785, the amount of allowable sideslip is scaled down by the ratio of the aileron deflection used in a given test to the aileron deflection needed to meet the roll performance requirements. This is a good parameter to use if the roll performance requirements can be met and if roll performance varies linearly with aileron deflection. To circumvent the problems that arise if either or both of these conditions do not hold, the parameter "k" has been related to the ratio of roll performance rather than to the ratio of control deflection. Further, to avoid interdependence of requirements, commanded roll performance in a test is ratioed to the roll performance requirements of Table IX so that the sideslip requirement can be applied even if the aircraft does not meet the roll performance requirements.

In certain cases, Reference A1 allows use of rudder pedals to meet the roll performance requirement. It is rational to determine "commanded roll performance" and k with the same rudder pedal use. But note that the rudder pedals cannot be used in testing for sideslip excursions. For example, take Class II, Category A, Level 1. From Table IX, $(\phi_t)_{requirement} = 45°$ in 1.4 sec. With 1/4 aileron and just enough rudder pedal to counteract adverse yaw, $(\phi_t)_{command} = 15°$ in 1.4 sec (from flight-test results), so $k = 15/45\ 0.333$. That value of k is used in determining compliance with 3.3.2.4.1 during rudder-pedals-free rolls. Other examples are given in 6.2.6.

To transform the Reference F1 data into terms of $\Delta\beta_{max}$ as presented in Figure 5 rather than $\Delta\beta_{max}/\phi_1$, the data were multiplied by the ϕ_1 required for the aircraft type (Class IV) and Flight Phase Category (A), which is $\phi_1 \approx$ 60 degrees in 1 second.

Comparing these curves with the MIL-F-8785 adverse yaw requirement of $\beta \doteq 15°$, it can be seen that for Level 2 operation, the adverse yaw requirement is identical. However, the figure shows that for Level 1 operation or for airplanes exhibiting proverse yaw, the allowable sideslip is considerably less.

In determining the Flight Category C requirements, data were examined from References F72, F73, B96 and B91, and from current aircraft. From these data, it was concluded that for airplanes exhibiting adverse yaw characteristics, satisfactory ratings could be obtained with up to 10° of sideslip, and acceptable ratings could be obtained with up to 15° of sideslip. Based on this, the Level 1 boundary for Flight Phase Category C was scaled up from the Level 1 boundary for Flight Phase Category A as shown in Figure 6 of Reference A1.

To check the validity of the requirements for Flight Phase Category C, as they relate to ϕ/β , the Level 1 and 2 boundaries were compared with the low $|\phi/\beta|_d$ data of References G10 and G11 (Figure 6). The boundaries were transformed into the form $\Delta\beta_{max}$ from $\Delta\beta_{max}/\phi_1$ by use of the ϕ_1 appropriate to Flight Phase Category C, that is, ϕ_1 = 30 degrees in 1 second. It can be seen from Figure 6 that the criteria are consistent with the available data. It should be noted that whereas the data of Figure 5 (and 7) are based on the CAL pilot rating scale, the data of Figure 6 are based on the Cooper scale. Thus Level 2 for Figure 5 (and 7) is at PR 6.5, whereas Level 2 for Figure 6 is at PR $\approx$ 5.

To determine the requirements for Flight Phase Category B, the data of Reference F22 were examined. The data (Figure 7) were transformed from the form $\Delta\beta_{max}/\phi_1$ to $\Delta\beta_{max}$ by use of the ϕ_1 believed to be appropriate to the entry mission. A value of $\phi_1 \approx$ 30 degrees in 1 second was selected. From Figure 7 it can be seen that the data are consistent with the curves selected for Category C Flight Phases.

The preceding discussion presents the rationale and the data which support the requirement of 3.3.2.4.1. As with the roll rate requirements of 3.3.2.2.1, the sideslip requirements of 3.3.2.4.1 are applicable for small inputs only. In order to be able to test for large control inputs, an additional but more lenient requirement (3.3.2.4) similar to that stated in MIL-F-8785 has been specified. In this way the more comprehensive requirement of 3.3.2.4.1 on sideslip limitations can be incorporated without losing the ability to flight test for compliance with large control inputs. Also, as in 3.3.2.2, it might have been desirable to include larger-amplitude rolls. This was not done for the reason discussed under 3.3.2.2.

There is no quantitative Level 3 requirement. Here also the rationale presented in the discussion of 3.3.2.2 applies.

Effect of Lateral Acceleration

Although the sideslip requirements are not a function of speed, they will only be limiting at low speed. At high speed, parameters such as side force and side acceleration will more probably limit the amount of sideslip that can be generated. To gain some insight on this area, the sideslip and side acceleration characteristics of eight current aircraft were examined and are presented in Figure 8. From this figure it can be seen that the amount of sideslip generated, following a step aileron input of sufficient size to meet the roll performance requirements, is such that the maximum lateral acceleration at the center of gravity is generally less than 0.2 g's.

No experimental data were found that pertained directly to this area; however, References F53, F37, M49 and F76 investigated the effect of side accelerations in connection with investigation of Dutch roll dynamics on flying qualities.

Reference F53 reports an investigation of lateral acceleration at the pilot's head (due to lateral, rolling and yawing accelerations of the aircraft) for Dutch roll periods between 1 and 4 seconds. An amplitude of acceleration of ± 0.02g was sometimes perceptible to the pilot. An amplitude of ±0.025g was always perceptible but was considered unsatisfactory for a long flight, while an amplitude of ±0.08g was considered very unsatisfactory for any mission this (fighter) aircraft might perform.

Reference F37 reports the ability of pilots to tolerate oscillatory lateral acceleration to which they were exposed during automatically controlled fighter intercepts. Since these runs were of fairly short duration and the pilot acted primarily as an observer, the results presented in Figure 9 differ somewhat from those of Reference F53.

For Reference M49, the safety pilot induced constant-amplitude rolling oscillations at various frequencies while a subject pilot rated the acceptability of the resulting motions, taking "into account the degree of which the condition would interfere with their normal (fighter) aircrew duties." The results of this program are presented in Figure 10 in terms of lateral acceleration at the pilot's head versus frequency.

Reference F76 is unable to correlate lateral acceleration with pilot rating in a program investigating the effects of various Dutch roll dynamic characteristics on flying qualities, in which ω_{n_d}, ζ_d, $|\phi/\beta|_d$ and $\angle\phi/\beta$ were varied over wide ranges. Reference F76 concludes:

> "The difficulty with the acceleration calculations is believed to be that the acceleration calculated above is not what the pilot feels. He is not a rigid body, and he is not rigidly attached to the airframe. The nature of his anatomy and of his attachment to the airplane are such that he receives some feel through his feet, hands, and back, but primarily through his

ischial tuberosities (his two seat bones), which are in effect attached to the airframe through relatively heavy vertical springs, and through relatively light transverse springs. He thus gets most of his feel through vertical forces, and couples, on his ischia. If the restraints were idealized to zero lateral restraint, he still would feel the moment about his own body axis, as the reacting couple of his ischia, independent of height. The problem is further complicated inasmuch as the pilot's reaction to the Dutch roll oscillation must be by sight as well as by feel."

Reference F76 also points out that phasing of the Dutch roll (phase angle between p and β) greatly affects the magnitude of lateral acceleration felt by the pilot, changing it by a factor of 4 or 5 in addition to affecting how "natural" or "unnatural" the motion feels.

The data of Reference E80 (Figures 11-14) were also examined for the F-86 and F-84 airplanes performing typical fighter maneuvers. By comparing Figures 11 and 12 with Figures 13 and 14, it can be seen that although the amount of sideslip generated fell off rapidly as speed increased, the amount of side acceleration generated was relatively constant, with peak values of approximately 0.3 g's.

From these observations, it can be seen that even if lateral acceleration is an important independent flying qualities parameter, not enough research has been performed to formulate a requirement on lateral acceleration at this time. Further research is being performed, however, to try and assess the influence of lateral acceleration on allowable roll-sideslip coupling and acceptable Dutch roll characteristics.

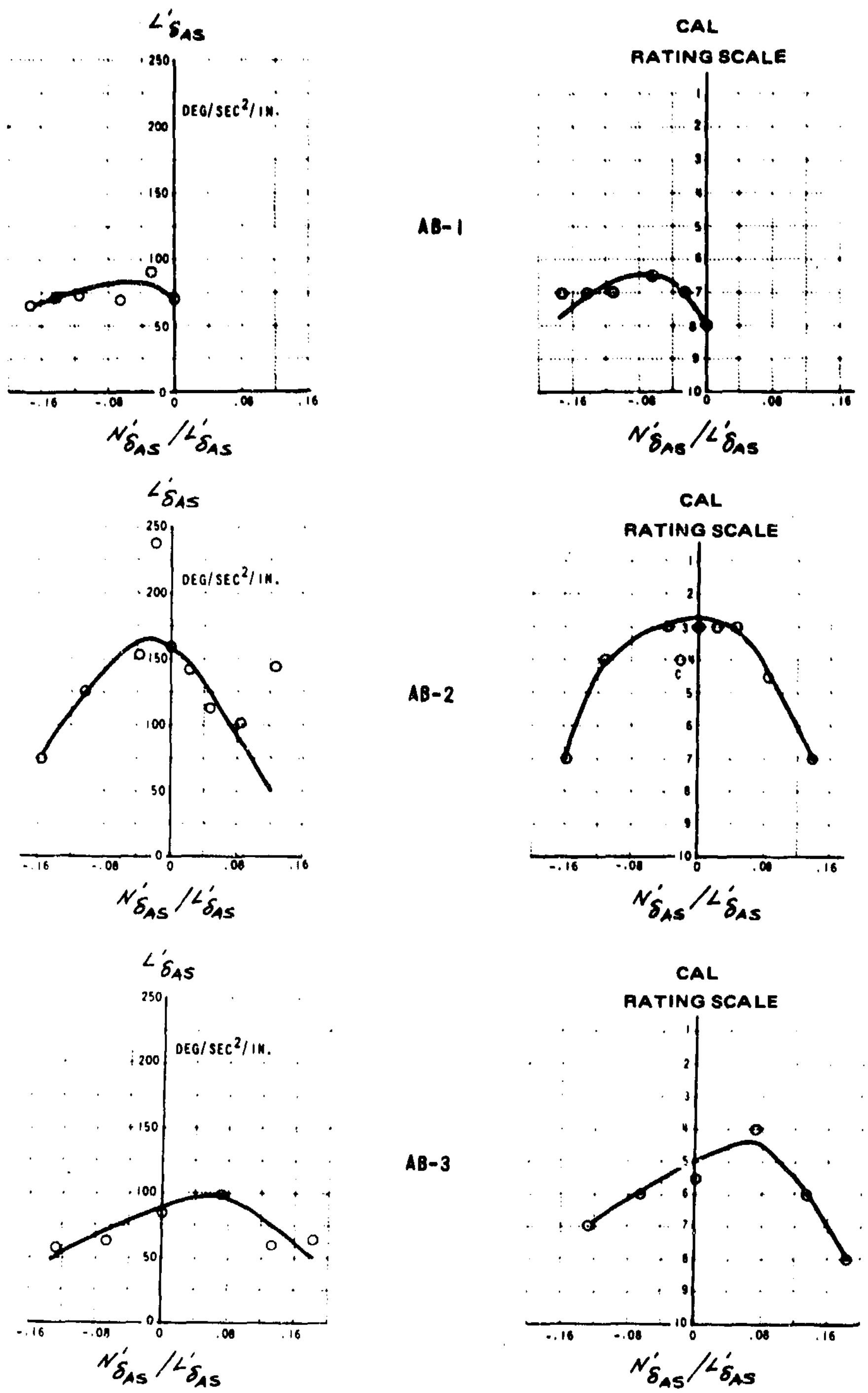


Figure 1 (3.3.2.4)
PILOT RATINGS AND OPTIMUM AILERON SENSITIVITY
(LOW $|\phi/\beta|_d$, MEDIUM τ_R) (FROM REFERENCE F1)

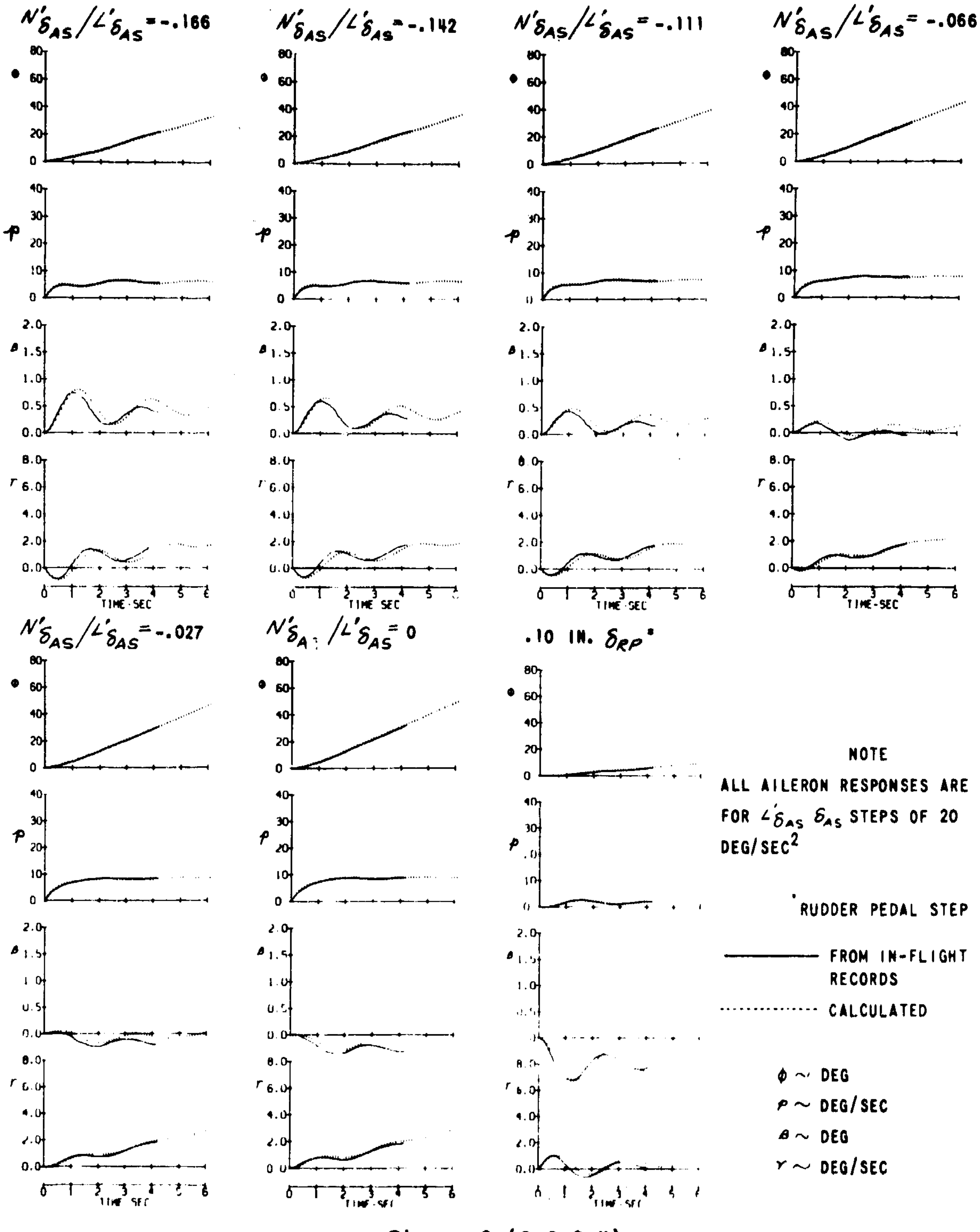


Figure 2 (3.3.2.4)

TRANSIENT RESPONSES FOR AB-1 CONFIGURATIONS (FROM REFERENCE F1)

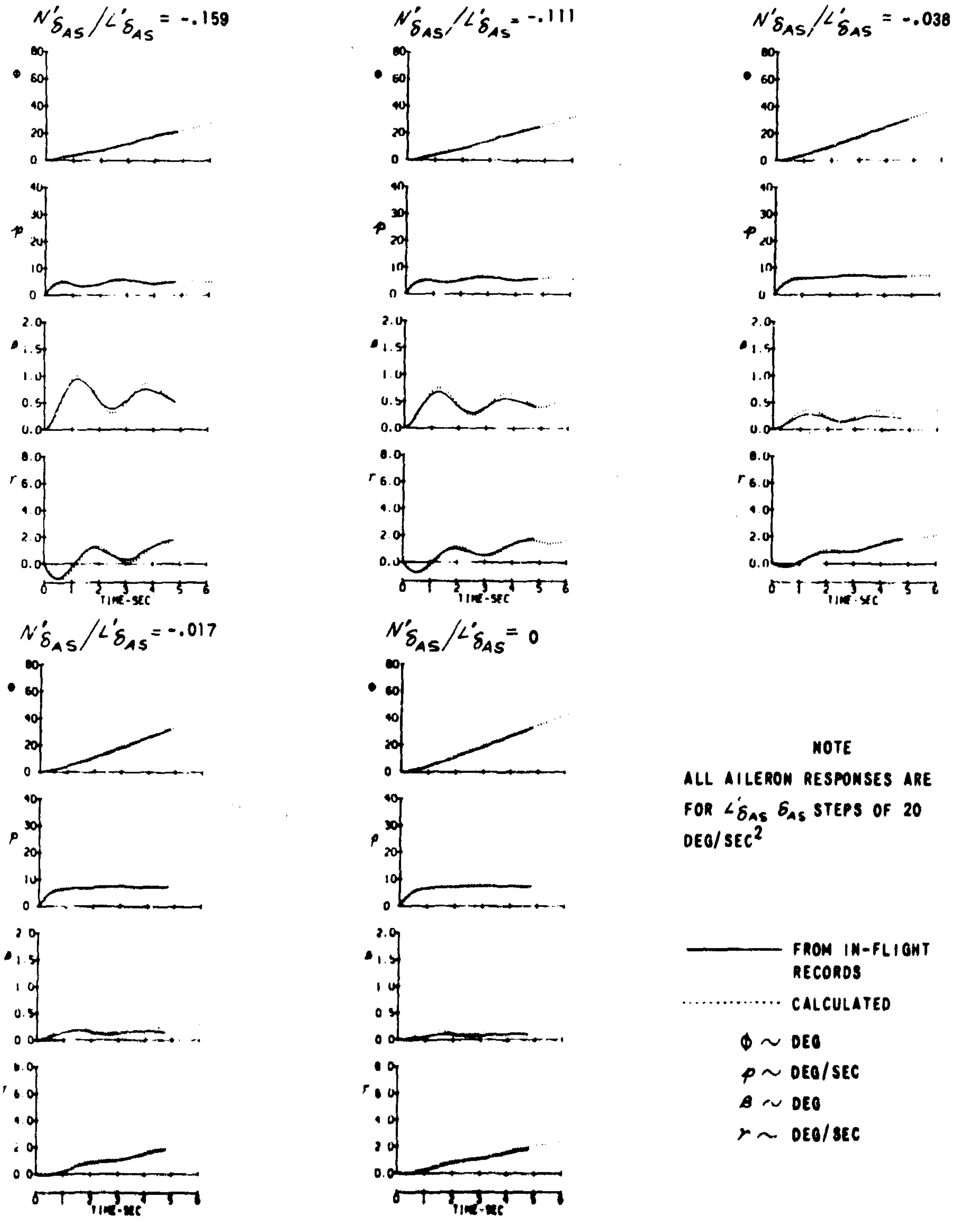


Figure 3 (3.3.2.4)
TRANSIENT RESPONSES FOR AB-2 CONFIGURATIONS (FROM REFERENCE F1)

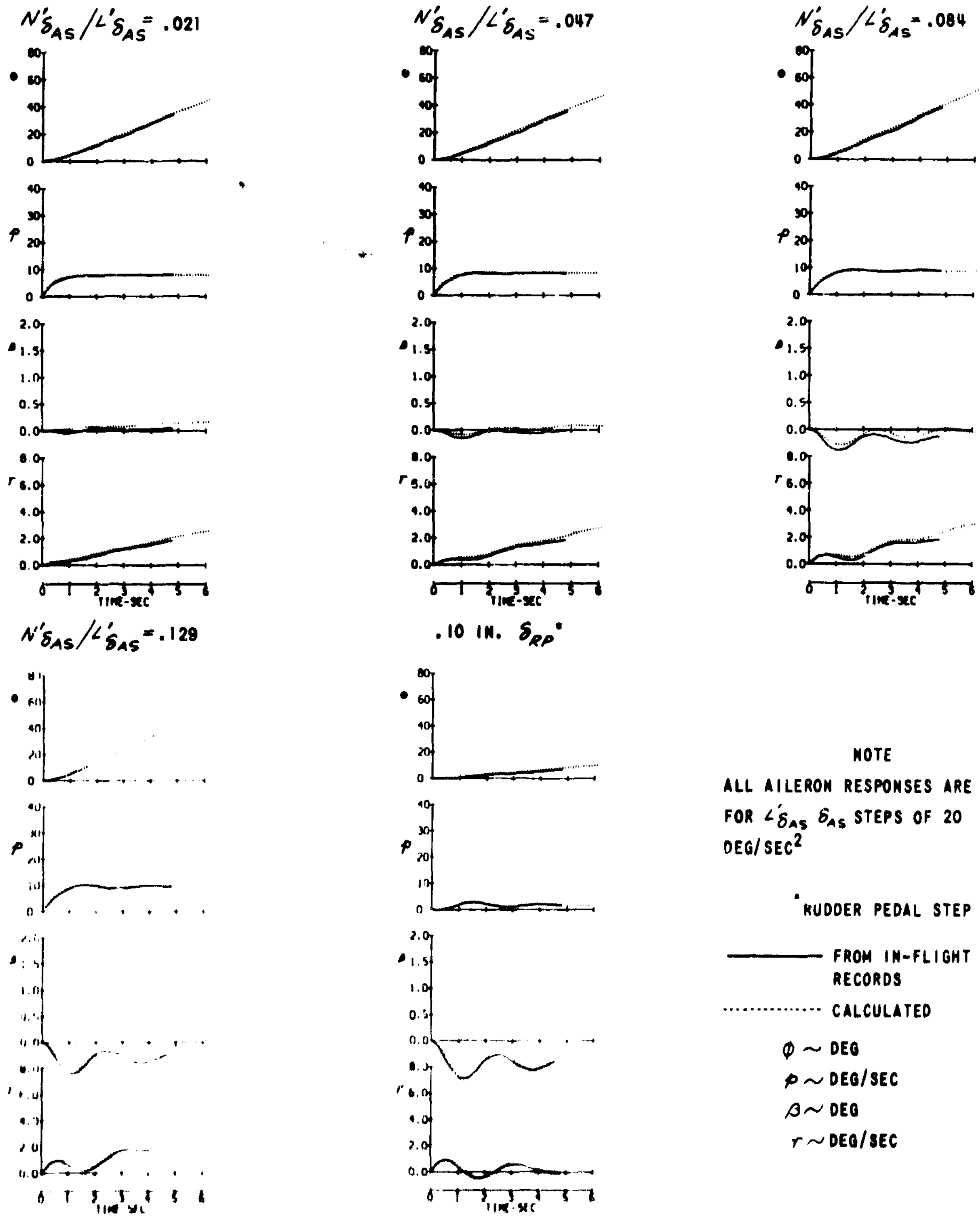


Figure 3 (3.3.2.4)
TRANSIENT RESPONSES FOR AB-2 CONFIGURATIONS (CONT)

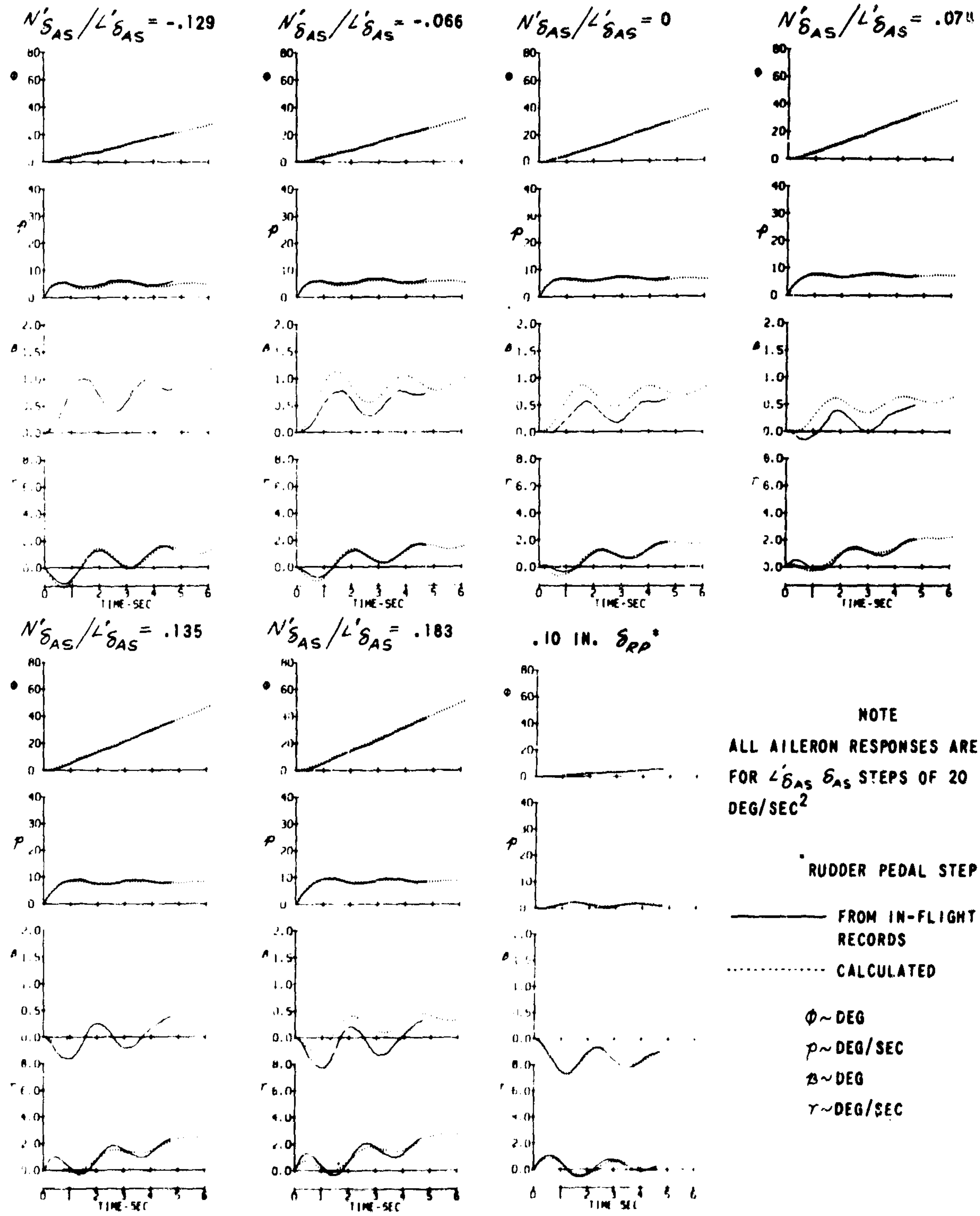


Figure 4 (3.3.2.4)
TRANSIENT RESPONSES FOR AB-3 CONFIGURATIONS (FROM REFERENCE F1)

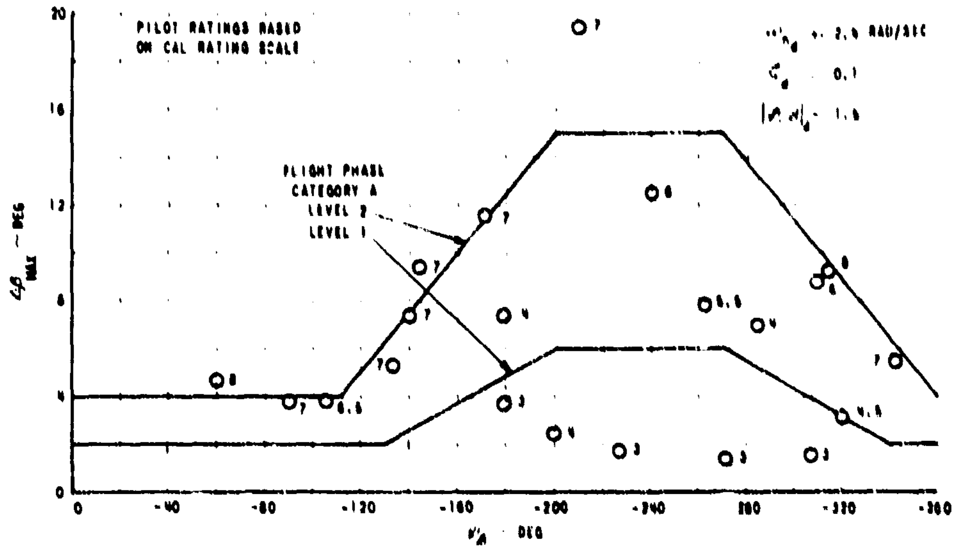


Figure 5 (3.3.2.4)
FLIGHT PHASE CATEGORY A DATA FROM REFERENCE F1

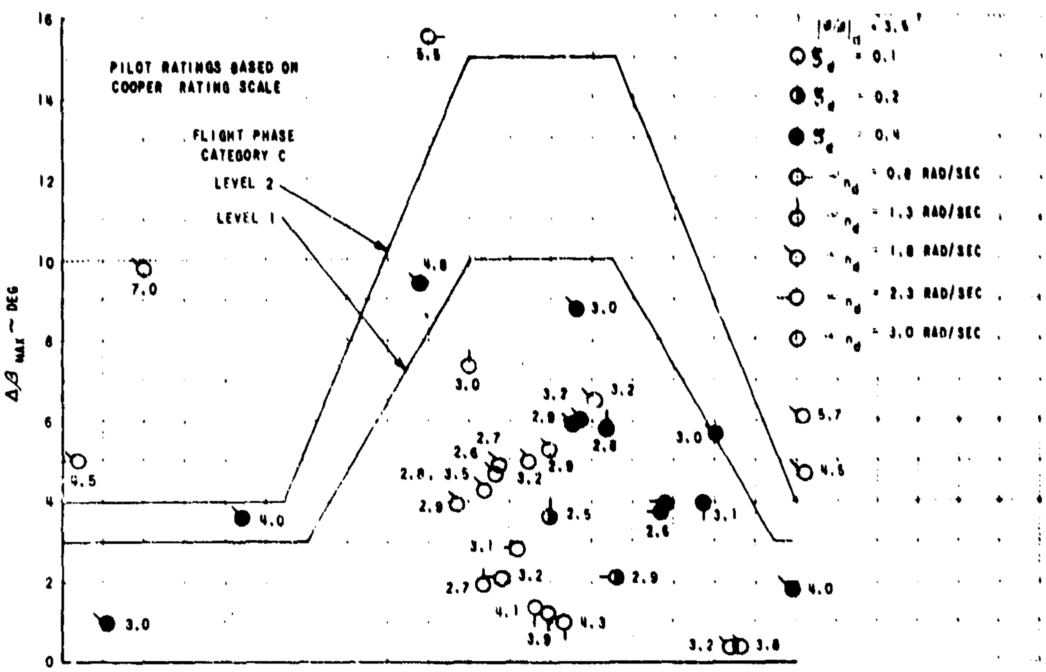


Figure 6 (3.3.2.4)
FLIGHT PHASE CATEGORY C DATA FROM REFERENCES G10 AND G11

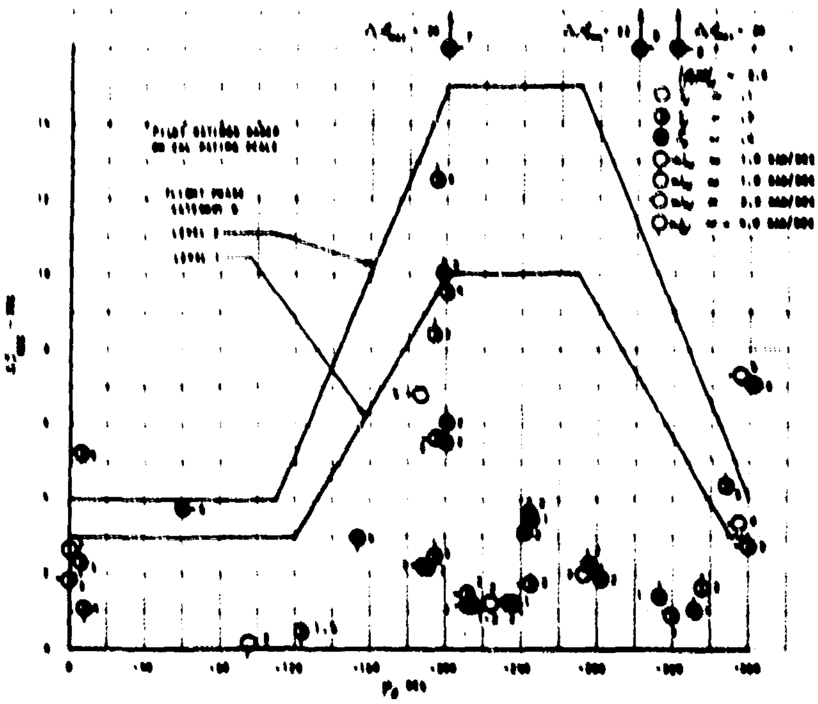


Figure 7 (3.3.2.4)
FLIGHT PHASE CATEGORY B DATA FROM REFERENCE F22

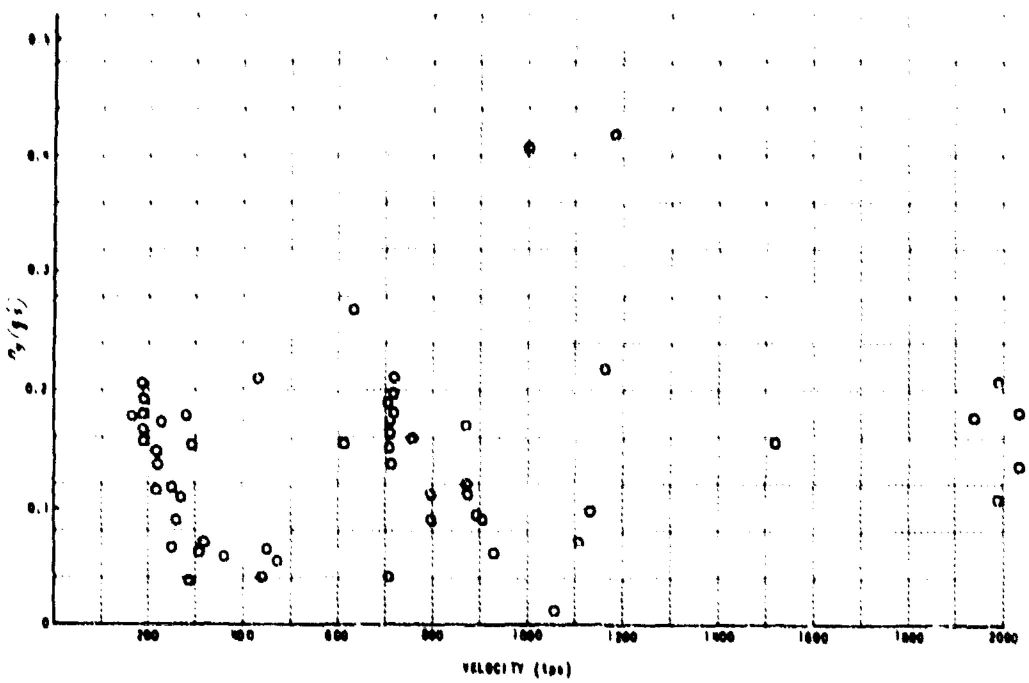


Figure 8 (3.3.2.4)
LATERAL ACCELERATION OF EIGHT CURRENT AIRCRAFT FOLLOWING A STEP AILERON INPUT OF SUFFICIENT MAGNITUDE TO MEET THE REFERENCE A1 ROLL PERFORMANCE REQUIREMENTS

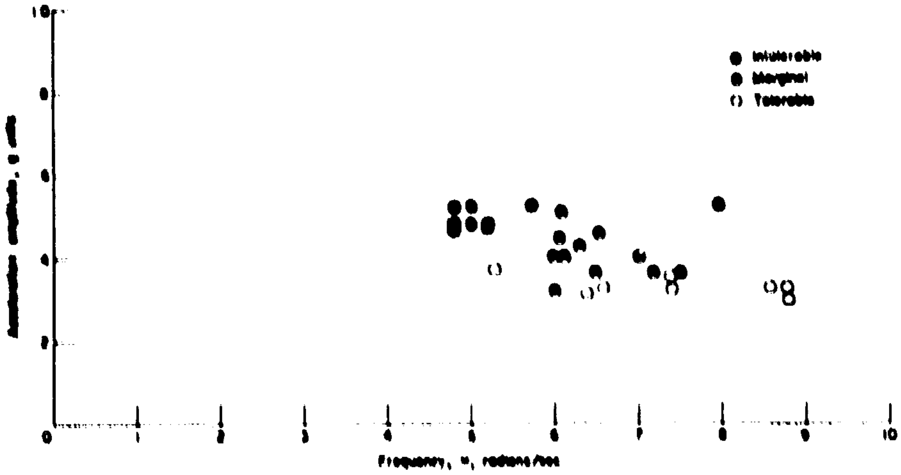


Figure 9 (3.3.2.4)
DISTRIBUTION OF DATA POINTS CONCERNING TOLERABLE OR INTOLERABLE LATERAL ACCELERATIONS DUE TO OSCILLATORY ROLLING MOTIONS.
(FROM REFERENCE F37)

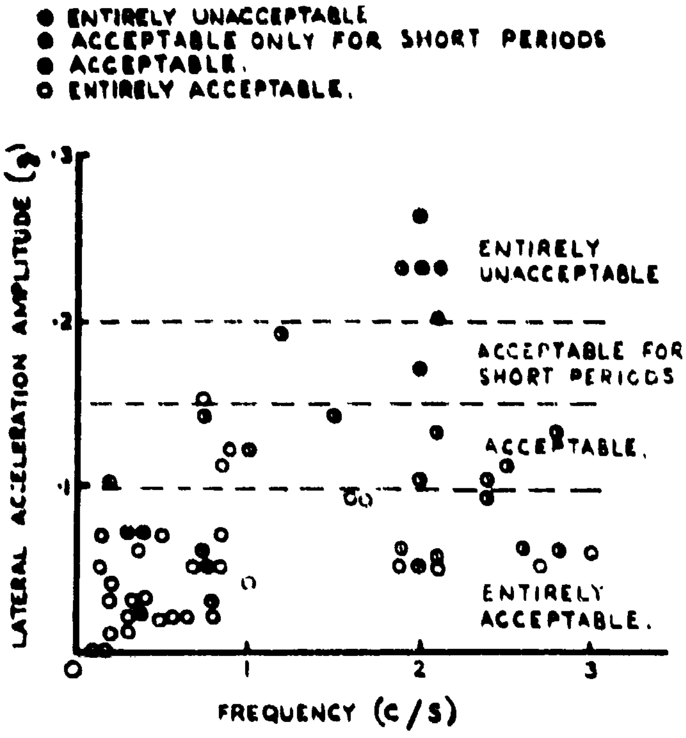


Figure 10 (3.3.2.4)
SUBJECTIVE TOLERANCE TO OSCILLATORY MOTION IN ROLL.
(FROM REFERENCE M49)

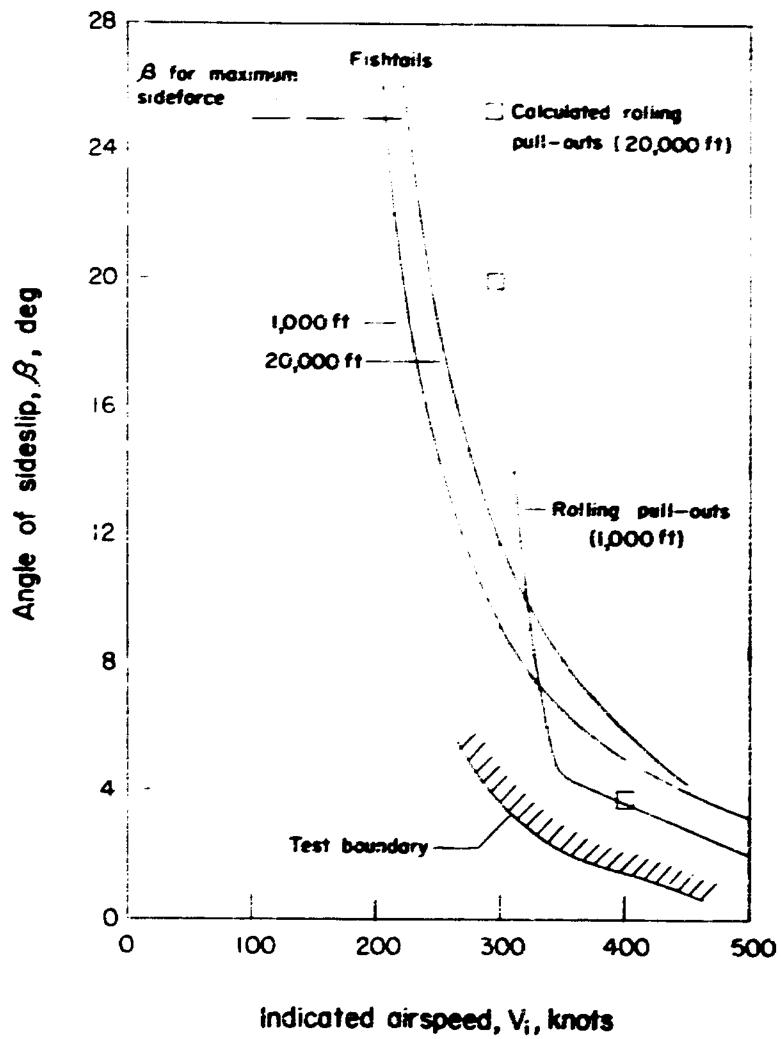


Figure 11 (3.3.2.4)
COMPARISON OF TEST RESULTS FOR THE F-86A AIRPLANE WITH MAXIMUM CALCULATED VALUES OF SIDESLIP DURING FISHTAIL AND ROLLING PULL-OUT MANEUVERS (FROM REFERENCE B80)

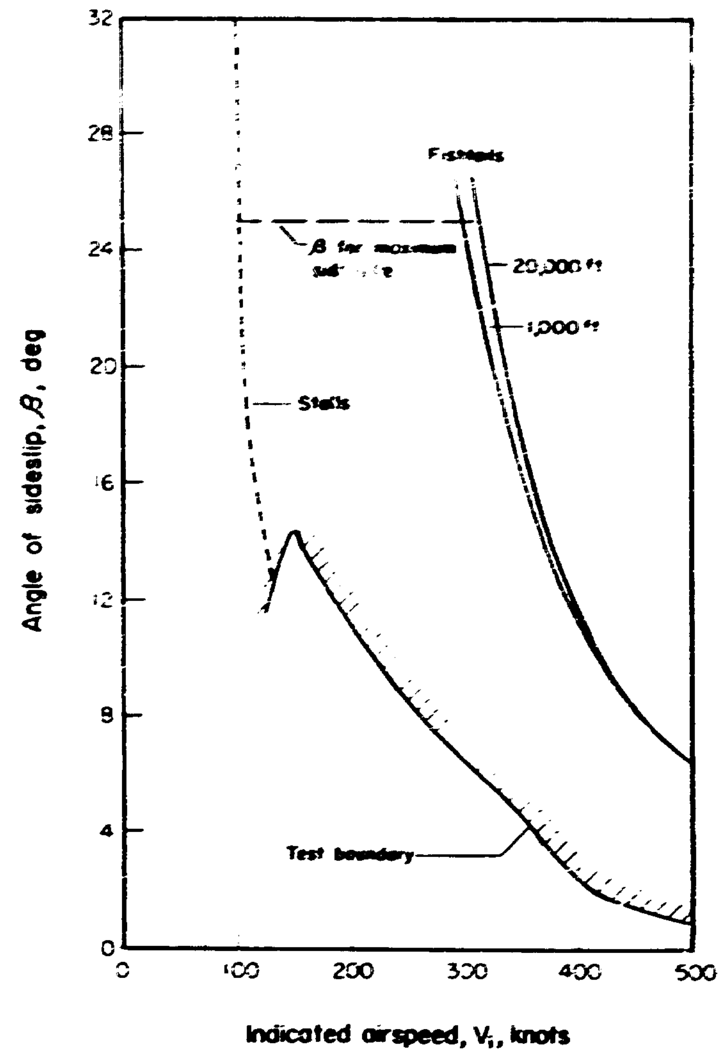


Figure 12 (3.3.2.4)
COMPARISON OF TEST RESULTS FOR THE F-84 AIRPLANE WITH MAXIMUM CALCULATED VALUES OF SIDESLIP DURING FISHTAIL MANEUVERS (FROM REFERENCE B80)

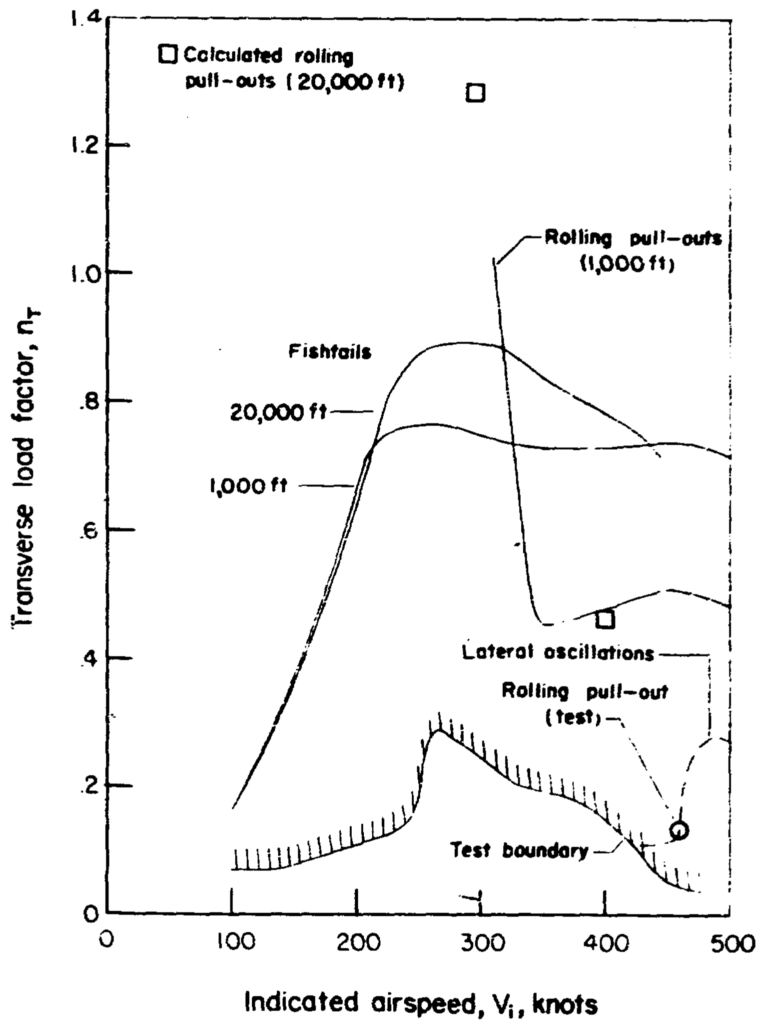


Figure 13 (3.3.2.4)
COMPARISON OF TEST RESULTS FOR THE F-86A AIRPLANE WITH MAXIMUM CALCULATED VALUES OF TRANSVERSE LOAD FACTOR DURING FISHTAIL AND ROLLING PULL-OUT MANEUVERS. (FROM REFERENCE B80)

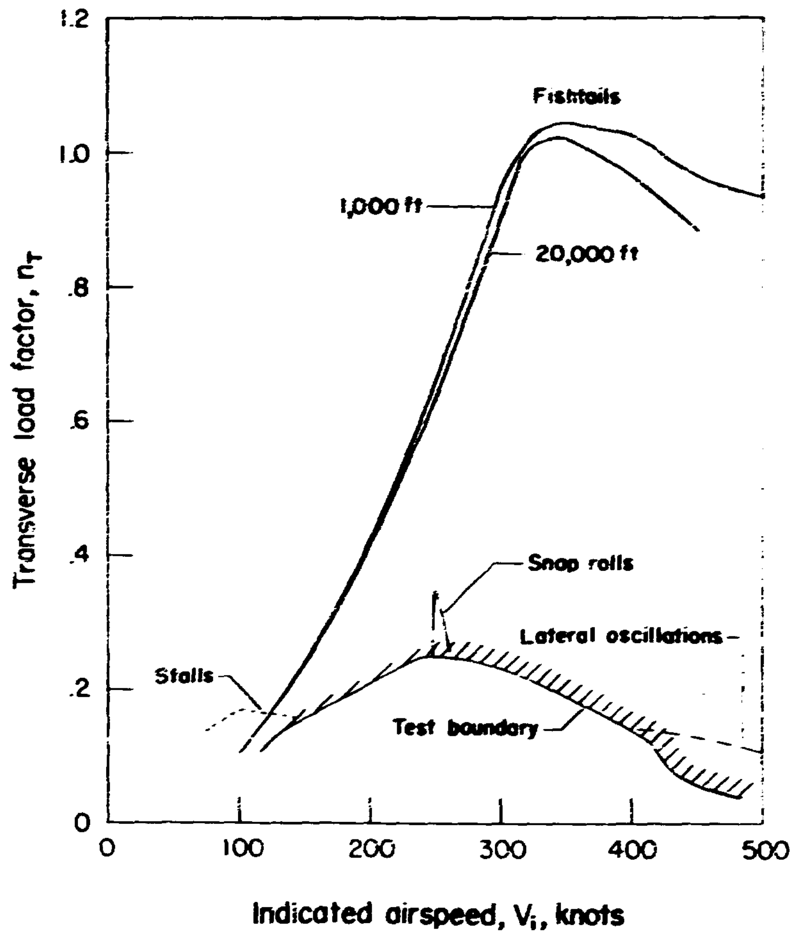


Figure 14 (3.3.2.4)
COMPARISON OF TEST RESULTS FOR THE F-84 AIRPLANE WITH MAXIMUM CALCULATED VALUES OF TRANSVERSE LOAD FACTOR DURING FISHTAIL MANEUVERS. (FROM REFERENCE B80)

3.3.2.5 CONTROL OF SIDESLIP IN ROLLS

REQUIREMENT

3.3.2.5 Control of sideslip in rolls. In the rolling maneuvers described in 3.3.4, but with the rudder pedals used for coordination for all Classes, directional-control effectiveness shall be adequate to maintain zero sideslip with a rudder pedal force not greater than 50 pounds for Class IV airplanes in Flight Phase Category A, Level 1, and 100 pounds for all other combinations of Class, Flight Phase Category, and Level.

RELATED MIL-F-8785 PARAGRAPHS

3.4.14

DISCUSSION

This paragraph, like Paragraph 3.4.14 in MIL-F-8785, is aimed at ensuring that full coordination can be achieved during rapid turn entries with reasonable rudder-pedal forces. The allowable rudder-pedal forces have been reduced from 180 pounds to the specified values since 180 pounds was considered to be excessive, particularly for Class IV airplanes. Since the rudder-pedal deflections can be rather intricate, the requirement is stated in terms of control effectiveness rather than a demonstration in which the pilot keeps zero sideslip.

3.3.2.6 TURN COORDINATION

REQUIREMENT

3.3.2.6 Turn coordination. It shall be possible to maintain steady coordinated turns in either direction, using 60 degrees of bank for Class IV airplanes, 45 degrees of bank for Class I and II airplanes, and 30 degrees of bank for Class III airplanes, with a rudder pedal force not exceeding 40 pounds. It shall be possible to perform steady turns at the same bank angles with rudder pedals free, with an aileron stick force not exceeding 5 pounds or an aileron wheel force not exceeding 10 pounds. These requirements constitute Levels 1 and 2 with the airplane trimmed for wings-level straight flight.

RELATED MIL-F-8785 PARAGRAPHS

None.

DISCUSSION

This requirement is new and was introduced as a result of recommendations received during meetings with industry.

The objective of the requirement is to ensure that only modest rudder pedal forces are required when performing coordinated turns and only modest aileron control forces are required when rudder pedals are not used. The steepness of the turn is a function of airplane Class to correspond with normal operational use.

As mentioned in the discussion of 3.3.1.3, there is a close tie between turn coordination and spiral stability.

3.3.3 LATERAL-DIRECTIONAL PILOT-INDUCED OSCILLATIONS

REQUIREMENT

3.3.3 Pilot-induced oscillations. There shall be no tendency for sustained or uncontrollable lateral-directional oscillations resulting from efforts of the pilot to control the airplane.

RELATED MIL-F-8785 PARAGRAPHS

None.

DISCUSSION

This paragraph has been added on the recommendation of Reference A2. It is very similar to the requirement of Paragraph 3.3.5.2 in the Longitudinal section of MIL-F-8785. The requirement applies to all levels since zero or negative closed-loop damping are to be avoided under any flight condition or failure state.

3.3.4 ROLL CONTROL EFFECTIVENESS

3.3.4.1 ROLL PERFORMANCE FOR CLASS IV AIRPLANES

REQUIREMENT

3.3.4 Roll control effectiveness. Roll performance in terms of bank angle change in a given time, ϕ_t , is specified in table IX and in 3.3.4.1. Aileron control commands shall be initiated from zero roll rate in the form of abrupt inputs, with time measured from the initiation of control-force application. Rudder pedals shall remain free for Class IV airplanes for Level 1, and for all carrier-based airplanes in Category C Flight Phases for Levels 1 and 2; but otherwise, rudder pedals may be used to reduce sideslip that retards roll rate (not to produce sideslip that augments roll rate) if rudder pedal inputs are simple, easily coordinated with aileron-control inputs, and consistent with piloting techniques for the airplane Class and mission. Roll control shall be sufficiently effective to balance the airplane in roll throughout the Service Flight Envelope in the atmospheric disturbances of 3.7.3 and 3.7.4.

3.3.4.1 Roll performance for Class IV airplanes. Additional or alternate roll performance requirements are specified for Class IV airplanes in 3.3.4.1.1 through 3.3.4.1.4. These requirements take precedence over table IX.

3.3.4.1.1 Air-to-air combat. For Class IV airplanes in Flight Phase CO, the roll performance requirements are:

	Time to roll through 90 degrees	Time to roll through 360 degrees
a. Level 1 - - - - - - - - - - -	1.0 second	2.8 seconds
b. Level 2 - - - - - - - - - - -	1.3 seconds	3.3 seconds
c. Level 3 - - - - - - - - - - -	1.7 seconds	4.4 seconds

3.3.4.1.2 Ground attack with external stores. The roll performance requirements for Class IV airplanes in Flight Phase GA with large complements of external stores may be relaxed from those specified in table IX, subject to approval by the procuring activity. For any external loading specified in the contract, however, the roll performance shall not be less than:

a. Level 1 - - - - - - - - - - - 90 degrees in 1.7 seconds

b. Level 2 - - - - - - - - - - - 90 degrees in 2.6 seconds

c. Level 3 - - - - - - - - - - - 90 degrees in 3.4 seconds.

For any asymmetric loading specified in the contract, aileron control power shall be sufficient to hold the wings level at the maximum load factors specified in 3.2.3.2 in the atmospheric disturbances of 3.7.3.

3.3.4.1.3 Roll rate characteristics for ground attack. Class IV airplanes in Flight Phase GA shall be able to roll through 180 degrees in not more than twice the time to roll through 90 degrees. This requirement specifies Level 1 with the rudder pedals remaining free throughout the maneuver and Levels 2 and 3 with the rudder pedals employed to reduce sideslip in the manner described in 3.3.4.

3.3.4.1.4 Roll response. Stick-controlled Class IV airplanes in Category A Flight Phases shall have a roll response to aileron control force not greater than 15 degrees in 1 second per pound for Level 1, and not greater than 25 degrees in 1 second per pound for Level 2. For Category C Flight Phases, the roll sensitivity shall be not greater than 7.5 degrees in 1 second per pound for Level 1, and not greater than 12.5 degrees in 1 second per pound for Level 2. In case of conflict between the requirements of 3.3.4.1.4 and 3.3.4.2, the requirements of 3.3.4.1.4 shall govern.

MIL-F-8785B

TABLE IX. Roll Performance Requirements

Class	Flight Phase Category	Level 1	Level 2**	Level 3
I	A	$\phi_t = 60°$ in 1.3 sec	$\phi_t = 60°$ in 1.7 sec	$\phi_t = 60°$ in 2.6 sec
	B	$\phi_t = 60°$ in 1.7 sec	$\phi_t = 60°$ in 2.5 sec	$\phi_t = 60°$ in 3.4 sec
	C†	$\phi_t = 30°$ in 1.3 sec	$\phi_t = 30°$ in 1.8 sec	$\phi_t = 30°$ in 2.6 sec
II	A	$\phi_t = 45°$ in 1.4 sec	$\phi_t = 45°$ in 1.9 sec	$\phi_t = 45°$ in 2.8 sec
II	B	$\phi_t = 45°$ in 1.9 sec	$\phi_t = 45°$ in 2.8 sec	$\phi_t = 45°$ in 3.8 sec
II-L	C†	$\phi_t = 30°$ in 1.8 sec	$\phi_t = 30°$ in 2.5 sec	$\phi_t = 30°$ in 3.6 sec
II-C	C†	$\phi_t = 25°$ in 1.0 sec	$\phi_t = 25°$ in 1.5 sec	$\phi_t = 25°$ in 2.0 sec
III	A	$\phi_t = 30°$ in 1.5 sec	$\phi_t = 30°$ in 2.0 sec	$\phi_t = 30°$ in 3.0 sec
	B	$\phi_t = 30°$ in 2.0 sec	$\phi_t = 30°$ in 3.0 sec	$\phi_t = 30°$ in 4.0 sec
	C†	$\phi_t = 30°$ in 2.5 sec	$\phi_t = 30°$ in 3.2 sec	$\phi_t = 30°$ in 4.0 sec
IV	A*	$\phi_t = 90°$ in 1.3 sec	$\phi_t = 90°$ in 1.7 sec	$\phi_t = 90°$ in 2.6 sec
	B	$\phi_t = 90°$ in 1.7 sec	$\phi_t = 90°$ in 2.5 sec	$\phi_t = 90°$ in 3.4 sec
	C†	$\phi_t = 30°$ in 1.0 sec	$\phi_t = 30°$ in 1.3 sec	$\phi_t = 30°$ in 2.0 sec

* Except as the requirements are modified in 3.3.4.1

† For takeoff, the required bank angle can be reduced proportional to the ratio of the maximum rolling moment of inertia for the maximum authorized landing weight to the rolling moment of inertia at takeoff, but the Level 1 requirement shall not be reduced below the listed value for Level 3.

**At altitudes below 20,000 feet at the high-speed boundary of the Service Flight Envelope, the Level 3 requirements may be substituted for the Level 2 requirements with all systems functioning normally.

RELATED MIL-F-8785 PARAGRAPHS

3.4.16, 3.4.16.1, 3.4.16.7, 3.7.5

DISCUSSION

General

Roll control effectiveness determines the maneuverability of an airplane in roll, analogous to the way in which limit load factor or elevator effectiveness determines the maneuverability of an airplane in pitch. It is thus a parameter of fundamental importance.

The roll performance requirements of MIL-F-8785 are stated in terms of average $pb/2V$, peak $pb/2V$, $pb/2$, bank angle in one second and steady-state roll rate. Aircraft industry representatives voiced a great deal of dissatisfaction with the existing lateral control requirements, particularly with the parameter $pb/2V$, and suggested that a complete revision was in order. They also expressed dissatisfaction with the large number of parameters that is currently used, in addition to those listed above, to specify roll performance. According to Reference G5,

> "A new criterion for lateral control power should be determined which would ideally be applicable to both large and small airplanes, to both slow and fast airplanes, and to both conventional and unconventional airplane configurations."

A difficulty in attempting to specify rational, realistic roll performance requirements in this general specification is that roll performance is probably more closely related to airplane type and mission than is any other characteristic. A detailed analysis revealed that there should logically be far more sets of roll performance requirements than there are adequate data to support. A great deal of weight has therefore been placed on relationships between roll performance required to maneuver and to counter the response to atmospheric disturbances, and on those aspects of the MIL-F-8785 roll performance requirements that did stand the test of time.

The characteristics of a large number of current aircraft were examined (References P1 to P52). Where sufficient data were found, the roll performance characteristics of an airplane were related to its acceptability in performing its operational missions. The roll characteristics were also compared to the applicable MIL-F-8785 requirement and to the applicable requirement of Reference A1. The results of this study are presented in Appendix VD. The study showed that although the $pb/2V$ requirements are adequate for conventional propeller-driven aircraft, they are too stringent for high-speed or small-wing-span airplanes, and are too lenient for low-speed or large-wing-span airplanes. The study also strongly supports the roll performance requirements of Reference A1 even though current aircraft have, in general, slightly poorer roll performance capabilities than will be required by Reference A1.

Bank Angle in a Specified Time, ϕ_t

In Reference F8, it was concluded from an extensive survey of roll flying qualities that, "For combat and cruise conditions, the pilot opinion aspects of roll performance are most accurately and conveniently measured in terms of the bank angle achievable in a given time in response to an abrupt full aileron (stick) input." In keeping with this concept, for fighter aircraft in combat, Reference F8 uses the parameter ϕ_1 (bank angle in 1 sec), and for heavy bombers or transports in cruise uses the parameter ϕ_2 (bank angle in 2 sec). For all airplanes in approach and landing, Reference F8 uses steady roll rate to cover maneuvering requirements, and the parameter $t_{30°}$, (time to roll through 30°) as the most practical parameter to cover the gust recovery aspects. Reference A14 (see Figure 1) has incorporated similar ideas into a proposed requirement for fighter aircraft in which steady roll rate is plotted against τ_R for power and combat configurations.

In revising the roll performance section of the specification, the main objectives were to select parameters that were direct and meaningful to the pilot, and that most precisely defined the physical situation appropriate to the aircraft type and Flight Phase Category. Although no single parameter was found to be adequate to specify requirements for the large combination of aircraft types and flight phases, the number of parameters which have to be considered has been minimized. Roll performance requirements have been specified in terms of time to bank to characteristic bank angles, as a function of airplane Class and Flight Phase Category. In this way the same type of parameter can be used throughout, yet each parameter can be tailored to the basic maneuverability characteristics. The selected bank angles for up-and-away flight (Flight Phase Categories A and B) and landing approach (Flight Phase Category C) are indicated below.

Flight Phase Category	Class			
	I	II	III	IV
A and B	60°	45°	30°	90°
C	30°	30°	30°	30°

Although the parameter t_ϕ is used throughout this discussion, for practical considerations the parameter ϕ_t is used in Reference A1 for specifying roll performance requirements. Not only is the numerical value of ϕ_t proportional to roll performance (whereas t_ϕ is inversely proportional to roll performance), but when roll-sideslip coupling affects the bank angle time history, the bank angle achieved in a specified time is a more meaningful measure of roll performance than is the time to reach a specified bank angle.

In Reference B80 (Figures 2-5), a study was made of the actual rates and amounts of control motion used by service pilots while performing squadron operational training missions with fighter aircraft. From the data it can be seen that of the three performance parameters $pb/2V$, p_{max} and $t_{90°}$, the values of the parameter $t_{90°}$ that were available and that were achieved remained far more constant over most of the speed range than did those of $pb/2V$ or p_{max}. Since in general the available performance as measured by $t_{90°}$ was greater than the achieved performance, this would indicate that for those aircraft performing that role, the pilots desired a given value of $t_{90°}$. Since the achieved values of $t_{90°}$ were in general between 1 and 2 seconds, the parameter ϕ_1 would also be descriptive of the physical situation and would also be a valid parameter to use. On the other hand, for large aircraft which have longer maneuver times, performance during the first second is not as critical, so their roll performance requirements should more properly be based on a longer period of time. Thus, by basing individual airplane Class requirements on a realistic physical situation, the chance of specifying a poor requirement is reduced.

Further indication of the merits of t_ϕ or ϕ_t parameters can be obtained from examination of the data and iso-rating curves of Reference F30 for fighter airplanes (Figures 6-8). The curves of constant pilot rating of Figure 8 were more closely approximated by curves of $t_{90°}$ or ϕ_1 than by curves of p_{ss}. On the other hand, curves of very poor pilot rating, where the pilot was in danger of losing control of the aircraft and where maneuvering would be expected to be restricted, are more closely approximated by curves of constant time to 30 degrees.

An approach sometimes taken is to specify roll performance as a function of such parameters as $L'_{\delta_a}\delta_{a_{max}}$ or p_{ss} versus τ_R. However, for non one-degree-of-freedom rolling motions where the Dutch roll and roll mode are coupled, the roll performance can vary widely from that indicated by a point or curve on the $L'_{\delta_a}\delta_{a_{max}}$ or p_{ss} versus τ_R plot. This can be seen, for example, in the roll rate time histories presented in Figure 27 of the discussion of Paragraph 3.3.2.2; all time histories have the same values of $L'_{\delta_a}\delta_{a_{max}}$ and τ_R. Thus a direct requirement on desired performance is not only simpler, but more precise.

The range of load factors over which all requirements apply (3.1.7, 3.1.8) is from $n_o(-)$ to $n_o(+)$ or from $n(-)$ to $n(+)$; but commonly demonstration is at 1g. AFFTC pilots recommend that the roll requirements apply up to 2/3 or 3/4 n_L, at least for Class IV airplanes (in rolling pullouts, the MIL-A-8861 structural limit is 0.8 n_L). But a few apparently satisfactory operational airplanes do employ elevons, with reduced roll-control authority at aft stick positions. However, no systematic study seems to have been made. For lack of sufficient data, we have left the matter open in this paragraph of the general specification. Of course, when bank-to-bank maneuvers are used for flight demonstration, moderate load factors up to 1.4 (for 90° total change in bank angle) will be involved. Certain tactics require high roll control capability at $n \neq 1$: for example, rolling onto a ground target from a pop-up maneuver or avoiding an obstacle during a pull-up from a ground-attack run.

Discussion of the quantitative criteria specified in Table IX and 3.4.4.1 of Reference A1 is given in the following sections by airplane Class. To aid in correlating data, plots of initial roll acceleration versus τ_R

were prepared, which shows lines of constant steady roll rate following a step aileron input based on a one-degree-of-freedom rolling motion, that is, $p_{ss} = \dot{p}_{step} \tau_R$. Curves of constant ϕ_1 , t_{90° , t_{60° , t_{45° , and t_{30° were calculated from the equation $\phi_t = \dot{p} \tau_R \left[t - \tau_R(1 - e^{-t/\tau_R})\right]$ and were superimposed on these plots based on a 0.2-second aileron ramp input for Class I and IV airplanes, a 0.4-second aileron ramp input for Class II airplanes, and a 0.6-second aileron ramp input for Class III airplanes. The effect of the ramps was calculated by inserting step inputs at one half the ramp time. These ramps were selected to at least partially take into account flight control system effects when comparing data which use different parameters. The roll performance and roll damping requirements are presented in graphical form in Figures 9, 10, 11 and 12 for Class I, II, III and IV airplanes, respectively. The Flight Phase Categories and Levels are denoted by the symbols A_1, A_2, B_1, B_2, C_1 and C_2.

Requirements for Class IV Airplanes

Flight Phases Categories A and B

Comments from industry pertaining to roll performance requirements indicated that, in their opinion, the present requirement of ϕ_1 = 90° is too high. Two manufacturers specifically recommended a value of ϕ_1 = 60°, and several manufacturers stated that even ϕ_1 = 60° was higher than was needed for ground attack or than could be obtained with many present day fighters when carrying a large load of external stores. Reference F8 concluded that, "For fighter airplanes in combat condition, ... ϕ_1 greater than about 50° appears to be a reasonably well supported requirement from both the standpoint of pilot rating and usable maneuvering capability." The Reference F30 data (Figure 13) indicate that $\phi_1 \approx$ 100° is considered optimum, but that $\phi_1 \approx$ 40° to 50° is still rated as satisfactory. From consideration of the Reference F30 data and from the requirements in MIL-F-8785, Reference A14 (Figure 14) proposed a lower limit equivalent to ϕ_1 = 40° and an upper limit equivalent to ϕ_1 = 90°. Finally, Reference B80 (Figure 5) shows that fighter pilots utilize ϕ_1's as great as 90° over at least portions of the flight envelope, and Reference F31 (Figure 15) shows that, at least for the F-100, pilots require far greater roll acceleration and roll rate capability in performing aerial combat maneuvers than in carrying out air-to-ground gunnery and bombing.

Consideration of these factors and of the fact that fighter aircraft normally operate over a wide speed range, carry a great variety of external stores, and perform many different tasks, dictates that the roll performance requirements for Class IV airplanes must be flexible. This flexibility is achieved through the application of Flight Phase Categories and Levels, and through special requirements directed at ground attack with external stores and at air-to-air combat.

A bank angle of 60 degrees in one second (ϕ_1 = 60°) was selected as the basic Flight Phase Category A, Level 1 requirement. Since, as explained previously, requirements for Class IV airplanes in up-and-away flight are to be expressed in terms of rolling maneuvers through 90 degrees, the roll performance requirement as obtained from Figure 12 is ϕ_t = 90 degrees in 1.3 seconds. It should be noted that, since this requirement must be met throughout the Operational Flight Envelope, in general greater roll performance than ϕ_t = 90 degrees in 1 second may be achieved when not operating near the limits of the Operational Flight Envelope (Figure 5).

Maneuverability is extremely important in fighter airplanes and, as we have seen, most critical for air-to-air combat. The fighter pilots who were queried were strongly in favor of keeping the ϕ_t = 90° requirement for that case. Therefore, to ensure that fighters in air-to-air combat have the roll performance capability that experience shows is required, requirements directed specifically at the CO Flight Phase have been specified in 3.3.4.1.1. These require that during a 360 degree roll the airplane be able to roll through the first 90 degrees in:

(a) for Level 1: 1 second (ϕ_t = 90°) and average 150 degrees per second through the remaining 270 degrees,

(b) for Level 2: 1.3 seconds ($\phi_t \approx$ 60°) and average 135 degrees per second through the remaining 270 degrees, and

(c) for Level 3: 1.7 seconds ($\phi_t \approx$ 40°) and average 100 degrees per second through the remaining 270 degrees.

The requirements at ϕ > 90° are to prevent undue interference of roll limiters or roll-sideslip coupling with rolling performance. It appears that 360° rolls are actually used in air-to-air combat.

Many factors were considered in determining the Level 2 requirements. In MIL-F-8785 the roll performance requirements for operation on the alternate control system are, in general, half the requirements for normal operation. In Reference F30, (Figure 13) the Cooper rating 5 curve corresponds to about 50% of the roll performance of the Cooper rating 3.5 curve. Since, as explained previously, these Cooper ratings roughly correspond to the Level 2 and Level 1 boundaries respectively, the factor of two is again evident. Using this as a criterion, the Level 2 requirement for Flight Phase Category A would be half the Level 1 requirement, or ϕ_t = 30 degrees. Other factors, though, must be considered.

Under the concept used in Reference A1, there are conditions under which the requirement of ϕ_t = 90 degrees in 1.3 seconds would not have to be met, even if all systems were functioning normally and the aircraft were in up-and-away flight: (1) in a B Category Flight Phase, or (2) outside the Operational Flight Envelope but within the Service Flight Envelope. To ensure that, regardless of planned missions or usage a fighter aircraft will always be able to perform fighter missions throughout its Service Envelope, the Flight Phase Category A, Level 2 requirements have been made the same as the Flight Phase Category B, Level 1 requirements, and both are somewhat more stringent than would be required if the sole criterion were that the flying qualities be acceptable in the event of a probable, but infrequent, emergency.

Thus the problem reduces to determination of the minimum flying qualities required to perform any fighter mission adequately. Several reports, References F27, F32 and F38, which were directed specifically at this area, were examined. These reports, which are discussed on pp. 27-29 of Reference F8, indicate that surprisingly low roil performance is required for most fighter missions. From consideration of all these factors, the value proposed by Reference A14 of $\phi_1 = 40°$ was selected as the basis for the Flight Phase Category B, Level 1 requirement and the Flight Phase Category A, Level 2 requirement. From Figure 12, the equivalent value in terms of rolling maneuvers through 90 degrees is $\phi_t = 90°$ in 1.7 seconds.

ϕ_t = 90 degrees in 1.7 seconds was also used as the basic roll performance requirement for the ground attack flight phase with large complements of external stores (3.3.4.1.2). This relaxation is proposed in deference to the severe design problem with large rolling moments of inertia, since available data and discussions with pilots tend to support some relaxation from the ϕ_t = 90 degrees in 1.3 second requirement. Further, since asymmetric store loadings both intentional and as a result of failures, are a fact of life, the requirement that the pilot be able to hold the wings level while pulling "g" is a reasonable and necessary requirement. In 3.3.4.1.3 roll rate requirements for ground attack have been specified to ensure that the airplane can roll effectively through the large (180 degrees) bank angles involved in that Flight Phase.

The reduction in flying qualities in going from Level 1 to Level 2 for Flight Phase Category B was made about the same as going from Level 1 to Level 2 for Flight Phase Category A. This resulted in a requirement of $\phi_t = 90°$ in 2.5 seconds for Flight Phase Category B, Level 2.

From Reference F30, (Figure 13), a Cooper rating of 6.5 corresponds roughly with a steady roll rate of 15 deg/sec. The curve almost exactly corresponds to a $t_{30°}$ = 2.6 seconds curve. Since, as discussed previously, a Cooper rating of 7 was selected as the Level 3 criterion, a requirement of ϕ_t = 30 degrees in 3 seconds, which corresponds roughly to a steady-state roll rate of 12 deg/sec, is indicated as the Level 3 requirement from these data. Operational personnel considered this limited roll capability to be far too low to actually retain control in many present-day missions. The Level 3 requirements are therefore more stringent than is indicated by the Reference F30 data. They were selected instead to require approximately half the rolling moment required for Level 2.

No maximum roll performance limit has been specified. Although the data of Reference F30 (Figures 6-8) indicate that an upper limit exists, it is believed that the degradation in flying qualities is caused by oversensitivity rather than too much control power. Requirements on sensitivity to cover this condition have been specified in Paragraphs 3.3.4.1.4 and 3.3.4.2 of Reference A1 (see the discussion of 3.3.4.2).

Flight Phase Category C

The requirement of MIL-F-8785 specifies average $pb/2V$ = 0.05 for the first 30 degrees of bank. For an airplane with b = 35 ft and V = 200 fps, $p_{AV} \approx$ 33°/sec for the first 30°. Thus $\phi_t \approx$ 30 degrees in 1 second.

In Reference G10 (Figure 16), which reports on lateral-directional flying qualities for the power approach, ϕ_1 = 30 degrees was required to achieve a Cooper rating of 3.5 - 4.0 and ϕ_1 = 20 degrees was required to achieve a Cooper rating of 5.0 - 5.5. These ratings correspond to Level 1 and Level 2, respectively.

In Reference G5 (Figure 17), another in-flight program using fighter aircraft in the landing approach, it was found that for all carrier-based aircraft, ϕ_1 = 20° was required to obtain performance between marginal and unsatisfactory. "Marginal" was defined as "barely enough response to pick up a wing with no control to spare," and unsatisfactory was defined as "insufficient response to pick up a wing consistently to assure a safe landing." This ties in well with what is required for Level 3 flying qualities. Reference G5 recommends this minimum requirement for land-based fighter airplanes as well, but lacks supporting data.

The values of ϕ_1 = 30° and ϕ_1 = 20° have thus been selected as the basis for the Flight Phase Category C requirements. As previously discussed, rolling maneuvers through 30 degrees were selected to express Flight Phase Category C requirements, so the requirements as obtained from Figure 12 are: ϕ_t = 30 degrees in 1 second and ϕ_t = 30 degrees in 1.3 seconds for Level 1 and Level 2, respectively.

As for the A and B Category Flight Phases, the Level 3 requirements were selected to provide approximately half the rolling acceleration provided by the Level 2 requirements.

Requirements for Class III airplanes

Flight Phase Categories A and B

In discussing required characteristics of large airplanes, Reference F8 concludes: "For heavy bombers or transports in cruise, bank angle in two seconds, ϕ_2 , greater than about 25° - 30° for normal loadings seem to be indicated by the little available data (Table II)" - see Figure 18. Since the condition described pertains to Flight Phase Category B, and since no conflicting data have been uncovered, ϕ_t = 30 degrees in 2.0 seconds was selected as the Flight Phase Category B, Level 1 requirement.

Little data were found that pertained to requirements for Category A Flight Phases, such as high-speed, low-level flight. However, one manufacturer of large aircraft suggested a value of ϕ_1 = 30° for this situation. In Reference F27 (Figures 19-21), the problem of terrain and collision avoidance is examined, and from Figures 19, 20 and 21 it can be seen that the point of diminishing return on roll performance is at lower values of average roll rate for lower normal-acceleration capability. While this study indicates that average roll rates up to 60 deg/sec would still provide significant improvement in avoidance capability for maneuvers initiated at short range, the resulting requirement could possibly be unrealistic. If p_{AV} = 30 deg/sec is selected as a value that provides reasonable payoff for a 2g avoidance maneuver, the corresponding roll performance in terms of time to 30°, assuming $\Delta\phi$ = 60° and τ_R = 1 second, is approximately $t_{30°}$ = 1.5 seconds. For lack of more definite data, and since $t_{30°}$ = 1.5 second would make the A and B Flight Phase Category requirements for Class III airplanes in the same ratio as was found necessary for Class IV airplanes, ϕ_t = 30 degrees in 1.5 seconds was selected as the Flight Phase Category A, Level 1 requirement.

As with Class IV airplanes, it was felt advisable to make the Flight Phase Category A, Level 2 requirements the same as the Flight Phase Category B, Level 1 requirements; so far Flight Phase Category A, Level 2, ϕ_t = 30 degrees in 2.0 seconds was selected.

For Flight Phase Category B, Level 2, the MIL-F-8785 approach of halving requirements was used, giving a ϕ_t = 30 degrees in 3.0 seconds for this condition. Level 3 requirements are approximately half the corresponding Level 2 requirements.

Flight Phase Category C

In the last few years, considerable data pertaining to roll performance requirements for large aircraft in the landing approach have been generated. References C2 and C7 report on in-flight and fixed-base evaluation studies and Reference B39 presents pertinent data for 21 large aircraft.

Reference C2 (Figures 22-24) concluded that ϕ_1 from 4 to 5 degrees in 1 second is satisfactory. Although there is considerable scatter, the data indicate that ϕ_1 = 2 to 3 degrees in 1 second is required for acceptable flying qualities.

Reference C7 (Figure 25) shows that with a maximum wheel throw of ±60 degrees (the value specified in Reference A1), a Cooper rating of 3.5 corresponds to $\phi_1 \approx$ 4.5 degrees in 1 second, while a Cooper rating of 5 corresponds to $\phi_1 \approx$ 2.0 degrees in 1 second. These Cooper ratings correspond to Level 1 and Level 2 flying qualities respectively. When the data of Figure 26 (also from Reference C7) are examined, however, it can be seen that more control power is required to achieve a given level of flying qualities in the presence of atmospheric disturbances. The data indicate that, for the level of atmospheric disturbance simulated in this program, $\phi_1 \approx$ 6 degrees in 1 second is required for Level 1 and a $\phi_1 \approx$ 4 degrees in 1 second is required for Level 2. The results from the programs of Reference C2 and C7 are presented in Figure 29 at their respective values of τ_R.

The boundaries proposed by Reference B39 (Figures 27 and 28) are also presented in this figure. The presentation is valid since the input aileron ramp time assumed on the figure (0.6 seconds) is almost the same as the ramp time assumed by Reference B39 (0.5 seconds). The upper boundary is drawn between ratings of marginal and unacceptable. The definitions of the terms as used in the Reference B39 study (see p. 17 of Reference B39) suggest that the upper boundary corresponds more closely with Level 2 than Level 1 flying qualities, while the lower boundary closely corresponds to Level 3 flying qualities.

Interpreting the data presented in Figure 29 in terms of the philosophy discussed in "Interpretation of Pilot Rating Data," $\phi_t \approx 30$ degrees in 2.5 seconds was selected as the Level 1 criterion, $\phi_t \approx 30$ degrees in 3.2 seconds was selected as the Level 2 criterion, and $\phi_t \approx 30$ degrees in 4 seconds was selected as the Level 3 criterion.

This criterion is consistent with the conclusions reached in Reference F8 that: "For large airplanes on approach, the most accurate and convenient metric, generally descriptive of pilot desires, is the time required to roll through 30°, $t_{30°}$, following an abrupt maximum aileron (wheel or stick) input. The data available indicated that values of $t_{30°}$ greater than about 3 to 3.5 sec are unacceptable," and that "Large-airplane approach maneuvering requirements seem to demand minimum steady roll rates, p_s, greater than about 12° - 15°/sec." Comparison of the selected criterion with that of Reference F8 (shown as a crosshatched area in Figure 29) shows that the selected Level 2 criteria generally lie within the acceptable/unacceptable band of Reference F8.

Although available data indicate that different requirements should apply to large than to small aircraft for the landing approach, it may be questioned why this should be so. Should not the requirements be the same when the aircraft are doing the same thing?

There are at least two reasons why they should not be expected to be the same: one is the differing response to atmospheric disturbances and the other is tactics. The type of approach and landing pattern that the specification must allow for is far different for fighters than for transports. It must be anticipated that fighters will employ flat breaks or battle breaks, whereas transports will normally make straight-in approaches or employ rectangular patterns. Also, since fighters often take off and land in formation, provision must be made for hitting severe jet wash. This, coupled with the fact that the response to a given level of atmospheric disturbances will normally be more severe for fighters than for transports, dictates that fighters have greater available roll power in the landing approach than do transports.

Requirements for Class I and II airplanes

Since there are very little research data pertaining directly to these classes of airplanes, the approach taken in defining requirements is based on general basic differences in size, maneuverability and response to atmospheric disturbances of the four classes of airplanes.

Historically and rationally, maneuverability in roll has been related to maneuverability in pitch: six-g airplanes have far greater roll performance than two-g airplanes. From examination of structural specifications (listed in Section 6.8 of Reference A1), it was observed that in general, limit load factors required for basic missions associated with Class IV airplanes were high and that limit load factors required for basic missions associated with Class III airplanes were relatively low. Further, limit load factors for basic missions associated with Class I and II airplanes were generally intermediate to those of Class III and IV airplanes, with basic missions associated with Class I airplanes generally requiring higher load factors than basic missions associated with Class II airplanes. Thus, on the basis of maneuverability, roll performance capability (from greatest to least) should be in the order of Class IV, I, II and III. This order is consistent with MIL-F-8785 and is further justified when response to atmospheric disturbances is considered, since response to atmospheric disturbances is generally greatest for Class I and IV airplanes and is progressively less for larger, higher moment of inertia airplanes, that is, Class II and III airplanes.

Thus, on the basis of maneuverability requirement and response to atmospheric disturbances, it was decided that Class I and II roll response requirements should be between those of Class III and Class IV airplanes. This approach is supported by Reference F8 in analyzing roll performance requirements for airplanes intermediate in size between fighters and heavy bombers or transports. Reference F8 states that for intermediate airplane types:

"a. In cruise conditions 'satisfactory' values of p_0 steadily diminish in going from light trainers (T-37A through T-39) to small utility transports (MAC-119A) to medium bombers (B-66B) or fighter-bombers (F-105B).

b. In approach conditions the data are too sparse to show trends but it appears that values of t_{30}, intermediate to those for fighters and heavy bombers are permissible. For example, the B-66B with p_0 and ϕ_1 almost identical to the F-101A, but with a t_{30} of 1.7, is rated excellent, whereas the F-101A with a t_{30} of 1.3 is rated satisfactory."

Since so little data on Class I and II airplanes are available, quantitative requirements were obtained by selecting Class I and II roll performance requirements such that roll performance increases in equal multiples in going from Class III, to Class II, to Class I, to Class IV airplanes. These values were then checked against what data are available to obtain at least partial verification.

Some data on Class I and small Class II airplanes were obtained from Reference B96. In data presented in Figure 30, it could be determined that the airplane the report describes as being sluggish had a ϕ_1 of approximately 17 degrees in the landing approach. The report states that "for satisfactory handling qualities, airplanes of this class should produce helix angles on the order of 0.07 radian, ..." Assuming representative wing spans, approach speeds and roll mode time constants, this transforms to $\phi_1 \approx 20$ degrees. A ϕ_1 of 20 degrees checks almost exactly with the selected value of ϕ_t = 30 degrees in 1.3 seconds for Flight Phase Category C, Level 1 for Class I airplanes. The Class I, Flight Phase Category A and B requirements also look consistent with a $pb/2V$ = 0.07 requirement, even if minimum operational speeds are considered to be quite low.

In Appendix VD, the roll performance characteristics of thirteen Class II airplanes are discussed. These data generally strongly support the specified roll performance requirements for Class II airplanes.

Takeoff

The roll performance requirements for takeoff are relaxed from the Flight Phase Category C requirements as a function of the rolling moment of inertia. A relaxation is possible since normally there is less lateral-directional maneuvering at takeoff than during an approach. Moreover, a relaxation is desirable since a given ϕ_t requirement may be difficult to meet at high takeoff gross weights and rolling moments of inertia. On the other hand, since at high inertias gust response is less than at low inertias, it is possible to reduce the specified Flight Phase Category C ϕ_t requirements proportional to the ratio of rolling moments of inertia, while retaining roughly the same gust recovery capability.

Roll Response

The roll response requirements for Class IV airplanes are discussed in Section 3.3.4.2, Aileron control forces.

Balancing Rolling Moments

The statement in 3.3.4 on balancing rolling moments i[illegible] necessary catch-all requirement. By incorporating specific atmospheric [illegible]sturbances it gives increased assurance of controllability in severe turbulence and discrete gusts.

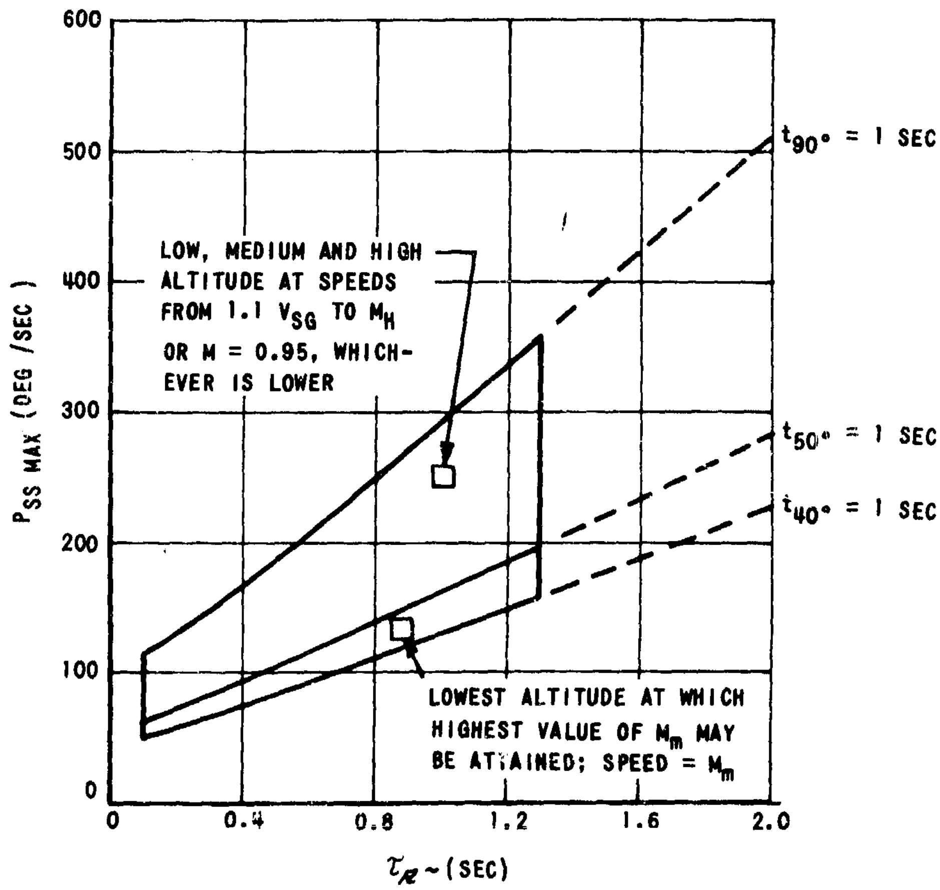


Figure 1 (3.3.4)

PROPOSED ROLL PERFORMANCE REQUIREMENTS FOR (MIL-F-8785) CLASS III AIRCRAFT (FROM REFERENCE A14)

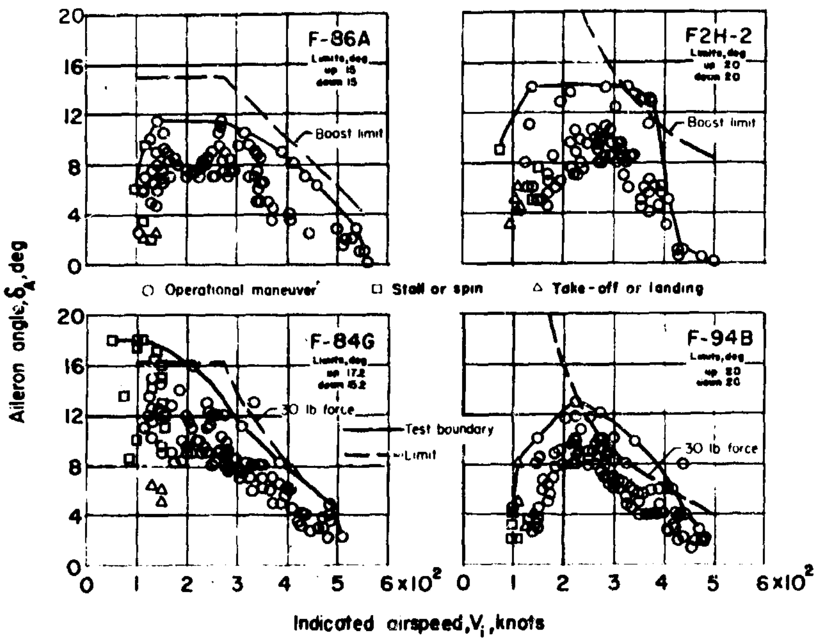


Figure 2 (3.3.4)

COMPARISON OF TEST RESULTS WITH MAXIMUM UP OR DOWN AILERON ANGLES OBTAINABLE IN ABRUPT AILERON ROLLS (FROM REFERENCE B80)

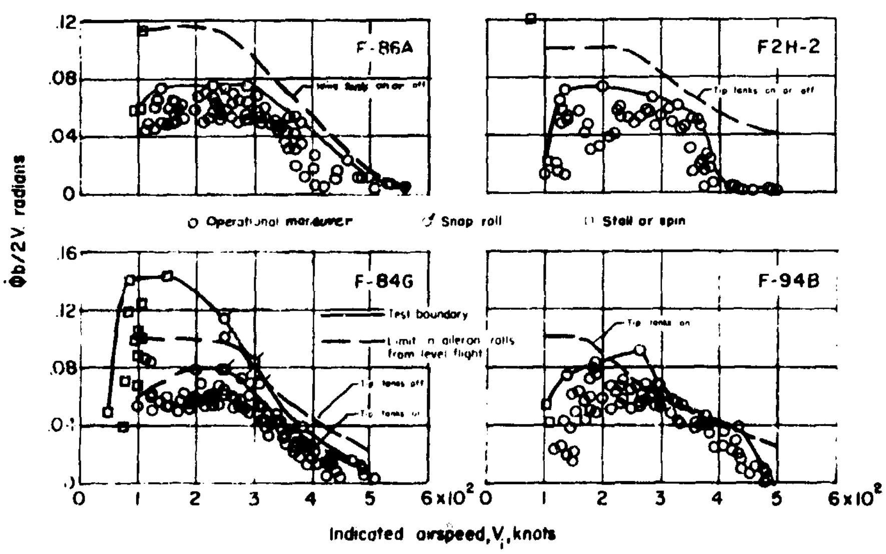


Figure 3 (3.3.4)

COMPARISON OF TEST RESULTS WITH MAXIMUM WING-TIP HELIX ANGLES OBTAINABLE IN ABRUPT AILERON ROLLS (FROM REFERENCE B80)

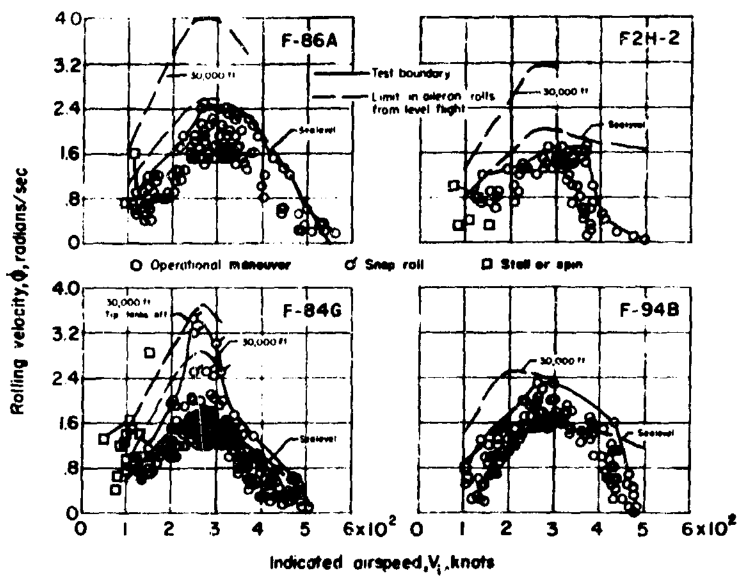


Figure 4 (3.3.4)

COMPARISON OF TEST RESULTS WITH MAXIMUM ROLLING VELOCITIES OBTAINABLE IN ABRUPT AILERON ROLLS (FROM REFERENCE B80)

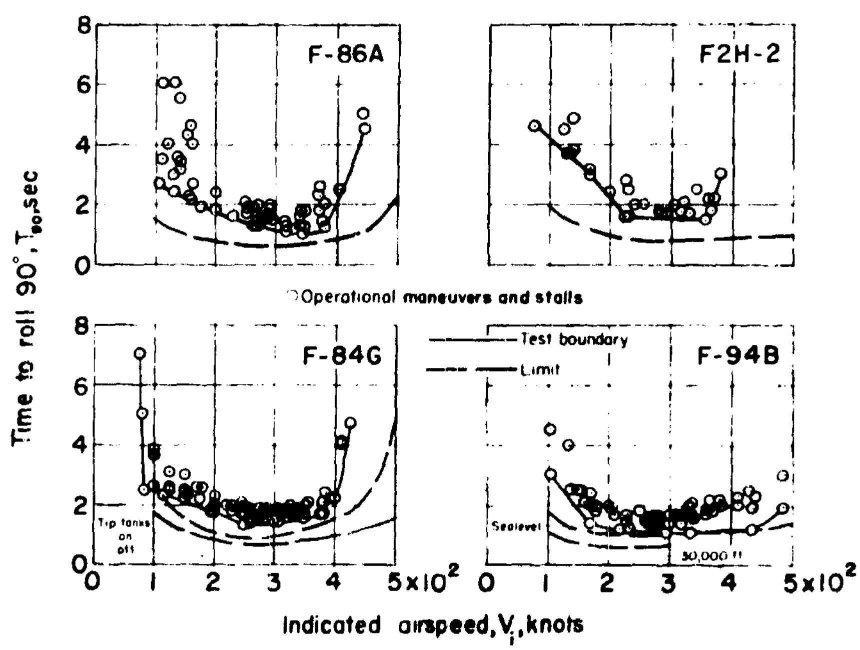


Figure 5 (3.3.4)

COMPARISON OF TEST RESULTS WITH CALCULATED MINIMUM TIMES TO ROLL 90° IN ABRUPT AILERON ROLLS AT SEA LEVEL (FROM REFERENCE B80)

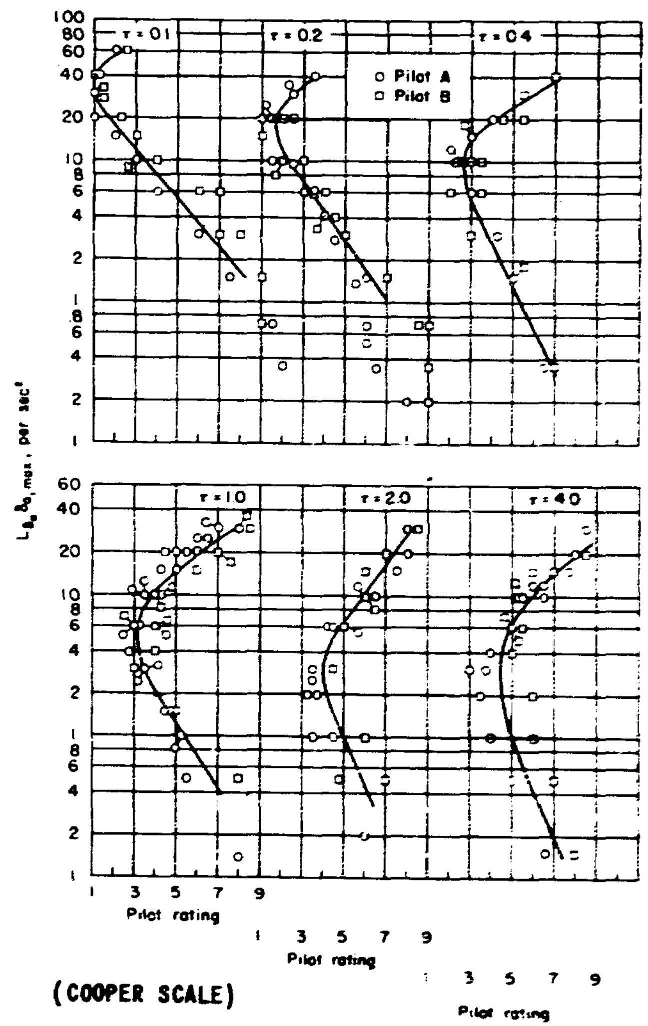


Figure 6 (3.3.4)
VARIATION OF PILOT OPINION WITH $L_{\delta a}\delta_{a\ max}$
FOR CONSTANT VALUES OF τ AS OBTAINED FROM THE
STATIONARY FLIGHT SIMULATOR (FROM REFERENCE F30)

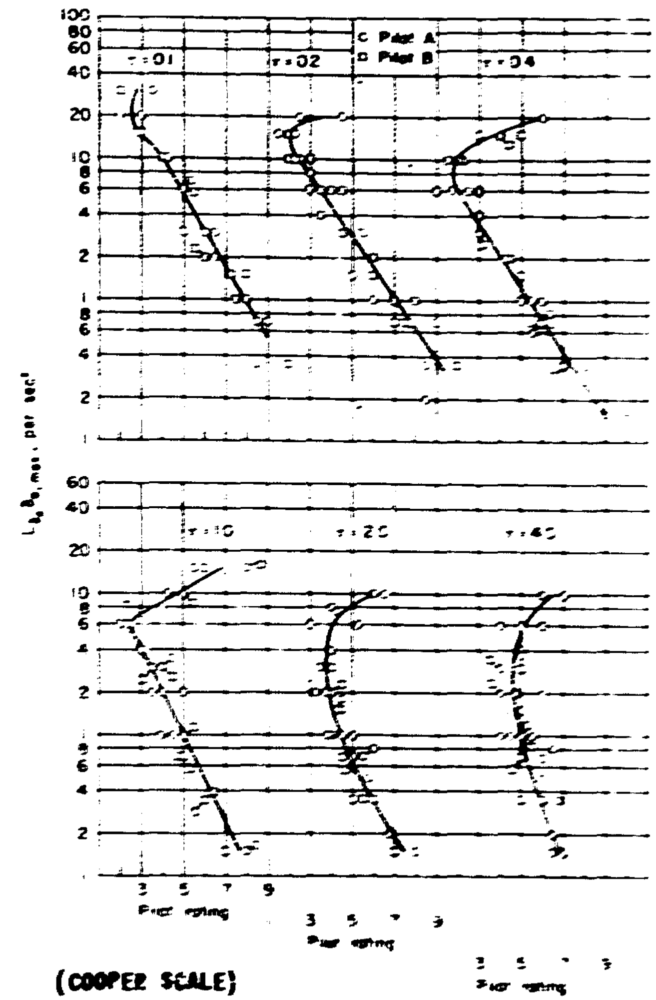


Figure 7 (3.3.4)
VARIATION OF PILOT OPINION WITH $L_{\delta a}\delta_{a\ max}$
FOR CONSTANT VALUES OF τ AS OBTAINED FROM THE
MOVING FLIGHT SIMULATOR (FROM REFERENCE F30)

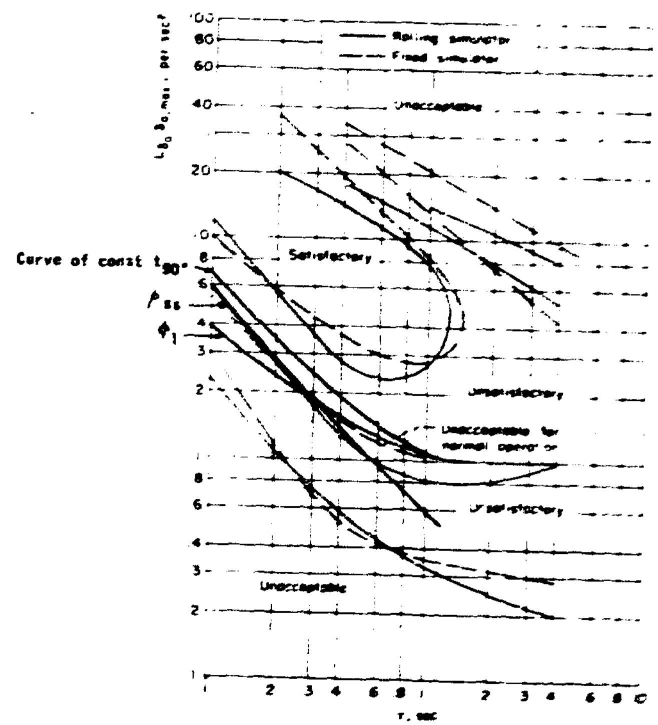


Figure 8 (3.3.4)
COMPARISON OF PILOT OPINION BOUNDARIES OBTAINED FROM THE FIXED AND MOVING FLIGHT SIMULATORS (FROM REFERENCE F30) WITH CURVES OF CONSTANT ϕ_1, $t_{90°}$ AND p_{ss}

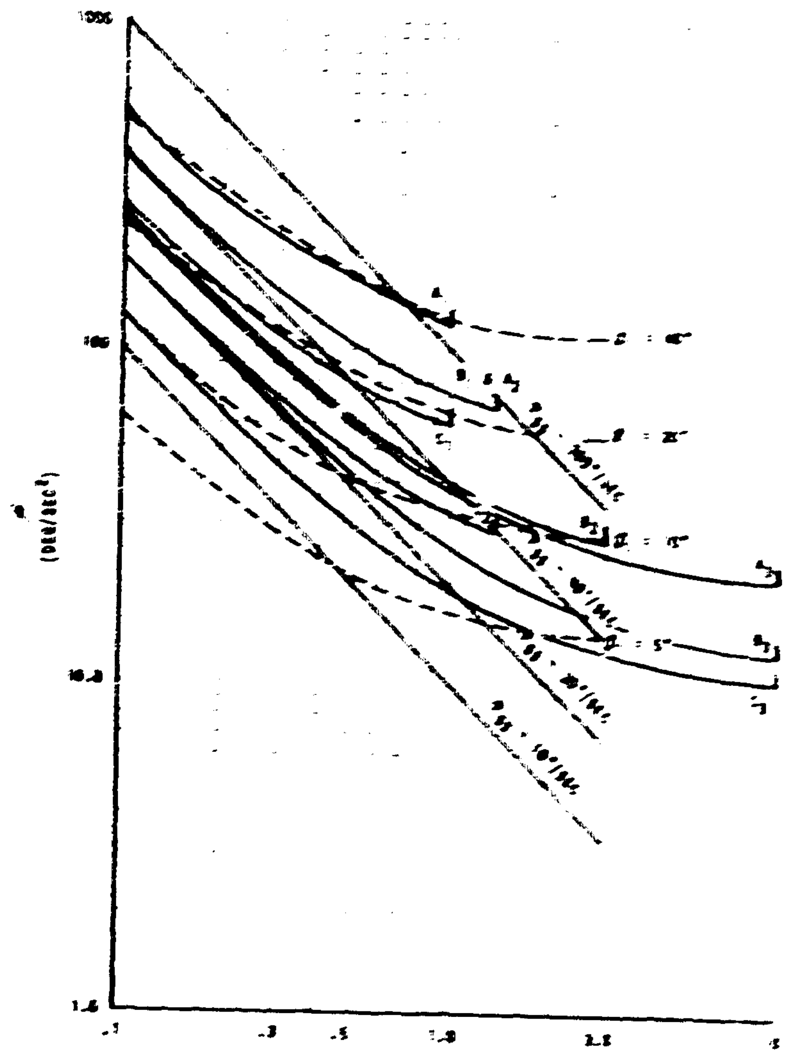


Figure 9 (3.3.4)
ROLL CONTROL EFFECTIVENESS PARAMETERS FOR CLASS I AIRPLANES

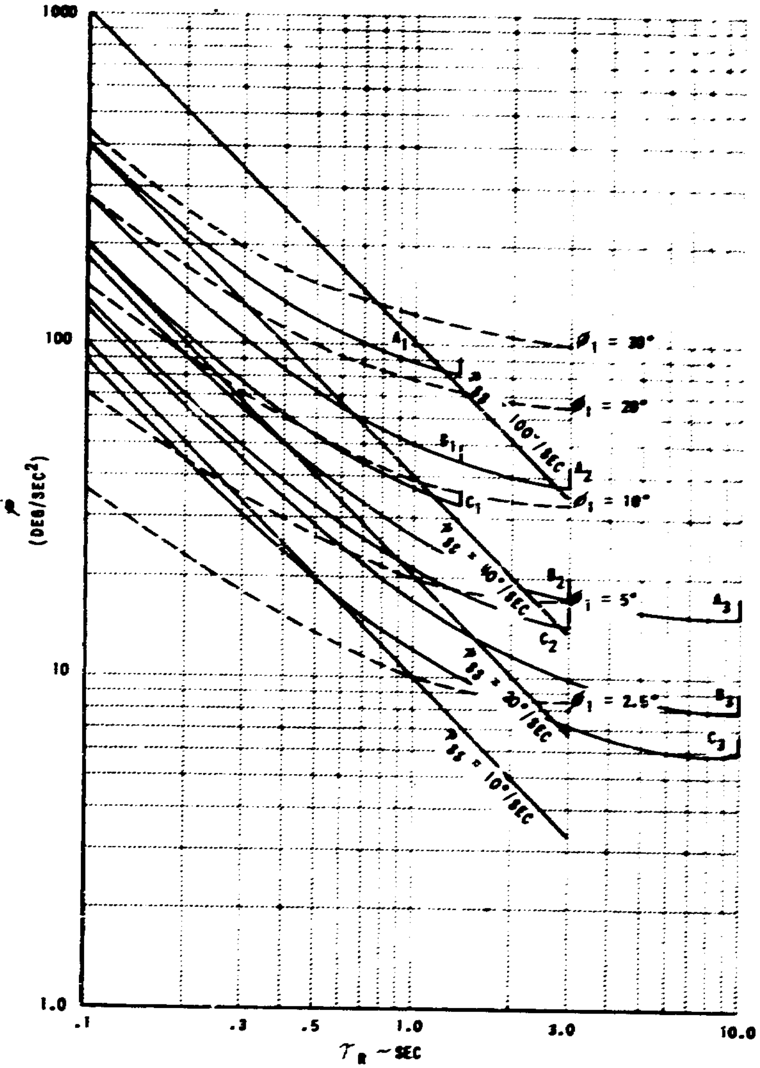


Figure 10a (3.3.4)
ROLL CONTROL EFFECTIVENESS
PARAMETERS FOR CLASS II-L AIRPLANES

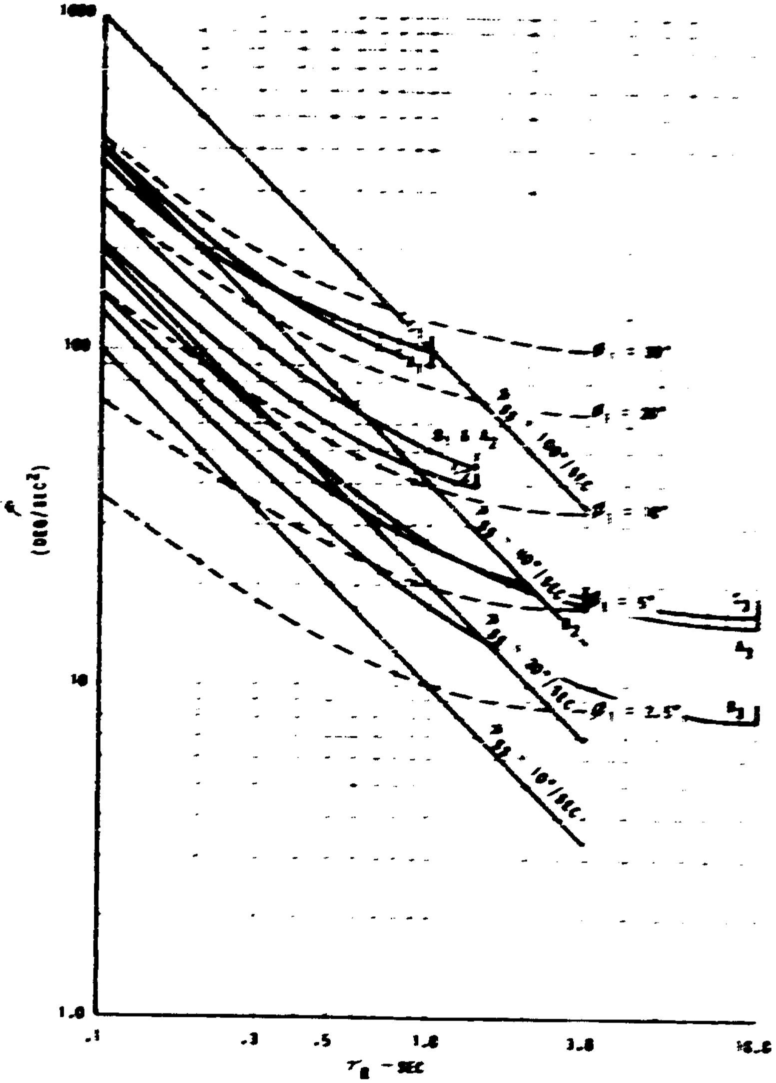


Figure 10b (3.3.4)
ROLL CONTROL EFFECTIVENESS
PARAMETERS FOR CLASS II-C AIRPLANES

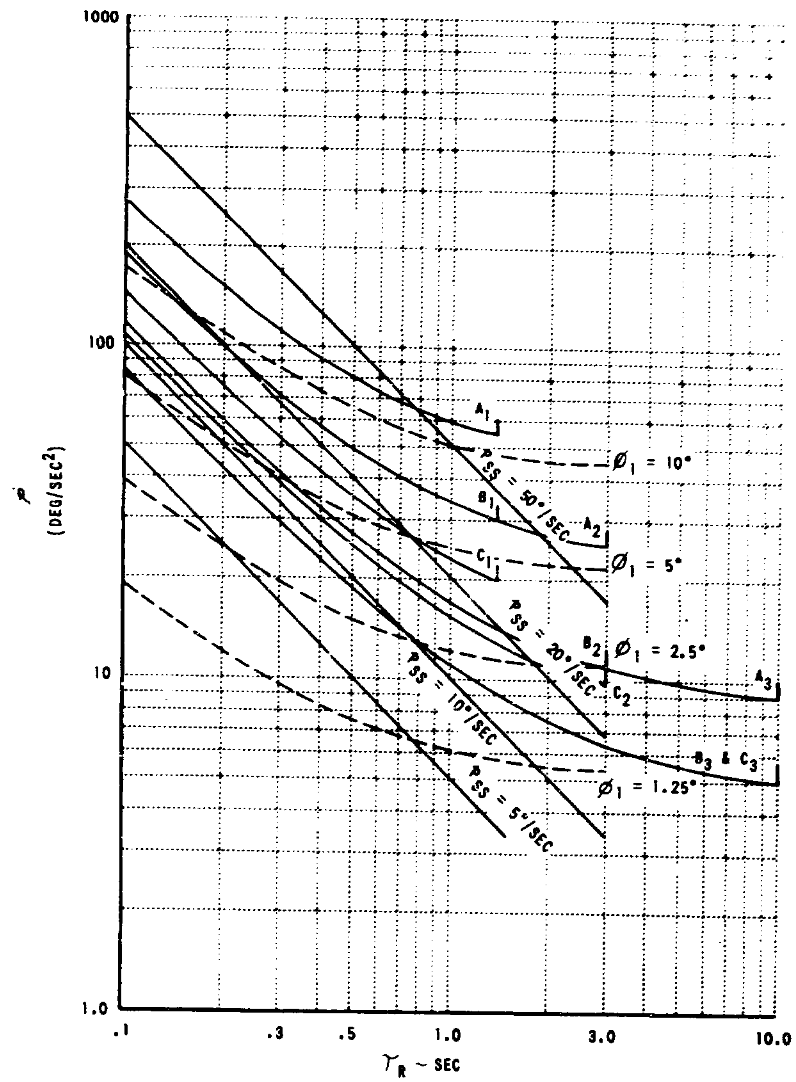


Figure 11 (3.3.4)
ROLL CONTROL EFFECTIVENESS
PARAMETERS FOR CLASS III AIRPLANES

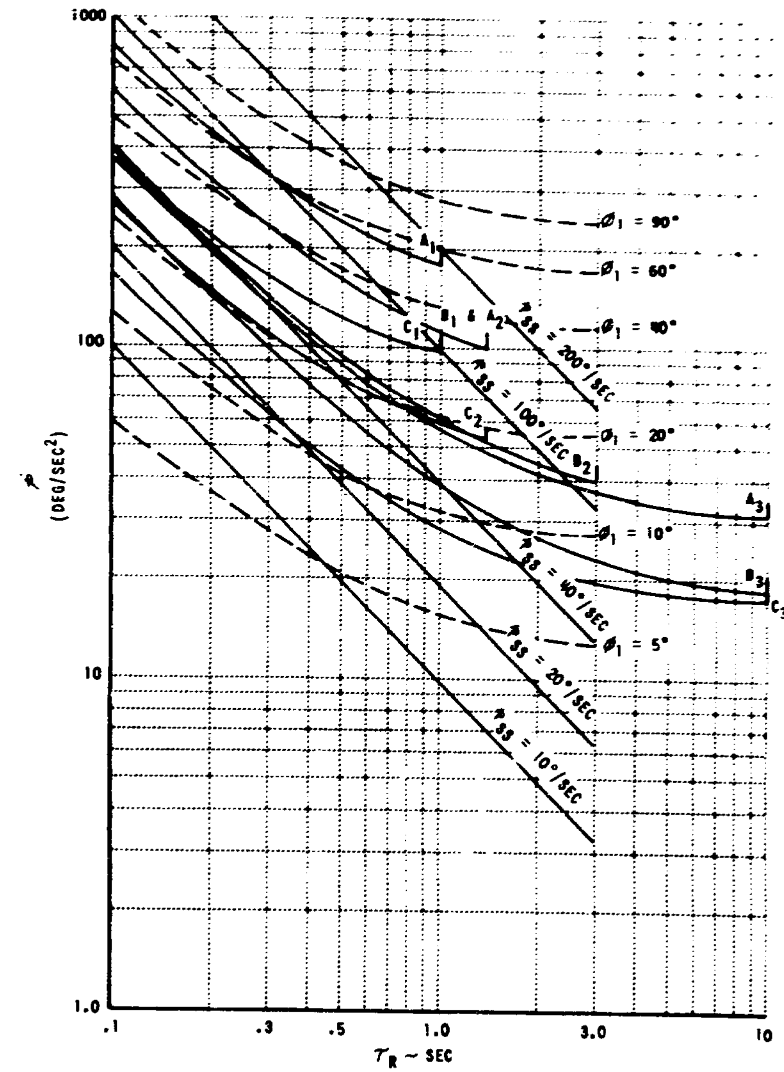


Figure 12 (3.3.4)
ROLL CONTROL EFFECTIVENESS
PARAMETERS FOR CLASS IV AIRPLANES

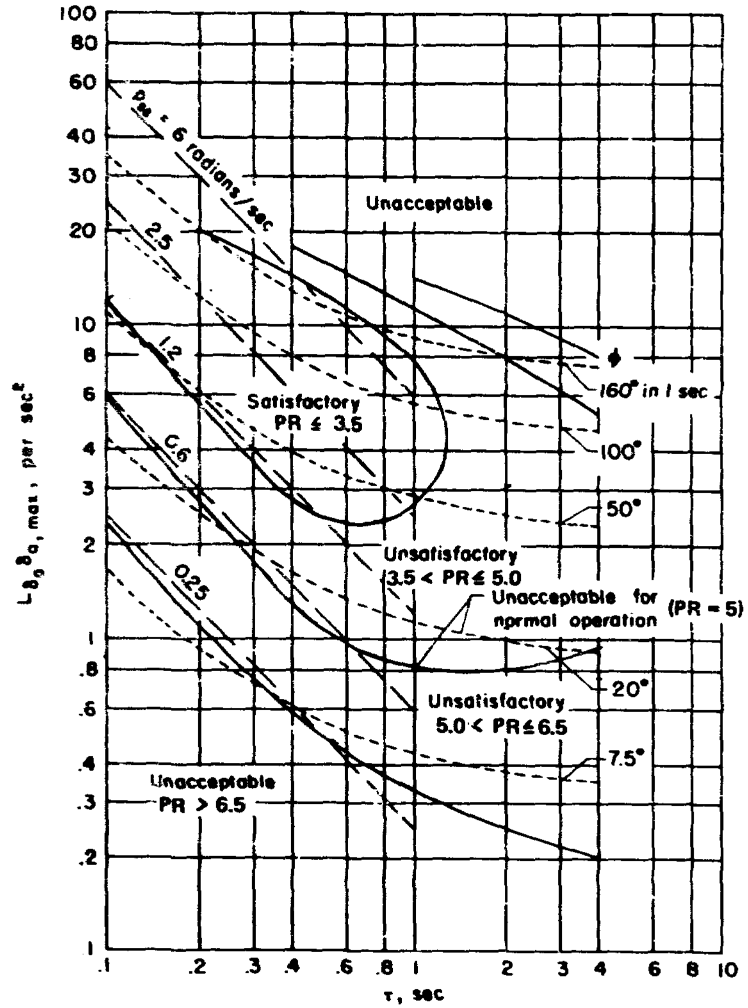


Figure 13 (3.3.4)

COMPARISON OF THE DERIVED ROLL PERFORMANCE CRITERION WITH THE ROLL PERFORMANCE CONCEPT BASED ON BANK ANGLE DISPLACEMENT AT END OF 1 SECOND . (FROM REFERENCE F30)

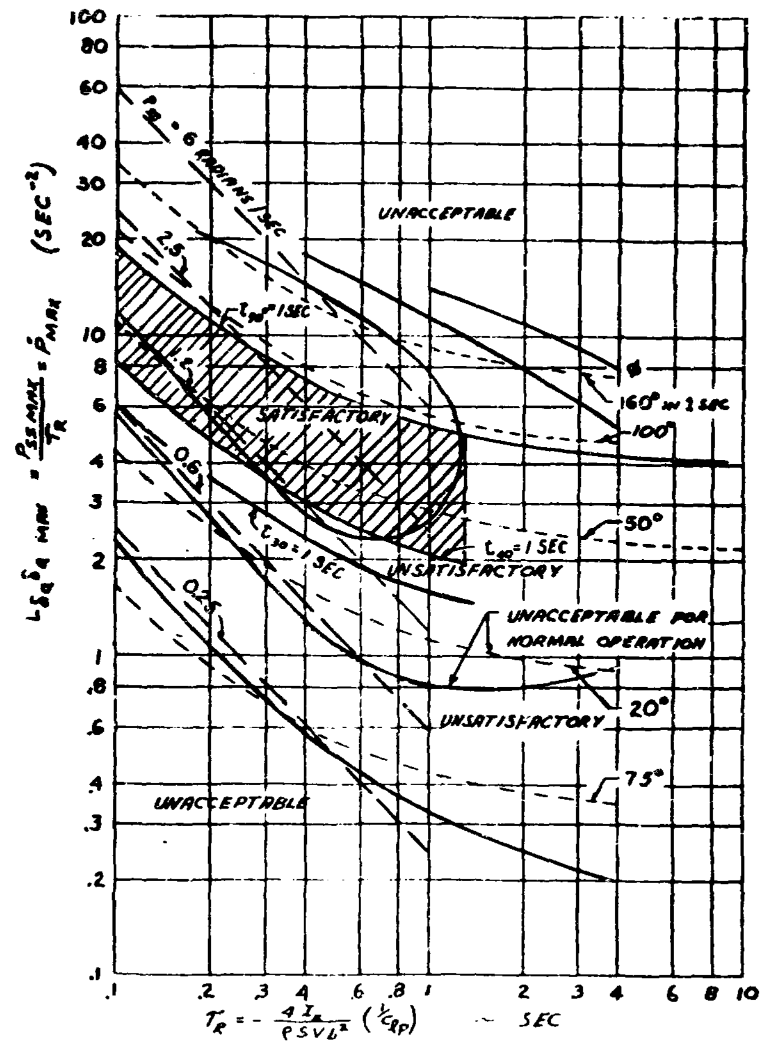
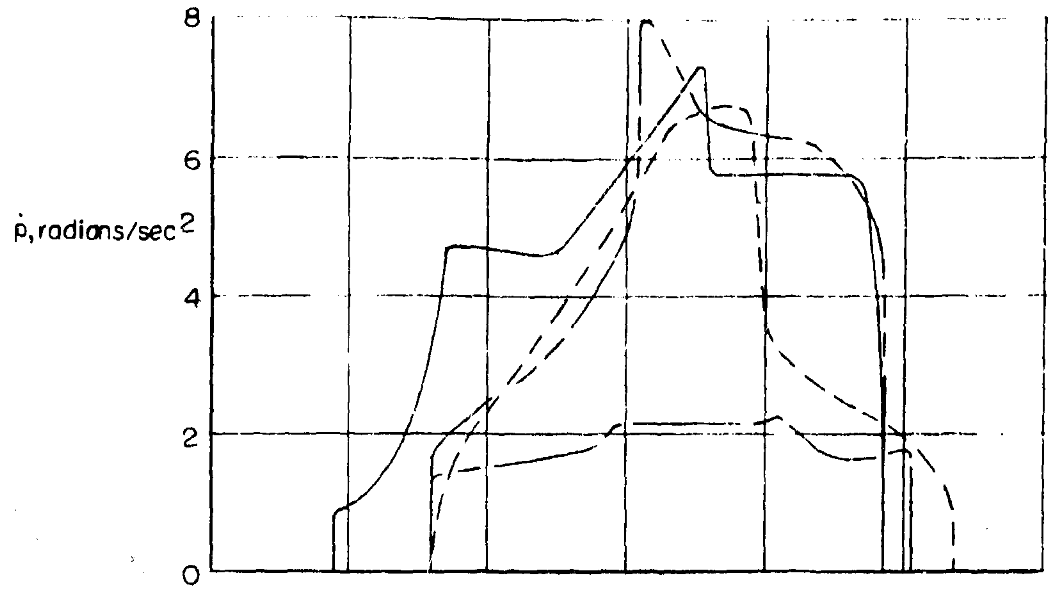
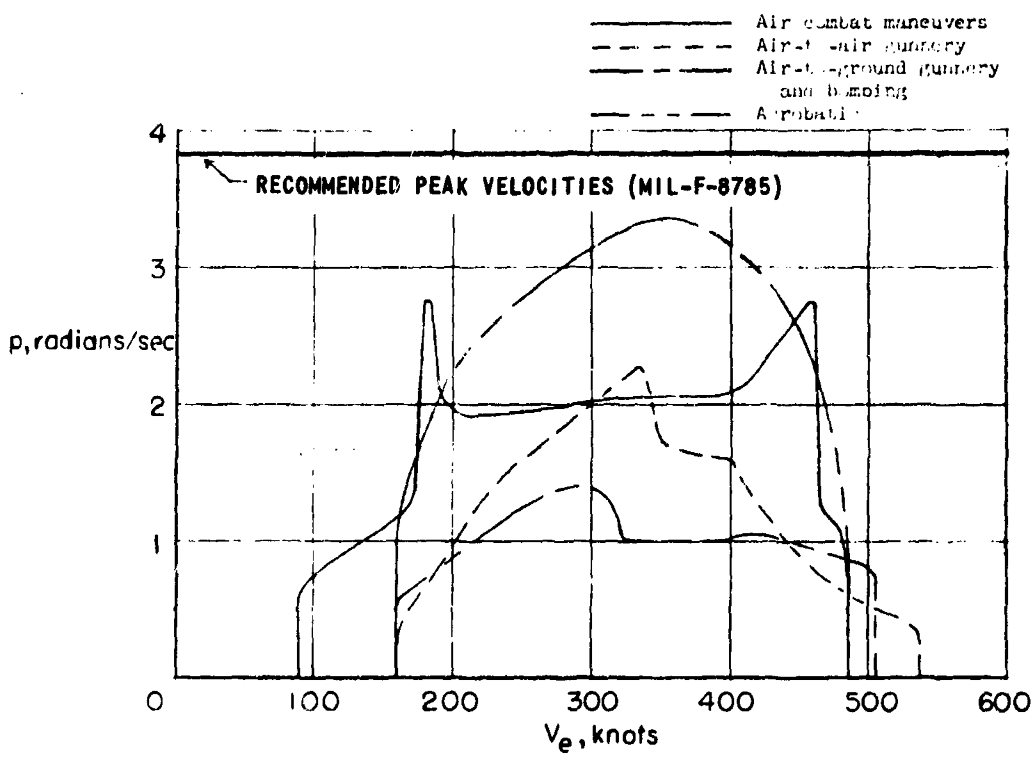


Figure 14 (3.3.4)

A COMPARISON OF THE RECOMMENDED ROLL PERFORMANCE REQUIREMENTS OF REFERENCE (F30) WITH PRESENT REQUIREMENTS (FROM REFERENCE A14)



(a) Rolling acceleration.



(b) Rolling velocity.

Figure 15 (3.3.4)

SUMMARY OF AIRPLANE ROLL CHARACTERISTICS BOUNDARIES AS A FUNCTION OF EQUIVALENT AIRSPEED FOR ALL MANEUVERS. (FROM REFERENCE F31)

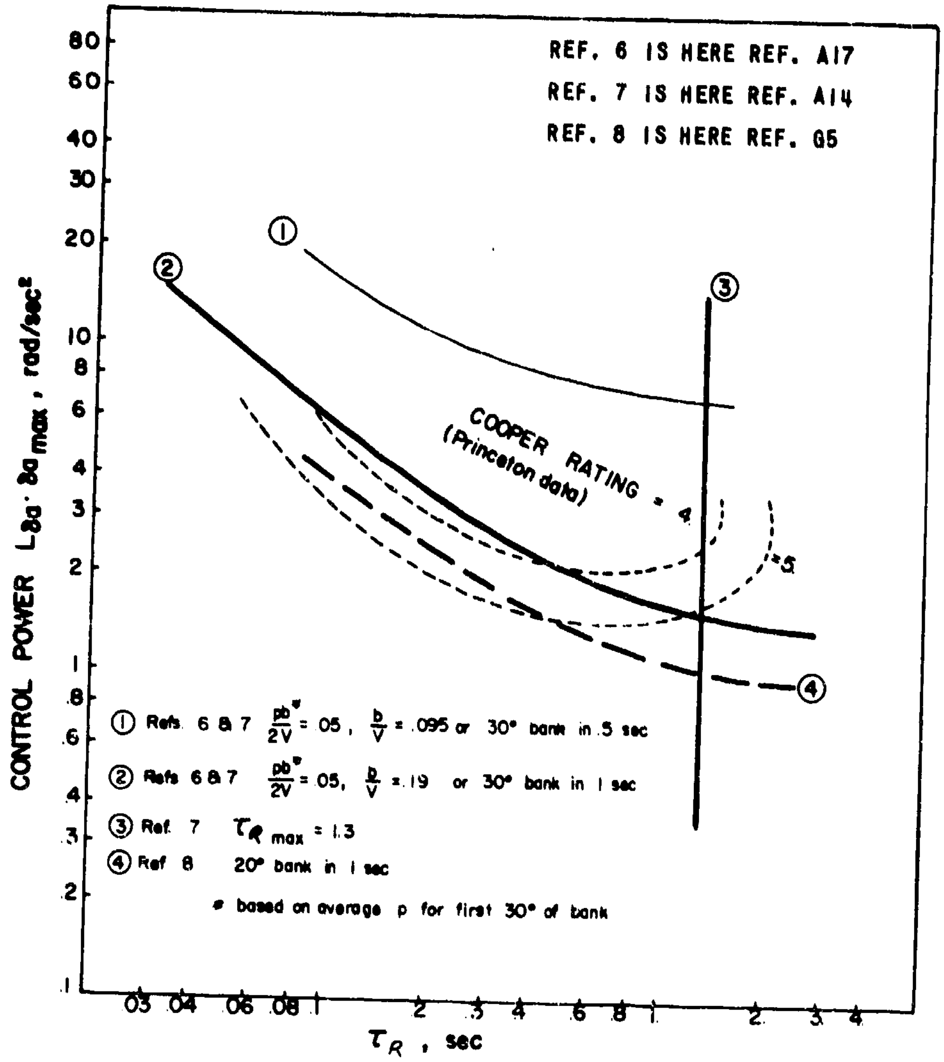


Figure 16 (3.3.4)

COMPARISON OF PRINCETON DATA WITH LATERAL CONTROL DATA OF REFERENCES A14, A17 AND G5. (FROM REFEPENCE G10)

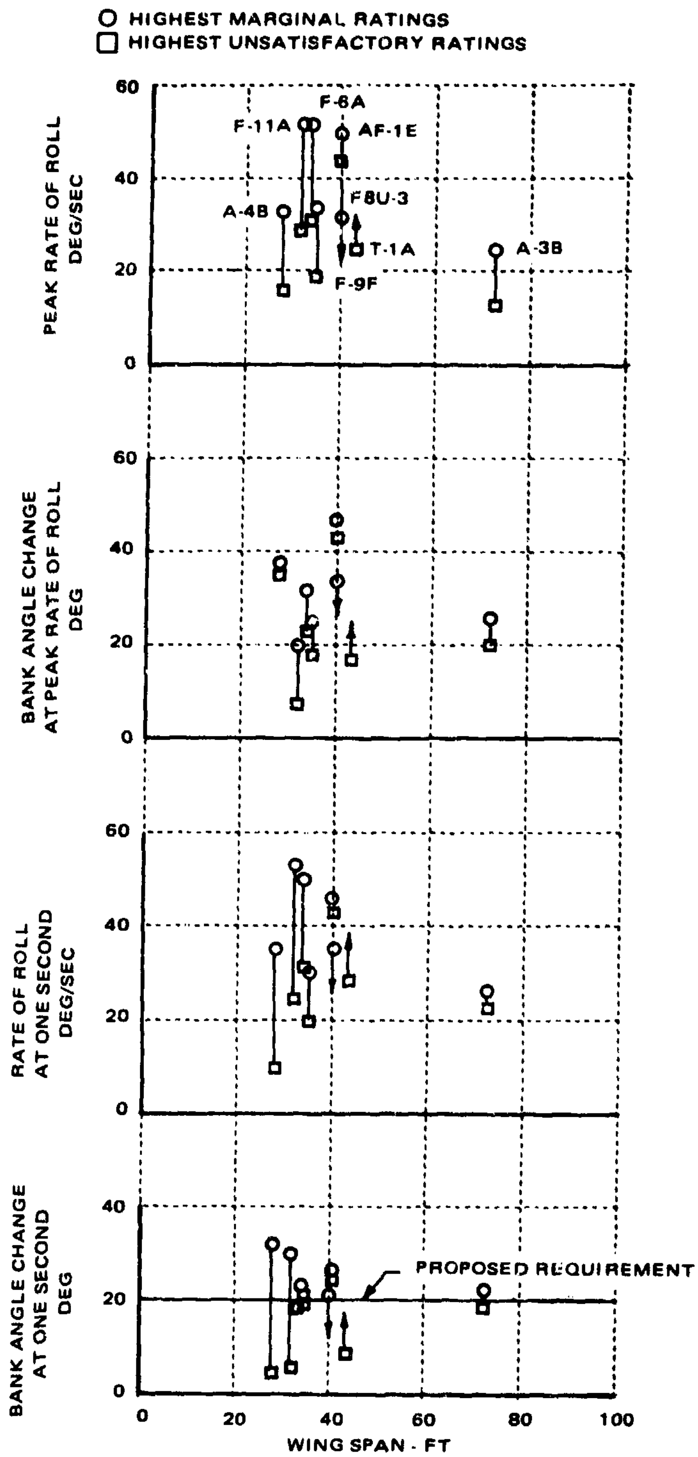


Figure 17 (3.3.4)

RANGE OF MARGINAL RATINGS AT APPROACH AIRSPEED (CARRIER-BASED AIRPLANES). (FROM REFERENCE G5)

AIRPLANE	SPAN (ft)	DESIGN WEIGHT (1000 lb)	COMBAT OR CRUISE p_o	ϕ_1	ϕ_2	t_{30^o}	LANDING APPROACH p_o	ϕ_1	t_{30^o}	EVALUATION COMMENTS
F-86H	39	23.8	85	45			70			Unsatisfactory for combat, OK for approach
F-100C	39	30.8	150+	40+			No data			Inadequate — poor control response, large aileron yaw for approach conditions
F-101A(1)	40	48.7	Maximum roll restricted				50	16	1.3	Satisfactory for approach
F-101B	40	45.1	180+	80+			80		1	Satisfactory
F-102A	38	28	160 – 220	80 – 130			85	35		Good roll capability
F-102A	38	28		100+			p_{30^o} = 34		1.1	Adequate
F-102A(2)	38	28	110 – 180	70+			70		0.8	10^o aileron limit acceptable for the flight conditions investigated
XF-104	22	18.5	250+	100+			80(3)	30(3)		Very satisfactory
F-104B(4)	25	20.9	100+	40+			80		1.3	Unsatisfactory for combat; satisfactory for approach
F-105B	35	49	150	40 – 60			60	24		Adequate for fighter-bomber mission
F-106A	38	35	150	60			No data			Satisfactory
YB-60	206	410	8				No data			Very poor in general; marginal for landing approach
RB-52C	185	371,440(5)	25	5	25 – 17(5)	2.3 – 2.8(5)	11(5)	1(5)	4.1(5)	Satisfactory
C-130B	133	135	20 – 30	10 – 13	30^o	2.0 – 1.9	12	4	2.9	Satisfactory in general
C-133A	180	255	22 – 24	6 – 7	27^o	2.2	14	3	3.9	Adequate
C-133B	180	286	Same as C-133A			2.2	12	4	3.5	Control during approach considered minimum acceptable
KC-135A	131	300	33	9	57^o	1.8	18	6	2.5	Very good for cruise; good for approach
T-37A	34	6.2	125+	75+			40+	12+		Ample roll capability in general
T-38A	25	11.5	150 – 190	70 – 140			75	45	0.75	Excellent
HA-246 (T-39)	42	11.6	90	65			25			Satisfactory in general
HAC-119A	57	45	70	44			34			Excellent
B-66B	72	96	60 – 70	25 – 30			47	16	1.7	Excellent throughout entire speed range
SA-16B	97	34	30				20			Satisfactory in general
YAC-1DH (Caribou)	96	26	No data				8			Unsatisfactory

(1) With external stores
(2) With enlarged fin and aileron limited to 10^o
(3) Takeoff configuration
(4) With 20-inch tip tanks
(5) External tanks full

Figure 18 (3.3.4)
FULL AILERON ROLL PERFORMANCE OF USAF AIRPLANES (FROM REFERENCE F8)

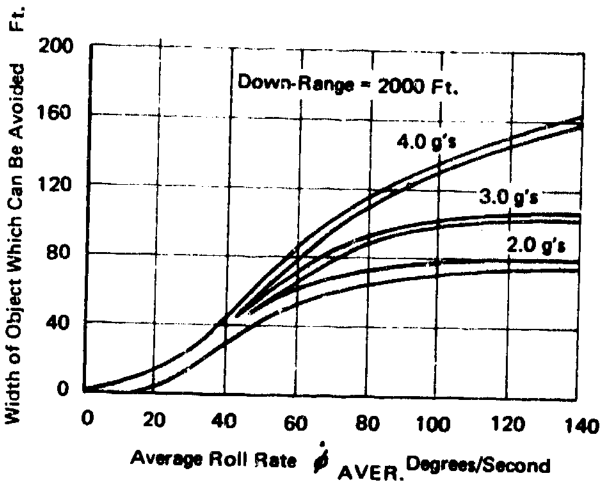


Figure 19 (3.3.4)

EFFECT OF AVERAGE ROLL RATE ON TERRAIN AVOIDANCE CAPABILITIES (FROM REFERENCE F27)

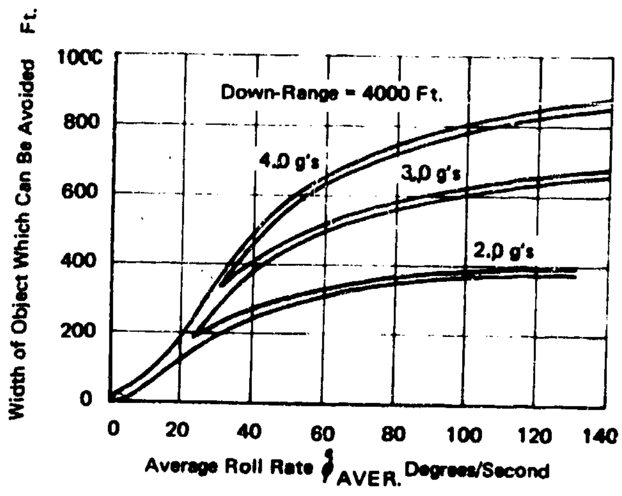


Figure 20 (3.3.4)

EFFECT OF AVERAGE ROLL RATE ON TERRAIN AVOIDANCE CAPABILITIES (FROM REFERENCE F27)

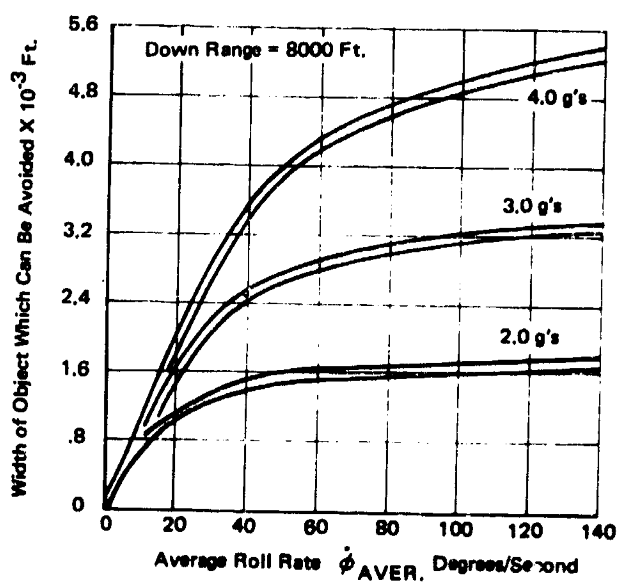


Figure 21 (3.3.4)

EFFECT OF AVERAGE ROLL RATE ON TERRAIN AVOIDANCE CAPABILITIES (FROM REFERENCE F27)

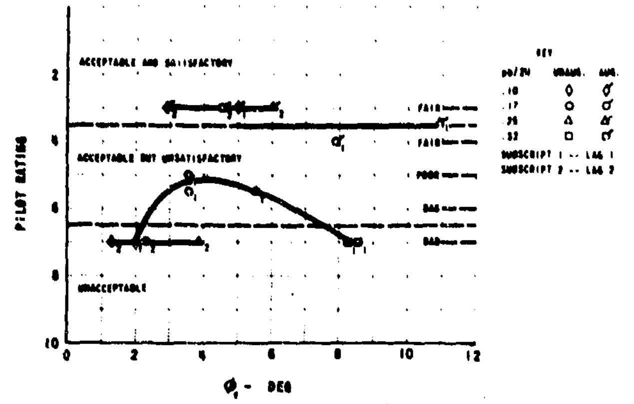


Figure 22 (3.3.4)

LATERAL-DIRECTIONAL EVALUATION -- PILOT A, EFFECT OF RATINGS INCLUDE LONGITUDINAL CHARACTERISTICS (FROM REFERENCE C2)

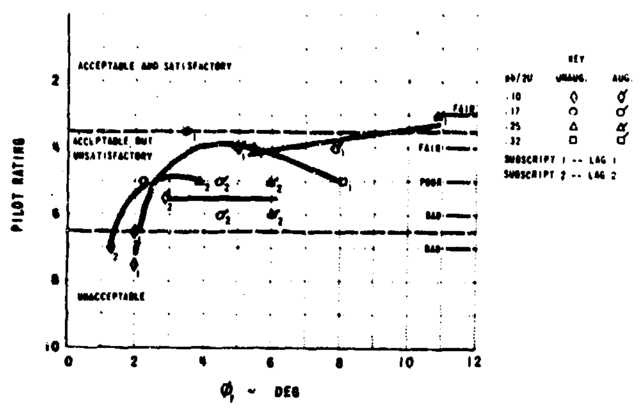


Figure 23 (3.3.4)

LATERAL-DIRECTIONAL EVALUATION -- PILOT B, EFFECT OF RATINGS INCLUDE LONGITUDINAL CHARACTERISTICS (FROM REFERENCE C2)

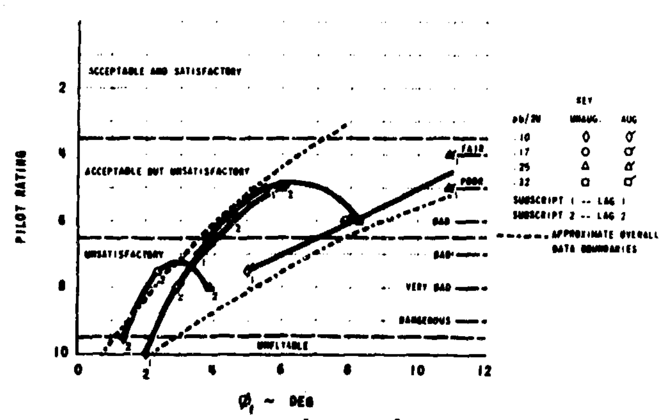


Figure 24 (3.3.4)

LATERAL-DIRECTIONAL EVALUATION -- PILOT C, EFFECT OF RATINGS INCLUDE LONGITUDINAL CHARACTERISTICS (FROM REFERENCE C2)

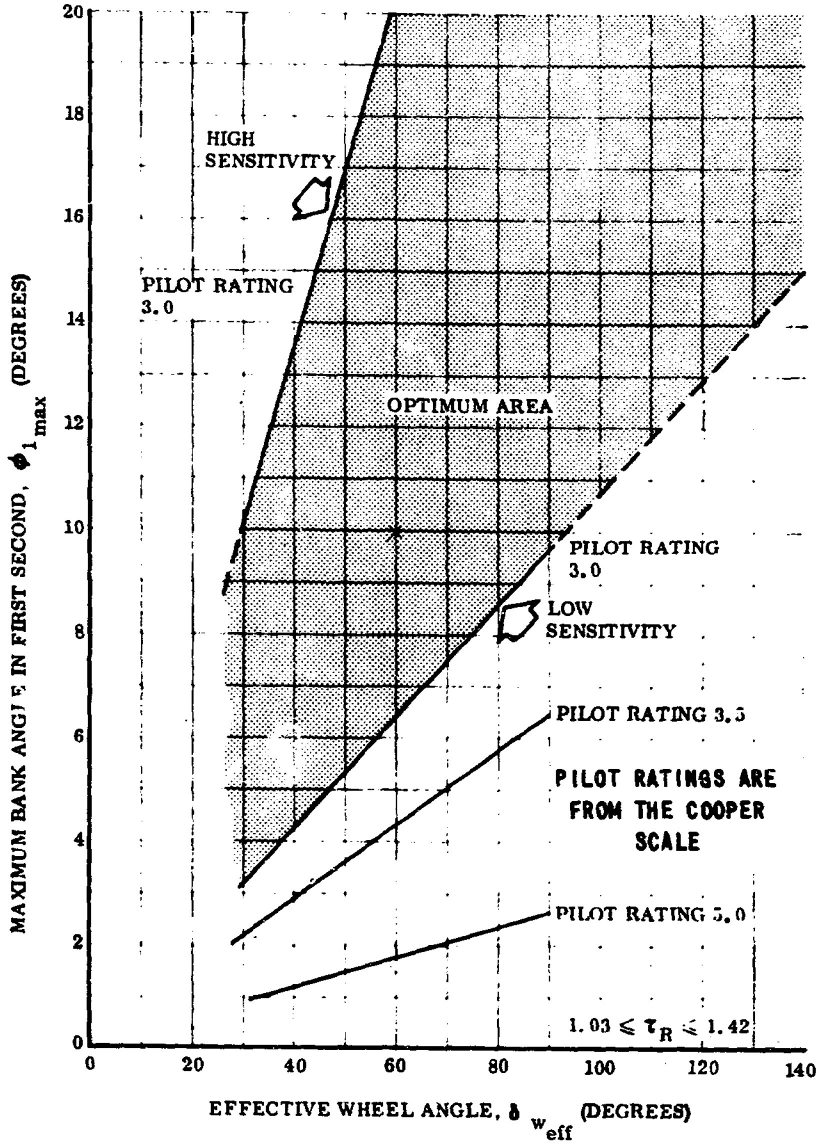


Figure 25 (3.3.4)
PILOT RATING BOUNDARIES FOR ROLL RESPONSE SENSITIVITY (FROM REFERENCE C7)

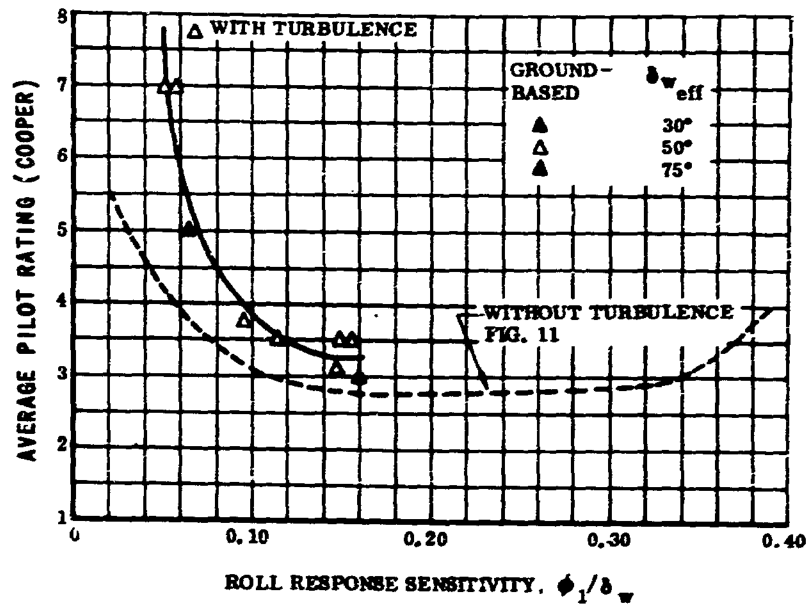


Figure 26 (3.3.4)
VARIATION OF PILOT RATING WITH ROLL RESPONSE SENSITIVITY IN TURBULENCE (FROM REFERENCE C7)

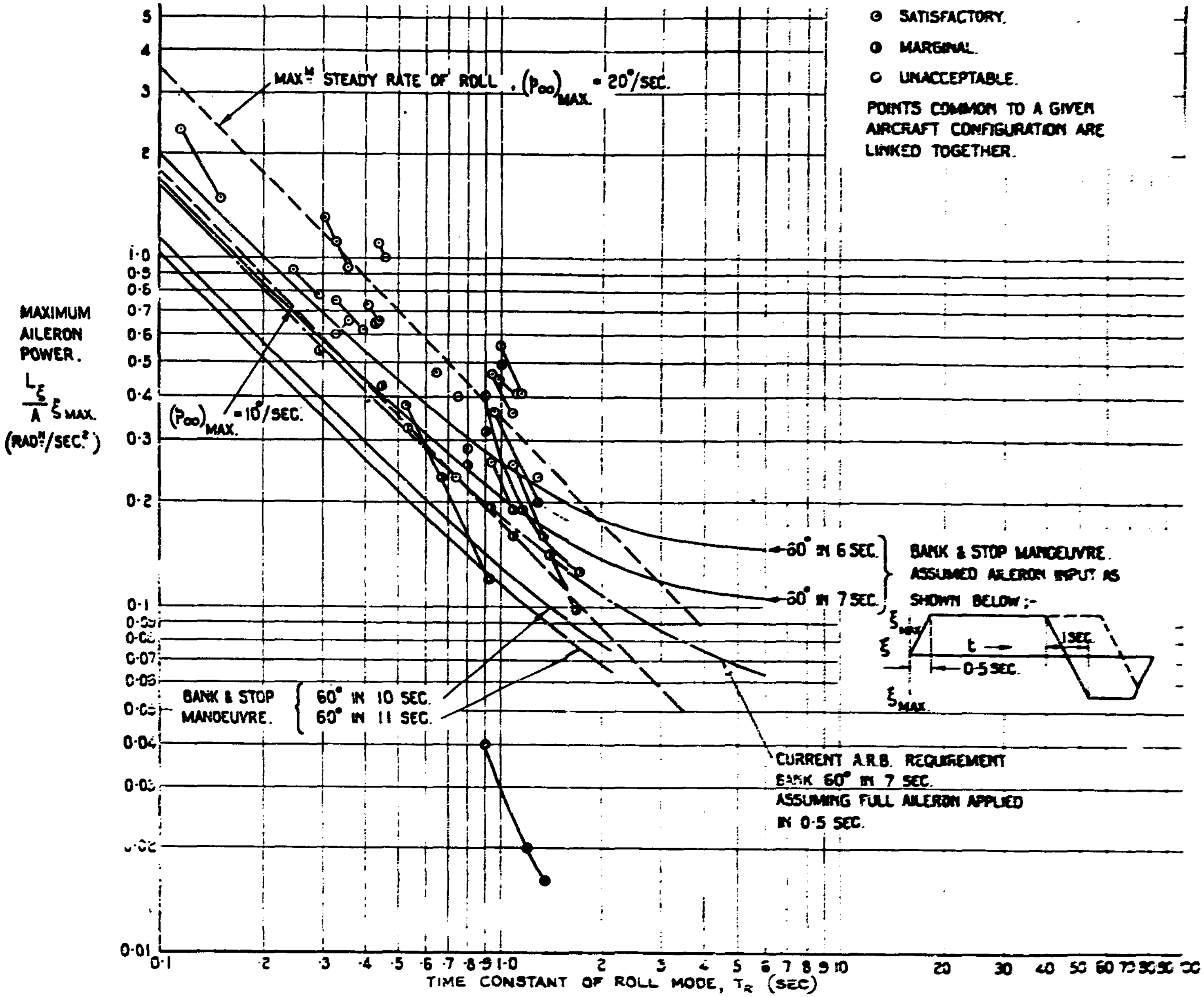


Figure 27 (3.3.4)

ROLL RESPONSE, 'ARGE AIRCRAFT IN APPPOACH (FROM REFERENCE B39)

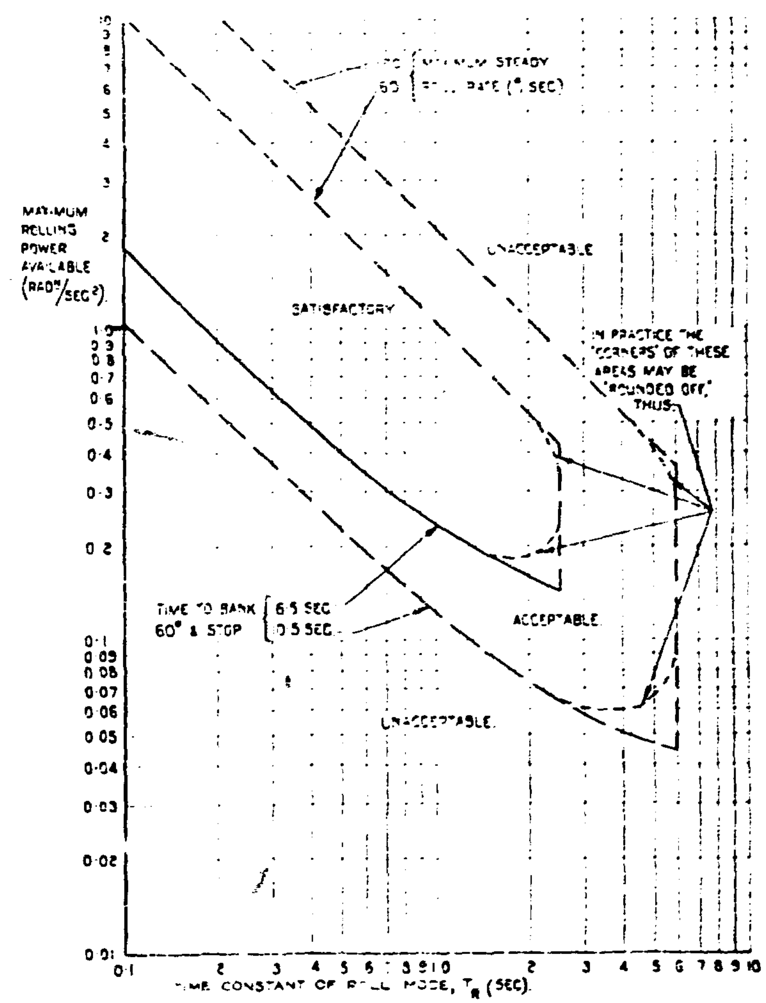


Figure 28 (3.3.4)
SUGGESTED ROLL-RESPONSE BOUNDARIES FOR LARGE AIRCRAFT (APPROACH CONDITIONS) (FROM REFERENCE B39)

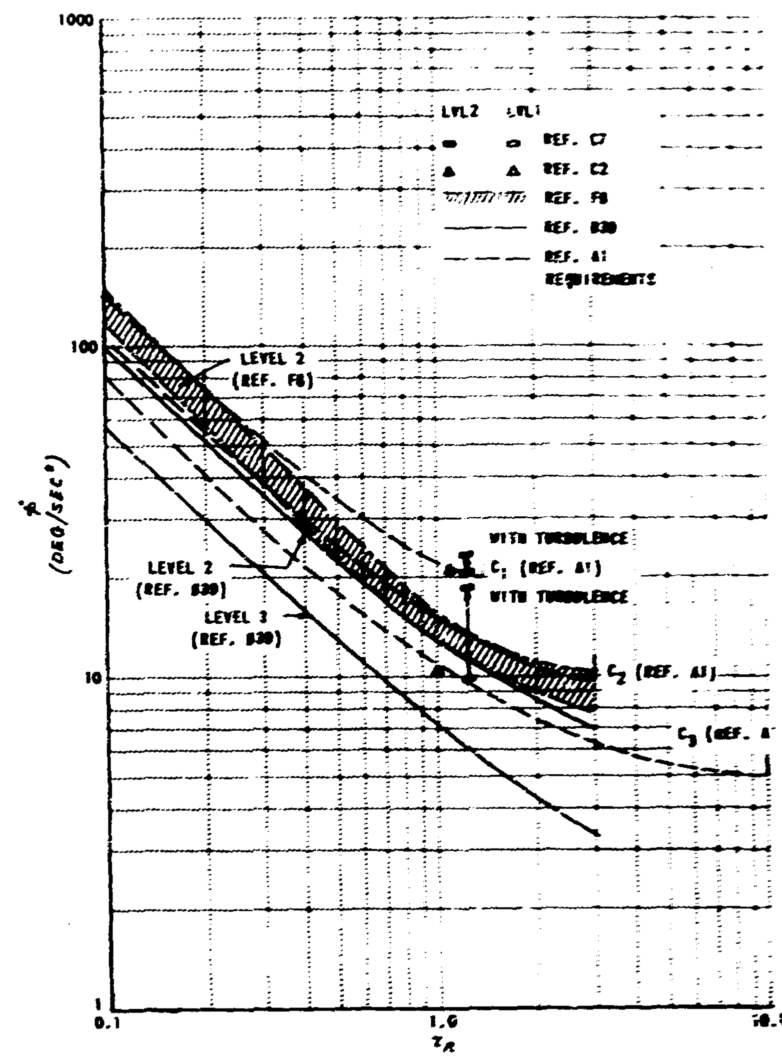


Figure 29 (3.3.4)
DATA ON CLASS III AIRPLANES FOR FLIGHT PHASE CATEGORY C

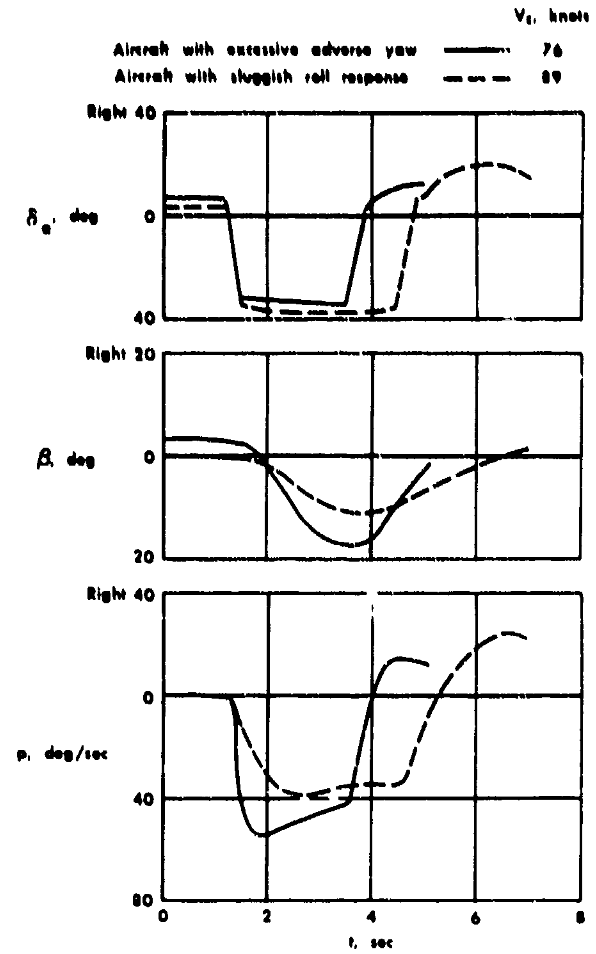


Figure 30 (3.3.4)
TIME HISTORIES OF ABRUPT RUDDER-FIXED AILERON ROLLS.
APPROACH CONFIGURATION. (FROM REFERENCE B96)

3.3.4.2 AILERON CONTROL FORCES

REQUIREMENT

3.3.4.2 Aileron control forces. The stick or wheel force required to obtain the rolling performance specified in 3.3.4 and 3.3.4.1 shall be neither greater than the maximum in table X nor less than the breakout force plus:

a. Level 1 -- one-fourth the values in table X

b. Level 2 -- one-eighth the values in table X

c. Level 3 -- zero

TABLE X. Maximum Aileron Control Force

Level	Class	Flight Phase Category	Maximum Stick Force (lb)	Maximum Wheel Force (lb)
1	I, II-C, IV	A, B	20	40
		C	20	20
	II-L, III	A, B	25	50
		C	25	25
2	I, II-C, IV	A, B	30	60
		C	20	20
	II-L, III	A, B	30	60
		C	30	30
3	All	All	35	70

RELATED MIL-F-8785 PARAGRAPHS

3.4.16.3, 3.7.5

DISCUSSION

Maximum Forces

The maximum stick and wheel forces specified in 3.4.16.3 of MIL-F-8785, in terms of the airplane Classes defined in Reference A1, are as follows:

Class I: 25-lb stick force or 50-lb wheel force

Classes II and III: 25-lb stick force or 50-lb wheel force

Class IV: 20-lb stick force or 40-lb wheel force

Classes II-C and IV-C in landing configuration: 20-lb stick force or 20-lb wheel force.

The maximum stick and wheel forces specified in 3.7.5 of MIL-F-8785 for control on the alternate system are as follows:

All classes: 30-lb stick force or 60-lb wheel force, with the exception that in the landing configuration the maximum for Class II-C and IV is 20-lb stick force or 20-lb wheel force.

The maximum stick and wheel forces for Levels 1 and 2 of Reference A1 have been mapped directly from MIL-F-8785 into Table X with the following three exceptions:

(1) In MIL-F-8785, land-based Class IV airplanes with wheel controllers in the landing configuration were allowed 20-lb force when operating on the alternate system and 40-lb force when operating normally; in Reference A1 they are restricted to 20-lb force for both Level 1 and Level 2 operation.

(2) Since it is anticipated that, in general, Class I airplanes will be utilized in missions where they will be required to maneuver considerably in roll, the maximum forces have been made the same as those specified for Class II-C and IV airplanes, rather than the same for Class II-L and III as is done in MIL-F-8785.

(3) To permit one-handed operation in the landing approach, maximum forces for all classes for Flight Phase Category C are 30 lb or less.

The recommended maximum stick force for Level 3 is the same as that required for spin recovery, and is consistent with physiological data presented in Reference B20. The maximum wheel force retains the ratio of 2:1 between wheels and sticks that is used throughout MIL-F-8785.

Minimum Forces (Roll response sensitivity)*

The minimum stick and wheel force requirements and the roll response requirements of 3.3.4.1.4, which are really roll response sensitivity requirements, replace the requirement in paragraph 3.4.16.3 of MIL-F-8785 which states: "At 0.8 V_H , the peak lateral control force required to obtain the rolling performance specified in Table VI shall not be less than half the above values." The Reference A1 minimum force requirements are far more lenient (and it is believed far more realistic) than the MIL-F-8785 requirements.

In specifying sensitivity requirements, a decision had to be made whether to specify it in terms of aileron control force or displacement. Reference F8 studied this question but found no data that clearly resolved the problem. Although undoubtedly both force and displacement cues are important to the pilot, recent experimental data indicate that sensitivity in terms of force (at least for stick controllers) is of more significance than sensitivity in terms of displacement.

To determine minimum stick forces, the results of five experimental programs employing stick controlled aircraft or simulators were examined: Reference F1, Reference F30, Reference G10, Reference F5, and Reference F22.

Program of Reference F1 (Flight Phase Category A)

In this in-flight lateral-directional flying qualities program for a typical fighter mission, the pilots were allowed to select the sensitivity of the aileron control. The resulting "optimum" sensitivities are presented in Figure 1. The data have been replotted in terms of rolling acceleration per inch of stick deflection, $L'_{\delta_{AS}}$, in Figure 2; and in terms of rolling acceleration per pound of stick force, $L'_{F_{AS}}$, in Figure 3. The spring rate, F_{AS} / δ_{AS} = 3.81 lb/in.

Program of Reference F30 (Flight Phase Category A)

In this program which utilized a rolling simulator and several fighter aircraft, a parametric variation of $L_{\delta_a} \delta_{a_{max}}$ and τ_R was made to determine lateral control requirements for fighter aircraft performing fighter missions. As such, the results of this program should be directly comparable to the results of the program of Reference F1.

*NOTE: In this text "roll sensitivity" refers to L'_{δ} whereas "roll response sensitivity" refers to ϕ_t / δ .

Although the results of this program are presented in Figure 4 in terms of $L_{\delta_a}\delta_{a_{max}}$ vs τ_R, it is believed that in the region of high $L_{\delta_a}\delta_{a_{max}}$, the pilot rating boundaries are determined more from sensitivity considerations than from maximum control power considerations. To quote from Reference F30: "His control difficulties in the overall region of high $L_{\delta_a}\delta_{a_{max}}$ values were further compounded because of the extreme stick sensitivity in this region wherein small stick deflections commanded large roll rates and roll accelerations."

That the pilot was concerned with control sensitivity rather than control power is further confirmed by the study (supported by simulation) of Reference F27. By noting whether or not the pilot either: (a) never hit the control stops, (b) sometimes hit the control stops, or (c) always hit the control stops during bank angle change maneuvers, it was possible to determine the region on the $L_{\delta_a}\delta_{a_{max}}$ vs τ_R plane where the aircraft was control-power limited. From Figures 5 through 7, where the results of this program are superimposed on the results of Reference F30, it can be seen that the configurations were control-power limited for low values of $L_{\delta_a}\delta_{a_{max}}$ and τ_R, whereas for large $L_{\delta_a}\delta_{a_{max}}$ and τ_R, other factors degraded the flying qualities.

Plotting the data of Reference F30 for optimum configurations in terms of sensitivity parameters rather than in terms of maximum control power, the data of Reference F30 can be compared directly with the data of Reference F1. The Reference F30 data are plotted in terms of roll acceleration per inch in Figure 2 ($\delta_{a_{max}} = \pm 5$ in.) and in terms of roll acceleration per pound in Figure 3. Optimum values and the values corresponding to Level 1 and Level 2 flying qualities are also shown. The spring rate, F_{AS}/δ_{AS}, is 2 lb/in.

Since the stick force gradient in the program of Reference F1 (F_{AS}/δ_{AS} = 3.81 lb/in.) was approximately twice that in Reference F30, and since the type of aircraft and type of control tasks were very similar (Flight Phase Category A), a direct comparison between the data of the two programs can be made to determine the relative merits of force and displacement sensitivity. In comparing the optimum-sensitivity data of Reference F1 for τ_R = 0.4 seconds with that of Reference F30, greater weight should be given to the high-optimum-sensitivity points of the Reference F1 data than to the low-sensitivity points, since the low-sensitivity points are associated with configurations of poor pilot rating. For the configurations with poor flying qualities, the gain was reduced to minimize the effect of the undesirable characteristics (for example, aileron yaw).

From consideration of the optimum-sensitivity data of References F1 and F30, as presented in Figures 2 and 3, it can be seen that the correlation between the data in terms of roll acceleration per force is better than in terms of roll acceleration per displacement. Furthermore, it can be seen that the data points of constant pilot rating lie approximately along lines of constant ϕ_1 (bank angle in one second). This suggests that, at least for Class IV aircraft performing fighter missions, roll response sensitivity can be best expressed in terms of ϕ_1 / F_{AS} (bank angle in 1 second per pound). It can further be seen that the data points from Reference F30 for maximum satisfactory roll response sensitivity lie along a curve of $\phi_1 / F_{AS} \approx$ 15 degrees/lb; for acceptable flying qualities, $\phi_1 / F_{AS} \approx$ 25 degrees/lb. Both sets of data indicate that for optimum roll response sensitivity ϕ_1 / F_{AS} should be about 10 degrees/lb. A possible exception is indicated by the low-τ_R data of Reference F1, where somewhat lower optimum roll response sensitivities were selected by the pilots.

These data were used directly to specify the roll response requirements of 3.3.4.1.4 that apply to stick-controlled Class IV airplanes performing Category A Flight Phases.

Program of Reference G10 (Flight Phase Category C)

In order now to compare the fighter-airplane data for up-and-away flight with data for the landing approach, consider the in-flight data of Reference G10 shown in Figures 8 through 13. The values of optimum, Level 1 and Level 2 control sensitivities for several values of τ_R were replotted onto Figure 14 from Figure 8. Using a force gradient of 4.4 lb/in., the data were then plotted on Figure 15 in terms of force.

From comparison of Figures 3 and 15 it can be seen that the optimum roll response sensitivity, maximum satisfactory roll response sensitivity, and maximum acceptable roll response sensitivity, in terms of rolling acceleration per force for the landing approach, are about half that for respective levels of flying qualities for Flight Phase Category A. These data were used directly to specify the roll response requirements of 3.3.4.1.4 that apply to stick-controlled Class IV airplanes performing Category C Flight Phases.

Program of Reference F5 (Flight Phase Category B)

Reference F5 reports on an in-flight investigation of the flying qualities of re-entry vehicles performing maneuvers characteristic of Flight Phase Category B. Although aileron yaw was the primary variable, aileron sensitivity was also varied and pilot comments pertaining directly to control sensitivity were obtained. From examination of the pilot comment data for Configurations B-2, B-2A, B-3A, and B-4A on Figure 16, it can be seen that for:

Configuration B-2 - the sensitivity is too high

Configuration B-2A - no comments are made regarding sensitivity, so presumably it is satisfactory

Configuration B-3A - specific comments on the desirable sensitivity

Configuration B-4A - requests greater control sensitivity

From the characteristics of the configuration given on pages 3-61, 3-62, and 3-63 of Reference F5, the sensitivity characteristics for these four configurations are plotted on Figure 18 in terms of roll acceleration per inch and on Figure 19 in terms of roll acceleration per pound.

Program of Reference F22 (Flight Phase Category B)

Reference F22 reports on another in-flight investigation of the flying qualities of re-entry vehicles, so is also applicable to Flight Phase Category B. From the data presented in Figure 17, it can be seen that the flying qualities are satisfactory over a wide range of sensitivities. By plotting the data on Figures 18 and 19 and comparing with the data from Reference F5, it can be seen that optimum roll response sensitivity occurs around $\phi_1 / F_{AS} = 5$ degrees/lb, which is similar to that for Flight Phase Category C but somewhat lower than for Flight Phase Category A. On the other hand, it can be seen that the maximum satisfactory value of roll response sensitivity more nearly corresponds to that for Flight Phase Category A at approximately $\phi_1 / F_{AS} \approx 18$ degrees/lb. The spring rate for this program was $F_{AS} / \delta_{AS} = 2$ lb/in.

Minimum Stick Forces

In order now to convert the roll response sensitivity requirements from the parameter ϕ_1 / F_{AS} to the parameter used in MIL-F-8785 (minimum control force required to meet the roll performance requirements), it is necessary to examine the roll performance requirements in paragraph 3.3.4 of Reference A1. From Table IX, for fighter aircraft in Flight Phase Category A, $\phi_t = 90$ degrees in 1.3 seconds for Level 1. From the discussion of paragraph 3.3.4, $\phi_t = 90$ degrees in 1.3 seconds is very similar to $\phi_1 = 60$ degrees. From the previous discussion, it was determined that maximum satisfactory roll response sensitivity for fighter aircraft in Flight Phase Category A was $\phi_1 / F_{AS} \approx 15$ degrees/lb. Thus, the minimum stick force to meet the roll performance requirements is

$$\frac{60 \text{ degrees in 1 sec}}{15 \text{ degrees in 1 sec/lb}} = 4 \text{ lb.}$$

Considering now the landing approach requirements from Table IX of Reference A1, for fighter aircraft in Flight Phase Category C, $\phi_t = 30$ degrees in 1 sec for Level 1. From the previous discussion it was determined that maximum satisfactory roll response sensitivity for fighter aircraft in Flight Phase Category C was about half that for Flight Phase Category A, that is, $\phi_1 / F_{AS} \approx 7.5$ degrees/lb. Thus, the minimum stick force to meet the roll performance requirements is

$$\frac{30 \text{ degrees in 1 sec}}{7.5 \text{ degrees in 1 sec/lb}} = 4 \text{ lb.}$$

These data therefore indicate that, at least under some conditions, the minimum forces can be much less than those specified in MIL-F-8785. But these maximum sensitivities would result in somewhat higher minimum forces over much of the Operational Flight Envelope if, as would be expected, roll performance over much of the envelope is better than required. Thus, since 4 pounds represents the lowest satisfactory minimum stick force, a minimum force value of one quarter the maximum values specified in Table X of Reference A1 have been specified for Flight Phase Categories A and C, which results in minimum forces between 5 and 6 pounds. In addition to the requirements of 3.3.4.2, direct roll response sensitivity requirements in terms of the parameter ϕ_1 / F_{AS} have been specified in 3.3.4.1.3 for stick-controlled Class IV airplanes.

Examination of the Flight Phase Category B sensitivity data and roll performance requirements indicates that the roll response sensitivity requirements for Flight Phase Category B could be somewhat more lenient (that is, lower minimum F_{AS}) than those for the other Flight Phase Categories. However, because of the interaction effects of breakout force with sensitivity, and in order to minimize complexity, the Flight Phase Category B roll response sensitivity requirement has been made the same as that for the other Flight Phase Categories.

Available data show that maximum acceptable roll response sensitivity is almost twice as great as maximum satisfactory roll response sensitivity. Since maximum acceptable stick forces specified in Table X are slightly greater than maximum satisfactory stick forces, minimum stick forces for Level 2 have been made one-eighth the maximum specified Level 2 stick forces.

Actual minimum satisfactory or acceptable stick forces will depend strongly upon the breakout force and other control system characteristics. For example, Reference B39 states, "With regard to breakout force, it is known that an aircraft with a breakout force of roughly 50% of the force to apply full control can be flown, but is generally unpleasant to handle and lacks precision of control." Thus, maximum satisfactory and acceptable sensitivity will be different for each airplane and must be determined on an individual basis.

It should be noted that by relating sensitivity requirements to roll performance requirements, the conflict of whether to relate sensitivity to roll rate or roll acceleration is avoided (see discussion in Section II of Reference F8). At low τ_R, where ailerons essentially command roll rate, the curves approximate curves of constant roll rate; whereas for large τ_R , where ailerons are more accelerating-ordering, the t_ϕ curves more closely approximate lines of constant initial roll acceleration. Thus the Reference A1 roll response sensitivity requirements take into account the changing nature of the roll response as a function of τ_R.

Minimum Wheel Forces

The proposed minimum wheel force requirements are based on discussions and data in References B39 and C7.

From a review of the literature and from pilot comments pertaining to large aircraft in the landing approach, Reference B39 suggests a sensitivity limit for satisfactory flying qualities of 1.5 degrees per sec/lb. Since the Reference A1 roll performance requirement for Class III aircraft in the landing approach is approximately 20 deg/sec in terms of steady roll rate, the force required to meet this roll performance at a sensitivity of 1.5 degree per sec/lb is

$$\frac{20}{1.5} \approx 13 \text{ pounds}$$

Reference C7 found, from ground-based simulation results, that maximum satisfactory roll response sensitivity occurred at $\phi_1/\delta_{AW} = 0.37$ deg/deg of wheel (Figure 20). The spring gradient was 0.28 lb/deg. Thus, for an aircraft with these characteristics, the roll response sensitivity would convert to

$$\frac{p_{SS}}{F_{AW}} = \frac{1.24}{.28} = 4.4\text{(deg/sec)/lb.}$$

At this sensitivity, the force required to attain a steady roll rate of 20 deg/sec is

$$\frac{20}{4.4} \approx 5 \text{ pounds.}$$

One aircraft manufacturer has found minimum wheel forces as low as 8 pounds to be satisfactory.

These data thus indicate quite a spread in minimum wheel forces. This suggests that for wheel-controllers, roll response per wheel deflection may be a better parameter than roll response per wheel force. Since there were insufficient data to resolve the question, however, the minimum wheel force requirements were specified in the same manner as for center sticks. Further, since the ratio of minimum to maximum forces for sticks was relatively consistent with the data on wheels, the same ratios were applied to wheels as were applied to sticks.

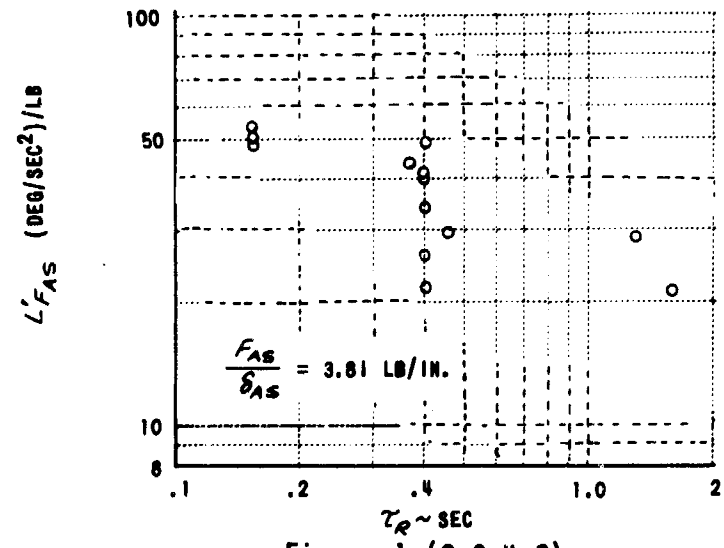


Figure 1 (3.3.4.2)

ROLL SENSITIVITY (FROM REFERENCE F1)

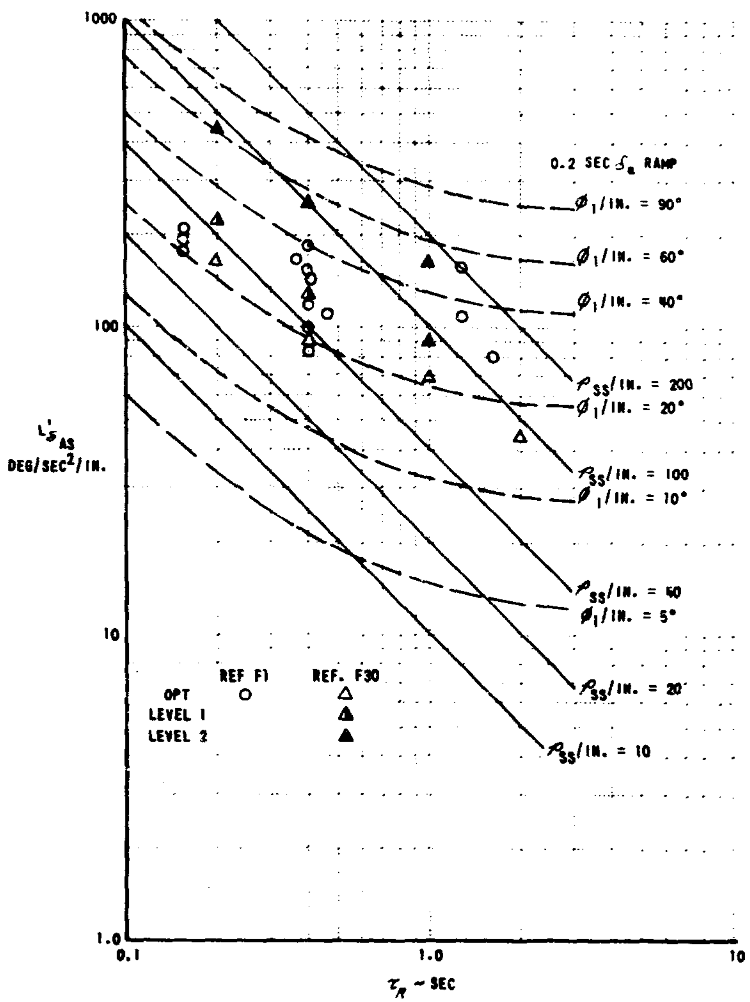


Figure 2 (3.3.4.2)
FLIGHT PHASE CATEGORY A - DEFLECTION SENSITIVITY

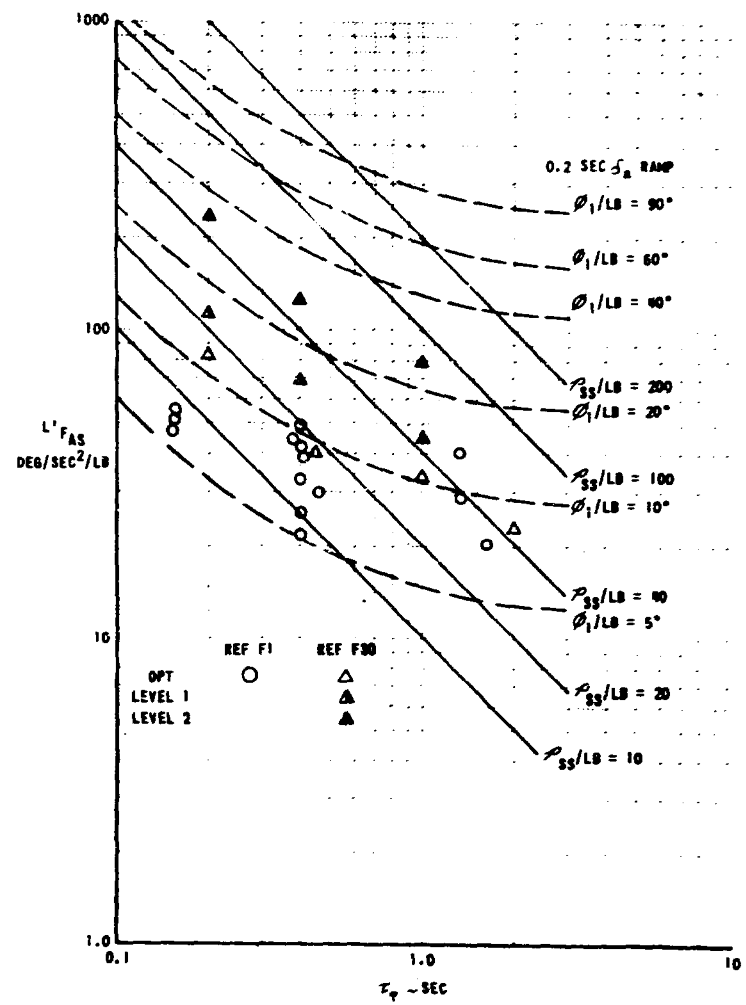


Figure 3 (3.3.4.2)
FLIGHT PHASE CATEGORY A - FORCE SENSITIVITY

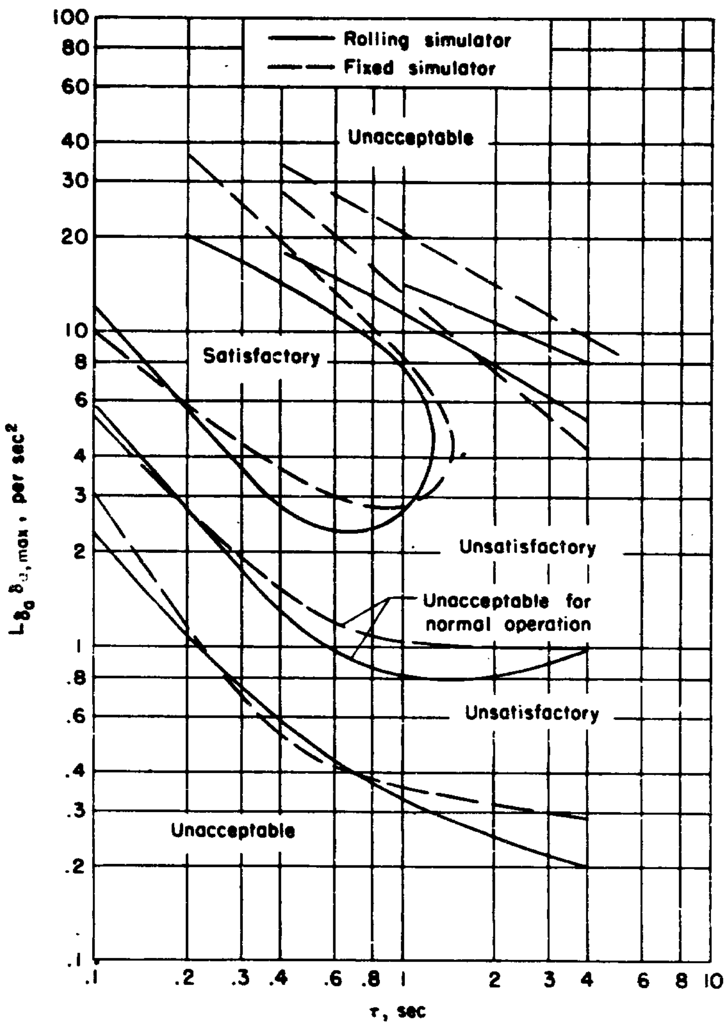


Figure 4 (3.3.4.2)
COMPARISON OF PILOT OPINION BOUNDARIES OBTAINED FROM THE FIXED AND MOVING FLIGHT SIMULATORS. (FROM REFERENCE F30)

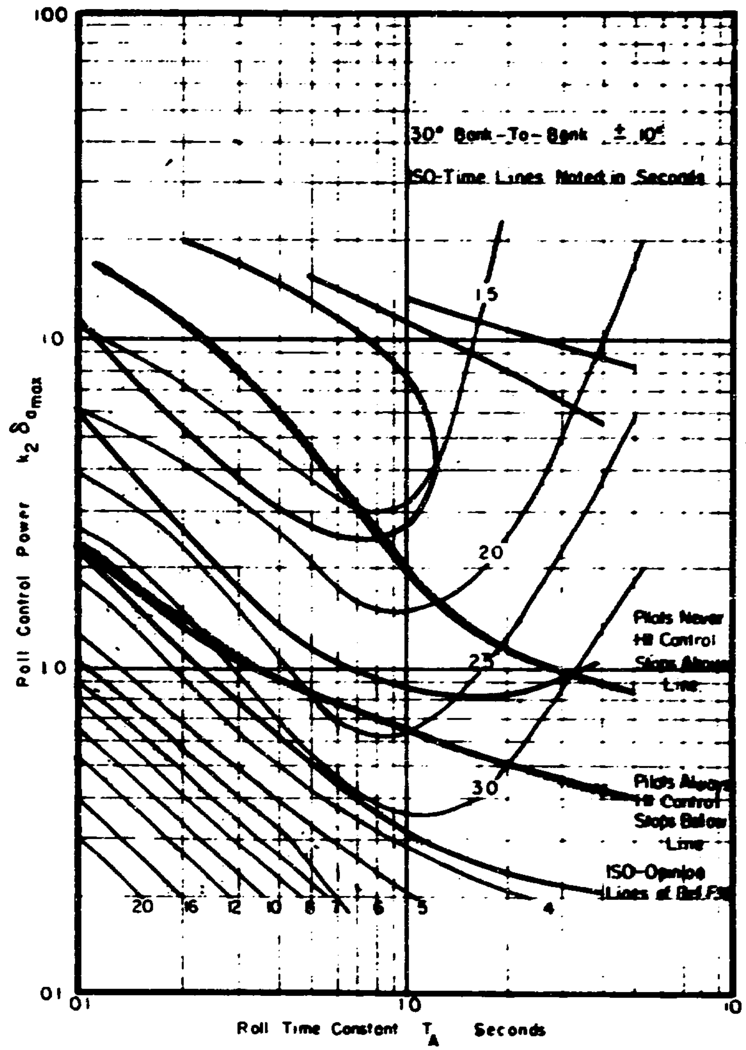


Figure 5 (3.3.4.2)
ISO-TIME CONTOURS FOR A BANK-AND-STOP MANEUVER (FROM REFERENCE F27)

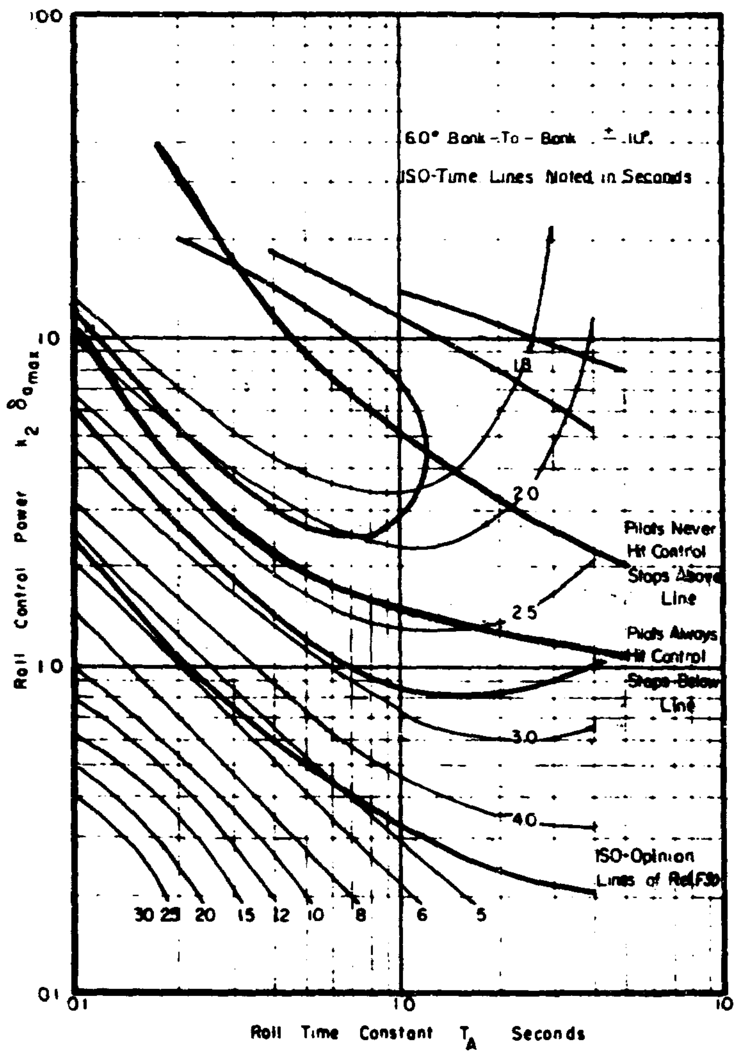


Figure 6 (3.3.4.2)

ISO-TIME CONTOURS FOR A BANK-AND-STOP MANEUVER

(FROM REFERENCE F27)

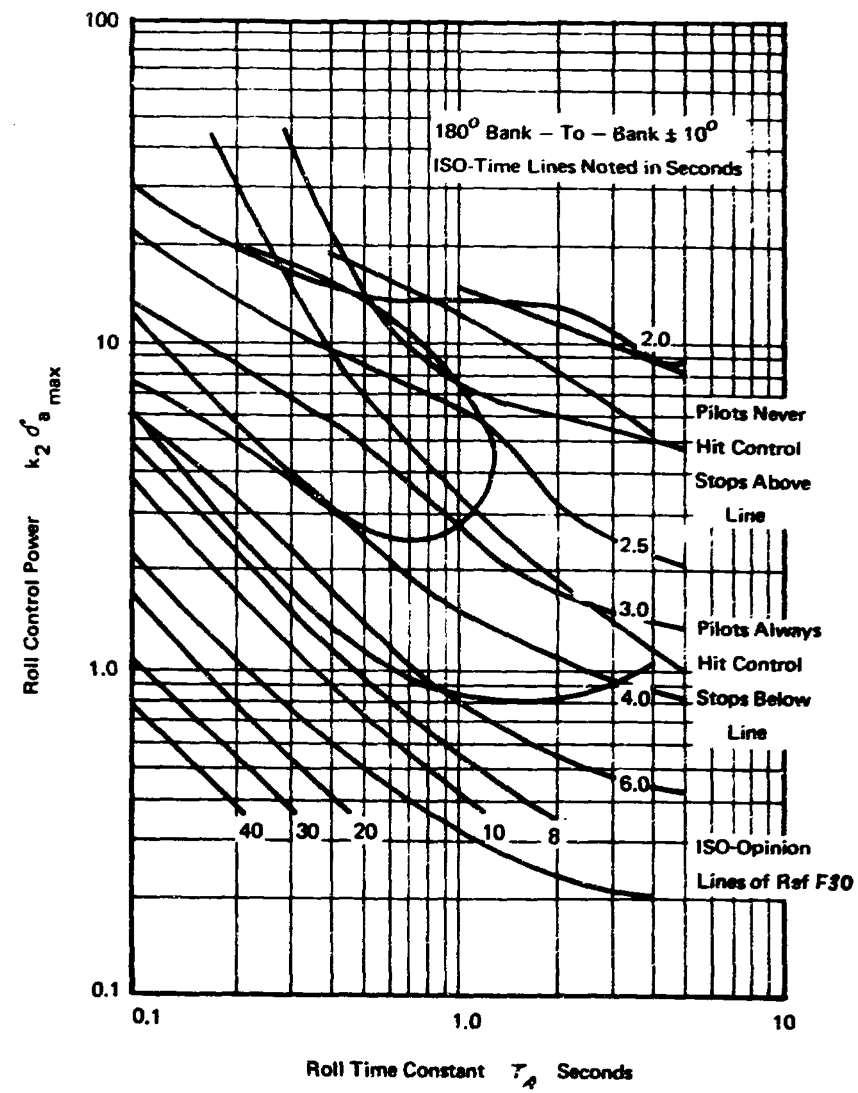


Figure 7 (3.3.4.2)

ISO-TIME CONTOURS FOR A BANK-AND-STOP MANEUVER

(FROM REFERENCE F27)

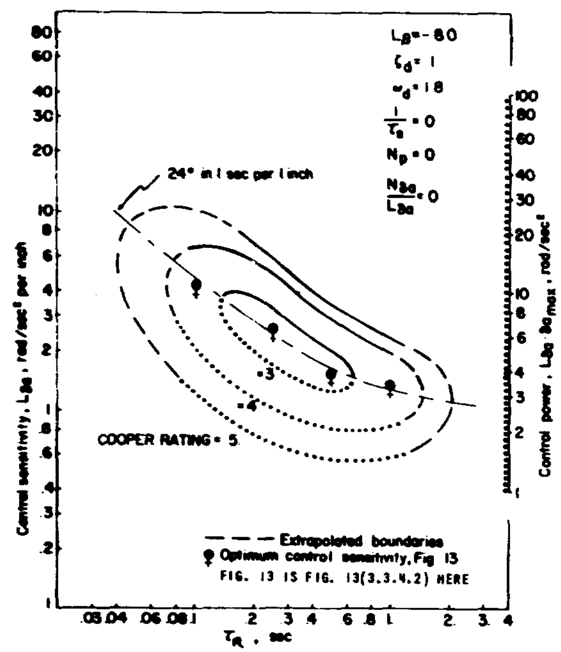


Figure 8 (3.3.4.2)
LATERAL CONTROL BOUNDARIES
(FROM REFERENCE G10)

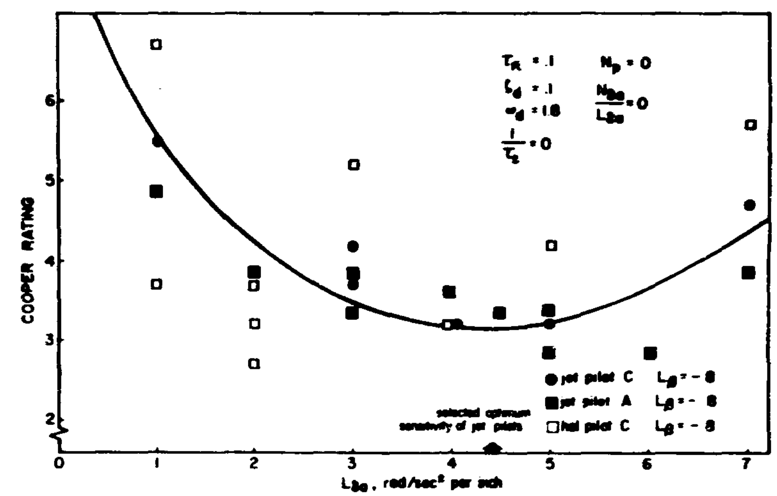


Figure 9 (3.3.4.2)
LATERAL CONTROL SENSITIVITY BOUNDARY, $\tau_R = 0.1$
(FROM REFERENCE G10)

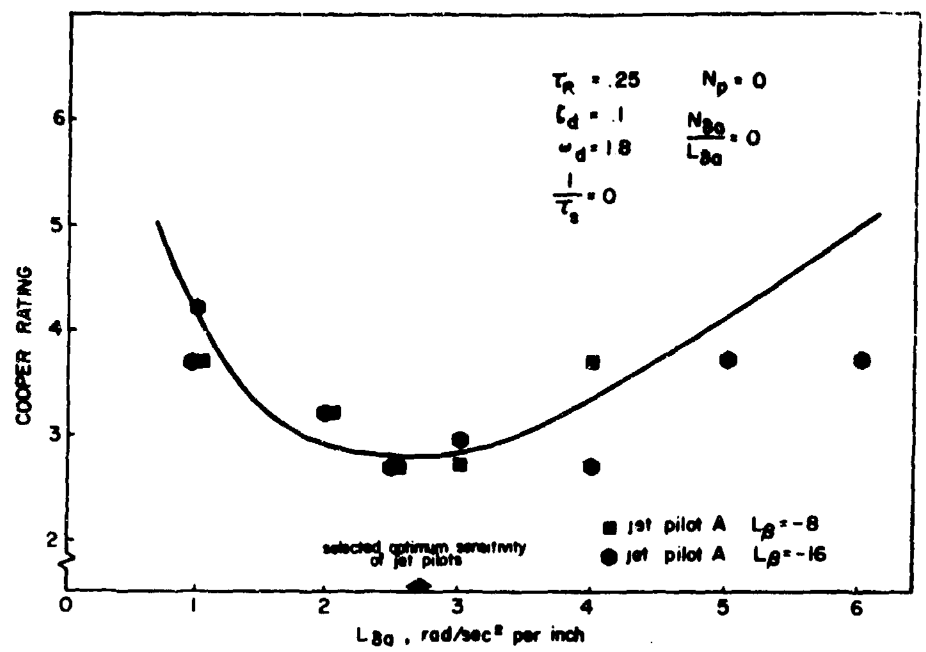


Figure 10 (3.3.4.2)

LATERAL CONTROL SENSITIVITY BOUNDARY, τ_R = 0.25
(FROM REFERENCE G10)

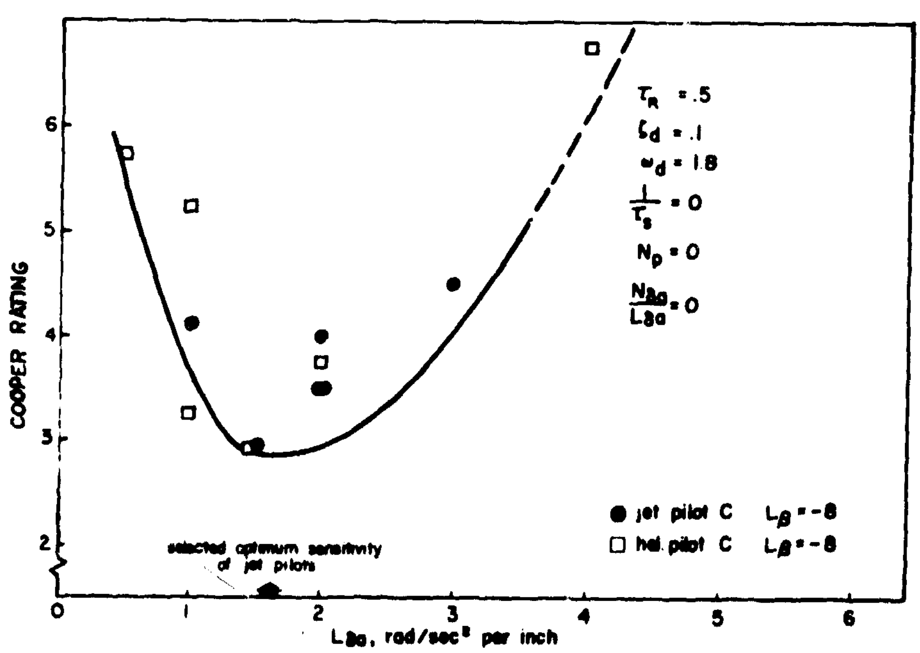


Figure 11 (3.3.4.2)

LATERAL CONTROL SENSITIVITY BOUNDARY, τ_R = 0.5
(FROM REFERENCE G10)

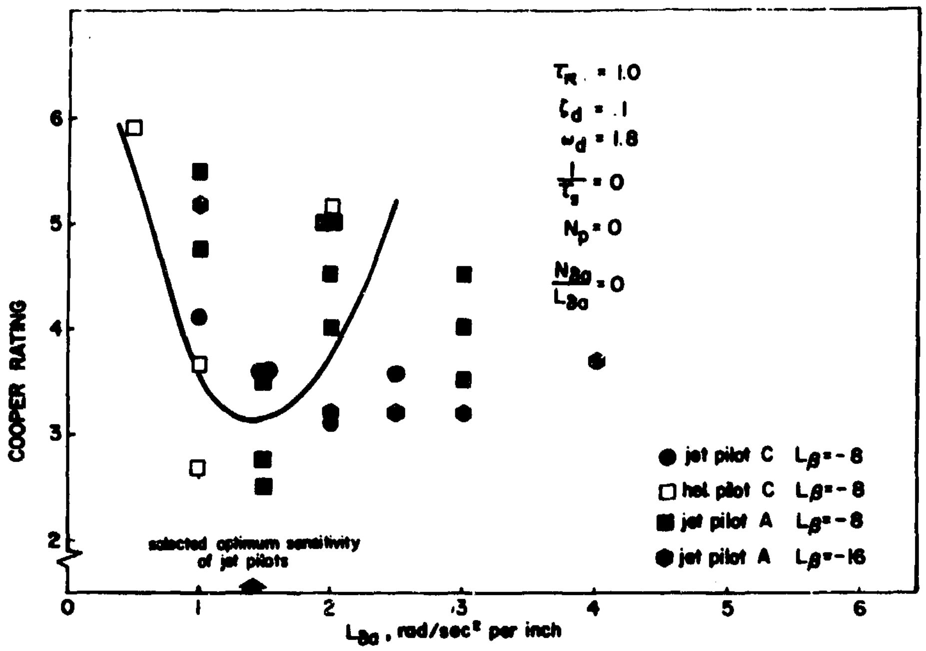


Figure 12 (3.3.4.2)
LATERAL CONTROL SENSITIVITY BOUNDARY, τ_R = 1.0
(FROM REFERENCE G10)

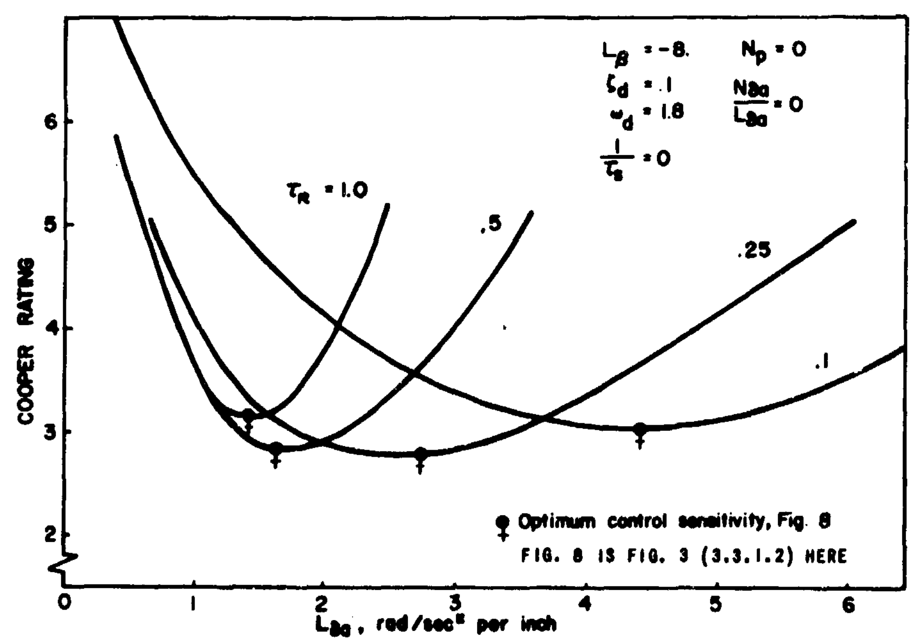


Figure 13 (3.3.4.2)
LATERAL CONTROL SENSITIVITY BOUNDARIES (FROM REFERENCE G10)

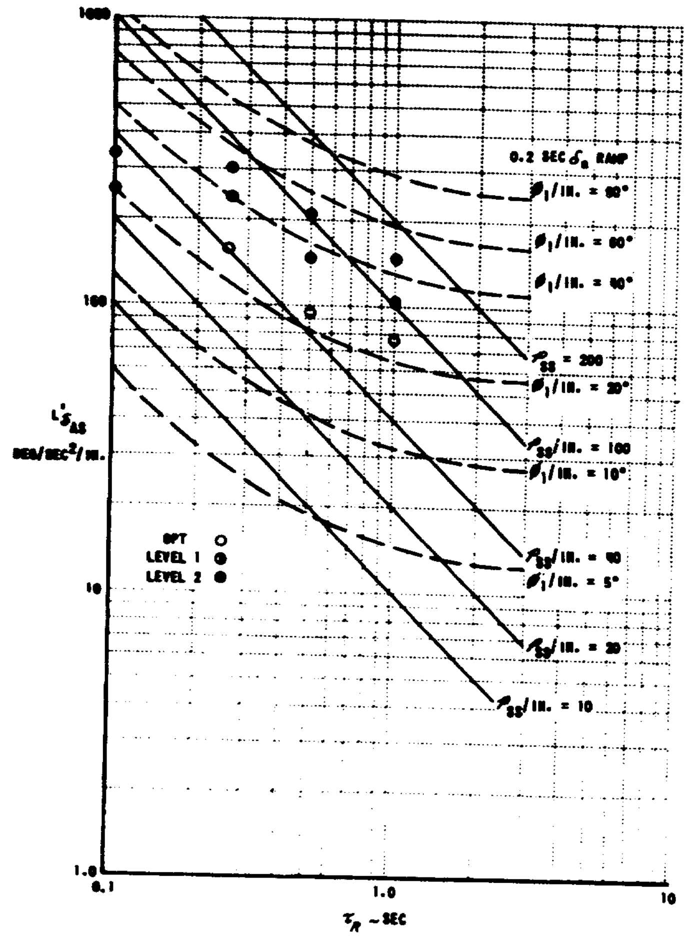


Figure 14 (3.3.4.2)
FLIGHT PHASE CATEGORY C- DEFLECTION SENSITIVITY (FROM REFERENCE G10)

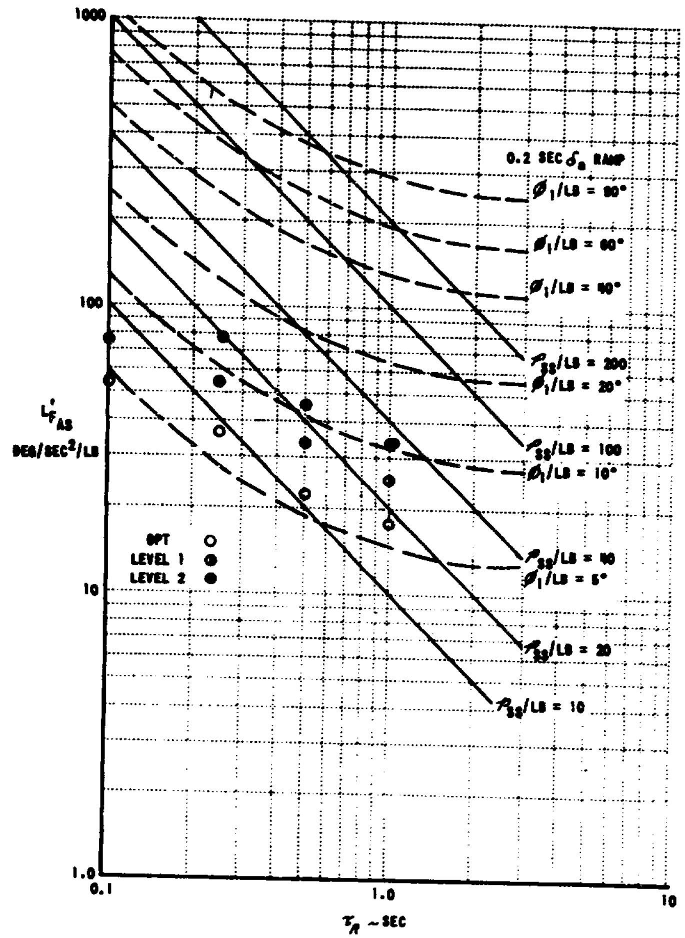


Figure 15 (3.3.4.2)
FLIGHT PHASE CATEGORY C-FORCE SENSITIVITY (FROM REFERENCE G10)

CONFIG.	AILERON CONTROL				FAVORABLE FEATURES	OBJECTIONABLE FEATURES	SPEC T
	GENERAL	AILERON YAW	COORDINATION	FEEL			
B-1	AILERON RESPONSE INITIALLY ABRUPT, THEN SIDESLIP COMES IN AND ROLL RATE SLOWS DOWN. ENCOUNTERED ROLL REVERSAL DUE TO THE QUANTITY OF ADVERSE YAW GENERATED REQUIRING EXCESSIVE LATERAL CONTROL. THIS CAUSED THE SIDESLIP TO BE INTOLERABLE. LIMITED TO SMALL STEP TYPE INPUTS.	TREMENDOUS AMOUNT OF ADVERSE YAW.	NO COMMENT.	HIGH SENSITIVITY, LOW FRICTION BAND, LOW FORCE GRADIENT, LOW BREAKOUT FORCE. STICK CENTERING OK.	NONE.	HIGH ADVERSE YAW RESULTING IN SIDESLIP DISTURBANCES THAT CULMINATE IN SIDESLIP DIVERGENCE WITH OVER 30° OF BANK. LATERAL CONTROL RESTRICTED TO SLOW SMALL INPUTS.	MAKE ONLY VER
B-2	ROLL CONTROL TOO SENSITIVE. TOO MUCH ROLL RATE ACCELERATION FOR A GIVEN STICK DEFLECTION. EVEN SMALL INPUTS INDUCE ZERO DAMPED SIDESLIP OSCILLATIONS. CONTINUOUS DUTCH ROLL OSCILLATIONS. COULD NOT MAINTAIN BANK ANGLE.	STRONG ADVERSE YAW	PILOT A - REFUSE TO FLY IT ON THE MISSION WITHOUT RUDDER PEDALS. PILOT B - NO RUDDERS WOULD BE REQUIRED IF THEY WERE AVAILABLE TO COORDINATE SMALL BANK ANGLE CHANGES.	HIGH SENSITIVITY, HIGH CONTROL POWER, INSIGNIFICANT BREAKOUT FORCE, SMALL FRICTION BAND, POOR STICK CENTERING.	STABILIZED HEADING CONTROL IS QUITE GOOD.	SENSITIVE LATERAL CONTROL, LOW ROLL DAMPING, STRONG ADVERSE YAW AND RESULTING HIGHLY OSCILLATORY DUTCH ROLL MODE. LACK OF DIRECTIONAL STABILITY.	SLOW SMALL LA
B-2A	SLIGHTEST DEFLECTION CAUSES DUTCH ROLL OSCILLATION.	VERY HIGH ADVERSE YAW.	I THINK THAT RUDDERS WOULD NOT HELP.	LOW BREAKOUT FORCE, LIGHT FORCE GRADIENT, POOR STICK CENTERING, MEDIUM TO HIGH LATERAL CONTROL POWER,	HEADING CONTROL GOOD.	THE INTOLERABLE MAGNITUDE OF SIDESLIP. VERY HIGH ADVERSE YAW WITH LATERAL CONTROL INPUT AND LARGE $\|\phi/\beta\|$ DUTCH ROLL OSCILLATION. HIGH ROLL DUE TO SIDESLIP.	$\dot{\beta}$ TECHNIQUE
B-3	UNPREDICTABLE ROLL RESPONSE DUE TO SIDESLIP INDUCED. CONTROL TOO SENSITIVE, ABRUPT, TEND TO OVERSHOOT IN SMALL BANK ANGLE CHANGES. ROLL ACCELERATION CAPABILITY TOO GREAT. YOU GET A SUSTAINED SIDESLIP OSCILLATION THAT IS NEITHER CONVERGENT NOR DIVERGENT.	OBJECTIONABLE ADVERSE YAW.	PILOT A - RUDDER PEDAL DESIRABLE AND WOULD IMPROVE RATING. PILOT B - NO COORDINATION WITH RUDDER REQUIRED.	ROLL SENSITIVITY IS SO HIGH YOU TEND TO OPERATE IN THE BREAKOUT REGION ALL THE TIME. LIGHT FORCE GRADIENT, POOR STICK CENTERING.	NONE.	HIGH SENSITIVITY. ABRUPT INITIAL ROLL ACCELERATION AND UNDEFINABLE ROLL RATES. OBJECTIONABLE ADVERSE YAW. ALMOST COMPLETE LACK OF ROLL DAMPING. LARGE ROLL DUE TO SIDESLIP AND LOW DAMPED DUTCH ROLL MODE.	USE SMALL INPU NIQUE HELPS
B-3A	GOOD LATERAL CONTROL POWER IN CONJUNCTION WITH THE LATERAL SENSITIVITY. OK FOR SMALL BANK ANGLE CHANGES BUT OBJECTIONABLE FOR LARGE CHANGES. FOR SMOOTH CHANGES IN BANK ANGLE YOU ARE RESTRICTED TO 2½ INCH DEFLECTION.	GENERALLY PRETTY SMALL AND ADVERSE. IF I LOOK AT THE YAW RATE INDICATOR, THE YAWING ACCELERATION DUE TO ROLL CONTROL IS ZERO BUT IT DOES INDUCE ADVERSE SIDESLIP.	PILOT A - LACK OF RUDDER CONTROL DEGRADES CONFIGURATION. NOT NECESSARY FOR SLOW CAUTIOUS MANEUVER. PILOT B - RUDDERS NOT REQUIRED OR DESIRED.	LOW BREAKOUT FORCE, LOW FRICTION BAND, VERY LIGHT GRADIENT, MEDIUM POWER CONTROL, POOR STICK CENTERING.	HEADING CONTROL NOT TOO BAD. DESIRABLE CONTROL POWER AND SENSITIVITY.	LARGE STRONG SIDESLIP INDUCED BY ROLLING MOMENTS THAT TRIGGERS OBJECTIONABLE DUTCH ROLL OSCILLATIONS. STRONG ROLL DUE TO SIDESLIP. GENERATION OF ADVERSE YAW RESTRICTS USE OF LATERAL CONTROL WITH RESPECT TO PRECISE MANEUVERING. LOW DIRECTIONAL STIFFNESS.	NONE REQUIRED.
B-4	TOO SENSITIVE AND ABRUPT WITH TOO MUCH ROLL ACCELERATION AND ROLL RATE FOR A GIVEN DEFLECTION.	QUITE SMALL ADVERSE YAW. IF YOU LOOK AT YAW RATE INDICATOR IT'S ABOUT ZERO. RELATIVELY INSIGNIFICANT SIDESLIP INDUCED BY AILERON.	NO COMMENT.	HIGH SENSITIVITY, HIGH CONTROL POWER, LOW BREAKOUT FORCE, LOW FORCE GRADIENT	CAN MAINTAIN HEADING REASONABLY WELL.	LATERAL CONTROL SENSITIVITY TOO HIGH. TOO MUCH ROLL ACCELERATION AND ROLL RATE FOR A GIVEN STICK DEFLECTION. LARGE ROLL DUE TO SIDESLIP. UNCERTAIN ROLLING VELOCITY. NOT STIFF ENOUGH DIRECTIONALLY.	KEEP THE GAIN
B-4A	COULD ESTABLISH AND MAINTAIN BANK ANGLE OR HEADING EASILY. COULD USE FULL DEFLECTION. COULD BE A LITTLE MORE SENSITIVE.	VERY SLIGHT ADVERSE YAW. JUST A LITTLE SIDESLIP EXCITATION.	NOT REQUIRED.	LOW BREAKOUT FORCE, LOW FRICTION BAND, LOW TO MEDIUM FORCE GRADIENT. LOW TO MEDIUM SENSITIVITY, LOW TO MEDIUM CONTROL POWER.	OVERALL BANK ANGLE CONTROL IS QUITE GOOD. HEADING CONTROL IS QUITE GOOD. LATERAL CONTROL SUFFICES QUITE WELL FOR THE NORMAL MANEUVERING TASKS. DUTCH ROLL EXCITATION IS VERY SLIGHT.	NO MAJOR OBJECTIONS. WOULD LIKE A LITTLE GREATER LATERAL CONTROL SENSITIVITY.	NONE REQUIRED.
B-5	MUCH TOO SENSITIVE. CAN USE ONLY SMALL INPUTS.	BAD PROVERSE YAW.	NO COMMENT.	LOW BREAKOUT FORCE, SMALL FRICTION BAND, LIGHT FORCE GRADIENT, HIGH SENSITIVITY, MEDIUM TO HIGH CONTROL POWER, POOR STICK CENTERING.	NONE.	PROVERSE YAW THAT TRIGGERS HIGHLY OSCILLATORY DUTCH ROLL MODE. MUCH TOO SENSITIVE LATERAL CONTROL. HIGH ROLL DUE TO SIDESLIP.	EVERYTHING MUST WITH SMALL INP NIQUE WAS USE TURE OF RANDOM
B-5X	COULDN'T ACHIEVE A TRIMMED FLIGHT CONDITION. USE OF AILERONS CAUSED THE AIRPLANE TO BE DIVERGENT CLOSED LOOP IN SIDESLIP (OSCILLATORY DIVERGENCE). DID NOT INDUCE SIDESLIP IN STEEP TURNS LIKE THEY DID IN STRAIGHT AND LEVEL FLIGHT. OVERLY SENSITIVE. INITIALLY ABRUPT.	SMALL AND PROVERSE	WOULD LIKE TO HAVE RUDDER PEDALS.	OVERLY SENSITIVE.	NONE.	UNABLE TO FLY IT OPEN LOOP IN BANK ANGLE BECAUSE OF UNSTABLE DIVERGENT SIDESLIP OSCILLATIONS. CONTROLLABLE ONLY WITH COMPLETE ATTENTION. LARGE INITIAL ROLL ACCELERATION. ROLL CONTROL TOO SENSITIVE. HEADING CONTROL DIFFICULT.	NO COMMENT.

A

	OBJECTIONABLE FEATURES	SPECIAL PILOTING TECHNIQUES	CONTROL IN PRESENCE OF DISTURBANCES	OVERALL OPINION	LATERAL-DIRECTIONAL FREE RESPONSE	LONGITUDINAL HANDLING
	HIGH ADVERSE YAW RESULTING IN SIDESLIP DISTURBANCES THAT CULMINATE IN SIDESLIP DIVERGENCE WITH OVER 30° OF BANK. LATERAL CONTROL RESTRICTED TO SLOW SMALL INPUTS.	MAKE ONLY VERY SMALL SLOW INPUTS.	DISTURBANCES WOULD SET OFF SIDESLIP, UNFLYABLE.	UNFLYABLE FOR THE MISSION DUE TO THE COMPLETE RESTRICTION ON LATERAL CONTROL.	LIGHTLY DAMPED DUTCH ROLL. ZERO DAMPED SIDESLIP OSCILLATION, MEDIUM $\|\phi/\beta\|$.	HIGHLY DAMPED SHORT PERIOD MODE. LOW BREAKOUT FORCE. LOW FORCE GRADIENT, LOW SENSITIVITY. MEDIUM CONTROL POWER WITH GOOD STICK CENTERING.
[...] QUITE	SENSITIVE LATERAL CONTROL, LOW ROLL DAMPING, STRONG ADVERSE YAW AND RESULTING HIGHLY OSCILLATORY DUTCH ROLL MODE. LACK OF DIRECTIONAL STABILITY.	SLOW SMALL LATERAL CONTROL INPUTS.	CONTINUOUS DUTCH ROLL OSCILLATIONS. CLOSED-LOOP ATTEMPTS TO CONTROL HEADING OR BANK ANGLE GREATLY INCREASES THE PROBLEM.	AIRCRAFT EXHIBITS CLOSED-LOOP DIVERGENT DIRECTIONAL OSCILLATIONS.	LOW DAMPED DUTCH ROLL MODE, HIGH $\|\phi/\beta\|$.	
	THE INTOLERABLE MAGNITUDE OF SIDESLIP. VERY HIGH ADVERSE YAW WITH LATERAL CONTROL INPUT AND LARGE $\|\phi/\beta\|$ DUTCH ROLL OSCILLATION. HIGH ROLL DUE TO SIDESLIP.	$\dot{\beta}$ TECHNIQUE HELPS STOP OSCILLATIONS.	QUITE A BIT OF ACTIVATION THROUGH THE RUDDER CHANNEL. CONTINUOUS LATERAL INPUTS REQUIRED TO MAINTAIN ±10° STRAIGHT AND LEVEL OR GIVEN BANK ANGLE. AT TIMES INPUTS TENDED TO MAGNIFY SIDESLIP MAGNITUDE.	VERY HIGH ADVERSE YAW AND INTOLERABLE SIDESLIP CAUSES CONTINUOUS DUTCH ROLL OSCILLATIONS.		
	HIGH SENSITIVITY. ABRUPT INITIAL ROLL ACCELERATION AND UNDEFINABLE ROLL RATES. OBJECTIONABLE ADVERSE YAW, ALMOST COMPLETE LACK OF ROLL DAMPING. LARGE ROLL DUE TO SIDESLIP AND LOW DAMPED DUTCH ROLL MODE.	USE SMALL SMOOTH INPUTS. $\dot{\beta}$ TECHNIQUE HELPS DAMP SIDESLIP.	RANDOM OSCILLATIONS MAKE IT DIFFICULT TO HOLD A BANK ANGLE OR STRAIGHT AND LEVEL.	ROLL RESPONSE IS ABRUPT AND UNPREDICTABLE. DIFFICULT TO FIND THE AILERON INPUT THAT WILL GIVE ZERO ROLL RATE.	LOW DIRECTIONAL STIFFNESS. LOW DAMPED DUTCH ROLL MODE. MEDIUM TO HIGH ROLL DAMPING. HIGH $\|\phi/\beta\|$.	
[...] DESIR- [...]IVITY.	LARGE STRONG SIDESLIP INDUCED BY ROLLING MOMENTS THAT TRIGGERS OBJECTIONABLE DUTCH ROLL OSCILLATIONS. STRONG ROLL DUE TO SIDESLIP. GENERATION OF ADVERSE YAW RESTRICTS USE OF LATERAL CONTROL WITH RESPECT TO PRECISE MANEUVERING. LOW DIRECTIONAL STIFFNESS.	NONE REQUIRED.	CARRIES OVER IN HIGH $\|\phi/\beta\|$ CHARACTERISTIC AND CAUSES BANK ANGLE CONTROL PROBLEMS.	DIFFICULT TO ESTABLISH A GIVEN BANK ANGLE.		
[...]LY WELL.	LATERAL CONTROL SENSITIVITY TOO HIGH. TOO HIGH ROLL ACCELERATION AND ROLL RATE FOR A GIVEN STICK DEFLECTION. LARGE ROLL DUE TO SIDESLIP. UNCERTAIN ROLLING VELOCITY. NOT STIFF ENOUGH DIRECTIONALLY.	KEEP THE GAIN DOWN ON PILOT INPUTS.	RATING NOT NOTICEABLY AFFECTED.	ROLL ACCELERATION IS QUITE ABRUPT. ROLL RATE TOO HIGH. ROLL CONTROL OVERLY SENSITIVE. CONFIGURATION IS MARGINAL FOR THE MISSION.	LOW DAMPED DUTCH ROLL MODE. HIGH $\|\phi/\beta\|$, LOW DIRECTIONAL STIFFNESS.	
[...] QUITE [...]TE GOOD. [...] WELL [...]ED. [...] SLIGHT.	NO MAJOR OBJECTIONS. WOULD LIKE A LITTLE GREATER LATERAL CONTROL SENSITIVITY.	NONE REQUIRED.	RANDOM NOISE EVALUATION NOT MADE. HOWEVER IN LIGHT TO MODERATE TURBULENCE THERE WERE NO MAJOR OBJECTIONS.	BANK ANGLE AND HEADING CONTROL QUITE GOOD. LOW ROLL DAMPING FORCES ME TO MAKE AN OPPOSITE CONTROL INPUT IN STOPPING. QUITE HIGH ROLL DUE TO YAW. ACCEPTABLE SATISFACTORY, FAIR TO GOOD.		
	PROVERSE YAW THAT TRIGGERS HIGHLY OSCILLATORY DUTCH ROLL MODE. MUCH TOO SENSITIVE LATERAL CONTROL. HIGH ROLL DUE TO SIDESLIP.	EVERYTHING MUST BE DONE QUITE SLOWLY WITH SMALL INPUTS. INVERSE $\dot{\beta}$ TECHNIQUE WAS USEFUL IN DECREASING AMPLITUDE OF RANDOM OSCILLATIONS.	PROBLEM NOT ALTERED SIGNIFICANTLY BY RANDOM NOISE.	CONSTANT ATTENTION REQUIRED TO KEEP WINGS LEVEL. CAN DO PRETTY GOOD JOB IN A STEEP TURN. HEADING CONTROL FAIR. HIGH ROLL DUE TO SIDESLIP. SENSITIVE LATERAL CONTROL WITH PROVERSE YAW TRIGGERS HIGHLY OSCILLATORY DUTCH ROLL MODE.	HIGH $\|\phi/\beta\|$. QUITE LOW DUTCH ROLL DAMPING, PARTICULARLY AT LOW FREQUENCIES.	
	UNABLE TO FLY IT OPEN LOOP IN BANK ANGLE BECAUSE OF UNSTABLE DIVERGENT SIDESLIP OSCILLATIONS. CONTROLLABLE ONLY WITH COMPLETE ATTENTION. LARGE INITIAL ROLL ACCELERATION. ROLL CONTROL TOO SENSITIVE. HEADING CONTROL DIFFICULT.	NO COMMENT.	NOT BAD IF PILOT GAIN IS KEPT LOW.	DIFFICULT TO FLY IN ROLL. AIRCRAFT WENT DIVERGENT CLOSED LOOP IN SIDESLIP DUE TO AILERON INPUTS. MORE FLYABLE IN STEEP TURNS. ROLL CONTROL TOO SENSITIVE. ROLL DUE TO SIDESLIP PRETTY LARGE. UNABLE TO MAINTAIN BANK ANGLE IN STRAIGHT AND LEVEL FLIGHT. CONTROLLABLE WITH COMPLETE ATTENTION.		

Figure 16 (3.3.4.2)
SUMMARY OF PILOT COMMENTS FOR IN-FLIGHT CONFIGURATIONS
(FROM REFERENCE F5)

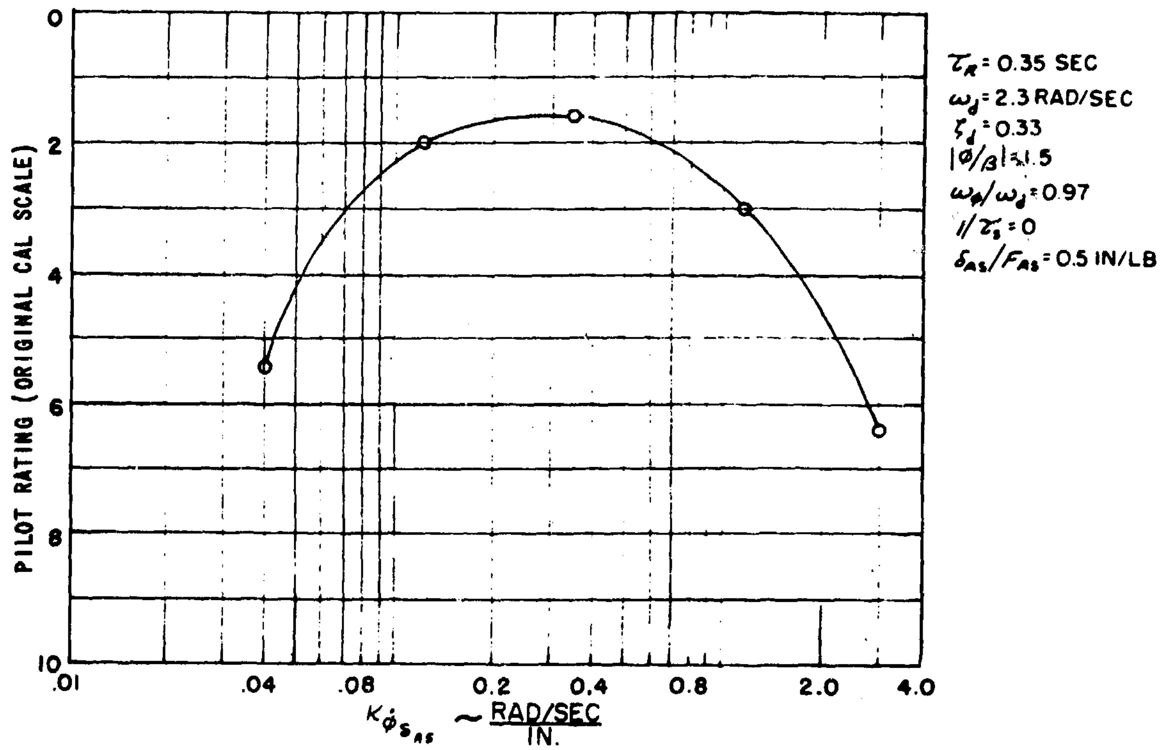


Figure 17 (3.3.4.2)

PILOT RATING VERSUS STEADY STATE ROLL RATE PER INCH OF AILERON STICK DISPLACEMENT (FROM REFERENCE F22)

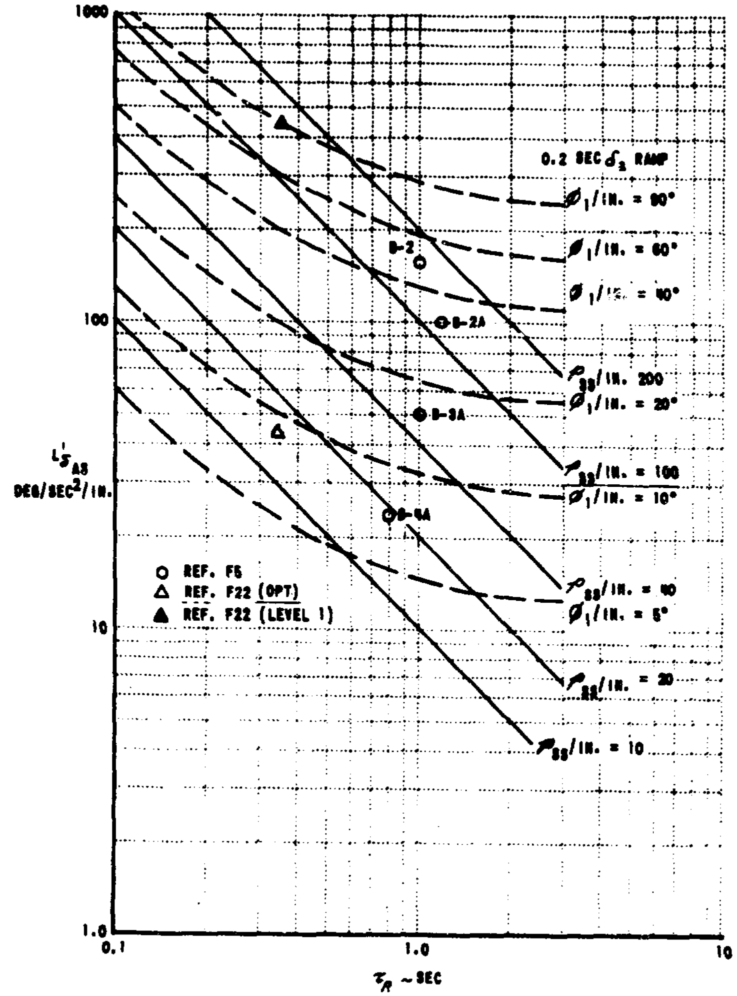


Figure 18 (3.3.4.2)
FLIGHT PHASE CATEGORY B - DEFLECTION SENSITIVITY

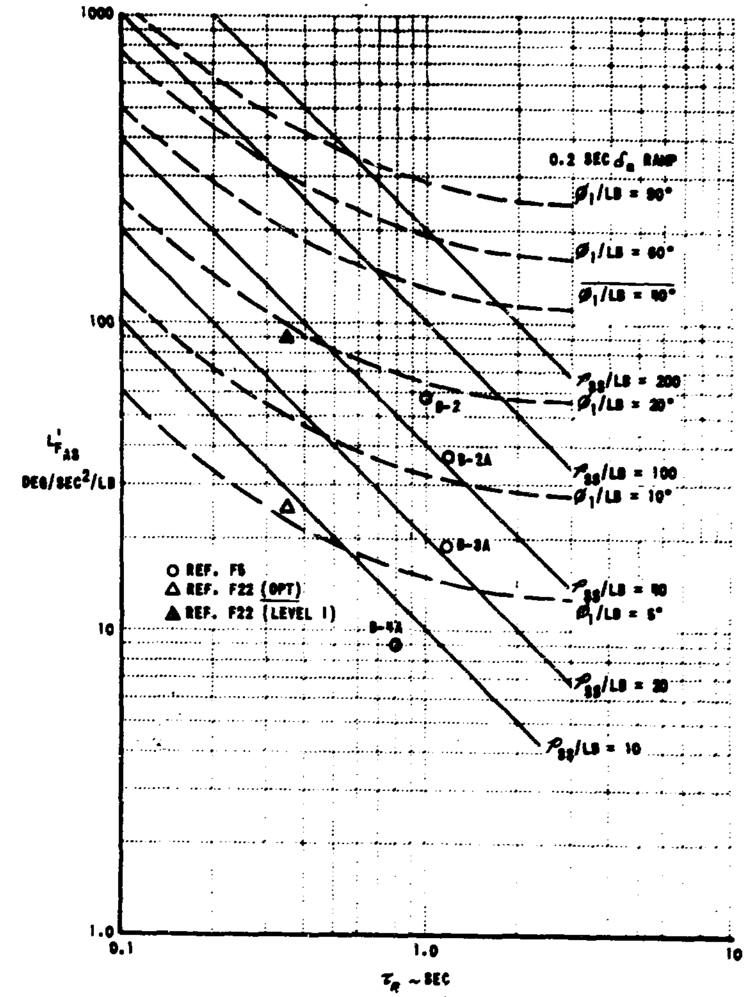


Figure 19 (3.3.4.2)
FLIGHT PHASE CATEGORY B - FORCE SENSITIVITY

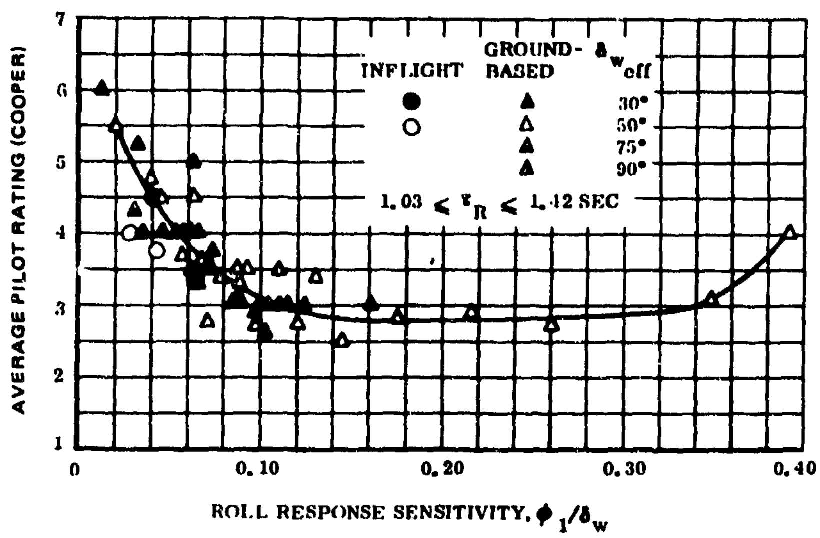


Figure 20 (3.3.4.2)
VARIATION OF PILOT RATING WITH ROLL RESPONSE SENSITIVITY
(FROM REFERENCE C7)

3.3.4.3 LINEARITY OF ROLL RESPONSE

REQUIREMENT

3.3.4.3 Linearity of roll response. There shall be no objectionable nonlinearities in the variation of rolling response with aileron control deflection or force. Sensitivity or sluggishness in response to small aileron control deflections or forces shall be avoided.

RELATED MIL-F-8785 PARAGRAPH

3.14.16.8

DISCUSSION

This requirement is similar to MIL-F-8785 paragraph 3.4.16.8, but has been generalized to prohibit objectional nonlinearities for any size aileron control input. The requirement is directed at precision of control. Objectionable nonlinearities have been detents, nonlinear force gradients, nonlinear $C_n(\delta_{AS})$ or $C_\ell(\delta_{AS})$, spoiler lag, etc. It has not been possible to specify values for tolerable levels of such nonlinearities, so reliance must be placed on avoidance or on pilots' evaluations.

Tests using 1/4, 1/2, 3/4 and full aileron are commonly used to demonstrate compliance. Such tests can also be used to help determine k for use in 3.3.2.4 and 3.3.2.4.1.

3.3.4.4 WHEEL CONTROL THROW

REQUIREMENT

3.3.4.4 Wheel control throw. For airplanes with wheel controllers, the wheel throw necessary to meet the roll perf rmance requirements specified in 3.3.4 shall not exceed 60 degrees in either direction. For completely mechanical systems, the requirement may be relaxed to 80 degrees.

RELATED MIL-F-8785 PARAGRAPH

3.4.16.4

DISCUSSION

This paragraph replaces paragraph 3.4.16.4 of MIL-F-8785. The wheel control throw has been reduced from the MIL-F-8785 value of ±90 degrees to ±60 degrees in keeping with the trend toward lower wheel throw angles. A smaller wheel throw facilitates flying, particularly for one-handed operation. Also, as the Reference C7 data indicate (Figure 1), to maintain a desirable roll response sensitivity in terms of roll performance per degree of wheel deflection, the smaller the wheel throw, the lower the required roll performance. (This is valid providing roll effectiveness is equal to or greater than the specified roll effectiveness requirements.)

Reference A8 makes recommendations concerning the amount of wheel throw for one-handed operation, and, although the comments pertain to VTOL vehicles, the recommendation may well be of general applicability. The Reference A8 recommendation is that for one-handed operation, the wheel throw should not exceed 60 degrees in each direction.

A wheel throw of ±80 degrees for completely mechanical systems has been specified in deference to the airplane designer.

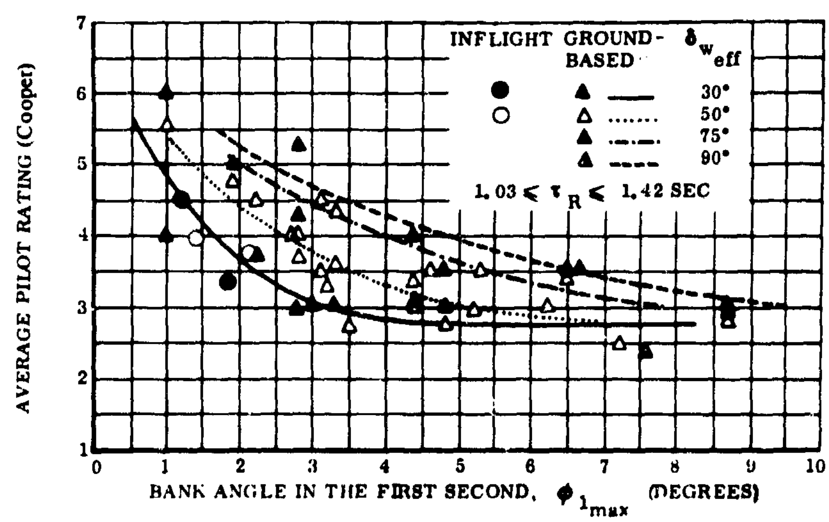


Figure 1 (3.3.4.4)

VARIATION OF PILOT RATING WITH BANK ANGLE IN THE FIRST SECOND FOR FOUR VALUES OF EFFECTIVE ANGLE (FROM REFERENCE C7)

3.3.4.5 RUDDER-PEDAL-INDUCED ROLLS

REQUIREMENT

3.3.4.5 Rudder-pedal-induced rolls. For Levels 1 and 2, it shall be possible to raise a wing by use of rudder pedal alone, with right rudder pedal force required for right rolls and left rudder pedal force required for left rolls. For Level 1, with the aileron control free, it shall be possible to produce a roll rate of 3 degrees per second with an incremental rudder pedal force of 50 pounds or less. The specified roll rate shall be attainable from coordinated turns at up to ±30 degrees bank angle with the airplane trimmed for wings-level, zero-yaw-rate flight.

RELATED MIL-F-8785 PARAGRAPHS

None.

DISCUSSION

This requirement is new and is specified for the reasons given in the discussion of requirements pertaining to sideslips under paragraph 3.3. It does not absolutely require stable C_{ℓ_β} because it can be met with a rudder-aileron interconnect. For the same reason, this requirement does not necessarily have implications on survivability or vulnerability.

A number of pilots who once preferred a "completely uncoupled airplane" now, after further experience and reflection, support this requirement.

3.3.5 DIRECTIONAL CONTROL CHARACTERISTICS

REQUIREMENT

3.3.5 Directional control characteristics. Directional stability and control characteristics shall enable the pilot to balance yawing moments and control yaw and sideslip. Sensitivity to rudder pedal forces shall be sufficiently high that directional control and force requirements can be met and satisfactory coordination can be achieved without unduly high rudder pedal forces, yet sufficiently low that occasional improperly coordinated control inputs will not seriously degrade the flying qualities.

RELATED MIL-F-8785 PARAGRAPHS

3.4.11, 3.4.14, 3.4.15

DISCUSSION

This is a general catch-all requirement on balancing yawing moments. Requirements on rudder pedal forces for specific conditions are given in the paragraphs referenced below:

3.3.5.1	(speed change)
3.3.5.2	(wave-off/go-around)
3.3.6	(steady sideslips)
3.3.7, 3.3.7.1, 3.3.7.2	(cross winds)
3.3.8	(dives)
3.3.9.1, 3.3.9.2, 3.3.9.5	(asymmetric thrust)
3.4.3	(spins)

The related MIL-F-8785 requirements were examined for validity and changed as necessary. A qualitative requirement on another important flying qualities parameter, rudder-pedal sensitivity, has been included. A quantitative requirement could not be specified because of insufficient data.

3.3.5.1 DIRECTIONAL CONTROL WITH SPEED CHANGE

REQUIREMENT

3.3.5.1 Directional control with speed change. When initially trimmed directionally with symmetric power, the trim change of propeller-driven airplanes with speed shall be such that straight flight can be maintained over a speed range of ±30 percent of the trim speed or ±100 knots equivalent airspeed, whichever is less (except where limited by boundaries of the Service Flight Envelope) with rudder pedal forces not greater than 100 pounds for Levels 1 and 2 and not greater than 180 pounds for Level 3, without retrimming. For other airplanes, rudder pedal forces shall not exceed 40 pounds at the specified conditions for Levels 1 and 2 nor 180 pounds for Level 3.

3.3.5.1.1 Directional control with asymmetric loading. When initially trimmed directionally with each 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 rudder pedal forces not greater than 100 pounds for Levels 1 and 2 and not greater than 180 pounds for Level 3, without retrimming.

RELATED MIL-F-8785 PARAGRAPHS

3.4.11

DISCUSSION

This requirement is based on the requirement stated in the first sentence of paragraph 3.4.11 of MIL-F-8785. In response to comments from industry and governmental agencies, the allowable rudder pedal forces have been reduced and have been made a function of type of propulsion, that is, either turbojet or propeller-driven. In the absence of propeller slipstream effects a more stringent requirement, which is closer to pilots' desires, is feasible to meet.

An additional requirement directed specifically at airplanes with asymmetric loading has been specified in 3.3.5.1.1. Its aim is to help keep pilot workload within bounds during the Flight Phases of an operational mission.

3.3.5.2 DIRECTIONAL CONTROL IN WAVE-OFF (GO-AROUND)

REQUIREMENT

3.3.5.2 Directional control in wave-off (go-around). For propeller-driven Class IV, and all propeller-driven carrier-based airplanes, the response to thrust, configuration, and airspeed change shall be such that the pilot can maintain straight flight during wave-off (go-around) initiated at speeds down to $V_S(PA)$ with rudder pedal forces not exceeding 100 pounds when trimmed at $V_{o_{min}}(PA)$. For other airplanes, rudder pedal forces shall not exceed 40 pounds for the specified conditions. The preceding requirements apply for Levels 1 and 2. For all airplanes the Level 3 requirement is to maintain straight flight in these conditions with rudder pedal forces not exceeding 180 pounds. For all levels, bank angles up to 5 degrees are permitted.

RELATED MIL-F-8785 PARAGRAPHS

3.4.11

DISCUSSION

This requirement is based on the requirement stated in the last sentence of paragraph 3.4.11 of MIL-F-8785. Allowable rudder pedal forces have been made a function of type of propulsion as in paragraph 3.3.5.1 of Reference A1. Changes have been made to conform more closely to the actual flight situation during a wave-off (Navy) or go-around (AF), and the requirement now applies to all airplanes. The need is obvious.

Asymmetries of configuration or thrust may be normal conditions (such as some asymmetric store loadings) or the result of Failure States (such as propulsion failure). The bank angle permitted is the same as for the specific requirements on control for asymmetric thrust (3.3.9.2).

3.3.6 LATERAL-DIRECTIONAL CHARACTERISTICS IN STEADY SIDESLIPS

REQUIREMENT

3.3.6 Lateral-directional characteristics in steady sideslips. The requirements of 3.3.6.1 through 3.3.6.3.1 and 3.3.7.1 are expressed in terms of characteristics in rudder-pedal-induced steady, zero-yaw-rate sideslips with the airplane trimmed for wings-level straight flight. Paragraphs 3.3.6.1 through 3.3.6.3 apply at sideslip angles up to those produced or limited by:

a. Full rudder pedal deflection, or

b. 250 pounds of rudder pedal force, or

c. Maximum aileron control or surface deflection,

except that for single-propeller-driven airplanes during wave-off (go-around), rudder pedal deflection in the direction opposite to that required for wings-level straight flight need not be considered beyond the deflection for a 10-degree change in sideslip from the wings-level straight flight condition.

RELATED MIL-F-8785 PARAGRAPH

3.4.3

DISCUSSION

The paragraphs within this section of Reference A1 were obtained primarily from a reorganization of paragraphs 3.4.3, 3.4.4, 3.4.5, 3.4.6, 3.4.6.1, 3.4.6.2, 3.4.7, 3.4.7.1 and 3.4.8 of MIL-F-8785. These were pulled together into one section since they all relate to airplane characteristics in steady rudder-pedal-induced sideslips. Furthermore, it is believed that the paragraphs have been most logically and precisely divided according to content by considering them as relating either to rolling moments, yawing moments or side forces in steady sideslips. The requirements have been altered very little, since this type of requirement is needed and in general has stood the test of time in the form of required airplane response and aileron-control inputs during steady rudder-pedal induced sideslips. Some operational uses of sideslips are listed in the discussion of 3.3. It should be noted that, although paragraphs 3.4.10 and 3.4.11.1 of MIL-F-8785 also refer to steady sideslip conditions, they are not covered in this section because of special implications to other areas of flying qualities. Paragraph 3.4.10 was omitted since it does not relate to rudder-pedal-induced sideslips (see Reference A1, 3.3.9.4), and 3.4.11.1 was omitted since it has special implications to cross-wind landings (see Reference A1, 3.3.7.1).

Paragraph 3.3.6 was obtained from a reorganization of paragraph 3.4.3 of MIL-F-8785 and, to avoid any ambiguity, pertains to rudder-pedal induced sideslips only. The type of airplane to which the wave-off (go-around) exception applies has been clarified.

Structural limitation is not given as one of the limitations on the size of sideslip that it is necessary to consider since the 250 pound rudder pedal force is well within the 300 pound force required to meet the structural specification requirements (MIL-A-8861).

Note that it must be possible to reach one of these limits to meet these requirements. For example, an airplane that diverges uncontrollably at a sideslip angle within all these limits is clearly unsatisfactory.

3.3.6.1 YAWING MOMENTS IN STEADY SIDESLIPS

REQUIREMENT

3.3.6.1 Yawing moments in steady sideslips. For the sideslips specified in 3.3.6, right rudder pedal deflection and force shall produce left sideslips and left rudder pedal deflection and force shall produce right sideslips. For Levels 1 and 2 the following requirements shall apply. The variation of sideslip angle with rudder pedal deflection shall be essentially linear for sideslip angles between +15 degrees and -15 degrees. For larger sideslip angles, an increase in rudder pedal deflection shall always be required for an increase in sideslip. The variation of sideslip angle with rudder pedal force shall be essentially linear for sideslip angles between +10 degrees and -10 degrees. Although a lightening of rudder pedal force is acceptable for sideslip angles outside this range, the rudder pedal force shall never reduce to zero.

RELATED MIL-F-8785 PARAGRAPHS

3.4.4, 3.4.5

DISCUSSION

The requirement combines paragraphs 3.4.4 and 3.4.5 of MIL-F-8785. At the request of industry, the range of linearity of rudder pedal force was reduced from ±15 degrees of sideslip to ±10 degrees of sideslip. This change does not seem objectionable to pilots, as long as rudder pedal deflections for larger sideslip angles remain stable. It is consistent with the rudder pedal force requirement. Rudder overbalance is still prohibited.

If one of the factors listed in 3.3.6 limits attainable sideslip to less than ±15 degrees, the requirements of 3.3.6.1 of course do not apply beyond that smaller sideslip angle.

Because of possible control cross-coupling, meeting this requirement will not necessarily assure static directional stability. However, the control-surface-fixed requirement of 3.3.1.1 should assure a stable "dynamic C_{n_β}."

3.3.6.2 SIDE FORCES IN STEADY SIDESLIPS

REQUIREMENT

3.3.6.2 Side forces in steady sideslips. For the sideslips of 3.3.6, 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.

RELATED MIL-F-8785 PARAGRAPH

3.4.8

DISCUSSION

This requirement replaces paragraph 3.4.8 of MIL-F-8785 and is worded almost identically. It is possible, though unlikely, to design an airplane that will not meet this requirement (Reference B11 shows $\frac{d\phi}{d\beta}$ in terms of stability and control derivatives). While there is some evidence that pilots do not object to zero bank in straight sideslips, opposite bank seems to be disconcerting. This was particularly apparent, according to an Air Force witness, in B-15 landing approaches.

3.3.6.3 ROLLING MOMENTS IN STEADY SIDESLIPS

REQUIREMENT

3.3.6.3 Rolling moments in steady sideslips. For the sideslips of 3.3.6, left aileron-control deflection and force shall accompany left sideslips, and right aileron-control deflection and force shall accompany right sideslips. For Levels 1 and 2, the variation of aileron-control deflection and force with sideslip angle shall be essentially linear.

RELATED MIL-F-8785 PARAGRAPHS

3.4.6, 3.4.7

DISCUSSION

This requirement combines paragraph 3.4.6 and most of paragraph 3.4.7 of MIL-F-8785 and specifies the sense of required aileron-control inputs in essentially the same way as MIL-F-8785. It does not necessarily require positive effective dihedral, because of possible control cross-coupling effects.

In reviewing this requirement, consideration was given to putting some lower limit on dihedral effect since data such as those presented in Reference G10 (Figure 1) indicate that zero or low L_β is undesirable. Reference G10 indicates that the zero L_β configurations were down-rated because the pilots were forced to use rudder pedals to damp the Dutch roll oscillations. Fighter pilots, in particular, desired some dihedral to enable them to damp the Dutch roll using ailerons alone. This has implications extending far beyond the scope of paragraph 3.3.6.3. The implications are that:

(a) the $|\phi/\beta|_d$ response ratio is sufficiently large that Dutch roll oscillations will show up in roll, and

(b) the combination of ψ_β (yaw due-to-aileron, etc.) and $\measuredangle p/\beta$ (phasing of the free Dutch roll oscillation) is such that aileron control inputs to damp the roll oscillations will generate yawing moments that damp β oscillations.

It is hoped that the requirements under 3.3.2 will cope with these implications. More work and data are necessary, however.

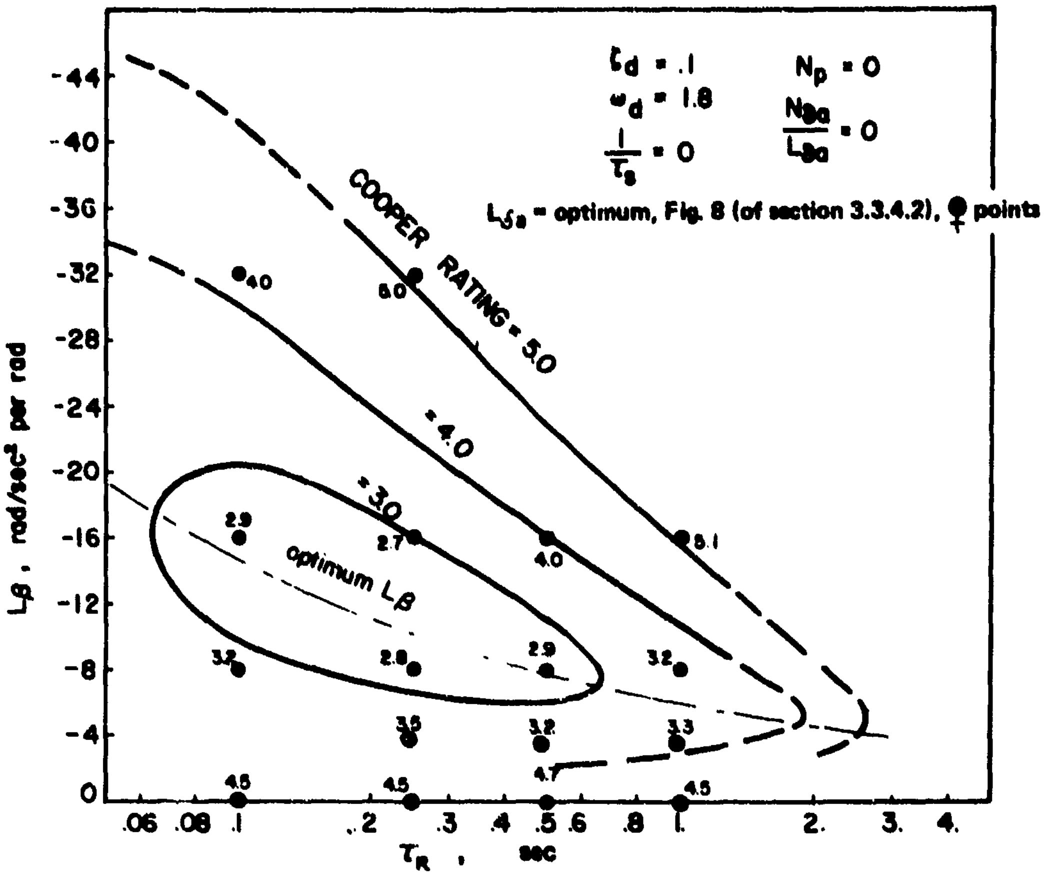


Figure 1 (3.3.6.3)

LATERAL FLYING QUALITIES BOUNDARIES

(L_β VS. τ_R , ζ_d=.1) (FROM REFERENCE G10)

3.3.6.3.1 EXCEPTION FOR WAVE-OFF (GO-AROUND)

REQUIREMENT

3.3.6.3.1 Exception for wave-off (go-around). The requirement of 3.3.6.3 may, if necessary, be excepted for wave-off (go-around) if task performance is not impaired and no more than 50 percent of roll control power available to the pilot, and no more than 10 pounds of aileron-control force, are required in a direction opposite to that specified in 3.3.6.3.

RELATED MIL-F-8785 PARAGRAPHS

3.4.6.1, 3.4.7.1

DISCUSSION

This requirement combines paragraphs 3.4.6.1 and 3.4.7.1 of MIL-F-8785. Such relaxation has been found both necessary on occasion and tolerable. In this paragraph, allowable aileron-control force is not made a function of the type of controller, since one-handed operation must be assumed for the wave-off or go-around maneuver. The phrase "available to the pilot" is used to take into account the fact that control surface position can be determined by both the pilot and the stability augmentation system. The pilot must be able to cope with disturbances during this low-altitude maneuver, so a control margin is provided. See the discussion of control power under 3.3.6.3.2.

3.3.6.3.2 POSITIVE EFFECTIVE DIHEDRAL LIMIT

REQUIREMENT

3.3.6.3.2 Positive effective dihedral limit. For Levels 1 and 2, positive effective dihedral (right aileron control for right sideslip and left aileron control for left sideslip) shall never be so great that more than 75 percent of roll control power available to the pilot, and no more than 10 pounds of aileron-stick force or 20 pounds of aileron-wheel force, are required for sideslip angles which might be experienced in service employment.

RELATED MIL-F-8785 PARAGRAPHS

3.4.6.2, 3.4.7

DISCUSSION

The requirement specifies allowable control power necessary for the sideslips and allowable aileron-control forces as a function of type controller. Since this requirement relates directly to aircraft usage, that is, the size of sideslip which "might be experienced in service employment," and since this is a very strong function of aircraft type, the requirement is tied to normal operational usage as was the corresponding requirement in MIL-F-8785. A margin of control power must be left for the pilot to cope with disturbances.

As defined in 6.2.4, control power is expressed in terms of moment-producing capability. There generally is a known or measurable relationship between surface deflection and control moment. The margin stated must be available to the pilot for effective control, over and above any surface deflection used for stability augmentation. As noted in the discussion of 3.5.4.2, saturation of augmentation will not be allowed to prohibit safe use of this control margin for maneuvering and compensating for disturbances.

3.3.7 LATERAL-DIRECTIONAL CONTROL IN CROSS WINDS

GENERAL DISCUSSION

Paragraphs 3.3.7, 3.3.7.1, 3.3.7.2.1 and 3.3.5.2.2 of Reference A1 replace paragraphs 3.4.13, 3.4.13.1, 3.4.13.2, 3.4.11.1, 3.7.6 and parts of paragraphs 3.4.6.2, 3.4.7 and 3.4.16.5 of MIL-F-8785. Reference A1 thus incorporates into one section all requirements relating maximum allowable aileron control and rudder pedal forces for takeoffs and landings in cross wind conditions, thus greatly reducing the amount of cross referencing. The main paragraphs within this section have been identified according to phase of operation, that is: final approach (3.3.7.1), takeoff run (3.3.7.2), landing rollout (3.3.7.2) and taxi (3.3.7.3).

The requirements are based on the philosophy that for Level 1 and 2 it must be possible to perform the severe task, although for Level 2 a degradation in flying qualities is accepted. Thus the cross winds under which a landing must be accomplished are the same for Levels 1 and 2, although cockpit forces may increase for Level 2. For Level 3, both the severity of the cross wind and the maximum allowable cockpit forces have been relaxed from those specified for Level 1.

REQUIREMENT

3.3.7 Lateral-directional control in cross winds. It shall be possible to take off and land with normal pilot skill and technique in 90-degree cross winds, from either side, of velocities up to those specified in table XI. Aileron-control forces shall be within the limits specified in 3.3.4.2, and rudder pedal forces shall not exceed 100 pounds for Level 1 nor 180 pounds for Levels 2 and 3. This requirement can normally be met through compliance with 3.3.7.1 and 3.3.7.2.

TABLE XI. Cross-Wind Velocity

Level	Class	Cross Wind
1 and 2	I	20 knots
	II, III, & IV	30 knots
	Water-based airplanes	20 knots
3	All	one-half the values for Levels 1 and 2

RELATED MIL-F-8785 PARAGRAPHS

3.4.13, 3.4.13.1, 3.4.13.2, 3.4.11.1, 3.7.6, 3.4.6.2, 3.4.7, 3.4.16.5

DISCUSSION

The most significant change in this requirement is the reduction in maximum specified cross wind for Class II, III and IV airplanes, from 40 knots to 30 knots. A relaxation of this requirement was recommended by several aircraft manufacturers and 30 knots was officially proposed by the Aerospace Industries Association (AIA), in its proposed revision of MIL-F-8785 (Reference A2).

That some relaxation of the requirement is warranted has been substantiated by a study performed by the USAF Environmental Technical Applications Center (ETAC) in direct support of the MIL-F-8785 revision effort. In the resulting ETAC report (Reference G2, Appendix II), the following recommendation is made: "The requirement that aircraft be able to land in cross-winds up to 40 knots is too severe for existing military airfields. This requirement could be relaxed to 25 knots and still achieve at least 99.5% operational effectiveness." This recommendation was made on the basis of examination of wind velocity data (based, in general, on averages for one-minute intervals taken hourly over a 5 to 10 year period) of 266 locations in the contiguous United States and 36 overseas locations. The Azores, which were considered critical in developing the MIL-F-8785 requirement, were not among the locations studied.

Reference A1 has specified a reduction to 30 knots rather than the suggested 25 knots since:

a) 30 knots represents a significant decrease in the maximum specified cross wind (decrease of 25%).

b) Although overall operational effectiveness may be 99.5%, the operational effectiveness at individual locations such as Shemyo and Thule, based on actual conditions, is somewhat below that figure. In addition, operational effectiveness depends upon forecast as well as actual conditions, and there may well be a sizable difference between the percentage of the time the wind at a given base is forecast to be above a certain speed and the percentage of the time that it is actually above that speed.

c) AIA recommended a 30-knot cross wind.

The requirement generally has been interpreted to specify ability to land in the stated cross wind with essentially zero crab. The side-load capacity of conventional landing gear is normally designed (MIL-A-8862) only to allow a margin for decrabbing inaccuracies in service use, in attempted zero-crab touchdowns.

3.3.7.1 FINAL APPROACH IN CROSS WINDS

REQUIREMENT

3.3.7.1 Final approach in cross winds. For all airplanes except land-based airplanes equipped with cross-wind landing gear, or otherwise constructed to land in a large crabbed attitude, rudder and aileron-control power shall be adequate to develop at least 10 degrees of sideslip (3.3.6) in the power approach with rudder pedal forces not exceeding the values specified in 3.3.7. For Level 1, aileron control shall not exceed either 10 pounds of force or 75 percent of control power available to the pilot. For Levels 2 and 3, aileron-control force shall not exceed 20 pounds.

RELATED MIL-F-8785 PARAGRAPHS

3.4.11.1, 3.4.6.2, 3.4.7

DISCUSSION

This paragraph combines the requirements of paragraph 3.4.11.1 and parts of 3.4.6.2 and 3.4.7 of MIL-F-8785, and incorporates them into the cross-wind section. The requirements are relatively unchanged but have been expanded to cover various levels. The concession to airplanes with cross-wind gear was retained intact as a practical matter, although its deletion was suggested so that there would be some requirement for such airplanes. In operational experience, 10 degrees of sideslip has often been needed as a bare minimum capability. See the discussion of control power under 3.3.6.3.2.

3.3.7.2 TAKEOFF RUN AND LANDING ROLLOUT IN CROSS WINDS

REQUIREMENT

3.3.7.2 Takeoff run and landing rollout in cross winds. Rudder and aileron-control power, in conjunction with other normal means of control, shall be adequate to maintain a straight path on the ground or other landing surface. This requirement applies in calm air and in cross winds up to the values specified in table XI, with cockpit control forces not exceeding the values specified in 3.3.7.

RELATED MIL-F-8785

3.4.13

DISCUSSION

This requirement is taken relatively unchanged from paragraph 3.4.13 of MIL-F-8785. Although propeller torque and slipstream prevented some World War II fighters from meeting such a requirement, that lack was always undesirable.

3.3.7.2.1 COLD- AND WET-WEATHER OPERATION

REQUIREMENT

3.3.7.2.1 Cold- and wet-weather operation. The requirements of 3.3.7.2 apply on wet runways for all airplanes, and on snow-packed and icy runways for airplanes intended to operate under such conditions. If compliance is not demonstrated under these adverse runway conditions, directional control shall be maintained by use of aerodynamic controls alone at all airspeeds above 50 knots for Class IV airplanes and above 30 knots for all others. For very slippery runways, the requirement need not apply for cross-wind components at which the force tending to blow the airplane off the runway exceeds the opposing tire-runway frictional force with the tires supporting all of the airplane's weight.

RELATED MIL-F-8785 PARAGRAPHS

3.4.13.2

DISCUSSION

This requirement is similar to, but expands upon, paragraph 3.4.13.2 of MIL-F-8785 to take into account operation from wet or very slippery runways. The need for the requirements is obvious. The low-speed limits for aerodynamic control are based upon operational experience.

Some airplanes having large side area tend to be blown sideways when there are high cross-wind components combined with very slippery runways. Under these circumstances it would be unreasonable to require the airplane to take off, so the cross-wind component for which the requirement applies is reduced from that contained in Table XI to that value above which the airplane would be blown off the runway with the tires supporting all of the airplane's weight. When analyzing an airplane design for compliance with this requirement, however, the expected variations in lift, side force, and cornering force (tire-runway side force) with speed and load should be used.

3.3.7.2.2 CARRIER-BASED AIRPLANES

REQUIREMENT

3.3.7.2.2 Carrier-based airplanes. All carrier-based airplanes shall be capable of maintaining a straight path on the ground without the use of wheel brakes, at airspeeds of 30 knots and above, during takeoffs and landings in a 90-degree cross wind of at least 10 percent $V_S(L)$. Cockpit control forces shall be as specified in 3.3.7.

RELATED MIL-F-8785

3.4.13.1

DISCUSSION

This requirement is taken relatively unchanged from paragraph 3.4.13.1 of MIL-F-8785.

3.3.7.3 TAXIING WIND SPEED LIMITS

REQUIREMENT

3.3.7.3 Taxiing wind speed limits. It shall be possible to taxi at any angle to a 35-knot wind for Class I airplanes and to a 45-knot wind for Class II, III, and IV airplanes.

RELATED MIL-F-8785 PARAGRAPHS

3.5.2

DISCUSSION

This paragraph replaces and expands upon paragraph 3.5.3 of MIL-F-8785. The conditions under which it must be possible to taxi have been specified since there is generally no point in being able to take off or land in a given cross wind if the aircraft cannot be taxied. The wind speeds specified are a compromise between what is desired and what is reasonable to require.

3.3.8 LATERAL-DIRECTIONAL CONTROL IN DIVES

REQUIREMENT

3.3.8 Lateral-directional control in dives. Rudder and aileron control power shall be adequate to maintain wings level and sideslip zero, without retrimming, throughout the dives and pullouts of 3.2.3.5 and 3.2.3.6. In the Service Flight Envelope, aileron control forces shall not exceed 20 pounds for propeller-driven airplanes nor 10 pounds for other airplanes. Rudder pedal forces shall not exceed 180 pounds for propeller-driven airplanes nor 50 pounds for other airplanes.

RELATED MIL-F-8785 PARAGRAPHS

3.4.15, 3.4.16.6

DISCUSSION

This paragraph replaces paragraphs 3.4.15 and 3.4.16.6 of MIL-F-8785 and has combined them into one paragraph. The major change is that allowable rudder pedal forces have been made a function of type of propulsion as was done in paragraphs 3.3.5.1 and 3.3.5.2 of Reference A1. Otherwise, the required levels of controllability have stood the test of time.

3.3.9 LATERAL-DIRECTIONAL CONTROL WITH ASYMMETRIC THRUST

REQUIREMENT

3.3.9 Lateral-directional control with asymmetric thrust. Asymmetric loss of thrust may be caused by many factors including engine failure, inlet unstart, propeller failure, or propeller-drive failure. Following sudden asymmetric loss of thrust from any factor, the airplane shall be safely controllable. The requirements of 3.3.9.1 through 3.3.9.4 apply for the appropriate Flight Phases when any single failure or malperformance of the propulsive system, including inlet or exhaust, causes loss of thrust on one or more engines or propellers, considering also the effect of the failure or malperformance on all subsystems powered or driven by the failed propulsive system.

RELATED MIL-F-8785 PARAGRAPHS

3.4.10, 3.4.12

DISCUSSION

The requirements in this section replace and expand upon paragraphs 3.4.10 and 3.4.12 of MIL-F-8785. Circumstances are recognized in which failure or malperformance of one item can have multiple consequences. Several of the new requirements are based upon existing, or proposed, civil aviation requirements and have been incorporated into the specification upon the recommendation of governmental agencies.

Generally, all the possible consequences of propulsion-system failures must be considered. For example, inlet unstart may cause a pitch disturbance. In that case the qualitative requirement of 3.4.9 must be met. Another kind of failure is represented by damage to other parts of the airplane caused by thrown turbine blades: for example, hydraulic lines should be routed (or enough armor used) so that thrown engine, fan or propeller parts cannot sever all hydraulic systems needed for flight control.

3.3.9.1 THRUST LOSS DURING TAKEOFF RUN

REQUIREMENT

3.3.9.1 Thrust loss during takeoff run. It shall be possible for the pilot to maintain control of an airplane on the takeoff surface following sudden loss of thrust from the most critical factor. Thereafter, it shall be possible to achieve and maintain a straight path on the takeoff surface without a deviation of more than 30 feet from the path originally intended, with rudder pedal forces not exceeding 180 pounds. For the continued takeoff, the requirement shall be met when thrust is lost at speeds from the refusal speed (based on the shortest runway from which the airplane is designed to operate) to the maximum takeoff speed, with takeoff thrust maintained on the operative engine(s), using only elevator, aileron, and rudder controls. For the aborted takeoff, the requirement shall be met at all speeds below the maximum takeoff speed; however, additional controls such as nosewheel steering and differential braking may be used in either case.

RELATED MIL-F-8785 PARAGRAPHS

3.4.10

DISCUSSION

This requirement expands on the first sentence of paragraph 3.4.10 of MIL-F-8785 by incorporating some ideas from paragraph 3.4.2.3 of ICAO Circular 75-AN/65 (Reference A7), paragraph 25.149 of FAR 25 (Reference A6), and aircraft manufacturers' comments. While the requirement of Reference A1 is more specific than the corresponding requirement in MIL-F-8785, it is still basically a qualitative rather than a quantitative requirement. This is considered to be the best approach at this time in light of the many variables that would have to be considered in specifying a strictly quantitative requirement. The objective of the requirement is to ensure that, following loss of thrust during the takeoff run, the pilot can either safely abort or safely continue the takeoff.

3.3.9.2 THRUST LOSS AFTER TAKEOFF

REQUIREMENT

3.3.9.2 Thrust loss after takeoff. During takeoff, it shall be possible without a change in selected configuration to achieve straight flight following sudden asymmetric loss of thrust from the most critical factor at speeds from V_{min} (TO) to V_{max}(TO), and thereafter to maintain straight flight throughout the climb-out. The rudder pedal force required to maintain straight flight with asymmetric thrust shall not exceed 180 pounds. Aileron control shall not exceed either the force limits specified in 3.3.4.2 or 75 percent of available control power, with takeoff thrust maintained on the operative engine(s) and trim at normal settings for takeoff with symmetric thrust. Automatic devices which normally operate in the event of a thrust failure may be used, and the airplane may be banked up to 5 degrees away from the inoperative engine.

RELATED MIL-F-8785 PARAGRAPHS

3.4.12

DISCUSSION

This requirement is based primarily on paragraph 3.4.12 of MIL-F-8785 but borrows ideas from paragraph 25.139 of FAR Part 25 (Reference A6) and paragraph 3.4.2 of ICAO Circular 75-AN/65 (Reference A7). The object of the requirement is to ensure that following thrust loss after takeoff, the pilot can safely continue climb-out. The intent is that V_{min}(TO) normally should be set by other considerations and adequate control provided down to that speed. This replaces the 1.2 V_S(TO) minimum control speed of MIL-F-8785. The straight flight path is not required to be parallel to the runway. Because the pilot must have at least 25% excess roll control power, Reference A1, like MIL-F-8785, is more stringent than the FAR (see the discussion of control power under 3.3.6.3.2).

The effect of thrust loss on rate of climb is a subject for performance specifications, rather than flying qualities specifications. The concern of Reference A1 and its predecessors has been to assure control of the flight path, whatever its inclination.

3.3.9.3 TRANSIENT EFFECTS

REQUIREMENT

3.3.9.3 Transient effects. The airplane motions following sudden asymmetric loss of thrust shall be such that dangerous conditions can be avoided by pilot corrective action. A realistic time delay (3.4.9) of at least 1 second shall be considered.

RELATED MIL-F-8785 PARAGRAPHS

3.4.10

DISCUSSION

This requirement expands on the first sentence of paragraph 3.4.10 of MIL-F-8785 by incorporating some ideas from paragraph 3.4.2.3 of Reference A7, paragraph 25.149 of Reference A6, and from aircraft manufacturers' comments. While the requirement of Reference A1 is more specific than the corresponding requirement in MIL-F-8785, it is still basically a qualitative rather than a quantitative requirement. This is considered to be the best approach at this time in light of the many variables that would have to be considered in specifying a strictly quantitative requirement. From 3.4.9,

> "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."

Depending upon expected initial pilot alertness and tightness of control, the magnitude, timing and unambiguity of pilot cues, and the type and variety of pilot action required, one second might be quite unrealistically short.

3.3.9.4 ASYMMETRIC THRUST - RUDDER PEDALS FREE

REQUIREMENT

3.3.9.4 Asymmetric thrust - rudder pedals free. The static directional stability shall be such that at all speeds above 1.4 V_{min}, with asymmetric loss of thrust from the most critical factor while the other engine(s) develop normal rated thrust, the airplane with rudder pedals free may be balanced directionally in steady straight flight. The trim settings shall be those required for wings-level straight flight prior to the failure. Aileron-control forces shall not exceed the Level 2 upper limits specified in 3.3.4.2 for Levels 1 and 2 and shall not exceed the Level 3 upper limits for Level 3.

RELATED MIL-F-8785 PARAGRAPHS

3.4.10

DISCUSSION

This requirement replaces most of paragraph 3.4.10 of MIL-F-8785. Although paragraph 3.4.10 of MIL-F-8785 is a requirement on transient response following sudden engine failure, the requirement is also directed at static directional stability and is thus of fundamental importance. The transient response requirement has been covered directly in paragraph 3.3.9.3 as discussed previously. The present wording in paragraph 3.3.9.4 of Reference A1 is almost identical to that in paragraph 5.2.4 of Reference A18, the predecessor of MIL-F-8785, and ensures a match between upsetting yawing moments due to asymmetric thrust and restoring moments from static directional stability. The wording of paragraph 5.2.4 of SR-119B and 1815B is:

> "5.2.4 The amount of rudder-free directional stability on multi-engine airpl[illegible]es in configuration P shall be such that at all speeds a[illegible]e 1.4 V_{S_G} with the more critical outboard engine inoperative (propeller windmilling, with propeller pitch control in the "low-pitch" setting) and the other engine or engines developing normal rated power, the airplane with rudder free may be balanced directionally in steady straight flight by sideslipping and banking. The weight shall be as specified in 5.2.1 and the trim settings shall be those required for wings-level straight flight in the test condition prior to cutting the engine."

Bank angle is a more generally prominent and natural parameter for the pilot to control in this situation than is yaw or sideslip.

3.3.9.5 TWO ENGINES INOPERATIVE

REQUIREMENT

3.3.9.5 Two engines inoperative. With any engine initially failed, it shall be possible upon failure of the most critical remaining engine to stop the transient motion at the one-engine-out speed for maximum range, and thereafter to maintain straight flight from that speed to the speed for maximum range with both engines failed. In addition, it shall be possible to effect a safe recovery at any service speed above $V_{o_{min}}$ (CL) following sudden simultaneous failure of the two critical failing engines.

RELATED MIL-F-8785 PARAGRAPH

None

DISCUSSION

This requirement is new and is based on paragraph 25.69 of Reference A6 and paragraph 3.5.6 of Reference A7. A requirement in the area has been included at the request of governmental agencies.

3.4 - MISCELLANEOUS FLYING QUALITIES

3.4 MISCELLANEOUS FLYING QUALITIES

DISCUSSION

Section 3.4 consists of those flying qualities aspects which are important, but which defy classification as primarily longitudinal, lateral-directional or control-system characteristics. Because of the complex nature of the subject treated, most of the requirements are qualitative in nature.

The subjects treated in this section were taken from various parts of MIL-F-8785, primarily Section 3.6. The subjects include stalls, spins, buffet, inertial coupling, control harmony, release of stores, and armament delivery. Most of the changes from the requirements of MIL-F-8785 were the direct result of comments from industry, the Air Force, and the Navy.

3.4.1 APPROACH TO DANGEROUS FLIGHT CONDITIONS

REQUIREMENT

3.4.1 Approach to dangerous flight conditions. Dangerous conditions may exist where the airplane 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.

3.4.1.1 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 stall warning (which requires pitching down or increasing speed), Mach buffet (which may indicate a need to decrease speed), and normal airplane vibration (which indicates no need for pilot action). If a warning or indication device is required, functional failure of the device shall be indicated to the pilot.

3.4.1.2 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, normal or inadvertent, shall never be possible. For Levels 1 and 2, neither hazardous nor nuisance operation shall be possible.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

Paragraphs 3.4.1 and 3.4.1.1 were deemed necessary to ensure that the pilot is properly warned when approaching dangerous flight conditions, particularly near the extremes of the flight envelopes. Wide latitude for approval or disapproval has been given the procuring activity because, first, the need for warning may not become apparent until late in the development program (or after it) and, second, because generally each such device will have to be tailored to a specific set of conditions. The procuring activity thus has a responsibility to establish specific criteria at the earliest possible time, to supplement these general requirements.

Paragraph 3.4.1.2 is designed to discourage prevention devices which create more problems than they solve.

These requirements clearly apply to stall warning and prevention devices, as well as to other cases.

3.4.2 STALLS

REQUIREMENT

3.4.2 Stalls. The requirements of 3.4.2 through 3.4.2.4.1 are to assure that the airflow separation induced by high angle of attack, which causes loss of aerodynamic lift or control about any one axis, does not result in a dangerous or mission-limiting condition. The stall is further defined in terms of speed and angle of attack in 6.2.2 and 6.2.5 respectively.

RELATED MIL-F-8785 PARAGRAPHS

3.6.2, 3.6.2.1, 3.6.2.2

DISCUSSION

The requirements under Section 3.4.2 are a rework of the requirements under Section 3.6 of MIL-F-8785. Many minor changes have been made, including a more complete definition of the stall warning range. The major changes are in the definition of stall speed.

Since the definition of stall speed is not a requirement, it was moved to 6.2 of the notes, where it belongs. There are three major changes in the definition of V_S:

1. The aspects of the definition of V_S discussed in 3.6.2.2 of MIL-F-8785 (limited elevator effectiveness, visibility, rate of sink, etc.) seemed more appropriate to define minimum service speed than V_S, and were therefore included in 3.1.8.2.

2. In order to define a single value of V_S for a given configuration and loading, a thrust setting is specified for the configurations associated with each flight phase.

3. Because the stall speed of interest is the speed at which loss of control is achieved with unity load factor normal to the flight path, a correction factor devised by the ICAO is to be used in flight test when $n_f \neq 1$ at the stall.

A subtle change in the definition of V_S is that MIL-F-8785 referred to the speed at which $C_{L_{max}}$ occurs and the speed at which loss of control occurs as if the two speeds were the same. The new definition of V_S picks the higher speed, recognizing that these two events do not necessarily occur at the same speed. A further limit recognized for V_S is intolerable buffet, which might start at an even higher speed. This change in definition is important for some airplanes because designers tend to define V_S only in terms of $C_{L_{max}}$.

Note that 6.2.2 defines V_S in essentially constant-speed flight, to minimize unsteady aerodynamic effects. Appreciable rates of speed reduction generally result in demonstrating a "V_S" lower than the required value.

3.4.2.1 REQUIRED CONDITIONS

REQUIREMENT

3.4.2.1 Required conditions. The requirements for stall characteristics apply for all Airplane Normal States in straight unaccelerated flight, and in turns and pullups with normal acceleration up to $n_{o_{max}}$. Specifically, the Airplane Normal States associated with the configurations, throttle settings, and trim settings of 6.2.2 shall be investigated; also, the requirements apply to Airplane Failure States that affect stall characteristics.

RELATED MIL-F-8785 PARAGRAPHS

3.6.1

DISCUSSION

Paragraph 3.4.2.1 is a rewrite of 3.6.1 of MIL-F-8785. All references to configurations, loadings, and stores were deleted because they would be redundant in view of 3.1.6.

The power settings associated with the configurations mentioned in 3.6.1 of MIL-F-8785 are very specific and rather arbitrary. Since the power setting can have a strong influence on the stalling characteristics, especially for propeller-driven airplanes or airplanes with boundary-layer control, it was deemed necessary to choose settings consistent with the Flight Phase under consideration.

Another change from MIL-F-8785 is that failures of the airplane systems must be considered. Sticking of one leading-edge slat in dive-bombing pull-outs is one recent example of an intolerable failure or malfunction. The resulting roll is so severe that airplanes have been lost.

3.4.2.2 STALL WARNING REQUIREMENTS

REQUIREMENT

3.4.2.2 Stall warning requirements. The stall approach shall be accompanied by an easily perceptible warning. Acceptable stall warning for all types of stalls consists of shaking of the cockpit controls, buffeting or shaking of the airplane, or a combination of both. The onset of this warning shall occur within the ranges specified in 3.4.2.2.1 and 3.4.2.2.2 but not within the Operational Flight Envelope. The increase in buffeting intensity with further increase in angle of attack shall be sufficiently marked to be noted by the pilot. This arning may be provided artifically only if it can be shown that natural stall warning is not feasible. These requiiements apply whether V_S is as defined in 6.2.2 or as allowed in 3.1.9.2.1.

3.4.2.2.1 Warning speed for stalls at lg normal to the flight path. Warning onset for stalls at lg normal to the flight path shall occur between the following limits:

Flight Phase	Minimum Stall Warning Speed	Maximum Stall Warning Speed
Approach	Higher of $1.05V_S$ or $V_S + 5$ knots	Higher of $1.10V_S$ or $V_S + 10$ knots
All Other	Higher of $1.05V_S$ or $V_S + 5$ knots	Higher of $1.15V_S$ or $V_S + 15$ knots

3.4.2.2.2 Warning range for accelerated stalls. Onset of stall warning shall occur outside the Operational Flight Envelope associated with the Airplane Normal State and within the following angle-of-attack ranges:

Flight Phase	Minimum Stall Warning Angle of Attack	Maximum Stall Warning Angle of Attack
Approach	$\alpha_0 + 0.82\ (\alpha_S - \alpha_0)$	$\alpha_0 + 0.90\ (\alpha_S - \alpha_0)$
All Other	$\alpha_0 + 0.75\ (\alpha_S - \alpha_0)$	$\alpha_0 + 0.90\ (\alpha_S - \alpha_0)$

where α_S is the stall angle of attack and α_0 is the angle of attack for zero lift (α_S is defined in 6.2.5; α_0 may be estimated from wind-tunnel tests).

RELATED MIL-F-8785 PARAGRAPHS

3.6.3, 3.6.3.1, 3.6.3.2

DISCUSSION

Operational experience shows that stall warning continues to be an important aspect of flying qualities for all airplanes. It is particularly critical if the stall and recovery characteristics are not entirely satisfactory.

Paragraphs 3.4.2.2 and 3.4.2.2.1 are a rewording of 3.6.3 and 3.6.3.1 of MIL-F-8785. An absolute speed margin has been introduced in 3.4.2.2.1 so that the margin is more realistic for airplanes with very low stall speeds, such as STOL's. The 5-knot minimum stall-warning margin, and the 10-knot maximum margin, are consistent with the minimum approach speeds reported in Reference C38: the higher of 1.1 V_S and V_S + 10 kt.

Paragraph 3.4.2.2.2 has been introduced to ensure that the pilot has adequate stall warning in maneuvers, as well as in straight flight. Angle of attack, rather than load factor, was used as the stall-warning reference here because some airplanes at certain flight conditions exhibit a rather wide range of angle of attack over which the lift coefficient changes relatively little. Stall warning loses its impact and interferes with maneuvering if it occurs at an angle of attack too far below the stall, as has been demonstrated on current operational airplanes. This requirement is most necessary for airplanes of Classes I and IV, but is important for Classes II and III as well.

Flight measurement of α_o is to be preferred, and often is easy; but an estimate based on wind-tunnel data is acceptable.

3.4.2.3 STALL CHARACTERISTICS

REQUIREMENT

3.4.2.3 Stall characteristics. In the unaccelerated stalls of 3.4.2.1, the airplane shall not exhibit uncontrollable rolling, yawing, or downward pitching at the stall in excess of 20 degrees for Classes I, II, and III, or 30 degrees for Class IV airplanes. It is desired that no pitch-up tendencies occur in unaccelerated or accelerated stalls. In unaccelerated stalls, mild nose-up pitch may be acceptable if no elevator control force reversal occurs and if no dangerous, unrecoverable, or objectionable flight conditions results. A mild nose-up tendency may be acceptable in accelerated stalls if the operational effectiveness of the airplane is not compromised and:

a. The airplane has adequate stall warning

b. Elevator effectiveness is such that it is possible to stop the pitch-up promptly and reduce the angle of attack, and

c. At no point during the stall, stall approach, or recovery does any portion of the airplane exceed structural limit loads.

The requirements apply to all stalls resulting from rates of speed reduction up to 4 knots per second. The stall characteristics will be considered unacceptable if a spin is likely to result.

RELATED MIL-F-8785 PARAGRAPHS

3.6.4

DISCUSSION

The requirements of 3.4.1.3 are an expansion of 3.6.4 of MIL-F-8785 to include accelerated stalls and to treat the pitch-up problem in more detail. Otherwise, the limits on uncontrollable rolling or pitching seem to have withstood the passage of time. It seemed rational to add a limit on uncontrollable yawing, in view of problems that have been experienced with post-stall gyrations and incipient spins.

Application at rates of speed reduction up to 4 kt/sec is an attempt to account for any deleterious effects of unsteady aerodynamics on stall characteristics.

3.4.2.4 STALL RECOVERY AND PREVENTION

REQUIREMENT

3.4.2.4 Stall recovery and prevention. It shall be possible to prevent the complete stall by moderate use of the controls at the onset of the stall warning. It shall be possible to recover from a complete stall by use of the elevator, aileron, and rudder controls with reasonable forces, and to regain level flight without excessive loss of altitude or buildup of speed. Throttles shall remain fixed until speed has begun to increase when an angle of attack below the stall has been regained. In the straight-flight stalls of 3.4.2.1, with the airplane trimmed at a speed not greater than $1.4V_S$ and with a speed reduction rate of at least 4.0 knots per second, elevator control power shall be sufficient to recover from any attainable angle of attack.

3.4.2.4.1 One-engine-out stalls. On multiengine airplanes, it shall be possible to recover safely from stalls with the critical engine inoperative. This requirement applies with the remaining engines at up to thrust for level flight at $1.4V_S$, but these engines may be throttled back during recovery.

RELATED MIL-F-8785 PARAGRAPHS

3.6.4.1

DISCUSSION

Paragraph 3.4.2.4 is essentially the same as 3.6.4.1 of MIL-F-8785. While the requirement is still qualitative, it is more explicit about allowable use of controls. As in 3.4.2.3, a range of rates of speed reduction is specified. It is particularly important that devices such as pitch-up inhibitors have enough anticipation in rapidly entered stalls.

In the second sentence no attempt is being made to legislate the primary flight control technique to be used in recovering from a complete stall. In fact, in some airplanes an attempt to use rudder and aileron control before regaining a speed margin from stall (through use of the elevator control) is an invitation to precipitate a spin. All the flight controls should be of sufficient effectiveness and power, however, that their use in a safe, orderly manner will result in prompt recovery from the stall.

From the wording of the last sentence of 3.4.2.4, compliance with that part of the requirement can be demonstrated by other means if flight test is judged too dangerous. Note that here high rates of entry ($\dot{V} \geq 4$ kt/sec) are required, compared to the lower rates ($\dot{V} \leq 4$ kt/sec) in 3.4.2.3.

Paragraph 3.4.2.4.1 is a new requirement which was added because some multi-engine airplanes have exhibited rather violent, unacceptable rolling tendencies in engine-out stalls. FAR Part 25 (Reference A6) and Part 23 (Reference A19) each have a similar requirement.

3.4.3 SPIN RECOVERY

REQUIREMENT

3.4.3 Spin recovery. If spin demonstration is required by MIL-S-25015 or MIL-D-8708, consistent prompt recoveries shall be possible from all modes of incipient and fully developed erect and inverted spins, using controls as required by the referenced specifications. If such controls include a special spin recovery device, that device shall satisfy the following additional requirements: required pilot action shall be easy, consistent, and simple; the device shall be immediately reusable for several spins on the same flight. Recovery control forces shall not exceed 250 pounds rudder, 75 pounds elevator, or 35 pounds aileron.

RELATED MIL-F-8785 PARAGRAPHS

3.5.1

DISCUSSION

The requirements of 3.5.1 of MIL-F-8785 have been reworded and expanded somewhat. Since spins are not normal operational maneuvers for most airplanes, auxiliary recovery devices are sometimes permitted; but on operational airplanes they must not be one-shot devices such as spin-recovery parachutes. Since spins are normal maneuvers for trainers, however, auxiliary devices would not normally be approved for trainers.

It is good, recommended practice to require spin analysis, spin-model tests, or both, even when no flight demonstration is required. Large airplanes too are susceptible to inadvertent spins, so pilots should know at least the characteristics in spin entry and the effectiveness of recovery techniques.

3.4.4 ROLL-PITCH-YAW COUPLING

REQUIREMENT

3.4.4 Roll-pitch-yaw coupling. For Class I and IV airplanes in rudder-pedal-free, elevator-control-fixed, maximum-performance rolls through 360 degrees, entered from straight flight or from turns, pushovers, or pullups ranging from 0g to 0.8 n_L, the resulting yaw or pitch motions and sideslip or angle of attack changes shall neither exceed structural limits nor cause other dangerous flight conditions such as uncontrollable motions or roll autorotation. During combat-type maneuvers involving rolls through angles up to 360 degrees, the yawing and pitching shall not be so severe as to impair the tactical effectiveness of the maneuver. These requirements define Level 1 and Level 2 operation. For Class II and Class III airplanes, these requirements apply in rolls through 120 degrees.

RELATED MIL-F-8785 PARAGRAPHS

3.5.7

DISCUSSION

These requirements are similar to those of 3.5.7 of MIL-F-8785. Since the maneuver is intended to expose any inertial coupling problems, it should be a reasonably violent maneuver. For this reason, the maneuver is to be done with full lateral control, the rudder pedals free, and the elevator stick fixed. If other control inputs are found to be more critical, these too should be investigated.

The maximum load factor specified for maximum-performance rolls was changed from 2/3 n_L in MIL-F-8785 to 0.8 n_L, in order to make the load factor compatible with the accelerated roll maneuvers required by the structures specification (MIL-A-8861). The maneuvers are still not entirely compatible with the structural requirements, however, because the bank angles required by the accelerated rolls of MIL-A-8861 are limited to 180 degrees. In Reference A1, the bank angle change specified for combat-type rolls was increased from 180 degrees to 360 degrees. This change was made because AFFTC pilots pointed out that 360 degree accelerated rolls are currently used as combat maneuvers.

Paragraph 3.5.7 of MIL-F-8785 did not specify which airplane classes the requirements applied to, but surely the intent was not to call for 180° rolls in transport-type airplanes. The maneuvers required for Classes II and III were therefore limited to 120 degree rolls, which are more typical of extreme maneuvers for low-to-medium-load-factor airplanes.

The dynamics of the pitch and yaw coupling associated with rapid rolls are complex and nonlinear. In general, the dynamics involve interactions among the airplane inertia properties, aerodynamic properties, and the kinematics of the rolling motion. Because of the complexities involved, no attempt has been made in this discussion to explain the mechanism of the various types of pitch-roll-yaw coupling. Instead, the reader is referred to the references in the L section of the Bibliography. Also, References B42 and B71 give brief qualitative explanations of the subject.

3.4.5 CONTROL HARMONY AND CONTROL FORCE COORDINATION

REQUIREMENT

3.4.5 Control harmony. The elevator and aileron force and displacement sensitivities and breakout forces shall be compatible so that intentional inputs to one control axis will not cause inadvertent inputs to the other.

3.4.5.1 Control force coordination. The cockpit control forces required to perform maneuvers which are normal for the airplane should have magnitudes which are related to the pilot's capability to produce such forces in combination. The following control force levels are considered to be limiting values compatible with the pilot's capability to apply simultaneous forces:

Type Control	Elevator	Aileron	Rudder
Center-stick	50 pounds	25 pounds	175 pounds
Wheel	75 pounds	40 pounds	175 pounds

RELATED MIL-F-8785 PARAGRAPHS

6.5

DISCUSSION

These paragraphs represent a complete rework of 6.5 of MIL-F-8785, since there are several aspects of 6.5 which seem undesirable. In the first place it would be preferable that no rudder inputs at all be required for rolling pullouts. Secondly, the maneuver described is too precisely defined for the very qualitative tone of the paragraph.

Control harmony has several aspects, which have been stated in the new requirements. One problem is that the elevator and aileron forces must be in the proper ratio for gross unsymmetrical maneuvers, to enhance proper coordination of the maneuver. Another problem is that unless the elevator and aileron control sensitivities and breakout forces are properly matched, intentional inputs to one control can result in inadvertent inputs to the other. For example, many heavy airplanes with unboosted controls have had aileron forces which were much too high with respect to the elevator forces. As a result, it was difficult to control pitch attitude accurately when rolling rapidly into a turn. The intent of 3.4.5 is to prevent this situation.

A third aspect of control harmony is that the pilot cannot apply forces simultaneously to all three controls which are as large as those forces which can be applied to one control at a time. Paragraph 3.4.5.1 is concerned with this aspect of the problem. Since there was no indication that the numbers should be changed, the values were taken directly from MIL-F-8785.

3.4.6 BUFFET

REQUIREMENT

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

RELATED MIL-F-8785 PARAGRAPHS

3.1.3

DISCUSSION

This requirement is extracted from 3.1.3 of MIL-F-8785. The need for such a requirement seems obvious, and it is not possible to be more definitive at the present time.

3.4.7 RELEASE OF STORES

3.4.8 EFFECTS OF ARMAMENT DELIVERY AND SPECIAL EQUIPMENT

REQUIREMENT

3.4.7 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.

3.4.8 Effects of armament delivery and special equipment. Operation of moveable parts such as bomb bay doors, cargo doors, armament pods, refueling devices, and rescue equipment, or firing of weapons, release of bombs, or delivery or pickup of cargo shall not cause buffet, trim changes, or other characteristics which impair the tactical effectiveness of the airplane under any pertinent flight condition. These requirements shall be met for Levels 1 and 2.

RELATED MIL-F-8785 PARAGRAPHS

3.1.7, 3.1.6

DISCUSSION

Paragraph 3.4.7 is a rewording of 3.1.7 of MIL-F-8785. It is a necessary catchall requirement. Because of the variety of possibilities it must be left qualitative.

Paragraph 3.4.8 is an expansion of 3.1.6 of MIL-F-8785. The slight difference in tone between 3.4.7 and 3.4.8 is the result of design and operational experience.

3.4.9, 3.4.10 FAILURES

REQUIREMENT

3.4.9 Transients following failures. The airplane motions following sudden airplane system or component failures shall be such that dangerous conditions can be avoided by pilot corrective action. A realistic time delay between the failure and initiation of pilot corrective action shall be incorporated when determining compliance. 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.

3.4.10 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) are excepted. The crew member concerned shall be provided with immediate and easily interpreted indications whenever failures occur that require or limit any flight crew action or decision.

RELATED MIL-F-8785 PARAGRAPHS

3.2.5

DISCUSSION

These paragraphs are an expansion of the last sentence of 3.2.5 of MIL-F-8785. The requirements should be self-explanatory, and the need self-evident.

A pilot, especially if he is not alert for failures, may not detect a failure or adapt immediately. In some cases his consequent inability to adapt can result in a pilot-airframe closed-loop instability, even if the airplane itself remains stable (Reference J18). Allowance for this phenomenon should be made in deciding the suitability of any required pilot corrective action. The required failure indications depend on operational rules. Consistent maintenance and check out capability and rules should be established. The flight control system specification should also be consulted, as should flight safety, maintenance, and reliability requirements.

The kinds of failures that might be excepted from the requirement of 3.4.10 are indicated in the discussion of 3.1.6.1.

3.5 - CHARACTERISTICS OF THE PRIMARY FLIGHT CONTROL SYSTEM

3.5 - CHARACTERISTICS OF THE PRIMARY FLIGHT CONTROL SYSTEM

3.5 CHARACTERISTICS OF THE PRIMARY FLIGHT CONTROL SYSTEM

3.5.1 GENERAL

REQUIREMENT

3.5.1 General characteristics. As used in this specification, the term primary flight control system includes the elevator, aileron and rudder controls, stability augmentation systems, and all mechanisms and devices that they operate. The requirements of this section are concerned with those aspects of the primary flight control system which are directly related to flying qualities. These requirements are in addition to the requirements of the applicable control system design specification, e.g., MIL-F-9490 or MIL-C-18244.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

Section 3.5 deals with the primary flight controls, i.e., the elevator, aileron, and rudder systems in a broad sense. The requirements include the various control system and stability augmentation system requirements which were scattered throughout MIL-F-8785, many of which were essentially duplications of each other.

In addition to changes which were primarily organizational, several new requirements have been added and control augmentation has been recognized. The major additions are in the paragraphs dealing with dynamic characteristics.

Note the expanded scope of the primary flight control system as defined in 3.5.1. Although the MIL-F-9490C(USAF) definition is somewhat different from this, the next revision to that specification is planned to include critical augmentation functions as part of the primary flight controls.

Autopilot specifications often contain additional flying qualities requirements.

3.5.2 MECHANICAL CHARACTERISTICS

REQUIREMENT

3.5.2 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 3.5.2.1 through 3.5.2.4. Meeting these separate requirements, however, will not necessarily ensure that the overall system will be satisfactory; the mechanical characteristics must be compatible with the non-mechanical portions of the control system and with the airframe dynamic characteristics.

RELATED MIL-F-8785 PARAGRAPHS

6.11

DISCUSSION

Some of the discussion in 6.11 of MIL-F-8785 has been rewritten and incorporated into 3.5.2 to form an introduction to the following four paragraphs dealing with mechanical characteristics.

3.5.2.1 CONTROL CENTERING AND BREAKOUT FORCES

REQUIREMENT

3.5.2.1 Control centering and breakout forces. Longitudinal, lateral, and directional controls should exhibit positive centering in flight at any normal trim setting. Although absolute centering is not required, the combined effects of centering, breakout force, stability, and force gradient shall not produce objectionable flight characteristics, such as poor precision-tracking ability, or permit large departures from trim conditions with controls free. Breakout forces, including friction, preload, etc., shall be within the limits of table XII. The values in table XII refer to the cockpit control force required to start movement of the control surface in flight for Levels 1 and 2; the upper limits are doubled for Level 3.

Table XII. Allowable Breakout Forces, Pounds

Control		Classes I, II-C, IV		Classes II-L, III	
		min	max	min	max
Elevator	Stick	1/2	3	1/2	5
	Wheel	1/2	4	1/2	7
Aileron	Stick	1/2	2	1/2	4
	Wheel	1/2	3	1/2	6
Rudder		1	7	1	14

Measurement of breakout forces on the ground will ordinarily suffice in lieu of actual flight measurement, provided that qualitative agreement between ground measurement and flight observation can be established.

RELATED MIL-F-8785 PARAGRAPHS

3.2.1, 3.2.1.1, 3.2.1.2, 3.4.16.8

DISCUSSION

These requirements are primarily a restatement of 3.2.1, 3.2.1.1, and 3.2.1.2 of MIL-F-8785, with the class breakdown of 3.2.1 altered somewhat because of the new format.

With absolute centering, a cockpit control will always return exactly to its trim position when released. Positive centering is a tendency to return: upon release, the control will move toward the trim position but friction may prevent absolute centering.

Although there are many indications that breakout forces should be a function of control force sensitivity (angular acceleration per pound of force) or some other force gradient, this approach was not used. The main reason for this is that there are not enough data (relating breakout forces and sensitivity) to justify the additional complication, especially when measurement of breakout forces is usually quite imprecise anyway.

There were several comments from airplane manufacturers stating that the rudder breakout forces should be increased, at least for Classes II-C and IV. A rather detailed study of the problem was conducted in Reference A14, however, which concluded that the present requirements were quite adequate. In view of this study, the rudder breakout forces were not increased.

3.5.2.2 COCKPIT CONTROL FREE PLAY

REQUIREMENT

3.5.2.2 Cockpit control free play. The free play in each cockpit control, that is, any motion of the cockpit control which does not move the control surface in flight, shall not result in objectionable flight characteristics, particularly for small-amplitude control inputs.

RELATED MIL-F-8785 PARAGRAPHS

3.2.4

DISCUSSION

This requirement is a straightforward rewrite of 3.2.4 of MIL-F-8785. No numerical value has yet been found that appears generally adequate. The allowable free play would seem to be a function of control-deflection sensitivity (angular acceleration per inch or degree of movement) and possibly control-force sensitivity as well.

3.5.2.3 RATE OF CONTROL DISPLACEMENT

REQUIREMENT

3.5.2.3 Rate of control displacement. The ability of the airplane to perform the operational maneuvers required of it shall not be limited in the atmospheric disturbances specified in 3.7 by control surface deflection rates. For powered or boosted controls, the effect of engine speed and the duty cycle of both primary and secondary controls together with the pilot control techniques shall be included when establishing compliance with this requirement.

RELATED MIL-F-8785 PARAGRAPHS

3.2.3, 3.7.1

DISCUSSION

Paragraph 3.2.3 and the last sentence of 3.7.1 of MIL-F-8785 have been included in this paragraph. An additional statement was included (on the basis of comments from aircraft manufacturers) to point out that auxiliary hydraulic devices may use up significant portions of the available hydraulic power during critical phases of the mission. For example, actuation of landing gear, flaps, slats, etc., during the landing approach when the engines are operating at relatively low power settings could drain enough hydraulic power to make it difficult for the pilot to make a safe approach, especially in turbulence. In other flight conditions with less auxiliary demand or higher engine thrust, however, that same hydraulic system might be more than adequate.

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 D7, E5, and D12). This technique is likely to be used when the short-period frequency is less than the minimum specified or if the phugoid is unstable.

While specific disturbances are listed, the evaluation remains somewhat qualitative.

The "required operational maneuvers" are commensurate with the particular level of flying qualities under consideration. The maneuvers required in Level 3 operation, for example, will normally be less precise and more gradual than for Level 1 and 2 operation. In some cases this may result in lower demands on control authority and rates for Level 3 operation. Note, however, that when the handling characteristics of the airplane are near the Level 3 limits, increased control activity may occur, even though the maneuvers are more gradual.

3.5.2.4 ADJUSTABLE CONTROLS

REQUIREMENT

3.5.2.4 Adjustable controls. When a cockpit control is adjustable for pilot physical dimensions or comfort, the control forces defined in 6.2 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.

RELATED MIL-F-8785 PARAGRAPHS

3.2.2

DISCUSSION

This paragraph is essentially the same as 3.2.2 of MIL-F-8785. Some such requirement is needed, and this one appears reasonable in all respects.

3.5.3 DYNAMIC CHARACTERISTICS

REQUIREMENT

3.5.3 Dynamic characteristics. The response of the control surfaces in flight shall not lag the cockpit control force inputs by more than the angles shown in table XIII, for frequencies equal to or less than the frequencies shown in table XIII.

Table XIII. Allowable Control Surface Lags

Level	Allowable Lag ~ deg	
	Category A and C Flight Phases	Category B Flight Phases
1 and 2	30	45
3	60	

Control	Upper Frequency ~ rad/sec
elevator	$\omega_{n_{sp}}$
rudder & aileron	ω_{n_d} or $1/\tau_R$ (whichever is larger)

The lags referred to are the phase angles obtained from steady-state frequency responses, for reasonably large-amplitude force inputs. The lags for very small control-force amplitudes shall be small enough that they do not interfere with the pilot's ability to perform any precision tasks required in normal operation.

RELATED MIL-F-8785 PARAGRAPHS

6.11, 3.7.1

DISCUSSION

This paragraph is an expansion of the last sentence of 6.11 of MIL-F-8785, which is essentially a more precise way of expressing the last sentence of 3.7.1 of MIL-F-8785. In other words, the pilot should get prompt control response when he applies a force at the stick.

In some cases, compliance with 3.5.3 (and 3.5.3.1) can be demonstrated by obtaining the required frequency responses on the ground. In most cases, however, the effects of control-surface aerodynamic loads, SAS feedback, or control system mass unbalance will make flight testing necessary. If flight tests are required, they can be combined with the tests of 3.2.2.3.1. The techniques for obtaining in-flight frequency responses are briefly discussed in Appendix IVF.

In a simple, approximately linear control system, the lag at reasonably large input amplitudes might be estimated from the time delay in surface response to a rapid cockpit-control input, multiplied by the appropriate upper frequency or inverse time constant, and converted to degree measure.

There are two basic sources of lags in the control system which can cause problems. The first is the nonlinear effect of friction and free play. Normally, friction and free play introduce appreciable phase lag only at low control input amplitudes, and therefore cause the most trouble during small precision maneuvers such as tracking. The second source of lags is introduced by the linear dynamics of the control system.

In the 1940's, control systems were generally quite fast; and the research efforts of the NACA and others were aimed primarily at the effects of friction and free play. If these effects were kept under control, the control system was usually satisfactory. The effects of friction and free play are hopefully limited by 3.5.2.1, 3.5.2.2, and the parts of 3.5.3 and 3.5.3.1 dealing with very small force inputs.

With the advent of fully powered, highly augmented control systems, the lags introduced by the linear dynamics of the control system have become very important.

There are suggestions from several sources that an overly sensitive airplane can be improved by filtering the pilot's input to the control surfaces. References B59, J32, and J34, for example, indicate that stick-force command filtering having corner frequencies near or below $\omega_{n_{sp}}$ can significantly improve the response characteristics of an airframe having low ζ_{sp} or high $\omega_{n_{sp}}$.

A recent systematic in-flight evaluation of the effects of control system dynamics on longitudinal flying qualities is the experiment of Reference J59. The results of this experiment are rather startling and are in direct contradiction to the ideas discussed above. That is, even relatively small lags in the control system can cause control difficulties and moderate lags can cause very pronounced pilot-induced oscillations. In fact, the trend indicated is that the allowable lag <u>decreases</u> with increasing $\omega_{n_{sp}}$. It is interesting to note that the pronounced piloting difficulties observed in this experiment did not occur in a similar experiment with the same airplane stationary on the ground (see Reference J2).

The pilot rating data of Reference J59 show very good correlation with either of two somewhat equivalent control system parameters. The first is an equivalent time delay divided by the period of the short-period mode. The second is the phase lag at $\omega = \omega_{n_{sp}}$ from a frequency response of $[\delta_e(j\omega)/F_s(j\omega)]$ The latter parameter seems to be more useful for both design and test purposes, so the requirement of 3.5.3 is stated in this form.

The results of Reference J59 are presented in Figures 1 through 3. Some of the data are for the landing approach and the rest are for Category A Flight Phases. No attempt was made to plot the approach and Category A data separately because there is no significant difference between the two sets. Pilot B used the standard CAL rating scale (see the discussion of 1.5). Pilot H used the standard CAL rating scale; but in addition, he rated the susceptibility of the airplane to pilot-induced oscillations using the following rating scale:

PIO TENDENCY RATING SCALE (Reference D3)

Description	Numerical Rating
No tendency for pilot to induce undesirable motions.	1
Undesirable motions tend to occur when pilot initiates abrupt maneuvers or attempts tight control. These motions can be prevented or eliminated by pilot technique.	2
Undesirable motions easily induced when pilot initiates abrupt maneuvers or attempts tight control. These motions can be prevented or eliminated but only at sacrifice to task performance or through considerable pilot attention and effort.	3
Oscillations tend to develop when pilot initiates abrupt maneuvers or attempts tight control. Pilot must reduce gain or abandon task to recover.	4
Divergent oscillations tend to develop when pilot initiates abrupt maneuvers or attempts tight control. Pilot must open loop by releasing or freezing the stick.	5
Disturbance or normal pilot control may cause divergent oscillation. Pilot must open control loop by releasing or freezing the stick.	6

The phase angle limit of -30 degrees for Category A and C flight Phases (Levels 1 and 2) was determined from Figures 1 and 2 for a pilot rating of 3.5. It might seem logical to allow a value of -50 to -60 degrees for Level 2, corresponding to a pilot rating of 6.5. This was not done because the rating degradation for lags of 50 or 60 degrees was largely due to PIO tendencies, as can be seen from the associated PIO ratings of 3 to 4 (from Figure 3). The decision was made not to allow this type of behavior for Level 1 or 2 operation. The pilot ratings of Figures 1 and 2 are little help in establishing limits for Category B Flight Phases because the degradations in rating are primarily due to difficulties in performing Category A Flight Phases. The PIO ratings of Figure 3 can be used to establish limits for Category B, however. When a PIO rating of 3 is given to a configuration, it means that the pilot is beginning to have control problems when "just flying around." A PIO rating of 2, on the other hand, seems adequate when precise control is not needed. The limit for Category B Flight Phases, Levels 1 and 2, was therefore established by using a PIO rating of 2.5. From Figure 3, this yields a phase lag of 45 degrees. The limit for Level 3 could be set by using a pilot rating of 9. Since PIO's are involved, however, a PIO rating of 4 was used as the limit for

Level 3, since this seemed compatible with the requirement of 3.2.2.3. Using Figure 3, a PIO rating of 4 yields a phase lag of 60 degrees.

The same phase lags allowable at $\omega = \omega_{n_{sp}}$ for elevator control were applied rather arbitrarily to aileron and rudder control at $\omega = 1/\tau_R$ and $\omega = \omega_{n_d}$, in the absence of definitive data.

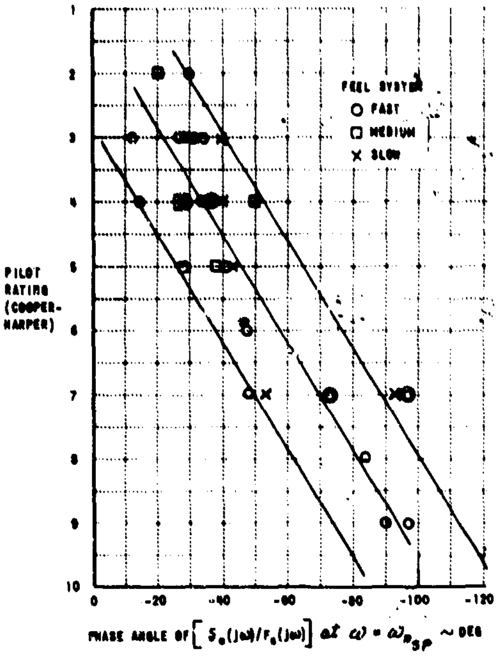


Figure I (3.5.3)
CATEGORY A AND C FLIGHT PHASES
(T-33, PILOT B, REFERENCE J59)

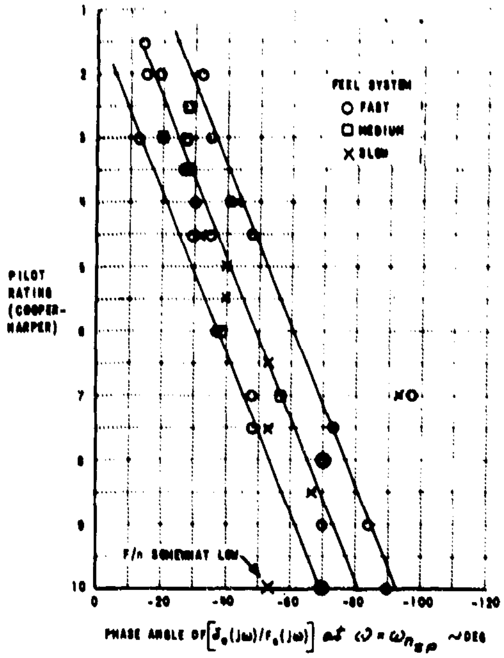


Figure 2 (3.5.3)
CATEGORY A AND C FLIGHT PHASES
(T-33, PILOT H, REFERENCE J59)

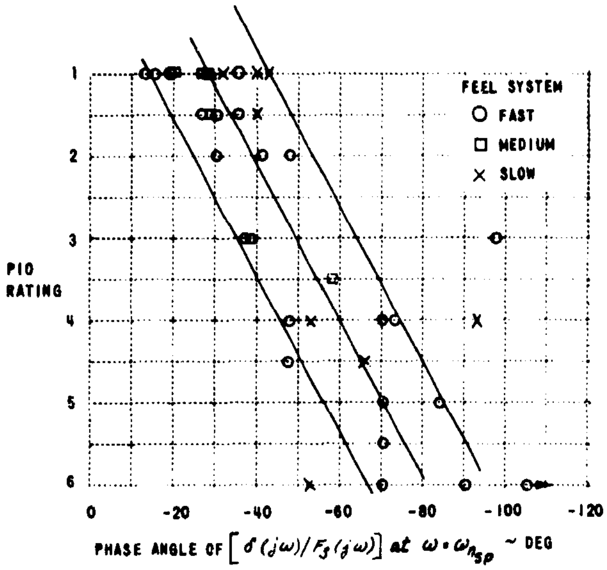


Figure 3 (3.5.3)
CATEGORY A AND C FLIGHT PHASES (T-33, PILOT H, REFERENCE J59)

3.5.3.1 CONTROL FEEL

REQUIREMENT

3.5.3.1 Control feel. In flight, the cockpit-control deflection shall not lead the cockpit-control force for any frequency or force amplitude. This requirement applies to the elevator, aileron, and rudder controls. In flight, the cockpit-control deflection shall not lag the cockpit-control force by more than the angles listed in 3.5.3, for frequencies equal to or less than those listed in 3.5.3, for reasonably large force inputs. The lags for very small control-force amplitudes shall not interfere with the pilot's ability to perform precision tasks required in normal operation.

RELATED MIL-F-8785 PARAGRAPHS

6.11

DISCUSSION

The first requirement of 3.5.3.1 is aimed at normal-acceleration bobweights and other devices which feed back the airplane's responses into the feel system. When the feel system response is altered by such devices so that the control force lags the control motion (the reverse situation occurs with normal control systems), the effects are undesirable. This factor is discussed in more detail under 3.2.2.3.

The other requirements of 3.5.3.1 are an application of 3.5.3 to the feel system alone. This was done because lags in the feel system can cause control problems (see Reference J59). It is possible for a stability-augmented airplane to have excessive feel system lags while still meeting the requirements of 3.5.3.

3.5.3.2 DAMPING

REQUIREMENT

3.5.3.2 Damping. All control system oscillations shall be well damped, unless they are of such an amplitude, frequency, and phasing that they do not result in objectionable oscillations of the cockpit controls or the airframe during abrupt maneuvers and during flight in the atmospheric disturbances specified in 3.7.3 and 3.7.4.

RELATED MIL-F-8785 PARAGRAPHS

3.3.5.1, 3.5.3

DISCUSSION

This paragraph is a combination of the requirements of 3.3.5.1 and 3.5.3 of MIL-F-8785. Many airplanes, including airplanes using stability augmentation, have second-order control-system modes which exhibit low damping. If these modes have natural frequencies considerably above the airframe natural frequencies, or if there are control-system zeros near the poles, the low damping is often not even noticeable to the pilot. There are other cases, however, where the low control-system damping is very objectionable or even dangerous.

Normal-acceleration bobweights have a tendency to cause trouble in this area, especially if the control system employs an irreversible elevator actuator. In this situation, the basic control system usually has very low damping ($2\zeta\omega_n$). Without a bobweight, this is normally no problem because the natural frequency (ω_n) of the control system is much higher than $\omega_{n_{sp}}$. When a bobweight is added, however, the natural frequency of the control system is significantly reduced because of the increased control system inertia. On several airplanes, the bobweight used was large enough to reduce the control-system natural frequency to the point that rapid control movements caused objectionable high-frequency control-system oscillations.

Certain types of adaptive control systems employ modes having very low damping. If the natural frequency of such a mode is too low, objectionable control-system oscillations may result.

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.

RELATED MIL-F-8785 PARAGRAPHS

3.2.5

DISCUSSION

This paragraph is simply a restatement of 3.2.5 of MIL-F-8785. Although it appears to say very little at first glance, it is reminder to the designer to make sure that the introduction of a control-system device to improve a particular undesirable characteristic of the airplane does not also introduce undesirable side effects. Unpleasant side effects are often caused by the use of the more commonplace "simple" augmentation devices. The following is a discussion of some devices which have been used to remedy obvious problems, without proper investigation of the effects on the dynamic characteristics of the airframe and control system.

Normal-acceleration bobweights introduced to meet steady-state stick force per g requirements can contribute to pilot-induced oscillations. Bobweights have a tendency to increase the control-system inertia, decrease the control-free short-period damping ratio, and provide poor control feel characteristics in abrupt maneuvers, turbulence, and taxiing. More detail is given in the discussions of paragraphs 3.2.2.3.1, 3.5.3.1, and 3.5.3.2.

Devices which apply a steady unbalancing force to the control system are often employed to correct an unstable variation of elevator control force with airspeed. As increasing amounts of nose-down unbalancing force are added, the gradient becomes first zero and then stable. When the slope becomes stable, a lightly damped phugoid mode will appear (control free). As the force gradient is made even more stable, the phugoid frequency will increase and the damping ratio may increase also. However, as more and more unbalancing moment is a added, the phugoid damping ratio will usually decrease again, often becoming negative. In addition, the high phugoid frequency may be undesirable by itself, especially in turbulence (see Reference E5). From the foregoing discussion, it can be seen that just the right amount of unbalancing force must be applied to the control system to avoid ending up with a poorer airplane than the basic one. Since the proper amount of unbalancing force changes rapidly with airspeed, the use of a fixed mechanical downspring to improve the force gradients at moderate or high speed will often result in negative phugoid damping in the landing approach. It should also be pointed out that a bobweight which is not statically balanced with an "upspring" will behave like a fixed downspring.

Spring interconnects between the rudder and aileron controls introduced to meet steady sideslip requirements or to improve coordination in turn entries can result in objectionable control forces during takeoffs and landings. In addition, a fixed spring can accomplish the desired effect only over a very limited angle-of-attack range.

Yaw-rate-sensing dampers introduced to improve Dutch roll damping can cause heavy rudder forces in steady turns. Acceptable wash-out may be difficult to devise if ω_{n_d} is very low. A yaw damper also may also aggravate $\Delta\beta_{max}$ while improving ζ_d and even the β response to gusts (Reference J4).

A pitch rate damper with low authority will tend to saturate in steady turns. Up to the point of saturation, it will also increase the stick force per g. These effects are most prominent at low speed where, kinematically, $\frac{q}{n}$ is high. To compensate, a high-pass filter is sometimes used to wash out the steady pitch-rate signal. Such augmentation can be effective; but short-period motion, maneuvering stability, and actuator dynamics should all be examined over the entire speed-altitude-load factor spectra of the flight envelopes to assure that there is no excessive degradation in any parameter.

Improper location of response sensors for augmentation systems can result in objectionable dynamic characteristics in rapid maneuvers. The location of accelerometers is especially important, and the location of any type of sensor can be critical if structural dynamics are involved.

The C^* criterion of Reference D11 has recently received considerable attention in the design of SAS systems. The criterion is in the form of time-history or frequency-response envelopes on a weighted sum of pitch-rate and n at the pilot's station. This criterion was developed to facilitate the design of SAS systems employing pitch-rate and normal-acceleration feedback. These feedbacks can indeed be effective in improving the short-period dynamics, since n feedback stiffens $\omega_{n_{sp}}$ and $\dot{\theta}$ feedback improves ζ_{sp} (there is also a strong influence on the phugoid mode). The C^* criterion itself, however, is not an adequate substitute for the short-period requirements of 3.2.2.1.1 and 3.2.2.1.2 or the control system requirements of 3.5.3 and 3.5.3.1 (see References D56, F80, and J75).

At high angle of attack, stability augmentation can be destabilizing. For example a roll damper can induce spinning by actuating the ailerons in a stall approach, where a pilot would be particularly careful not to command large or abrupt aileron inputs.

Unless special precautions are taken, series stability-augmentation signals to a common valve might cause unwanted cockpit-control movement at times when primary-control rate command "bottoms" the valve.

As stated in 6.4, "Changes of mechanical gearings and stability augmentation gains in the primary flight control system are sometimes accomplished by scheduling the changes as a function of the settings of secondary control devices, such as flaps or wing sweep. This practice is generally acceptable, but gearings and gains normally should not be scheduled as a function of trim control settings since pilots do not always keep airplanes in trim."

Redundant systems may be subject to nuisance warning or disengagement.

These examples are far from being exhaustive, but rather they serve to illustrate the kind and complexity of consideration that must be given to design of stability and control augmentation.

3.5.4.1 PERFORMANCE OF AUGMENTATION SYSTEMS

REQUIREMENT

3.5.4.1 Performance of augmentation systems. Performance degradation of augmentation systems caused by the atmospheric disturbances of 3.7.3 and 3.7.4 and by structural vibrations shall be considered, when such systems are used.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

This requirement was added partially as a result of flight tests of 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.

3.5.4.2 SATURATION OF AUGMENTATION SYSTEMS

REQUIREMENT

3.5.4.2 Saturation of augmentation systems. Limits on the authority of augmentation systems or saturation of equipment shall not result in objectionable flying qualities. In particular, this requirement shall be met during rapid large-amplitude maneuvers, during operation near V_S, and during flight in the atmospheric disturbances of 3.7.3 and 3.7.4.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

This requirement has been introduced 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 ζ_{sp} in light turbulence and for precision tracking tasks, but the nonlinearity of the airplane's response for a pullup due to saturation of the rate damper might be extremely objectionable.

Some requirements of Reference A1 specify a minimum control power (for takeoff, landing, maneuvering flight, roll control, etc.) or a minimum control margin (for sideslip, cross-wind landing, asymmetric thrust, etc.) available to the pilot. Saturation of augmentation must not prevent the safe utilization of that control power or margin for maneuvering and compensating for disturbances.

Although these requirements are necessarily, at this time, qualitative, the airplane must be evaluated in specific turbulence and discrete-gust environments.

3.5.5 FAILURES

REQUIREMENT

3.5.5 Failures. If the flying qualities with any or all of the augmentation devices inoperative are dangerous or intolerable, special provisions shall be incorporated to preclude a critical single failure. 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.

3.5.5.1 Failure transients. With controls free, the airplane motions due to failures described in 3.5.5 shall not exceed the following limits for at least 2 seconds following the failure, as a function of the Level of flying qualities after the failure transient has subsided:

Level 1 (after failure)	±0.05g normal or lateral acceleration at the pilot's station and ±1 degree per second in roll
Level 2 (after failure)	±0.5g at the pilot's station, ±5 degrees per second roll, and the lesser of ±5 degrees sideslip or the structural limits
Level 3 (after failure)	No dangerous attitude or structural limit is reached, and no dangerous alteration of the flight path results from which recovery is impossible.

3.5.5.2 Trim changes due to failures. The control forces required to maintain attitude and zero sideslip for the failures described in 3.5.5 shall not exceed the following limits for at least 5 seconds following the failure:

Elevator ------------------ 20 pounds
Aileron ------------------ 10 pounds
Rudder ------------------ 50 pounds

RELATED MIL-F-8785 PARAGRAPHS

3.7.3, 3.7.3.1, 3.7.3.2

DISCUSSION

The requirements of 3.7.3, 3.7.3.1, and 3.7.3.2 of MIL-F-8785 are a mixture of limitations on transients and trim changes. Since there are different reasons for having each, requirements on transients have been included in 3.5.5.1 and trim changes in 3.5.5.2.

Transients are the dynamic responses of the airplane to a control system failure. The purpose of a requirement in this area is to ensure that the short-term response of the airplane does not get out of hand before the pilot can react. The requirement of 3.5.5.1 specifies that the responses be measured with the controls free. This technique seems to best simulate the situation in which the pilot is most likely to be caught off-guard by the transient. The actual numerical limits of 3.5.5.1 are modifications to the requirements of 3.7.3 of MIL-F-8785, in an attempt to make the requirements more meaningful. The Level 1 requirement is quite stringent, the intent being that the pilot should hardly notice such a failure. The larger the transient, the less frequently it should occur; tying the transient magnitude to the Level after failure accomplishes this. (Small transients upon reversion to poor flying qualities are quite desirable but hardly seem feasible in general.) If there should turn out to be no other way to meet the Level 1 requirement than to add considerable complexity to the flight control system, the procuring activity should consider relaxing the requirement for that particular design.

In addition to the controls-free response transients caused by a transfer, it is necessary to have limits on the control forces required to minimize the airplane response. To this end, the requirements of 3.7.3.1 and 3.7.3.2 of MIL-F-8785 have been generalized to apply for all Flight Phases. The elevator and aileron forces are unchanged. No distinction has been made between land- and carrier-based airplanes in the rudder forces because there seems to be no need. It seems reasonable to state a time limit during which this requirement applies. Two seconds generally should be time enough for the pilot to detect a significant transient and react, and it should be possible to retrim after 5 seconds.

3.5.6 TRANSFER TO ALTERNATE CONTROL MODES

REQUIREMENT

3.5.6 Transfer to alternate control modes. The transient motions and trim changes resulting from the intentional engagement or disengagement of any portion of the primary flight control system by the pilot shall be small and, gradual enough that dangerous flying qualities never result.

3.5.6.1 Transients. With controls free, the transients resulting from the situations described in 3.5.6 shall not exceed the following limits for at least 2 seconds following the transfer:

Within the Operational Flight Envelope	±0.05g normal or lateral acceleration at the pilot's station and ±1 degree per second roll
Within the Service Flight Envelope	±0.5g at the pilot's station, ±5 degrees per second roll, and the lesser of ±5 degrees sideslip or the structural limit

These requirements apply only for Airplane Normal States.

3.5.6.2 Trim changes. The control forces required to maintain attitude and zero sideslip for the situations described in 3.5.6 shall not exceed the following limits for at least 5 seconds following the transfer:

Elevator ---------------- 20 pounds
Aileron ------------------ 10 pounds
Rudder ------------------ 50 pounds

These requirements apply only for Airplane Normal States.

RELATED MIL-F-8785 PARAGRAPHS.

3.7.3, 3.7.3.1, 3.7.3.2

DISCUSSION

Paragraphs 3.7.3, 3.7.3.1, and 3.7.3.2 of MIL-F-8785 deal with intentional transfers to alternate control modes, as well as failures. The failure aspects have been handled in 3.5.5, 3.5.5.1, and 3.5.5.2. The intentional tranfer aspects were developed in a similar manner and included in 3.5.6, 3.5.6.1, and 3.5.6.2. Application of Levels for intentional actions might be confusing, so the applicable flight envelopes are specified instead.

3.6 · CHARACTERISTICS OF SECONDARY CONTROL SYSTEMS

3.6 - CHARACTERISTICS OF SECONDARY CONTROL SYSTEMS

3.6 CHARACTERISTICS OF SECONDARY CONTROL SYSTEMS

DISCUSSION

Section 3.6 deals with the effects of operation of secondary control devices on flying qualities. Secondary control systems include the trim system, the throttle system, and all control devices which alter the loading or external geometry of the airplane (with the exception of the primary control system). The section includes various requirements on secondary controls which were scattered throughout MIL-F-8785.

The secondary-control paragraphs of MIL-F-8785 were examined for current validity. Most of the changes were made to fit the requirements into the new organizational framework. However, the requirements on longitudinal trim changes have been extensively modified.

3.6.1 TRIM SYSTEM

REQUIREMENT

3.6.1 Trim system. In straight flight, throughout the Operational Flight Envelope the trimming devices shall be capable of reducing the elevator, rudder, and aileron control forces to zero for Levels 1 and 2. For Level 3, the untrimmed cockpit control forces shall not exceed 10 pounds elevator, 5 pounds aileron, and 20 pounds rudder. 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 and MIL-F-18372.

3.6.1.1 Trim for asymmetric thrust. For all multiengine airplanes, it shall be possible to trim the elevator, rudder, and aileron control forces to zero in straight flight with up to two engines inoperative following asymmetric loss of thrust from the most critical factors (3.3.9). This requirement defines Level 1 in level-flight cruise at speeds from the maximum-range speed for the engine(s)-out configuration to the speed obtainable with normal rated thrust on the functioning engine(s). Systems completely dependent on the failed engines shall also be considered failed.

3.6.1.2 Rate of trim operation. Trim devices shall operate rapidly enough to enable the pilot to maintain low control forces under changing conditions normally encountered in service, yet not so rapidly as to cause over-sensitivity or trim precision difficulties under any conditions. Specifically, it shall be possible to trim the elevator control forces to less than ±10 pounds for center-stick airplanes and ±20 pounds for wheel-control airplanes throughout (a) dives and ground attack maneuvers required in normal service operation and (b) level-flight accelerations at maximum augmented thrust from 250 knots or $V_{R/C}$, whichever is less to V_{max} at any altitude when the airplane is trimmed for level flight prior to initiation of the maneuver.

3.6.1.3 Stalling of trim systems. Stalling of a trim system due to aerodynamic loads during maneuvers shall not result in an unsafe condition. Specifically, the longitudinal trim system shall be capable of operating during the dive recoveries of 3.2.3.6 at any attainable permissible n, at any possible position of the trimming device.

3.6.1.4 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 landing flaps, or by the operation of an augmentation device. If an automatic interconnect or augmentation device is used in conjunction with a trim device, provision shall be made to ensure the accurate return of the device to its initial trim position on completion of each interconnect or augmentation operation.

RELATED MIL-F-8785 PARAGRAPHS

3.5.4, 3.5.5, 3.5.6, 3.7.7, 6.4

DISCUSSION

The trim authority requirements of 3.5.4 of MIL-F-8785 were rewritten in the language of the new format. Also included were the trim system failure requirements of 3.5.6 and 3.7.7 of MIL-F-8785. For Level 3 conditions the residual forces which are permitted after trimming are arbitrary, but seem more reasonable than zero. They are small enough to be held for some time on rare occasions. Since engine failures are a unique type of failure, the engine-out requirements of 3.5.4 of MIL-F-8785 were included separately in 3.6.1.1.

The requirements of 3.6.1 come from 6.4 and the last sentence of 3.5.4 of MIL-F-8785. Since the quantitative dive requirement of 3.5.4 of MIL-F-8785 was aimed at operational dives, and since operational dives for many current airplanes are performed at speeds considerably lower than maximum operational speed, the new wording seems appropriate. This quantitative requirement was also expanded to cover level-flight accelerations because some high-performance airplanes can out-accelerate the trim rate.

Paragraph 3.6.1.3 is a new requirement, added at the suggestion of several government and industry sources on the basis of flight experience.

Paragraph 3.6.1.4 is essentially the same as 3.5.5 of MIL-F-8785. It certainly is necessary.

3.6.2 SPEED AND FLIGHT-PATH CONTROL DEVICES

REQUIREMENT

3.6.2 Speed and flight-path control devices. The effectiveness and response times of the fore-and-aft force controls, in combination with the other longitudinal controls, shall be sufficient to provide adequate control of flight path and airspeed at any flight condition within the Operational Flight Envelope. This requirement may be met by use of devices such as throttles, thrust reversers, auxiliary drag devices, and flaps.

RELATED MIL-F-8785 PARAGRAPHS

3.1.8, 6.10

DISCUSSION

The requirements of 3.6.2 are a combination of paragraphs 3.1.8 and 6.10 of MIL-F-8785. The requirements on speed control devices a qualitative in nature due to the complex nature of the problem, but the design of these devices is nevertheless very important.

One consideration in the design of speed control devices is to make sure that the thrust response time is short enough for good go-around (wave-off) capability. For pure jet airplanes, the thrust response is usually very poor at low thrust settings and improves at the higher settings. It may be necessary to provide such airplanes with auxiliary approach drag devices to allow the use of high thrust settings during the power approach.

Another consideration is to ensure that the airspeed can be stabilized in dives at the desired angle and airspeed. A stabilized airspeed in the ground attack maneuver means that the pilot need not worry about control force changes with speed or making sure that the airspeed does not exceed safe limits.

It had been hoped to include more specific, more quantitative requirements. But on the basis of industry comments, more definitive requirements that apply to a wide variety of mechanizations seem not to be feasible at this time for a general flying qualities specification.

3.6.3 TRANSIENTS AND TRIM CHANGES

REQUIREMENT

3.6.3 Transients and trim changes. The transients and steady-state trim changes for normal operation of secondary control devices (such as throttle, flaps, slats, speed brakes, deceleration devices, dive recovery devices, wing sweep, and landing gear) shall not impose excessive control forces to maintain the desired heading, altitude, attitude, rate of climb, speed or load factor without use of the trimmer control. This requirement applies to all in-flight configuration changes and combinations of changes made under service conditions, including the effects of asymmetric operations such as unequal operation of landing gear, speed brakes, slats, or flaps. In no case shall there be any objectionable buffeting or oscillation of such devices. More specific requirements on secondary control devices are contained in 3.6.3.1, 3.6.4, and 3.6.5 and in MIL-F-9490 and MIL-F-18372.

RELATED MIL-F-8785 PARAGRAPHS

3.1.10, 3.2.3, 3.3.18, 3.5.3, 6.4

DISCUSSION

This paragraph is a restatement of the several qualitative requirements contained in 3.1.10, 3.2.3, 3.3.18, 3.5.3, and 6.4 of MIL-F-8785. It was decided not to specify rates of operation directly, since these rates are called out in the control systems specifications.

3.6.3.1 PITCH TRIM CHANGES

REQUIREMENT

3.6.3.1 Pitch trim changes. The pitch trim changes caused by operation of secondary control devices shall not be so large that a peak elevator control force in excess of 10 pounds for center-stick controllers or 20 pounds for wheel controllers is required when such configuration changes are made in flight under conditions representative of operational procedure. Generally, the conditions listed in table XIV will suffice for determination of compliance with this requirement. (For airplanes with variable-sweep wings, additional requirements will be imposed consistent with operational employment of the vehicle.) With the airplane trimmed for each specified initial condition, the peak force required to maintain the specified parameter constant following the specified configuration change shall not exceed the stated value for a time interval of at least 5 seconds following the completion of the pilot action initiating the configuration change. The magnitude and rate of trim change subsequent to this time period shall be such that the forces are easily trimmable by use of the normal trimming devices. These requirements define Level 1. For Levels 2 and 3, the allowable forces are increased by 50 percent.

TABLE XIV

Pitch Trim Change Conditions

	Flight Phase	Initial Trim Condition					Configuration Change	Parameter to be held constant
		Altitude	Speed	Landing Gear	High-lift Devices & Wing Flaps	Thrust		
1	Approach	$h_{o_{min}}$	Normal pattern entry speed	Up	Up	TLF	Gear down	Altitude and airspeed*
2				Up	Up	TLF	Gear down	Altitude
3				Down	Up	TLF	Extend high-lift devices and wing flaps	Altitude and airspeed*
4				Down	Up	TLF	Extend high-lift devices and wing flaps	Altitude
5				Down	Down	TLF	Idle thrust	Airspeed
6			$V_{o_{min}}$	Down	Down	TLF	Extend approach drag device	Airspeed
7				Down	Down	TLF	Takeoff thrust	Airspeed
8	Approach		$V_{o_{min}}$	Down	Down	TLF	Takeoff thrust plus normal clean-up for wave-off (go-around)	Airspeed
9	Takeoff			Down	Take-off	Take-off thrust	Gear up	Pitch attitude
10			Minimum flap-retract speed	Up	Take-off	Take-off thrust	Retract high-lift devices and wing flaps	Airspeed
11	Cruise and air-to-air combat	$h_{o_{min}}$ and $h_{o_{max}}$	Speed for level flight	Up	Up	MRT	Idle thrust	Pitch attitude
12				Up	Up	MRT	Actuate deceleration device	
13				Up	Up	MRT	Maximum augmented thrust	
14			Speed for best range	Up	Up	TLF	Actuate deceleration device	

*Throttle setting may be changed during the maneuver

Notes: - Auxiliary drag devices are initially retracted, and all details of configuration not specifically mentioned are normal for the Flight Phase.

- If power reduction is permitted in meeting the deceleration requirements established for the mission, actuation of the deceleration device in #12 and #14 shall be accompanied by the allowable power reduction.

RELATED MIL-F-8785 PARAGRAPHS

3.3.19

DISCUSSION

The statement of the requirement is basically the same as that of 3.3.19 of MIL-F-8785, with appropriate wording changes to reflect the new format. The allowable elevator forces have been broken down according to type of controller, rather than class. The force values remain appropriate, based on flight experience. The table describing the individual maneuvers to be evaluated has been altered to make the maneuvers more representative of actual practice and to make the table more readable.

In Table IV of MIL-F-8785, maneuvers 1 through 3 are aimed at the trim changes which normally occur in the traffic pattern. Accordingly, maneuvers 1 through 5 of Table XIV, Reference A1, all start with the airplane trimmed in level flight at the normal pattern speed. First the gear is extended, and the pilot holds altitude or adds thrust to hold his altitude and airspeed. Then the flaps are extended, and the pilot holds altitude or adds thrust to hold his altitude and airspeed. Next, the pilot reduces thrust to begin the descent to final approach.

In Table IV of MIL-F-8785, maneuvers 4 and 10 are at a lower speed, as on final approach; they are usually most critical at the minimum approach speed. Accordingly, maneuvers 6 through 8 of Table XIV, Reference A1, all start with the airplane trimmed at $V_{o_{min}}$. At the beginning of the final approach, with the airplane trimmed at the approach speed, the approach drag device might be extended for the final descent. Then with the airplane actually trimmed in the power approach, a go-around (wave-off) might be necessary. During the initial stage of this maneuver, the pilot will probably try to hold airspeed constant.

In Table IV of MIL-F-8785, maneuvers 5 and 6 deal with cleaning up the airplane on takeoff. These maneuvers are probably most critical during maximum-performance takeoffs where the pilot is trying to clear an obstacle. In this type of takeoff, the pilot will probably hold pitch attitude as the gear starts up, then hold airspeed fairly constant during the climb. The gear-up and flaps-up maneuvers are so described in 9 and 10 of Table XIV, Reference A1.

In Table IV of MIL-F-8785, maneuvers 7, 8 and 11 are aimed at rapid speed changes during high-speed flight. Item 9 in Table IV of MIL-F-8785 is a similar maneuver for cruise. For these maneuvers, it seems appropriate to hold pitch attitude constant initially. These four maneuvers are so described in items 11 through 14 of Table XIV, Reference A1.

3.6.4 AUXILIARY DIVE RECOVERY DEVICES

REQUIREMENT

3.6.4 Auxiliary dive recovery devices. Operation of any auxiliary device intended solely for dive recovery shall always produce a positive increment of normal acceleration, but the total normal load factor shall never exceed 0.8 n_L, controls free.

RELATED MIL-F-8785 PARAGRAPHS

3.3.17

DISCUSSION

This requirement is essentially the same as 3.3.17 of MIL-F8785. Special pitching-moment-producing devices are sometimes used for dive recovery on subsonic airplanes when Mach number effects severely reduce elevator effectiveness. The requirement seems reasonable for this type of device.

3.6.5 DIRECT NORMAL-FORCE CONTROL

REQUIREMENT

3.6.5 Direct normal-force control. Use of devices for direct normal-force control shall not produce objectionable changes in attitude for any amount of control up to the maximum available. This requirement shall be met for Levels 1 and 2.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

This new paragraph requires the designer to minimize pitching moments associated with the use of any direct-lift control device, so that the pilot is provided with an essentially pure lift control. The designer may accomplish this end by careful design of the control itself, or by an interconnect to the elevator.

3.7 - ATMOSPHERIC DISTURBANCES

3.7 - ATMOSPHERIC DISTURBANCES

3.7 ATMOSPHERIC DISTURBANCES

3.7.1 USE OF TURBULENCE MODELS

REQUIREMENT

3.7.1 Use of turbulence models. Paragraphs 3.7.2 through 3.7.5 specify a continuous random turbulence model and a discrete turbulence model that shall be used in analyses to determine compliance with those requirements of this Specification that refer to 3.7 explicitly, to assess:

a. The effect of turbulence on the flying qualities of the airplane;

b. The ability of a pilot to recover from the effects of discrete gusts.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

There are several possible uses for turbulence models, as far as flying qualities requirements are concerned. For example, the models might be used in:

(1) Piloted simulation and closed-loop analysis to assess the effects of turbulence on ride qualities, flying qualities, and controllability.

(2) Analysis to determine the open-loop airplane turbulence response (including the effects of stability augmentation and structural mode excitation), for comparison with ride qualities criteria.

(3) Analysis to ensure that the performance of control systems and SAS equipment does not degrade appreciably during flight in turbulence.

(4) Design of control surfaces, and of pilot and SAS authority, to ensure that the airplane has sufficient control effectiveness to be manageable during flight in turbulence.

It became obvious fairly early in the development of Section 3.7, however, that piloted simulation, a most important use for turbulence models, was a difficult subject to write quantitative requirements for. In addition, although some quantitative criteria do exist for assessment of ride qualities and have been applied in specific instances such as AMSA, it seemed premature to introduce general ride qualities criteria. It was decided, therefore, that turbulence models would be presented in MIL-F-8785B, to be used in any analysis and simulation of flying qualities and ride qualities that the contractor performs. Either those models or the ones specified for structural analysis can be used to investigate interactions of structures and flying qualities, as appropriate. The contractor is bound to use the atmospheric disturbances of this section in showing compliance with paragraphs 3.3.4, 3.3.4.1.2, 3.5.3.2, 3.5.4.1, and 3.5.4.2.

The state of the art in describing atmospheric disturbances is, like that in specifying flying qualities, imperfect. Limitations on the validity of the models presented are imposed by:

- The atmospheric data used
- The mathematical description of turbulence and its effects
- The generalizations made for design purposes
- The need to avoid excessive complication
- The possible incompatibility with models used in structural analysis

Also, this work was mostly done before deciding to emphasize the von Karman spectral form. Use of the Dryden form in deriving the requirements should not, it is felt, contribute significant error.

In the discussions of Section 3.7 there are three main purposes. Since this is the first appearance of atmospheric disturbance models in a flying qualities specification, turbulence, the models, and their application have been described in some detail. Then many assumptions are inherent, of varying validity, in deriving quasi-rational requirements; the assumptions used have been stated, and the requirements derived. Also, substantiation data have been presented or referenced.

3.7.2 TURBULENCE MODELS

REQUIREMENT

3.7.2 Turbulence models. Where feasible, the von Karman form shall be used for the continuous random turbulence model, so that the flying qualities analyses will be consistent with the comparable structural analyses. When no comparable structural analysis is performed or when it is not feasible to use the von Karman form, use of the Dryden form will be permissible. In general, both the continuous random model and the discrete model shall be used. The scales and intensities used in determining the gust magnitudes for the discrete model shall be the same as those used in the Dryden continuous random model.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

Background

MIL-F-8785B provides for both continuous and discrete approaches to atmospheric disturbances (turbulence). These are both described in a statistical way (i.e., power spectral densities, probability density functions, etc.) because of the nature of atmospheric turbulence, which is a continuous vector random process that varies in three space dimensions and also in time. As a result, a very complicated statistical model would be required to describe adequately all that is currently known about atmospheric turbulence. Still, at present, much uncertainty remains about both its mathematical description and its numerical data. Also, known and suspected variations of atmospheric disturbances with such factors as location, time of day, wind direction, lapse rate and latitude can hardly be used directly in an airplane design specification. So, in the interest of simplicity of development, description and application of turbulence models for MIL-F-8785B, several commonly used simplifying assumptions are made and are described in the following paragraphs.

The continuous random turbulence model will be described first and its underlying assumptions explained. Then the discrete model, which is based on the continuous model, will be described. Numerical data are given and explained in the discussion of 3.7.3. Both models specify true gust velocities, not equivalent velocities.

Atmospheric turbulence should be described mathematically in an axis (coordinate) system related explicitly to the turbulence field itself; but instead, for MIL-F-8785B, the turbulence is described relative to the airplane body-axis system. If the turbulence were completely isotropic, mathematically it would be described exactly the same way in both axes systems, by the definition of isotropy: invariance of the statistical properties with axis translation, rotation and reflection. However, complete isotropy is not assumed here for all altitudes. Near the ground level the turbulence is assumed isotropic (and therefore, because of the translational independence, homogeneous) only in planes that are nearly horizontal (i.e., two-dimensional isotropy and homogeniety). At higher altitudes complete isotropy is assumed. The implication of these assumptions is that flight paths must be within a degree or so of being horizontal near the ground; otherwise special consideration should be given to the anisotropic and nonhomogeneous nature of the turbulence. One way to sidestep this difficulty is to use average values of the turbulence model parameters as discussed in 3.7.3. Figure 1 from Reference M3 clearly shows anisotropy and nonhomogeniety for the last 300 feet of a landing approach.

Complete isotropy is a property of turbulence that, when present, extends over all turbulence wavelengths. Realistically, however, atmospheric turbulence is at most only "locally" isotropic; that is, the turbulence is isotropic only over some finite range of wavelengths, usually those smaller than some given value which depends on meteorological conditions and nearness of the ground. Energy is fed anisotropically into the turbulence by winds which have wavelengths greater than those in the range of local isotropy. Near the ground, the limiting (longest) wavelength of the locally isotropic range is proportional to the height above the ground. Certain high-altitude clear air turbulence (CAT) is significantly anisotropic (References M69 and M71); but, like the atmospheric boundary layer turbulence, it too has a locally isotropic range. The continuous random turbulence models of MIL-F-8785B are designed to exhibit some locally isotropic properties at low altitudes. This has been done because turbulence data do indeed show a locally isotropic range that is within the range of wavelengths significant in the response of airplanes.

For the purposes of MIL-F-8785B, lack of isotropy at the lower altitudes implies that the statistical properties of the turbulence differ among the three turbulent velocity components and that statistical cross correlations exist among the components. It is assumed for simplicity, however, that the cross correlations are negligible. In other words, the turbulent velocity components are assumed statistically independent of one another even though the turbulence model permits anisotropy for low altitudes. There is some evidence that the cross correlations are small and insignificant enough to make this a reasonable assumption in most cases (Reference M81).

Tests for homogeniety and isotropy have been made frequently during gust measurement programs, and the validity of these assumptions has been adequately demonstrated for purposes of airplane response analyses except for flight near the terrain. Hence, the homogeniety and isotropy of gusts have been dignified by frequent use. Furthermore, homogeniety in horizontal planes is a reasonable assumption for flight over homogeneous terrain at low altitudes and for homogeneous meteorological conditions at higher altitudes. However, as Figure 1 suggests, homogeniety in the vertical direction is not a valid assumption, except perhaps for very shallow layers, regardless of the altitude. Nevertheless, for reasons to be brought out later, it is necessary to assume at least horizontal homogeniety.

Taylor's hypothesis (References M54 and M57), that time variations are statistically equivalent to distance variations in traversing the turbulence field, is also assumed. This almost universally employed assumption may be visualized as a gust field that is frozen in time and space. Taylor's hypothesis is implicit, for example, in use of gust data from tower measurements for airplane design. The implication of this hypothesis is that the turbulence-induced responses of the airplane result only from the motion of the airplane relative to the turbulent field. Experience has shown that the frozen field concept is entirely acceptable for those cases in which the mean wind velocity and the root-mean-square turbulence velocity are small relative to the ground speed of the airplane (Reference M57).

Since the airplane moves through the horizontally homogeneous turbulent field, the turbulence it senses is stationary; that is, the statistical properties of the turbulence as sensed by the airplane are independent of time in any given "patch" of turbulence. Therefore, stationary statistical methods may be used in analyses. This is the great simplification afforded by the assumptions of horizontal homogeniety and the frozen field concept.

The turbulent velocity field is assumed to be a zero-mean Gaussian (Normal) random process represented by a joint conditional (conditioned on the standard deviations of the velocity components) probability density function. Although Reference M83 presents ample evidence to invalidate the Gaussian assumption, this assumption is employed for MIL-F-8785B because of the simplicity it affords. Also, the Gaussian assumption greatly facilitates both mathematical analyses for linear systems and also simulations. (The experiments upon which a number of the flying qualities requirements are based used either filtered Gaussian noise or a random-appearing sum of sine waves to simulate turbulence. Neither technique produces enough extreme gusts or many of the "spikes" which are apparent in some turbulence records; that is one good reason for supplementing the MIL-F-8785B Gaussian continuous random turbulence model with the discrete gusts of 3.7.2.3.)

There are several good sources of general information on the description of atmospheric turbulence and its application in airplane response and analyses: see, for example, References B70, M32, M56, M58, M55, M54, M59, M81 and M83.

Gust spectra

In order to describe how much of the total root-mean-square turbulent velocity is contributed by a given band of wavelengths or spatial frequencies ($\Omega = 2\pi/\lambda$), the power spectral density of turbulence must be specified. Experimental and theoretical evidence suggest that the one-dimensional power spectral density of a turbulent velocity component should approach a constant value asymptotically at the lowest frequencies (longest wavelengths) and should decrease asymptotically according to the -5/3 power of frequency for high spatial frequencies. These "high" frequencies occur in a range called the "inertial subrange", in which energy is neither fed into nor dissipated from the turbulence (References M54 and M57). In a meteorological sense this range may not occur at all, depending on conditions but experimental data nearly always show a range of frequencies where the -5/3 power variation does occur. Often this -5/3 variation occurs when meteorological conditions do not indicate an inertial subrange.

MIL-F-8785B includes two spectral forms for the random continuous turbulence model:

(1) The von Karman spectral form (References M55 and M56), which has the asymptotic characteristics described above.

(2) The Dryden spectral form (References M55 and M56), which mathematically has the same low-frequency asymptote as the von Karman form but which has a high-frequency asymptote that is directly proportional to the -2 power of spatial frequency.

Both of these mathematical forms, named after the scientists who first proposed them, have been used in the past and are still accepted by many workers, although the trend is to adopt the scientifically more pleasing von Karman form. Other military and civil specifications, particularly proposed structural specifications, require the use of this spectral form. The Dryden spectra are rational, which greatly simplifies analysis and computation. There is every indication that, though the von Karman form seems to fit the available data somewhat better, for flying qualities work the two forms yield much the same results. Where feasible, the von Karman spectral form (3.7.2.1) is to be used in analyses with the continuous random turbulence model so that the flying qualities analyses will be consistent with the comparable structural analyses required by other specifications. However, when no comparabl structural analysis is performed or when it is not feasible to use the von Karman form, the Dryden form (3.7.2.2) may be used. It should be mentioned that the rms intensity data in 3.7.3 were developed specifically for the Dryden model but are used without modification for the von Karman model as well. However, the scales are specified differently for the two models in 3.7.3.1 and 3.7.3.2. Some consequences of this imperfect arrangement are described in the discussion of 3.7.3.1 and 3.7.3.2.

MIL-F-8785B specifies both continuous random models and a discrete model. However, these models are not entirely separate entities. The discrete model has been derived from the Dryden random continuous one and, as such, the discrete model requires the use of length scales and rms intensities which are the same as those used for the Dryden random continuous model.

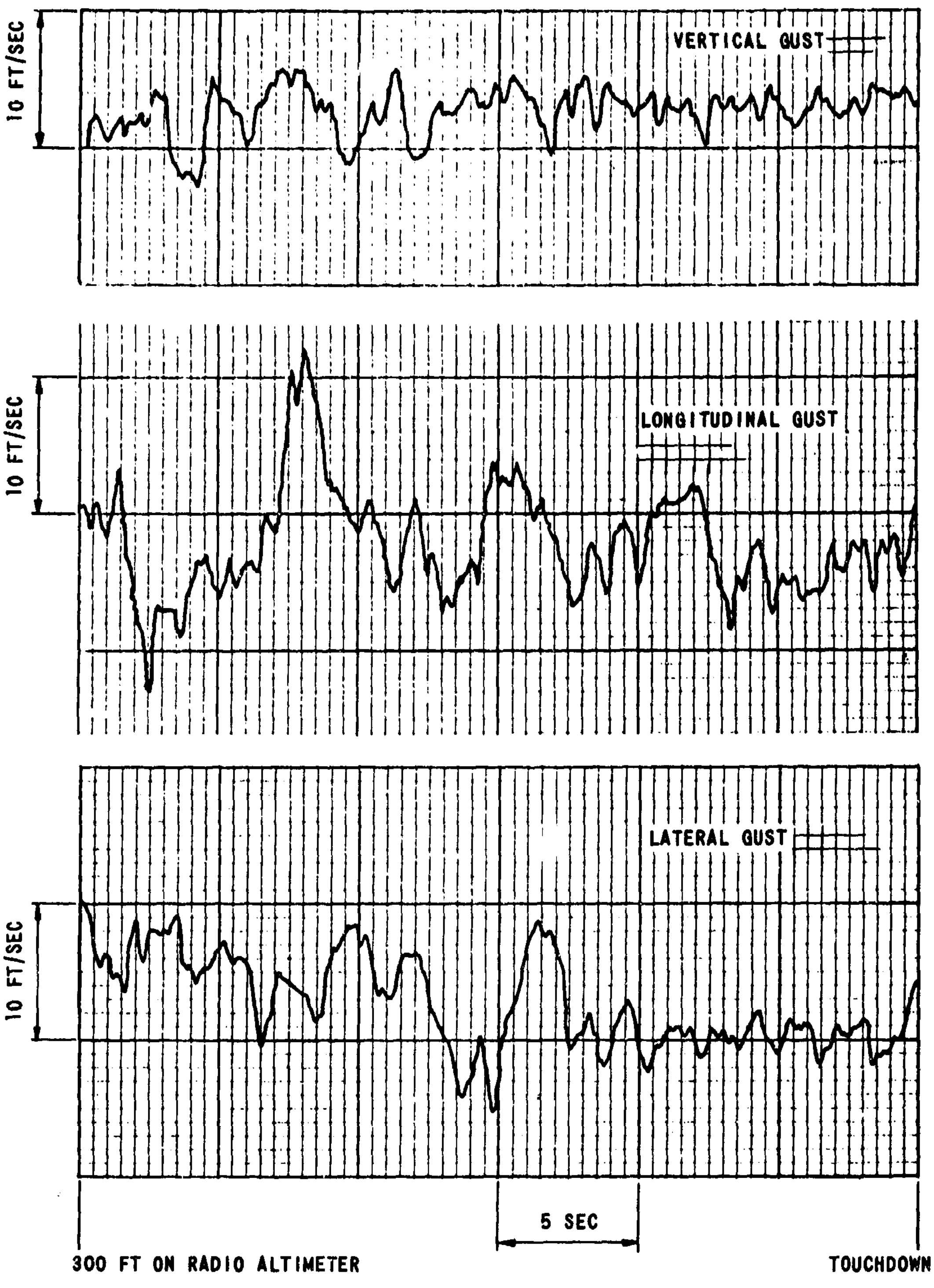


Figure 1 (3.7.2)
TURBULENCE VELOCITY COMPONENTS DURING A LANDING APPROACH

3.7.2.1, 3.7.2.2 CONTINUOUS RANDOM TURBULENCE MODEL

REQUIREMENT

3.7.2.1 Continuous random model (von Karman form). The von Karman form of the spectra for the turbulence velocities is:

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

$$\Phi_{v_g}(\Omega) = \sigma_v^2 \frac{L_v}{\pi} \frac{1 + \frac{8}{3}(1.339 L_v \Omega)^2}{\left[1 + (1.339 L_v \Omega)^2\right]^{11/6}}$$

$$\Phi_{w_g}(\Omega) = \sigma_w^2 \frac{L_w}{\pi} \frac{1 + \frac{8}{3}(1.339 L_w \Omega)^2}{\left[1 + (1.339 L_w \Omega)^2\right]^{11/6}}$$

3.7.2.2 Continuous random model (Dryden form). The Dryden form of the spe.tra for the turbulence velocities is:

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

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

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

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

The formulas for the one-dimensional power spectral densities apply to the longitudinal (x-axis), lateral (y-axis) and vertical (z-axis) gust-velocity components. Note, as mentioned earlier, that the Dryden form exhibits the Ω^{-2} variation at high frequencies instead of the $\Omega^{-5/3}$ variation of the von Karman form. Figure 1 shows plots of the two spectral forms (only the vertical gust velocity) for several values of scale L_w. The formulas for the spectra are listed, for example, in References M55 and M56 along with their corresponding autocovariance functions (sometimes called autocorrelation functions) and other pertinent formulas. They are defined such that the mean-square turbulence velocity (or variance, since the mean turbulence velocity fluctuation is assumed zero) is given by integrating the power spectrum over all positive spatial frequencies:

$$\sigma_i^2 = \int_0^\infty \Phi_i(\Omega)\, d\Omega \ , \quad i = u, v, \text{ or } w$$

The root-mean-square (rms) velocity σ (or standard deviation) is simply the square root of the integral.

The frozen field concept (Taylor's hypothesis) mentioned previously implies, in the frequency domain, that the temporal frequency ω (rad/sec) sensed by the airplane is related to the spatial frequency by the true air-speed V; that is, $\omega = \Omega V$. Therefore the spectral densities are transformed to functions of ω as follows:

$$\phi_i(\omega) = \frac{1}{V}\, \Phi_i\left(\Omega = \frac{\omega}{V}\right) \ , \quad i = u, v, \text{ or } w$$

It follows from this that

$$\sigma_i^2 = \int \phi_i(\omega)\, d\omega \ , \qquad i = u, v, \text{ or } w$$

Both the von Karman and Dryden forms satisfy all the mathematical requirements for isotropic atmospheric turbulence. For isotropi turbulence, the scales and the mean-square intensities are the same for the three velocity components; i.e., for isotropic turbulence

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

and

$$L_u = L_v = L_w$$

In MIL-F-8785B these same spectral forms $[\Phi_i(\Omega)$ and $\phi_i(\omega)]$ are used also for anisotropic turbulence, near the ground, as well as for isotropic turbulence. For anistropic turbulence, neither the three intensities nor the three scales need be equal. This is the reason for the u, v, w subscripts on σ and L.

In certain theoretical analyses it may be desirable to use the two-or three-dimensional versions of the von Karman and Dryden spectral forms. In particular, the two-dimensional spectrum of the vertical turbulence component has proven to be useful in the past. It depends on two spatial frequencies, Ω_1 and Ω_2 corresponding with x and y dimensions respectively (the spatial frequency Ω used in the one-dimensional formulas above is really Ω_1, but the subscript has been dropped for simplicity). Also, it should be noted that Ω_1 is the only spatial frequency that can be converted to a temporal frequency so far as this specification is concerned. The reason is that, to a first-order approximation, the airplane velocity is considered to be in the x direction. The two-dimensional formulas may be found for both the Dryden and the von Karman spectral forms in Reference M55, Section 9.4. To put the formulas of Reference M55 into a form consistent with the spectra in MIL-F-8785B, use the following conversion formulas:

(a) one-dimensional spectra

$$\Phi_i(\Omega) = \frac{1}{2\pi} S_i\left(k = \frac{\Omega}{2\pi}\right), \quad i = u, v, \text{ or } w$$

(b) two-dimensional spectrum

$$\Phi_w(\Omega_1, \Omega_2) = \frac{1}{4\pi^2} S_w\left(k_1 = \frac{\Omega_1}{2\pi}, \quad k_2 = \frac{\Omega_2}{2\pi}\right)$$

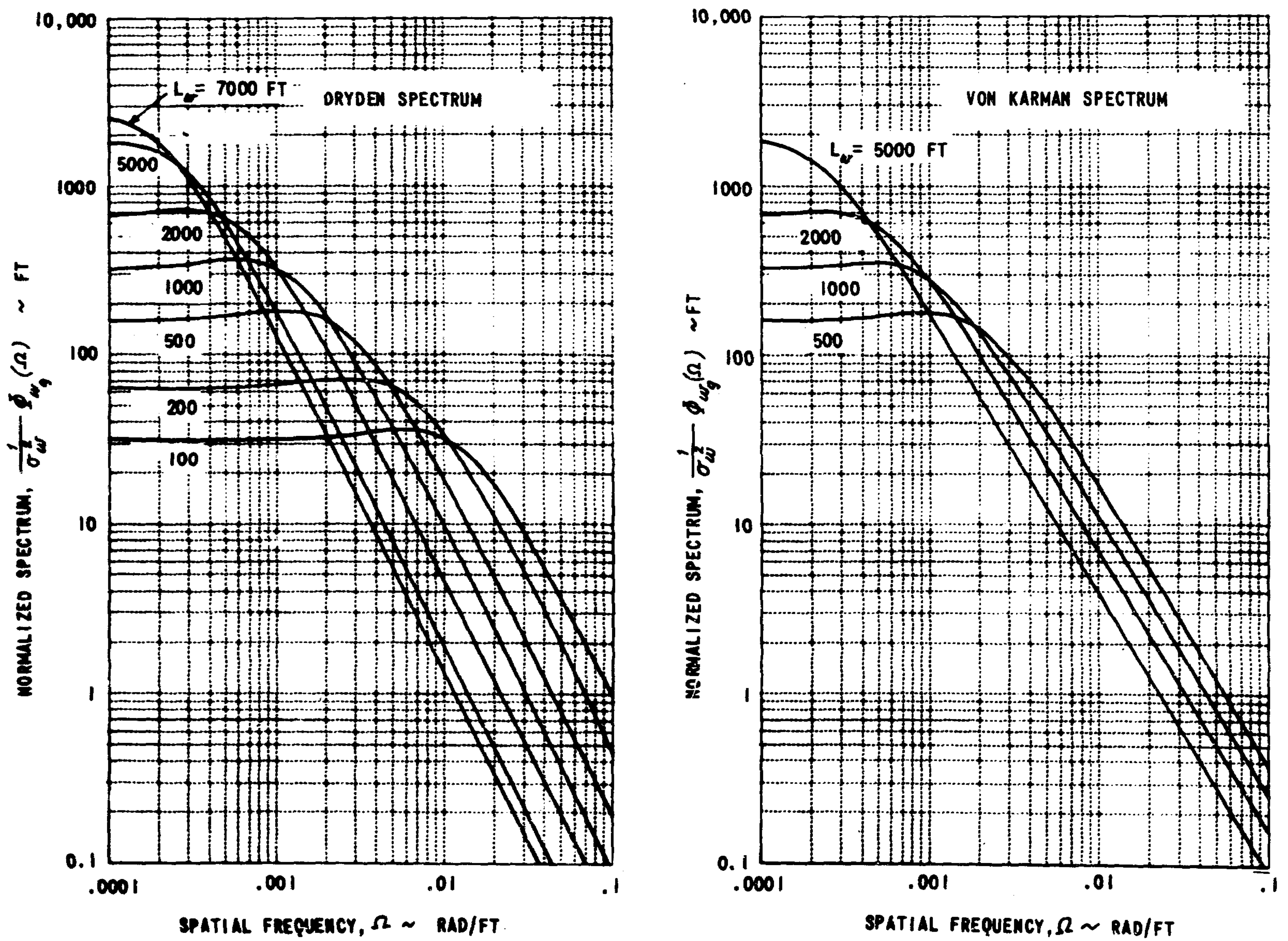


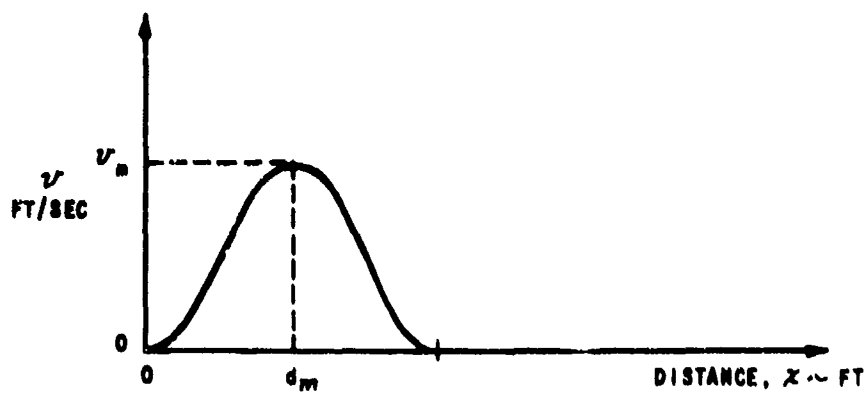
Figure 1 (3.7.2.1, 3.7.2.2)
THE DRYDEN AND VON KARMAN SPECTRAL FORMS
FOR THE VERTICAL TURBULENT VELOCITY

3.7.2.3 DISCRETE MODEL

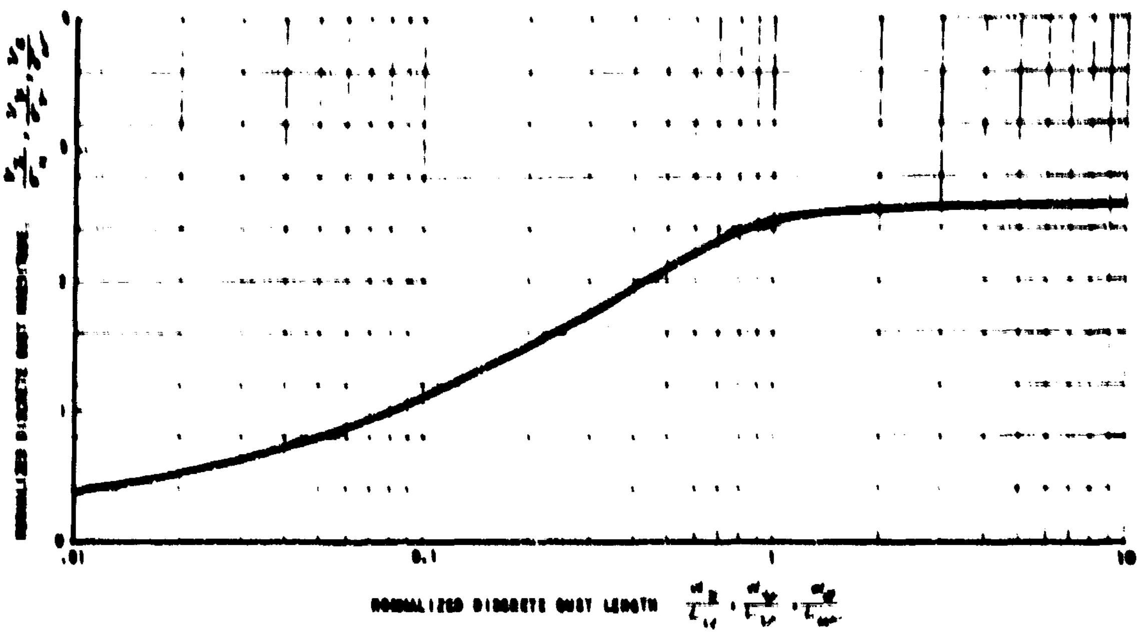
REQUIREMENT

3.7.2.3 Discrete model. The discrete turbulence model may be used for any of the three gust-velocity components. The discrete gust has the "1-cosine" shape:

$$
\begin{aligned}
v &= 0 \qquad &, \; x < 0 \\
&= \frac{v_m}{2}\left(1 - \cos\frac{\pi x}{d_m}\right) \qquad &, \; 0 \le x \le 2d_m \\
&= 0 \qquad &, \; x > 2d_m
\end{aligned}
$$



Several values of d_m shall be used, each chosen so that the gust is tuned to each of the natural frequencies of the airplane and its flight control system (higher-frequency structural modes may be excepted). The magnitude v_m shall then be chosen from figure 7. The parameters L and σ to be used with figure 7 are the Dryden scales and intensities from 3.7.3 or 3.7.4 for the velocity component under consideration.



MIL-F-8785B Figure 7. MAGNITUDE OF DISCRETE GUSTS

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

For analyses of some airplane operations, a discrete gust model is more appropriate than a continuous random model. The discrete model provides spike-type inputs that may not be apparent in the simulated Gaussian random turbulence. These exercise the vehicle and its flight control system in a different way but still in a way that is likely to be encountered. Thus the random and discrete-gust analyses are complementary. An example of design use is the requirement of 3.3.4: roll control power must be effective enough to balance the airplane when it encounters discrete, as well as random, gusts. Both might be particularly important in a landing approach. A discrete model is equally applicable to airplane structural design as well as airplane handling and control.

Since it is well known that the gust velocity gradient is as important as the gust magnitude, the discrete gust model should jointly treat both parameters. Hence, this section describes a form of discrete model that has a gust velocity $v(x)$ defined spatially in terms of a magnitude v_m which occurs at a distance $x = d_m$. Together, v_m and d_m determine the velocity gradient properties of the discrete gust. A conditional probability density, which describes the evolution of the random gust velocity, is developed by using certain results of the theory of linear mean-square estimation, as presented in Reference M82, Chapters 11 and 14, along with the Dryden random continuous turbulence model. Details of the development are provided in References M61 and M60; but because these references are not generally available, particularly Reference M61, a brief outline of the development is presented here. To make use of the mean-square estimation theory, it is necessary only that the gust velocity components be Gaussian, which is assumed to be the case. The random continuous turbulence need be neither isotropic nor homogeneous; however, it has been assumed homogeneous for the simplicity afforded to the results.

The discrete gust model of MIL-F-8785B makes use of the rather arbitrary but well known "1-cosine" shape which has been used often, and still is, in both military and civil structural design specifications. In the formula given in 3.7.2.3, the gust magnitude v_m may be either positive or negative, but the gust length d_m is always taken to be positive. The formula for the discrete gust shape and other formulas and data in this discussion are assumed to apply equally well to any of the three gust velocity components even though, in all of the pertinent formulas of this discussion, the x coordinate is always used for the distance traversed by the airplane, and d_m is always the gust length in the x direction. To distinguish among the values of v_m and d_m for the three components of the turbulent velocity in Figure 7 of MIL-F-8785B, the m subscript is dropped and x, y and z subscripts are used for longitudinal, lateral and vertical discrete gusts respectively.

Section 14-5 of Reference M82 is followed closely in the material developed here. (Also, see Example 11-1 starting on page 394 of Reference M82.) The gust velocity is assumed to be homogeneous, and it has zero mean, i.e.

$$E\{v(x)\} = 0$$

where E is the expectation operator. If the velocity at $x = x_1$ is $v(x_1)$ then the conditional mean of $v(x)$ is

$$E\left\{v(x) \mid v(x_1)\right\}, \quad x > x_1$$

In Reference M82, Equation 7-111 on Page 224, it is shown that this conditional mean is proportional to $v(x_1)$, i.e.,

$$E\left\{v(x) \mid v(x_1)\right\} = a\, v(x_1)$$

This conditional mean is also the best mean-square estimate of $v(x)$ in terms of $v(x_1)$. Now since the difference $v(x) - a\,v(x_1)$ is statistically orthogonal to $v(x_1)$, i.e.,

$$E\left\{\left[v(x) - a\,v(x_1)\right] v(x_1)\right\} = 0,$$

the following result is obtained:

$$a = \frac{R(d)}{R(0)}$$

where $d = x - x_1$ and $R(d)$ is the autocovariance function of the homogeneous gust velocity $v(x)$, assuming $v(x_1)$, is

$$e = E\left\{\left[v(x) - a\,v(x_1)\right]^2 \mid v(x_1)\right\} = R(0) - a\,R(d)$$

where the orthogonality condition above has been used; then, with the value of a also given above,

$$e = R(0) - \frac{R^2(d)}{R(0)}$$

Because $v(x)$ is Gaussian, the conditional probability density of $v(x)$ is also Gaussian:

$$p_v\left[v(x) \mid v(x_1)\right] = \frac{1}{\sqrt{2\pi e}} \exp\left\{\frac{-1}{2e}\left[v(x) - a\,v(x_1)\right]^2\right\}$$

This conditional probability density is the key to the discrete gust model, but in order to evaluate the density, some further assumptions are necessary.

Although $v(x_1)$ may have any value and may be a random variable, it is taken as zero since the "1-cosine" gust shape is generally considered to start from zero velocity. Furthermore, in the development of the conditional density, it is necessary to assume a mathematical form for the autocovariance of the turbulent velocity component. For this purpose, the autocovariance corresponding with the longitudinal (u_g) Dryden spectral form is used. It leads to the simplest mathematical form for the conditional density above. Although this is only approximate for the other velocity components, it is accurate enough, considering the very limited data available for substantiation. Thus, a and e become

$$a = \exp\left(-\frac{d}{L}\right), \quad d \geq 0$$

and

$$e = \sigma^2\left[1 - \exp\left(-\frac{2d}{L}\right)\right]$$

where σ^2 and L are the mean-square intensity and scale for the particular gust velocity component, respectively. Furthermore, if $v(x) = v(x_1+d_m)$ is interpreted as the magnitude v_m, the conditional probability density may be written

$$p_{v_m}\left[v_m \mid v(x_1)=0\right] = \frac{1}{\sqrt{2\pi e}} \exp\left(-\frac{1}{2}\frac{v_m^2}{e}\right)$$

where

$$e = \sigma^2\left[1 - \exp\left(-\frac{2d_m}{L}\right)\right], \quad d_m \geq 0$$

So far, the conditional density has been treated as if it were conditioned only on $v(x_1)$; however, it is also conditioned on the random variable σ which appears in the expression for . (The statistics of the rms intensity σ are described later in the discussion of 3.7.3.) Then, since explicit dependence on v (x_1) no longer appears, the density function may be written as $p(v_m \mid \sigma)$ for brevity. The appropriate subscript u, v or w is used in Figure 7 of MIL-F-8785B to denote the longitudinal, lateral or vertical gust respectively.

Since the turbulent velocity fluctuations have a Gaussian probability density function (as was assumed to begin this development), the conditional density $p(v_m/\sigma)$ and its cumulative distribution also have the Gaussian form. The cumulative probability distribution $P(v_m \mid \sigma)$ is plotted in Figure 1 in a normalized form. The curve of Figure 7 of MIL-F-8785B has been obtained simply by cross-plotting the values of v_m/σ and d_m/L from Figure 1 that correspond to the probability $P(v_m \mid \sigma) = 0.01$. Strictly speaking, it is not proper to use the value of v_m/σ obtained this way and simply multiply it by a value of σ from Figure 8 of MIL-F-8785B. The reason for this is that $P(v_m \mid \sigma)$ is a conditional cumulative distribution. The value of v_m should properly be obtained, first, by determining $P(v_m)$, with the condition removed as follows:

$$P(v_m) = \int_0^\infty P(v_m \mid \sigma)\, p(\sigma)\, d\sigma$$

where $p(\sigma)$ is obtained from the discussion of 3.7.3, and then by determining v_m as a function of altitude for a given value of $P(v_m)$. However, it is felt that the method used in MIL-F-8785B yields reasonable values of v_m for the present purposes even though the values used are not necessarily the same as those that have traditionally been used in other discrete gust models.

It is now evident that the discrete gust magnitude, v_m, depends on three parameters: d_m, L and σ. Several values of d_m should be used, each chosen so that the gust is tuned to each of the rigid-body natural frequencies of the airplane and its flight control system. The lower-frequency structural bending modes are also to be considered in analyses, but the higher structural-mode frequencies are excepted. Usually it should be sufficient to consider the primary fuselage vertical and side bending modes, and the primary wing symmetric and antisymmetric modes. However, specific requirements for structural bending modes may be stated by the procuring activity. The values of σ and L to be used with the discrete gust model depend on altitude and are the same as those used with the Dryden random continuous turbulence model which are found in 3.7.3 and 3.7.4. The values of d_m, L and σ obtained according to these instructions, used together with Figure 7 of MIL-F-8785B, will provide the required values of v_m for analyses.

Thus, although the (1-cos) form of the discrete gust remains arbitrary, the gust magnitude has been related in a rational manner to the expected intensity of continuous random turbulence. There is still, of course, no rational relation to some of the sources of discrete disturbances: buildings or terrain features, aircraft wakes, shear layers, jet streams, etc., in either form or magnitude.

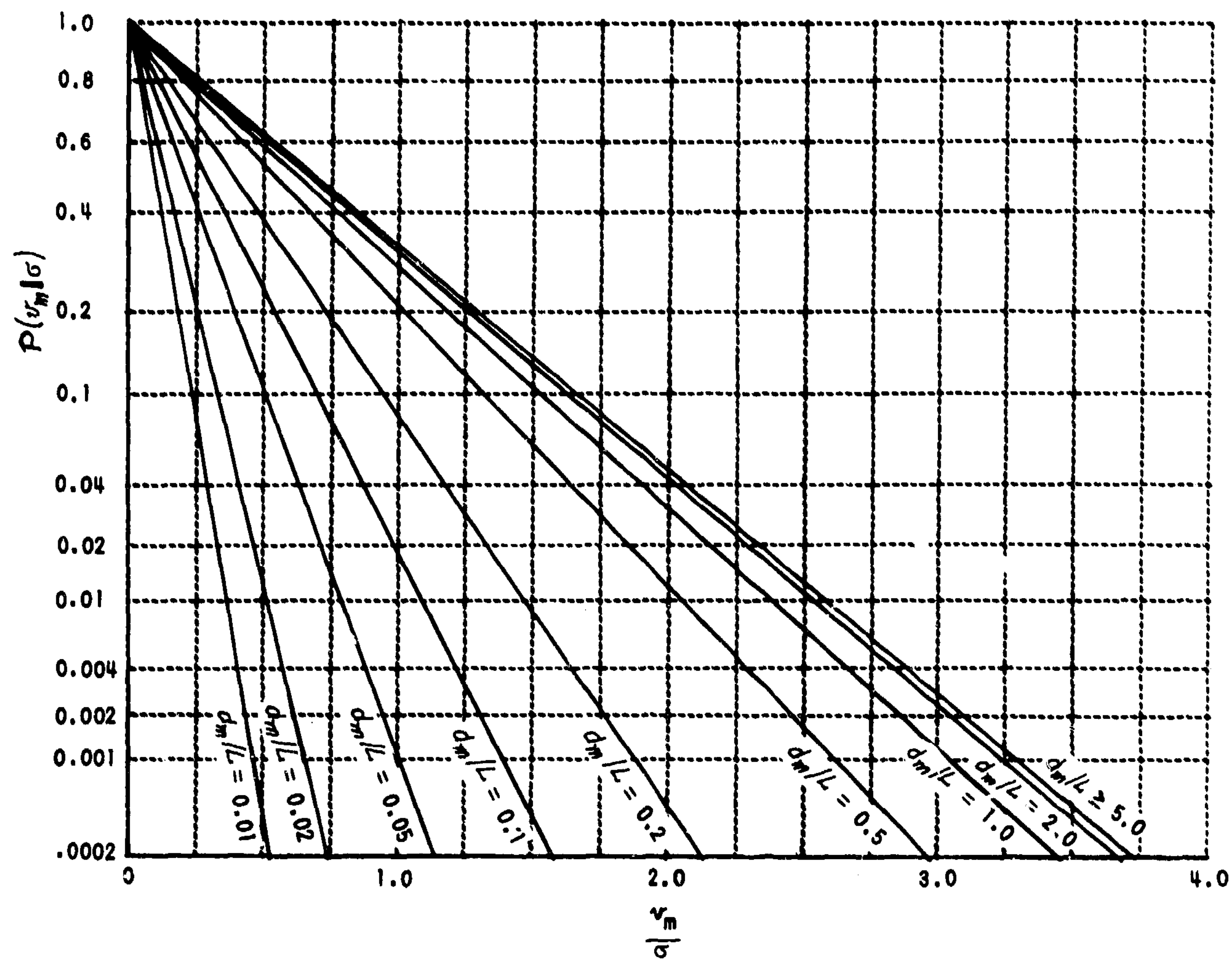


Figure 1 (3.7.2.3)

PROBABILITY OF EQUALLING OR EXCEEDING A GIVEN GUST MAGNITUDE v_m FOR VARIOUS VALUES OF d_m/L

3.7.3 SCALES AND INTENSITIES (CLEAR AIR TURBULENCE)

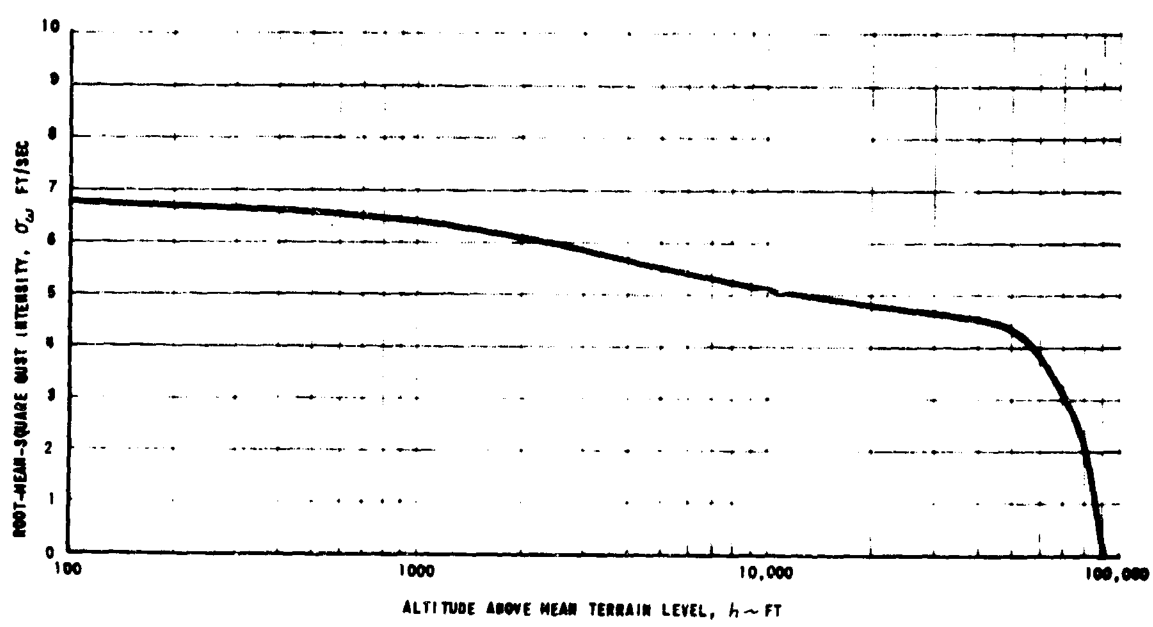
REQUIREMENT

3.7.3 Scales and intensities (clear air turbulence). The root-mean-square intensity σ_w for clear air turbulence is defined on figure 8 as a function of altitude. The intensities σ_u and σ_v may be obtained using the relationships

$$\frac{\sigma_u^2}{L_u^{2/3}} = \frac{\sigma_v^2}{L_v^{2/3}} = \frac{\sigma_w^2}{L_w^{2/3}} \qquad \text{(von Karman form)}$$

$$\frac{\sigma_u^2}{L_u} = \frac{\sigma_v^2}{L_v} = \frac{\sigma_w^2}{L_w} \qquad \text{(Dryden form)}$$

The scales for clear air turbulence are defined in 3.7.3.1 and 3.7.3.2 as a function of altitude. The altitude shall be defined consistently with any applicable terrain models specified in the contract. For those Flight Phases involving climbs and descents, a single set of scales and intensities based on an average altitude may be used. If an average set of scales and intensities is used for Category C Flight Phases, it shall be based on an altitude of 500 feet.



MIL-F-8785B Figure 8. INTENSITY FOR CLEAR AIR TURBULENCE

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

Introduction

The turbulence model of MIL-F-8785B has been developed by rather arbitrarily choosing reasonable values of the scales and then determining values for the intensity so that the mathematical spectral form (the Dryden form) fits the measured spectral data (from References M68, M69, M70, M71 and other sources listed in Reference M7). This approach is explained in a few simple steps.

(1) Measured atmospheric turbulence spectra seldom extend to low enough spatial frequencies to exhibit the constant low-frequency asymptote (however, some tower spectra measured very close to the ground do show the low-frequency asymptote.)

(2) The measured spectra, therefore, can determine no more than some minimum value of scale and some factor which combines both the scale and the rms intensity.

(3) This factor, σ^2/L, is obtained from the high-frequency asymptotic form of the Dryden spectrum. It is the same factor as in the equations relating the three intensities σ_u, σ_v and σ_w. (See 3.7.3.) (Although it was not used in the development of the data for MIL-F-8785B, the corresponding factor for the high-frequency asymptotic form of the von Karman spectrum is $\sigma^2/L^{2/3}$.)

(4) Consequently, having chosen a Dryden scale (Section 3.7.3.2) which is not less than some minimum, and having determined the value of the factor σ^2/L, one then simply obtains the value of σ^2.

This is the basic procedure that was followed for evaluating the rms vertical intensity (σ_w) of the clear air turbulence model, but the σ_w data chosen for the model were also checked as described later. The scales and intensities of the random turbulence model are interrelated by the equations in 3.7.3. These equations are used to calculate σ_u and σ_v after σ_w has been obtained from Figure 8, because MIL-F-8785B provides data explicitly only for σ_w. The following discussion is devoted primarily to the development of Figure 8. It starts by developing the statistical properties of σ_w.

Statistical Properties of σ_{w}

Now, there is a finite discrete probability, P_0, of there being no turbulence at all, while there is also a discrete probability of encountering turbulence, P_1. The total probability density function for σ_w is, therefore,

$$p(\sigma_w) = P_0\,\delta(\sigma_w) + P_1\,\hat{p}\,(\sigma_w)$$

where $\delta(\sigma_w)$ is a Dirac delta function which is zero for any $\sigma_w \neq 0$ and for which

$$\int_0^{\infty} \delta(\sigma_w)\,d\sigma_w = 1$$

The probability density function for the rms vertical turbulence intensity, σ_w, once turbulence has been encountered, is assumed to be of the Rayleigh form:

$$\hat{p}(\sigma_w) = \frac{\sigma_w}{c^2} \exp\left(-\frac{1}{2}\frac{\sigma_w^2}{c^2}\right)$$

$$0 \leq \sigma_w \leq \infty, \quad c > 0$$

where c^2 is one-half the expected value of σ_w^2 (i.e., $\frac{1}{2}E\left\{\sigma_w^2\right\}$). (In some other current turbulence models which employ the Gaussian probability density defined for $\sigma_w \geq 0$, i.e.,

$$\hat{p}\,(\sigma_w) = \sqrt{\frac{2}{\pi}}\,\frac{1}{b_1} \exp\left(-\frac{1}{2}\frac{\sigma_w^2}{b_1^2}\right), \quad b_1 > 0$$

the parameter b_1^2 is equal to $E\left\{\sigma_w^2\right\}$; therefore, to have the same expected value of σ_w^2, $b_1 = \sqrt{2}\,c$.)

The Rayleigh cumulative probability distribution for σ_w , which here means the probability of <u>equalling or exceeding</u> a given value of σ_w in a patch of turbulence, is

$$\hat{P}(\sigma_w) = 1 - \int_0^{\sigma_w} \hat{p}(\sigma_w)\, d\sigma_w = \exp\left(-\frac{1}{2}\,\frac{\sigma_w^2}{c^2}\right)$$

(This is <u>not</u> the usual definition of the cumulative probability distribution that is given in probability theory texts such as Reference M82, but this is the definition used in most of the atmospheric turbulence literature.)

Figure 1, which is adapted from Reference M62, illustrates that the Rayleigh probability density function fits reasonably well the relative frequency distributions of σ . The value of c used in the figure is that used for MIL-F-8785B, and its evaluation is described in the following paragraphs. For purposes of MIL-F-8785B, the Rayleigh density function is as convenient to use as any other previously employed or previously proposed functions. It is used in the absence of general agreement on the form of the distribution of σ_w .

In MIL-F-8785B, it is assumed that a single value of the parameter c is valid for clear air turbulence at all altitudes. Previous atmospheric turbulence models based on $b_1^2 = E\left\{\sigma_w^2\right\}$, which is proportional to c, have indicated that although b_1 is not really constant for nonstorm turbulence it is nearly enough constant so that for simplicity it may be assumed constant.

The measured spectra for clear air turbulence from many sources, such as References M31, M68, M69, M70, M71 and other sources as listed in Reference M7, were used to obtain values of σ^2/L for the Dryden spectral form. This ratio is related to c as follows:

$$E\left\{\frac{\sigma^2}{L}\right\} = \frac{1}{L}\,E\left\{\sigma^2\right\}$$

since L is not considered a random variable here. Then, since

$$E\left\{\sigma^2\right\} = 2c^2$$

there results

$$c = \sqrt{\frac{L}{2} E\left\{\frac{\sigma^2}{L}\right\}}$$

Thus, from a population of values of σ^2/L for clear air turbulence it is possible to estimate the value of the parameter c. The value of $\hat{c}$ thus obtained for MIL-F-8785B is 2.3 ft/sec. The cumulative probability $\hat{P}(\sigma_w)$ is plotted in Figure 2 with this value of c.

A quick check of the validity of the value c = 2.3 ft/sec and the assumption that c is independent of altitude may be obtained from the data given in Reference M63. Figure 1 from that reference compares the estimates of $b_1 = \sqrt{2}\, c$ from several sources. It shows that b_1 is reasonably independent of altitude and that the value b_1 = 3.25 ft/sec (or c = 2.3 ft/sec) is somewhat on the low end of the indicated range. However, Figure 4 of the same reference, which was obtained after a re-analysis of the basic data, shows a value of b_1 that is quite close to 3.25 ft/sec. This provides one independent check of the value of $c = b_1/\sqrt{2}$ = 2.3 ft/sec used in MIL-F-8785B.

The discrete probability (P_1) of encountering turbulence at all, which is often called the proportion of time spent in turbulence, is a function of altitude. As used here, P_1 has exactly the same meaning as it has had in previous turbulence models and specifications. Figure 3 gives the P_1 used here as the solid line. The curve is basically that of Reference M7, Figure 21, but revised to reflect some new data points that have more recently become available. The circled data points are directly from Reference M7, whereas the B-52 and B-70 points are the new ones (References M64 and M65 respectively). Above about 75,000 ft the curve in Figure 3 is an extrapolation. Between 50,000 and 70,000 feet the curve is approximately an average of the U-2 (References M66 and M67) and B-70 points. The large discrepancy between the U-2 and B-70 data points remains to be explained; in fact, this is the reason for simply using the "average" of the two for MIL-F-8785B. Between 25,000 and 50,000 feet the preferred B-52 values and the U-2 values were also roughly averaged.

For altitudes below 2500 ft above ground level (AGL), the abscissa (altitude) in Figure 3 should be interpreted as AGL because of the influence of the terrain on the probability of encountering turbulence. For altitudes above 2500 ft AGL it is permissible to interpret altitude as either pressure altitude or altitude above mean sea level (MSL). However, the altitude to be used shall be defined consistently with any applicable terrain models that may be specified by the procuring activity.

It should be clear at this stage that the model described here neglects any effects on the turbulence due to terrain roughness, atmospheric stability (lapse rate), wind shears, mean wind magnitude or any other meteorological factor except height (altitude). This means that the model describes an average of all conditions for clear air turbulence. Too few data are currently available to permit incorporation of these complicating features.

Specification of σ_w

So far, the turbulence model has been described strictly in a probabilistic fashion. For MIL-F-8785B, it is necessary to provide something concrete. A value of σ_w is specified as a function of altitude for clear air turbulence that can only be equalled or exceeded with a probability of 0.01. In other words, 99% of all the time spent in flight at a given altitude will be spent either in turbulence with less than the specified σ_w or in turbulence-free air ($\sigma_w = 0$). Let $P(\sigma_w)$ denote the cumulative probability that σ_w equals or exceeds a given value; then

$$P(\sigma_w) = P_1 \hat{P}(\sigma_w) = 0.01$$

where P_1 and $\hat{P}(\sigma_w)$ are given in Figures 3 and 2 respectively. The probability $P(\sigma_w) = 0.01$ has been chosen rather arbitrarily. Although it seems reasonable, only time and trial will prove its ultimate reasonableness. The values of σ_w obtained using the above equation are presented as a function of altitude in Figure 8 of MIL-F-8785B. Since the value of P_1 for altitudes above 90,000 feet is less than 0.01, the value of σ_w above 90,000 feet is exactly zero. This means simply that at h = 90,000 ft, $\sigma_w = 0$ is the only rms intensity that can be equalled or exceeded with exactly the overall probability 0.01. Above 90,000 ft, the same value $\sigma_w = 0$ applies since σ_w is never less than zero.

For those Flight Phases involving climbs and descents, a single set of scales and intensities based on an average altitude may be used. In particular, at altitudes where the turbulence is isotropic and for Category A and B (nonterminal) Flight Phases, the scales $L_u = L_v = L_w$ are constant, and an average value of $\sigma_u = \sigma_v = \sigma_w$ may be used. However, if the overall range of altitude is large, it may be necessary to break that range into several smaller ranges over which the rms intensity is relatively constant. If an average set of scales and intensities is used for Category C (terminal) Flight Phases, the set should be based on an altitude of 500 ft, whether Dryden or von Karman scales are used.

The probability data and, in particular, the value of c developed for 3.7.3 have been based on the curve-fitting of spectral data with the high-frequency asymptote of the Dryden spectral form. Nevertheless, the value $c = 2.3$ ft/sec is to be used without change with both the Dryden and von Karman spectral forms.

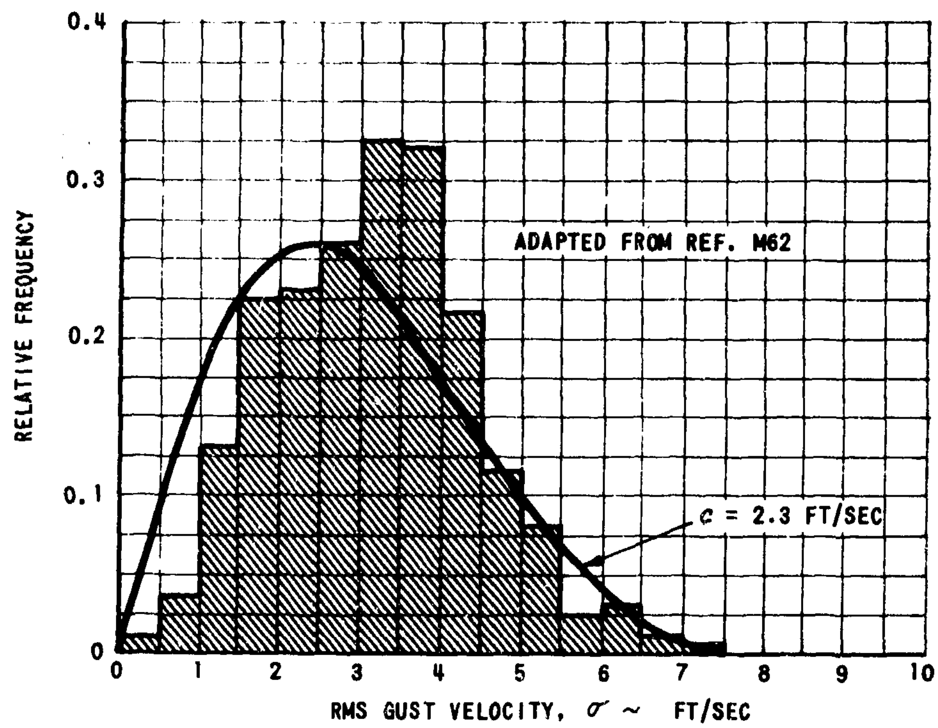
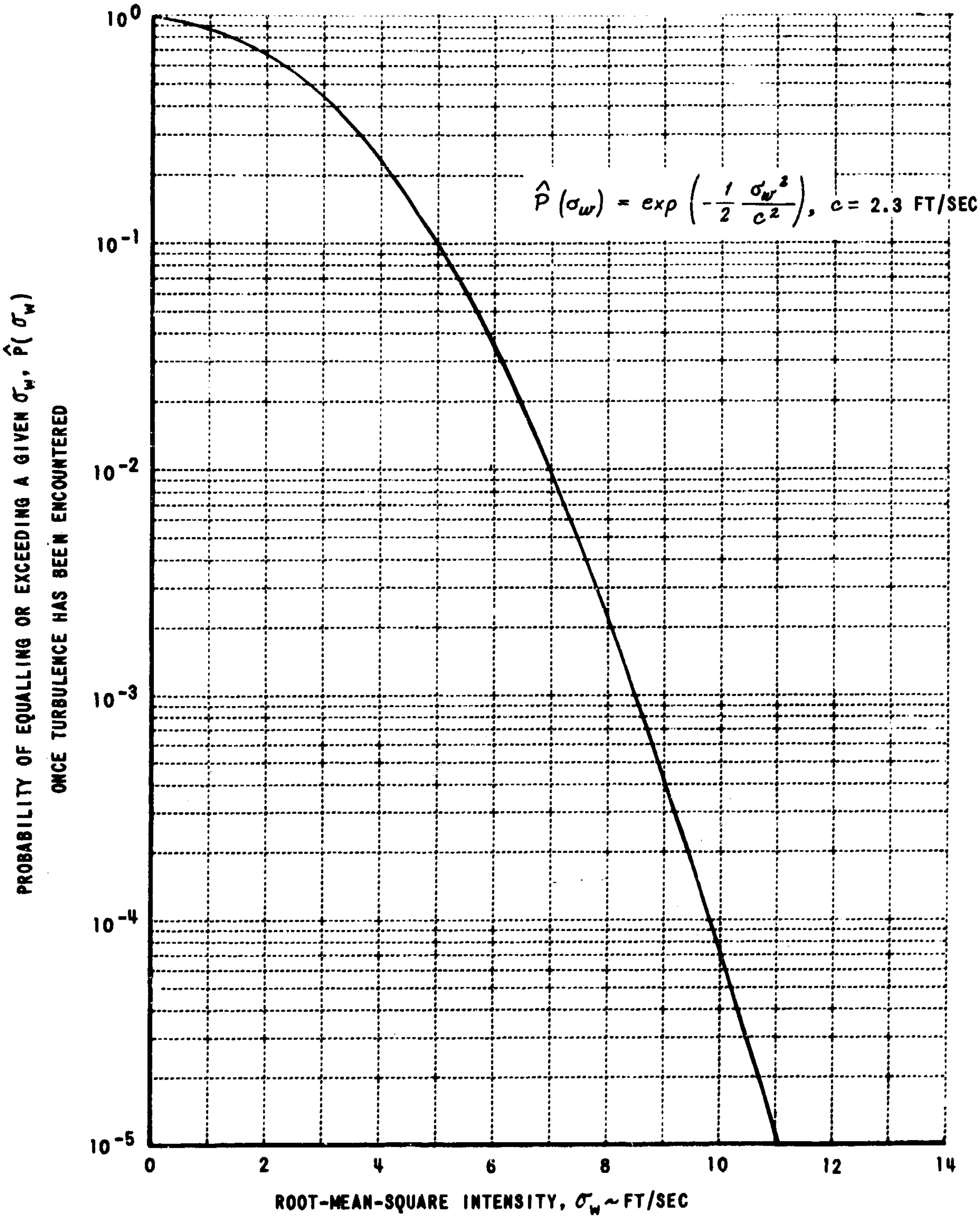


Figure 1 (3.7.3)
RELATIVE FREQUENCY DISTRIBUTIONS OF RMS GUST VELOCITY FROM B-66 LOW-LEVEL PROGRAM



$\hat{P}(\sigma_w) = exp\left(-\frac{1}{2}\frac{\sigma_w^2}{c^2}\right)$, $c = 2.3$ FT/SEC
10^0
10^{-1}
10^{-2}
10^{-3}
10^{-4}
10^{-5}
0
2
4
6
8
10
12
14
PROBABILITY OF EQUALLING OR EXCEEDING A GIVEN σ_w, $\hat{P}(\sigma_w)$
ONCE TURBULENCE HAS BEEN ENCOUNTERED
ROOT-MEAN-SQUARE INTENSITY, $\sigma_w \sim$ FT/SEC

Figure 2 (3.7.3)

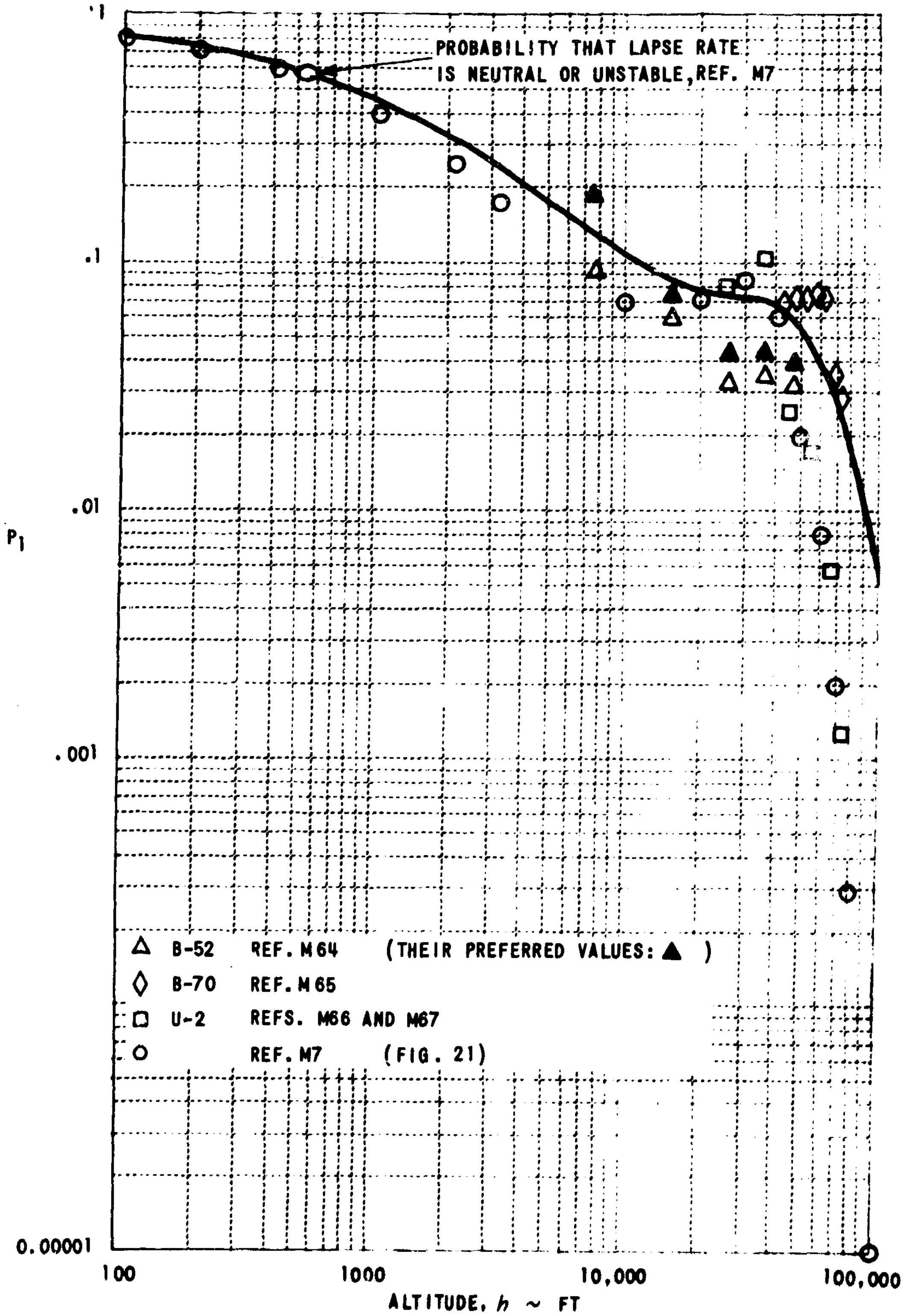


Figure 3 (3.7.3)

PROBABILITY OF ENCOUNTERING TURBULENCE

3.7.3.1, 3.7.3.2 CLEAR AIR TURBULENCE (SCALES)

REQUIREMENT

3.7.3.1 Clear air turbulence (von Karman scales). The scales for clear air turbulence using the von Karman form are:

Above h = 2500 feet: $L_u = L_v = L_w = 2500$ feet

Below h = 2500 feet: $L_w = h$ feet

$L_u = L_v = 184\ h^{1/3}$ feet

3.7.3.2 Clear air turbulence (Dryden scales). The scales for clear air turbulence using the Dryden form are:

Above h = 1750 feet: $L_u = L_v = L_w = 1750$ feet

Below h = 1750 feet; $L_w = h$ feet

$L_u = L_v = 145\ h^{1/3}$ feet

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

The three von Karman scales are all equal to 2500 ft for isotropic clear air turbulence above 2500 ft AGL. There are no compelling reasons for this choice, except that it is currently being used in other military and civil specifications. In addition, L = 2500 ft describes well measured spectra at these altitudes. At altitudes below 2500 ft AGL, a number of sources claim that the turbulence is not isotropic since the scales of the horizontal velocity components are not equal to the scale of the vertical component. (However, there are some data to the contrary in Reference M81.) Different scales have been used here. The formulas for the scales at low altitudes are adjusted to force all three scales to be exactly equal to 2500 ft at an altitude of 2500 ft AGL.

The low-altitude formulas for the Dryden scales are adjusted to make all three scales the same at the altitude h = 1750 ft AGL. Again there are no strong reasons for choosing L = 1750 ft as the scale for isotropic clear air turbulence above 1750 ft AGL. This value is larger than that commonly used in the past for the Dryden spectral models (i.e., L = 1000 ft), but it provides a good fit to all existing, measured, clear-air-turbulence spectral data. Furthermore, spectral measurements (e.g. Reference M56) made with improved instrumentation indicate rather conclusively that scales larger than 1000 ft must be used to best fit the spectra. Too small a scale would greatly underestimate the energy of the turbulence at the longer wavelengths that contribute significantly to the gust response in supersonic flight. (There is further discussion of this Dryden scale selection later.)

It should be noted that recent data taken during the HICAT (Reference M69) and TOPCAT (Reference M71) programs provide conclusive evidence that high-altitude clear air turbulence is significantly anisotropic at the longer wavelengths. Unfortunately, no clearly established trends amenable to simple mathematical modeling are evident. As a result of this situation, and also to provide the utmost in simplicity, the model of MIL-F-8785B is completely isotropic at the higher altitudes in both the scales and the rms intensities.

The variation of L_u and L_v at low altitudes according to the one-third power of altitude above ground level is simply a mechanism that forces the scales of the two horizontal components to be larger than the vertical scale. Although these formulas produce the correct trends, there are little data available that can be used to substantiate the $h^{1/3}$ as used in MIL-F-8785B. It is merely a formula that produces reasonable results. The variation of $L_w = h$ appears to be reasonably well established (Reference M55, Figure 2.12).

Although two sets of scales have been specified, one for the von Karman spectral form and the other for the Dryden form, there are not two different specifications for the rms intensity σ_w. This rather imperfect situation is the result of the way in which the data of MIL-F-8785B were developed. In the first place, the rms intensity data for σ_w were developed by curve-fitting the Dryden form to the measured spectral data, both for clear air turbulence and thunderstorm turbulence (3.7.4), using the assumed scales given in 3.7.3.2 and 3.7.4.2. Later in the development of Section 3.7, it was decided that the von Karman spectral form should also be incorporated with its corresponding scale and rms intensity data. Because of the differences in the high-frequency asymptotic nature of the Dryden ($\sim \Omega^{-2}$) and von Karman ($\sim \Omega^{-5/3}$) spectra, it is impossible to obtain exact agreement of the magnitudes of the two spectral forms over all spatial frequencies or wavelengths. However, the two spectral forms can be made to match reasonably well over the range of spatial frequencies covered by measured spectra, which admittedly do not extend to small enough frequencies for gust response analyses of supersonic (up to Mach 3 and higher) airplanes. It was discovered quite accidentally, however, that a reasonably good fit of the von Karman spectra to the Dryden spectra could be obtained by using the rms intensity data developed for the Dryden spectra, L = 1750 ft, along with the currently popular von Karman scale, L = 2500 ft, given in 3.7.3.1 and 3.7.4.1. This curve fitting of the two spectral forms is reasonable for isotropic turbulence over the band of spatial frequencies from zero to about 1 rad/ft. However, the curve fit is not as good below 2500 ft as it is above that height because the scales vary differently for the two spectral forms.

The decision was made, therefore, to use this particular fitting of the two spectral forms. The result, then, is that the MIL-F-8785B random, continuous turbulence model has:

(1) two spectral forms, the Dryden and the preferred von Karman

(2) a single rms vertical gust intensity specification for clear air turbulence that is valid for both spectral forms,

(3) two sets of equations relating the mean-square intensities and the scales are provided, namely,

$$\frac{\sigma_u^2}{L_u} = \frac{\sigma_v^2}{L_v} = \frac{\sigma_w^2}{L_w} \qquad \text{(Dryden)}$$

for the Dryden model, and

$$\frac{\sigma_u^2}{L_u^{2/3}} = \frac{\sigma_v^2}{L_v^{2/3}} = \frac{\sigma_w^2}{L_w^{2/3}} \qquad \text{(von Karman)}$$

for the von Karman model.

These are based on factors occurring in the high-frequency asymptotic forms of the respective spectra. At the higher altitudes, both spectral models exhibit complete isotropy. At the lower altitudes (near the ground), both spectral models have a locally isotropic range (at the high end of the frequency spectrum) as forced by the sets of equations above.

3.7.4 SCALES AND INTENSITIES (THUNDERSTORM TURBULENCE)

REQUIREMENT

3.7.4 Scales and intensities (thunderstorm turbulence). The root-mean-square intensities σ_u, σ_v, and σ_w are all equal to 21 feet per second for thunderstorm turbulence. The scales for thunderstorm turbulence are defined in 3.7.4.1 and 3.7.4.2. These values are to be used when evaluating the airplane's controllability in severe turbulence, but need not be considered for altitudes above 40,000 feet.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

The thunderstorm turbulence is to be used to evaluate airplane controllability in severe turbulence. Thunderstorms, of course, are not the only source of severe atmospheric disturbances. In the absence of other requirements, then, this thunderstorm turbulence can also be used to get some feeling for both the random and discrete effects of penetrating other severe turbulence such as in mountain waves, etc. Generally, less precise control is to be expected of an airplane in severe turbulence. Nevertheless, flying qualities must not degrade to the extent that the airplane becomes uncontrollable or that any attempt of the pilot to control the airplane is liable to lead to overstress.

Thunderstorm turbulence is assumed to be completely isotropic; therefore, the rms intensities are all equal ($\sigma_u = \sigma_v = \sigma_w$), and the scales are likewise equal ($L_u = L_v = L_w$). All measured thunderstorm spectra indicate that isotropy is a valid assumption (References M72 and M73). The value of the rms intensities is $\sigma_u = \sigma_v = \sigma_w = 21$ ft/sec (true). This value is approximately the median of the Rayleigh distribution for σ_w with c = 18 ft/sec. The parameter c was determined by fitting the measured spectra of References M72 and M73 to obtain σ^2/L as was done for the clear air turbulence. As with the clear air turbulence, the specified rms intensities are used with both the Dryden and von Karman spectral forms, although they were developed specifically for the Dryden form.

Since only the most severe thunderstorms penetrate to altitudes much higher than about 40,000 ft, and, even when they do, they are often most intense below 40,000 ft, it has been arbitrarily decided that thunderstorm turbulence need not be considered above this altitude. Another factor contributing to this decision is that the storms may be avoided more easily at the higher altitudes. Of course, if it proves necessary, the thunderstorm turbulence model may be employed for the higher altitudes.

3.7.4.1, 3.7.4.2 THUNDERSTORM TURBULENCE (SCALES)

REQUIREMENT

3.7.4.1 Thunderstorm turbulence (von Karman scales). The scales for thunderstorm turbulence using the von Karman form are $L_u = L_v = L_w = 2500$ feet.

3.7.4.2 Thunderstorm turbulence (Dryden scales). The scales for thunderstorm turbulence using the Dryden form are $L_u = L_v = L_w = 1750$ feet.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

The values of scales given in the requirement are compatible with the spectral data of References M72 and M73, despite the fact that Reference M73 lists scale values larger than 2500 feet for the von Karman fits of measured spectra for severe thunderstorms. Since these measured spectra exhibit only the high-frequency asymptotic slope, they permit only the definition of some minimum scale. The scale of 3.7.4.1 is greater than the minimum indicated by the measured spectra and is used for consistency with the von Karman clear air turbulence scale.

3.7.5 APPLICATION OF THE TURBULENCE MODELS IN ANALYSES

REQUIREMENT

3.7.5 Application of the turbulence models in analyses. The gust velocities shall be applied to the airplane equations of motion through the aerodynamic terms only, and the direct effect of the gust on the aerodynamic sensors shall be included when such sensors are part of the airplane augmentation system. When using the discrete model, all significant aspects of the penetration of the gust by the airplane shall be incorporated in the analyses. Application of the continuous random model depends on the range of frequencies of concern in the analyses of the airframe. When structural modes are significant, the exact distribution of the gust velocities over the airframe should be considered. For this purpose, it is acceptable to consider u_g and v_g as being one-dimensional functions only of x, but w_g shall be considered two-dimensional, a function of both x and y, for the evaluation of aerodynamic forces and moments. When structural modes are not significant, airframe rigid-body responses may be evaluated by considering uniform gust immersion along with linear gradients of the gust velocities. The uniform immersion is accounted for by u_g , v_g , and w_g defined at the airplane center of gravity. The angular velocities due to the turbulence are equivalent in effect to the airplane angular velocities. These angular velocities are defined (precisely at very low frequencies only) as follows:

$$p_g = -\frac{\partial w_g}{\partial y} \quad , \quad q_g = +\frac{\partial w_g}{\partial x} = -\dot{\alpha}_g \quad , \quad r_g = -\frac{\partial v_g}{\partial x}$$

$$\Phi_{p_g}(\Omega) = \frac{\sigma_w^2}{L_w} \frac{0.8\left(\frac{\pi L_w}{4b}\right)^{1/3}}{1+\left(\frac{4b}{\pi}\Omega\right)^2} \qquad \text{where } b = \text{wing span}$$

$$\Phi_{q_g}(\Omega) = \frac{\Omega^2}{1+\left(\frac{4b}{\pi}\Omega\right)^2}\Phi_{w_g}(\Omega)$$

$$\Phi_{r_g}(\Omega) = \frac{\Omega^2}{1+\left(\frac{3b}{\pi}\Omega\right)^2}\Phi_{v_g}(\Omega)$$

The turbulence velocities u_g, v_g, w_g, p_g, q_g, and r_g are then applied to the airplane equations of motion through the aerodynamic terms. For longitudinal analyses u_g, w_g, and q_g gusts should be employed. For lateral-directional analyses v_g, p_g, and r_g should be used. The gust velocity components u_g, v_g, and w_g shall be considered mutually independent (uncorrelated) in a statistical sense. However, q_g is correlated with w_g, and r_g is correlated with v_g. The rolling velocity gust p_g is statistically independent of all the other gust components.

RELATED MIL-F-8785 PARAGRAPHS

None

DISCUSSION

General

Paragraphs 3.7.1, 3.7.2, 3.7.3, and 3.7.4 have specified the pertinent statistics for a random continuous and a discrete model of turbulence in terms of true turbulence velocities. As such, the turbulence velocities are to be considered as direct increments to the velocity components of the airplane at all points of the airplane. For purposes of analysis, the continuous turbulence is usually defined relative to the center of gravity of the airplane, which is the origin of the airplane body axes, but any suitable reference point may be used for convenience in application. Then, with respect to this chosen reference point, the continuous turbulence varies randomly in space over the entire length and span of the airplane at any given instant in time. Furthermore, the distribution of turbulent velocity over the vehicle changes as the airplane moves through the turbulent field. Likewise, any part or all of the airplane may be immersed in a discrete gust as the airplane penetrates the gust. Thus the distribution of the turbulent velocity over the airplane is a function of location on the airplane, whichever mode is used; and this distribution is important in turbulence response analyses. Because of the small vertical dimension of the airplane relative to the length and span, it is reasonable to ignore variations of the turbulence over the vertical dimension of the airplane. Again, this comment applies to both the random continuous and the discrete gust models.

The turbulence velocities produce incremental aerodynamic forces and moments that act on the airplane. Since all analyses, whether they be ultimately in the frequency or time domain, start with the airplane equations of motion, it is important to understand that the turbulence velocities, either random continuous or discrete, are applied to the equations of motion only through the aerodynamic terms. Turbulence increments are not applied to inertial (acceleration) terms in the equations. However, when aerodynamic sensors, such as angle-of-attack vanes, are considered in analyses of airplane augmentation and automatic flight control systems, the direct effect of the gust on the sensor must be included. Inertial sensors, such as accelerometers and rate gyros, do not sense the turbulence directly. These comments also, of course, apply to whichever turbulence model is used.

The turbulence velocity components are defined in MIL-F-8785B as being positive along the positive airplane body axes. As a result of this, the incremental angles of attack and sideslip are defined, using the conventional small-angle approximations, respectively as:

$$\alpha_g = -\frac{w_g}{V} \quad , \ rad$$

$$\beta_g = -\frac{v_g}{V} \quad , \ rad$$

With these definitions, the total angles of attack and sideslip to be used with the aerodynamic terms in the equations of motion are

$$\alpha_A = \alpha + \alpha_g$$

$$\beta_A = \beta + \beta_g$$

where $\alpha = w/V$ and $\beta = v/V$ are the nondimensional inertial velocity components. Since w_g and v_g have some distribution over the length and span of the airplane, so also do α_g and β_g. Furthermore α_g and β_g are not necessarily uniform over the airplane because of the angular velocities (p, q, and r) of the airplane and because of structural flexibility, if this is considered. Consequently, α_A and β_A vary over the length and span of the airplane.

The definitions of α_g and β_g given above permit definition of the one-dimensional spectral densities of α_g and β_g in terms of the one-dimensional spectra of w_g and v_g respectively:

$$\Phi_{\alpha_g}(\Omega) = \frac{1}{V^2}\,\Phi_{w_g}(\Omega)$$

$$\Phi_{\beta_g}(\Omega) = \frac{1}{V^2}\,\Phi_{v_g}(\Omega)$$

The two-dimensional spectrum of α_g is obtained similarly from the two-dimensional w_g spectrum. It is useful to note also that rms intensities are also related:

$$\sigma_{\alpha_g} = \frac{1}{V}\sigma_w \text{, rad}$$

$$\sigma_{\beta_g} = \frac{1}{V}\sigma_v \text{, rad}$$

Therefore, Figure 8 of 3.7.3 may be re-interpreted as a requirement for σ_{α_g}

Specific application of the random continuous turbulence model depends on the range of frequencies of concern in the analyses of the airframe, that is, whether the airframe is rigid or flexible. As stated in 3.7.5, when structural modes are significant, the exact distribution of the turbulence velocity components over the airframe should be considered. In general, each of the three gust velocity components is a function of three dimensions, x, y and z. For all ordinary airplanes, the z variation of the gust is negligibly small and will be ignored. For purposes of MIL-F-8785B, it is acceptable to consider u_g and v_g as being one-dimensional functions of x, but w_g must be considered two-dimensional, a function of both x and y, for evaluation of the aerodynamic forces and moments. Although for flying quality analyses, the y variations of u_g and v_g are generally of relatively minor importance, it is not necessarily so for analyses of airframe structural loads. It should be noted that for some unusual airframe configurations, particularly at the low altitudes of landing approach, the y variations of u_g may be significant, in which case the procuring activity will state specific requirements for u_g (if necessary). The spanwise (y) variation of w_g often produces significant lateral responses of the airplane and is, therefore, always to be considered, except possibly for very slender configurations when approved by the procuring activity. This idealization of the turbulence field permits idealization of the airframe, for calculation of the (generalized) aerodynamic forces and moments, as a two-dimensional lifting surface (in the x-y plane) for the vertical component w_g and as one-dimensional force distributions (along the x axis) for the horizontal components u_g and v_g. Interactions of lifting surfaces caused, for example, by turbulence-induced downwash fields must be considered in the analyses. This is the approach to be used generally. Many examples illustrating this approach for flexible airplanes in a more or less complete form may be found in the literature, e.g., References M74, M75, M76, M78, M26 and M30.

Random Gust Simplification for Rigid-Body Dynamics

A simpler approach, utilizing one-dimensional spectra, may be used for rigid-body responses of the airframe when the dynamics of structural modes are not significant. (The "rigid-body" derivatives may be corrected for static aeroelastic effects.) This approach, based on the work of Etkin in References B70, M79 and M37, assumes that the exact turbulent velocity distribution over the airframe may be adequately approximated by no more than the constant and linear terms of a Taylor's series expansion (about the airplane center of gravity) of the exact gust distribution. Consistent with the dimensionality assumptions mentioned previously, the Taylor's series, to first order, are:

$$u_g(x) = u_g(0) + \left.\frac{\partial u_g}{\partial x}\right|_0 x$$

$$v_g(x) = v_g(0) + \left.\frac{\partial v_g}{\partial x}\right|_0 x$$

$$w_g(x,y) = w_g(0,0) + \left.\frac{\partial w_g}{\partial x}\right|_0 x + \left.\frac{\partial w_g}{\partial y}\right|_0 y$$

where the 0 subscript on the first partial derivatives means that they are evaluated at the airplane center of mass. The uniform immersion is accounted for by u_g, v_g and w_g defined at the airplane center of gravity (c.g.). The linear gradients are accounted for by the four partial derivatives.

In their aerodynamic effect on the airframe, three of the linear gradients are considered equivalent to the airplane angular velocity components. (This situation is discussed in detail later in this paragraph).

Thus, the gradient $+\partial w_g / \partial x$, evaluated at the airplane c.g., is

equivalent aerodynamically to the inertial pitching velocity q , and similarly for two other gradients evaluated at the c.g.:

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

$$q_g = +\frac{\partial w_g}{\partial x}$$

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

As used in connection with the spectra for the linear gradients, however, these definitions are precise only at very low spatial or temporal frequency. The g subscript is used to distinguish these so-called gust angular velocities from the inertial angular velocities p, q and r. This equivalence of aerodynamic effects means that aerodynamic terms in the equations of motion that involve the angular velocity components are simply multiplied by the sum of the inertial and gust angular velocities (References M79 and B70). Therefore, as with α_A and β_A the total angular velocities are:

$$p_A = p + p_g$$

$$q_A = q + q_g$$

$$r_A = r + r_g$$

It should be mentioned here that the aerodynamic effects of gust gradients are not quite the same as the effects of the airplane angular velocities. For example, in rolling, the vertical tail experiences a side gust gradient in the z direction; this z gradient, however, is not present in p_g.

Therefore, strictly speaking, it is not correct to use the complete airplane derivative L_p to determine the gust-induced rolling moment. Generally, however, for p_g and q_g, effects such as these are small, so that the simple replacement of p and q with p_A and q_A in the aerodynamic terms is very satisfactory. When the airplane yaws, a significant spanwise gradient, $\partial u_g / \partial y$, is impressed on the wing; however, $\partial v_g / \partial x$ has no appreciable effect on the wing unless sweepback is extreme. Fortunately, the entire derivative Y_r is usually negligible unless the airplane has a large vertical tail (the effect of which would be accounted for by $\partial v_g / \partial x$), and the wing contribution to N_r is generally small compared to the vertical tail effect. Thus the effect of this discrepancy is normally limited to L_r which is difficult to estimate and may still be small anyway. So, for flying quality analyses, the simplified r_g (involving only the lengthwise gradient of the side gust) may well be adequate. In special cases where it is not, gust derivatives for each of the two gradients, $\partial u_g / \partial y$ and $\partial v_g / \partial x$ may be estimated separately and used.

In the Taylor's series expansion of the longitudinal or head-on gust, u_g, one linear gradient, $\partial u_g / \partial x$, has not yet been discussed. This gradient, which makes u_g different over the length of the airplane (e.g., different at the tail from at the wing), has no counterpart in the commonly used airplane stability derivatives. If $\partial u_g / \partial x$ is thought to be significant, a gust derivative may be derived easily that will permit incorporation of that effect. However, the use of $\partial u_g / \partial x$ is not required by MIL-F-8785B because its effect on flying qualities is believed to be negligible at the flight speeds of conventional airplanes.

The one unsteady aerodynamic effect that must be considered in this simpler approach for a rigid airframe is the effect of the time-rate-of-change of the vertical gust on the pitching moment. This is represented by

$$\dot{\alpha}_g = \frac{d\alpha_g}{dt} = \frac{\partial \alpha_g}{\partial x}\frac{dx}{dt} = \frac{\partial\left(\frac{-w_g}{V}\right)}{\partial x} V = -\frac{\partial w_g}{\partial x}$$

where $\frac{dx}{dt} = V$; therefore, $\dot{\alpha}_g = -q_g = \frac{-\partial w_g}{\partial x}$

This equation indicates a convenient relationship between $\dot{\alpha}_g$ and q_g that will be used here.

As an example to illustrate the use of the gust variables, the following linearized, two-degree-of-freedom longitudinal equations of motion are given:

LIFT $$\dot{\alpha} = q - L_\alpha(\alpha + \alpha_g) - L_\delta \delta$$

PITCHING MOMENT $$\dot{q} = M_\alpha(\alpha + \alpha_g) + M_{\dot{\alpha}}(\dot{\alpha} + \dot{\alpha}_g) + M_q(q + q_g) + M_\delta \delta$$

It should be emphasized that the gust variables, with the g subscript, appear only in the aerodynamic terms, not in the inertial terms. These equations also illustrate that only standard airplane stability derivatives (here in dimensional form) are used in this simpler approach for the flying quality analyses of assumed rigid airframes.

Spectra for Simplified p_g, q_g, r_g and $\dot{\alpha}_g$

Reference M37 presents expressions for the spectral densities of

$$\frac{\partial w_g}{\partial y} = -p_g, \quad \frac{\partial w_g}{\partial x} = q_g = -\dot{\alpha}_g \quad \text{and} \quad \frac{\partial v_g}{\partial x} = -r_g$$

which are based on the Dryden spectral form. The expressions are rather complicated algebraic and transcendental functions of the longitudinal spatial frequency Ω_1 ($\Omega_1 = k_1/L$, using the notation of Reference M37) and of the parameter k_2', which depends on the wing span of the airplane. For use in the flying quality analyses of assumed rigid airframes, the expressions for p_g, q_g and r_g can be simplified appreciably by curve fitting the exact expressions with simpler expressions that are functions of Ω^2 (the 1 subscript has been dropped.) The simplified expressions, which fit the exact expressions for the spectra over a wide range of Ω, are as given in 3.7.5. The three spectra depend on the wing span b.

The spectra of q_g and r_g are seen to be the spectra of w_g and v_g, respectively, multiplied by a factor that corresponds with an approximate differentiation of w_g and v_g by x. Ordinarily the spectra for q_g and r_g would be obtained according to the following simple formulas that represent exact differentiation:

$$\Phi_{q_g}(\Omega) = \Omega^2 \Phi_{w_g}(\Omega)$$

$$\Phi_{r_g}(\Omega) = \Omega^2 \Phi_{v_g}(\Omega)$$

but, if the spectra for w_g and v_g are those of either 3.7.2.1 or 3.7.2.2, the mean-squares of q_g and r_g are not defined since Φ_{q_g} and Φ_{r_g} would not tend to zero as Ω approaches infinity. To overcome this problem, the factors (representing the approximate differentiation) as shown in 3.7.5 are used to force the spectra of q_g and r_g to approach zero more rapidly as spatial frequency increases towards infinity. These factors are obtained, as mentioned before, by curve fitting the results of Reference M37. In Reference M37, the spectra for q_g and r_g are actually obtained as indicated by the simple formulas above (multiplication by Ω^2). The problem of undefined mean-squares for q_g and r_g does not occur in Reference M37 because the spectra of w_g and v_g in that reference decrease more rapidly to zero than the spectra for w_g and v_g in 3.7.2.2.

The spectrum of p_g given in 3.7.5 is a rational function that is a curve-fit of the very complicated algebraic and transcendental function given in Reference M37. As in the reference, it depends on the wing span b . In fact, both the high and low frequency asymptotes depend on the wing span. As a result of this, the p_g spectrum is not simply the w_g spectrum multiplied by some factor as is the case with the q_g and r_g spectra.

The spectral expression for p_g is to be used without change regardless of which spectral forms, von Karman or Dryden, are used for u_g , v_g and w_g . But, since the expressions for the q_g and r_g spectra in 3.7.5 are just modified versions of w_g and v_g spectra respectively, it would be possible to use either von Karman or Dryden spectra for w_g and v_g . However, for simplicity the spectra used for w_g and v_g may also be used in the expressions for q_g and r_g respectively. This is permissible because as Reference M37 states, the higher the order of the spectrum in terms of derivative of the velocity component, the smaller is its contribution to the overall airplane response. In other words, the spectra of u_g , v_g and w_g contribute appreciably greater portions of the airplane gust response than do p_g , q_g and r_g . Thus the significance of the inaccuracies in the curve fits is minimized. The significance of the differences between the p_g , q_g , and r_g spectra as would be developed from the von Karman and Dryden spectral forms is also minimized. Actually, in the formulas of Reference M37 the magnitude of the p_g spectrum decreases asymptotically according to Ω^{-3} at high frequencies, and the magnitudes of q_g and r_g decrease as Ω^{-1} . In 3.7.5 however, the Dryden spectra for p_g , q_g , and r_g are only approximations since all decrease asymptotically as Ω^{-2} at high frequencies. Although it is recommended that the formulas given in 3.7.5 be used, if more precise results are desired, the exact formulas of Reference M37 may be used with the appropriate value of k_z' . The spectra given in 3.7.5 are simpler, and they may be spectrally factored as will be described later.

Random Gust Simulation and Analysis

The random turbulent velocity components have been assumed to be statistically independent at all altitudes (see discussion for 3.7.3), with the following consequences for MIL-F-8785B:

(1) The three gust velocity components, u_g , v_g (or β_g) and w_g (or α_g) are mutually statistically independent (uncorrelated). This means that in simulations of airplane gust response, each of these components requires a separate (independent) noise source.

(2) Since q_g is linearly related to w_g (or α_g), q_g is correlated with w_g (or α_g); similarly r_g is correlated with v_g (or β_g).

(3) Also $\dot{\alpha}_g$ is correlated with q_g , since $\dot{\alpha}_g = -q_g$

(4) The rolling velocity gust p_g is statistically independent of (uncorrelated with) any and all of the other gust variables.

(5) The gust gradient $\partial u_g / \partial x$ (not generally required) is correlated with u_g .
(See Reference M37 for spectrum of $\partial u_g / \partial x$ and for the cross-spectral density between $\partial u_g / \partial x$ and u_g).

These statistical conclusions concerning the gust variables are of great importance in either frequency-domain or time-domain analyses of airplane gust response using the continuous random turbulence model. In all, for a six-degree-of-freedom simulation, mechanization of the turbulence model would require four independent random noise sources. For three-degree-of-freedom longitudinal analyses, the u_g, w_g (or α_g) and $q_g = -\dot{\alpha}_g$ (and possibly $\partial u_g/\partial x$) gusts are used, requiring two independent random noise sources in simulations. For three-degree-of-freedom lateral-directional analyses, the v_g (or β_g), p_g and r_g gusts are used, requiring again two independent noise sources for simulations.

Time history simulations of atmospheric turbulence, whether analog or digital, are probably most often obtained by passing a Gaussian random "white" noise signal through a process represented by a transfer function T (s) involving the Laplace transform variable s. It can easily be shown that the spectral density of the resulting time history is proportional to

$$T^*(j\omega)\,T(j\omega) = \left|T(j\omega)\right|^2$$

where s has been replaced by $j\omega$ ($j = \sqrt{-1}$) and the asterisk denotes the complex conjugate. Thus, if a power spectral density $\Phi(\omega)$ is a ratio of polynomials in ω^2, then it can be spectrally factored into the product of T $(j\omega)$ and its complex conjugate; and T $(j\omega)$ with s replacing $j\omega$ is the Laplace transform of a linear constant-parameter system. The Dryden spectral forms for u_g, v_g, w_g, p_g, q_g and r_g are spectrally factorable:

$$\Phi_{u_g}(\omega):\quad T_{u_g}(s) = \sigma_u\sqrt{\frac{2L_u}{\pi V}}\;\frac{1}{1+\frac{L_u}{V}s}$$

$$\Phi_{v_g}(\omega):\quad T_{v_g}(s) = \sigma_v\sqrt{\frac{L_v}{\pi V}}\;\frac{1+\frac{\sqrt{3}\,L_v}{V}s}{\left(1+\frac{L_v}{V}s\right)^2} = -V\,T_{\beta_g}(s)$$

$$\Phi_{w_g}(\omega):\quad T_{w_g}(s) = \sigma_w\sqrt{\frac{L_w}{\pi V}}\;\frac{1+\frac{\sqrt{3}\,L_w}{V}s}{\left(1+\frac{L_w}{V}s\right)^2} = -V\,T_{\alpha_g}(s)$$

$$\Phi_{p_g}(\omega):\quad T_{p_g}(s) = \sigma_w\sqrt{\frac{1}{L_w V}}\sqrt{\frac{0.8\left(\frac{\pi L_w}{4b}\right)^{\frac{1}{3}}}{1+\frac{4b}{\pi V}s}}$$

$$\phi_{q_g}(\omega): \quad T_{q_g}(s) = \frac{\frac{1}{V}s}{1 + \frac{4b}{\pi V}s} \cdot T_{w_g}(s) = -T_{\dot{\alpha}_g}(s)$$

$$\phi_{r_g}(\omega): \quad T_{r_g}(s) = \frac{-\frac{1}{V}s}{1 + \frac{3b}{\pi V}s} \cdot T_{v_g}(s)$$

Because q_g is correlated with w_g and r_g is correlated with v_g, the transfer function formulas above provide the relationships necessary to obtain cross-spectral densities of q_g and w_g, and of r_g and v_g, that are consistent with the given spectra:

$$\Phi_{q_g w_g}(\Omega) = \frac{-j\Omega}{1 - j\frac{4b}{\pi}\Omega} \Phi_{w_g}(\Omega), \quad \Phi_{w_g q_g} = \Phi^*_{q_g w_g}$$

$$\Phi_{r_g v_g}(\Omega) = \frac{+j\Omega}{1 + j\frac{3b}{\pi}\Omega} \Phi_{v_g}(\Omega), \quad \Phi_{v_g r_g} = \Phi^*_{r_g v_g}$$

These spectra are approximations of the more complicated formulas in Reference M37. The spectra above are converted to functions of ω in the same manner as previously described, and to functions of α_g and β_g as follows.

$$\Phi_{q_g \alpha_g} = -\frac{1}{V} \Phi_{q_g w_g}, \quad \Phi_{r_g \beta_g} = -\frac{1}{V} \Phi_{r_g v_g}$$

The von Karman spectra for u_g, v_g, and w_g in 3.7.2.1 are not spectrally factorable. In time-domain simulations, therefore, either the Dryden spectra must be used or, alternatively, the von Karman spectra must be curve-fitted to a satisfactory degree of approximation with a factorable spectral form for which a transfer function in s may be obtained. For p_g, only one spectral form is specified. As mentioned previously, w_g and v_g may be used to generate q_g and r_g respectively. See Reference M81 for the details of time history simulation of random continuous turbulence.

Frequency-domain expressions relating response spectra to the above input gust spectra are developed in References M56, M80 and M81, as well as M37. The simplified random continuous turbulence approach just described, although recommended for rigid-airframe analyses, should not preclude more detailed and precise approaches if such approaches are preferred or seem necessary.

Other Considerations

It has been noted that some data show significantly non-Gaussian gust distributions. Perhaps an exponential distribution is more realistic (Reference M83). Reference M84, which is now in publication, describes a way to operate on white noise to generate exponentially-distributed disturbances that have Dryden spectra.

For some analyses, much time can be saved by recognizing that a linearly-filtered white-noise input has an equivalent deterministic form (Reference M85). Thus, where Gaussian-distributed turbulence with Dryden spectra are acceptable, for example the equivalent deterministic input can reduce solution time from minutes to seconds in an analog computer analysis to determine rms aircraft responses.

Discrete Gust Analyses and Simulation

Finally, as stated in 3.7.5, all significant aspects of the penetration of the discrete gust by the airplane should be incorporated in analyses. This implies, also, that aerodynamic sensors such as angle-of-attack vanes may penetrate any part of the discrete gust at a different time than, for example, the wing or the tail of the airplane.

In analyses, the discrete gust is ordinarily used for one axis at a time, partly because of different length tuning requirements that may apply for each axis and partly because this has been traditional with discrete gusts. This usage is all that is required. However, the discrete gust may be used along two or more axes simultaneously in order to simulate the effect of a gust arbitrarily inclined relative to the airplane. If an inclined gust is used, its length d_m should be tuned in turn to each of the appropriate natural frequencies of the airplane and its flight control system (again, higher-frequency structural modes may be excepted). The magnitude of the inclined gust v_m is chosen from Figure 7 of 3.7.2.3, using consistent values of σ_w and L_w from 3.7.3 or from 3.7.4. These values of σ_w and L_w are to be used regardless of the direction of the inclined discrete gust relative to the airplane, in spite of the fact that σ_w and L_w refer to the vertical gust velocity component. Then, for use in the airplane equations of motion, the inclined gust velocity must be resolved in components along the pertinent airplane axes.

4 - QUALITY ASSURANCE

4. QUALITY ASSURANCE

REQUIREMENT

4.1 Compliance demonstration. Compliance with the requirements of Section 3 may be demonstrated through:

Analysis

Simulation

Flight Test

The methods for demonstrating compliance shall be established by agreement between the procuring activity and the contractor. 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 accurate extrapolation to the other conditions at which the requirements apply. Table XV gives general guidelines, but the peculiarities of the specific airplane design may require additional or alternate test conditions. The required failure analyses shall be thorough, excepting only approved Special Failure States (3.1.6.2.1).

4.2 Airplane States

4.2.1 Weights and moments of inertia. Terms in table XV such as "heaviest weight" and "greatest moment of inertia" mean the heaviest and greatest consistent with 3.1.2 and 3.1.3. When a critical center-of-gravity position is identified, the airplane weight and associated moments of inertia shall correspond to the most adverse service loading in which that critical center-of-gravity position is obtained.

4.2.2 Center-of-gravity positions. Terms in table XV such as "most forward c.g." and "most aft c.g." mean the most forward or aft consistent with 3.1.2. When a critical weight or moment of inertia is identified, the center-of-gravity position shall correspond to the most adverse service loading in which that critical weight or moment of inertia is obtained.

4.2.3 Thrust settings. Thrust settings shall be as listed in table XVI.

4.3 Design and test conditions

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

4.3.2 Special conditions. In addition to the flight conditions previously indicated, the speed-altitude combinations that result in the following shall all be investigated, where applicable:

a. Maximum normal acceleration response per degree of elevator deflection

b. Maximum normal acceleration response per pound of stick force

c. Highest dynamic pressure and highest Mach number.

4.4 Interpretation of qualitative requirements. In several instances throughout the specification, qualitative terms such as "objectionable flight characteristics", "realistic time delay", and "normal pilot technique", have been employed to permit latitude where absolute quantitative criteria might be unduly restrictive. Final determination of compliance with requirements so worded will be made by the procuring activity (1.5).

TABLE XV

DESIGN AND TEST CONDITION GUIDELINES

REQ'MT. NO.	TITLE	CRITICAL LOADING (4.2.1, 4.2.2)	LOAD FACTOR	ALTITUDE (4.3.1)	SPEED	FLIGHT PHASE
SECTION 3.2	LONGITUDINAL FLYING QUALITIES					
3.2.1.1	Longitudinal static stability	Most aft c.g.	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max} & transonic	CO,CR,LO,RR, FF,RT,PA,L, WO,TO,CT
3.2.1.1.1	Relaxation in transonic flight	↓	↓			
3.2.1.1.2	Elevator control force variations during rapid speed changes	—	As required		$V_{o_{min}}$ to $V_{o_{max}}$ & transonic	CO,GA,DE
3.2.1.2	Phugoid stability	Most forward c.g.†	1.0		V_{min} to V_{max}	CR,LO,PA,RT
3.2.1.3	Flight-path stability	—			$V_{o_{min}}$, $V_{o_{min}}$ - 5 kt	PA
3.2.2.1.1	Short-period frequency and acceleration sensitivity	Most forward c.g.† and most aft c.g.††			V_{min} to V_{max}	*,CR,RT,PA,L, CT
3.2.2.1.2	Short-period damping	Most forward c.g.			V_{min} to V_{max}	*,CR,RT,PA,L CT
3.2.2.1.3	Residual oscillations	—	↓		$V_{o_{min}}$ to $V_{o_{max}}$	*,PA
3.2.2.2	Control feel and stability in maneuvering flight	Most aft c.g.	n(-) to n(+)		V_{min} to V_{max}	*,RT,CR,PA,L, CT
3.2.2.2.1	Control forces in maneuvering flight	Most forward c.g.† and most aft c.g.††	n_o(-) to n(+)			
3.2.2.2.2	Control motions in maneuvering flight	Most forward c.g.†	n_o(-) to n(+)			↓
3.2.2.3	Longitudinal pilot-induced oscillations	—	Min. permissible to max. permissible			—
3.2.2.3.1	Transient control forces	Most forward c.g.† Most aft c.g.††	1.0			*,RT,CR,PA,L, CT
3.2.3.1	Longitudinal control in unaccelerated flight	Most forward c.g.	1.0		↓	—
3.2.3.2	Longitudinal control in maneuvering flight	Most forward c.g.†	As required	↓	$V_{o_{min}}$ to $V_{o_{max}}$	CO,GA,AR,TF, CR,PA
3.2.3.3	Longitudinal control in takeoff	Most forward c.g. for nose-wheel airplanes, most aft c.g. for tail-wheel airplanes	1.0	low	As required	TO
3.2.3.3.1	Longitudinal control in catapult takeoff	Most forward c.g. and most aft c.g.	As required		Min. safe launch speed to min. +30	CT
3.2.3.3.2	Longitudinal control force and travel in takeoff	Most forward c.g. and most aft c.g.	As required		0 to V_{max} (TO)	TO,CT
3.2.3.4	Longitudinal control in landing	Most forward c.g.	1.0		V_S (L) or geometric limit	L
3.2.3.4.1	Longitudinal control forces in landing	Most forward c.g.	1.0	↓		L
3.2.3.5	Longitudinal control forces in dives -Service Flight Envelope	Most forward c.g.† and most aft c.g.††	As required	2000 ft above MSL to h_{max}	V_{min} to V_{max}	D,ED,CO,CR
3.2.3.6	Longitudinal control forces in dives -Permissible Flight Envelope	↓	As required	As required	V_{MAT} to max permissible	D,ED,CO,CR
3.2.3.7	Longitudinal control in sideslips	—	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	CO,CR,PA,L

†Combined with heaviest weight

††Combined with lightest weight

TABLE XV (Cont.)

REQ'MT. NO.	TITLE	CRITICAL LOADING (4.2.1, 4.2.2)	LOAD FACTOR	ALTITUDE (4.3.1)	SPEED	FLIGHT PHASE
SECTION 3.3	LATERAL-DIRECTIONAL FLYING QUALITIES					
3.3.1.1	Lateral-directional oscillations (Dutch roll)	Greatest yawing moment of inertia	1.0 and $n_o(+)$	$h_{o_{min}}$, medium,	V_{min} to V_{max}	*,CR,RT,PA,L
3.3.1.2	Roll mode	Greatest rolling moment of inertia	1.0 and $n_o(+)$	$h_{o_{max}}$	V_{min} to V_{max}	*,CR,PA,L
3.3.1.3	Spiral stability	—	1.0		V_{min} to V_{max}	*,CL,CR,LO,RT DE,PA,L
3.3.1.4	Coupled roll-spiral oscillation	—	1.0 and $n_o(+)$			*,CR,PA,L
3.3.2.1	Lateral-directional response to atmospheric disturbances	—	1.0			—
3.3.2.2	Roll rate oscillations	—	1.0 and $n_o(+)$			*,CR,PA,L
3.3.2.2.1	Additional roll rate requirement for small inputs	—				
3.3.2.3	Bank angle oscillations	—				
3.3.2.4	Sideslip excursions	Greatest yawing and rolling moment of inertia	1.0			
3.3.2.4.1	Additional sideslip requirement for small inputs		1.0			
3.3.2.5	Control of sideslip in rolls	Greatest rolling moment of inertia	As required			CO,GA,AR,TF, CR,PA,L
3.3.2.6	Turn coordination	—			$V_{o_{min}}$	CO,CR,LO,PA
3.3.3	Pilot-induced oscillations	—	Min. permissible to max. permissible	MSL to h_{max}	V_{min} to V_{max}	—
3.3.4	Roll control effectiveness	Greatest rolling moment of inertia	As required (not above 0.8 n_L)	$h_{o_{min}}$, medium,		CO,GA,AR,TF, CR,PA,L
3.3.4.1.1	Air-to-air combat			$h_{o_{max}}$		CO
3.3.4.1.2	Ground attack with external stores			$h_{o_{min}}$		GA
3.3.4.1.3	Roll rate characteristics for ground attack					GA
3.3.4.1.4	Roll response	Smallest rolling moment of inertia		$h_{o_{min}}$, medium		—
3.3.4.2	Aileron control forces	Greatest and smallest rolling moment of inertia		$h_{o_{max}}$		CO,GA,AR,TF, CR,PA,L
3.3.4.3	Linearity of roll response					—
3.3.4.4	Wheel control throw	Greatest rolling moment of inertia				CO,GA,AR,TF, CR,PA,L
3.3.4.5	Rudder-pedal-induced rolls					CL,CR,D,PA
3.3.5	Directional control characteristics	—	n(-) to n(+)	$h_{o_{min}}$, medium,		*,CR,PA,L
3.3.5.1	Directional control with speed change	—	1.0	$h_{o_{max}}$		CO,GA,CR,D, PA,L
3.3.5.1.1	Directional control with asymmetric loading	—	1.0		$V_{o_{min}}$ to $V_{o_{max}}$	—
3.3.5.2	Directional control in wave-off (go-around)	Lightest weight	1.0	low	V_{min} (PA) or guaranteed landing speed	WO
3.3.6 (3.3.6.1, 3.3.6.2, 3.3.6.3, 3.3.6.3.1, 3.3.6.3.2)	Lateral-directional characteristics in steady sideslips		1.0	$h_{o_{min}}$, medium $h_{o_{max}}$	V_{min} to V_{max}	CO,CR,PA,L

TABLE XV (Cont.)

REQ'MT. NO.	TITLE	CRITICAL LOADING (4.2.1, 4.2.2)	LOAD FACTOR	ALTITUDE (4.3.1)	SPEED	FLIGHT PHASE
3.3.7	Lateral-directional control in cross winds	—	1.0	low	As required	TO,L
3.3.7.1	Final approach in cross winds	—	1.0		V_{min} to V_{max}	PA
3.3.7.2 (3.3.7.2.1 3.3.7.2.2)	Takeoff run and landing rollout in crosswinds	—	As required		As required	TO,L
3.3.7.3	Taxiing wind speed limits	—	As required		All taxiing speeds	TAXI
3.3.8	Lateral-directional control in dives	—	As required	2000 ft above MSL to h_{max}	V_{MAT} to V_{max}	D,ED
3.3.9.1	Thrust loss during takeoff run	Lightest weight	1.0	$h_{o_{min}}$	0 to max takeoff speed	TO
3.3.9.2	Thrust loss after takeoff		1.0		Down to V_{min}(TO)	TO,CT
3.3.9.3	Transient effects	Lightest weight	1.0	All	V_{min} to V_{max}	CO,GA,TF,CR, CL,TO,CT
3.3.9.4	Asymmetric thrust - rudder pedals free		1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	1.4 V_{min}	CR
3.3.9.5	Two engines inoperative		1.0		V_{range} (1 & 2 engines out)	—
SECTION 3.4	MISCELLANEOUS FLYING QUALITIES					
3.4.2.2 (3.4.2.2.1, 3.4.2.2.2)	Stall warning requirements	—	As required	MSL to h_{max}	1g stall warning speed to speed for which α_{stall} results in $n = n_{o_{max}}$	CO,AD,AR,TO, PA,CT WO,L
3.4.2.3 and 3.4.2.4 (3.4.2.4.1)	Stall characteristics and Stall recovery and prevention	—			V_s to speed for which α_{stall} results in $n = n_{o_{max}}$	
3.4.3	Spin recovery	—	As required	As required	V_s (to initiate spin)	—
3.4.4	Roll-pitch-yaw coupling	—	0 to 0.8 n_L	$h_{o_{min}}$, medium, $h_{o_{max}}$	$V_{o_{min}}$ to $V_{o_{max}}$	CO,GA,AR,TF
3.4.5 (3.4.5.1)	Control harmony	—	$n_o(-)$ to $n_o(+)$			—
3.4.6	Buffet	—				*
3.4.7	Release of stores	—				CO,GA,WD,AD
3.4.8	Effects of armament delivery and special equipment	—				*,RT
3.4.9	Transients following failures	—	all		all	—

TABLE XV (Cont.)

REQ'MT. NO.	TITLE	CRITICAL LOADING (4.2.1, 4.2.2)	LOAD FACTOR	ALTITUDE (4.3.1)	SPEED	FLIGHT PHASE
SECTION 3.5	CHARACTERISTICS OF THE PRIMARY FLIGHT CONTROL SYSTEM					
3.5.2 (3.5.2.1, 3.5.2.2, 3.5.2.3)	Mechanical characteristics	—	$n_o(-)$ to $n_o(+)$	$h_{o_{min}}$ and $h_{o_{max}}$	V_{min} to V_{max}	—
3.5.3 (3.5.3.1, 3.5.3.2)	Dynamic characteristics	most forward c.g. & lowest values of rolling and yawing moments of inertia	1.0			—
3.5.5 (3.5.5.1, 3.5.5.2)	Failures	—	all			—
3.5.6 (3.5.6.1, 3.5.6.2)	Transfer to alternate control modes	—	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$		—
SECTION 3.6	CHARACTERISTICS OF SECONDARY CONTROL SYSTEMS					
3.6.1	Trim system	most forward c.g. and most aft c.g.	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	—
3.6.1.1	Trim for asymmetric thrust	most forward c.g. and most aft c.g.	1.0	$h_{o_{min}}$ and max. attainable	V_{range} to V_{NRT} (with 1 & 2 engines out)	CR
3.6.1.2	Rate of trim operation	—	1.0	As required	As required	CO,GA,D,ED
3.6.1.3	Stalling of trim systems	most forward c.g. combined with heaviest weight	As required	As required	Start of dive recovery to V_{max}	D,ED,CO,CR
3.6.1.4	Trim system irreversibility	—	1.0	MSL to h_{max}	V_{min} to V_{max}	—
3.6.2	Speed and flight-path control devices	—	1.0 to $n_o(+)$	$h_{o_{min}}$, medium, $h_{o_{max}}$	$V_{o_{min}}$ to $V_{o_{max}}$	*,RT,ED,DE, PA,WO,GA
3.6.3	Transients and trim changes	—	$n_o(-)$ to $n_o(+)$	$h_{o_{min}}$, medium, $h_{o_{max}}$	$V_{o_{min}}$ to $V_{o_{max}}$	—
3.6.3.1	Pitch trim changes	—	As required	As required	As required	CO,CR,PA,TO,CT
3.6.4	Auxiliary dive recovery devices	most aft c.g. combined with lightest weight	As required	MSL to h_{max}	$V_{o_{min}}$ to V_{max}	D,ED
3.6.5	Direct normal-force control	—	1.0 ±maximum DLC authority	$h_{o_{min}}$, medium, $h_{o_{max}}$	$V_{o_{min}}$ to $V_{o_{max}}$	—
SECTION 3.7	ATMOSPHERIC DISTURBANCES	—	1.0	MSL to h_{max}	V_{min} to V_{max}	—

NOTES:

(1) a dash (-) indicates no general guidance can be provided.

(2) The phrase "as required" means the flight conditions are specified in the requirement or are determined by the nature of the test maneuver.

(3) An asterisk (*) means all applicable Category A Flight Phases.

RELATED MIL-F-8785 PARAGRAPHS

None specifically related

DISCUSSION

General

The philosophy underlying MIL-F-8785B is that the requirements should apply under those conditions in which the airplane operates. The requirements therefore apply in those flight regimes, with those loadings and external store combinations, and in those geometric configurations required by the airplane's missions plus failure considerations. Since normal or critical values of these parameters for each requirement are generally different for each airplane, it was not possible to specify them in this general specification.

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. Section 4 was drafted in order to provide guidelines to aid in the selection of design or flight test points.

In Table XV, guidance on loading, flight condition (speed, altitude load factor) and Flight Phase has been provided. The information on loading relates to critical weight, center-of-gravity and/or moment-of-inertia conditions with respect to each requirement. Information on flight condition relates to those ranges of speed, altitude, and load factor that are operationally significant and do not necessarily represent most critical conditions. (A more detailed discussion of load factor is presented below.) The Flight Phases specified are those which are either most critical or which are most intimately related to a given requirement.

The general approach taken in MIL-F-8785 was to specify loadings and flight conditions at which the requirement should be examined, as in Table II of MIL-F-8785 for example. In order to evaluate the merits of this approach, the specific information relating to each requirement in MIL-F-8785 has been tabulated in Table I of this section. The format of Table I is similar to that of Table XV in MIL-F-8785B so that the degree and type of guidance provided in MIL-F-8785B can be readily compared with that provided in MIL-F-8785. Examination of Table I reveals that, although MIL-F-8785 is specific in many areas, there are many requirements where no, or only partial, guidance is provided. In those areas where specific conditions are stated, it can be seen that in many cases the values quoted are not generally applicable. Thus, although MIL-F-8785 does limit the number of points to be examined for some requirements, the values selected are arbitrary. Furthermore, from examination of flight test reports and from a great many discussions with design and test organizations, it is obvious that many of the specified conditions in MIL-F-8785 are ignored. In fact these discussions have revealed that, although MIL-F-8785 is used as a guide, the points to be examined are generally determined from analysis of the characteristics of each airplane.

Section 4 of MIL-F-8785B attempts to recognize these facts of life, while providing general guidance.

Often it will be manifestly impossible to flight test all Airplane Failure States at all flight conditions. Since a comprehensive failure effect analysis (3.1.10.1) should exist at that stage, it can be used as a guide to more frequent failures and critical items that should be checked in flight. As with all other flight test planning, it will be necessary to use judgment to apply available resources most effectively.

When a flight condition is stated that varies with loading, flight test should be as near the critical loading as feasible, and the results corrected to the critical condition. An example of a requirement for which this is often necessary is 3.2.1.3, flight path stability: $V_{o_{min}}$ may be determined by a margin from stall speed and thus vary with gross weight.

Normal Load Factor as a Handling Qualities Parameter

Introduction

Although MIL-F-8785B is stated to apply at all positive and negative load factors in the Flight Envelopes, the similar requirements of -8785 have never been fully enforced. In fact, the only investigations (analytical or flight test) at $n \neq 1$ have often been of longitudinal maneuvering stability, and sometimes of inertia coupling. There have been some investigations of accelerated stalls, also, and plots of C_{n_β}, $C_{n_{\delta_a}}$, etc. at high Mach number vs. n or α.

Generally one would expect the effect of normal acceleration to be primarily the change in stability and control derivatives with angle of attack at constant Mach number. In incompressible flow the aerodynamic effect of angle of attack is largely reflected in equivalent-airspeed variation at 1g, with α proportional to $1/V_e^2$ (the exceptions are thrust and possible viscous-flow or aeroelastic effects which create significant u-derivatives). Static stability and control, therefore, are usually only weakly dependent on airspeed and altitude explicitly. The dependence becomes much stronger at higher Mach number. At transonic and supersonic speed, compressibility changes the derivatives with Mach number. But hypersonically, the derivatives again tend to be invariant with Mach number. So, to some extent, equivalent-airspeed variation has the same kind of effect as load-factor variation.

Dynamic characteristics, however, are functions of inertial as well as aerodynamic parameters. Consequently, in general they are functions of both angle of attack and airspeed even in incompressible flow. Neglecting changes in nondimensional stability derivatives except for square-law drag, the explicit dependence is approximately shown, for example, by these characteristic and control-response parameters:

$$\omega_{n_{sp}},\ 1/T_{h_2},\ 1/T_{h_3},\ \omega_{n_d},\ \omega_\phi \propto V_T\sqrt{\sigma} = V_e$$

$$\zeta_{sp},\ \zeta_d^*,\ \zeta_\phi \propto \sqrt{\sigma}$$

$$\omega_{n_p},\ 1/T_s,\ (\dot{\theta}/n)_{ss} \propto 1/V_T = \sqrt{\sigma}/V_e$$

$$1/T_{\theta_2},\ 1/T_R \propto V_T\sigma = V_e\sqrt{\sigma}$$

$$\zeta_p \propto V_T^2\sigma + k/(V_T^2\sigma) = V_e^2 + k/V_e^2$$

$$1/T_{h_1},\ d\gamma/du \propto V_T\sigma - k/(V_T^3\sigma) = V_e/\sigma - k\sqrt{\sigma}/V_e^3$$

$$n/\alpha \propto V_T^2\sigma = V_e^2$$

$$|\phi/\beta|_d \sim \text{independent of } V_T,\ V_e,\ \sigma$$

$$\phi_t \propto \frac{1}{\sigma}\left(kV_T\sigma t - 1 + e^{-kV_T\sigma t}\right) = \frac{1}{\sigma}\left(kV_e\sqrt{\sigma}t - 1 + e^{-kV_e\sqrt{\sigma}t}\right)$$

Wide differences in speed and altitude dependence are seen, and they are largely different from the $1/V_e^2$ dependence of α for 1-g flight. Variation of the nondimensional stability derivatives and I_{XZ} with angle of attack can also be extremely important; some of these considerations are mentioned later in this discussion. Since compliance must be demonstrated at several altitudes, the following discussion is restricted to speed-load-factor relationships.

With flight-test time becoming more expensive, it is unrealistic to demand full flight tests at $n \neq 1$. The saving grace is the computer, which can quickly survey the flight envelope, much less expensively, to determine critical points for flight test. Hopefully the flight tests will be extensive enough to verify the computer data.

Survey of Requirements at $n \neq 1$

Traditionally roll performance has been demonstrated at low g's -- starting from 1-g flight or, more commonly, from coordinated turns at bank angles up to half the requirement value (there was once a requirement for fighters to roll 100° in the first second; MIL-F-8785B calls for 30° to 90° bank-angle changes). Operationally, pilots commonly bank to pull g's. But if roll performance at high g's has been a problem on a given aircraft, it has normally been related to angle-of-attack effects on C_{l_β}, aileron yaw, etc., or to inertial coupling, rather than to decreased control effectiveness. Even elevons, that have reduced roll authority at large pitch deflections, have been satisfactory. But, on the other hand, for some maneuvers roll performance can be critical at $n > 1$ (See 3.3.4).

*ζ_d tends to decrease somewhat at low speed because of the contribution of $(N'_p - g/V_T)$.

Investigation of longitudinal short-period characteristics at $n > 1$ should be made for Class IV aircraft in Category A Flight Phases. Recent studies have shown that air-to-air tracking accuracy has deteriorated at $n > 1$ due to decreasing short-period damping ratio. Assurance of adequate longitudinal short-period characteristics should be obtained to the load factor limits if the Operational and Service Flight Envelopes. "Static" maneuvering stability at all load factors assures short-period stability at least, and there are separate pitch-up requirements. Positive-g stalls are important. Load factors less than 0 are generally avoided in operation and in flight test, but can be examined analytically for "static" maneuver characteristics.

Phugoid stability could be important in prolonged turns, as in large course changes at supersonic speeds; but the 1-g case has lower drag (less ζ_p) and so is more critical. There generally appears to be no point in investigating phugoid characteristics at $n \neq 1$.

Dynamic lateral-directional stability <u>should</u> be investigated at $n > 1$. For swept and low-aspect-ratio wings, $-C_{\ell_\beta}$ characteristically increases with α. One common supersonic phenomenon is the reduction in C_{n_β} with increasing α. $C_{n_{\delta_a}}$ also can vary with α, changing the roll-yaw coupling. These characteristics commonly deteriorate with increasing angle of attack and are important in turn entries and "steady" turns. With dynamic stability assured, there is no apparent need to check static lateral-directional stability at $n > 1$. Aircraft generally are not flown at appreciable sideslip angles in combination with n much different than 1.

Asymmetric thrust is an emergency condition. Though the possibility of trouble exists, loss of an engine in accelerated flight has not been investigated. The lateral-directional dynamics investigations at high g offer a measure of assurance here. Anyway, a pilot would quickly reestablish 1-g flight if he lost an engine. The 1-g investigation appears sufficient.

Trimmability is most important at 1-g. But especially at high speed, where prolonged turns are necessary to change heading, trimming into a turn may be desirable. That should be investigated, but normally would not be critical because large control deflections must be trimmed at low speed anyway. The transonic trim change, however, may well be aggravated by higher than 1-g angle of attack.

Transfer to an alternate control-system mode could well be more critical at $n > 1$, where a higher angle of attack is attained by pulling g's rather than by slowing down. The tendency to increased control sensitivity, $\Delta n / \Delta \delta$, with increasing speed would emphasize any failure transient that becomes more critical at high angle of attack.

Conclusions

In summary, for flight investigations of flying qualities at $n > 1$ the major emphasis should be on:

Stall, pitch-up, buffet

Maneuvering stability and control effectiveness

Longitudinal short-period dynamics for Class IV, Category A Flight Phases

Inertial coupling

Dynamic lateral-directional stability

Transonic trim change

Dive recoveries

Failure transients

Longitudinal short-period dynamics normally need not be investigated for other Classes and Categories at other than 1-g trim, but large-amplitude perturbations should be considered. At least qualitative short-period investigations should be made at $n < 1$, but in view of operational practice it seems unnecessary to conduct flight tests at steady-state negative g's. Qualitative evaluation of roll performance should usually be enough at high g's.

These generalities cannot be considered universally applicable. Intelligent use of wind tunnels and computers can indicate analytically the validity of these statements in a given case, and should be a valuable aid in flight-test planning as well.

Table I (Section 4)

MIL-F-8785 DESIGN AND TEST CONDITIONS

REQ'MT. NO.	TITLE	LOADING	LOAD FACTOR	ALTITUDE	SPEED	CONFIGURATION
3.1.6	Effects of armament provisions	Any flight condition in which operation of devices may be required in the conduct of airplane mission for normal operation.				
3.1.7	Release of stores	-	-	-	-	-
3.1.8	Deceleration devices	-	-	-	-	-
3.1.10	Effects of asymmetry	-	-	normal operation	normal operation	-
3.2.1	Control friction and breakout force	-	-	all attainable	all attainable	-
3.2.3	Rate of control displacement	-	-	-	-	-
3.2.5	Artificial stability devices	-	-	-	-	-
Section 3.3	<u>Longitudinal stability and control</u>					
3.3.1	Elevator-fixed static stability	aft critical	-	H, M, L	1.4 V_{SG} to V_{NRP}	CR
			-		all dive speeds	D
			-		V_{SG} to V_H	G
			-		V_{SL} to structural limit	L
			-		.75 V_{NRP} to V_H	P
			-		.85 $V_{R/C}$ (or 1.15 V_{SG}, whichever is larger) to 1.3 $V_{R/C}$	P(climb)
			-		V_{NRP} to V_M	CO
			-		V_{SL} (for -C, lower of V_{SL} or max. design arresting V) to structural limit speed	PA
3.3.1.1	(M_u stability)	aft critical	-		1.4 V_{SG} to V_{NRP}	CR
			-		all dive speeds	D
			-		V_{SG} to V_H	G
			-		V_{SL} to structural limit	L
			-		.75 V_{NRP} to V_H	P
			-		.85 $V_{R/C}$ (or 1.15 V_{SG}, whichever is larger) to 1.3 $V_{R/C}$	P(climb)
			-		V_{NRP} to V_M	CO
			-		V_{SL} (for -C, lower of V_{SL} or max. design arresting V) to structural limit speed	PA
			-		1.15 V_{SL}	WO
3.3.2	Elevator-free static stability	aft critical	-		1.4 V_{SG} to V_{NRP}	CR
			-		all dive speeds	D
			-		V_{SG} to V_H	G
			-		V_{SL} to structural limit	L
			-		.75 V_{NRP} to V_H	P
			-		V_{NRP} to V_M	CO
3.3.2.1	(Stick force vs. speed stability)	aft critical	-		$V_{max.\ range}$, 2 add'l.	CR
			-		1 representative V	D
			-		1.4 V_{SG}, 1 add'l.	G
			-		1.4 V_{SL}	L
			-		V_{NRP}, 1 add'l.	P
			-		$V_{R/C}$	P(climb)
			-		V_H, 1 add'l.	CO
			-		1.15 V_{SL}	PA
3.3.3	Exception in transonic flight	aft critical	(modifies requirements of 3.3.1, 3.3.1.1, 3.3.2, 3.3.2.1)		transonic range	

Table I (Section 4)

MIL-F-8785 DESIGN AND TEST CONDITIONS (Cont.)

REQ'MT. NO.	TITLE	LOADING	LOAD FACTOR	ALTITUDE	SPEED	CONFIGURATION
3.3.4	Stability in accelerated flight	-	all attainable	all	all	all
3.3.5 (3.3.5.1, 3.3.5.2, 3.3.5.3)	Short-period oscillations	all permissible	-	H, M, L	all permissible	-
3.3.6	Long-period oscillations	-	-	H, M, L	-	-
3.3.7	Control effectiveness in unaccelerated flight	all permissible	1.0	any	any permissible	all
3.3.8	Control effectiveness in accelerated flight	forward critical	see req'mt.	any permissible	any permissible	CR, D, G, L, P, P(climb), CO, PA
3.3.9 (3.3.9.2, 3.3.9.3, 3.3.9.4, 3.3.9.5)	Control forces in steady accelerated flight:					
	(stable stick force vs. n)	-	all attainable	H, M, L	-	-
	(magnitude of F_s/n_z)	-	all operational up to .85 n_L	H, M, L	all operational	P, CO, D, PA
3.3.9.1	(minimum F_s/n_z)	-	all permissible	H, M, L	all permissible	-
3.3.10	Control forces in sudden pullups	-	-	H, M, L	-	-
3.3.11	Control effectiveness in takeoff	most nose-heavy (for nose-wheel airplanes)	-	L	$V_{S_{TO}}$	TO
		most tail-heavy (for tail-wheel airplanes)				
3.3.12	Control in catapult takeoff	-	-		min. safe launch speed to 25 kt. higher than min.	TO
3.3.13	Control forces in takeoff	most nose-heavy (for nose-wheel airplanes)	-		$V_{S_{TO}}$ to 1.3 $V_{S_{TO}}$	TO
		most tail-heavy (for tail-wheel airplanes)				
3.3.14	Control effectiveness in landing	forward critical	-		V_{S_L}	L
3.3.15	Control force in landing	forward critical	-		V_{S_L}	D
3.3.16	Control forces in dives	-	-	H, M, L	any operational	D
3.3.16.1	(forces in dives to V_D)	-	-		any permissible	D
3.3.17	Auxiliary dive recovery device	aft critical	see req'mt.		any speed	D
3.3.18	Effects of drag devices	-	-		-	-
3.3.19	Longitudinal trim changes	-		See Table IV		
3.3.20	Longitudinal trim change caused by sideslip	-	-	H, M, L	$V_{max. range}$, 2 add'l.	CR
			-		1 represent. V	D
			-		1.4 V_{S_G}, 1 add'l.	G
			-		1.4 V_{S_L}	L
			-		V_{NRP}, 1 add'l.	P
			-		$V_{R/C}$	P(climb)
			-		V_H, 1 add'l.	CO
			-		1.15 V_{S_L}	PA
Section 3.4	<u>Lateral-directional stability and control</u>					
3.4.1	Damping of the lateral-directional oscillations	-	-	H, M	$V_{max. range}$	CR
			-	H, M, L	V_H	P
(3.4.1.1, 3.4.1.2)			-	H, M	(a) 0.9 V_H (b) stabilized speed in 50° dive	D
			-	H, M, L	V_H	CO
			-	L	1.15 V_{S_L}	PA
			-	L	1.4 V_{S_L}	L

Table I (Section 4)

MIL-F-8785 DESIGN AND TEST CONDITIONS (Cont.)

REQ'MT. NO.	TITLE	LOADING	LOAD FACTOR	ALTITUDE	SPEED	CONFIGURATION
3.4.2	Spiral stability		-	H, M	speed for max. range	CR
			-	H, M, L	0.75 V_{NRP} to V_H	P
			-	H, M, L	0.85 $V_{R/C}$ to 1.3 $V_{R/C}$	P
			-	H, M, L	V_{NRP} to V_M	CO
			-	H, M, L	V_{S_G} to V_H	G
			-	H, M, L	all dive speeds	D
			-	L	V_{S_L} to limit structural	L
			-	L	1.15 V_{S_L}	PA
3.4.3 (3.4.4, 3.4.5, 3.4.6, 3.4.7, 3.4.8)	Steady sideslip conditions	lightest normal service	-	H, M, L	$V_{max.\ range}$ plus 2 others	CR
			-	H, M, L	V_{NRP} plus 1 other	P
			-	H, M, L	$V_{R/C}$	P(climb)
			-	H, M, L	V_H plus 1 or more others	CO
			-	H, M, L	1.4 V_{S_G} plus 1 or more others	G
			-	H, M, L	1 or more dive speeds	D
			-	L	1.4 V_{S_L}	L
			-	L	1.15 V_{S_L}	PA
			-	L	1.15 V_{S_L}	WO
3.4.9	Adverse yaw	-	-	H, M, L	1.4 $V_{S_{CR}}$	CR
			-	L	1.4 $V_{S_{PA}}$	PA
3.4.10	Asymmetric power (rudder free)	lightest normal service	-	H, M, L	all speeds above 1.4 V_{S_G}	P
3.4.11	Directional control (symmetric power)	-	-	H, M, L	1.4 V_{S_G} to V_{NRP}	CR
			-	H, M, L	0.75 V_{NRP} to V_H	P
			-	H, M, L	0.85 $V_{R/C}$ to 1.3 $V_{R/C}$	P(climb)
			-	H, M, L	V_{NRP} to V_M	CO
			-	H, M, L	V_{S_G} to V_H	G
			-	H, M, L	all dive speeds	D
			-	L	V_{S_L} to limit structural	L
			-	L	V_{S_L} to limit structural	PA
			-	L	all speeds down to $V_{S_{PA}}$	WO
3.4.11.1	(10 degree sideslip requirement)	-		L	1.1 V_{S_L}	L
3.4.12	Directional control (asymmetric)	lightest normal service	-	L	all speeds above $V_{S_{TO}}$	TO
3.4.13 (3.4.13.2)	Directional control during takeoff and landing	-	-	L	-	TO
			-	L		L
3.4.13.1	(Carrier-based airplanes)		-	L	30 kts and above	TO
		-	-	L	30 kts and above	L
3.4.14	Direct. Cont. to counteract adverse yaw	-	-	H, M, L	1.4 $V_{S_{CR}}$	CR
			-	L	1.4 $V_{S_{PA}}$	PA
3.4.15	Directional Control in dives	-	-	H, M, L	any attainable operational	P
3.4.16 (3.4.16.1, 3.4.16.2, 3.4.16.3, 3.4.16.4, 3.4.16.5)	Lateral control	Class IV-P, CO - all spanwise weight distribution; L - all normal TO and land. loadings. Class II-L for 3.4.16.2 - light weight, heavy outboard concentration of spanwise weight.	-	H, M, L	1.1 V_{S_G} to M_H	P
				H, M, L	1.1 V_{S_G} to M_H	CO
				L	1.1 V_{S_G}	L
				L	1.1 V_{S_L}	PA

Table I (Section 4)

MIL-F-8785 DESIGN AND TEST CONDITIONS (Cont.)

REQ'MT. NO.	TITLE	LOADING	LOAD FACTOR	ALTITUDE	SPEED	CONFIGURATION
3.4.16.6	(Lateral control in dives)	-	-	H, M, L	any attainable operational	P
3.4.16.7	(high speed)	-	-	all	M_D	-
3.4.16.8	(linearity)	-	-	-	-	-
Section 3.5	General control and trimmability requirements					
3.5.1	Control for spin recovery	-	-	H, M, L	-	G, L
3.5.2	Control for taxiing	-	-	-	-	-
3.5.3	Control surface oscillations	-	-	H, M, L	all permissible	CR, D, G, L, P, P(climb), CO, PA
3.5.4	Primary flight control trimmability	forward and aft critical	-	H, M, L	1.2 V_{SCR} to V_H	P
			-	L	1.4 V_{SL} to structural limit	L
			-	L	1.4 V_{SL} (1.15 V_{SL} for -C) to struct. limit	PA
			-	H, M, L	$V_{max.\ range}$ to V_{NRP}	CR (2 engines out)
			-	H, M, L	any operational	D
3.5.5	Irreversibility of trim controls	-	-	H, M, L	-	-
3.5.6	Trim system failure	-	-	H, M, L	-	-
3.5.7	Roll-pitch-yaw coupling	-	0 to 2/3 n_L	all	all permissible	-
Section 3.6	Stall characteristics					
3.6.3 (3.6.3.1, 3.6.3.2)	Stall warning requirements	all permissible	1.0 to max. operational	H, M, L	stall warning speed	G, CR, L, PA
3.6.4 (3.6.4.1)	Requirements for acceptable stalling characteristics	all permissible	1.0 to max. operational	H, M, L	stall speed	G, CR, L, PA
Section 3.7	Requirements for power- and boost-control systems					
3.7.1	Normal control system operation	-	-	H, M, L	-	-
3.7.2	Power or boost failure	-	-	H, M, L	-	-
3.7.3	Transfer to alternate control system - trim change	-	-	H, M, L	level flight	-
			-	H, M, L	to V_H	D
			-	H, M, L	level flight	P
3.7.3.1	(transfer in configuration PA)	-	-	H, M, L	1.15 V_{SL}	PA
3.7.3.2	(transfer at low altitude)	-	-	sea level	1.4 V_{SL} and V_H	P
3.7.4	Longitudinal control on alternate syst.	most forward C.G. for combat loadings	see req'mt.	sea level	max. level flight	-
3.7.4.1	(dive recovery)	-	-	initiated from service ceiling	initiated from max. level flight speed	-
3.7.4.2	(landing)	most forward C.G. for normal service loadings	-	L	1.4 V_{SL} (1.15 V_{SL} for -C) to landing speed	PA
3.7.5	Lateral control on alternate system	-	-	H, M, L	-	-
			-	L	1.1 V_{SL}	L
3.7.6	Directional control on alternate syst.	-	-	L	1.4 V_{SL} (1.15 V_{SL} for -C) to landing speed	PA
3.7.7	Ability to trim on alternate system	-	-	H, M, L	all level flt. above 1.2 V_{SCR}	P
			-	L	all level flt. above 1.4 V_{SL}	L
			-	L	all level flt. above 1.4 V_{SL} (1.15 V_{SL} for -C)	PA
			-	H, M, L	all level flt. above max. range speed	CR
3.7.8	Feel system failure	-	-	H, M, L	-	-

Table I (Section 4)

MIL-F-8785 DESIGN AND TEST CONDITIONS (Cont.)

Note: A dash (-) means no specifics mentioned in requirement -

Section 3.1 provides the following:

(a) Loading (ref. 3.1.1): optional if not otherwise specified

(b) Flight conditions (ref. 3.1.2, 3.1.3): the requirements apply to all load factor - speed combinations within the operational V-n envelopes at the following 4 altitudes:

(1) Low (sea level)

(2) Medium (lower of 40% of service ceiling, or 40,000 feet)

(3) High (80% of service ceiling)

(4) Combat ceiling

(2)–(4) need not be considered for L, PA, MO, TO

5. PREPARATION FOR DELIVERY

REQUIREMENT

5.1 General. Not applicable to this specification.

6 - NOTES

6. NOTES

DISCUSSION

This section is basically an expansion of Section 6 of MIL-F-8785 to include clarification of the new parameters and concepts contained in MIL-F-8785B.

Many new definitions have been introduced into MIL-F-8785B. 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 V_S and the roll-sideslip coupling parameters, are explained more thoroughly in the discussions of the requirements to which they apply. Symbols used in Reference A1 are also listed alphabetically, along with other symbols used in this document, in the front of this volume.

Paragraphs 6.5, Engine considerations, and 6.6, Effects of aeroelasticity, control equipment, and structural dynamics, give general design guidance that is not discussed elsewhere in this document. Several new notes have been added, including a clarification of the stick force per g requirements (6.3), a note on gain scheduling (6.4), a clarification of the concept of Levels and a discussion of the computation of failure probabilities (6.7), a list of related documents (6.8), and marginal indicia (6.9).

For completeness, Section 6 of Reference A1 is given below. No further discussion seems necessary because the notes themselves are explanatory in nature.

RELATED MIL-F-8785 PARAGRAPHS

Section 6

REQUIREMENT

6.1 Intended use. This specification contains the flying qualities requirements for piloted airplanes and forms one of the bases for determination by the procuring activity of airplane acceptability. The specification serves 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. The requirements are intended to assure adequate flying qualities regardless of design implementation or flight control system mechanization. To the extent possible, this specification should be met by providing an inherently good basic airframe. Where that is not entirely feasible, or where inordinate penalties would result, a mechanism is provided herein to assure that the flight safety, flying qualities and realiability aspects of dependence of stability augmentation and other forms of system complication will be considered fully.

6.2 Definitions. Terms and symbols used throughout this specification are defined as follows:

6.2.1 General

S	- Wing area
s	- Laplace operator
q	- dynamic pressure
MSL	- mean sea level
T_2	- time to double amplitude; $T_2 = \left(\frac{-.693}{\zeta \omega_n}\right)$ for oscillations, $T_2 = -.693\tau$ for first-order divergences.
Airplane Normal States	- the nomenclature and format of table XVI shall be used in defining the Airplane 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	- altutude at a given airspeed at which rate of climb is 500 ft/min at stated weight and engine thrust
Cruising ceiling	- altitude at a given airspeed at which rate of climb is 300 ft/min at NRT at stated weight
h_{max}	- maximum service altitude (defined in 3.1.8.3)
$h_{o_{max}}$	- maximum operational altitude (3.1.7)
$h_{o_{min}}$	- minimum operational altitude (3.1.7)
c.g.	- airplane center of gravity

6.2.2 Speeds

Equivalent airspeed	- true airspeed multiplied by $\sqrt{\sigma}$, where σ is the ratio of free-stream density at the given altitude to standard sea-level air density
Calibrated airspeed	- airspeed-indicator reading corrected for position and instrument error but not for compressibility
Refusal speed	- the maximum speed to which the airplane can accelerate and then stop in the available runway length
M	- Mach number
V	- airspeed (where appropriate, V may be replaced by M in this specification), along the flight path

Table XVI

AIRPLANE NORMAL STATES

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

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

  a. speed for steady straight flight at $C_{L_{max}}$, the first local maximum of the curve of lift coefficient (L/qS) vs. angle of attack which occurs as C_L is increased from zero

  b. speed at which abrupt uncontrollable pitching, rolling or yawing occurs; i.e., loss of control about a single axis

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

  (Note that 3.1.9.2.1 allows an alternative definition of V_S in some cases.)

The airplane shall be initially trimmed at approximately 1.2 V_S with the following settings, after which the trim and throttle settings shall be held constant:

Conditions for Determining V_S

Flight Phase	Thrust Setting	Trim Setting
Climb (CL)	Normal climb	For straight flight
Descent (D)	Normal descent	For straight Flight
Emergency descent (ED)	Idle	For straight flight
Emergency deceleration (DE)	Idle	For straight flight
Takeoff (TO)	Takeoff	Recommended takeoff setting
Approach (PA)	Normal approach	For normal approach
Wave-off/go-around (WO)	Takeoff	For normal approach
Landing (L)	Idle	For normal approach
All other	TLF at 1.2 V_S	For straight flight

In flight test, it is necessary to reduce speed very slowly (typically 1/2 knot per second or less) to minimize dynamic lift effects. The load factor will generally not be exactly 1g when stall occurs; when this is the case, V_S is defined as follows:

$$V_S = \frac{V}{\sqrt{n_f}}$$

where V and n_f are the measured values at stall, n_f being the load factor normal to the flight path.

$V_S(X)$, $V_{min}(X)$, $V_{max}(X)$	- short-hand notation for the speeds V_S, V_{min}, V_{max} for a given configuration, weight, center-of-gravity position, and external store combination associated with Flight Phase X. For example, the designation $V_{max}(TO)$ is used in 3.2.3.3.2 to emphasize that the speed intended (for the weight, center of gravity, and external store combination under consideration) is V_{max} for the configuration associated with the takeoff Flight Phase. This is necessary to avoid confusion, since the configuration and Flight Phase change from takeoff to climb during the maneuver.
V_{trim}	- trim speed
V_{end}	- speed for maximum endurance
$V_{L/D}$	- speed for maximum lift-to-drag ratio
$V_{R/C}$	- speed for maximum rate of climb
V_{range}	- speed for maximum range in zero wind conditions
V_{NRT}	- high speed, level flight, normal rated thrust
V_{MRT}	- high speed, level flight, military rated thrust
V_{MAT}	- high speed, level flight, maximum augmented thrust
V_{max}	- maximum service speed (defined in 3.1.8.1)
V_{min}	- minimum service speed (defined in 3.1.8.2)
$V_{o_{max}}$	- maximum operational speed (3.1.7)
$V_{o_{min}}$	- minimum operational speed (3.1.7)

6.2.3 Thrust and power

Thrust and power	- For propeller-driven airplanes, the word "thrust" shall be replaced by the word "power" throughout the specification

TLF	- thrust for level flight
NRT	- normal rated thrust, which is the maximum thrust at which the engine can be operated continuously
MRT	- military rated thrust, which is the maximum thrust at which the engine can be operated for a specified period
MAT	- maximum augmented thrust: maximum thrust, augmented by all means available for the Flight Phase
Takeoff thrust	- maximum thrust available for takeoff

6.2.4 Control parameters

Elevator, aileron, rudder controls	- The stick or wheel and rudder pedals manipulated by the pilot to produce pitching, rolling, and yawing moments respectively; the cockpit controls
Elevator 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 perpendicualr to a line between the center of the stick grip or wheel and the stick or control column pivot
Aileron 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.
Rudder pedal 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
Control surface	- A device such as an external surface which is positioned by a cockpit control or stability augmentation to produce aerodynamic or jet-reaction type forces for controlling the attitude of the airplane. As used in this specification the elevator surface, aileron surface, and rudder surface are the control surfaces

or devices which are controlled by the stick or wheel and rudder pedals, and automatically by stability augmentation systems.

Direct normal force control - A device producing direct normal force for the primary purpose of controlling the flight path of the airplane. Direct normal force control is the descriptive title given to the concept of directly modulating the normal force on an airplane by changing its lifting capabilities at a constant angle of attack and constant airspeed or by controlling the normal force component of such items as jet exhausts, propellers, and fans.

Control power - Effectiveness of control surfaces in applying forces or moments to an airplane. For example, 50% of available aileron control power is 50% of the maximum rolling moment that is available to the pilot with allowable aileron control force.

6.2.5 Longitudinal parameters

ζ_{sp} - damping ratio of the short-period oscillation

$\omega_{n_{sp}}$ - undamped natural frequency of the short-period oscillation

ζ_p - damping ratio of the phugoid oscillation

ω_{n_p} - undamped natural frequency of the phugoid oscillation

n - normal acceleration or normal load factor, measured at the c.g.

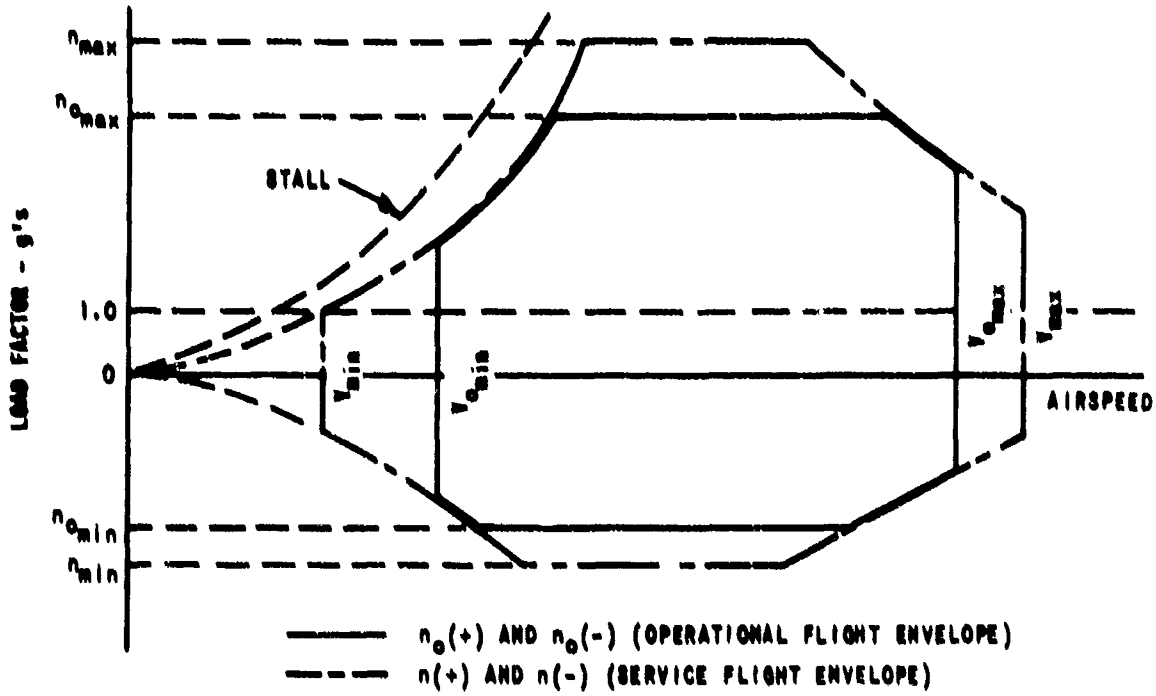
n_L - symmetrical flight limit load factor for a given Airplane Normal State, based on structural considerations

n_{max}, n_{min} - maximum and minimum Service load factors

$n(+)$, $n(-)$ - for a given altitude, the upper and lower boundaries of n in the V-n diagrams depicting the Service Flight Envelope

$n_{o_{max}}$, $n_{o_{min}}$ - maximum and minimum Operational load factors

$n_o(+)$, $n_o(-)$ - for a given altitude, the upper and lower boundaries of n in the V-n diagrams depicting the Operational Flight Envelope



α - angle of attack; the angle in the plane of symmetry between the fuselage reference line and the tangent to the flight path at the airplane center of gravity

α_S - the stall angle of attack at constant speed for the configuration, weight, center-of-gravity position and external-store combination associated with a given Airplane Normal State; defined as the lowest of the following:

a. Angle of attack for the highest steady load factor, normal to the flight path, that can be attained at a given speed or Mach number

b. Angle of attack, for a given speed or Mach number, at which abrupt uncontrollable pitching, rolling or yawing occurs, i.e., loss of control about a single axis

c. Angle of attack, for a given speed or Mach number, at which intolerable buffeting is encountered

d. An arbitrary angle of attack allowed by 3.1.9.2.1.

n/α - the steady-state normal acceleration change per unit change in angle of attack for an incremental elevator deflection at constant speed (airspeed and Mach number)

F_S/n	- gradient of steady-state elevator control force versus n at constant speed (3.2.2.2.1)
γ	- climb angle, $\gamma = \sin^{-1} \frac{\text{vertical speed}}{\text{true airspeed}}$, positive for climbing flight
L	- aerodynamic lift plus thrust component, normal to the flight path

6.2.6 <u>Lateral-directional parameters</u>

δ_{AS}	- displacement of the aileron control stick or wheel along its path
τ_R	- first-order roll mode time constant, positive for stable mode
τ_S	- first-order spiral mode time constant, positive for stable mode
λ_R	- $-1/\tau_R$
λ_S	- $-1/\tau_S$
ω_ϕ	- undamped natural frequency of numerator quadratic of ϕ/δ_{AS} transfer function
ζ_ϕ	- damping ratio of numerator quadratic of ϕ/δ_{AS} transfer function
ω_{n_d}	- undamped natural frequency of the Dutch roll oscillation
ζ_d	- damping ratio of the Dutch roll oscillation
T_d	- damped period of the Dutch roll, $T_d = \frac{2\pi}{\omega_{n_d}\sqrt{1-\zeta_d^2}}$
ϕ	- bank angle measured in the y-z plane, between the y-axis and the horizontal (6.2.1)
ϕ_t	- bank angle change in time t, in response to control deflection of the form given in 3.3.4
p	- roll rate about the x-axis (6.2.1)
$\frac{p_{osc}}{p_{AV}}$	- a measure of the ratio of the oscillatory component of roll rate to the average component of roll rate following a rudder-pedals-free step aileron control command:

$$\zeta_d \leq 0.2: \quad \frac{p_{osc}}{p_{AV}} = \frac{p_1 + p_3 - 2p_2}{p_1 + p_3 + 2p_2}$$

$$\zeta_d > 0.2: \quad \frac{p_{osc}}{p_{AV}} = \frac{p_1 - p_2}{p_1 + p_2}$$

where p_1, p_2 and p_3 are roll rates at the first, second and third peaks, respectively. (figures 9 and 10).

$\frac{\phi_{osc}}{\phi_{AV}}$ - a measure of the ratio of the oscillatory component of a bank angle to the average component of bank angle following a rudder-pedals-free impulse aileron control command:

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

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

where ϕ_1, ϕ_2 and ϕ_3 are bank angles at the first, second and third peaks, respectively.

β - sideslip angle at the center of gravity, angle between undisturbed flow and plane of symmetry. Positive, or right, sideslip corresponds to incident flow approaching from the right side of the plane of symmetry.

$\Delta\beta_{max}$ - maximum sideslip excursion at the c.g., occurring within two seconds or one half-period of the Dutch roll, whichever is greater, for a step aileron-control command

k - ratio of "commanded roll performance" to "applicable roll performance requirement" of 3.3.4 or 3.3.4.1, where:

(a) "Applicable roll performance requirement", $(\phi_t)_{requirement}$, is determined from 3.3.4 and 3.3.4.1 for the Class, Flight Phase Category and Level under consideration.

(b) "Commanded roll performance", $(\phi_t)_{command}$, is the bank angle attained in the stated time for a given step aileron command with rudder pedals employed as specified in 3.3.4 and 3.3.4.1

$$k = \frac{(\phi_t)_{command}}{(\phi_t)_{requirement}}$$

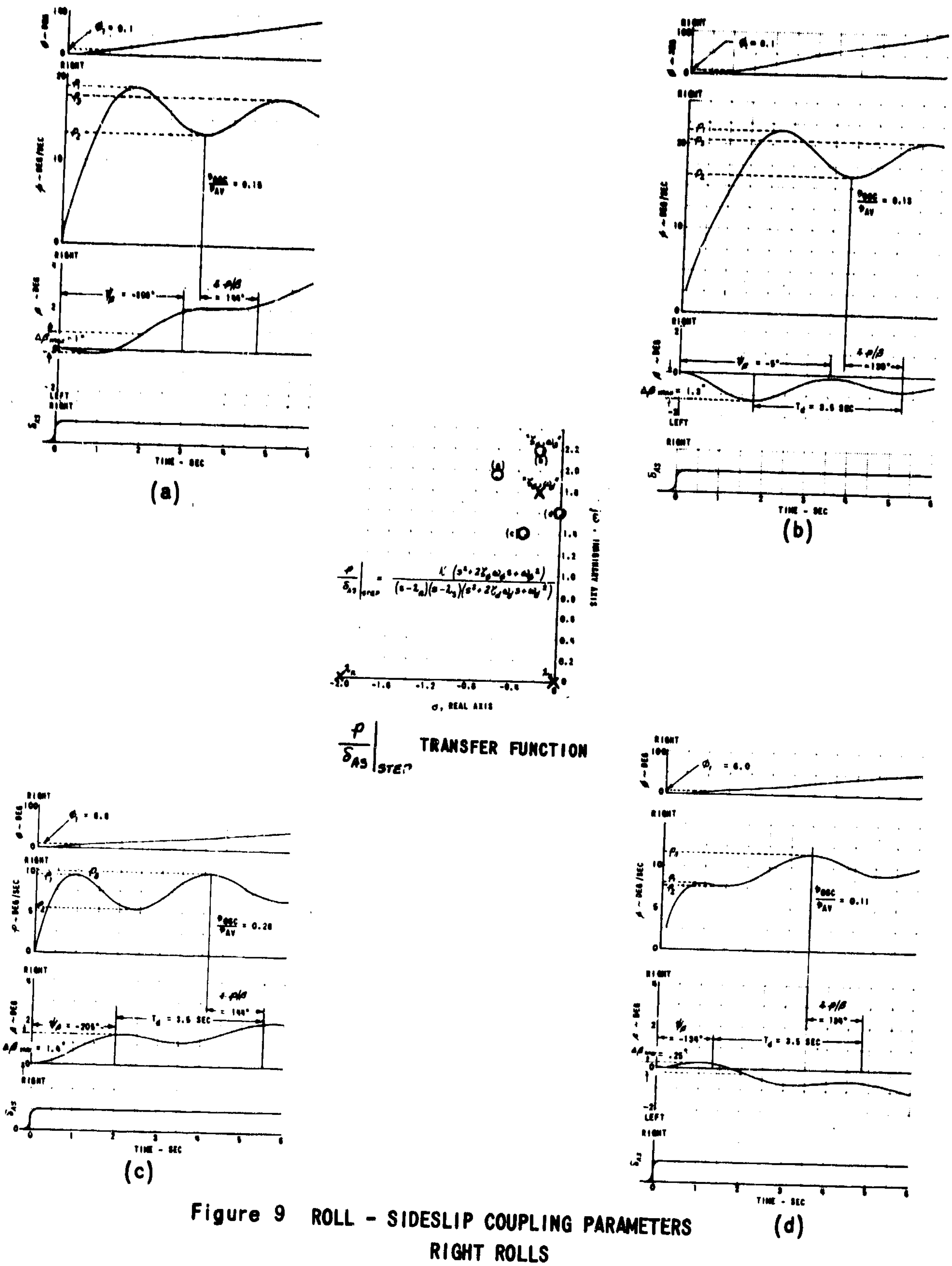


Figure 9 ROLL - SIDESLIP COUPLING PARAMETERS RIGHT ROLLS (d)

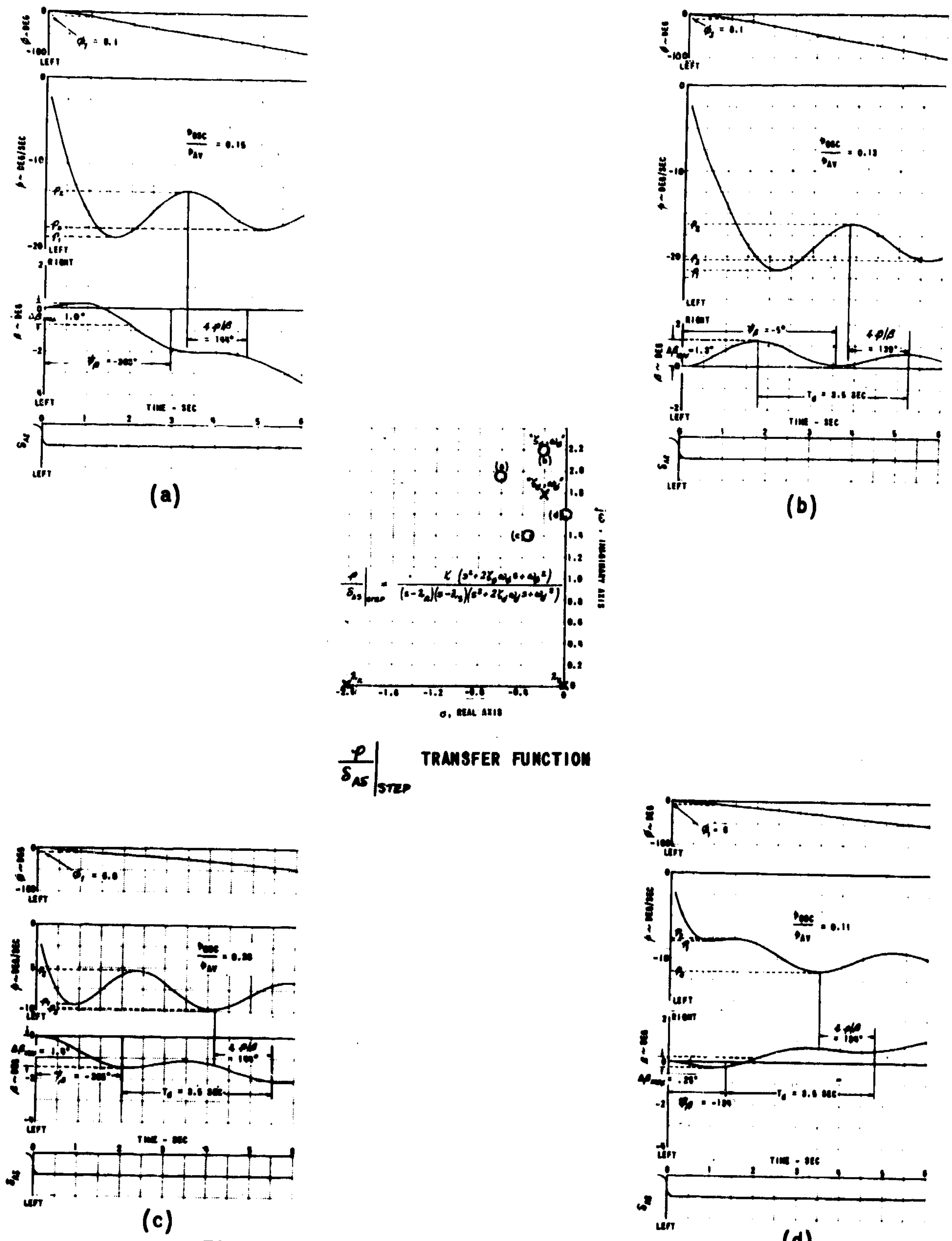


Figure 10 ROLL - SIDESLIP COUPLING PARAMETERS
LEFT ROLLS

t_{n_β} - time for the Dutch roll oscillation in the sideslip response to reach the n^{th} local maximum for a right step or pulse aileron-control command, or the n^{th} local minimum for a left command. In the event a step control input cannot be accomplished, the control shall be moved as abruptly as practical and, for purposes of this definition, time shall be measured from the instant the cockpit control deflection passes through half the amplitude of the commanded value. For pulse inputs, time shall be measured from a point halfway through the duration of the pulse.

ψ_β - phase angle in a cosine representation of the Dutch roll component of sideslip - negative for a lag

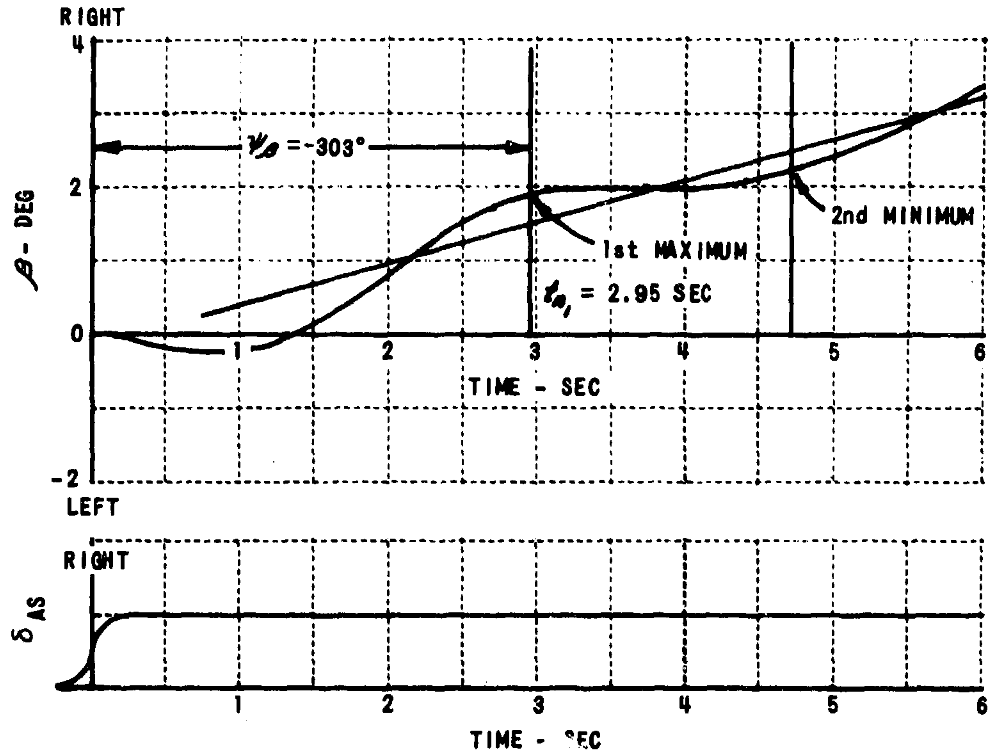
$$\psi_\beta = \frac{-360}{T_d} t_{n_\beta} + (n-1)360 \text{(degrees)}$$

with n as in t_{n_β} above

$\measuredangle \frac{p}{\beta}$ - phase angle between roll rate and sideslip in the free Dutch roll oscillation. Angle is positive when p leads β

$|\phi/\beta|_d$ - at any instant, the ratio of amplitudes of the bank-angle and sideslip-angle envelope in the Dutch roll mode

Examples showing measurement of roll-sideslip coupling parameters are given in figure 9 for right rolls and figure 10 for left rolls. Since several oscillations of the Dutch roll are required to measure these parameters, and since for proper identification large roll rates and bank angle changes must generally be avoided, for flight test, step aileron inputs should generally be small. It should be noted that since ψ_β is the phase angle of the Dutch roll component of sideslip, care must be taken to select a peak far enough downstream that the position of the peak is not influenced by the roll mode. In practice, peaks occurring one or two roll mode time constants after the aileron input will be relatively undistorted. Care must also be taken when there is ramping of the sideslip trace, since ramping will displace the position of a peak of the trace from the corresponding peak of the Dutch roll component. In practice, the peaks of the Dutch roll component of sideslip are located by first drawing a line through the ramping portion of the sideslip trace and then noting the times at which the vertical distance between the line and the sideslip trace is the greatest. (See following sketch for Case (a) of figures 9 and 10.)



Since the first local maximum of the Dutch roll component of the sideslip response occurs at t = 2.95 seconds,

$$\psi_\beta = \frac{-360}{T_d} t_{n_\beta} + (n-1)360 = \frac{-360}{3.5}(2.95) = -303^\circ$$

Level 1 flying qualities of a Class IV airplane in the approach are under examination; so the roll performance requirement from table IX upon which the parameter "k" in the sideslip excursion requirement (figure 6) is based, is ϕ_t = 30 degrees in 1 second with rudder pedals free (as in the rolls of 3.3.2.4). From the definitions, "k" for this condition is,

$$k = \frac{(\phi_1)\ command}{(\phi_1)_{requirement}}$$

Therefore from figures 9 and 10 for:

Case (a), $k = \frac{9.1}{30} = 0.30$

Case (b), $k = \frac{8.1}{30} = 0.27$

Case (c), $k = \frac{6.8}{30} = 0.23$

Case (d), $k = \frac{6.0}{30} = 0.20$

6.2.7 Atmospheric disturbances parameters

Ω	- spatial (reduced) frequency (radians per foot)
ω	- temporal frequency (radians per second), where $\omega = \Omega V$
u_g	- random gust velocity along the x body axis (feet per second)
v_g	- random gust velocity along the y body axis (feet per second)
w_g	- random gust velocity along the z body axis (feet per second)
	Note: u_g, v_g, w_g have Gaussian (normal) distributions, and are defined positively along the positive airplane body axes.
σ	- root-mean-square gust intensity, where $\sigma^2 = \int_0^\infty \Phi(\Omega)\, d\Omega = \int_0^\infty \phi(\omega)\, d\omega$
σ_u	- root-mean-square intensity of u_g
σ_v	- root-mean-square intensity of v_g
σ_w	- root-mean-square intensity of w_g
L_u	- scale for u_g (feet)
L_v	- scale for v_g (feet)
L_w	- scale for w_g (feet)
$\Phi_{u_g}(\Omega)$	- spectrum for u_g, where $\Phi_{u_g}(\Omega) = V\phi_{u_g}(\omega)$
$\Phi_{v_g}(\Omega)$	- spectrum for v_g, where $\Phi_{v_g}(\Omega) = V\phi_{v_g}(\omega)$
$\Phi_{w_g}(\Omega)$	- spectrum for w_g, where $\Phi_{w_g}(\Omega) = V\phi_{w_g}(\omega)$
v_m	- generalized discrete gust velocity, positive along the positive airplane body axes, m = x, y, z (feet per second)
d_m	- generalized discrete gust length (always positive) m = x, y, z (feet)

6.3 <u>Interpretation of the F_s/n limits of table V.</u> Because the limits on F_s/n are a function of both n_L and n/α, table V is rather complex. To illustrate its use, the limits are presented on figure 11 for an airplane having a center-stick controller and $n_L = 7.0$.

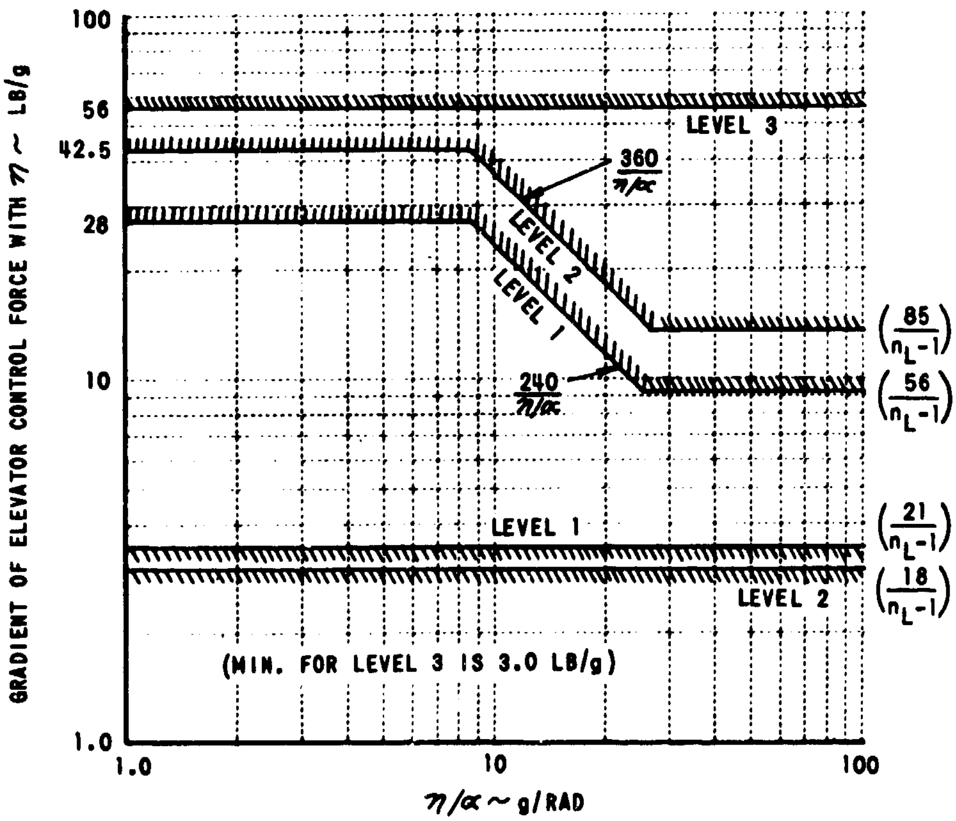


Figure 11 EXAMPLE OF ELEVATOR MANEUVERING FORCE GRADIENT LIMITS: CENTER-STICK CONTROLLER, $n_L = 7.0$

6.4 Gain scheduling. Changes of mechanical gearings and stability augmentation gains in the primary flight control system are sometimes accomplished by scheduling the changes as a function of the settings of secondary control devices, such as flaps or wing sweep. This practice is generally acceptable, but gearings and gains normally should not be scheduled as a function of trim control settings since pilots do not always keep airplanes in trim.

6.5 Engine considerations. Secondary effects of engine operation may have an important bearing on flying qualities and should not be overlooked in design. These considerations include such effects as engine gyroscopic moments influencing airframe dynamic motions, the effects of engine operation on spin characteristics and spin recovery, and the variation of engine-derived power for actuating the flight controls with engine speed.

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

6.7 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.

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

a. Identify those Airplane 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 Airplane 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 Airplane Failure State in terms of Levels as defined in the specific requirements.

e. Determine the most critical Airplane 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 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. An example which illustrates an approximate estimate of the probabilities of encounter follows: if the failures are all statistically independent, determine the sum of the probabilities of encountering all Airplane Failure States which degrade flying qualities to Level 2 in the Operational Envelope. This sum must be less than 10^{-2} per flight.

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

a. Improve the airplane 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 Airplane Failure States (with the exception of Special Failure States) the flying qualities shall not degrade below Level 3.

6.7.2 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. 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 Airplane Normal States (3.1.10.1). Airplane Failure States that do not degrade flying qualities beyond this common boundary are not considered in meeting the requirements of 3.1.10.2. Airplane 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.

6.7.3 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. 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 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 Airplane Failure States might fall in the Airplane Special Failure State classification (3.1.6.2.1).

6.8 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-A-8860	Airplane Strength and Rigidity - General Specification for
MIL-A-8861	Airplane Strength and Rigidity - Flight Loads
MIL-S-38130	Safety Engineering of Systems and Associated Subsystems, and Equipment, General Requirements for
MIL-G-38478	General Requirements for Angle of Attack Based Systems

PUBLICATION

USAF

HIAD-Handbook of Instructions for Airplane Designers

6.9 Marginal indicia. Asterisks are not used in this revision to identify changes with respect to the previous issue due to the extensiveness of the changes.

BIBLIOGRAPHY

BIBLIOGRAPHY

Reports are arranged by subject into sections as indicated below. Within each section, the reports are, in general, listed in chronological order with most recent reports first. New reports which have been published most recently are added to the end of each section, and are indicated by a dashed line in the list.

Section		Page
A	Proposed and Current Specification and Specification Reviews	502
B	General Flying Qualities Topics	505
C	General Landing Approach Topics	518
D	Longitudinal Flying Qualities	526
E	Longitudinal Flying Qualities - Landing Approach	533
F	Lateral-Directional Flying Qualities	537
G	Lateral-Directional Flying Qualities - Landing Approach	547
H	Pilot-Induced Oscillations	549
J	Control Systems and SAS Equipment	551
K	Spins and Stalls	560
L	Aerodynamic and Inertia Coupling Effects	562
M	Structural Mode Response, Turbulence Response, and Ride Qualities	563
N	Simulation Techniques, Pilot Models, and Human Factors	573
P	Test Data on Specific Airplanes - USAF Flight Test Center	579

A. PROPOSED AND CURRENT SPECIFICATION AND SPECIFICATION REVIEWS

A1. Anon: Military Specification, Flying Qualities of Piloted Airplanes. MIL-F-8785B(ASG), 7 August 1969.

A2. Anon: Revision of Military Airplane Flying Quality Requirements, MIL-F-8785 (ASG)-4. Proposed by Aerospace Industries Association (AIA) Materials and Structures Committee (MSC) Task Group. January 1967.

A3. Anon: Flying Qualities Conference, Wright-Patterson Air Force Base, Ohio, 5 and 6 April 1966. AFFDL-TR-66-148, USAF, December 1966.

A4. Carlson, John W. and Wilson, Richard K.: (AF Flight Dynamics Laboratory): Flying Qualities Criteria Problems and Some Proposed Solutions. FDCC TM 66-8, (AGARD Stability and Control Meeting, Cambridge, England, September 1966.)

A5. Westbrook, C. W.: The Status and Future of Flying Qualities Requirements. AIAA ' per 65-313, July 1965.

A6. Anon: Airworthiness Standards: Transport Category Airplanes. FAR Part 25, FAA, 1 February 1965 (Through Change 16, Effective 20 June 1968).

A7. Anon: Provisional Acceptable Means of Compliance - Aeroplane Flying Qualities. ICAO Circular 75-AN/65, ICAO Air orthiness Committee, 1965.

A8. Anon: Recommendations for V/STOL Handling Qualities. Report 408A, NATO Advisory Group for Aeronautical Research and Development, October 1964.

A9. Anon: Design Objectives for Flying Qualities of Civil Transport Aircraft. SAE ARP 842, Subcommittee S-7M of SAE Committee S-7, 1 August 1964.

A10. Anon: Design Requirements for Aircraft for the Royal A.F. and Royal Navy - Part 6, Aerodynamics and Flying Qualities. AV.P. 970, Vol. I, British Ministry of Aviation, Amended July 1964.

A11. Anon: Supersonic Transport Aeroplane Flying Qualities. TSS Standard 5, Anglo-French SST Specifications, 22 May 1964.

A12. Anon: Flying Qualities Specification. L-O-1R10, SAAB Report, Sweden, October 1963.

A13. Anon: Tentative Airworthiness Objectives and Standards for Supersonic Transport Design Proposals. Flight Standards Service, FAA, 15 August 1963.

A14. Mazza, C.J., Becker, William, et al.: "Flying Qualities of Piloted Airplanes" (MIL-F-8785ASG) With Substantiating Test. NADC-ED-6282, Navy, 22 July 1963.

A15. Koven, W. and Wasicko, R.: Flying Qualities Requirements for U.S. Navy and Air Force Aircraft. AGARD Report 336, 10 April 1961.

A16. Magruder, W.: A Summary of Past and Present Military and Civil Stability and Control Specification Requirements for Piloted Aircraft. Douglas Report DEV-2970, 15 May 1959.

A17. Anon: Military Specification, Flying Qualities of Piloted Airplanes. MIL-F-8785(ASG). 1 September 1954 (4 amendments thru 17 April 1959).

A18. Flying Qualities of Piloted Airplanes. SR-119B, Navy BuAer Specification, and C-1815B, USAF Specification, 1 June 1948.

A19. Airworthiness Standards: Normal, Utility, and Acrobatic Category Airplanes. FAR Part 23, FAA, 1 February 1965 (Through Change 6, Effective 1 August 1967).

A20. High-Speed Characteristics. FAA Advisory Circular AC25.253-1, 24 November 1965 (CH1, 10 January 1966).

B. GENERAL FLYING QUALITIES TOPICS

B1. Analysis of Several Handling Quality Topics Pertinent to Advanced Manned Aircraft. AFFDL-TR-67-2 USAF, Systems Technology, Inc., June 1967.

B2. O'Hara, F: "Handling Criteria". Journal of the Royal Aeronautical Society, Vol. 71, April 1967.

B3. White, A.S. and Anderson, P.H.: Operational Experience at Mach 3. AGARD Stability and Control Meeting, Cambridge, England, September 1966.

B4. Breuhaus, W.O.: Recent Experience with In-Flight Simulation. AGARD Stability and Control Meeting, Cambridge, England, September 1966.

B5. Harper, R.P. and Cooper, G.E.: A Revised Pilot Rating Scale for the Evaluation of Handling Qualities. Cornell Aeronautical Laboratory Report No. 153, (AGARD Stability and Control Meeting, Cambridge, England, September 1966.)

B6. Caiger, B.: Some Problems in Control Arising From Operational Experiences with Jet Transports. NAE Misc. 41, National Aeronautical Establishment, Ottawa, Canada, August 1966, (AGARD Stability and Control Meeting, Cambridge, England, September 1966.)

B7. Vaucheret, Xavier: Stabilité Dynamique sur Maquettes d'Avion par La Methode des Oscillations Libres. T.P. n°390, Office National d'Etudes et de Recherches Aerospatiales, Chatillon, France, (AGARD Stability and Control Meeting, Cambridge, England, September 1966.)

B8. Wiley, Harleth G.: The Significance of Nonlinear Damping Trends Determined for Current Aircraft Configurations. AGARD Stability and Control Meeting, Cambridge, England, September 1966.

B9. Christopher, P. A. T.: Studies in Stability and Control Analysis of Airframes Having Nonlinear Aerodynamic Characteristics. CoA Note Aero. No. 171, The College of Aeronautics, (AGARD Stability and Control Meeting, Cambridge, England, September 1966.)

B10. Parkin, G. B.: Wide Angle Motions on Buccaneer. Note No. YWT. 1424 LS., Hawker Siddeley Aviation Ltd., Brough, Yorkshire, England, July 1966, (AGARD Stability and Control Meeting, Cambridge, England, September 1966.)

B11. Woodcock, R. J. and Drake, D. E.: Estimation of Flying Qualities of Piloted Airplanes. AFFDL-TR-65-218, USAF, April 1966.

B12. Carlson, J. W.: C-5A Handling Qualities Simulations. SAE Committee A-18, Aero-Space Vehicle Flight Control Systems, 19 January 1966.

B13. Leyman, C. and Mittall E. R.: A Survey of Aircraft Handling Criteria. Current Paper 833, British Aeronautical Research Council, 1966.

B14. Bisgood, P. L.: Results of Flight Tests on a Slender Wing Low Speed Research Aircraft (H. P. 115). AGARD Report 535, 1966.

B15. Hoey, Robert G.: Correlation of X-15 Simulation Experience with Flight Test Results. AGARD Report 530, 1966.

B16. Eldridge, William M. and Crane, Harold L.: Use of a Large Jet Transport as an Inflight Dynamic Simulator. AGARD Report 528, 1966.

B17. DiFranco, D. A.: In-Flight Parameter Identification by the Equations-of-Motion Technique -- Application to the Variable Stability T-33 Airplane. TC-1921-F-3, Cornell Aeronautical Laboratory, Inc., 15 December 1965.

B18. Mabli, R. A., Carlson, J. W., Sickeler, R. O.: Results of Handling Qualities Research for the C-5A. AIAA 65-740, November 1965.

B19. Aircraft Control Task Allocation (Phase I - Feasibility). CDG Project 64-1A, USAF Instrument Pilot Instruction School, November 1965.

B20. Rhoads, D. W. and Eckhart, F. F.: In-Flight Simulation of Low Speed and Cruise Handling Qualities of a Supersonic Transport Airplane. TB-1992-F-1, Cornell Aeronautical Laboratory, Inc., October 1965.

B21. Curtiss, H. C., Jr. (Princeton University): An Analytical Study of the Dynamics of Aircraft in Unsteady Flight. USAAVLABS Technical Report 65-48, U. S. Army, October 1965.

B22. Carlson, J. W. and Sweeney, T. P. (Air Force): "Defining Stability and Control for the C-5A." Astronautics and Aeronautics, August 1965.

B23. Sadoff, Melvin and Wempe, Thomas E.: Pilot-Vehicle System Simulation for Low-Altitude, High-Speed Flight. NASA Technical Note D-2793, June 1965.

B24. Hacker, T.: Current Aspects of the Theory of Stability and Control of Aircraft - Part 1. Library Translation No. 1106, Royal Aircraft Establishment, May 1965.

B25. Hacker, T.: Some Recent Aspects of the Theory of Stability and Control of Aircraft - Part 2. Problems of Stability and Control Connected with Certain Present-Day Flight and Operating Characteristics. Library Translation No. 1107, Royal Aircraft Establishment, April 1965.

B26. Gertsen, W.M. and Shomber, H.A.: Critical Analysis of B-52 Stability Augmentation and Flight Control Systems for Improved Structural Life: Part V.A., B-52 Handling Qualities Design Goals. Boeing Report D3-6420, 1 April 1965.

B27. Thelander, J.A.: Aircraft Motion Analysis. FDL-TDR-64-70, March 1965.

B28. Anderson, Seth B., Quigley, Hervey C., Innis, Robert C.: Stability and Control Considerations for STOL Aircraft. AGARD Report 504, 1965.

B29. Anon: NASA Conference on Aircraft Operating Problems. NASA Special Publication 83, NASA Langley Research Center, 1965.

B30. Quinlivan, R.P. and Tye, G.: Analytical Investigation of Control Requirements for High Speed Low Altitude Penetration; Final Report Brief. AFFDL-TDR-64-104, December 1964.

B31. Kastner, T.M. (Cdr) and Soderquist, R.: Flight Evaluation of a Side Hand Controller in an F-4A Airplane. NATC Report FT2123-62R-64, U.S. Navy, 15 October 1964.

B32. Abrams, C.R. and Farr, T.P.: Controller Design Criteria and Flying Qualities Requirements for a Class of Advanced High-Performance Stability Augmented Aircraft. NADC-ED-6471, Navy, 7 October 1964, (also SAE A-18 Meeting No. 15).

B33. Brown, L.W.: An Investigation of the Dynamic Stability and Control Characteristics for a Transport Cruising at a Mach Number of 3. NASA TN-C-2483, October 1964.

B34. Abel, J.M.: Damping in Slightly Nonlinear Systems: Implications to Dynamic Stability Testing. AFFDL-TDR-64-110, October 1964.

B35. Anon: Supersonic Transport Aeroplane Flying Qualities. TSS Standard No. 5, Joint British-French Civil Specification, October 1964.

B36. Khoker, T.: Current Aspects of the Theory of Stability and Control of Aircraft. USAF FTD-TT-64-396, 18 September 1964.

B37. Rudov, Melvin H.: Pilot Aircraft Aiming Performance, AMRL Technical Report 64-87, AF, Aerospace Medical Division, September 1964.

B38. McLaughlin, Milton D. and Whitten, James B.: Pilot Evaluation of Dynamic Stability Characteristics of a Supersonic Transport in Cruising Flight Using a Fixed-Base Simulator. NASA TN D-2436, September 1964.

B39. Bisgood, P.L.: A Review of Recent Research on Handling Qualities, and Its Application to the Handling Problems of Large Aircraft. Part I - Observations on Handling Problems and Their Study. Part II - Lateral-Directional Handling. A.R.C.R. & M. No. 3458, June 1964.

B40. Quigley, Hervey C., Innis, Robert C., Holzhauser, Curt A.: A Flight Investigation of the Performance, Handling Qualities, and Operational Characteristics of a Deflected Slipstream STOL Transport Airplane Having Four Interconnected Propellers. NASA TN D-2231, March 1964.

B41. Barnes, A.G.: Stability and Control for Instrument Flight. AGARD Report 490, 1964.

B42. Anon: Stability and Control. FTC-TIH-64-2004, USAF TPS (Edwards AFB), Revised November 1964.

B43. Eckhart, F. and Harper, R. P., Jr.: Analysis of Longitudinal Responses of Unstable Aircraft. BA-1610-F-1, Cornell Aeronautical Laboratory, Inc., 30 September 1964.

B44. Kidd, E. A., Harper, R. P.: Fixed-Base and In-Flight Simulations of Longitudinal and Lateral-Directional Handling Qualities for Piloted Re-Entry Vehicles. ASD-TDR-61-362, February 1964.

B45. Anon: Control-Display Pilot Factors Program. Instr. Eval. Proj. Report 63-1, Air Force Instrument Pilot Instructor School, December 1963.

B46. Bird, D.: Flight Control Studies in the Small Stick Deflection Area. AIAA Simulation for Aerospace Flight Conference, August 1963.

B47. McRuer, D. and Graham, D.: Pilot-Vehicle Control System Analysis. AIAA 63-310, August 1963.

B48. White, Maurice D., Vomaske, Richard F., McNeill, Walter E., Cooper, George E.: A Preliminary Study of Handling-Qualities Requirements of Supersonic Transports in High-Speed Cruising Flight Using Piloted Simulators. NASA TN D-1888, May 1963.

B49. Tymczyszyn, Joseph J. and Spiess, Paul C.: The Effects of Supersonic Transport Flight Characteristics on Performance Requirements. SAE Paper 674D, April 8-11, 1963.

B50. A'Harrah, R. C. and Schulze, R. P.: An Investigation of Low-Altitude, High-Speed Flying and Riding Qualities of Aircraft. Report No. NA62H-397, North American Aviation, February 1963.

B51. Quigley, Hervey C. and Innis, Robert C.: Handling Qualities and Operational Problems of a Large Four-Propeller STOL Transport Airplane. NASA TN D-1647, January 1963.

B52. Sadoff, Melvin and Harper, Charles W.: A Critical Review of Piloted Flight Simulator Research, I.A.S. Paper 62-186, August 10-11, 1962.

B53. Barnes, A.G.: Review of Stability and Handling Criteria, With Particular Reference to Marginal Conditions. Aero, Servo, Mechanisms Technical Note ASV/158, English Electric Aviation, March 1962.

B54. Ashkenas, I.L. and McRuer, D.T.: "A Theory of Handling Qualities Derived From Pilot-Vehicle System Considerations." Aerospace Engineering, February 1962.

B55. White, M.D. and Sadoff, M.: Assessment of Critical Problem Areas of the Supersonic Transport by Means of Piloted Simulators. I.A.S. Paper 62-20, January 22-24, 1962.

B56. Schaufele, R.D.: Flying Qualities Design Considerations for Supersonic Transport Aircraft. 6th Annual Symposium, Society of Experimental Test Pilots, 1962.

B57. White, M.D., Bray, R.S., Cooper, G.E.: Some Design Problems of Supersonic Transports as Identified in Piloted-Simulator Studies. ICAS 3rd Congress, Chapter 49, 1962.

B58. Taylor, Lawrence W., Jr. and Day, Richard E.: Flight Controllability Limits and Related Human Transfer Functions as Determined From Simulator and Flight Tests. NASA TN D-746, May 1961.

B59. McFadden, Norman M., Vomaske, Richard F., Heinle, Donovan R.: Flight Investigation Using Variable-Stability Airplanes of Minimum Stability Requirements for High-Speed, High-Altitude Vehicles. NASA TN D-779, April 1961.

B60. Determination of Lateral Stability and Control Characteristics From Flight Tests in Non-Stationary Flight. TN V. 1883, NLR-AMSTERDAM, February 1961.

B61. McKee, John W.: A Three-Axis Fixed-Simulator Investigation of the Effects on Control Precision of Various Ways of Utilizing Rate Signals. NASA TN D-525, January 1961.

B62. Weil, Joseph: Application of Analytical Techniques to Flight Evaluations in Critical Control Areas. AGARD Report 369, 1961.

B63. Hafer, X.: Influence of Deflection of the Control Surfaces on the Free-Flight Behaviour of an Aeroplane. AGARD Report 355, 1961.

B64. Allwright, H. J.: Design Aims for Stability and Control of Piloted Aircraft. AGARD Report 337, 1961.

B65. Lee, G. H.: The Aeroplane Designer's Approach to Stability and Control. AGARD Report 334, 1961

B66. Ashkenas, I. and McRuer, D.: "Optimization of the Flight-Control, Airframe System." Journal of the Aerospace Sciences, Vol. 27, No. 3, P. 197, March 1960.

B67. Breuhaus, W., Reynolds, P., Kidd, E.: Handling Qualities Requirements for Hyper-Velocity Aircraft. (Title Unclassified). TC-1332-F-1, Cornell Aeronautical Laboratory, Inc., 30 September 1959. CONFIDENTIAL.

B68. Westbrook, C. and McRuer, D.: "Handling Qualities and Pilot Dynamics." Aerospace Engineering, May 1959.

B69. Butchart, Stanley P., Fischel, Jack, Tremant, Robert A., Robinson, Glenn H.: Flight Studies of Problems Pertinent to High-Speed Operation of Jet Transports. NASA Memo 3-2-59H, April 1959.

B70. Etkin, B.: Dynamics of Flight. John Wiley and Sons, Inc., 1959.

B71. Flack, N. D.: AFFTC Stability and Control Techniques. FTC TN59-21, USAF, 1959.

B72. Doyle, G. B., USMC (Revised by Petersen, Navy): Pilot Techniques for Stability and Control Testing. U. S. Navy Test Pilot School, 1958.

B73. Ashkenas, I. and McRuer, D.: Approximate Airframe Transfer Functions and Application to Single Sensor Control Systems. WADC TR 58-82, June 1958.

B74. Westbrook, Charles B. and McRuer, Duane T.: Aircraft Handling Qualities and Pilot Response Characteristics. AGARD Report 125, 1957.

B75. Abzug, Malcolm J.: High-Speed Stability and Control Problems as They Affect Flight Testing. AGARD Report 120, 1957.

B76. Cooper, G. E.: Understanding and Interpreting Pilot Opinion. IAS Preprint 683, January 1957.

B77. McNeill, W. and Creer, B.: A Summary of Results Obtained During Flight Simulation of Several Aircraft Prototypes with Variable-Stability Airplanes. NACA RM A56C08, 25 May 1956.

B78. Williams, Walter, C. and Phillips, William H.: Some Recent Research on the Handling Qualities of Airplanes. NACA RM H55L29, 21 February 1956.

B79. Perkins, C.D. (Editor) and Dommasch, D.O. (Associate Editor): AGARD Flight Test Manual, Volume II: Stability and Control. 1956.

B80. Mayer, John P. and Hamer, Harold A.: A Study of Service-Imposed Maneuvers of Four Jet Fighter Airplanes in Relation to Their Handling Qualities and Calculated Dynamic Characteristics. NACA RM L55E19, August 1955.

B81. Abzug, M. J. and Daros, C. J.: Equations of Motion for Large Disturbances from Steady Flight. Report No. ES17935, Douglas, 1 May 1955.

B82. Rathert, George, A., Jr., Abramovitz, Marvin and Gadeberg, Burnett L.: The Effects of Powered Controls and Fire-Control Systems on Tracking Accuracy. NACA RM A55D12a, May 1955.

B83. Abzug, M. J.: "Effects of Certain Steady Motions of Small-Disturbance Airplane Dynamics". Journal of Aeronautical Science, Volume 21, No. 11, November 1954.

B84. Douglas Aircraft Company: Comments on Proposed MIL Specification "Flying Qualities of Piloted Airplanes, Requirements for," Dated August 1953. Report No. ES 17466, Douglas, 26 October 1953.

B85. Rathert, George A., Jr., Gadeberg, Burnett L., Ziff, Howard L.: An Analysis of the Tracking Performances of Two Straight-Wing and Two Swept-Wing Fighter Airplanes with Fixed Sights in a Standardized Test Maneuver. NACA RM A53H12, October 22, 1953.

B86. Cheatham, Donald C., Mathews, Charles W., Harper, John A.: A Study of Visual Interception Attacks on a Nonmaneuvering Airplane Target. NACA RM L53E01, July 2, 1953.

B87. Williams, W. C. and Crossfield, A. S.: Handling Qualities of High-Speed Airplanes. NACA RM L52A08, January 1952.

B88. Anon: Flight Path Stability and Control in Relation to All-Weather Flying. Memo Report WCT-2372, WADC USAF, 29 October 1951.

B89. Phillips, W. H.: Appreciation and Prediction of Flying Qualities. NACA Report 927, 1949.

B90. Williams, Walter C., Forsyth, Charles M., Brown, Beverly P.: General Handling-Qualities Results Obtained During Acceptance Flight Tests of the Bell XS-1 Airplane. NACA RM No. L8A09, April 1948.

B91. Kayten, G.G., Koven, W.: Comparison of Wind-Tunnel and Flight Measurements of Stability and Control Characteristics of a Douglas A-26 Airplane. NACA ARR No. L5H11a, March 1946.

B92. Donlan, Charles J.: An Interim Report on the Stability and Control of Tailless Airplanes. NACA Report 796, 1944.

B93. Gilruth, R.R.: Requirements for Satisfactory Flying Qualities of Airplanes. NACA Report 755, 1943.

B94. Jones, R.T. and Cohen, D.: An Analysis of the Stability of an Airplane with Free Controls. NACA Report 709, 1941.

B95. Soulé, H.A.: Preliminary Investigation of the Flying Qualities of Airplanes. NACA Report 700, 1940.

B96. Barber, M.R., Jones, C.K., Sisk, T.R., Haise, F.W.: An Evaluation of the Handling Qualities of Seven General-Aviation Aircraft. NASA TND-3726, November 1966.

B97. Draper, C.S., McKay, W., Lees, S.: Instrument Engineering, Vol. II Methods for Associating Mathematical Solutions with Common Forms, 1953.

B98. Dolbin, B.H.: Study of Some Hand-Computing Techniques to Determine the Approximate Short Period Mode from Airplane Responses, FDM No. 371, Cornell Aeronautical Laboratory, Inc., March 1966.

B99. Neal, T.P.: "Frequency and Damping from Time Histories: Maximum Slope Method", Journal of Aircraft, Volume 4, number 1, Jan. - Feb., 1967.

B100. Behm, D.D.: Analog Matching Technique for Longitudinal Responses, Cornell Aeronautical Laboratory, Inc., FRM No. 409, October 1967.

B101. Hall, G.W.: A Method for Matching Flight Test Records with the Output of an Analog Computer, Cornell Aeronautical Laboratory, Inc., FDM No. 400, September 1966.

B102. Powers, Bruce G.: A Review of Transport Handling-Qualities Criteria in Terms of Preliminary XB-70 Flight Experience. NASA FRC TM-X-1584, May 1968.

B103. Feistel, T.W. and Innis, R.C.: Results of a Brief Flight Investigation of a COIN-Type STOL Aircraft. NASA (Ames) TN D-441, August 1967.

B104. Kotik, M.G. et al.: Flight Testing of Aircraft. NASA TT F-442, April 1967.

B105. O'Leary, C.O. and Perry, D.H.: A Piloted Simulator Study of the Study of the Take-Off Maneuver of a Large Aircraft With and Without a Take-Off Director. A.R.C. R. & M. No. 3524, August 1966.

B106. Mullins, Denver W.: Maneuver Load Data from F-104A. WADD TR 61-16, January 1961.

B107. McRuer, D., Ashkenas, I., Graham, D.: Aircraft Dynamics and Automatic Control, Naval Air Systems Command Technical Report No. 129-1, August 1968.

B108. General Dynamics Report FZM-12-2652, 9 December 1968.

B109. "Industry Observer", Aviation Week and Space Technology, p. 13, 1 April 1968.

B110. Wilkie, Lloyd E., et al.: The Determination of Aircraft Stability Coefficients from Flight Test Data - Part I - Theoretical Considerations. WADC TN 57-410, Part I, April 1958.

B111. Wolowicz, C.H., and Holleman, E.C.: Stability Derivative Determination from Flight Data, AGARD Report 224, October 1958.

B112. Yauncey, R.B., et al.: Aerodynamic-Derivative Characteristics of the X-15 Research Airplane as Determined from Flight Tests for Mach Numbers from 0.6 to 3.4. NASA TN D-1060, January 1962.

B113. Cooper, G.E., and Harper, R.P., Jr.: The Use of Pilot Rating in the Evaluation of Aircraft Handling Qualities, NASA TN D-5153, April 1969.

C. GENERAL LANDING APPROACH TOPICS

C1. Patterson, G.A., Jr.: Criteria for Determination of Minimum Usable Approach Speed. Paper 67-578, AIAA, August 1967.

C2. Rhoads, D.W.: In-Flight Simulation and Pilot Evaluation of Selected Landing Approach Handling Qualities of a Large Logistics Transport Airplane. AFFDL-TR-67-51, May 1967.

C3. Durand, T.S.: Carrier Landing Analysis. Systems Technology Inc., Technical Report 137-2, February 1967.

C4. Wilson, R.J.: Statistical Analysis of Landing Contact Conditions of the X-15 Airplane. NASA TN D-3801, January 1967.

C5. McLaughlin, R.B., and Williams, W.J.: Statistical Presentation of Landing Parameters for Models F-4B, A-6A, A-4C, E-1B, and S-2D Aircraft During Field Mirror Landing Practice at NALF, Frentress, Virginia. Naval Air Engineering Center, Aeronautical Structures Laboratory Report 1098, 13 December 1966.

C6. Walchli, R.M. and Wible, R.V.: Large Aircraft Validation of a Force Wheel Steering Technique With a Split Axis Configuration. AFFDL TR-66-129, November, 1966.

C7. Condit, P.M., Kimbrel, L.G., and Root, R.G.: In-Flight and Ground-Based Simulation of Handling Qualities of Very Large Airplanes in Landing Approach. NASA CR-635, October 1966.

C8. Bisgood, P.L.: Low-Speed Investigations Using Freely-Flying Models. AGARD Stability and Control Meeting, Cambridge, England, September 1966.

C9. Schade, R.O. and Vomaske, R.F.: Study of Low-Speed Flying Qualities of Very Large Airplanes by Means of an In-Flight Simulator. (Title Unclassified) Conference of Aircraft Aerodynamics, NASA SP-124, May 1966. CONFIDENTIAL

C10. Catanese, R.M.: Statistical Presentation of Landing Parameters for Models F-4B, RF-8A, F-8E, A-6A, T-2A, T-28C, TS-2A, and C-1A Aircraft Aboard the USS F.D. Roosevelt (CVA-42). Naval Air Engineering Center, Aeronautical Structures Laboratory Report 1089, 29 April 1966.

C11. Rhoads, D.W.: In-Flight Simulation and Pilot Evaluation of the Landing Approach Flying Qualities of a Supersonic Transport Airplane. Cornell Aeronautical Laboratory, Inc. Report No. TM-2091-F-1, January 1966.

C12. Newell, F.D., Parrag, M.L.E., and Bull, G.: Simulated Landing Approaches of an Unaugmented C-5A Configuration. AFFDL-TR-65-210, December 1965.

C13. Drinkwater, Fred J. III, and Rolls, L. Stewart: Flight Experience With the Ogee Wing at Low Speed. North Atlantic Treaty Organization, Advisory Group for Aerospace Research and Development Report 538, 1966.

C14. Klein, R.H., Archer, R.B., and Lew, D.W.: Supersonic Transport Handling Characteristics During Approach and Landing Flight Regimes. USAF (Sponsored by FAA) AFFDL-TR-65-227, December 1965.

C15. Durand, T., and Wasicko, R.: An Analysis of Carrier Landing. AIAA Paper 65-791, November 1965.

C16. Anon.: Carrier Landing Analysis. Systems Technology, Inc. Progress (6/1 - 8/31/65) Report No. 21, September 1965.

C17. Schade, R.O. and Crane, H. L.: Low-Speed Flight Investigation of a Jet Transport With a Powered-Lift Boundary-Layer Control System. NATO AGARD Report, June 1965.

C18. Eldridge, W.: A Simulator and Flight Evaluation of the Longitudinal and Lateral Control Requirements of the C-5A for the Landing Approach Task. Boeing Company D6-10725, 18 May 1965.

C19. Hall, A. W. and McGinley, D.J.: Flight Investigation of Steep Instrument Approach Capabilities of a T-33 Airplane Under Manual Control. NASA TN D-2775, April 1965.

C20. Bowe, J. T. and Scheindlinger, S.: Statistical Presentation of Landing Parameters for Models A-3B, A-4C/E, F-4B, and RF-8A/F-8C Aircraft Aboard the USS Midway (CVA-41) in the WESTPAC Area. Navy, Naval Air Engineering Center Report ASL-1074, March 1965.

C21. Behan, R.A., Smith, E.E. and Price, H.E.: Pilot Acceptance Factors Related to Information Requirements and Display Concepts for All-Weather Landing. NASA CR-189, March 1965.

C22. Perry, B.L.: The Rainbow Optical Landing Aid. Naval Research Laboratory, Report 6184, February 1, 1965.

C23. Hall, Albert W., and McGinley, Donald J., Jr.: Flight Investigation of Steep Instrument Approach Capabilities of a C-47 Airplane Under Manual Control. NASA TN D-2559, January 1965.

C24. McLean, W., and Gannett, G. R.: SST Handling Characteristics During Approach and Landing Flight Regimes. Boeing-North American Vol. II, QUART. T.R. No. 4, July 1964.

C25. Price, Harold E., Smith, Ewart E., and Gartner, Walter B., A Study of Pilot Acceptance Factors in the Development of All-Weather Landing Systems. NASA CR-34, April 1964.

C26. Stapleford, R. L. (Systems Technology, Inc.): Application of Sensitivity Analysis to SST Handling Qualities in Landing Approach. SAE A-18 Meeting No. 12, (Systems Technology Inc. Working Paper No. M-3), July 1963.

C27. Hoy, W. W.: A Method for Establishing Landing Design Criteria for Carrier-Based Airplanes; Phase 2; Final Report. Chance Vought Corporation Aero. & Missile Division Report No. 2-53400/3R460, June 1963.

C28. Barnes, A.G.: Simulator Assessments of Take-Off and Landing. AGARD Report 432, 1963.

C29. Farnum, D.D. and Montooth, J. W.: APN-114 Manual and Automatic Landing System: Terminal Control Theory, System Integration Concepts, Flight Test Results, and Future Developments. AGARD Report 425, 1963.

C30. Nelson, J.H. and Griffin, G.M.: United States Navy Pilot-Controlled Landing Procedure and Associated Equipment. AGARD Report 423, 1963.

C31. Pinsker, W.J.G.: Features of Large Transport Aircraft Affecting Control During Approach and Landing. AGARD Report 421, 1963.

C32. Hoy, W. W.: A Method for Establishing Landing Design Criteria for Carrier-Based Airplanes; Phase 1; Final Report. Chance Vought Corporation, Aero. & Missiles Division Report No. 2-53400/2R367, August 1962.

C33. Geoffrion, D.R., and Kibardin, V.M.: Statistical Presentation of Operational Landing Parameters for Transport Jet Airplanes. Flight Standards Serv. FAA, June 1962.

C34. Markowitz, S.A. and Negro, F.G.: Statistical Presentation of Fleet Operational Landing Parameters for Models F8U-1, F9F-8P, F3H-2N, A4D-1, and A3D-2 Aircraft aboard the USS Saratoga (CVA-60) in the Mediterranean Sea Area. U.S. Naval Air Material Center, Aeronautical Structures Laboratory Report - 1035, August 1961.

C35. Matranga, Gene J.: Analysis of X-15 Landing Approach and Flare Characteristics Determined From the First 30 Flights. NASA TN D-1057, July 1961.

C36. Gaines, L.M., and Surber, T.E.: Prediction of Optimum Approach and Landing Techniques for Manned Re-Entry Gliders. Paper No. 61-115-1809, Joint IAS/ARS Meeting, June 1961.

C37. Innis, Robert C., and Quigley, Hervey C.: A Flight Examination of Operating Problems of V/STOL Aircraft in STOL-Type Landing and Approach. NASA TN D-862, June 1961.

C38. Innis, Robert C.: Factors Limiting the Landing Approach Speed of Airplanes From the Viewpoint of a Pilot. AGARD Report 358, 1961.

C39. Spence, A. and Lean, D.: Some Low-Speed Problems of High-Speed Aircraft. AGARD Report 357, 1961.

C40. Eberle, R.B., Schoelerman, D.G., et al: Criteria for Predicting Landing Approach Speed Based on an Analog Computer Analysis of 21 Jet-Propelled Aircraft. Chance Vought Aircraft Report EOR-13202, October 25, 1960.

C41. Markowitz, S.A. and Negro, F.G.: Statistical Presentation of Landing Parameters for Models F8U-1, F3H-2M/2N, F4D-1, F11F-1, A4D-1 and AD-5N/5W Airplanes During Carrier Qualification Aboard the USS F.D Roosevelt (CVA-42) and the USS Hancock (CVA-19). U.S. Naval Air Material Center, Aeronautical Structures Laboratory Report 1033, 26 September 1960.

C42. Bray, Richard S., Drinkwater, Fred J. III, and White, Maurice D.: A Flight Study of a Power-Off Landing Technique Applicable to Re-Entry Vehicles. NASA TN D-323, July 1960.

C43. Adams, James J.: An Analog Study of an Airborne Automatic Landing-Approach System. NASA TN D-105, December 1959.

C44. Markowitz, S.A. and Negro, F.G.: Statistical Presentation of Landing Parameters on Models A3D-1 and AD-6/5 Airplanes During Carrier Qualification Aboard the USS Forrestal (CVA-59). NAMATCEN-ASL-1020, 19 November 1959.

C45. Riebe, J.M.: Consideration of Some Aerodynamic Characteristics During Take-Off and Landing of Jet Airplanes. NASA TN D-19, September 1959.

C46. White, M.D., and Innis, R.C.: A Flight Investigation of the Low-Speed Handling Qualities of a Tailless Delta-Wing Fighter Airplane. (Title Unclassified) NASA Memo 4-15-59A, May 1959, CONFIDENTIAL.

C47. Fischel, Jack, Butchart, Stanley P., Robinson, Glenn H., and Tremant, Robert A.: Flight Studies of Problems Pertinent to Low-Speed Operation of Jet Transports. NASA Memo 3-1-59H, April 1959.

C48. Lindquist, Dean G.: A Statistical Evaluation of Airplane Structural Landing Parameters in Mirror-Aid Landing Operations Aboard Aircraft Carriers. ASTIA AD-212713, Navy BuAer, Airframe Design Division Report No. AD-224-1, April 1959.

C49. Lane, J.C. and Cumming, R.W.: Pilot Opinions and Practices on the Approach to Landing: A Questionary Survey Among Australian Civil and Military Pilots. Aeronautical Research Laboratory, Human Engineering Report No. 1, April 1959.

C50. Drinkwater, Fred J. III, and Cooper, George E.: A Flight Evaluation of the Factors Which Influence the Selection of Landing Approach Speeds. NASA Memo 10-6-58A, December 1958.

C51. White, Maurice D., Schlaff, Bernard A. and Drinkwater, Fred J. III: A Comparison of Flight-Measured Carrier-Approach Speeds With Values Predicted by Several Different Criteria for 41 Fighter-Type Airplane Configurations. NACA RM A57L11, May 1958.

C52. Lina, Lindsay J., Morris, Garland J., and Champine, Robert A.: Flight Investigation of Factors Affecting the Choice of Minimum Approach Speed for Carrier-Type Landings of a Swept-Wing Jet Fighter Airplane. NACA RM L57F13, September, 1957.

C53. White, M.D. and Drinkwater, F.J. III: A Comparison of Carrier Approach Speeds as Determined From Flight Tests and From Pilot-Operated Simulator Studies. NACA RM A57D30, 19 June 1957.

C54. Calvert, E.S. and Sparke, J.W.: The Effect on Weather Minima of Approach Speed, Cockpit Cut-Off Angle and Type of Approach Coupler for a Given Landing Success Rate and Level of Safety. Royal Aircraft Establishment Technical Note El. 130, January 1957.

C55. Lane, J.C. and Cumming, R.W.: The Role of Visual Cues in Final Approach to Landing. Aeronautical Research Laboratory (Australia) Human Engineering Note 1, May 1956.

C56. Stillwell, Wendell H.: Results of Measurements Made During the Approach and Landing of Seven High-Speed Research Airplanes. NACA RM H54K24, February 4, 1955.

C57. Staff of Langley Research Center: Determination of Flight Characteristics of Supersonic Transports During the Landing Approach with a Large Jet Transport In-Flight Simulator. NASA TN D-3971, June 1967.

C58. Crane, H.L., Sommer, R.W. and Healy, F.M.: Effects of Reduced Airspeed for Landing Approach on Flying Qualities of a Large Jet Transport Equipped With Powered-Lift. NASA TN D-4804, October 1968.

D. LONGITUDINAL FLYING QUALITIES

D1. Stapleford, R. L. and Ashkenas, I. L.: Effects of Manual Altitude Control and Other Factors on Short-Period Handling Quality Requirements. AIAA Paper 67-579, August 1967.

D2. Eney, J. A.: Navy Variable-Stability Studies of Longitudinal Handling Qualities. AIAA Paper 67-576, August 1967.

D3. DiFranco, D. A.: Flight Investigation of Longitudinal Short-Period Frequency Requirements and PIO Tendencies. AFFDL-TR-66-163, April 1967.

D4. Tobak, Murray: On Nonlinear Longitudinal Dynamic Stability. AGARD Stability and Control Meeting, Cambridge, England, September 1966.

D5. Stapleford, R. L. and Ashkenas, I. L.: Analysis of Several Handling Qualities Topics Pertinent to Advanced Manned Aircraft. Pt.IV, Longitudinal Short-Period Handling Quality Requirements. Technical Report No. 156, Systems Technology, Inc., September 1966.

D6. Klung, H. A., Jr.: Frequency Response Method of Determining Aircraft Longitudinal Short-Period Stability and Control System Characteristics in Flight. FTC-TR-66-24, USAF, August 1966.

D7. Bihrle, W., Jr.: A Handling Qualities Theory for Precise Flight-Path Control. AFFDL-TR-65-198, USAF, June 1966.

D8. Saunders, T. B.: Handling Qualities of Aircraft with Marginal Longitudinal Stability. C.P. 837, British Aeronautical Research Council, 1966.

D9. Tomlinson, B. N. and Judd, M.: Some Calculations of the Take-Off Behaviour of a Slender-Wing Supersonic Transport Design Constrained to Follow a Specified Pitch-Attitude Time History. Technical Report 65174, Royal Aircraft Establishment, Britain, August 1965.

D10. Shomber, H. A. and Gertsen, W. M.: Longitudinal Handling Qualities Criteria: An Evaluation. AIAA Paper 65-780, November 1965.

D11. Malcom, L. G. and Tobie, H. N.: New Short Period Handling Quality Criterion for Fighter Aircraft. Boeing Report D6-17841 T/N, 15 September 1965.

D12. Higgins, H. C.: A Note on Longitudinal Control Response. Boeing Report D6-10732 T/N, 25 June 1965.

D13. Hong, J.: Resolution of Speed Stability for Supersonic Aircraft. AIAA Paper 65-622, June 1965.

D14. Tuomela, C. H.: Angle of Attack as an Aid to Flying. SAE Paper 650270, April 1965.

D15. Dolkas, C. B. and Stewart, J. D.: Effect of Combined Linear and Oscillatory Acceleration on Pilot Attitude-Control Capabilities. NASA TN D-2710, March 1965.

D16. Christopher, P. A. T.: The Stability of the Short-Period Motion of an Airframe Having Non-Linear Aerodynamic Characteristics in Pitch and Subject to a Step-Function Elevator Deflection. CoA Report Aero. 178, College of Aeronautics, Cranfield, England, January 1965.

D17. Tobak, Murray and Pearson, Walter E.: A Study of Nonlinear Longitudinal Dynamic Stability. NASA TR R-209, September 1964.

D18. A'Harrah, R. C.: "Low-Altitude, High-Speed Handling and Riding Qualities". Journal of Aircraft, Volume I, No. 1, January-February 1964, (Also AIAA Paper 63-299, North American Aviation 62H-397, AGARD 443).

D19. Chalk, C. R.: Fixed-Base Simulator Investigation of the Effects of L_α and True Speed on Pilot Opinion of Longitudinal Flying Qualities. ASD-TDR-63-399, November 1963.

D20. Ball, J. N. and Rynaski, E. G.: Longitudinal Flight Control for Military Aircraft - A Study of Requirements and Design Concepts. ID-1757-F-1, Cornell Aeronautical Laboratory, Inc., October 1963.

D21. Jex, Henry R. and Cromwell, Charles H., III: Theoretical and Experimental Investigation of Some New Longitudinal Handling Qualities Parameters. Technical Report 61-26, AF, Aeronautical Systems Division, June 1962.

D22. Bowman, D. R. and Mathews, R. H.: A Moving Base Flight Simulator Evaluation of Model F4H-1 Longitudinal Control Characteristics and Flying Qualities. Report 8397, McDonnell Aircraft Corporation, September 29, 1961.

D23. Kidd, E. A. and Bull, G.: Handling Qualities Requirements as Influenced by Pilot Evaluation Time and Sample Size. TB-1444-F-1, Cornell Aeronautical Laboratory, Inc., February 1962.

D24. Hall, Albert W. and Harris, Jack E.: A Simulator Study of the Effectiveness of a Pilot's Indicator Which Combined Angle of Attack and Rate of Change of Total Pressure as Applied to the Take-Off Rotation and Climbout of a Supersonic Transport. NASA TN D-948, September 1961.

D25. Faber, Stanley and Crane, Harold L.: A Longitudinal Control Feel System for In-Flight Research on Response Feel. NASA TN D-632, January 1961.

D26. Sadoff, Melvin, McFadden, Norman M., Heinle, Donovan R.: A Study of Longitudinal Control Problems at Low and Negative Damping and Stability with Emphasis on Effects of Motion Cues. NASA TN D-348, January 1961.

D27. Hurt, George J., Jr. and Whitten, James B.: Flight Investigation of an Automatic Pitchup Control. NASA TN D-114, August 1960.

D28. Brown, Beverly Porter, Johnson, Harold I., Mungall, Robert G.: Simulator Motion Effects on a Pilot's Ability to Perform a Precise Longitudinal Flying Task. NASA TN D-367, May 1960.

D29. Brissenden, Roy F., Alford, William L., Mallick, Donald L.: Flight Investigation of Pilot's Ability to Control an Airplane Having Positive and Negative Static Longitudinal Stability Coupled with Various Effective Lift-Curve Slopes. NASA TN D-211, February 1960.

D30. McRuer, D. T., Ashkenas, I. L., Guerre, C. L.: A Systems Analysis View of Longitudinal Flying Qualities. WADD TR 60-43, USAF, January 1960.

D31. Bull, G.: Minimum Flyable Longitudinal Handling Qualities of Airplanes. TB-1313-F-1, Cornell Aeronautical Laboratory, Inc., December 1959.

D32. Miele, A.: Minimal Maneuvers of High-Performance Aircraft in a Vertical Plane. NASA TN D-155, September 1959.

D33. Sadoff, Melvin: Pitch-Up Problem - A Criterion and Method of Evaluation. NASA Memo 3-7-59A, February 1959.

D34. Chalk, C. R.: Additional Flight Evaluations of Various Longitudinal Handling Qualities in a Variable-Stability Jet Fighter. WADC TR 57-719, July 1958.

D35. Faber, Stanley: Qualitative Simulator Study of Longitudinal Stick Forces and Displacements Desirable During Tracking. NACA TN 4202, February 1958.

D36. Sadoff, Melvin, Stewart, John D., Cooper, George E.: Analytical Study of the Comparative Pitch-Up Behavior of Several Airplanes and Correlation with Pilot Opinion. NACA RM A57D04, June 1957.

D37. Harper, R. P., Jr.: Flight Evaluations in Variable-Stability Airplanes of Elevator Control Motion Gradients for High-Speed Bombers, WADC TR 56-258, November 1956.

D38. Newell, F. D. and Rhoads, D. W.: Flight Evaluations of the Effect of Variable Phugoid Damping in a JTB-26B Airplane. WADC TR 56-223, October 1956.

D39. Matthews, Howard F. and Merrick, Robert B.: A Simulator Study of Some Longitudinal Stability and Control Problems of a Piloted Aircraft in Flights to Extreme Altitude and High Speed. NACA RM A56F07, September 1956.

D40. Faber, Stanley: Ground-Simulator Study of the Effects of Stick Force and Displacement on Tracking Performance. NACA TN 3428, April 1955.

D41. Harper, R. P., Jr.: Flight Evaluations of Various Longitudinal Handling Qualities in a Variable-Stability Jet Fighter. WADC TR 55-299, July 1955.

D42. Anderson, S. B. and Bray, R. S.: A Flight Evaluation of the Longitudinal Stability Characteristics Associated with the Pitch-Up of a Swept-Wing Airplane in Maneuvering Flight at Transonic Speeds. NACA Report 1237, 1955.

D43. Newell, F. D. and Campbell, G.: Flight Evaluations of Variable Short-Period and Phugoid Characteristics in a B-26. WADC-TR-54-594, December 1954.

D44. Newell, F. D. and Campbell, G.: Evaluations of Elevator Force Gradients and Types of Force Feel in a B-26. WADC TR 54-442, November 1954.

D45. Weil, Joseph and Gray, W. H.: Recent Design Studies Directed Toward Elimination of Pitch-Up. NACA RM L53I23c.

D46. Abramovitz, Marvin, Schmidt, Stanley F., VanDyke, Rudolph D., Jr.: Investigation of the Use of a Stick Force Proportional to Pitching Acceleration for Normal-Acceleration Warning. NACA RM A53E21, August 1953.

D47. Oswald, T. W.: "The Effect on Non-Linear Aerodynamic Characteristics on the Dynamic Response to a Sudden Change in Angle of Attack". Journal of the Aeronautical Sciences, Volume 19, No. 5, May 1952.

D48. Donlan, C. J.: Current Status of Longitudinal Stability. NACA RM L8A28, 24 May 1948.

D49. Greenberg, Harry and Sternfield, Leonard: A Theoretical Investigation of Longitudinal Stability of Airplanes with Free Controls Including Effect of Friction in Control System. NACA TR 791, 1944.

D50. Phillips, W. H.: A Flight Investigation of Short-Period Longitudinal Oscillations of an Airplane with Free Elevator. NACA Wartime Report L-444 (Original ARR), May 1942.

D51. Soulé, H. A.: Flight Measurements of the Dynamic Longitudinal Stability of Several Airplanes and a Correlation of the Measurements with Pilots' Observations of Handling Characteristics. NACA Report 578, 1936.

D52. Hall, G. W., In-Flight Investigation of Longitudinal Short-Period Handling Characteristics of Wheel Control Airplanes, AFFDL-TR-68-91, July 1968.

D53. Chase, T. W., Faulkner, V. L. and Helfinstine, R. F.: Study and Simulation Program to Investigate the Mechanization of an Aircraft Flight Control System that Employs Direct Lift Control. AFFDL-TR-68-69 (3 volumes), June 1968.

D54. Klein, D. G., Lt. USN: A Preliminary Investigation of the Use of Lift Control in Maneuvering Flight. Report 808, Princeton University, Dept. of Aerospace and Mechanical Science, June 1967.

D55. Bisgood, P. L.: A Review of Recent Handling Qualities Research, and Its Application to the Handling Problems of Large Aircraft, Part III - Longitudinal Handling. RAE Technical Report 68022, January 1968.

D56. McCormick, R. L., and Koepcke, W. W., Capabilities of In-Flight Thrust Reversing on Tactical Aircraft, AFFDL-TR-67-120, October 1967.

E. LONGITUDINAL FLYING QUALITIES - LANDING APPROACH

E1. A'Harrah, R. C. and Siewert, R. F.: A Study of Terminal Flight Path Control in Carrier Landings. NA66H-289, North American Aviation, February 1967.

E2. Brockhaus, R.: Mixed Pilot and Automatic Longitudinal Control During Landing Approach. AGARD Stability and Control Meeting, Cambridge, England, September 1966.

E3. Buhrman, J., Douwes Dekker, F. E., Kalkman, C. M.: A Few Notes on the Problem of Speed Stability. AGARD Stability and Control Meeting, Cambridge, England, September 1966.

E4. Eney, J. A.: Comparative Flight Evaluation of Longitudinal Handling Qualities in Carrier Approach. Report No. 777, Princeton University, May 1966.

E5. Chalk, C. R.: Flight Evaluation of Various Phugoid Dynamics and $1/T_{h_1}$ Values for the Landing Approach Task. AFFDL-TR-66-2, February 1966.

E6. Smith, L. R. and Prilliman, F. W.: Direct-Lift Control as a Landing-Approach Aid. AIAA Paper 66-14, January 1966.

E7. Miller, D. P. and Clark, J. W.: Investigation of the Use of Vectored Thrust During Carrier Landings. AIAA Paper 65-792, November 1965.

E8. Drake, D. E.: Direct Lift Control During Landing Approaches. AIAA Paper (Douglas), July 1965.

E9. Mazza, C. J., Spector, A., Kaniuka, W. W.: A Performance and Mechanical Design Analysis of Direct Lift Control for Improving the Carrier Landing Characteristics of Five Naval Airplanes. Report No. NADC-ED-6460, U. S. Naval Air Development Center, June 1965.

E10. Sullivan, E. R.: An Investigation of an Automatic Airspeed Control System for the KC-135 Approach and Landing Systems. GGC/EE/65-15, Air Force Institute of Technology, June 1965.

E11. Etheridge, J. D.: Direct Lift Control as a Landing Approach Aid in the F-8C Airplane - Simulator and Flight Tests. LTV Report 2-53310/4R-175, 31 December 1964.

E12. Chalk, C. R.: Flight Evaluation of Various Short-Period Dynamics at Four Drag Configurations for the Landing Approach Task. FDL-TDR-64-60, October 1964.

E13. Kehrer, W. T.: Longitudinal Stability and Control of Large Supersonic Aircraft at Low Speeds. ICAS Paper 64-586, Paris, France, August 1964.

E14. Powers, Bruce G. and Matheny, Neil W.: Flight Evaluation of Three Techniques of Demonstrating the Minimum Flying Speed of a Delta-Wing Airplane. NASA TN D-2337, July 1964.

E15. Urich, W. F. W.: The Landing Flare of a Slender-Winged Supersonic Transport Aircraft. Technical Note No. Aero 2961, Royal Aircraft Establishment, May 1964.

E16. Bray, Richard S.: A Piloted Simulator Study of Longitudinal Handling Qualities of Supersonic Transports in the Landing Maneuver. NASA TN D-2251, April 1964.

E17. Evans, R. E. and Schuppe, R. H.: An Investigation of the Effect of Simplified Equations of Motion and Various Inputs on Automatic Throttle Devices. AD 621548, U. S. Naval PostGraduate School, 1964.

E18. McNeill, Walter, E.: A Piloted Simulator Study of the Loss of Altitude by a Jet Transport in a Go-Around From an Instrument-Landing Approach. NASA TN D-2060, November 1963.

E19. Bray, R. S.: Piloted Simulator Studies Pertaining to the Low-Speed Longitudinal Handling Qualities of a Supersonic Transport Airplane. AIAA Simulation for Aerospace Flight Conference, August 1963.

E20. Ashkenas, I. and Durand, T.: Simulator and Analytical Studies of Fundamental Longitudinal Control Problems in Carrier Approach. AIAA Simulation for Aerospace Flight Conference, August 1963.

E21. Anon.: Simulator Study of Direct Lift Control During Carrier Landing Approaches. Douglas Report LB-31253, 8 April 1963.

E22. Staples, K. J.: Flight Measurements of the Influence of Speed Stability on the Landing Approach. AGARD Report 420, 1963.

E23. Abrams, C. R.: Comparison of Two Automatic Throttle Controllers for the F8U-2N Aircraft. NADC-ED-L6292, Navy, 30 November 1962.

E24. Cromwell, C. and Ashkenas, I.: A Systems Analysis of Longitudinal Piloted Control in Carrier Approach. TR 124-1, Systems Technology, Inc., June 1962.

E25. Bezanson, A. F.: Effects of Pilot Technique on Minimum Approach Speed. Report No. 1, Final Report, Navy Project TED No. PTR AD-3089, 17 February 1961.

E26. White, Maurice D. and Schlaff, Bernard A.: Airplane and Engine Responses to Abrupt Throttle Steps as Determined from Flight Tests of Eight Jet-Propelled Airplanes. NASA TN D-34, September 1959.

E27. Lina, Lindsay J., Champine, Robert A., Morris, Garland J.: Flight Investigation of an Automatic Throttle Control in Landing Approaches. NASA Memo 2-10-59L, March 1959.

E28. Newell, F.: Investigation of Phugoid Effects on GCA Landings. WADC TR 57-650, December 1957.

E29. Dickinson, R. P.: Some Possible Implications of Minimum Drag Speed at Altitude. AGARD Report 131, 1957.

E30. Lean, D. and Eaton, R.: The Influence of Drag Characteristics on the Choice of Landing Approach Speeds. AGARD Report 122, 1957.

E31. Bihrle, W.: A Study of Longitudinal Control Surface Pumping During Landing Flare-Outs. EAR 313, Republic Aviation, 15, December 1956.

E32. Neumark, S.: Problems of Longitudinal Stability Below Minimum Drag Speed, and Theory of Stability Under Constraint. (Title Unclassified). Royal Aircraft Establishment Report Aero. 2504, 1953. CONFIDENTIAL.

E33. Bihrle, W., Jr. and Stone, R. W., Jr.: Analytical Studies of the Response to Longitudinal Control of Three Airplane Configurations in Landing Approaches. (Title Unclassified). NACA RM L53B10, 9 April 1953. CONFIDENTIAL.

E34. Lovell, J. Calvin and Lipson, Stanley: An Analysis of the Effect of Lift-Drag Ratio and Stalling Speed on Landing-Flare Characteristics. NACA TN 1930, September 1949.

E35. Staples, K. J.: Speed Stability and the Landing Approach. (Appendix of Avro 707A Longitudinal Characteristics). RAE Technical Report 67322, December 1967.

E36. Bray, R. S., Snyder, C. T., and Drinkwater, F. J., III: A Preliminary Flight Test and Simulator Study of the Use of Wing Spoilers for Direct-Lift Control in the Approach and Landing of a Jet Transport Airplane. Working Paper No. 224, NASA (Ames), 1967.

F. LATERAL-DIRECTIONAL FLYING QUALITIES

F1. Meeker, J. I. and Hall, G. W.: In-Flight Evaluation of Lateral-Directional Handling Qualities for the Fighter Mission. AFFDL-TR-67-98, July 1967.

F2. Mesniere, M. (Nord-Aviation): Critere d'Instabilité dans L'Étude du Couplage Roulis-Tangage. AGARD Stability and Control Meeting, Cambridge, England, September 1966.

F3. Taylor, Lawrence W., Jr. and Iliff, Kenneth W.: Recent Research Directed Toward the Prediction of Lateral-Directional Handling Qualities. AGARD Report 531, 1966.

F4. Ashkenas, I. L.: Some Open- And Closed-Loop Aspects of Airplane Lateral-Directional Handling Qualities. AGARD Report, May 1966.

F5. Meeker, J. I.: Evaluation of Lateral-Directional Handling Qualities of Piloted Re-Entry Vehicles Utilizing Fixed-Base and In-Flight Evaluations. TC-1921-F-5, Cornell Aeronautical Laboratory, Inc., January 1966.

F6. Ross, A. J.: The Lateral Oscillation of Slender Aircraft. Current Paper 845, British Aeronautical Research Council, 1966.

F7. Smith, E. H. and DiFranco, D. A.: In-Flight Evaluation of Certain Lateral-Directional Handling Qualities of High-Performance Aircraft. AFFDL-TR 65-97, December 1965.

F8. Ashkenas, I. L.: A Study of Conventional Airplane Handling Qualities Requirements. Part I, Roll Handling Qualities. AFFDL-TR-65-138, November 1965.

F9. Ashkenas, I. L.: A Study of Conventional Airplane Handling Qualities Requirements. Part II, Lateral-Directional Oscillatory Handling Qualities. AFFDL-TR-65-138, November 1965.

F10. Masak, M.: Lateral Instability of Swept-Winged Aircraft Induced by Pitching Oscillations. AIAA Paper 65-315, July 1965.

F11. Ashkenas, I. L.: A Consolidation of Lateral-Directional Handling Qualities. AIAA Paper 65-314, July 1965.

F12. Newell, F. D.: Ground Simulator Evaluations of Coupled Roll-Spiral Mode Effects on Aircraft Handling Qualites. AFFDL-TR-65-39, March 1965.

F13. Masak, M.: On The Lateral Instabilities of Aircraft Due to Parametric Excitation. Technical Note No. 86, Institute for Aerospace Studies, University of Toronto, January 1965.

F14. Holladay, W. L.: Handling Qualities Considerations in Air Vehicle Designs. SAE A-18 Meeting No. 14, 8 July 1964.

F15. Ashkenas, I. L.: A Study of Lateral Oscillatory Handling Qualities Requirements. Working Paper 133-2, Systems Technology Inc., 18 June 1964.

F16. Sanctuary, G. E. and Roberto, A. M.: A Study of the Effect of Equivalent Airspeed on the Acceptability of Various Roll-to-Sideslip Ratios. Report FZM-12-435, General Dynamics, 27 January 1964.

F17. Quigley, Hervey C. and Lawson, Herbert F., Jr.: Simulator Study of the Lateral-Directional Handling Qualities of a Large Four-Propellered STOL Transport Airplane. NASA TN D-1773, May 1963.

F18. Pinsker, W.: The Lateral Motion of Aircraft, and in Particular, of Inertially Slender Configurations. R & M 3334, Aeronautical Research Council, Britain, 1963.

F19. Durand, T. S. and Jex, H. R.: Handling Qualities in Single-Loop Roll Tracking Tasks: Theory and Simulator Experiments. ASD-TDR-62-507, November 1962.

F20. Brown, L. W.: A Theoretical Investigation of the Effect of Cross-Control Derivatives on the Stability Characteristics of Airplanes Designed for Flight at High Mach Numbers. NASA TN D-1223, March 1962.

F21. Vomaske, Richard F., Sadoff, Melvin, Drinkwater, Fred J., III: The Effect of Lateral-Directional Control Coupling on Pilot Control of An Airplane as Determined in Flight and in a Fixed-Base Flight Simulator. NASA TN D-1141, November 1961.

F22. Harper, R. P.: In-Flight Simulation of the Lateral-Directional Handling Qualities of Entry Vehicles. WADD TR-61-147, November 1961.

F23. Aubin, W. M.: "Some Considerations of the Effects of Mission on Roll Requirements". Aerospace Engineering, Vol. 20, No. 9, Part I, September 1961.

F24. Phillips, William H.: Analysis of Effects of Interceptor Roll Performance and Maneuverability on Success of Collision-Course Attacks. NASA TN D-952, August 1961.

F25. Engel, G. R. and Rice, D. W.: Influence of Lateral-Directional Responses on Flying Qualities. Aeronautical Engineering Report 551, Princeton University, May 1961.

F26. Palmer, W. E.: A Flight Simulator Study of the Lateral-Directional Stability Requirements of Piloted Air Vehicles. Report 61H-241, North American Aviation, Inc., March 31, 1961.

F27. Kelley, H. J. and Aubin, W. M.: Mission Analysis to Determine Roll Performance Requirements. Research Report RE-125, Grumman, March 1960.

F28. Crone, R. M. and A'Harrah, R. C.: Development of Lateral-Directional Flying Qualities Criteria for Supersonic Vehicles, Based on a Stationary Flight Simulator Study. I. A. S. Paper No. 60-18, January 25-27, 1960.

F29. Ashkenas, I. L. and McRuer, D. T.: The Determination of Lateral Handling Quality Requirements From Airframe-Human Pilot System Studies. WADC TR 59-135, June 1959.

F30. Creer, Brent Y., Stewart, John D., Merrick, Robert B., Drinkwater, Fred J., III: A Pilot Opinion Study of Lateral Control Requirements for Fighter-Type Aircraft. NASA Memo 1-29-59A, March 1959.

F31. Matranga, Gene J.: Roll Utilization of An F-100A Airplane During Service Operational Flying. NASA Memo 12-1-58H, January 1959.

F32. Phillips, William H.: Analysis of Effects of Interceptor Roll Performance and Maneuverability on Success of Collision-Course Attacks. NACA RM L58E27, August 1958.

F33. Moul, Martin T. and Paulson, John W.: Dynamic Lateral Behavior of High-Performance Aircraft. NACA RM L58E16, August 1958.

F34. Kuehnel, Helmut A., Beckhardt, Arnold R., Champine, Robert A.: A Flight Investigation of the Effects of Varied Lateral Damping on the Effectiveness of a Fighter Airplane as a Gun Platform. NACA TN 4199, January 1958.

F35. Rhoads, D. W.: Flight Evaluation of the Effect of Variable Spiral Damping in a JTB-26B Airplane. TB-1094-F-1, Cornell Aeronautical Laboratory, Inc., October 1957.

F36. Woodling, C. H.: Theoretical Investigation of the Effect of Rudder and Stabilizer Deflections on the Angles of Attack and Sideslip in Rapid Rolls. NACA RM L57A30a, March 1957.

F37. Brissenden, R., Cheatham, D., Champine, R.: Tolerable Limits of Oscillatory Accelerations Due to Rolling Motions Experienced by One Pilot During Automatic-Interceptor Flight Tests. NACA RM L56K20, 25 January 1957.

F38. Adams, James J.: Flight and Analytical Study of Roll Requirements of a Fighter Airplane. NACA RM L56F12a, August 1956.

F39. Weil, Joseph and Day, Richard E.: Correlation of Flight and Analog Investigations of Roll Coupling. NACA RM H56F08, September 1956.

F40. Sisk, Thomas R. and Andrews, William H.: Flight Experience with a Delta-Wing Airplane Having Violent Lateral-Longitudinal Coupling in Aileron Rolls. NACA RM H55H03, October 1955.

F41. Kauffman, W. and Drinkwater, F.: "Variable-Stability Airplanes in Lateral Stability Research". Aeronautical Engineering Review. August 1955.

F42. Sternfield, L.: "A Vector Method Approach to the Analysis of the Dynamic Lateral Stability of Aircraft". Journal of Aeronautical Science, Volume 21, No. 4, April 1954.

F43. Gates, Ordway B., Jr. and Woodling, C. H.: A Method for Estimating Variations in the Roots of the Lateral-Stability Quartic Due to Changes in Mass and Aerodynamic Parameters of an Airplane. NACA TN 3134, January 1954.

F44. McNeill, Walter, E., Drinkwater, Fred J., III, Van Dyke, Rudolph D., Jr.: A Flight Study of the Effects on Tracking Performance of Changes in the Lateral-Oscillatory Characteristics of a Fighter Airplane. NACA RM A53H10, October 9, 1953.

F45. Pinsker, W. J. G.: Aileron Control of Small-Aspect-Ratio Aircraft; In Particular, Delta Aircraft. R & M 3188, Aeronautical Research Council, Britain, October 1953.

F46. Campbell, John P. and McKinney, Marion O., Jr.: A Preliminary Study of the Problem of Designing High-Speed Airplanes with Satisfactory Inherent Damping of the Dutch Roll Oscillation. NACA TN 3035, October 1953.

F47. Kuehnel, Helmut A., Beckhardt, Arnold R., Champine, Robert A.: A Flight Investigation of the Effects of Varied Lateral Damping on the Effectiveness of a Fighter Airplane as a Gun Platform. NACA RM L53F08a, August 1953.

F48. Schade, Robert O. and Hassell, James L., Jr.: The Effects of Dynamic Lateral Stability and Control of Large Artificial Variations in the Rotary Stability Derivatives. NACA TN 2781, October 1952.

F49. Bull, G.: A Flight Investigation of Acceptable Roll to Yaw Ratio of the Dutch Roll and Acceptable Spiral Divergence. TB-574-F-6, Cornell Aeronautical Laboratory, Inc., February 1952.

F50. Raikh, A.: Calculation of the Lateral-Dynamic Stability of Aircraft. NACA TM 1264, February 1952.

F51. Liddell, Charles J., Jr., Creer, Brent Y., Van Dyke, Rudolph D., Jr.: A Flight Study of Requirements for Satisfactory Lateral Oscillatory Characteristics of Fighter Aircraft. NACA RM A51E16, July 1951.

F52. Phillips, William H. and Cheatham, Donald C.: Ability of Pilots to Control Simulated Short-Period Yawing Oscillations. NACA RM L50D06, November 1950.

F53. Beckhardt, Arnold R., Harper, John A., Alford, William L.: A Preliminary Flight Investigation of the Effect of Snaking Oscillations on the Pilots' Opinions of the Flying Qualities of a Fighter Airplane. NACA RM L50E17a, September 1950.

F54. Winograd, Lee and Van Dyke, Rudolph D., Jr.: Flight Determination of the Effects of Rudder-Pedal-Force Characteristics on the Aiming Error in Azimuth of a Conventional Fighter Airplane. NACA RM A50D06, July 5, 1950.

F55. Graham, D. and James, C.: A Flight Investigation of Minimum Acceptable Lateral Dynamic Stability. Report No. TB-574-F-3, Cornell Aeronautical Laboratory, Inc., 30 April 1950.

F56. Liddell, Charles J., Jr., Van Dyke, Rudolph D., Jr., Heinle, Donovan R.: A Flight Determination of the Tolerable Range of Effective Dihedral on a Conventional Fighter Airplane. NACA TN 1936, August 1949.

F57. Sternfield, Leonard and Gates, Ordway B., Jr.: A Method of Calculating a Stability Boundary That Defines a Region of Satisfactory Period-Damping Relationship of the Oscillatory Mode of Motion. NACA TN No. 1859, April 1949.

F58. Kauffman, William M., Liddell, Charles J., Jr., Smith, Allan, Van Dyke, Rudolph D., Jr.: An Apparatus for Varying Effective Dihedral in Flight with Application to a Study of Tolerable Dihedral on a Conventional Fighter Airplane. NACA TR 948, 1949.

F59. Sternfield, Leonard and Gates, Ordway B., Jr.: A Simplified Method for the Determination and Analysis of the Neutral-Lateral-Oscillatory-Stability Boundary. NACA TR 943, 1949.

F60. Campbell, J. P. and Toll, T. A.: Factors Affecting Lateral Stability and Controllability. NACA RM L8A28a, 13 May 1948.

F61. Toll, Thomas A.: Summary of Lateral-Control Research. NACA TR 868, 1947.

F62. Drake, H. M.: Experimental Determination of Directional Stability and Rotary Damping in Yaw on Lateral Stability and Control Characteristics. NACA TN 1104, July 1946.

F63. McKinney, M. O., Jr.: Experimental Determination of the Effects of Dihedral, Vertical-Tail Area, and Lift Coefficient on Lateral Stability and Control Characteristics. NACA TN 1094, July 1946.

F64. McKinney, Marion O., Jr.: Experimental Determination of the Effect of Negative Dihedral on Lateral Stability and Control Characteristics at High Lift Coefficients. NACA ARR 477, January 1946.

F65. Bamber, M. J.: Effect of Changes in Aspect Ratio, Side Area, Flight-Path Angle, and Normal Acceleration on Lateral Stability. NACA ARR 121, December 1942.

F66. Fehlner, Leo F.: A Study of the Effect of Adverse Yawing Moment on Lateral Maneuverability at a High Lift Coefficient. NACA ARR 38, September 1942.

F67. Fehlner, Leo F.: A Study of the Effects of Vertical Tail Area and Dihedral on the Lateral Maneuverability of an Airplane. NACA ARR 587, October 1941.

F68. Bamber, M. J.: Effect of Some Present-Day Airplane Design Trends on Requirements for Lateral Stability. NACA TN 814, June 1941.

F69. Gilruth, R. R. and Turner, W. N.: Lateral Control Required for Satisfactory Flying Qualities Based on Flight Tests of Numerous Airplanes. NACA Report 715, 1941.

F70. Weick, F. E. and Jones, R. T.: Résumé and Analysis of NACA Lateral Control Research. NACA Report 605, 1937.

F71. Shortal, J. A.: Effect of Tip Shape and Dihedral on Lateral-Stability Characteristics. NACA Report 548, 1935.

F72. Quigley, H. C., Vomaske, R. F., Innis, R. C.: Lateral-Directional Augmentation Criteria for Jet Swept-Wing Transport Airplanes Operating at STOL Airspeeds. Conference on V/STOL and STOL Aircraft, NASA SP-116, April 4-5 1966.

F73. McNeill, W. E., Innis, R. C.: The Effect of Yaw Coupling in Turning Maneuvers of Large Transport Aircraft. Conference on Aircraft Operating Problems, NASA SP-83, May 10-12, 1965.

F74. Anderson, S. B., Quigley, H. C., Innis, R. C.: Some Performance and Handling-Qualities Consideration for Operation of STOL Aircraft. Conference on Aircraft Operating Problems, NASA SP-83, May 10-12, 1965.

F75. Crone, R. M., A'Harrah, R. C.: "A New Modified Acceptance Criterion for Lateral-Directional Flying Qualities." Aerospace Engineering, Vol. 19, No. 9, September 1960, pp. 24-29.

F76. Moore, N. B.: Artificial Stability Flight Tests of the XF-88A Airplane. WADC TR 52-298, July 1954.

F77. Eckhart, F. F., Hall, G. W., Martino, P. A.: Flight Evaluation of a Stability Augmentation System for Light Airplanes. Report No. IH-2154-F-1, Cornell Aeronautical Laboratory, Inc., November 1966.

F78. Chalk, C. R.: Some Observations Concerning Lateral-Directional Equations of Motion and Transfer Functions. Cornell Aeronautical Laboratory, Inc., FDM No. 397, August 1966.

F79. Taylor, L. W., Jr., Robinson, G. H., and Iliff, K. W.: Correlation of Pilot Ratings with Criteria Developed from Various Lateral-Directional Handling-Qualities Studies. Working Paper FWP-9, NASA FRC, 28 June 1968.

F80. Eney, J. A., "Navy Variable - Stability Studies of Longitudinal Handling Qualities", Journal of Aircraft, Volume 5, number 3, May-June 1968.

G. LATERAL-DIRECTIONAL FLYING QUALITIES - LANDING APPROACH

G1. Miller, G. E. and Franklin, J. A.: Lateral-Directional Flying Qualities for Power Approach. AIAA Paper 67-577, August 1967.

G2. Anon.: Wind Speed Information for MIL-F-8785. Report 5515, USAF Environmental Technical Applications Center, July 1966.

G3. Staples, K. J.: Some Flight Measurements of Crosswind Landings on a Small Delta Aircraft (AVR0707A). R&M No. 3476, Aeronautical Research Council, Britain, April 1965.

G4. O'Gara, J. F.: Aircraft Crosswind Performance. AGARD Report 492, 1964.

G5. Patterson, G. and Spangenberg, W.: The Provision of Adequate Lateral Control Power for Landing Approach Conditions. AGARD Report 419, January 1963.

G6. Perry, D. H., Port, W. G. A., Morrall, J. C.: A Flight Study of the Sidestep Manoeuvre During Landing. R & M No. 3347, Aeronautical Research Council, Britain, July 1961.

G7. McNeill, Walter, E. and Vomaske, Richard F.: A Flight Investigation to Determine the Lateral Oscillatory Damping Acceptable for an Airplane in the Landing Approach. NASA Memo 12-10-58A, February 1959.

G8. Faber, Stanley: Comparison of Effectiveness of Coordinated Turns and Level Sideslips for Correcting Lateral Displacement During Landing Approaches. NACA RM L9129, December 1949.

G9. Harmon, S. M.: Aerodynamic Factors Affecting the Ability of a Pilot to Return an Airplane to Level From a Banked Attitude by Use of the Rudder Alone and Without Change of Heading. NACA ARR 88, October 1941.

G10. Seckel, E., Miller, G. E., Nixon, W.B.: Lateral-Directional Flying Qualities for Power Approach. Princeton University Report No. 727, September 1966.

G11. Seckel, E., Franklin, J. A. and Miller, G. E.: Lateral-Directional Flying Qualities for Power Approach: Influence of Dutch Roll Frequency. Princeton University, Report No. 797, September 1967.

G12. Quigley, H.C., Vomaske, R.F., and Innis, R.C.: Lateral-Directional Augmentation Criteria for Jet Swept-Wing Transport Airplanes Operating at STOL Airspeeds. NASA SP-116, Conference on V/STOL and STOL Aircraft, April 1966.

H. PILOT-INDUCED OSCILLATIONS

H1. Keith, L. A. and Marrett, G. J.: F-4C Stability and Control Tests With a TAC Training Loading. AFFTC-TR-67-9, USAF, 1967.

H2. A'Harrah, R. C. and Siewert, R. F.: Pilot-Induced Instability. AGARD Stability and Control Meeting, Cambridge, England, September 1966.

H3. Williams, G. H. and Twinting, W. T.: F-4C Category II Stability and Control Test. AFFTC-TR-65-30, USAF, December 1965.

H4. Hirsch, D. and McCormick, R.: Experimental Investigation of Pilot Dynamics in a Pilot-Induced Oscillation Situation. AIAA Paper 65-793, November 1965.

H5. Hirsch, D.: Investigation and Elimination of PIO Tendencies in the Northrop T-38A. SAE Paper, New York, July 1964.

H6. Ashkenas, I., Jex, H., McRuer, D.: Pilot-Induced Oscillations: Their Cause and Analysis. NORAIR Report No. NOR-64-143, 20 June 1964.

H7. Finberg, Floyd, Lt. Col.: Report of the T-38 Flight Control System PIO Review Board. USAF ASD, 15 February 1963.

H8. Caporali, R. L., Lamers, J. P., Totten, J. R.: A Study of Pilot-Induced Lateral-Directional Instabilities. Aeronautical Engineering Report 604, Princeton University, May 1962.

H9. Taylor, Lawrence W., Jr.: Analysis of a Pilot-Airplane Lateral Instability Experienced with the X-15 Airplane. NASA TN D-1059, November 1961.

H10. Phillips, William H., Brown, B. Porter, Matthews, James T.: Review and Investigation of Unsatisfactory Control Characteristics Involving Instability of Pilot-Airplane Combination and Methods for Predicting these Difficulties from Ground Tests. NACA TN 4064, August 1957.

H11. Terrill, W. H., Wong, J. G., Springer, L. R.: Investigation of Pilot-Induced Longitudinal Oscillation in the Douglas Model A4D-2 Airplane. Report No. LB-25452, Douglas, 15 May 1957.

H12. Abzug, M. J. and Dietrick, H. B.: Interim Report on Elimination of Pilot-Induced Longitudinal Oscillations From the Douglas Model A4D-2 Airplane. Report No. ES26613, Douglas, 20 March 1957.

J. CONTROL SYSTEMS AND SAS EQUIPMENT

J1. Krachmalnick, F. M., Vetsch, G. J., Wendl, M. J.: Automatic Flight Control System for Automatic Terrain Following. AIAA Paper 67-569, August 1967.

J2. Parrag, Michael L.: Pilot Evaluations In a Ground Simulator of the Effects of Elevator Control System Dynamics in Fighter Aircraft. AFFDL-TR-67-19, July 1967

J3. Smith, R. M.: Analysis and Design of Space Vehicle Flight Control Systems - Volume X; Man in the Loop. CR-829, NASA, July 1967.

J4. Belcher, G.: The Design of Autostabilizers With Particular Reference to Low Speed Handling and Ground Attack Operations. AGARD Stability and Control Meeting, Cambridge, England, September 1966.

J5. Deque, R.: Problèmés Posés par L'Intégration du Système de Controle Automatique de Vol dans les Commandes de Vol d'un Avion a Hautes Performances. Exemple d'Application dans le Cadre du Projet de Transport Civil Supersonique "Concorde". AGARD Stability and Control Meeting, Cambridge, England, September 1966.

J6. Barron, R. L., McKechnie, R. M., III, Davies, J. M., Snyder, R. F.: Self-Organizing Control of Aircraft Pitch-Rate and Normal Acceleration. AFFDL-TR-66-41, USAF, August 1966.

J7. Bernadyn, J., Pfersch, G., McGough, J.: Off-Line Adaptive Flight Control System. AFFDL-TR-65-182, USAF, April 1966.

J8. Howell, G. C.: Flight Experience of Rate Demand Control Using Electric Signalling in the Avro 707C Aircraft. Report 536, North Atlantic Treaty Organization, 1966.

J9. Watts, M. R.: A Preliminary Theoretical Study of an Adaptive Manoeuvre Demand Aircraft Control System Using Sampled-Data Techniques. TR 65278, Royal Aircraft Extablishment, December 1965.

J10. Cromwell, C. and Koven, W.: Flying Qualities Requirements as Related to Control System Complexity. AIAA Paper 65-794, November 1965.

J11. Shipley, P. P., Engel, A. G., Jr., Hung, J. W.: Self-Adaptive Flight Control by Multivariable Parameter Identification. AFFDL-TR-65-90, USAF, August 1965.

J12. Kastner, T. M., Soderquist, R. H.: Final Report: Evaluation of A General Electric Self-Adaptive Flight Control System in the F-4A Airplane. FT2123-49R-64, Navy, NATC, 8 July 1964.

J13. McRuer, D., Ashkenas, I., Pass, H.: Analysis of Multi-Loop Vehicular Control Systems. ASD-TDR-62-1014, USAF, March 1964.

J14. Ashkenas, Irving L. and McRuer, Duane T.: Competing Flight Control Systems For Entry Glider Lateral Control. Technical Document Report 62-699, USAF, February 1964.

J15. Kislyakav, V. S.: Longitudinal Stability of an Airplane with Delayed-Action Autopilot. TT 64-335, USAF, 1964.

J16. Clark, D. C., Pritchard, F. E., Reynolds, P. A.: Application of Self-Adaptive Control Techniques to the Flexible Supersonic Transport. SAE, National Aeronautic and Space Engineering and Manufacturing Meeting - Los Angeles, September 1963.

J17. Anon.: Summary Report: TFX Handling Quality and Flight Control System Study. Working Paper 132-1, Systems Technology, Inc., 8 August 1963.

J18. Sadoff, M.: A Study of a Pilot's Ability to Control During Simulated Augmentation System Failures. NASA TND-1552, 1962.

J19. Crane, Harold L. and Sommer, Robert W.: Effects of Control-Feel Configuration on Airplane Longitudinal Control Response. NASA TN D-912, October 1961.

J20. Crane, Harold L.: Analog-Computer Investigation of Effects of Friction and Preload on the Dynamic Longitudinal Characteristics of A Pilot-Airplane Combination. NASA TN D-884, May 1961.

J21. McNeill, Walter E., McLean, John D., Hegarty, Daniel M., Heinle, Donovan R.: Design and Flight Tests of an Adaptive Control System Employing Normal-Acceleration Command. NASA TN D-858, April 1961.

J22. Gaynor, F. A.: Effects of Servo-Mechanism Characteristics on Aircraft Stability and Control. AGARD Report 361, 1961.

J23. Doetsch, K. H.: The Place of Servo-Mechanisms in the Design of Aircraft with Good Flight Characteristics. AGARD Report 360, 1961.

J24. Woodling, C. H. and Gates, Ordway B., Jr.: Theoretical Analysis of The Longitudinal Behavior of an Automatically Controlled Supersonic Interceptor During the Attack Phase Against Maneuvering and Nonmaneuvering Targets. NASA TN D-454, June 1960.

J25. McLean, John D. and Schmidt, Stanley F.: On the Design of a High-Gain Saturating Control System for Use as an Adaptive Autopilot. NASA TN D-305, February 1960.

J26. Short, M. D. and Patterson, G. A.: Evaluation of the Model F8U-1P Airplane Equipped with an Electronic Flight Control System, Report No. 1, Final Report. Project No. PTR AD-3091, NATC Navy, 4 December 1959.

J27. Russell, Walter R., Sjoberg, S. A., Alford, William L.: Flight Investigations of Automatic Stabilization of an Airplane Having Static Longitudinal Instability. NASA TN D-173, December 1959.

J28. Sjoberg, S. A.: Flying Qualities Associated with Several Types of Command Flight Control Systems. IAS Paper No. 59-134, October 5-7, 1959.

J29. Russell, Walter R. and Alford, William L.: Flight Investigation of a Centrally Located Rigid Force Control Stick Used with Electronic Control Systems in a Fighter Airplane. NASA TN D-102, September 1959.

J30. Moul, Martin T. and Brown, Lawrence W.: Effect of Artificial Pitch Damping on the Longitudinal and Rolling Stability of Aircraft with Negative Static Margins. NASA Memo 5-5-59L, June 1959.

J31. Potocki, W. J.: Broad Outline of Aircraft Feel - - Pilot's Appreciation. IAS Proprint No. 854, October 7-8, 195

J32. Sadoff, Melvin: The Effects of Longitudinal Control-System Dynamics on Pilot Opinion and Response Characteristics as Determined From Flight Tests and From Ground Simulator Studies. NASA Memo 10-1-58A, October 1958.

J33. Sjoberg, S.A., Russell, Walter R., Alford, William L.: Flight Investigation of a Normal-Acceleration Automatic Longitudinal Control System in a Fighter Airplane. (Title Unclassified) NASA Memo 10-26-58L, December 1958, CONFIDENTIAL.

J34. McFadden, Norman M., Pauli, Frank, A., Heinle, Donovan R.: A Flight Study of Longitudinal-Control-System Dynamic Characteristics By the Use of a Variable-Control-System Airplane. NACA RM A57L10, March 1958.

J35. Flügge-Lotz, I., Taylor, C. F., Lindberg, H. E.: Investigation of a Non-linear Control System. NACA Report 1391, 1958.

J36. Brown, B. Porter: Ground Simulator Studies of the Effects of Valve Friction, Stick Friction, Flexibility, and Backlash on Power Control System Quality. NACA TR 1348, 1958.

J37. Newell, F.: Effects of Breakout Force on Longitudinal Handling Qualities. WADC TR 51-155, USAF, December 1957.

J38. Harper, R. P.: Flight Evaluations in Variable-Stability Airplanes of Elevator Control Motion Gradients for High-Speed Bombers. TB-969-F-2, Cornell Aeronautical Laboratory, Inc., November 1956.

J39. Russell, Walter R., Sjoberg, S. A., Alford, William L.: A Flight Investigation of the Handling Characteristics of a Fighter Airplane Controlled Through A Rate Type of Automatic Control System. NACA RM L56F06, September 1956.

J40. Sjoberg, S. A., Russell, Walter R., Alford, William L.: A Flight Investigation of the Handling Characteristics of a Fighter Airplane Controlled Through an Attitude Type of Automatic Pilot. NACA RM L56A12, April 1956.

J41. Phillips, William H.: Analysis of an Automatic Control to Prevent Rolling Divergence. NACA RM L56A04, April 1956.

J42. Senders, J. W. and Bradley, J. V.: Effect of Backlash on Manual Control of Pitch of a Simulated Aircraft. WADC TR 56-107, USAF, March 1956.

J43. Abzug, M. J. and Thorndike, C. S.: Effects of Flight Control System Mechanical Characteristics on Airplane Controllability. ES26175, Douglas, 14 February 1956.

J44. Sjoberg, S. A.: A Flight Investigation of the Handling Characteristics of a Fighter Airplane Controlled Through Automatic-Pilot Control Systems. NACA RM L55F01b, September 1, 1955.

J45. Mathews, Charles W.: Analog Study of the Effects of Various Types of Control Feel on the Dynamic Characteristics of a Pilot-Airplane Combination. NACA RM L55F01a, August 1955.

J46. Investigation of Control "Feel" Effects on the Dynamics of a Piloted Aircraft System. Report GER-6726, Goodyear (BuAer Report AE-61-10), 25 April 1955.

J47. Curfman, H. J.: Theoretical and Analog Studies of the Effects of Non-Linear Stability Derivatives on the Longitudinal Motions of an Aircraft in Response to Step Control Deflections and to the Influence of Proportional Automatic Control. NACA Report 1241, 1955.

J48. Abramovitz, M. and Van Dyke, R.: The Effect of Stick-Force Gradient and Stick Gearing on the Tracking Accuracy of a Fighter Airplane. NACA RM A54I28, 17 December 1954.

J49. Stokes, Fred H. and Mathews, Charles W.: Theoretical Investigation of Longitudinal Response Characteristics of a Swept-Wing Fighter Airplane Having a Normal-Acceleration Control System and a Comparison with Other Types of Systems. NACA TN 3191, July 1954.

J50. Levine, Marvin: Tracking Performance as a Function of Exponential Delay Between Control and Display. WADC Technical Report 53-236, USAF, October 1953.

J51. Goldberg, J. H.: Effects of Spring and Inertia Devices on the Longitudinal Stability of Aircraft. WADC TR 53-350, USAF, July 1953.

J52. Adams, James J. and Whitten, James B.: Tests of a Centering Spring Used as an Artificial Feel Device on the Elevator of a Fighter Airplane. NACA RM L52G16, September 1952.

J53. Mathews, Charles W., Talmage, Donald B., Whitten, James B.: Effects on Longitudinal Stability and Control Characteristics of a Boeing B-29 Airplane of Variations in Stick-Force and Control-Rate Characteristics Obtained Through Use of a Booster in the Elevator-Control System. NACA Report 1076, 1952.

J54. Johnson, H. I.: Flight Investigation to Improve the Dynamic Longitudinal Stability and Control-Feel Characteristics of the P-63A-1 Airplane with Closely Balanced Experimental Elevators. Memo Report L6E20, NACA, July 1946.

J55. Phillips, W. H.: An Investigation of Additional Requirements for Satisfactory Elevator Control Characteristics. NACA TN 1060, June 1946.

J56. Jones, R. T. and Greenberg, H.: Effect of Hinge-Moment Parameters on Elevator Stick Forces in Rapid Maneuvers. NACA Report No. 798, November 1944.

J57. McAvoy, W. H.: Maximum Forces Applied by Pilots to Wheel-Type Controls. NACA TN 625, November 1937.

J58. Gough, M. N. and Beard, A. P.: Limitations of the Pilot in Applying Forces to Airplane Controls. NACA TN 550, January 1936.

J59. DiFranco, D. A.: In-Flight Investigation of the Effects of Higher-Order System Dynamics on Longitudinal Handling Qualities, AFFDL TR-68-90, July 1968.

J60. Newell, F.D. and Wasserman, R.: In-Flight Investigation of Pitch Acceleration and Normal Acceleration Bobweights. AFFDL TR 69-3, April 1969.

J61. Taylor, L.W., Jr. and Smith, J.W.: An Analysis of the Limit-Cycle and Structural Resonance Characteristics of the X-15 Stability Augmentation System. NASA TN D-4287, December 1967.

J62. Horsnell, J.R.: The SST Flight Control System Concept. AIAA Paper 67-570, August 1967.

J63. Miller, F.L. and Emfinger, J.E.: Fly-by-Wire Techniques. AFFDL-TR-67-53, July 1967.

J64. Harris, Pat: Some Aspects of the 727 Flight Control System. A-18 Meeting No. 14, Society of Automotive Engineers, July 1964.

J65. Bureau of Naval Weapons Failure Rate Data Handbook. Prepared by U.S. Naval Ordnance Laboratory, Corona, California (Updated periodically).

J66. Ad Hoc Committee Report on B-58 Controllability in Flight. Wright Air Development Division, Wright-Patterson AFB, Ohio, 2 April - 10 May 1960.

J67. Anderson, Ronald O.: A Second Analysis of B-58 Flight Control System Reliability. Flight Control Laboratory, Aeronautical Systems Division, Wright-Patterson AFB, Ohio, 6 November 1962.

J68. Gaul, J.W.: Application of Pilot-Controller Integration Techniques to a Representative V/STOL Aircraft. AFFDL-TR-65-200, October 1965.

J69. Cole, G., et al.; Study of Pilot - Controller Integration for Emergency Conditions. RTD TDR 63-4092, Air Force Systems Command, January 1964.

J70. Gottfried, P., et al.: Reliability Prediction Techniques for Flight Control Systems. AFFDL TR 67-20, April 1967.

J[illegible] Gottfried, P., et al.: Evaluation of Reliability Prediction Techniques for Entire Flight Control Systems. AFFDL TR 67-183, April 1968

J72. Lutskiy, V.A., Raschet nadezhnosti i effektivnosti radioelektronoy apparatury: spravochnik (calculating the reliability and efficiency of radio electronic apparatus: a handbook). Kiyev, Nankova Dumka; Foreign Technology Division Translation HT-23-493-68, 1966.

J73. Greenberg, H. and Sternfield, L.: A Theoretical Investigation of Longitudinal Stability of Airplanes with Free Controls Including Effect of Friction in Control System, NACA Report No. 791, 1944.

J74. Jones, R.T. and Cohen, D.: An Analysis of the Stability of an Airplane with Free Controls, NACA Report No. 709, 1941.

J75. Chalk, C.R., Flying Qualities and Fly-By-Wire, Paper presented at the Fly-By-Wire Flight Control System Conference, Wright-Patterson Air Force Base, Ohio, 16 and 17 November 1968.

K. SPINS AND STALLS

K1. Wykes, John H. and Casteel, Gilbert R.: Comparison of Computer and Flight Test Results for a Spinning Airplane. AGARD Stability and Control Meeting, Cambridge, England, September 1966.

K2. Wocke, Hans H.: Theoretical Considerations, Wind-Tunnel-Research and Flight Test Results on the Super-Stall Characteristics of the HFB 320 HANSA. AGARD Stability and Control Meeting, Cambridge, England, September 1966.

K3. Anglin, Ernie L. and Scher, Stanley H.: Analytical Study of Aircraft-Developed Spins and Determination of Moments Required for Satisfactory Spin Recovery. NASA TN D-2181, February 1964.

K4. Scher, S. H.: Post-Stall Gyrations and Their Study on a Digital Computer. AGARD Report 359, April 1961.

K5. Wimpenny, J. C.: Low-Speed Stalling Characteristics. AGARD Report 356, 1961.

K6. Neihouse, Anshal I., Klinar, Walter, J., Scher, Stanley H.: Status of Spin Research for Recent Airplane Designs. NASA TR R-57, 1960.

K7. Campbell, George S. and Weil, Joseph: The Interpretation of Non-linear Pitching Moments in Relation to the Pitch-Up Problem. NASA TN D-193, December 1959.

K8. Klinar, W. J. and Grantham, W. D.: Investigation of the Stability of Very Flat Spins and Analysis of the Effects of Applying Various Moments Utilizing 3-Moment Equations of Motion. NASA Memo 5-25-59L, June 1959.

K9. Klinar, Walter J.: Spinning and Related Problems at High Angles of Attack for High-Speed Airplanes. NACA RM L55L23a, March 19, 1956.

K10. Neihouse, Anshal I.: Effect of Current Design Trends on Airplane Spins and Recoveries. NACA RM L52A09, January 1952.

K11. Kamm, Robert W.: Spin-Tunnel Tests of Airplane Models with Extreme Variations in Mass Distribution Along the Three Body Axes. NACA ARR 402, March 1945.

K12. Neihouse, A. I.: A Mass-Distribution Criterion for Predicting the Effect of Control Manipulation on the Recovery From a Spin. NACA ARR 24, August 1942.

K13. Neihouse, A. I.: The Aileron as an Aid to Recovery from the Spin. NACA TN 776, September 1940.

K14. Seidman, Oscar and Donlan, Charles J.: An Approximate Spin Design Criterion for Monoplanes. NACA TN 711, June 1939.

K15. McAvoy, W. H.: Piloting Technique for Recovery from Spins. NACA TN 555, February 1936.

K16. Fuchs, R. and Schmidt, W.: The Dangerous Flat Spin and the Factors Affecting It. NACA TM 629, July 1931.

K17. Fuchs, R.: Mathematical Treatise on the Recovery From a Flat Spin. NACA TM 591, November 1930.

K18. Bihrle, W. Jr. and Heyman, A.C.: The Spin Behavior of Aircraft, Grumman Aircraft Engineering Corporation Report No. 394-68-1, December 1967.

L. AERODYNAMIC AND INERTIA COUPLING EFFECTS

L1. Cronvich, L. L. and Amsler, B. E.: Pitch-Yaw-Roll Coupling. AGARD Report 353, 1961.

L2. Pinsker, W. J. G.: Charts of Peak Amplitudes in Incidence and Sideslip in Rolling Manoeuvres Due to Inertia Cross Coupling. Report No. Aero 2604, Royal Aircraft Establishment, April, 1958.

L3. Bratt, R. W. and DaRos, C. J.: A Simplified Analysis of Inertia Coupling of a Rolling Airplane. Report No. ES26513, Douglas, 9 January 1957.

L4. Weil, Josepl and Day, Richard E.: An Analog Study of the Relative Importance of Various Factors Affecting Roll Coupling. NACA RM H56A06, April 1956.

L5. Weil, Joseph, Gates, Ordway B., Jr., Banner, Richard D., Kuhl, Albert E.: Flight Experience of Inertia Coupling in Rolling Maneuvers. NACA RM H55E17b, July 1955.

L6. Anon: Flight Experience with Two High-Speed Airplanes Having Violent Lateral-Longitudinal Coupling in Aileron Rolls. NACA RM H55A13, February 1955.

L7. Phillips, William H.: Effect of Steady Rolling on Longitudinal and Directional Stability. NACA TN 1627, June 1948.

M. STRUCTURAL MODE RESPONSE, TURBULENCE RESPONSE, AND RIDE QUALITIES

M1. Edinger, L. D.: Design of Elastic Mode Suppression Systems for Ride Quality Improvement and Application to an SST. AIAA Paper 67-571, August 1967.

M2. Wykes, J. H., Nardi, L. U., Knight, R. J.: Suggested Revisions to Military Specifications for Aircraft Utilizing a Gust Alleviation and Structural Dynamic Stability Augmentation System (Final Report). NA-67-189, North American Aviation, 15 June 1967.

M3. Hall, J. C.: The Effect of Low Altitude Gusts on Automatic Landing. AGARD Stability and Control Meeting, September 1966.

M4. Wasicko, Richard J.: NASA Research Experience on Jet Aircraft Control Problems in Severe Turbulence. AGARD Stability and Control Meeting, Cambridge, England, September 1966.

M5. Milne, R. D.: The Role of Flutter Derivatives in Aircraft Stability and Control. AGARD Stability and Control Meeting, Cambridge, England, September 1966.

M6. Evans, G. J. and Beele, B. J.: Auto-Aeroelastic Mode Coupling, A Comparison of Predicted and Actual Characteristics. AGARD Stability and Control Meeting, Cambridge, England, September 1966.

M7. Pritchard, F.E., Easterbrook, C.C., McVehil, G.E.: Spectral and Exceedance Probability Models of Atmospheric Turbulence for Use in Aircraft Design and Operation. AFFDL-TR-65-122, October 1965.

M8. Perry, D. H. and Burnham, J.: A Flight Simulation Study of Difficulties in Piloting Large Jet Transport Aircraft Through Severe Atmospheric Disturbances. Technical Report No. 65195, Royal Aircraft Establishment, September 1965.

M9. Bisgood, P. L. and Burnham, J.: A Review of the "Jet Upset" Problem. Technical Report 65192, Royal Aircraft Establishment, August 1965.

M10. Mori, A. S. and Wykes, J. H.: Control System Aspects of an Analytical Investigation into Controlling the Structural Modes of Flexible Aircraft. SAE Paper 650581, July 1965.

M11. Kadlec, P. W.: Flight Data Analysis of the Relationship Between Atmospheric Temperature Change and Clear Air Turbulence. AD 620 989, Eastern Air Lines, June 1965.

M12. Jones, J. G.: Simplified Spectral Analysis of the Motion of an Aircraft with a Saturating Yaw-Damper. Technical Report 65100, Royal Aircraft Establishment, May 1965.

M13. Hoblit, F. M. and Paul, N.: Development of a Power-Spectral Gust Design Procedure for Civil Aircraft. Report 18253, Lockheed-California Company, April 20, 1965.

M14. Lappe, U. O.: Design of a Low-Altitude Turbulence Model for Estimating Gust Loads on Aircraft. AFFDL-TR-64-170, USAF, March 1965.

M15. Burnham, J.: Measurements of the Rolling Response of a Fighter Aeroplane (Hunter MK. 6) to Turbulent Air and a Comparison with Theory. Technical Report 65033, Royal Aircraft Establishment, February 1965.

M16. Lappe, U. Oscar: A Low Altitude Turbulence Model for Estimating Gust Loads on Aircraft. AIAA Paper No. 65-14, January 1965.

M17. Dobrolenskii, Yu. P.: The Effect of an Autopilot on the Dynamics and Flight Safety of a Non-Rigid Aircraft in a Turbulent Atmosphere. Library Translation No. 1096, Royal Aircraft Establishment, January 1965.

M18. Lyster, H. N. C. and Pettit, K. G.: Low Altitude Turbulence Over Hilly Terrain. Counting Accelerometer Data From Jet Aircraft. NRC LR-417, National Research Council, Canada, December 1964.

M19. Chaney, R. E. and Parks D. L.: Tracking Performance During Whole-Body Vibration. D3-3512-6, Boeing. November 1964.

M20. Chaney, R. E. and Parks, D. L.: Visual-Motor Performance During Whole-Body Vibration. D3-3512-5, Boeing, November 1964.

M21. Jones, Jerry W.: High Intensity Gust Investigation. D-13273-333A, Boeing, October 24, 1964.

M22. Austin, William H., Jr.: Environmental Considerations in the Structural Design of a Low Level Strike Aircraft. AF, Systems Command, October 20-23, 1964.

M23. Beaupeurt, J. E.: Subjective Reaction to Whole-Body Vibration. D3-6474, Boeing, September 1964.

M24. Watts, Donald G.: Optimal Windows for Power Spectra Estimation. MRC Report No. 506, Mathematics Research Center, U. S. Army, September 1964.

M25. Piersol, A. G.: The Measurement and Interpretation of Ordinary Power Spectra for Vibration Problems. NASA CR-90 September 1964.

M26. Huntley, E.: The Longitudinal Response of a Flexible Slender Aircraft to Random Turbulence. Report No. Aero 2691, Royal Aircraft Establishment (also A.R.C. R & M 3454), August 1964.

M27. Sewell, R. T. and Pettit, K. G.: Analysis of V-G-H Records From A DC-4M Aircraft in The Vicinity of Frontal Systems and Troughs. NRC LR-402, National Research Council, Canada, July 1964.

M28. Fonda, A.G. and Frueh, F.J.: Flexible Flight Vehicle Transfer Function Factors for Use in Autopilot Preliminary Design. RTD-TDR-65-4218, April 1964.

M29. Clark, D., Notess, C., Pritchard, F., Reynolds, P., Schuler, J.: Application of Self-Adaptive Control Techniques to the Flexible Supersonic Transport. Summary Report. ASD-TDR 63-831, August 1963.

M30. Zbrozek, J. K.: Vertical Accelerations Due to Structural Vibrations of a Slender Aircraft Flying in Continuous Turbulence. T.N. Aero. 2901, Royal Aircraft Establishment, July 1963.

M31. Ridland, D. M.: The Measured Response of an Aircraft to the Vertical Velocity Component of Atmospheric Turbulence. T.N. Aero. 2866, Royal Aircraft Establishment, February 1963.

M32. Zbrozek, J. K.: Aircraft and Atmospheric Turbulence. T.N. Aero 2790, Royal Aircraft Establishment, October 1961.

M33. Beaupeurt, J. E. and Parks, D. L.: A Comparision of Sinusoidal and Random Vibration Effects on Human Performance. Report D3-3512, No. 2, Boeing, 28 July 1961.

M34. Beaupeurt, J. F., Parks, D. L., Snyder, F.W.: Human Reaction to Low Frequency Vibration. Report D3-3512, No. 1, Boeing, 24 July 1961.

M35. Johnson, J. M., Jr., Andrew, G. M., Olschausen, R.: Gust Alleviation Study for the B-70 Aircraft. Technical Memo 3441-12-50, North American Aviation, 11 May 1961.

M36. Zbrozek, J. K.: Theoretical Study of the Rolling Response of Aircraft to Turbulent Air. AGARD Report 373, April 1961.

M37. Etkin, B.: Theory of Flight of Airplanes in Isotropic Turbulence - Review and Extension. AGARD Report 372, 1961.

M38. Hancock, G. J.: Some Static Aeroelastic Considerations of Slender Aircraft. AGARD Report 352, 1961.

M39. Kalkman, C. M.: The Influence of Aeroelasticity on the Longitudinal Stability of a Swept Wing Subsonic Transport Aircraft. AGARD Report 351, 1961.

M40. Runyan, H. L., Pratt, K. G., Bennett, F. V.: Effects of Aeroelasticity on the Stability and Control Characteristics of Airplanes. AGARD Report 348, 1961.

M41. Anderson, K.: A Lateral-Directional Analog Study of a Flexible B-70 Airplane. NA 59-1820, North American Aviation, 30 November 1959.

M42. Etkin, B.: The Response of Slender Aeroplanes to Side-Gusts, and Means for its Suppression. Technical Note Aero. 2611, Royal Aircraft Establishment, April 1959.

M43. Simons, A. K. and Schmitz, M. A.: The Effect of Low Frequency, High Amplitude Whole Body Vibration on Human Performance. Progress Report No. 1, Bostrom Research Labs, January 1958.

M44. Tobak, Murray: On the Minimization of Airplane Responses to Random Gusts. NACA TN 3290, October 1957.

M45. Snyder, F. W.: Preliminary Study of Aircrew Tolerance to Low-Frequency Vertical Vibration. D3-1189, Boeing, 3 July 1957.

M46. Ribner, H. S.: "Spectral Theory of Buffeting and Gust Response: Unification and Extension". Journal of Aeronautical Science. Volume 23, No. 12, December 1956.

M47. Cole, Henry A., Jr., Brown, Stuart C., Holleman, Euclid C.: The Effects of Flexibility on the Longitudinal and Lateral-Directional Response of a Large Airplane. NACA RM A55D14, 1955.

M48. Payne, Chester B.: A Flight Investigation of Some Effects of Automatic Control on Gust Loac NACA Research Memo L53E14A, July 2, 1953.

M49. Jones, G. Melvill and Drazin, D. H.: Oscillatory Motion in Flight. Air Ministry Flying Personnel Research Committee, FPRC/1168, July 1961.

M50. Notess, C. B.: A Triangle -- Flexible Airplanes, Gusts, Crew. Cornell Aeronautical Laboratory FDM No. 343, May 1963.

M51. Wempe, Thomas E.: Effects of Gust-Induced and Maneuvering Acceleration Stress on Pilot-Vehicle Performance. NASA TM X-51748, May 1964.

M52. Davis, H. M. and Swaim, Dr. R. L.: Control of Flexible Airplane Response. Presented at NATO AGARD Meeting, Cambridge, England, September 1966.

M53. A'Harrah, R. C.: Maneuverability and Gust Response Problems Associated with Low-Altitude, High-Speed Flight. AGARD Report 556, October 1967.

M54. Panofsky, H. A., Press, H.: "Meteorological and Aeronautical Aspects of Atmospheric Turbulence" Progress in the Aeronautical Sciences, Volume 3. Edited by A. Ferris, D. Kuchemann and L.G.H. Sterne, Pergamon Press, 1962.

M55. Taylor, James: Manual on Aircraft Loads. Pergamon Press, 1965.

M56. Houbolt, John C., Steiner, Roy, Pratt, Kermit G.: Dynamic Response of Airplanes to Atmospheric Turbulence Including Flight Data on Input and Response. NASA TR R-199, June 1964.

M57. Lappe, U. Oscar, Davidson, Ben: "On the Range of Validity of Taylor's Hypothesis and the Kolmogoroff Spectral Law" Journal of the Atmospheric Sciences, Vol. 20, No. 6, November 1963.

M58. Press, H., Meadows, M. T., Hadlock, I.: A Reevaluation of Data on Atmospheric Turbulence and Airplanes Gust Loads for Application in Spectral Calculations. NACA Report 1272, 1956.

M59. Press, Harry: Atmospheric Turbulence Environment with Special Reference to Continuous Turbulence. AGARD Report 115, April - May 1957.

M60. Jones, J. G.: Gradient Properties of a Model of Stationary Random Turbulence. Royal Aircraft Establishment Technical Report 67134, June 1967.

M61. Pritchard, F. E.: A Discrete Model of Atmospheric Turbulence Based on the Continuous Random Model. CAL Flight Research Memorandum No. 411 (T-33 TM No. 42), November 1967.

M62. Austin, W. H., Jr.: A Summary of Some Recent Developments in the Description of Atmospheric Turbulence Used for Aircraft Structural Design. SEG-TR-66-45, August 1966.

M63. Firebaugh, John M.: "Evaluations of a Spectral Gust Model Using VGH and V-G Flight Data." Journal of Aircraft, Vol. 4, No. 6, January 1968.

M64. Coy, Richard G.: Atmospheric Turbulence Spectra from B-52 Flight Loads Data. AFFDL-TR-67-13, April 1967.

M65. Kordes, E. E. and Love, B. J.: Preliminary Evaluation of XB-70 Airplane Encounters with High-Altitude Turbulence. NASA TN D-4209.

M66. Coleman, T. L. and Steiner, R.: Atmospheric Turbulence Measurements Obtained from Airplane Operations at Altitudes Between 20,000 and 75,000 Feet for Several Areas in the Northern Hemisphere. NASA TN D-548, October 1960.

M67. Steiner, R.: A Review of NASA High-Altitude Clear Air Turbulence Sampling Programs. AIAA Paper No. 65-13, January 1965.

M68. Saunders, K. D.: B-66B Low Level Gust Study, Volume I - Technical Analysis. WADD Technical Report 60-305, Volume I, March 1961.

M69. Crooks, W. M., Hoblit, F. M., Prophet, D. T., et al.: Project HICAT An Investigation of High Altitude Clear Air Turbulence. AFFDL-TR-67-123, Volumes I, II, and III, November 1967.

M70. Gault, J. D.: Low Altitude Atmospheric Turbulence LO-LOCAT Mid-Term Technical Data Analysis. SEG-TR-67-35, August 1967.

M71. Burns, A., Rider, C.K.: PROJECT TOPCAT, Power Spectral Measurements of Clear Air Turbulence Associated with Jet Streams. Royal Aircraft Establishment Technical Report No. 65210, September 1965.

M72. Strom, J. A., Weathermon, T. G.: NB-66B High Altitude Gust Survey: Technical Analysis ASD-TDR-63-145, Volume I, June 1963.

M73. Rhyne, R. H. and Steiner, R.: Power Spectral Measurement of Atmospheric Turbulence in Severe Storms and Cumulus Clouds, NASA TN D-2469, October 1964.

M74. Lin, Y. K.: "Transfer Matrix Representation of Flexible Airplanes in Gust Response Study," Journal of Aircraft, Vol. 2, No. 2, March - April 1965.

M75. Ii, J. M., Sidwell, K. W.: Evaluation of Unsteady Aerodynamic Forces and Dynamic Response of Flexible Aircraft Structure to Continuous Turbulence in Supersonic Flight. Presented at AIAA/ASME 8th Structures, Structural Dynamics and Materials Conference, March 29-31, 1967.

M76. Bisplinghoff, R. L., Ashley, H., Halfman, R. L.: Aeroelasticity, Addison-Wesley Publishing Company, Inc., 1955.

M77. Bisplinghoff, R. L., Ashley, H.: Principles of Aeroelasticity, John Wiley and Sons, Inc., 1962.

M78. George, D., Smedfjed, J. B., Schriro, G. R.: An Engineering Evaluation of Airplane Gust Load Analysis Methods, Volume I - Theoretical Development, Grumman Report No. ADR 06-14-63.1, October 1963.

M79. Etkin, B.: "A Theory of the Response of Airplanes to Random Atmospheric Turbulence." Journal of the Aero/Space Sciences, July 1959.

M80. Enochson, L.D.: Frequency Response Functions and Coherence for Multiple Input Linear Systems. NASA Contractor Report CR-32, April 1964.

M81. Skelton, Grant B., Investigation of the Effects of Gusts on V/STOL Craft in Transition and Hover, AFFDL-TR-68-85, October 1968.

M82. Papoulis, Athanasios: Probability, Random Variables, and Stochastic Processes, McGraw-Hill Book Company, 1965.

M83. Burnham, J.: Atmospheric Gusts - A Review of the Results of Some Recent R.A.E. Research, Royal Aircraft Establishment TR 68244, October 1968.

M84. Reeves, P.M., A Non-Gaussian Turbulence Simulation, AFFDL TR 69-67, June 1969.

M85. Magdaleno, R., and Wolkovitch, J., Performance Criteria for Linear Constant-Coefficient Systems with Random Inputs, ASD TDR 62-470, January 1963.

N. SIMULATION TECHNIQUES, PILOT MODELS, AND HUMAN FACTORS

N1. Newell, F. D. : Human Transfer Characteristics in Flight and Ground Simulation for the Roll Tracking Task. TE-2256-F-1, Cornell Aeronautical Laboratory, Inc., June 1967.

N2. Stapleford, R., McRuer, D., and Magdaleno, R.: Pilot Describing Function Measurements in a Multiloop Task. IEEE Paper, 7th IEEE Symposium on Human Factors in Electronics, May 1966.

N3. Adams, J., Kincaid, J., and Bergeron, H.: Determination of Critical Tracking Tasks for a Human Pilot. NASA TN D-3242, February 1966.

N4. Graham, D. and Hofmann, L.: Investigations of Describing Function Technique. AFFDL-TR-65-137, Air Force Flight Dynamics Laboratory, February 1966.

N5. Wasicko, R. and Magdaleno, R.: Effects of Nonlinearities on Human Operator Tracking Performance: A Review of the Literature. AMRL-TR-65-158, Air Force Aerospace Medical Research Laboratories, October 1965.

N6. Hess, R. A. : An Investigation of the Human Operator as an Element in Both Time Variant and Equivalent Time-Invarient Systems. AFFDL-FDCC-65-42, September 1965.

N7. A'Harrah, R.C.: Flight Simulation, Past, Present and Future. AIAA Paper 65-480, July 1965.

N8. Hess R. A. : The Human Operator as an Element in a Control System with Time-Varying Dynamics. AFFDL-FDCC-TM-65-34, June 1965.

N9. Meiry, Jacob L. : The Vestibular System and Human Dynamic Space Orientation. N65-32033, Massachusetts Institute of Technology, D. Sci. Thesis (Man-Vehicle Control Laboratory), June 1965.

N10. McRuer, D. and Graham, D. : Human Pilot Dynamics in Compensatory Systems - Theory, Models, and Experiments With Controlled Element and Forcing Function Variations. AFFDL-TR-65-15, U. S. Air Force, January 1965.

N11. McRuer D. and Graham D. : Pilot Models for Single- and Multi-Loop Systems With Random Forcing Functions. Technical Report 134-1, Systems Technology, Inc., November 1964.

N12. Adams, J. and Bergeron, H. : Measurements of Human Transfer Function With Various Model Forms. NASA TN D-2394, August 1964.

N13. Muckler F. and Obermayer, R. : The Human Operator. International Science and Technology, July 1964.

N14. Obermayer, R. and Muckler, F. : Performance Measurement; In-Flight Simulation Studies. NASA CR-82, July 1964.

N15. Fry, D.E. and Watts M. R. : A Theoretical Assessment of Various Control Laws for Use in a Take-Off Director for Large Aircraft. Royal Aircraft Establishment (Britain) Technical Note I. E. E. 50, June 1964.

N16. Harper, C. W. : Simulation for Aerospace Research. North Atlantic Treaty Organization, Advisory Group for Aerospace Research and Development. AGARDograph 99, 1964.

N17. Obermayer, R. and Muckler, F.: Performance Measurement in Flight Simulation Studies. AIAA Simulation for Aerospace Flight Conference, August 1963.

N18. Krendel, E. and Bloom, J.: The Natural Pilot Model for Flight Proficiency Evaluation. NAVTRADEVCEN T.R. 323-1, Navy, 15 April 1963.

N19. Ryack, B. and Krendel, E.: Experimental Study of the Natural Pilot Flight Proficiency Evaluation Model. NAVTRADEVCEN T.R. 323-2, Navy, April 1963.

N20. Anon.: A Study of Selected Control-Display Problems for Manned Aerospace Vehicles. Interim Report, Report No. ASD-TDR-63-170. Aeronautical Systems Division, Dir/Aeromechanics, Flight Control Laboratory, Wright-Patterson AFB, Ohio. March 1963.

N21. van Oosterom, T.: A Flight Test Method for the Evaluation of Approach and Runway Lighting Effectiveness. AGARD Report 431, North Atlantic Treaty Organization, Advisory Group for Aeronautical Research and Development. 1963

N22. Newell, F.: Criteria for Acceptable Representation of Airplane Dynamic Responses in Simulators Used for Pilot Training. BM-1642-F-1, Cornell Aeronautical Laboratory, Inc. September 1962.

N23. Kuehnel, Helmut A.: Human Pilots' Dynamic-Response Characteristics Measured in Flight and on a Non-moving Simulator. NASA TN D-1229, March, 1962.

N24. Bekey, G.A.: An Investigation of Sampled Data Models of the Human Operator in a Control System. ASD-TDR-62-36, U. S. Air Force, February 1962.

N25. Francher, P.S. and Larrowe, V.L.: A Study of Engine Simulation Methods for Operational Flight Trainers. Final Report, Report No. 3591-19-F. Navy NAVTRADEVCEN 822-4. Institute of Science and Technology, University of Michigan, Ann Arbor. July 1961.

N26. Westbrook, C. B. : Simulation in Modern Aero-Space Vehicle Design. AGARD Report No. 366, April 1961.

N27. Rathert, G.A.Jr., Creer, B.Y., and Sadoff, M. : The Use of Piloted Flight Simulators in General Research. AGARD Report 365. North Atlantic Treaty Organization, Advisory Group for Aeronautical Research and Development. 1961.

N28. Kidd, E.A., Bull, G., and Harper, R.P. Jr. : In-Flight Simulation - Theory and Application. AGARD Report 366. North Atlantic Treaty Organization, Advisory Group for Aeronautical Research and Development. 1961.

N29. Jex, H., Cromwell, C., Siskind: Correlation of Experimental and Theoretical Limits for Pilot Control of Unstable Second Order Systems. Systems Technology Incorporated Technical Memo No. 56, 10 July 1960.

N30. Sheridan, T.: Time-Variable Dynamics of Human Operator Systems. AFCRC-TN-60-169 Air Force Cambridge Research Center. March 1960.

N31. Douvillier, Joseph G., Turner, Howard L., McLean, John D., and Heinle, Donovan R.: Effects of Flight Simulator Motion on Pilots' Performance of Tracking Tasks. NASA TN D-143. February 1960.

N32. Muckler, F. A. : Man-Machine Tracking Performance with Short-Period Oscillatory Control System Transients. WADD-TR-60-3, The Martin Company, Baltimore, Maryland. January 1960.

N33. McKee, John W. : Single-Degree-of-Freedom Simulator Investigation of Effects of Summing Display-Instrument Signals on Man-Machine Control. NASA TN D-148, December 1959.

N34. Silver, C.: Analysis of Human Capability in Control Tasks. Minneapolis-Honeywell Document U-ED-6140. November 1959.

N35. McRuer, Duane, T., Seckel, Edwards, Hall, Ian A. M., et al.: Human Pilot Dynamic Response in Flight and Simulator. WADC TR 57-520. August 1958.

N36. Eakin, G., and Campbell, G. : Human Operator Transfer Function Investigation. TE-665-F-5, Cornell Aeronautical Laboratory, Inc. 1 October 1957.

N37. Rockway, M. R. : Effects of Variations in Control Deadspace and Gain on Tracking Performance. WADC TR-57-326. U. S. Air Force (Aero Medical Laboratory). September 1957.

N38. Human Engineering - Application of Human Response Data to Aircraft Design. Control Specialists Inc., Memo Report No. 69. June 7, 1956.

N39. Cheatham, Donald C. : A Study of the Characteristics of Human-Pilot Control Response to Simulated Aircraft Lateral Motions. NACA Report 1197. 1954.

N40. **Warrick,** Melvin J: Effect of Transmission-Type Control Lags on Tracking Accuracy. Air Force Technical Report No. 5916. September 1949.

N41. Koegler, R. : A Study of Human Frequency Responses and Aircraft Stability and Control. Flight Research Memo No. 59, Cornell Aeronautical Laboratory, Inc. 15 December 1948.

N42. Jackson, C.T. and Snyder, C.T.: Validation of a Research Simulator for Investigating Jet Transport Handling Qualities and Airworthiness Criteria During Takeoff. NASA TN D-3565, October 1966.

P. TEST DATA ON SPECIFIC AIRPLANES - USAF FLIGHT TEST CENTER

P1. Rutan, E. L., Cuthill, F.J., and Twinting, W.T.: A-7A Limited Handling Qualities Evaluation. FTC-TR-68-1, March 1968.

P2. Keith, L.A., Richard, R.R., and Marrett, G.J.: Evaluation of Longitudinal Control Feel System Modifications Proposed for USAF F/RF-4 Aircraft. FTC-TR-67-19, December 1968.

P3. Van Norman, C.V., Bowline, J.D., Major: C-123K Limited Performance and Stability and Control Tests. FTC-TR-67-11 with Appendix, August 1967.

P4. Kurowski, J.A., Captain, and Cretney, F.D., Squadron Leader: KC-976 Performance and Limited Stability and Control Tests. FTC-TR-66-22, October 1967.

P5. McCabe, J.M., Captain, and Benefield, T.D., Major: C-141A Category II Performance and Stability Tests, FTC-TR-66-5, June 1966.

P6. Klung, H.A., Jr.. Captain, Fox, G.W., 1st Lt. , Adams, M.J., Major, and Knight, W.J., Major: F-5A Category II Supplementary Stability and Control Tests. FTC-TR-65-36, April 1966.

P7. Mack, J.B., Captain, Clem, G.O., Jones, G.E., Major and Cretney, F.D., Squadron Leader: B-26K Category II Performance and Qualitative Stability and Control Tests. FTC-TR-65-29, December 1965.

P8. Williams, G.H., Major and Twinting, W.T., Captain: F-4C Category II Stability and Control Tests. FTC-TR-65-30, December 1965.

P9. Hendrickson, C. L. and Smith, R. W.: NF-104A Aerospace Trainer Evaluation. FTC-TR-65-37, December 1965

P10. Martin, K. L., Henshaw, C. J., Captain and Cross, C. S., Major: Air Force Evaluation of the Jet Commander Model 1121. FTC-TR-65-28, September 1965.

P11. Broom, P. W., Henshaw, C. J., and Cross, C. S.: Air Force Evaluation of the Lear Jet Model 23 Aircraft. FTC-TR-65-21, August 1965.

P12. McKinzie, G. A. and Macleay, L.: F-5N Limited Flight Evaluation FTC-TR-65-22, August 1965.

P13. Abrams, R. and Bock, Jr., C. C.: B-58A Multiple Weapons Configuration Category II Performance and Stability Tests. FTC-TR-65-10, July 1965.

P14. Klung, H. A. and Parsons, R. K.: F-5B Category II Stability and Control Tests. FTC-TR-65-8, July 1965.

P15. Mitchell, J. F. and Parsons, R. K.: F-5A Category II Stability and Control Tests. FTC-TR-65-6, June 1965.

P16. Bailey, H. G. and Gordon, H. C.: F-5 A/B Systems Evaluation FTC-TR-65-3, June 1965.

P17. Reschak, R. L., Lt. USAF and Roger, R. L., Major USAF: YAT-28E Limited Category II Performance Stability and Control Tests. FTC-TR-64-39, January 1965.

P18. Kline, R. E., Capt. USAF and Twinting, W. T., Capt. USAF: F-105F Limited Category II Performance and Stability Tests. FTC-TR-64-33, January 1965.

P19. Yancey, M.H. and Cross, C.S., Major USAF: C-135B Limited Category II Stability and Control Tests. FTC-TR-64-6, July 1964.

P20. Martin, K.L. and Loewe, W.R., Capt. USAF: Limited C-130E Category II Stability and Control Tests. FTC-TR-64-2, June 1964.

P21. Adolph, C.E., Lt. USAF, et.al: B-52H Category II Performance and Stability Tests. FTC-TR-63-16, November 1963.

P22. Myers, R.H., Jr. Capt. USAF, and Cross, C.S., Major USAF C-140 Jet Star Flight Evaluation. FTC-TR-62-24, February 1963.

P23. Campbell, J.K., Major USAF: Pilot's Evaluation of the Dornier DO-28. FTC-TR-62-31, October 1962.

P24. Gobert, D. and Evans, N., Major USAF: F-105D Category II Performance and Stability Tests. FTC-TR-61-47, March 1962.

P25. Lusby, W. Capt. USAF and Hanks, N. Capt. USAF: T-38A Category II Stability and Control Tests. FTC-TDR-61-15, August 1961.

P26. C-54G Limited Flight Test Evaluation, April 1962.

P27. Johnson, R., Lt. USAF and Mosley, R., Capt. USAF: JC-131B Limited Flight Test Evaluation. FTC-TR-60-75, January 1961.

P28. McNamar L. and Rushworth, R., Capt. USAF: F-106A Category II Performance, Stability and Control Tests. FTC-TR-60-14, November 1960.

P29. Johns, S., Capt. USAF and Campbell, J. Capt. USAF: YAC-1DH Category II Performance and Stability Tests. FTC-TDR-60-41, November 1960.

P30. Miller, R., Major USAF: Noratlas FTC-TR-60-56, October 1960.

P31. Calderon, J. and Campbell, J., Capt. USAF: L-28A Air Force Flight Evaluation. FTC-TR-59-38, January 1960.

P32. Schweikhard, W. and Rushworth, R., Capt. USAF: F-104B Category II Performance and Stability Evaluation. FTC-TR-59-32, December 1959.

P33. Johnson, C., Lt. USAF and Cole, F., Capt. USAF: McDonnell Model 119A Air Force Flight Evaluation. FTC-TDR-59-25, September 1959.

P34. Schofield, B.L. and Bock, C., Capt. USAF: Category II Stability and Control Evaluation of the UB-58A S/N 55-665. FTC-TDR-59-18, July 1959.

P35. Leisy, D., Lt. USAF and Hunderwadel, H., Capt. USAF: F-100D Performance and Stability and Control. FTC-TDR-58-27, September 1958.

P36. Schweikhard, W. and Slayton, D. Capt. USAF: TF-102A Phase IV Performance and Stability. FTC-TDR-58-11, May 1958.

P37. Schweikhard, W. and Slayton, D. Capt. USAF: TF-102A Phase IV Performance and Stability. FTC-TDR-58-16, May 1958.

P38. Simmons, C.D. and Sorlie, D., Capt. USAF: F-101B Phase II Stability and Control Flight Evaluation. FTC-TDR-58-11, May 1958.

P39. Davidson, J. and Hodgson, W., Capt. USAF: L-27A Phase IV Performance. FTC-TDR-58-8, April 1958.

P40. Lusby, W., Lt. USAF and Cole, F., Capt. USAF: L-32D Phase IV Performance Tests. FTC-TDR-57-35, January 1958.

P41. Archer, D., Lt. USAF and Benefield, T., Capt. USAF: C-133A Phase IV Stability and Control Tests. FTC-TDR-57-34, January 1958.

P42. Simmons, C. and Ballentine, W., Maj. USAF: Phase IV Performance, Stability and Control Tests of C-130A. FTC-TDR-57-32, February 1958.

P43. Reed, J., Lt. USAF and Carlson, J., Capt. USAF: RB-66B Phase IV Limited Performance and Stability Tests. FTC-TDR-57-[illegible]9, September 1957. CONFIDENTIAL.

P44. Davidson, J. and Ferry, R., Capt. USAF.: TL-19D Phase IV Performance Tests, FTC-TDR-57-14, July 1957.

P45. Neyhart, C. Capt. USAF and Seigler, J. Maj. USAF: YC-134 Safety of Flight Evaluation. FTC-TDR-57-10, 1957.

P46. Seigler, J.P., Maj. USAF and Jackson, R.C.: C-123B Performance Tests. FTC-TDR-55-12, May 1955.

P47. Smith, D.R.: YDB-47E Phase II and IV Combined Tests. FTC-TR-55-17, May 1955. SECRET.

P48. Looney, J.L., Lt. USAF and Fulton, F.L., Major USAF: Phase IV Flight Tests of the Stability and Control Tests of the Glenn L. Martin B-57A Aircraft. FTC-TDR-54-20, September 1954. CONFIDENTIAL.

P49. Dienst, W.L., Capt. USAF and Julian, A.A., Major USAF: Phase II Tests of the North American F-86H Airplane, USAF S/N 52-1975. FTC-TDR-54-4, February 1954.

P50. Dienst, W. L., Capt. USAF and Julian, A. A., Major USAF: Phase II Flight Tests of the F-86H (Addendum I). FTC-TDR-54-4, May 1954.

P51. Anon.: Phase IV Flight Test of the T-28A Airplane No. 48-1371. FTC-TDR-51-16, January 1952.

P52. Keith, L. A., Rickard, R. R. and Marrett, G. J.: F-4C Category II Follow-on Stability and Control Tests. FTC-TR-67-26, May 1968.

APPENDIX I

(SUPERSEDED)

MIL-F-8785(ASG)
AMENDMENT-4
17 APRIL 1959

Superseding
BuAer SR-119B
1 June 1948
Air Force 1815-B
1 June 1948

MILITARY SPECIFICATION

(U) FLYING QUALITIES OF PILOTED AIRPLANES

This specification has been approved by the Department of the Air Force and by the Navy Bureau of Aeronautics.

1. SCOPE

1.1 Scope.- This specification contains the requirements for the flying qualities of U. S. military piloted airplanes.

1.2 Application.- The flying qualities for all airplanes proposed or contracted for shall be in accordance with the provisions of this specification, unless specific deviations are authorized by the procuring activity. Additional special requirements for stability and control may be specified by the procuring activity.

1.3 Classification.- For purposes of this specification, airplanes shall be divided into the following classes:

Class I - Primary trainer, observation, and other light airplanes specifically designated by the procuring activity.

Class II - Horizontal bomber, cargo, transport, glider, patrol, antisubmarine, early warning, minelayer, heavy attack, and trainers for class II airplanes.

Class III - Fighter, interceptor, general purpose attack, and trainers for class III airplanes.

An airplane not listed specifically among these class designations shall be considered to be in that class which includes airplanes of the most similar type. When peculiarities of intended mission or configuration so dictate, an airplane of one class may be required by the procuring activity to meet selected requirements ordinarily specified for airplanes of another class.

1.3.1 Land- or carrier-based designation.- The letter -L following a class designation identifies an airplane as land-based; carrier-based airplanes are similarly identified by the letter -C. When no such differentiation is made in a requirement, the requirement shall apply to both land-based and carrier-based airplanes.

2. APPLICABLE DOCUMENTS

2.1 Not applicable to this specification.

3. REQUIREMENTS

3.1 General

3.1.1 Airplane loadings.- Unless otherwise stated, the airplane weight for a specified cg (center of gravity) position shall be that corresponding to the normal service loading in which the specified cg is obtained. Similarly, normal service loading conditions shall govern the location of the cg for a specified weight. When not specified, loadings shall be optional.

3.1.2 Altitudes.- Unless otherwise stated, the requirements shall apply at all altitudes at which the airplane might be operated in each of the specified configurations. In general, compliance with this stipulation may be determined by investigation of three significant altitudes consistent with the airplane mission requirements. Unless otherwise established between the procuring activity and the contractor, these altitudes shall be defined as follows:

(a) Low altitude: For design purposes, low altitude shall be sea level.
(b) High altitude: An altitude not lower than 80 percent of the service ceiling.
(c) Medium altitude: Approximately 50 percent of high altitude, or 40,000 feet, whichever is lower. (Medium altitude need be investigated only when the service ceiling is 40,000 feet or higher.)

The high and medium altitude conditions may be excluded in consideration of configurations L, PA, WO, and TO.

3.1.3 Operational flight envelopes.- For the three altitudes defined in paragraph 3.1.2, and for combat ceiling, Mach-number - normal-acceleration envelopes for several significant airplane loading conditions shall be specified in the contract or otherwise established by agreement between the procuring activity and the contractor. Both positive and negative normal accelerations are to be included. These envelopes shall serve to define the boundaries within which the airplane is expected to be operational and within which the requirements of this specification therefore apply. Within these boundaries, there shall be no objectionable buffet, trim or stability changes, or other irregularities which might detract from the effectiveness of the airplane in executing its intended mission. The operational flight envelopes shall show cutoff points representing the highest Mach numbers at which the airplane is to be considered operational. These maximums shall be based on considerations of pullout recovery (reaching level flight at 2,000 feet above sea level), as well as attainable speeds. In the requirements of this specification, a curve of such cutoff speeds plotted against altitude is referred to as the maximum operational speed envelope. If necessary for adequate definition of this envelope, maximum speed points for various intermediate altitudes shall be included.

3.1.3.1 The operational flight envelopes for an airplane intended solely for missions at supersonic speeds need not include the transonic speed range, provided that satisfactory transition through the transonic speed range is assured.

3.1.4 Maximum permissible speed envelope.- A V_D (or M_D) altitude envelope shall be established in addition to the envelopes specified in paragraph 3.1.3. This maximum permissible speed envelope shall be derived from consideration of dives entered at V_H. Unless limited by structural considerations, this envelops shall define, at each altitude, the maximum speed from which a recovery can be made which will result in level flight at an altitude of not less than 2,000 feet above sea level without encountering intolerable buffet, loss of control, uncontrollable trim changes, or other dangerous airplane behavior during the entire dive or pullout. In establishing this maximum permissible speed, the pullout shall be governed by the requirements of paragraph 3.3.16.1.

3.1.4.1 The development of any dangerous flight conditions associated with the dive or pullout described in paragraph 3.1.4 shall be sufficiently gradual, in order that the pilot is amply warned.

3.1.5 External stores.- In preparation of the flight envelopes discussed in paragraph 3.1.3 and 3.1.4, external stores which are not normally droppable in flight, or which are intended to be carried during the primary mission, shall be considered as integral elements of the airplane configuration. When such stores contain expendable loads, the requirements shall, unless otherwise stated, apply throughout the range of store loadings. For other significant store installations, revisions to the flight envelopes and deviations from the flying qualities requirements shall be established by agreement between the procuring activity and the contractor in accordance with the mission requirements of the airplane with such stores installed. In establishing these agreements, consideration of reasonable single malfunctions, such as failure of release mechanism or failure of fuel feed, as well as normal initial asymmetric store installations, shall be included.

3.1.6 Effects of armament provisions.- Operation of bomb bay doors, armament pods or other movable protuberances, or firing of weapons, shall not cause objectionable buffet, trim changes, or other characteristics which impair the tactical effectiveness of the airplane under any flight condition in which operation of such devices may be required in the conduct of the airplane mission.

3.1.7 Release of stores.- The release of any stores intended to be released during normal operation of the airplane shall not result in dangerous or seriously objectionable flight conditions.

3.1.8 Deceleration devices.- Unless specifically exempted by the procuring activity, all class II and class III airplanes shall be capable of deceleration, dive-speed limitation, and constant-speed, glide-path control, to any degree desired by the pilot, within limits which shall be stated in the contract or otherwise agreed to by the procuring activity. These capabilities need not be provided by auxiliary devices, such as speed brakes, if other design features or provisions can be utilized to produce the desired characteristics. The term "deceleration device," as employed in this specification, shall apply to whatever brake, flap, or other feature is used to provide the desired incremental drag effect.

3.1.9 Configurations.- For purposes of this specification, the basic airplane configurations shall be as described herein. Items of configuration not specified, such as cockpit enclosure, cowl flaps, oil cooler flaps, gun turrets, blast tube covers, or bomb bay doors shall be in their normal settings for the particular configuration.

Configuration CR:	Cruise: Power for level flight at trim speed (See table II), flaps in cruise position, gear up.
Configuration D:	Dive: 25-Percent normal rated power or minimum operable power, whichever is the greater, flaps and gear up (unless normally used as speed brakes), speed brake extended.

Configuration O: Glide: Power off, unless otherwise specified; gear and flaps up.

Configuration L: Landing: Power off, gear down, flaps or other high lift device at landing setting.

Configuration P: Power on, clean: Normal rated power, flaps and gear up.

Configuration CO: Combat: Augmented power, airplane in combat configuration.

Configuration PA: Power approach: Gear down, flaps, other high lift device, canopy, and approach brake in normal approach position; power for level flight at $1.15V_{S_L}$ or normal approach speed, whichever is lower.

Configuration WO: Wave off: Gear down, flaps or other high lift device in landing position, takeoff power.

Configuration TO: Takeoff: Gear down, flaps or other high lift device at takeoff setting, takeoff power, including assist or augmentation used in normal takeoff.

3.1.10 Effects of asymmetry. - There shall be no dangerous or seriously objectional flight characteristics resulting from asymmetric flight conditions which may be encountered in normal operations (e.g., unequal flop or speed brake operation, manufacturing tolerances, tail-pipe eyelids, etc.)"

3.2 Mechanical characteristics of control systems.-

3.2.1 Control friction and breakout force.- Longitudinal, lateral, and directional controls shall exhibit positive centering in flight at any normal trim setting. Although absolute centering is not required, the degree of centering shall be such that the combined effects of centering, breakout force, stability, and force gradient do not produce objectionable flight characteristics, or permit large departures from trim conditions with controls free. Breakout forces, including friction, feel, preload, etc, shall be within the limits given in table I. These values refer to the pilot control force required to start movement of the control surface, and apply in flight at all attainable conditions of trimmed airspeed, altitude, temperature, and control deflection.

3.2.1.1 Measurement of the breakout forces on the ground will ordinarily suffice in lieu of actual flight measurement, provided that qualitative agreement between ground measurement and flight observation can be established to the satisfaction of the procuring activity.

TABLE I

Allowable breakout forces (including friction), pounds

Control		Classes I, II-C, and III		Class II-L	
		min	max	max	min
Elevator	Stick	1/2	3	1/2	5
	Wheel	1/2	4	1/2	7
Aileron	Stick	1/2	2	1/2	4
	Wheel	1/2	3	1/2	6
Rudder		1	7	1	14

3.2.1.2 For emergency manual operation upon failure of a power-operated or power-boosted control system, the allowable breakout forces specified in table I may be doubled.

3.2.2 Adjustable controls.- When a cockpit control is adjustable for pilot physical dimensions or comfort, the control force as defined in paragraph 6.2 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.

3.2.3 Rate of control displacement.- The ability of the airplane to perform the maneuvers expected of it shall not be limited by the rates of control surface deflection or auxiliary control operation, nor shall the rates of operation of either primary controls or auxiliary devices result in objectionable flight characteristics.

3.2.4 Cockpit control free play.- The free play in each cockpit control, i.e., the motion of the cockpit control, from the trim position, which does not move the control surface in flight, shall not be excessive.

3.2.5 Artificial stability devices.- Normal operation of an artificial device for improvement of any characteristic shall not introduce any objectionable flight or ground handling characteristics. Failure of such a device shall not result in a dangerous or intolerable flight condition. (See paragraphs 3.4.1.2, 3.5.7 and 6.6 for additional discussion.)

3.3 Longitudinal stability and control.-

3.3.1 Elevator-fixed static stability.- In the flight conditions and throughout the speed ranges listed in columns 1 and 2 of table II, the elevator-fixed neutral points shall be aft of the cg position in the aft critical loading.

3.3.1.1 At the aft critical loading, in the flight conditions and throughout the speed ranges listed in columns 1 and 2 of table II, the elevator-fixed static longitudinal stability with respect to angle of attack at constant speed shall be positive. This requirement shall also apply to configuration WO at 1.15 V_{S_L}.

3.3.2 Elevator-free static stability.- In the flight conditions and throughout the speed ranges listed in columns 1 and 2 of table II, the elevator-free neutral points shall be aft of the cg position in the aft critical loading. In general, this requirement shall be considered satisfied if the requirement of paragraph 3.3.2.1 is met. For configurations PA and P (climb), this requirement may be waived, provided paragraph 3.3.2.1 is met.

3.3.2.1 In the aft critical loading, with the airplane trimmed at the speeds listed in column 3 of table II, the variation of elevator control force with speed shall be a smooth curve, with a gradient which is stable through trim and remains stable throughout the specified speed range. (In configurations PA and P (climb), a reversal in slope may be permitted below the trim speed; if a reversal does occur, however, the force shall not decrease to less than 1 pound for classes I and III airplanes, or 3 pounds for class II airplanes.) This requirement applies throughout the speed ranges listed in column 2 of table II, but need be considered only at speeds within ±15 percent (or ±50 knots, whichever is less) of the trim speed, and need not be considered at speeds where the control force exceeds 50 lb. As used in this paragraph, the term gradient shall not include that portion of the force versus speed curve within the preloaded breakout force or friction range.

TABLE II

Required conditions for longitudinal static stability

Configuration	Speed range	"Trim speeds" 1/ for elevator-free stability
CR	1.4 V_{S_O} to V_{NRP}	Speed for maximum range, 2 additional trim speeds
P	0.75 V_{NRP} to V_H	V_{NRP}, 1 additional trim speed
P (climb)	0.85 $V_{R/C}$ or 1.15 V_{S_O}, whichever is greater, to 1.3 $V_{R/C}$	$V_{R/C}$
CO	V_{NRP} to V_M	V_H, 1 or more additional trim speeds
G	V_{S_O} to V_H	1.4 V_{S_O}, 1 or more additional trim speeds
D	All speeds normally attained in configuration D dives	1 or more representative configuration D dive speeds
L	V_{S_L} to limit structural speed in configuration L	1.4 V_{S_L}
PA	V_{S_L} to limit structural speed in configuration PA NOTE: For -C airplanes, lower speed shall be V_{S_L} or design maximum arresting speed, whichever is lower.	1.15 V_{S_L}

1/ Additional "trim speeds" shall be so selected that the trim speeds effectively span the specified speed range.

3.3.3 Exception in transonic flight.- The requirements of paragraphs 3.3.1 and 3.3.2 may be relaxed, if necessary, in the transonic-speed range, provided that any reversals in slope of elevator angle or elevator control force with speed are mild and gradual and not seriously objectionable to the pilot. However, on airplanes with cruising speeds or mission requirements necessitating prolonged operation at transonic speeds, the requirements of paragraph 3.3.2 shall be satisfied. For this purpose, the use of artificial means satisfactory to the procuring activity is permissible. The relaxation of paragraph 3.3.1 is not intended to include paragraph 3.3.1.1, which shall remain applicable throughout the entire speed range. (It is considered that a force reversal greater than 10 lb for class III or 15 lb for class II airplanes, or a gradient greater than 3 lb per incremental M of 0.01 for class III, or 5 lb per incremental M of 0.01 for class II airplanes, would be excessive.)

3.3.3.1 When the airplane is decelerated rapidly through the transonic speed range by actuation of the deceleration device and reduction of power or by maintaining an accelerated turn or pull-up, the magnitude and rate of the associated trim change shall be not so great as to cause difficulty in maintaining the desired normal acceleration by normal pilot techniques.

3.3.4 Stability in accelerated flight.- The slope of the curve of elevator deflection versus normal acceleration (g) at constant speed shall be stable (increasing up elevator required for increasing g) throughout the range of attainable load factors in all configurations and in all conditions of flight.

3.3.5 Short-period oscillations.- The dynamic oscillations of normal acceleration, which occur at approximately constant speed and which may be produced by abruptly deflecting and returning the elevator control to the trimmed position, shall damp to 1/10 amplitude in 1 cycle, and the magnitude of any residual oscillations shall not exceed ±0.05g at the pilot's location. Residual oscillations in angular attitude shall not be of objectionable magnitude and shall not adversely affect the tactical utility of the airplane. (For gunnery or bombing applications, pitch deviations greater than ± 5 mils are ordinarily considered excessive.) Any longitudinal oscillations with periods less than 6 seconds shall be governed by this requirement. For unarmed airplanes, or for primary damper inoperative conditions on airplanes which employ artificial damping, the degree of damping may be relaxed for altitudes above 30,000 feet. As a minimum, however, the oscillations at combat ceiling shall damp to 1/2 amplitude in 1 cycle.

3.3.5.1 When the elevator is abruptly deflected and released, the motion of the elevator following the release shall be essentially deadbeat, unless the elevator oscillations are of such frequency and amplitude that they do not result in an objectionable oscillation in normal acceleration.

3.3.5.2 There shall be no tendency for a sustained or uncontrollable oscillation resulting from efforts of the pilot to maintain steady flight.

3.3.5.3 The requirements of paragraphs 3.3.5, 3.3.5.1, and 3.3.5.2 shall apply at all permissible airspeeds and loadings, both in straight flight and in turns.

3.3.6 Long-period oscillations.- Although there is no specific requirement for damping of the conventional long-period, or phugoid oscillation which occurs at approximately constant angle of attack, there shall be no objectionable flight characteristics attributable to apparent poor phugoid damping. In addition, if the period of a longitudinal oscillation is less than 15 seconds, the oscillation shall be at least neutrally stable.

3.3.7 Control effectiveness in unaccelerated flight.- In erect unaccelerated flight at any altitude, the attainment of any permissible speed above the stalling speed V_S, as defined in paragraph 3.6.2, shall not be limited by the effectiveness of the longitudinal control, or controls. This requirement shall apply to all airplane configurations and permissible loading.

3.3.8 Control effectiveness in accelerated flight.- In the forward critical loading, when trimmed at any permissible speed and altitude in the configurations listed in table II, it shall be possible to develop at the trim speed, by the use of the elevator control alone, the limit load factor, the lift coefficient corresponding to V_S as defined in paragraph 3.6.2 or 3.6.2.2, or a load factor consistent with the operational flight envelope specified in paragraph 3.1.3.

3.3.9 Control forces in steady accelerated flight.- In steady turning flight and in pullouts, increases in pull force shall be required to produce increases in positive normal acceleration throughout the range of attainable accelerations. The variation of force with normal acceleration at all points beyond the breakout force shall be approximately linear, except that an increase in slope upward (such as might be introduced by an acceleration restrictor) is permissible above $0.85n_L$. In general, a departure from linearity resulting in a local gradient which differs from the average gradient by more than 50 percent is considered excessive. The average force gradient shall be within the limits specified in table III in configurations P, CO, D, and PA throughout the operational flight envelope up to $0.85n_L$.

3.3.9.1 In all configurations at all permissible speeds and accelerations, the local value of the force gradient shall never be less than 3 pounds per g.

TABLE III

Elevator control force gradient limits, lb per g

Class	Maximum	Minimum
I, III	$\frac{56}{n_L - 1}$	$\frac{21}{n_L - 1}$
II	$\frac{120}{n_L - 1}$	$\frac{45}{n_L - 1}$

3.3.9.2 For configurations P, CO, and D in airplanes intended primarily for high altitude missions, the maximum allowable force gradients specified in table III need not apply below the medium altitude. The maximum force gradients at the low altitude, however, shall be not more than 50 percent greater than the maximum values specified in table III.

3.3.9.3 Under conditions in which maximum attainable normal acceleration is less than n_L (e.g., limited by stall or control effectiveness), an increase in the maximum force gradient, up to a value no higher than 50 percent greater than that specified in table III, may be permitted.

3.3.9.4 For configurations P, CO, and D on class III airplanes with cg positions in combat loadings which are aft of the cg positions in other normal service loadings, the maximum allowable force gradients specified in table III shall not apply at cg positions forward of the most forward combat position. The maximum forces gradients in any normal service loading, however, shall be not more than 50 percent greater than the values specified in table III.

3.3.9.5 The requirements of paragraph 3.3.9 apply to negative as well as positive accelerations, except that the maximum force gradients specified in table III may be exceeded in the negative acceleration range. This increase, however, shall not exceed 50 percent of the value specified in table III.

3.3.10 Control forces in sudden pull-ups.- In sudden pull-ups from trimmed straight flight, in which the elevator cockpit control is rapidly deflected and returned to its initial position, the ratio of the maximum elevator control force to maximum (peak) change in normal acceleration shall never be less than the ratio of force to acceleration change obtained in steady accelerations under the same conditions. In investigating the sudden pull-up, several rates of cockpit control motion shall be considered, the elapsed time from start to return varying, for example, from 1/2 second to 6 seconds.

3.3.11 Control effectiveness in takeoff.- Elevator effectiveness shall not unduly restrict the takeoff performance of the airplane. As a minimum, elevator effectiveness shall be adequate to permit compliance with takeoff performance guarantees; if the takeoff performance is not specifically guaranteed in the airplane contract, it shall be possible, on a hard-surface runway at a minimum speed no greater than $V_{S_{TO}}$, to obtain takeoff attitude on nose-wheel airplanes or to maintain any attitude up to thrust line level on tail-wheel airplanes. (For propeller-powered airplanes, $V_{S_{TO}}$ may be estimated, with the concurrence of the procuring activity, on the basis of stall speeds determined with various amounts of power up to the highest feasible.) These requirements shall be met with the airplane loading which produces the most critical nose-heavy moment on nose-wheel types and the most critical tail-heavy moment on tail-wheel types. The loadings considered for this purpose shall include all full and partial loads which might normally be employed during training, as well as operational takeoffs. For class I tail-wheel airplanes, the required minimum speed for maintaining attitudes up to thrust line level shall be 0.5 $V_{S_{TO}}$ and shall be applicable on sod as well as hard-surface runways.

3.3.12 Control in catapult takeoff.- On airplanes designed for catapult takeoff, longitudinal control shall be sufficient to prevent pitch up or pitch down to undesirable attitudes in catapult launchings at speeds ranging from the minimum safe launching speed to a speed 25 knots higher than the minimum. Satisfactory catapult takeoff shall not be dependent upon predetermined control programming or unusual control manipulation by the pilot.

3.3.13 Control forces in takeoff.- With trim optional but constant, the elevator control forces required throughout the takeoffs described in paragraphs 3.3.11 and 3.3.12, and during the ensuing acceleration to a speed of 1.3 $V_{S_{TO}}$ (flaps, gear, and power held constant) shall be within the following limits:

Nose-wheel and bicycle-gear airplanes

Classes I, III-C	- 20-lb pull to 10-lb push.
Classes III-L, II-C	- 30-lb pull to 10-lb push.
Class II-L	- 50-lb pull to 20-lb push.

Tail-wheel airplanes

Classes I, II-C, and III	- 20-lb push to 10-lb pull.
Class II-L	- 35-lb push to 15-lb pull.

These requirements shall apply also in rocket-assisted or other power-augmented takeoffs, and shall include consideration of assist cessation.

3.3.14 Control effectiveness in landing.- At the forward critical loading, with the airplane trimmed for 1.2 V_{S_L} in configuration PA, longitudinal control shall be sufficiently effective, in order that in configuration L, V_{S_L} or the guaranteed landing speed, if such a guaranty is included in the contract, can be obtained in close proximity to the ground.

3.3.15 Control force in landing.- It shall be possible to meet the requirement of paragraph 3.3.14 with an elevator pull force not exceeding 35 lb for classes I, II-C, and III airplanes, or 50 lb for class II-L airplanes.

3.3.16 Control forces in dives.- With the airplane trimmed for level flight at V_H, the elevator control forces required in dives to any attainable speed within the operational flight envelope shall not exceed 50-lb push or 10 lb pull in class III airplanes, or 75 lb push or 15 lb pull in class II airplanes. In similar dives, but with trim optional following the dive entry, it shall be possible with normal piloting technique to maintain the forces within the limits of 10-lb push or pull in class III airplanes, or 20-lb push or pull in class II airplanes. The forces required for recovery from these dives shall be in accordance with paragraph 3.3.9.

3.3.16.1 With the airplane trimmed initially in level flight at V_H, but with trim optional in the dive, it shall be possible to maintain the elevator control forces within the limits of 50-lb push or 35-lb pull in dives to any attainable speed within the maximum permissible speed envelope. The forces required for recovery from these dives (see paragraph 3.1.4) shall not exceed 120 lb. Trim, deceleration devices, etc, may be used to assist in recovery provided that no unusual pilot technique is required.

3.3.17 Auxiliary dive recovery device.- Operation of an auxiliary device for dive recovery at any speed shall always produce a positive increment of normal acceleration, but the total normal load factor shall never be greater than 0.8n_L, controls free, at the most aft critical loading.

3.3.18 Effects of drag devices.- Operation of the speed brakes or other drag devices provided for deceleration, dive-speed limitation, glide-path control, etc, shall not produce objectionable buffet or other undesirable flight characteristics. This requirement shall apply to partial as well as full operation. Drag devices intended for employment in the landing approach shall not produce an objectionable nose-down trim change when operated during the approach. Additional requirements for trim change caused by drag devices are included in paragraph 3.3.19.

3.3.19 Longitudinal trim changes.- The longitudinal trim changes caused by changes in power, flap setting, gear operation, deceleration devices, etc, shall not be so large that peak longitudinal control forces in excess of 10 lb for classes I and III, or 20 lb for class II, are required when such configuration changes are made in flight under conditions representative of operational procedure. Generally, the conditions listed in table IV will suffice for determination of compliance with this requirement. With the airplane trimmed for each specified initial condition, the peak force required to maintain the specified constant parameter following the specified configuration change shall not exceed 10-lb push or pull for classes I and III airplanes, or 20-lb push or pull for class II airplanes. This requirement shall apply to a time interval of at least 5 seconds following the completion of the pilot action initiating the configuration change. The magnitude and rate of trim change subsequent to this time period shall be such that the forces are easily trimmable by use of the normal trimming devices.

TABLE IV

Longitudinal trim change conditions

Condition No.	Altitude	Initial Trim Condition				Configuration change	Parameter to be held constant
		Speed	Gear	Flaps	Power		
1	Low	$1.4V_{S_G}$	Up	Up	PLF	Gear down	Altitude
2	Low	$1.4V_{S_G}$	Down	Up	PLF	Flaps down	Altitude
3	Low	$1.4V_{S_L}$	Down	Down	PLF	Idle Power	Speed
4	Low	1/ $1.15V_{S_L}$	Down	Down	PLF	Takeoff power	Altitude
5	Low	$1.3V_{S_{TO}}$	Down	Takeoff	Takeoff	Gear up	Rate of climb
6	Low	$1.5V_{S_{TO}}$	Up	Takeoff	Takeoff	Flaps up	Rate of climb
7	Medium, high	Level flight	Up	Up	MRP	Idle power	Altitude
8 3/	Medium, high	Level flight	Up	Up	MRP	Actuate deceleration device	Point of aim,
9 3/	Low, medium	Speed for best range	Up	Up	PLF	Actuate deceleration device	Altitude
10	Low	$1.15\ V_{S_L}$	Down	Down	PLF	Extend approach drag device	Speed
11	Medium, high	Level flight	Up	Up	MRP	Augmented power	Altitude

Footnotes 1/, 2/, 3/ follow on page 11.

Footnotes to table IV:

1/ Normal approach speed, if lower than $1.15\ V_{S_L}$.

2/ Class III only. It is required that the normal acceleration changes, if any, not exceed $\pm 0.25g$.

3/ If power reduction is permitted in meeting the deceleration requirements established for the mission, actuation of the deceleration device shall be accompanied by the allowable power reduction.

3.3.20 Longitudinal trim change caused by sideslip.- With the airplane trimmed for straight flight in each of the configurations and at the trim speeds specified in table II, the longitudinal control force required to maintain constant speed in sideslips shall not exceed numerically the lowest force which in the same configuration would produce a normal acceleration change of 1.0g in the accelerated maneuvers of paragraph 3.3.9. In no event, however, shall the force exceed 10-lb pull or 3-lb push on classes I, III, and II-C airplanes, and 15-lb pull or 10-lb push on all others. The sideslips considered shall include angles up to the largest obtainable with 50 lb of rudder pedal force applied in either direction for wings-level trimmed flight. If a variation of longitudinal control force with sideslip does exist, it is preferred that increasing pull force accompany increasing sideslip, and that the magnitude and direction of the trim change be similar for right and left sideslips.

3.4 Lateral-directional stability and control.-

3.4.1 Damping of the lateral-directional oscillations.- In the configurations and over the corresponding speed ranges specified in table II, the damping of the lateral-directional oscillations, with controls fixed and with controls free, when excited by rudder pulses shall be such that the damping parameter $1/C_{\frac{1}{2}}$ has a value not less than that required by curve A of figure 1. Residual undamped oscillations may be tolerated only if the amplitude is sufficiently small that the motions are not objectionable. Generally, the conditions listed in table V will suffice for determination of compliance with these requirements. (See paragraph 6.8 for additional discussion.)

3.4.1.1 For armed airplanes in the firing or bombing configuration and under the critical flight conditions consistent with the tactical mission requirements, the damping parameter $1/C_{\frac{1}{2}}$ shall be at least that required by curve A of figure 1, or at least 1.73, whichever is higher. Under these conditions, the magnitude of any residual oscillation shall not be so great as to cause yaw or pitch deviations which adversely affect bombing or tracking accuracy. (For gunnery or bombing applications, deviations greater than ±5 mils are ordinarily considered excessive.) If it can be established to the satisfaction of the procuring activity that the armament system is such that provision of the degree of damping specified herein will afford no significant improvement in tactical effectiveness, this requirement shall be waived and the requirement of paragraph 3.4.1 shall apply.

3.4.1.2 If an artificial stabilization device is employed, the damping parameter $1/C_{\frac{1}{2}}$ with the artificial device inoperative, shall be at least 0.24 in all configurations. In configuration PA this parameter shall, moreover, have a value at least as high as that required by curve B of figure 1.

3.4.2 Spiral stability.- Spiral stability is not required, but if the spiral motion is divergent, the rate of divergence shall not be so great that, following a small disturbance in bank with controls fixed, the bank angle is doubled in less than 20 seconds in the PA and CR conditions of table V, or 4 seconds in any of the other flight conditions of table II.

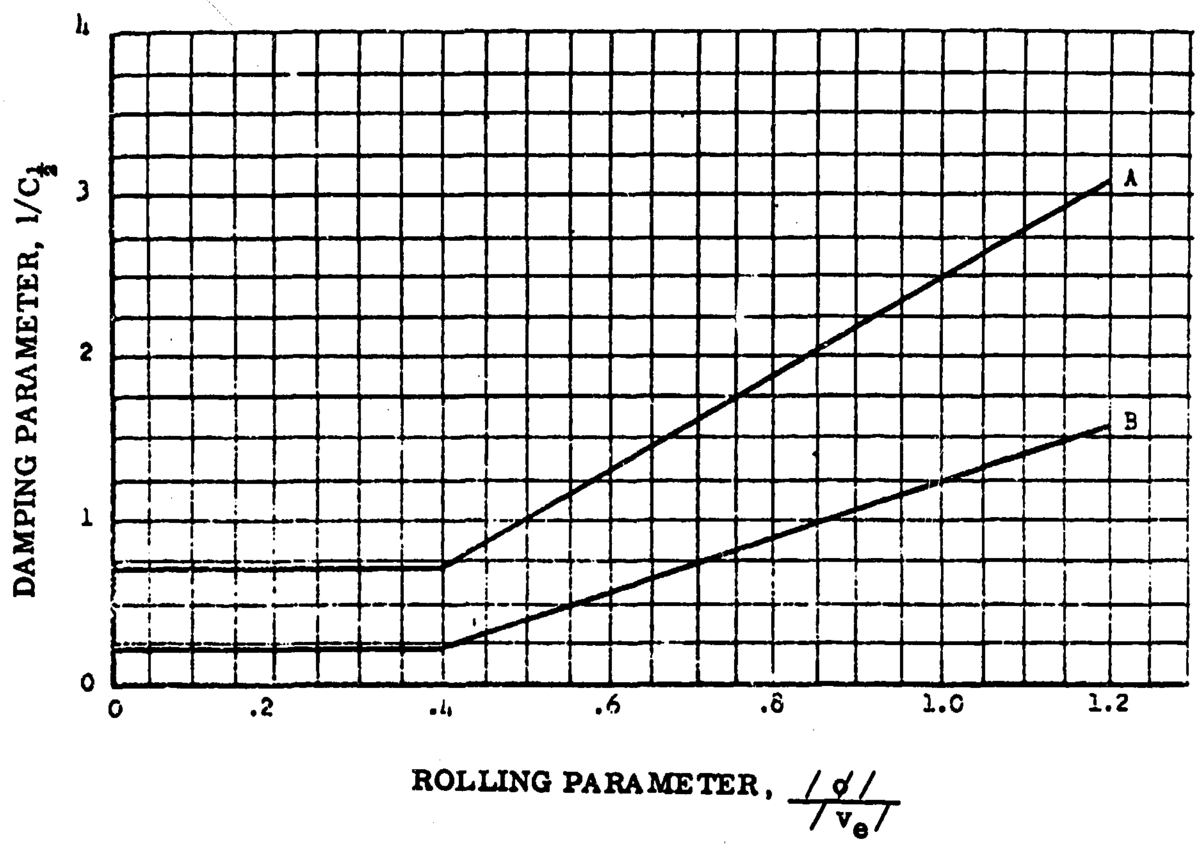


FIGURE 1. Lateral-directional damping requirements

TABLE V

Flight conditions for investigation
of lateral-directional damping

Configuration	Altitude	Speed
CR	Medium, high	Speed for maximum range
P	Low, medium, high	Speed for level flight
D	Medium, high	(a) 0.9 V_H (b) Stabilized speed in 50-degree dive entered from V_H at service ceiling
CO	Low, medium, high	V_H
PA	Low	$1.15V_{S_L}$
L	Low	$1.4V_{S_L}$

3.4.3 Steady sideslip conditions.- Requirements for static directional stability, dihedral effect, and side force variation are expressed in terms of characteristics in steady sideslips. Unless otherwise stated, such requirements shall apply in straight-path (zero turn rate) sideslips up to the sideslip angles produced by full rudder deflection or 250 lb of rudder force or full aileron deflection, whichever is reached first. The requirements shall be met at the lightest normal service loading, in the configurations and speed ranges specified in table II, with the airplane trimmed for wings-level straight flight. In addition, the requirements shall be met on class III and class II-C airplanes in configuration WO at all permissible speeds above $V_{S_{PA}}$, with the airplane trimmed for wings-level straight flight at $1.15V_{S_L}$ in configuration PA. On single-engine, single-rotation propeller airplanes in configuration WO, rudder deflection in the direction opposite to that required for wings-level straight flight need not be considered beyond the deflection for a 10- degree change in sideslip from the wings-level straight flight condition. Although the requirements apply over the entire specified speed range, investigation at the trim speeds specified in table II, and at $1.15V_{S_L}$ in configuration WO, will ordinarily suffice for determination of compliance.

3.4.4 Static directional stability (rudder position).- The airplane shall possess rudder-fixed directional stability such that, in the sideslips specified in paragraph 3.4.3, right rudder pedal deflection from the wings-level position is required in left sideslips, and left rudder pedal deflection is required in right sideslips. For angles of sideslip between ±15 degrees from the wings-level condition, the variation of sideslip angle with rudder pedal deflection shall be essentially linear. Throughout the remainder of the range of required pedal deflections, an increase in pedal deflection shall always be required for an increase in sideslip.

3.4.5 Static directional stability (rudder force).- The airplane shall possess rudder-free stability such that, in the sideslips specified in paragraph 3.4.3, right rudder force is required in left sideslip and left rudder force is required in right sideslip. For angles of sideslip between ±15 degrees from the wings-level, straight-flight condition, the variation of sideslip angle with rudder force shall be essentially linear. At greater angles of sideslip, a lightening of the rudder force is acceptable, but the rudder force shall never reduce to zero or overbalance.

3.4.6 Dihedral effect (aileron position).- The airplane shall exhibit positive control-fixed dihedral effect as indicated by the variation of aileron cockpit control deflection with sideslip in the sideslips specified in paragraph 3.4.3. Left aileron deflection shall be required for left sideslip, and right aileron deflection shall be required for right sideslip.

3.4.6.1 Configuration WO may, if necessary, be excepted from the requirement of paragraph 3.4.6. The aileron cockpit control deflections required in the sideslips of paragraph 3.4.3, however, shall never exceed one-half of full deflection in the negative-dihedral direction.

3.4.6.2 The positive effective dihedral shall never be so great that more than 75 percent of full aileron cockpit control deflection is required in any of the sideslips specified in paragraph 3.4.11.1, or in the sideslips specified in paragraph 3.4.3 up to the sideslip angles which might be required in normal operation or tactical employment.

3.4.6.3 Throughout rolls similar to those required in paragraph 3.4.16 but performed with rudder free, the rolling velocity shall always be in the correct direction.

3.4.7 Dihedral effect (aileron force).- The airplane shall exhibit positive control-free dihedral effect as indicated by the variation of aileron control force with sideslip in the sideslips specified in paragraph 3.4.3. Left aileron control force shall be required for left sideslip, and right aileron control force shall be required for right sideslip. The variation of aileron control force with sideslip angle shall be essentially linear. The aileron force required in the sideslips specified in paragraph 3.4.11.1 shall not exceed 10 lb for stick- or wheel-control airplanes.

3.4.7.1 Configuration WO may, if necessary, be excepted from the requirements of paragraph 3.4.7. The aileron control forces required in the sideslips specified in paragraph 3.4.3, however, shall never exceed 10 lb in the negative-dihedral direction.

3.4.8 Side force in sideslips.- The side force characteristics shall be such that in the sideslips specified in paragraph 3.4.3, an increase in right bank angle accompanies an increase in right sideslip, and an increase in left bank angle accompanies an increase in left sideslip.

3.4.9 Adverse yaw.- The angle of sideslip developed during a rudder-pedal-fixed abrupt roll out of a trimmed, level, steady 45-degree banked turn at $1.4V_{S_{CR}}$ in configuration CR, and at $1.4V_{S_{PA}}$ in configuration PA, shall not exceed 15 degrees. The roll shall continue until a bank angle of 45 degrees is reached in the opposite direction. The aileron deflection held during the roll shall be at least that required for compliance with the lateral control requirements of paragraph 3.4.16. In similar rolls with partial aileron deflections, the angle of sideslip shall be proportional to the aileron cockpit control deflection. If an automatic turn coordination device is employed, the rudder pedals may be free rather than fixed during the roll.

3.4.10 Asymmetric power (rudder free).- On multiengine airplanes, the airplane motions following sudden failure of one engine shall be such that dangerous flight conditions can be avoided by normal pilot corrective control action. As a measure of compliance with this requirement, the following conditions shall be fulfilled: In configuration P, with the most critical engine inoperative (with rpm and pitch simulating the static condition after an engine has failed in flight with no corrective action unless automatically provided), and with the other engine or engines developing normal rated power, it shall be possible at all speeds above $1.4V_{S_G}$, with rudder free, to maintain steady straight flight by sideslipping and banking. The weight shall be that corresponding to the lightest normal service loading, and trim shall be as required for wings-level straight flight with symmetric power. On airplanes with 2 or more engines connected to 1 propeller system by means of a common drive mechanism, "engine inoperative" shall connote complete loss of power to the propeller system.

3.4.11 Directional control (symmetric power).- For all airplanes, directional control shall be sufficiently effective to maintain wings-level straight flight in the configurations and speed ranges specified in table II, with rudder control forces not greater than 180 lb when the airplane is trimmed directionally at the trim speeds specified in table II. Additional requirements for directional control in dives are contained in paragraph 3.4.15. For class III and all carrier-based airplanes in configuration WO at the lightest normal loading, directional control shall be sufficiently effective to maintain wings-level straight flight at all speeds down to V_{SPA}, with rudder control force not exceeding 100 lb when trimmed in configuration PA at $1.15V_{SL}$.

3.4.11.1 For all airplanes, except land-based airplanes equipped with cross-wind landing gear, directional control shall be sufficient to permit development of at least 10 degrees of steady sideslip in configuration L at $1.1V_{SL}$, with rudder control forces not greater than 180 pounds.

3.4.12 Directional control (asymmetric power).- On all multiengine airplanes in configuration TO with the most critical outboard engine inoperative (with rpm and pitch simulating failure in flight with no corrective action unless automatically provided), it shall be possible, at the lightest normal takeoff loading and with takeoff power on the remaining engine, or engines, to achieve and maintain straight flight with a bank angle not greater than 5 degrees, at all speeds above $1.2V_{STO}$. Automatic devices which normally operate in the event of power failure may be used. With trim settings normally employed in a symmetric power takeoff, the rudder pedal force required to maintain straight flight with asymmetric power, as defined above, shall not exceed 180 lb.

3.4.13 Directional control during takeoff and landing.- The rudder control, in conjunction with other normal means of control, shall be adequate to maintain straight paths on the ground during normal takeoffs and landings. For class I airplanes, this requirement shall apply in calm air, and in 90-degree cross winds of at least 50 percent V_{SL} or 20 knots, whichever is less. For classes II and III airplanes, the requirement shall apply in calm air, and in 90-degree cross winds of at least 30 percent V_{SL} or 40 knots, whichever is less. For water-based airplanes, the requirement shall apply to straight paths on the water in calm air and in 90-degree cross winds of at least 20 percent V_{SL} or 15 knots, whichever is less. This requirement shall be met with not more than 180-lb pedal force.

3.4.13.1 Without the use of wheel brakes, classes II-C and III-C airplanes shall be capable of maintaining a straight path on the ground, at airspeeds of 30 knots and above, during takeoffs and landings in a 90-degree cross wind of at least 10 percent V_{SL}, without exceeding a pedal force of 180 lb.

3.4.13.2 For airplanes intended to operate under cold weather conditions, the requirements of paragraph 3.4.13 shall be applicable on snow-packed and ice-covered runways.

3.4.14 Directional control to counteract adverse yaw.- In the rolling maneuvers described in paragraph 3.4.9, but with the rudder employed for coordination rather than held fixed, directional control effectiveness shall be adequate to maintain zero sideslip, with rudder forces not greater than 180 lb.

3.4.15 Directional control in dives.- When trimmed directionally at the service ceiling in configuration P, the rudder control shall be capable of maintaining zero sideslip throughout the dives and pullouts of paragraph 3.3.16 without exceeding 50-lb rudder pedal force for classes I and III airplanes, or 180 lb for class II airplanes.

3.4.16 Lateral control.- Lateral control shall be adequate for compliance with the rolling performance specified in table VI. In cases where the flight conditions coincide, the highest rolling requirements shall govern. In those requirements involving measurement of time, the time shall be measured from the instant of initiation of pilot control action. Unless otherwise established by agreement between the contractor and the procuring activity on the basis of intended tactical employment limitations, the altitudes at which the rolling performance requirements are to be met shall be as specified in table VI. For those requirements which are specified in terms of peak pb/2V, the rate of roll need not exceed 220 deg/sec. In obtaining the required rolling performance, the rudder pedals on classes I-L and II-L airplanes may be held fixed in the positions required for steady flight prior to the roll, or may be employed to reduce adverse sideslip (not to produce favorable sideslip). On class III and all carrier-based airplanes, the rudder pedals shall remain fixed in the position required for steady flight prior to the roll. Automatic coordination devices are permissible, provided that no objectionable characteristics result. If such a device is employed, the rudder pedals may be free rather than fixed during the roll.

3.4.16.1 On class III airplanes, the lateral control requirements relative to configurations P and CO shall apply under all conditions of spanwise weight distribution which may be encountered in combat. On classes III and II-C airplanes in configuration L, the requirements shall apply to all normal takeoff and landing loadings, except that fuel tanks mounted externally at the wing tips or at outboard wing stations may be empty. When these tanks are full, a value of 0.03 may be substituted for 0.05 as the required average pb/2V value.

3.4.16.2 On class II-L airplanes, the rolling acceleration shall be such that in the normal loading condition which produces the most critical rolling moment of inertia (light weight, heavy outboard concentration of spanwise weight), it is possible to attain the peak rate of roll, corresponding to the pb/2V values specified in table VI, in no more than (0.5 +b/100) seconds after initiation of pilot control action, with peak control forces not greater than 25 lb (stick) or 50 lb (wheel).

3.4.16.3 The peak lateral control force required to obtain the rolling performance specified in table VI shall not exceed the following values:

Class I	-	25-lb stick force or 50-lb wheel force
Class II	-	25-lb stick force or 50-lb wheel force
Class III	-	20-lb stick force or 40-lb wheel force
Classes II-C and III-C in Configuration L	-	20-lb stick or wheel force

At 0.8 V_H the peak lateral control force required to obtain the rolling performance specified in table VI shall be not less than half the above values.

3.4.16.4 For all airplanes with wheel-type controls, the wheel throw necessary to meet the lateral control requirements shall not exceed 90 degrees in each direction.

3.4.16.5 Lateral control shall be sufficiently effective to balance the airplane laterally under the conditions specified in paragraphs 3.4.10, 3.4.11.1, 3.4.12, and 3.4.13, with aileron control forces not exceeding those spedified in paragraph 3.4.16.3.(See also paragraphs 3.4.6.2, and 3.4.7.)

3.4.16.6 When trimmed laterally at the service ceiling in configuration P, lateral control effectiveness shall be adequate to maintain the wings level throughout the dives and pullouts of paragraph 3.3.16, with aileron control forces not exceeding 10 lb for stick control or 20 lb for wheel control.

3.4.16.7 At all altitudes, lateral control at M_D shall be sufficient to produce a steady rolling velocity in the correct direction of at least 15 degrees per second without excessive pilot effort.

3.4.16.8 There shall be no objectional nonlinearities in the variation of rolling response with lateral deflection or force causing sensitivity or sluggishness in response to small cockpit control deflections or forces.

3.5 General control and trimmability requirements.-

3.5.1 Control for spin recovery.- In configurations G and L, the normal controls on classes I and III airplanes shall be adequate to provide consistent prompt recoveries from fully developed erect and inverted spins. Recovery shall require no abnormal effort on the part of the pilot, and recovery control forces shall not exceed 250 lb (rudder), 75 lb (elevator), or 35 lb (aileron). Spin recovery characteristics shall be adequate to permit spin demonstration as required by the procuring activity.

3.5.2 Control for taxiing.- It shall be possible to perform all normal taxiing operations without undue pilot effort or inconvenience.

3.5.3 Control surface oscillations.- All control surfaces, and surfaces such as flaps, slats, and speed brakes, shall be free of any tendency toward undamped oscillations apparent to the pilot under the flight conditions specified in table II, at all permissible speeds.

TABLE VI

Minimum rolling performance requirements

	Configurations P, CO (Speeds up to M_H)	Configurations P, CQ (M_M)	Configurations L, PA $1.1V_{S_L}$
Altitude	Low, medium, high	Lowest altitude at which highest value of M_M may be attained	Low
Class I	$\frac{pb}{2V} = 0.09$ at speeds up to 0.8 M_H.	$\frac{pb}{2V} = 0.03$	$\frac{pb}{2V} = 0.09$
Class II	Where V_H is less that 500 knots: $\frac{pb}{2V} = 0.07$ at speeds up to 0.8 M_H or 300 knots, whichever is lower Where V_H is greater than 500 knots: $\frac{pb}{2V} = 0.07$ up to $0.6M_H$; $\frac{pb}{2V} = 0.05$ at $0.8M_H$	$\frac{pb}{2V} = 0.015$	(a) $\frac{pb}{2} = 10$ ft per sec (b) Class II-L: $\frac{pb}{2V} = 0.07$ Class II-C: Same as Class III
Class III	(a) $\frac{pb}{2V} = 0.07$ between $1.1V_{S_G}$ and minimum combat speed (b)(1) At 7,500 feet altitude - From minimum subsonic combat speed (V_R/C maximum) to M_H or M = 0.95 whichever is lower.... Bank angle = 90 degrees in 1 second (2) Between 20,000 - 40,000 feet altitude - From minimum subsonic combat speed (V_R/C maximum) to M_H ....Bank angle = 90 degrees in 1 second.	Bank angle = 30 degrees in 1 second with a linear variation between M_H or M = 0.95 and M_M Bank angle = 50 degrees in 1 second with a linear variation between M_H and M_M	Average $\frac{pb}{2V} = 0.05$ for first 30 degrees of bank, where average $\frac{pb}{2V}$ is based on an average p obtained from the time required to reach 30 degrees of bank

See page 22 for definitions.

3.5.4 Primary flight control trimmability.- The trimming devices shall be capable of reducing the elevator, rudder, and aileron control forces to zero, at all speeds between the minimum trim speeds specified in table VII and the upper limits of the speed ranges specified in table II. In addition, the rate of operation and the effectiveness of the longitudinal trim device shall be such that the elevator control force can be maintained within 10-lb push or pull (20 lb for class II airplanes) throughout the dives specified in paragraph 3.3.16.

TABLE VII

Conditions for trimming to zero control forces

Condition	Configuration	Class	Minimum trim speed
1	Configuration P, at forward and aft critical loading	All	$1.2V_{SCR}$
2	Configuration L, at forward and aft critical loading	All	$1.4V_{S_L}$
3	Configuration PA, at forward and aft critical loading	Carrier-based	1/ $1.15V_{SL}$
		All others	$1.4V_{SL}$
4	Configuration CR, with up to two most critical engines on one side inoperative, wings level	All multiengine	Speed for max range

1/ Or normal approach speed, whichever is lower.

3.5.5 Irreversibility of trim controls.- All trimming devices shall maintain a given setting indefinitely, unless changed by the pilot, by a special automatic interconnect, such as to the landing flaps, or by the operation of an artificial stability device. If an artificial stability device or automatic interconnect is used in conjunction with a trim device, provision shall be made to insure the accurate return of the device to its initial trim position on completion of each artificial stabilization or automatic interconnect operation.

3.5.6 Trim system failure.- Failure of a power-actuated trim system (including sticking or runaway in either direction) shall not result in an unsafe flight condition. Following such failure, it shall be possible to cruise for extended periods and to make a safe landing (including carrier landing for carrier-based aircraft). The use of override provisions or alternate trim mechanisms normally available to the pilot shall be permissible. This requirement shall apply to both aerodynamic and feel-system trim devices.

3.5.7 Roll-pitch-yaw coupling.- In rudder- and elevator-cockpit control-fixed rolls through 360 degrees at all altitudes and permissible speeds, entered from straight flight or from turns, push-overs, or pull-ups ranging from 0g to $2/3n_L$, the resulting yaw motion , sideslip angle, and normal acceleration shall neither exceed structural limits nor cause other dangerous flight conditions such as uncontrollable oscillations. During combat-type maneuvers involving similar rolls through angles up to 180 degrees, the extent of the pitching and yawing shall be not so severe as to impair seriously the tactical effectiveness of the maneuver.

3.6 Stall characteristics.-

3.6.1 Required flight conditions.- The requirements for stall characteristics shall apply at all permissible cg positions, for configurations G, CR, L, and PA in straight unaccelerated flight, and with normal acceleration up to the limits of the operational flight envelopes. Unless otherwise specified by the procuring activity, all stall characteristics requirements apply for normal symmetric external store installations throughout the entire range of store loadings, as well as for the clean airplane if such stores are not installed for some training or tactical missions.

3.6.2 Definition of stalling speed, V_S.- The stalling speed, V_S, is defined as the minimum speed attainable in flight, and is normally associated with the breakdown of airflow over the wing immediately after attaining the maximum over-all trimmed lift coefficient of the airplane. In order to minimise dynamic lift effects, the rate of reduction of speed during an approach to the stall should be not greater than 1 knot per second. The complete stall is generally characterised by uncontrollable pitching or rolling, or by a decrease in normal acceleration in turning flight.

3.6.2.1 For some airplanes, the technique of paragraph 3.6.2 will not result in a true aerodynamic wing stall because of insufficient longitudinal control. Such airplanes, at a given weight, will have varying minimum speeds depending upon the cg position. For purposes of this specification, the stalling speed, V_S, for such an airplane shall be defined as the minimum speed attainable in the applicable configuration with the airplane loaded at its aft critical loading.

3.6.2.2 In the event that considerations other than wing maximum lift or available longitudinal control determine the minimum usable flying speed in any configuration (e.g., ability to perform altitude corrections, ability to take wave off, visibility, etc), the stalling speed V_S for that configuration shall, for the purposes of this specification, be defined as the minimum usable flying speed as agreed upon between the contractor and the procuring activity, provided, however, that such definition is consistent with the definition of stalling speed as employed in structural design considerations, performance guarantees, etc.

3.6.3 Stall-warning requirements.- The approach to the complete stall shall be accompanied by an easily perceptible stall warning which occurs between 1.05 and 1.15 times the stalling speed in configurations G, L, and CR, and between 1.05 and 1.10 times the stalling speed in configuration PA. Acceptable stall warning shall consist of shaking of the cockpit controls, buffeting or shaking of the airplane, or both.

3.6.3.1 Artificial stall warning closely simulating the warning required in paragraph 3.6.3 shall be permitted only if it can be shown that it is not feasible to provide aerodynamic stall warning, and if the artificial device is approved by the procuring activity.

3.6.3.2 For airplanes with limiting elevators described in paragraph 3.6.2.1, no stall warning is required provided a true aerodynamic stall cannot be obtained while loaded at the aft critical loading, and provided no dangerous flight characteristics or motions occur at the minimum attainable speed.

3.6.4 Requirements for acceptable stalling characteristics.- Although it is desired that no nose-up pitch occurs at the stall, a mild nose-up pitch may be accepted, provided that no dangerous or seriously objectionable flight conditions result. The stall shall be considered unacceptable if the airplane exhibits uncontrollable rolling or downward pitching at the stall in excess of 20 degrees from level for classes I and II airplanes, or 30 degrees from level for class III airplanes. These requirements shall apply not only to airplanes with conventional (maximum lift) stalling speeds, but also to complete stalls when attainable by any means on airplanes with stalling speeds as defined in paragraphs 3.6.2.1 or 3.6.2.2.

3.6.4.1 It shall be possible to prevent the complete stall by normal use of the controls at the onset of the 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, and without excessive loss of altitude or buildup of speed.

3.7 Requirements for power- and boost-control systems.-

3.7.1 Normal control system operation.- The control system shall satisfy the applicable mechanical design requirements of the procuring activity, as well as the requirements of this specification. The system shall be capable of providing rapid repeated control movements as might be required in very rough air operation.

3.7.2 Power or boost failure.- All airplanes employing power- or boost-control systems shall be provided with suitable means for control following complete loss of power or boost. The means for control following such failure (e.g., independent boost, direct mechanical control) is referred to herein as the alternate control system.

3.7.3 Transfer to alternate control system - Trim change.- The trim change associated with transfer to the alternate control system, when such transfer is either caused by control power failure or performed intentionally in accordance with routine procedure, shall never be such as to bring about dangerous flight conditions. This shall apply not only in trimmed level flight, but also in dives to V_M with elevator control force, prior to transfer, out of trim by as much as ±5 lb. If dual independent control systems are used, a transfer at cruising altitude in trimmed level flight in configuration P shall cause no perceptible trim change. If the alternate system is not an independent power system, the out-of-trim conditions resulting from a transfer at cruising altitude in trimmed level flight in configuration P shall be within the following limits:

(a) Pitch - with controls free, the change in normal acceleration shall not exceed ±0.5g.

(b) Roll - with controls free, the resulting rate of roll in either direction shall not exceed 5 degrees per second or 10 percent of the rate corresponding to the requirement of table VI, whichever is less.

(c) Yaw - the rudder control force required to maintain zero side-slip shall not exceed 100 lb.

3.7.3.1 Upon transfer to the alternate control system in configuration PA at $1.15V_{SL}$, with trim set for zero control forces prior to transfer, it shall be possible on all airplanes to maintain the airplane attitude with elevator control forces not exceeding 20 lb for all classes of airplanes, aileron control forces not exceeding 10 lb for all classes of airplanes, and rudder control forces not exceeding 50 lb for classes I, II-C, and III airplanes, and 100 lb for class II-L airplanes.

3.7.3.2 On airplanes intended for tactical employment at low altitude, it shall be possible upon transfer to the alternate control system at sea level in configuration P, to maintain straight level flight, at $1.4V_{S_L}$ and at V_H, with control forces not exceeding 20 lb elevator, 10 lb aileron, and 50 lb rudder (100 lb rudder on class II-L airplanes). The longitudinal trim change under these conditions shall never be nose down.

3.7.4 Longitudinal control on alternate system.- At maximum level flight speed at sea level, it shall be possible, with the primary control system inoperative, to obtain at least 3g on class III airplanes, and at least 1.5g or $0.6n_L$, whichever is less, on all other airplanes. Elevator control force in this maneuver, with the airplane trimmed for 1g flight, shall not exceed 75 lb for class I airplanes, 150 lb for class II airplanes, and 120 lb for class III airplanes. This requirement shall be met with the most forward cg location corresponding to a combat loading.

3.7.4.1 It is desired that longitudinal control on the alternate system be adequate to permit recovery from any dive condition normally attained in service operation. As a minimum requirement, it shall be possible on class III airplanes to recover from a dive of at least 50 degrees (or the dive angle resulting in V_D, whichever is less) initiated from service ceiling and maximum level flight speed, with the primary control power system rendered inoperative at 20,000 feet. With the elevator control force trimmed within 10-lb push or pull prior to the simulated failure, and with pilot trim setting operational during recovery, the elevator control force in the recovery shall not exceed 120 lb. The use of any auxiliary dive-recovery or deceleration device having an independent power supply and readily available to the pilot is permissible during the recovery.

3.7.4.2 With the primary control power or boost system inoperative and the elevator control force trimmed to within 5 lb at $1.4V_{S_L}$ ($1.15V_{S_L}$ for carrier-based airplanes) in configuration PA, it shall be possible to execute a safe landing with elevator control forces not exceeding 35 lb for classes I, II-C, and III airplanes, and 50-lb stick force or 80-lb wheel force for class II-L airplanes, when loaded at the most forward cg location corresponding to a normal service loading.

3.7.5 Lateral control on alternate system.- With the primary control power or boost system inoperative, it shall be possible to obtain a peak steady rolling velocity of 15 degrees per second or 50 percent of the pertinent requirement of table VI, whichever is less, with aileron control forces not exceeding 30-lb stick force or 60-lb wheel force. In addition, on class II-C and class III airplanes in configuration L at $1.1V_{S_L}$, the average helix angle obtainable over the first 30 degrees of bank, as defined in table VI under class III configuration L, shall be at least pb/2V (average) = 0.02, with aileron control force not exceeding 20-lb stick or wheel force. During these rolls, the aileron trim shall remain fixed in the wings-level setting, and requirements regarding use of the rudder shall be as specified in paragraph 3.4.16, except that on class III-L airplanes, the rudder requirements shall be as specified for class II-L airplanes in paragraph 3.4.16.

3.7.6 Directional control on alternate system.- With the primary control power or boost system inoperative, it shall be possible to perform the landing of paragraph 3.7.4.2 in a cross wind of 50 percent of the value specified in paragraph 3.4.13, with rudder control forces not exceeding 180 lb.

3.7.7 Ability to trim on alternate system.- With the airplane operating on the alternate control system, it shall be possible to trim the elevator, aileron, and rudder control forces to zero at all level-flight speeds above the minimum speeds specified in table VII.

3.7.8 Feel system failure.- Failure of an artificial feel system shall not result in unsafe flight conditions, and shall not impair the ability to effect a satisfactory landing (including carrier landing for carrier-based aircraft). This requirement may be waived only if it is established to the satisfaction of the procuring activity that the possibility of feel system failure is extremely remote.

4. QUALITY ASSURANCE PROVISIONS

4.1 Quality assurance provisions shall be as specified by the procuring activity.

5. PREPARATION FOR DELIVERY

5.1 Not applicable to this specification.

6. NOTES

6.1 Intended use.- This specification contains the flying qualities requirements for piloted airplanes and shall form one of the bases for determination, by the procuring activity, of airplane acceptability. The specification shall serve as design requirements, and as criteria for use in stability and control calculations, analysis of wind tunnel test results, and flight testing and evaluation.

6.2 Definitions.- Terms and symbols used throughout this specification are defined as follows:

$V_{R/C}$ - Speed for maximum rate of climb with normal rated power.

V_S - See paragraph 3.6.2. The subscripts, e.g., G, L, etc, refer to the airplane configurations described in paragraph 3.1.9.

M - Mach number

V_H, M_H - High speed (Mach number), level flight, augmented power.

V_{NRP} - High speed, level flight, normal rated power.

V_D, M_D - Maximum permissible speed (Mach number) as defined by the maximum permissible speed envelope of paragraph 3.1.4.

V_M, M_M - Maximum operational speed (Mach number) as defined by the maximum operational speed envelope of paragraph 3.1.3.

$\frac{pb}{2V}$ - The helix angle, in radians, described by a wingtip during a rolling maneuver: p is the rate of roll about the body axis in radians per second, b is the wingspan in feet, and V is the true airspeed in feet per second.

n - Normal load factor, in g units, normal to body axis.

n_L - Limit load factor for a given loading based on structural considerations.

$C_{\frac{1}{2}}$ - Number of cycles for the lateral oscillations to damp to half amplitude.

ϕ - Bank angle, degrees.

β - Sideslip angle, degrees.

$\frac{|\phi|}{|\beta|}$ - Ratio of amplitudes of bank and sideslip angles in oscillatory mode.

$\frac{|\phi|}{|v_e|}$ - Rolling parameter, degrees/feet per second.

$$\frac{|\phi|}{|v_e|} = \frac{57.3}{V_e} \cdot \frac{|\phi|}{|\beta|}$$, where V_e is equivalent airspeed in feet per second.

MRP - Military rated power, which is the maximum power (not including augmentation) at which the engine can be operated for a specified period.

NRP - Normal rated power, which is the maximum power at which the engine can be operated continuously.

PLF - Power for level flight at the specified condition.

Augmented power - Military rated power plus augmentation, e.g., jato, afterburner.

Power and thrust - For reaction-type engines, the word "power" shall be replaced by the word "thrust" throughout the specification.

Control surface - An external surface or device which is positioned by a cockpit control, and which produces aerodynamic or jet-reaction type forces in such manner as to control the attitude of the airplane. As used in this specification, the elevator, ailerons, and rudder are the control surfaces or devices which are controlled by the stick (or wheel) and rudder pedals to provide longitudinal, lateral, and directional control, respectively.

Control-fixed - A condition where the pilot's cockpit control is held firmly at a given position. Elevator-fixed, rudder-fixed, and aileron-fixed refer to the condition of the individual cockpit control.

Control-free - A condition where the cockpit control is unrestrained by the pilot. Elevator-free, rudder-free, and aileron-free refer to the condition of the individual cockpit control.

Elevator-fixed neutral point - The cg position for zero elevator cockpit control travel with change in speed, in straight flight at constant throttle.

Elevator-free neutral point - The cg position for zero elevator control force with change of speed for trim, in straight flight at constant throttle.

Elevator 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.

Rudder control force - Difference of push-force components, of the forces exerted by the pilot on the rudder pedals, lying in planes parallel to the plane of symmetry, measured along lines connecting the foremost point of the seat (at midadjustment) and the normal points of application of the pilot's instep on the respective rudder pedals.

Aileron 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 of the pilot's grip.

Aft critical loading - The normal service loading which results in a combination of weight and cg position producing minimum stability. (Ordinarily the lightest gross weight at which the most aft cg position can be obtained in a given configuration at a normal service loading.)

Forward critical loading - Ordinarily, the heaviest gross weight at which the most forward cg position can be obtained in a given configuration at a normal service loading.

Sideslip angle - Angle between the undisturbed flow and plane of symmetry of the airplane, measured in a plane parallel to the relative wind and perpendicular to the plane of symmetry. Plus, or right sideslip, corresponds to incident flow approaching from the right side of the plane of symmetry.

Power off - Reciprocating engine: Throttle closed, propeller windmilling; for configuration L, propeller pitch control in "low pitch" setting or in setting normally used in landing approach; for configuration G, propeller control setting optional.
Jet engines: Idling thrust.
Turboprop engines: Flight-idle setting; for configuration L, propeller control setting as normally employed in landing approach; for configuration G, propeller control setting optional.
Rocket engines: Thrust condition normally used in landing touchdown.

6.3 Interpretation of qualitative requirements.- In several instances throughout the specification requirements, qualitative terms, such as "objectionable flight characteristics," "unacceptable flight conditions," "unusual pilot technique," etc, have been employed as a means of permitting latitude where absolute quantitative criteria might be unduly restrictive. Final determination of compliance with requirements so worded will be made by the procuring activity.

6.4 Rates of operation of auxiliary aerodynamic devices.- Although it has not been considered feasible to include in this specification quantitative requirements for rates of operation of trim tabs, trimmable stabilizers, artificial feel trimmers, etc, or for rates of extension and retraction of flaps, speed brakes, etc, the influence of such rates on the flying qualities may be appreciable and is treated qualitatively in paragraph 3.2.3. In general, trim devices should be operable rapidly enough to enable the pilot to maintain trim under changing conditions as normally encountered in functional and tactical employment of the airplane, and yet must not be so rapid in operation as to induce over sensitivity or trim precision difficulties under any flight condition. Flaps and other high lift devices should operate at a rate sufficient to permit transition into the high lift configuration without undue delay, and yet must not operate so rapidly as to cause sudden or erratic trim or lift changes. This limitation on rate of operation applies also to speed brakes, which, nevertheless, must function at a rate sufficient to meet the tactical and operational needs.

6.5 Control force coordination.- The control forces required to perform maneuvers which are normal for the airplane should have magnitudes which are related to the pilot's capability to produce such forces. As a tentative guide on this subject, it is desired that the relative magnitudes of control forces in coordinated maneuvers should be approximately in the ratio of 50, 175, and 25 pound (or 2:7:1) for elevator, rudder, and aileron force, respectively, for a stick-control airplane. For a wheel-control airplane, the elevator and aileron control forces may be increased by 50 percent. These ratios refer to the peak forces obtained when, starting from level flight in configuration P at medium altitude, a rolling pullout maneuver is performed in which approximately 2/3 of the available rolling velocity is obtained simultaneously with a normal load factor of approximately $1 + 2/3\ (n_L - 1)$, maintaining zero sideslip with the rudder.

6.6 Artificial stability devices.- In general, the use of artificial devices, such as rate dampers or static-stability augmenters, should be considered only when provision of the required degree of stability by aerodynamic or simple mechanical means, such as bob weights, down springs, spring tabs, etc, is shown to be impossible or impracticable. When artificial devices are employed, it is ordinarily desired that, subject to reasonable limitations on weight and complexity, the improvement in the affected flight characteristics be such that an appreciable margin is provided beyond the pertinent minimum requirements. When extensive automatic provisions are incorporated (e.g, automatic pilot with control-stick steering), the requirements of this specification will ordinarily be augmented by specifications governing the procurement of the specialized equipment.

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

6.8 Lateral oscillations.- The inclusion of the roll parameter in the lateral-dynamic stability requirements of paragraph 3.4.1 is based on partial results of several research programs still in progress. Evidence indicates that for very short periods (i.e., below 1.8 seconds) and for values of $|\phi / v_e|$ greater than 1.2, the desired damping may be considerably greater than that specified in figure 1. Pending the incorporation of later research results in the requirements, periods and rolling parameters in these areas should be treated with caution, with the values shown in figure 1 employed as minimums.

6.9 Control position measurement.- In this specification, requirements involving control position have generally been written in terms of cockpit control rather than surface deflection because of the more direct influence of cockpit controls on pilot impressions. Because of the more basic engineering significance of, and need for, surface deflection data, proof of compliance with such requirements will ordinarily be accepted in terms of surface deflections unless linkage peculiarities, stretch, deformation, etc, appear to render such proof invalid.

6.10 Engine considerations.- In certain of the flying qualities design requirements, the effects of engine operation are obvious and are covered directly (e.g., trim changes with power, stability with various power settings, etc). Secondary or less clearly defined effects of engine operation may have an important bearing on flight characteristics and should not be overlooked in design. These considerations include: The effects of engine control and response; the effects of engine gyroscopic moments in airplane dynamic motions; and the effects of engine operation on spin characteristics and spin recovery.

6.11 Control system characteristics.- So many arrangements of flight control systems are possible, considering direct mechanical, power-boosted, and fully powered controls, artificial feel, artificial stabilization, autopilot tie-in, etc, that a limited set of requirements such as those specified in paragraph 3.2 hardly can be expected to rule out all undesirable characteristics. Some of the known important variables, even in a simple system, are friction in the control valve; friction, flexibility, back-lash, gear ratio, and inertia in the control system; viscous damping and preload in the control system or valve; rate limiting of the control actuator; and the level of aircraft static and dynamic stability. The introduction of nonlinear linkages or valve characteristics further multiplies the important variables. In general, the designer should make every effort to provide a linear or smoothly varying response to cockpit control deflection and to control force for all amplitudes of control input, including values of stick force within the range of allowable breakout forces (table I), and small control deflections such as those required in tracking. The phase lag between the cockpit control deflection or force and control surface deflection should be kept to a minimum for reasonably large amplitude motions at frequencies considerably above the airplane natural frequencies, and should not increase unduly at very small control amplitudes.

Custodians:
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Air Force

NUMERICAL INDEX

			Page
Section 1.		SCOPE	1
	1.1	Scope	1
	1.2	Application	1
	1.3	Classification	1
	1.3.1	Land- or carrier-based designation	1
Section 2.		APPLICABLE DOCUMENTS	2
Section 3.		REQUIREMENTS	2
	3.1	General	2
	3.1.1	Airplane loadings	2
	3.1.2	Altitudes	2
	3.1.3	Operational flight envelopes	2
	3.1.4	Maximum permissible speed envelope	3
	3.1.5	External stores	3
	3.1.6	Effects of armament provisions	3
	3.1.7	Release of stores	3
	3.1.8	Deceleration devices	3
	3.1.9	Configurations	3
	3.1.10	Effects of asymmetry	3
	3.2	Mechanical characteristics of control systems	4
	3.2.1	Control friction and breakout force	4
	3.2.2	Adjustable controls	5
	3.2.3	Rate of control displacement	5
	3.2.4	Cockpit control free play	5
	3.2.5	Artificial stability devices	5
	3.3	Longitudinal stability and control	5
	3.3.1	Elevator-fixed static stability	5
	3.3.2	Elevator-free static stability	5
	3.3.3	Exception in transonic flight	6
	3.3.4	Stability in accelerated flight	6
	3.3.5	Short-period oscillations	7
	3.3.6	Long-period oscillations	7
	3.3.7	Control effectiveness in unaccelerated flight	7
	3.3.8	Control effectiveness in accelerated flight	7
	3.3.9	Control forces in steady accelerated flight	7
	3.3.10	Control forces in sudden pull-ups	8
	3.3.11	Control effectiveness in takeoff	8
	3.3.12	Control in catapult takeoff	9
	3.3.13	Control forces in takeoff	9
	3.3.14	Control effectiveness in landing	9
	3.3.15	Control forces in landing	9
	3.3.16	Control forces in dives	9
	3.3.17	Auxiliary dive recovery device	9
	3.3.18	Effects of drag devices	10
	3.3.19	Longitudinal trim changes	10
	3.3.20	Longitudinal trim change caused by sideslip	11
	3.4	Lateral-directional stability and control	11
	3.4.1	Damping of the lateral-directional oscillations	11
	3.4.2	Spiral stability	11
	3.4.3	Steady sideslip condition	13
	3.4.4	Static directional stability (rudder position)	13
	3.4.5	Static directional stability (rudder force)	14
	3.4.6	Dihedral effect (aileron position)	14
	3.4.7	Dihedral effect (aileron force)	14

NUMERICAL INDEX (Continued)

		Page
3.4.8	Side force in sideslips	14
3.4.9	Adverse yaw	14
3.4.10	Asymmetric power (rudder free)	15
3.4.11	Directional control (symmetric power)	15
3.4.12	Directional control (asymmetric power)	15
3.4.13	Directional control during takeoff and landing	15
3.4.14	Directional control to counteract adverse yaw	16
3.4.15	Directional control in dives	16
3.4.16	Lateral control	16
3.5	General control and trimmability requirements	17
3.5.1	Control for spin recovery	17
3.5.2	Control for taxiing	17
3.5.3	Control surface oscillations	17
3.5.4	Primary flight control trimmability	19
3.5.5	Irreversibility of trim controls	19
3.5.6	Trim system failure	19
3.5.7	Roll-pitch-yaw coupling	19
3.6	Stall characteristics	20
3.6.1	Required flight conditions	20
3.6.2	Definition of stalling speed, V_S	20
3.6.3	Stall-warning requirements	20
3.6.4	Requirements for acceptable stalling characteristics	21
3.7	Requirements for power- and boost-control systems	21
3.7.1	Normal control system operation	21
3.7.2	Power or boost failure	21
3.7.3	Transfer to alternate control system - Trim change	21
3.7.4	Longitudinal control on alternate system	22
3.7.5	Lateral control on alternate system	22
3.7.6	Directional control on alternate system	22
3.7.7	Ability to trim on alternate system	22
3.7.8	Feel system failure	23
Section 4.	QUALITY ASSURANCE PROVISIONS	23
Section 5.	PREPARATION FOR DELIVERY	23
Section 6.	NOTES	23
6.1	Intended use	23
6.2	Definitions	23
6.3	Interpretation of qualitative requirements	25
6.4	Rates of operation of auxiliary aerodynamic devices	25
6.5	Control force coordination	26
6.6	Artificial stability devices	26
6.7	Effects of aeroelasticity, control equipment, structural dynamics, etc	26
6.8	Lateral oscillations	26
6.9	Control position measurement	26
6.10	Engine considerations	27
6.11	Control system characteristics	27

APPENDIX II

WIND SPEED INFORMATION FOR MIL-F-8785B

Summary of Information Supplied By:

Environmental Technical Application Center
USAF
Washington, D.C.

(See Reference G2 for full text)

July 1966

PREFACE

This report was prepared to answer a request from ASD (ASBW), Wright-Patterson AFB, Ohio. The Flight Control Division (FDCC) of the Flight Dynamics Laboratory required environmental data on the probability of high cross winds at existing airfields to aid in determining realistic cross wind landing criteria for MIL-F-8785B, "Flying Qualities of Piloted Airplanes."

Questions concerning this report or related problems should be directed to ASBW.

WIND SPEED INFORMATION FOR MIL-F-8785

I. INTRODUCTION

At present MIL-F-8785, "Flying Qualities of Piloted Airplanes" specifies that certain aircraft be able to land in cross winds up to 40 knots. The validity of this requirement has been questioned and it has been suggested that the requirement be relaxed. Data provided herein show that this requirement should be lowered considerably for flying operations conducted from existing military airfields.

II. DATA

Wind is defined as air in motion relative to the surface of the earth. For meteorological purposes, the surface wind is taken as the horizontal component of the wind at some height near the surface. This height has varied at each station and from station to station over the years. In recent years at USAF bases the anemometer height has been about 13 feet. Weather stations record the average speed over some time interval. Values given in this report are averages over a 1-minute period. There are large time and space variations in wind speed and terrain features exert a large influence. Data presented in this report are taken at airfield locations which generally have flat terrain and one mile or more of unobstructed flow upwind from the anemometer. Hourly 1-minute wind speeds were collected from 266 locations in the contiguous U. . and 36 overseas locations over the available period of record. The period of record varies but 5 years of record was available for almost all stations with a large percentage of the stations having in excess of 10 years. Three maps of the contiguous U.S. showing isolines of mean annual percent of time or mean annual hours that surface wind speeds exceed 22 knots, 28 knots, and 34 knots are given as Figure 1, Figure 2, and Figure 3, respectively.

III. RECOMMENDATION

The requirement that aircraft be able to land in cross winds up to 40 knots is too severe for existing military airfields. This requirement could be relaxed to 25 knots and still achieve at least 99.5% operational effectiveness.

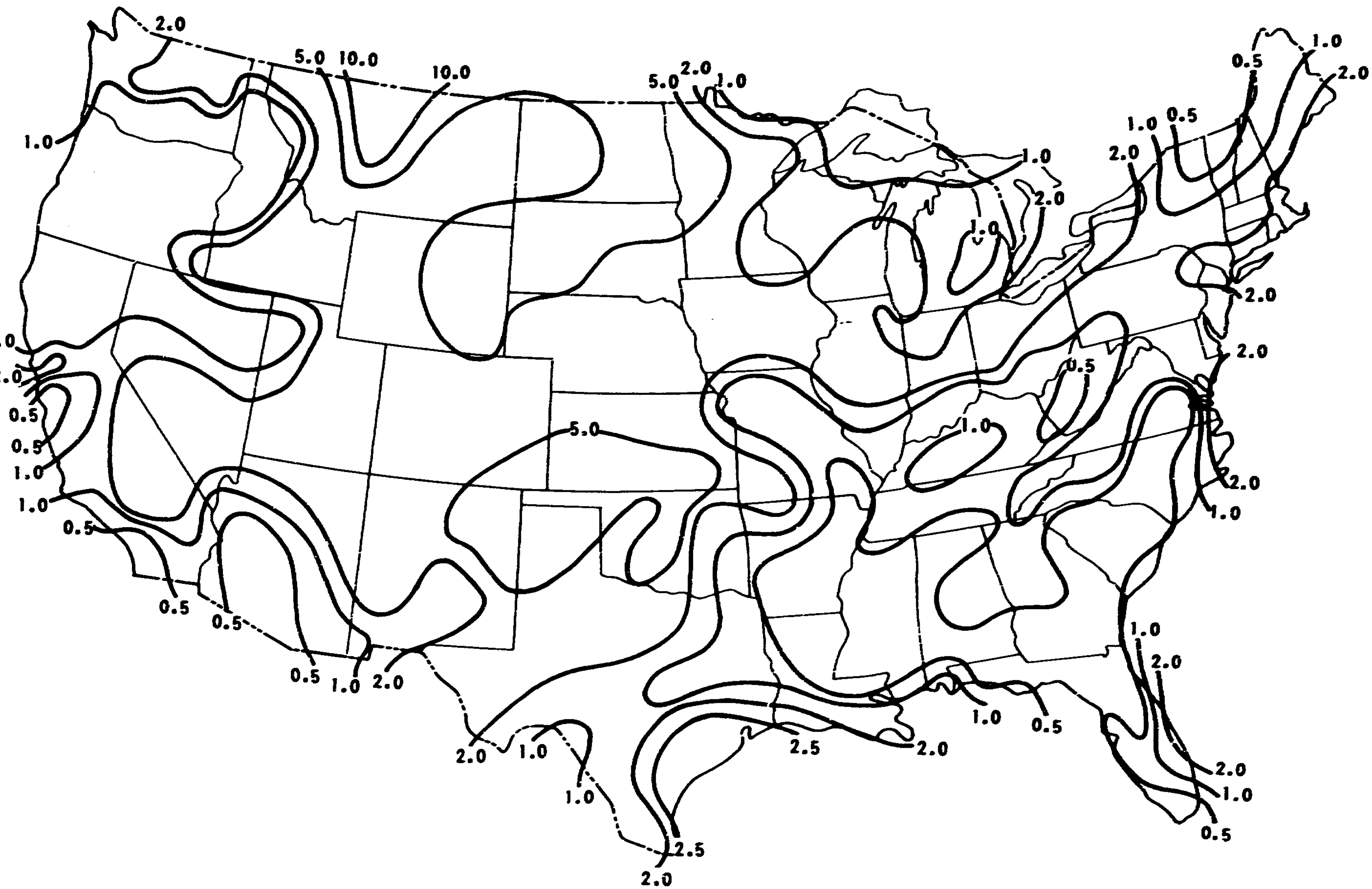


Figure 1 (APPENDIX II)

AVERAGE ANNUAL PERCENT OF TIME THAT SURFACE WIND SPEED (AVERAGE FOR A 1-MINUTE PERIOD) EQUALS OR EXCEEDS 22 KNOTS

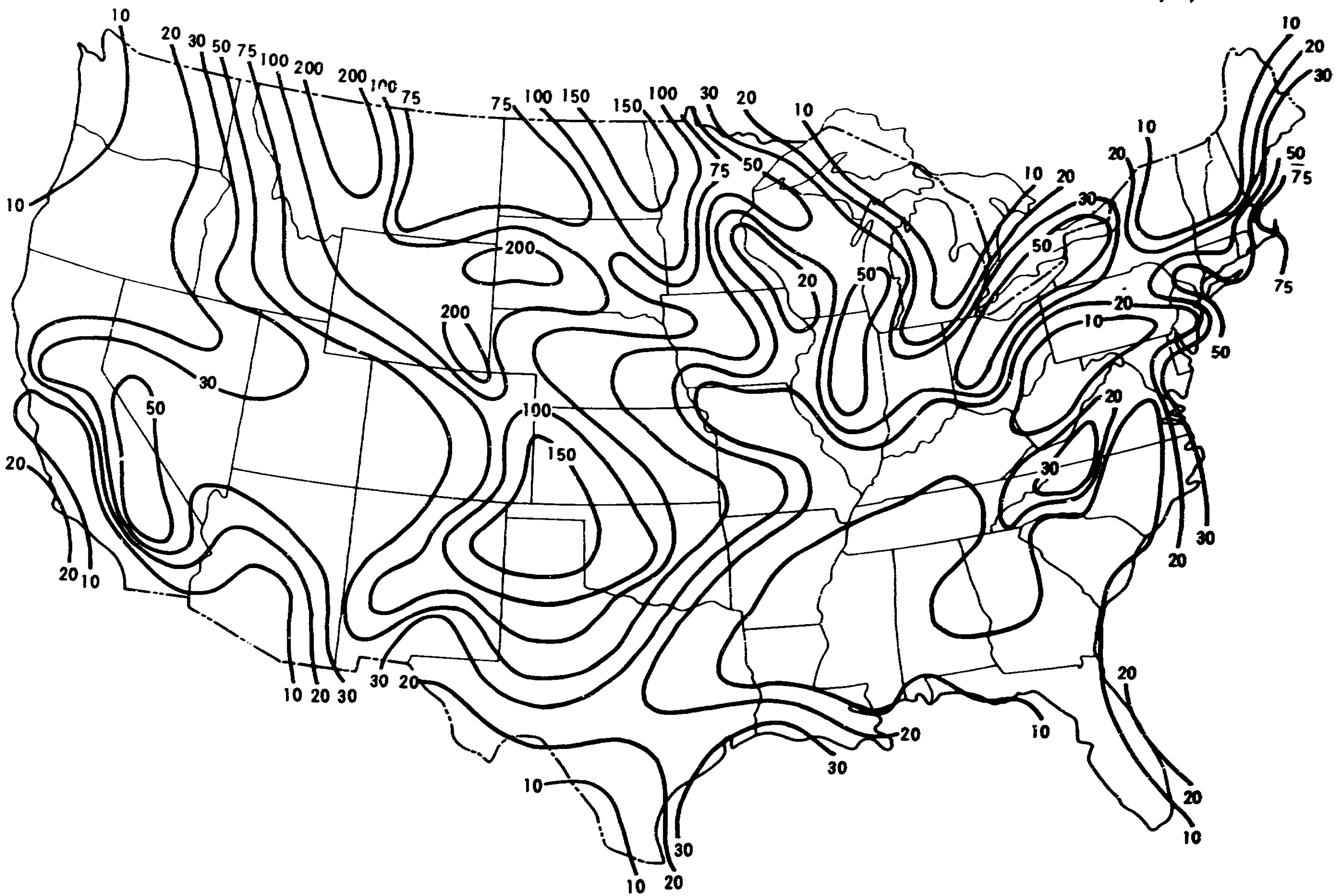


Figure 2 (APPENDIX II)
AVERAGE ANNUAL NUMBER OF HOURS THAT SURFACE WIND SPEED
(AVERAGE FOR A 1-MINUTE PERIOD) EQUALS OR EXCEEDS (28 KNOTS)

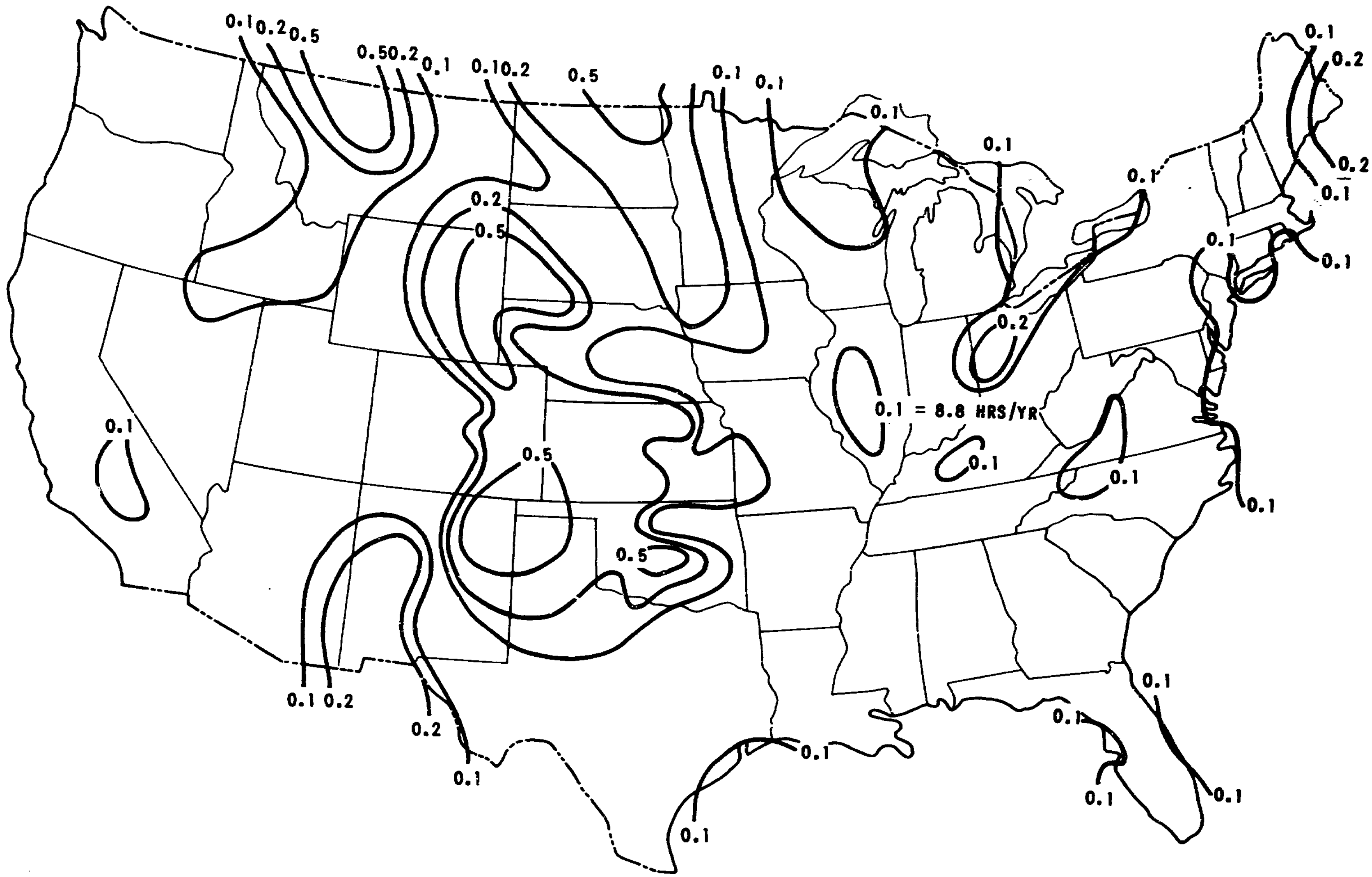


Figure 3 (APPENDIX II)

AVERAGE ANNUAL PERCENT OF TIME THAT SURFACE WIND SPEED
(AVERAGE FOR A 1-MINUTE PERIOD) EQUALS OR EXCEEDS 34 KNOTS

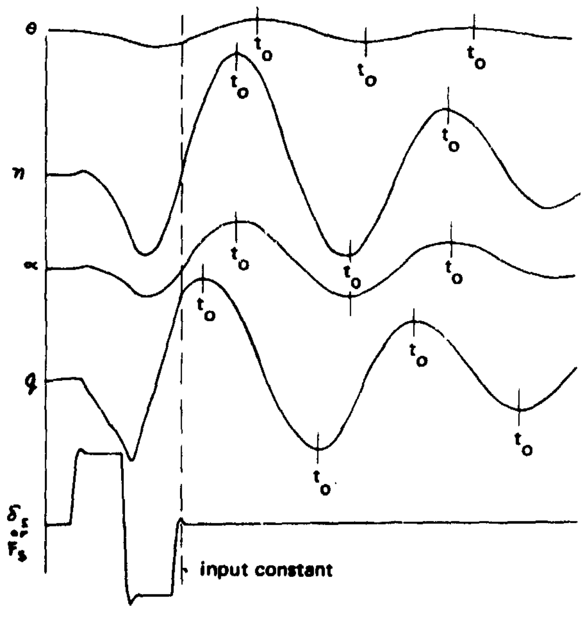
APPENDIX III

MEASUREMENT OF ω_n AND ζ OF SECOND-ORDER RESPONSES

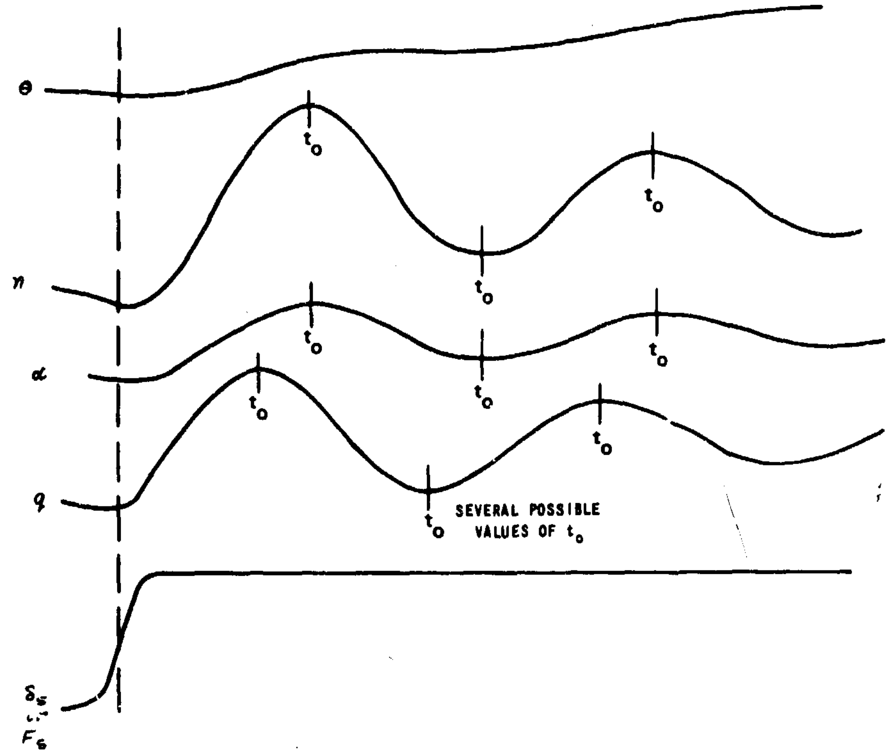
This appendix is a discussion of various hand-measurement techniques for determination of ω_n and ζ of second-order modes from time histories. If the response being measured is higher than second order, then these techniques can be successfully applied only if the residues of the unwanted roots are small or if the contribution of these roots has been identified and subtracted from the response. Analog-matching techniques (Appendix IV-D) can be used for those cases where the hand techniques cannot be used at all or where more accuracy is desired.

Several hand techniques are presented in Table I, covering different ranges of damping ratios. The transient peak ratio method is useful for damping ratios from -0.5 to +0.5. This is the most important method because ζ_p, ζ_{sp}, and ζ_d normally fall within this range. The time-ratio method and the maximum-slope method yield good results for damping ratios from 0.5 to 1.0, and may be useful up to ζ = 1.4. A method is also discussed which is specifically applicable to overdamped short-period modes and which can be used for damping ratios above 1.1 or 1.2. All the hand methods require that the section of time history being analyzed begin with zero slope at some time, t_o , after the input becomes steady state (the input is control force when controls-free dynamics are being measured, and control position when controls-fixed dynamics are being measured). Any of the airplane responses having a steady state can be analyzed provided this initial condition is met, as shown in the following examples for short-period response to elevator inputs.

<u>Time History of Doublet Input (recommended by Appendix IVC):</u>

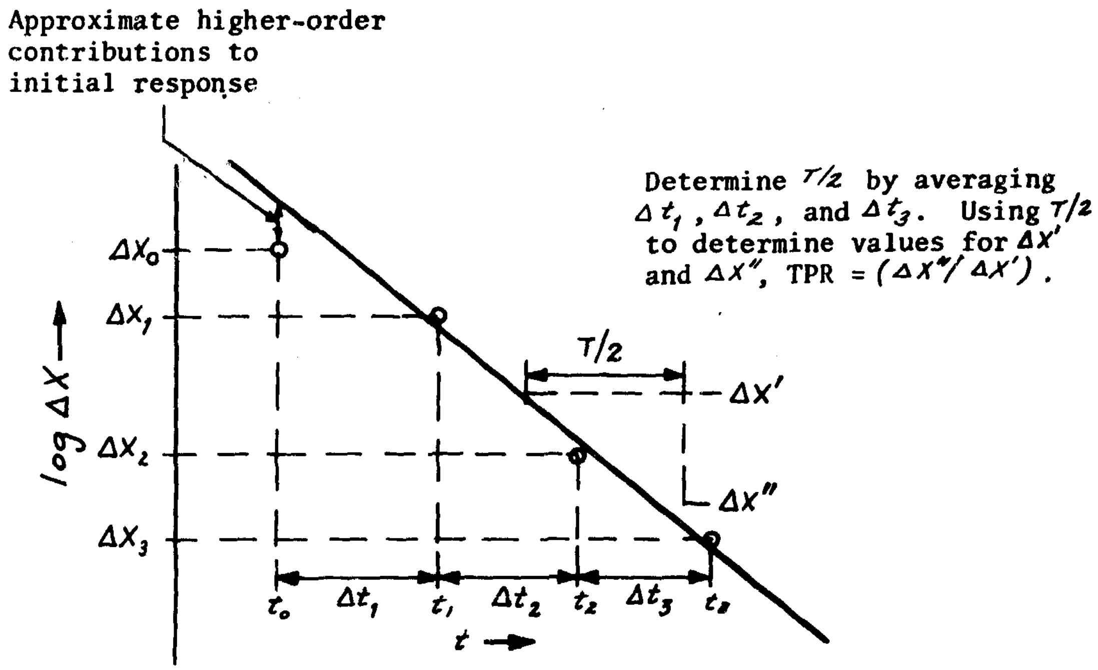


Time History of Step Input:



The application of the required initial conditions to these examples is straightforward, but one pitfall should be mentioned. The initial response of the airplane to nonideal step inputs is normally quite distorted. Therefore, use of the instant of the input as t_o can lead to errors in obtaining ω_n and ζ. Particular caution must be observed in analyzing pitch-rate response. Even for an ideal step input, the phasing of the pitch-rate response is such that t_o cannot be taken at the instant of the input. Failure to observe this caution will result in gross errors in ω_n and ζ.

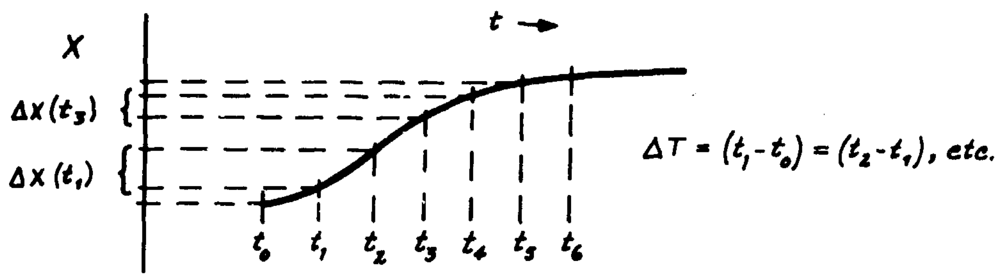
The second version of the transient peak-ratio (TPR) method shown in Table I is the easier of the two to use, and is generally more accurate because the TPR determined from the first version is very sensitive to how accurately the steady state can be determined. In general, the portion of the response immediately after the input becomes steady state is likely to be somewhat distorted from second order due to control system dynamics. The entire response may be slightly contaminated with higher-order effects or nonlinearities. If the damping ratio is low enough that several overshoots are apparent in the response, a plot of $\overline{\Delta X}$, $\overline{\Delta X}_2$, etc., or ΔX_0, ΔX_1, etc., versus t on semilog paper will give a feeling for the magnitude of these distortions. The TPR can then be determined by fairing a straight line through the data points, as follows:



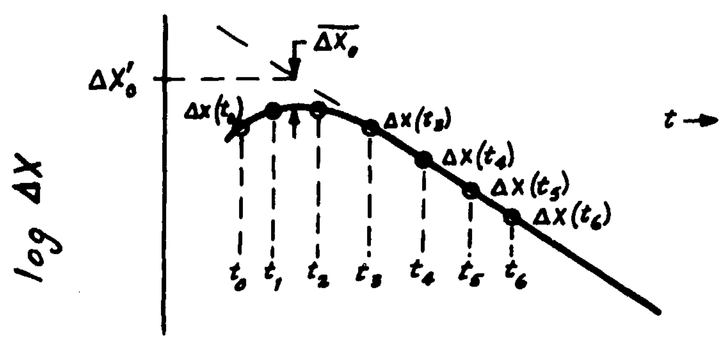
The time-ratio method provides three separate checks on ζ and ω_n, which is useful in getting a feeling for how much the time history is distorted from pure second order. Notice, however, that Δt_1 should generally be regarded as the least accurate of the time measurements because it is very sensitive to errors in choosing t_0, and because the early portion of the response is especially affected by control system dynamics. Remember also that Δt_3 is sensitive to errors in selection of the steady state.

The maximum-slope method was developed to eliminate some of the inherent inaccuracies of the time-ratio method by eliminating the need to determine the steady state and t_0. The method does introduce an inaccuracy of its own, however. The inaccuracy is caused by the fact that ΔX_1 is generally a small amplitude to measure, usually 12 to 24 percent of ΔX. This inaccuracy affects ζ primarily, but ω_n tends to be somewhat insensitive to errors in determining ΔX_1. In the original version of the method (Reference B98), a time ratio was computed in addition to $\Delta X_1/\Delta X$ as a cross-check on ζ; but this was eliminated in the present version because it was subject to considerable inaccuracy (similar to the inaccuracy of Δt_1 in the time-ratio method).

The method for separated real roots generally works well but is somewhat sensitive to errors in determining the steady state. Therefore, if the exact location of the steady state is in question (due to phugoid excitation, for instance), a modification of the basic method developed in Reference B98 can be employed. The only difference in the modified method is that the values of $\Delta X(t)$ to be plotted on semilog paper are measured so as to eliminate the need to determine the steady state. Each value of $\Delta X(t)$ is now defined as $[X(t+\Delta t) - X(t)]$, where ΔT is a fixed time increment.



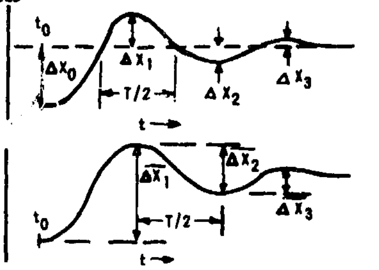
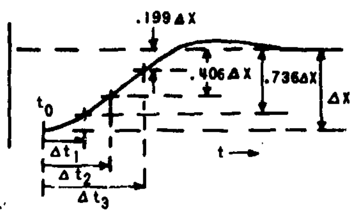
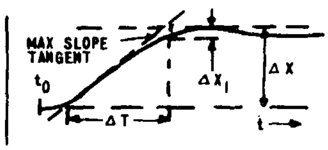
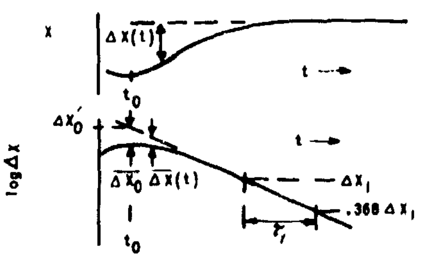
After plotting $\Delta X(t)$, defined in this manner, versus t on a semilog plot, the analysis proceeds exactly as shown in Table I for the basic method.



This modified method will exhibit more scatter in the data points on the semilog plot than will the basic method, but the slope of the line faired through the data may be more accurate. A word of caution is in order, however. If the basic method is applied to a response whose damping ratio is thought to be greater than 1.0 but is actually slightly less than 1.0, the error will probably be noticed because no part of the semilog plot will be found to which a straight line can be fitted (although this behavior can also result from an error in determining the steady state). With the modified method, however, a straight line can usually be fitted and two real roots obtained. Use of these roots to obtain ω_n and ζ will result in gross errors. Therefore, if the damping ratio is thought to be only slightly greater than 1.0, check the results of separated real-roots method by using the time-ratio or the maximum-slope method.

Table I (APPENDIX III)

HAND MEASUREMENT TECHNIQUES FOR DETERMINATION OF ζ & ω_n FOR SECOND-ORDER SYSTEMS

METHOD & ζ RANGE	METHOD PARAMETERS	USE OF PARAMETERS TO OBTAIN ζ & ω_n
TRANSIENT PEAK-RATIO METHOD $-0.5 < \zeta < 0.5$	GENERALIZED RESPONSE 	THE FIRST SKETCH ILLUSTRATES THE METHOD DESCRIBED IN REFERENCE 897. THE SECOND SKETCH ILLUSTRATES A MODIFICATION TO THIS METHOD (REFERENCE 898) WHICH DOES NOT REQUIRE DETERMINATION OF THE STEADY STATE RESPONSE. FIRST, DETERMINE THE TRANSIENT PEAK RATIO: TPR $= \Delta X_1/\Delta X_0 = \Delta X_2/\Delta X_1 = \Delta X_3/\Delta X_2 = \overline{\Delta X_2}/\overline{\Delta X_1} = \overline{\Delta X_3}/\overline{\Delta X_2}$, ETC. USE TPR TO COMPUTE ζ FROM FIGURE 1. DETERMINE THE DAMPED PERIOD, T, FROM THE TIME BETWEEN CROSSINGS OF THE STEADY STATE OR TIME BETWEEN PEAKS. $\omega_n = (2\pi/T\sqrt{1-\zeta^2})$.
TIME-RATIO METHOD $0.5 < \zeta < 1.2$		THE TIME-RATIO METHOD IS DESCRIBED IN REFERENCE 897. AFTER DETERMINING THE STEADY STATE AND Δt_1, Δt_2 Δt_3 COMPUTE THE TIME RATIOS $(\Delta t_2/\Delta t_1)$, $(\Delta t_3/\Delta t_1)$, $\left(\frac{\Delta t_3 - \Delta t_2}{\Delta t_2 - \Delta t_1}\right)$. USE OF THESE RATIOS ON FIGURE 2 YIELDS THREE VALUES OF ζ. IN PRACTICE, USE JUDGMENT OR AN AVERAGE OF THE THREE VALUES AS ζ. AGAIN, USE FIGURE 2 TO DETERMINE $(\omega_n t_1)$, $(\omega_n t_2)$, $(\omega_n t_3)$. DIVIDE BY t_1, t_2, t_3 RESPECTIVELY TO DETERMINE ω_n.
MAXIMUM-SLOPE METHOD $0.5 < \zeta < 1.2$		THE MAXIMUM-SLOPE METHOD WAS DEVELOPED IN REFERENCE 898 AND THE PRESENT VERSION PRESENTED IN REFERENCE 899. AFTER DETERMINING THE MAXIMUM-SLOPE TANGENT AND ΔX, ΔX_1, COMPUTE THE RATIO $(\Delta X_1/\Delta X)$. USE $(\Delta X_1/\Delta X)$ TO OBTAIN ζ AND $(\omega_n \Delta T)$ FROM FIGURE 3. THEN $\omega_n = (\omega_n \Delta T)/\Delta T$.
SEPARATED-REAL-ROOTS METHOD $1.1 < \zeta$		WHEN $\zeta > 1.0$, THE SECOND-ORDER RESPONSE IS ACTUALLY COMPOSED OF TWO UNEQUAL REAL ROOTS. A METHOD FOR DETERMINING THESE ROOTS IS DESCRIBED IN REFERENCE 897. DETERMINE THE STEADY STATE AND PLOT $\Delta X(t)$ VERSUS t ON SEMILOG PAPER. AFTER THE FASTER ROOT HAS DECAYED, THE SEMILOG RESPONSE WILL BE A STRAIGHT LINE, WHOSE SLOPE DETERMINES THE SLOWER ROOT (τ_1) AS SHOWN. THE FASTER ROOT $(\tau_2) = \tau_1 (\overline{\Delta X_0}/\Delta X_0')$, OR CAN BE DETERMINED FROM THE SLOPE OF A SEMILOG PLOT OF $\overline{\Delta X}(t)$ VERSUS t. THEN $\omega_n = \sqrt{\frac{1}{\tau_1}\frac{1}{\tau_2}}$, $\zeta = \frac{(\tau_1+\tau_2)\omega_n}{2}$ (SEE TEXT FOR A USEFUL VARIATION OF THIS METHOD)

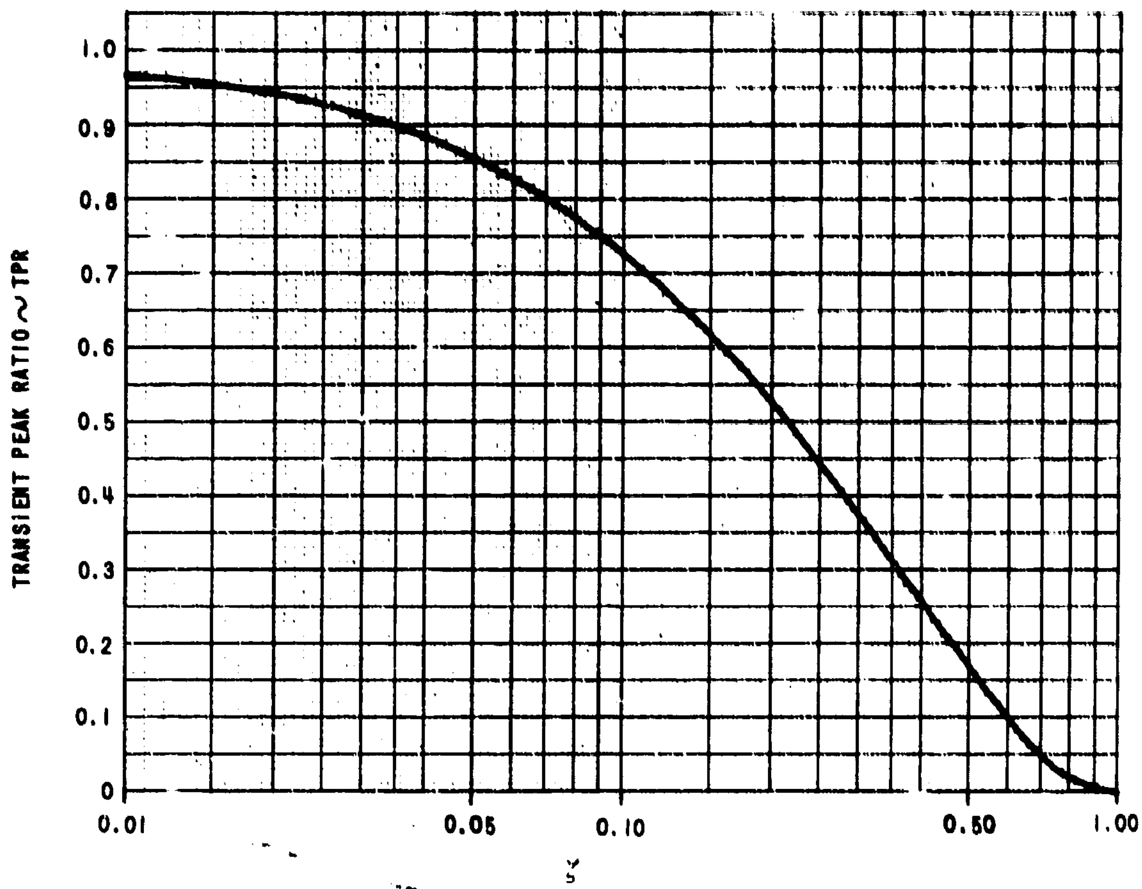


Figure 1 (APPENDIX III)
DETERMINING ζ BY TRANSIENT-PEAK-RATIO METHOD

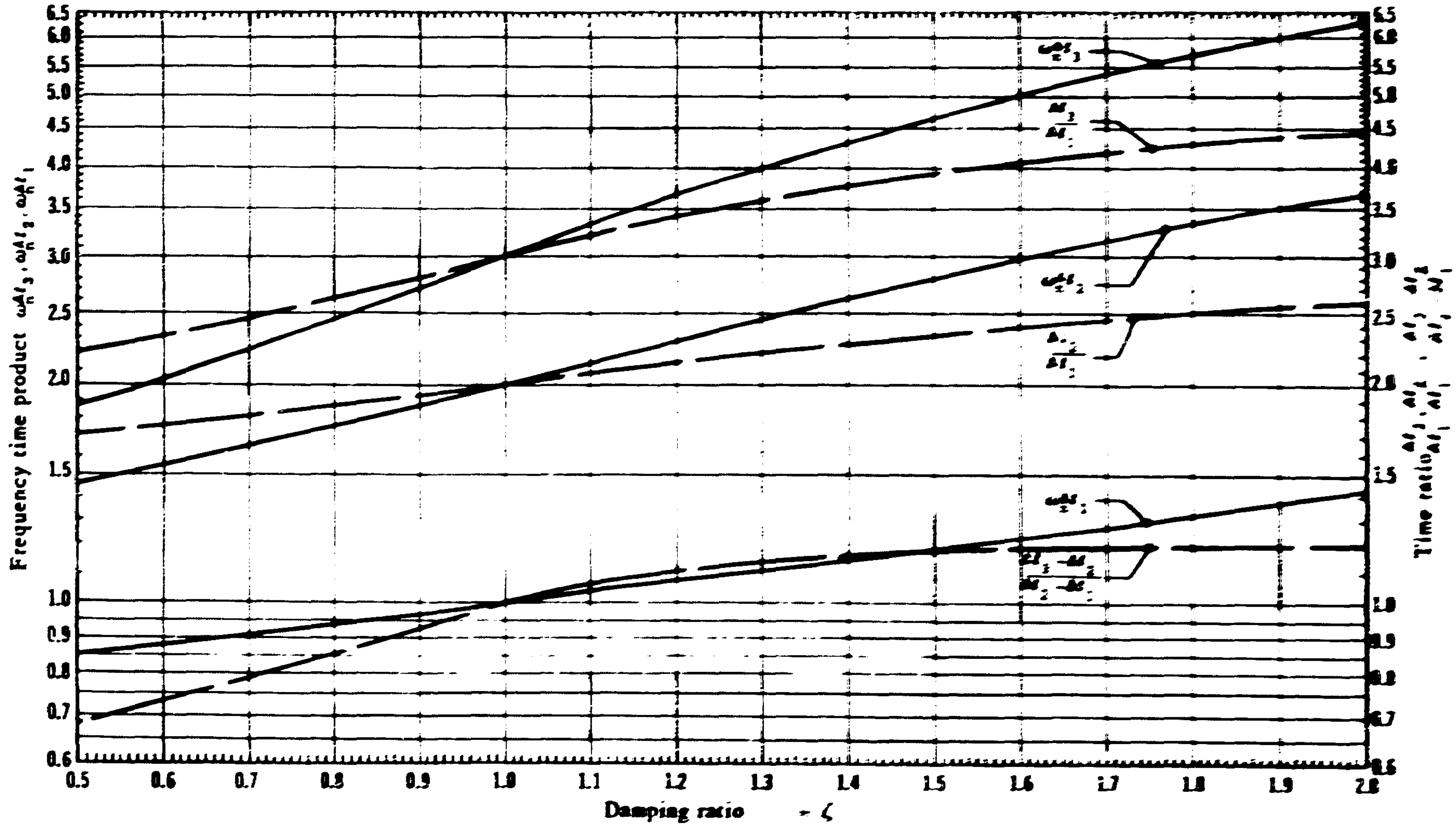


Figure 2 (APPENDIX III)

DETERMINING ζ AND ω_n BY TIME-RATIO METHOD

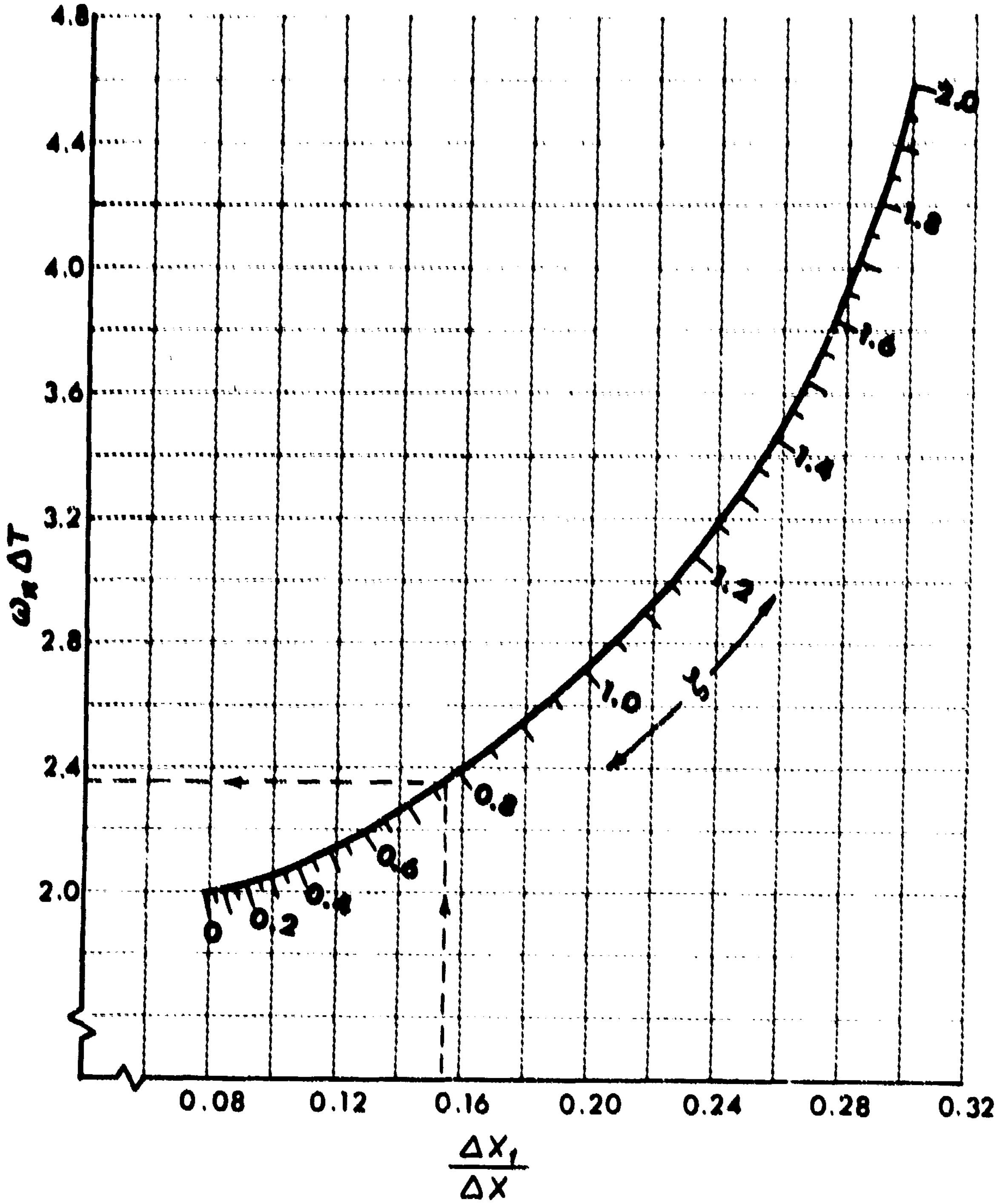


Figure 3 (APPENDIX III)
DETERMINING ζ AND ω_n BY MAXIMUM-SLOPE METHOD

APPENDIX IV

MEASUREMENT TECHNIQUES FOR SELECTED LONGITUDINAL PARAMETERS

IVA. LONGITUDINAL STATIC STABILITY

The obvious method for testing against 3.2.1.1 is to first trim the airplane; and then use elevator control alone to change and restabilize airspeed, leaving the throttle and trimmer controls at their trim settings. The altitude, of course, will vary constantly during this test; but careful programming of the test sequence can keep the altitude within reasonable bounds, for subsonic speeds at least. At low speed this test gives an excellent indication of phugoid stiffness or any divergent roots, though it is time-consuming. But at higher speeds, larger altitude changes are encountered during the runs. Then the lack of any unique relation between altitude and speed can cause difficulty, because compressibility effects are functions of both h and V. For example, gross differences in apparent "stability" are common transonically between results of airspeed variation at constant Mach number and results of Mach variation at constant airspeed. Neither of these latter two tests give the desired result which is an indication of long-term stability.

The acceleration-deceleration method is a popular method because it is the quickest. After trimming, the airplane is decelerated to the specified lower limit of the speed range by reducing power and holding altitude constant with the elevator. The airplane is next accelerated to the maximum specified speed and then decelerated to the trim speed. All this is done at constant altitude. The method is fast, and provides an almost infinite number of data points because data can be recorded continuously during the maneuver. One practical problem, however, is that unless the pilot changes power slowly and moves the elevator smoothly so that normal acceleration is held very close to 1.0, the data will include unwanted contributions from δ_e/n (constant speed).

At low speeds, the control force and position versus airspeed gradients obtained from the above two methods will be essentially equal. At high speeds, the gradients will differ by a factor which depends on altitude and airspeed. The difference is primarily due to the fact that large altitude changes accompany small airspeed changes during high-speed flight at constant throttle, as explained in the discussion of 3.2.1.2. This means that air density and the speed of sound will vary appreciably during static stability tests using the stabilized method, but not during acceleration-deceleration tests. It is not obvious which type of test most accurately measures "static stability." It is obvious, however, that the stabilized method is very time-consuming and exhibits poor repeatability for high-speed flight conditions. For this reason, the acceleration-deceleration method is generally preferable for testing at high speeds.

A possible source of error, which can accompany the acceleration-deceleration tests, should be mentioned. This error is often present because the tests are usually conducted using off-trim throttle settings. The pitching moment and vertical force changes with speed at an off-trim throttle setting may be significantly different from those obtained at the trim throttle setting. Unless the engine thrust and slipstream effects due to changing throttle and airspeed are known before the test, it is obvious that the control force and position data must be obtained with the throttles at their trim settings.

In view of the above discussion, the following techniques are recommended as a reasonable compromise between accuracy and practicality. At low speeds where the altitude changes associated with constant-throttle airspeed changes are small and where operation near the stall speed is required, the constant-power stabilized-airspeed method works very nicely. At high speeds (say $M > 0.4$) where the altitude excursions associated with the stabilized-airspeed method become larger, economy considerations dictate that some form of the acceleration-deceleration method be employed. To ensure that the results of the test give a reasonable indication of throttle-fixed stability, the following procedure should be used. After trimming the airplane, reduce throttle and allow the airplane to decelerate at constant altitude to the low end of the desired speed range, taking no data. When the desired speed is reached, advance the throttle to the trim setting and hold normal acceleration as close to 1.0 g as is possible without use of abrupt control movements. The reverse procedure should be used for speeds above the trim speed. Data should only be taken during the acceleration and deceleration runs where the throttle is at the trim setting.* For climbing or descending Flight Phases, other appropriate throttle settings should be used; but the acceleration-deceleration runs are still to be conducted in level flight.

In testing for compliance with paragraph 3.2.1.1 and 3.2.1.1.1, if the control gradients obtained for a number of trim points are stable over the specified speed range, relatively few additional trim points will be needed. If an unstable region is found far from the trim point, however, the test should be repeated with the airplane trimmed closer to the unstable region; the airplane may or may not be stable within the specified speed range about the new trim point.

IVB. FLIGHT-PATH STABILITY

The climb-angle-versus-airspeed data used to demonstrate compliance with 3.2.1.3 can be obtained during the stabilized-airspeed tests for static stability at low airspeeds.

By the nature of the way in which the climb-angle-versus-airspeed criterion was developed, the climb angle to be measured is the climb angle relative to the air, not the ground. This fact is mentioned for the benefit of flight-test engineers contemplating use of Doppler radar or ground-based tracking equipment to obtain the data. If such methods are used, the wind must be calm, or at least constant and accurately measured.

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*The combined effect of thrust and acceleration can be seen by comparing acceleration and deceleration data, but for showing specification compliance only the data for the trim throttle setting are pertinent.

The most straightforward method is probably to use a well-calibrated airspeed indicator and an accurate measure of vertical speed, such as a radar altimeter. The climb angle is then equal to

$$\sin^{-1}\left[\frac{\text{vertical speed}}{\text{true airspeed}}\right]$$

Still air is necessary in any case, to minimize data scatter. Because of thrust effects it has been found necessary to keep altitude excursions small (less than 1000 ft) to get an acceptably accurate curve of flight path angle versus speed. The trim flight-path angle can have a marked effect on the results.

IVC. SHORT-PERIOD RESPONSE MEASUREMENT

The best flight-test maneuver to measure the short-period response is normally a sharp doublet input to the stick (abruptly push the stick, then pull the stick through trim to a nearly equal deflection aft of trim, and then release the stick for the stick-free data or return it to trim for stick-fixed data). This type of input tends to excite the short-period mode very nicely while suppressing the phugoid mode (see Reference B98). A pull-and-hold (step) or a pull-and-return (pulse) input may work fairly well for certain flight conditions, but the resulting phugoid excitation will normally be large enough to distort the short-period response.

The limits on how fast the doublet input must be are normally fairly broad and are readily apparent to the pilot. If the input is too slow, the phugoid mode will be excited, and the airspeed changes may become excessive. If the input is too fast, the airplane may not respond enough to produce a measurable response.

The best airplane response for measuring the short-period natural frequency and damping ratio is a function of several theoretical and practical factors. Airspeed, of course, is obviously a poor response to use because it normally exhibits negligible short-period response.

The pitch responses of the airplane are attractive for measurement purposes because the entire instrumentation package can be mounted inside the airframe. Since rigid-body pitch responses are independent of the longitudinal location of the sensor, the sensor can be located so as to minimize the effects of structural modes. There are several fundamental problems associated with the use of pitch responses, however. The first problem is that when the phugoid and short-period modes are not well-separated in frequency, it becomes very difficult to excite the short-period mode without having the phugoid mode appear in the pitch response. This is primarily a problem at low speeds, such as in the landing approach, but can be of significance at any speed. The phugoid response can be minimized by using a symmetrical input such as a doublet, so that the pitch attitude returns to its trim value after the short-period transient. In addition, the numerator dynamics of the pitch-rate transfer function are such that it may be difficult to find a usable portion of the $\dot{\theta}$ response satisfying the initial conditions required by the hand techniques of Appendix C. In view of these problems, analog matching techniques, such as those discussed in Appendix IVD, should be used to analyze the pitch responses.

Normal acceleration response is relatively easy to measure because, as with the pitch responses, the sensor can be mounted inside the airframe. Normal acceleration is much less susceptible to phugoid excitation than is pitch response. One difficulty, however, is that the numerator dynamics of the n/δ_e transfer function are very sensitive to the longitudinal location of the sensor, so that sensor location must be chosen carefully to obtain a usable time history. Normal-acceleration sensors are very sensitive to structural bending modes, and especially to local structural vibrations. Since structural factors are often the overriding consideration in selecting a location for a normal-acceleration sensor, the numerator dynamics are likely to be such that analog matching techniques must be used to identify $\omega_{n_{sp}}$ and ζ_{sp}.

Angle of attack is by far the best indicator of the pure short-period mode, and is therefore the response specifically mentioned in the requirement of 3.2.2.1. The reasons for this can be described using the following transfer function:

$$\frac{\Delta\alpha}{\delta_e} = \frac{K_\alpha\left(s + \frac{1}{T_\alpha}\right)\left(s^2 + 2\zeta_\alpha\omega_\alpha s + \omega_\alpha^2\right)}{\left(s^2 + 2\zeta_{sp}\omega_{n_{sp}} s + \omega_{n_{sp}}^2\right)\left(s^2 + 2\zeta_p\omega_{n_p} s + \omega_{n_p}^2\right)}$$

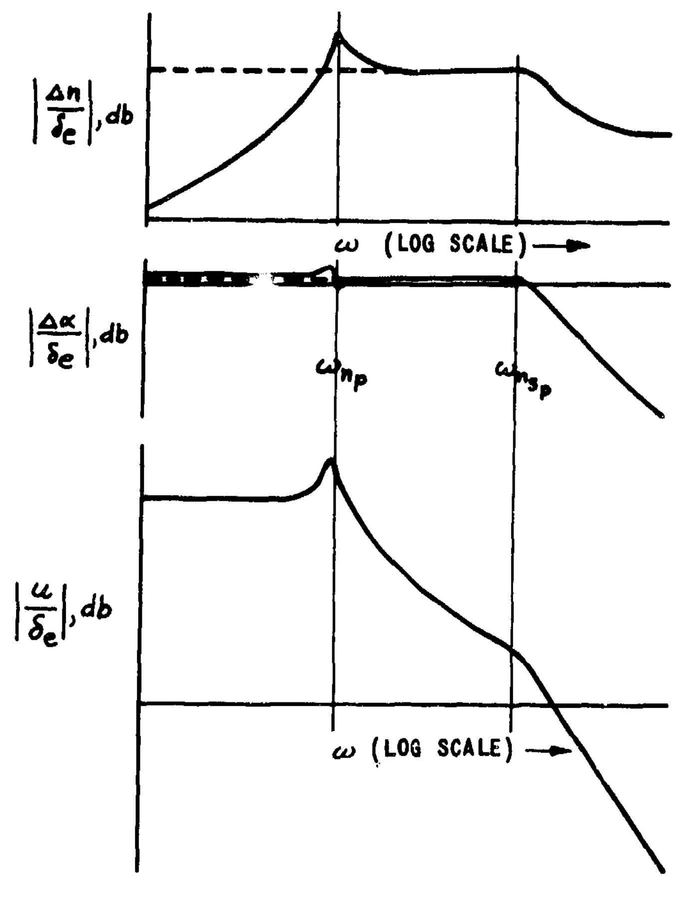
Generally speaking, $(1/T_\alpha)$ is so large that the $(s+\frac{1}{T_\alpha})$ is not important in the angle-of-attack response to control inputs. In addition, the second-order numerator term is nearly identical to the phugoid mode, under most circumstances. Therefore, the transfer function is quite accurately approximated by the following simple form:

$$\frac{\Delta\alpha}{\delta_e} = \frac{K_\alpha/T_\alpha}{\left(s^2 + 2\zeta_{sp}\omega_{n_{sp}} s + \omega_{n_{sp}}^2\right)}$$

In other words, even if the input used to excite the short period disturbs the phugoid mode, the amount of phugoid motion appearing in the angle-of-attack response will be small compared to the amount appearing in either pitch or normal-acceleration response. Therefore, the angle-of-attack response to control inputs most nearly represents the second-order short-period response in its purest form. If doublet control inputs are used to further minimize the phugoid excitation, the angle-of-attack response normally yields very good values of $\omega_{n_{sp}}$ and ζ_{sp}, even when simple hand techniques are used. There are some practical problems associated with the use of angle of attack, however. One problem is that it is difficult to find a location for the sensor where its indications are not influenced by the flow field around the airplane. If the flow field is well-behaved so that the variation of indicated angle of attack with true angle of attack is linear however, this problem is of little consequence. A more serious problem results if the sensor is not longitudinally located near the center of gravity. If the sensor is very far displaced from the center of gravity, especially fore or aft, it will sense angle of attack plus a significant contribution from pitch rate. This problem can be remedied by mounting the sensor high above the c.g. on a pylon (u variations have a smaller effect on α), or by cancelling the pitch-rate contribution with a signal from a pitch-rate sensor (requiring accurate calibration of the α sensor).

In summary, the best way to determine $\omega_{n_{sp}}$ and ζ_{sp} is normally to analyze the angle-of-attack response to a doublet control input. If the angle-of-attack response of a particular airplane is very difficult to measure, however, a suitable analog-matching technique can be used to obtain $\omega_{n_{sp}}$ and ζ_{sp} from the θ , $\dot{\theta}$, or n responses (see Appendix IVD).

The parameter n/α is defined (6.2.5) as the ratio of normal-acceleration change to angle-of-attack change with control deflection, in the steady state at constant speed. Actually, any control input will likely disturb speed as well as angle of attack and normal acceleration. Typical Bode plots of the three-degree-of-freedom n/δ_e, α/δ_e and u/δ_e transfer functions are sketched (Reference B107).



The dashed lines give the short-period approximation derived from the two-degree-of-freedom equations (with u suppressed) presented in the discussion of 3.2.2.1. What is desired is the n/α ratio of this two-degree-of-freedom approximation. It is seen that when the short-period and phugoid frequencies are well separated, as in this example, n/α can be determined by measuring $|n|$ and $|\alpha|$ while oscillating the elevator at a frequency in the range of flat response between ω_{n_p} and $\omega_{n_{sp}}$.

Measuring the steady-state responses to a step control input would lead to error because of speed changes. Generally some speed change also occurs with an impulse control input (see, for example, the reference cited); but an even more fundamental deficiency of impulse elevator inputs is that the n and α steady-state responses are zero.

Alternatively, n/α can be measured in the turns or pull-ups used to check maneuvering stability (Appendix IVE). These steady-state techniques can give n/α as a function of n or α to account for a nonlinear lift curve slope, etc.

IVD. ANALOG MATCHING TECHNIQUES FOR SHORT-PERIOD DATA REDUCTION

To begin with, there are two basic methods for analog matching airplane responses. One method involves working with the equations of motion, varying the coefficients until a match is obtained. The other method works with the transfer functions resulting from the equations of motion, varying the transfer function parameters until a match is obtained. The equations-of-motion approach is the more general of the two, since no assumptions need to be made concerning the linearity of the response. This is a small advantage, however, if a reasonable match cannot be obtained with linear equations, the response characteristics are probably nonlinear enough that the parameters ω_n and ζ no longer adequately describe the short-period response. The equations-of-motion technique has the disadvantage that there are a large number of unknown stability derivatives to vary, and the choice of a coefficient to produce a desired change in the time history is largely a trial-and-error process.

The transfer-function approach, on the other hand, automatically reduces the number of possible unknowns by combining the stability derivatives into lumped parameters. By making assumptions as to the form of the transfer-function poles and zeros, two other advantages result. First, the factored form of the transfer functions facilitates rational selection of the appropriate parameters to vary so as to effect a desired change in the time history. Second, the factored form permits the use of a priori knowledge to reduce the number of unknown parameters. For example, a lightly damped phugoid mode could be identified from an airspeed time history, using the simple hand-measurement techniques of Appendix C. The parameters ζ_p and ω_p could then be introduced into the θ/δ_e or $\dot{\theta}/\delta_e$ transfer function to determine ζ_{sp} and $\omega_{n_{sp}}$ by analog matching. For these reasons, only the factored-transfer-function approach will be discussed in this appendix.

Assuming that the techniques of Appendix IVC are used to effectively suppress the phugoid mode, a very simple analog-matching technique can be used to obtain ω_n and ζ from α , n , or θ responses. Set up a transfer function of the following form on the analog:

$$\frac{X}{F_s} \text{ or } \frac{X}{\delta_s} = \frac{K}{\left(\frac{s^2}{\omega_{n_{sp}}^2} + \frac{2\zeta_{sp}}{\omega_{n_{sp}}} s + 1\right)}$$

Picking a value for t_o on the time history (see Appendix III), apply a step input to the analog computer at t_o . Adjust K to match the steady state properly, and then adjust ζ_{sp} and $\omega_{n_{sp}}$ until the response shape is matched.

If the above simple method does not result in a good match, a more elaborate method must be used. Still using the assumption that the phugoid mode is effectively suppressed, the constant-speed form of the transfer functions can be used:

$$\frac{\Delta\alpha}{F_s} \text{ or } \frac{\Delta\alpha}{\delta_s} = \frac{K_\alpha (T_\alpha s + 1)}{\left(\frac{s^2}{\omega_{n_{sp}}^2} + \frac{2\zeta_{sp}}{\omega_{n_{sp}}} s + 1\right)} \qquad (T_\alpha \ll 1 \text{ usually})$$

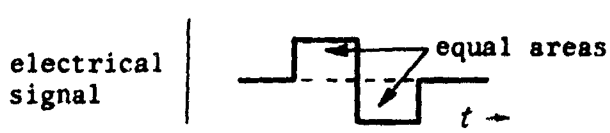
$$\frac{\Delta\theta}{F_s} \text{ or } \frac{\Delta\theta}{\delta_s} = \frac{K_\theta (T_{\theta_2} s + 1)}{s\left(\frac{s^2}{\omega_{n_{sp}}^2} + \frac{2\zeta_{sp}}{\omega_{n_{sp}}} s + 1\right)}$$

$$\frac{\Delta\dot{\theta}}{F_s} \text{ or } \frac{\Delta\dot{\theta}}{\delta_s} = \frac{K_\theta (T_{\theta_2} s + 1)}{\left(\frac{s^2}{\omega_{n_{sp}}^2} + \frac{2\zeta_{sp}}{\omega_{n_{sp}}} s + 1\right)}$$

$$\frac{\Delta n}{F_s} \text{ or } \frac{\Delta n}{\delta_s} = \frac{K_n (T_{n_1} s + 1)(-T_{n_2} s + 1)}{\left(\frac{s^2}{\omega_{n_{sp}}^2} + \frac{2\zeta_{sp}}{\omega_{n_{sp}}} s + 1\right)}$$

(The two real zeros sometimes become a pair of complex zeros, depending on the location of the n sensor and the point of control application.)

If the phugoid natural frequency is well separated from $\omega_{n_{sp}}$, as is often the case at high speeds, a step input can be used to excite the airplane and an ideal step or a truncated ramp used as the analog input. The entire response is then matched, starting from the time when input is initiated. This technique is discussed in more detail in Reference B100. An improvement to this method would be to feed an electrical doublet into the elevator or feel system servo.



This same electrical doublet can be used as the input for the analog. Notice that the exact form of the n transfer function zeros is dependent on the location of the n sensor. Also, if the longitudinal location of the α sensor is not at the center of gravity, the measured α response contains a contribution from $\dot{\theta}$.

$$\frac{\Delta\alpha_{VANE}}{\delta_s} = K_V\left[\frac{\Delta\alpha}{\delta_s} - \left(\frac{X}{V}\right)\frac{\dot{\theta}}{\delta_s}\right]$$

where X = distance of sensor ahead of c.g.

and K_V = $(\alpha\text{vane}/\alpha)$ for $\dot{\theta} = 0$

$$\frac{\Delta\alpha_{VANE}}{\delta_s} = K_V \frac{(K_\alpha T_\alpha s + K_\alpha) - \left(\frac{X}{V} K_\theta T_{\theta_2} s + \frac{X}{V} K_\theta\right)}{\left(\frac{s^2}{\omega_{n_{sp}}^2} + \frac{2\zeta_{sp}}{\omega_{n_{sp}}} s + 1\right)}$$

$$= K_V \frac{\left(K_\alpha T_\alpha - \frac{X}{V} K_\theta T_{\theta_2}\right)s + \left(K_\alpha - \frac{X}{V} K_\theta\right)}{\left(\frac{s^2}{\omega_{n_{sp}}^2} + \frac{2\zeta_{sp}}{\omega_{n_{sp}}} s + 1\right)}$$

$$\frac{\Delta\alpha_{VANE}}{\delta_s} = \frac{K'_\alpha\,(T'_\alpha s + 1)}{\left(\frac{s^2}{\omega_{n_{sp}}^2} + \frac{2\zeta_{sp}}{\omega_{n_{sp}}} s + 1\right)}$$

Since this transfer function has the same form as (α/δ_s), the analog setup will not be affected by the α-sensor location. However, the $(T'_\alpha s + 1)$ zero can become quite important in matching the time history if X is large; whereas it has a very small influence if $X = 0$.

Control system effects can often be included in the match by representing them with a pure time delay. If the elevator servo dynamics are fast enough, the effects of a slow feel system can sometimes be eliminated from the experimental time history by putting the input into the elevator servo electrically. If neither of these techniques result in a good match, the control-system poles and zeros must be included in the analog transfer functions.

Airplanes in the landing approach will sometimes have a phugoid frequency which approaches $\omega_{n_{sp}}$. When this is the case, it is very difficult to suppress the phugoid mode by proper shaping of the control input, and the constant-speed forms of the transfer functions can no longer be used. The complete transfer function forms must then be used. Ignoring sensor and recording system dynamics and gains, and control system dynamics, these transfer functions are of the following form:

$$\frac{\Delta\alpha}{F_s} \text{ or } \frac{\Delta\alpha}{\delta_s} = \frac{K_\alpha (T_\alpha s + 1)\left(\frac{s^2}{\omega_\alpha^2} + \frac{2\zeta_\alpha}{\omega_\alpha} s + 1\right)}{\left(\frac{s^2}{\omega_{n_{sp}}^2} + \frac{2\zeta_{sp}}{\omega_{n_{sp}}} s + 1\right)\left(\frac{s^2}{\omega_{n_p}^2} + \frac{2\zeta_p}{\omega_{n_p}} s + 1\right)}$$

$$\frac{\Delta\theta}{F_s} \text{ or } \frac{\Delta\theta}{\delta_s} = \frac{K_\theta (T_{\theta_1} s + 1)(T_{\theta_2} s + 1)}{\left(\frac{s^2}{\omega_{n_{sp}}^2} + \frac{2\zeta_{sp}}{\omega_{n_{sp}}} s + 1\right)\left(\frac{s^2}{\omega_{n_p}^2} + \frac{2\zeta_p}{\omega_{n_p}} s + 1\right)}$$

$$\frac{\Delta\dot{\theta}}{F_s} \text{ or } \frac{\Delta\dot{\theta}}{\delta_s} = \frac{K_\theta s (T_{\theta_1} s + 1)(T_{\theta_2} s + 1)}{\left(\frac{s^2}{\omega_{n_{sp}}^2} + \frac{2\zeta_{sp}}{\omega_{n_{sp}}} s + 1\right)\left(\frac{s^2}{\omega_{n_p}^2} + \frac{2\zeta_p}{\omega_{n_p}} s + 1\right)}$$

$$\frac{\Delta n}{F_s} \text{ or } \frac{\Delta n}{\delta_s} = \frac{K_n s (T_{n_1} s + 1)(T_{n_2} s + 1)(T_{n_3} s + 1)}{\left(\frac{s^2}{\omega_{n_{sp}}^2} + \frac{2\zeta_{sp}}{\omega_{n_{sp}}} s + 1\right)\left(\frac{s^2}{\omega_{n_p}^2} + \frac{2\zeta_p}{\omega_{n_p}} s + 1\right)}$$

In this situation, considerable time can be saved if ω_{n_p} and ζ_p are first identified from the airspeed response.

If both the phugoid and control system modes must be considered, the situation becomes even more complex. For this reason, it is worth considerable effort to obtain responses which are free of phugoid response and hopefully not badly contaminated by control system modes, even if an automatic input device must be used.

IVE. CONTROL FORCES AND MOTIONS IN PULLUPS AND STEADY TURNS

There are several methods for obtaining the control force and control motion data required for 3.2.2.2, 3.2.2.2.1, and 3.2.2.2.2. The best method to use depends primarily on the speed range under consideration. A major factor in determining the appropriate method for a given speed range is that control gradients with n are for constant speed, by definition. The method selected must therefore result in zero or small speed changes with n, or at least include a means for eliminating the effects of any speed changes. At speeds where characteristics vary significantly with Mach number, "speed" should be interpreted as "Mach No."

One method is to use a series of alternating symmetric pullups and pushovers, sequenced so as to minimize the airspeed and altitude changes. The control is held fixed after each input until the short-period motion becomes steady state, and measurements are taken from the steady-state values at a near-level attitude. An alternate version of this method is to stabilize the airplane holding various amounts of out-of-trim control force, and suddenly releasing the control.

Another method is to perform a series of stabilized turns after trimming the airplane in level flight. The load factor can be changed by changing the bank angle, and the airspeed held constant by using a different rate of descent for each load factor. The throttle and trimmer settings should be left at their trim settings throughout the maneuver to minimize the possibility of introducing extraneous pitching moments. The gradients obtained in this manner will not be quite as linear as with the symmetric pullup method; but, with the possible exception of a more stable slope near 1 g in the turns, the differences are generally small enough to be within the measurement errors.

A third method that is sometimes used involves a windup turn. After trimming in level flight, a turn of a certain number of g's is initiated, and the speed is allowed to decrease slowly as the g-level and altitude are held constant. The test is then repeated at several other g-levels until the complete range is covered. In this way, control gradient data can be obtained rapidly for several speeds. Again, the trimmer and throttle should be left at the trim settings and the rate of change of airspeed controlled by changing the rate of descent. The major disadvantage of this method is that it is less accurate because more careful pilot technique is required.

In general, the symmetric pullup method will work well at high speeds, but the airspeed changes will be excessive if the method is used at low speeds. On the other hand, the turn methods work well at low speeds, but can cause excessive altitude changes at high speeds. Also, it is impossible to obtain data for n less than 1.0 using turn methods. Notice that n/α can be obtained from these tests for use on Figures 1 and 2 of Reference A1. The $\Delta\alpha$ readings must be corrected for sensor position and local flow angularity (see Appendix IVC).

IVF. TRANSIENT CONTROL FORCES AND CONTROL SYSTEM DYNAMICS

To obtain data for 3.2.2.3.1, 3.5.3, and 3.5.3.1, it will be necessary for the pilot to pump the stick sinusoidally at various frequencies. Several techniques have been employed to aid the pilot in this task. One method is described in Reference D6, where the pilot visually follows an oscillating spot on the instrument panel. In other studies, oscillating aural tones have been fed to the pilot through earphones.

If the frequencies desired are not too high or too low, pilots can do an amazingly good job of moving the stick sinusoidally with no aids whatsoever. In addition, if the damping ratio is not too high, the pilot can find the resonant dip in the F_s/n versus frequency curve fairly accurately, by pumping at the frequency which gives the most airplane response for the least effort.

APPENDIX V

MEASUREMENT TECHNIQUES AND DISCUSSION OF SELECTED LATERAL-DIRECTIONAL PARAMETERS

VA. SIMPLIFIED ROLL RESPONSE TO SIDE GUSTS

The material here is extracted from Reference A5. Note that only side gusts are considered, and that these appear in the equations of motion in a simpler form than that required by paragraph 3.7 of Reference A1. Nevertheless, it is hoped that this exposition may aid in understanding some of the phenomena mentioned in paragraph 3.3.2.1, Lateral-directional response to atmospheric disturbances, and of the coupled roll-spiral mode. Response to turbulence has been found an especially important factor in pilots' ratings of high-$|\phi/\beta|_d$ configurations, which would be expected to respond more to side gusts than to roll gusts. In any case, 3.7 specifies that the side and roll gusts used in analysis should be uncorrelated.

In the flight program of Reference F5, there were frequent references in the pilot comment data to the aircraft response to turbulence or disturbances. The major complaint voiced was the large roll response for sideslip disturbances experienced for some configurations. In Reference F5, transient responses to disturbances were generated as indicated below to obtain a measure of the susceptibility of a configuration to turbulence.

The input disturbance used was equivalent to a gust along the aircraft y axis.

Equation A-1 from Appendix A of Reference F5 is shown below for the controls fixed, that is, no pilot input:

$$\begin{bmatrix} Y_\beta - s & -1 & \frac{g}{V_o} + \alpha_o s \\ N'_\beta + N'_{\dot\beta}\, s & N'_r - s & N'_p s \\ L'_\beta + L'_{\dot\beta}\, s & L'_r & (L'_p - s)s \end{bmatrix} \begin{bmatrix} \beta \\ r \\ \phi \end{bmatrix} = 0 \qquad (1)$$

The assumption that the air mass is nonaccelerating, that is, the air mass is a satisfactory inertial reference, is implicit in the equation. When the air mass is allowed to have motion along the aircraft y axis, this must be accounted for and the equation can be written as the following set:

$$\begin{aligned} Y_\beta \beta_A - s(\beta_A - \beta_G) \quad & - r \quad + \left(\frac{g}{V_o} + \alpha_o s\right)\phi = 0 \\ (N'_\beta + N'_{\dot\beta}\, s)\beta_A \quad & + (N'_r - s) r \quad + (N'_p s)\phi = 0 \\ (L'_\beta + L'_{\dot\beta}\, s)\beta_A \quad & + L'_r r \quad + s(L'_p - s)\phi = 0 \end{aligned} \qquad (2)$$

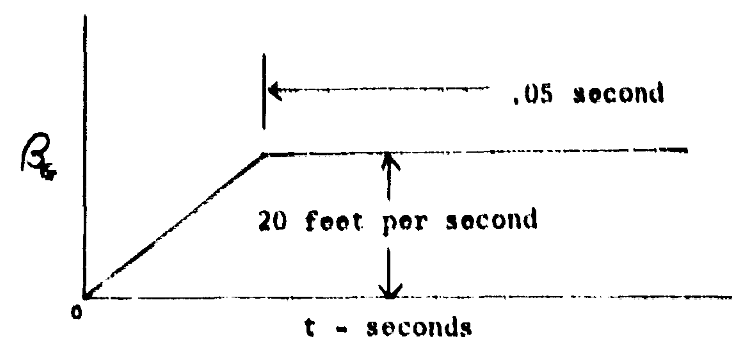
β_A - the aerodynamic sideslip angle or $\frac{1}{V_0}$ times the velocity of the aircraft with respect to the air mass along the y axis. (This is the sideslip angle displayed to the pilot.)

β_G - the sideslip gust or $\frac{1}{V_0}$ times the velocity of the air mass with respect to the earth along the negative y axis. (A positive β_G disturbance gives a positive β_A indication to the pilot.)

The set of equations can be replaced by the following equation where β_G appears as an input.

$$\begin{bmatrix} Y_\beta - s & -1 & \frac{g}{V_0} + \alpha_0 s \\ N'_\beta + N'_{\dot\beta} s & N'_r - s & N'_p s \\ L'_\beta + L'_{\dot\beta} s & L'_r & (L'_p - s)s \end{bmatrix} \begin{bmatrix} \beta_A \\ r \\ \phi \end{bmatrix} = \begin{bmatrix} -s \\ 0 \\ 0 \end{bmatrix} \beta_G \qquad (3)$$

This equation was solved on a digital computer for the side gust input shown below to generate the gust response presented in Reference F5. It should be noted that the sideslip angle β_A is the same angle that would be sensed by a sideslip vane for display to the pilot.



The transfer function for bank angle response to a β_G input, determined from Equation 3, is shown below.

$$\frac{\phi}{\beta_G} = \frac{1}{\Delta}(-s)\left[L'_{\dot\beta} s^2 + (L'_r N'_{\dot\beta} - L'_{\dot\beta} N'_r + L'_\beta)s + (L'_r N'_\beta - L'_\beta N'_r)\right]$$

$$\Delta = \left(s + \frac{1}{\tau_s}\right)\left(s + \frac{1}{\tau_R}\right)\left(s^2 + 2\zeta_d \omega_{nd} s + \omega_{nd}^2\right) \qquad (4)$$

The spiral-mode root, $1/\tau_s$, was essentially zero for Parts I and II configurations in the experiment described in Reference F5, which means that the term $(L'_r N'_\beta - L'_\beta N'_r)$ was also near zero. $L'_{\dot\beta}$ was also zero, and L'_β was large compared to $L'_r N'_{\dot\beta}$ for these configurations. For these conditions, the transfer function becomes:

$$\frac{\phi}{\beta_G} \approx \frac{-L'_\beta s}{\left(s + \frac{1}{\tau_R}\right)\left(s^2 + 2\zeta_d \omega_{nd} s + \omega_{nd}^2\right)} \qquad (5)$$

or

$$\frac{\phi}{\beta_G} \approx \tau_R \left(\frac{L'_\beta}{\omega_{n_d}^2} \right) \frac{s}{(\tau_R s + 1)\left(\frac{s^2}{\omega_{n_d}^2} + \frac{2\zeta_d}{\omega_{n_d}} s + 1 \right)} \qquad (6)$$

From the transfer function of Equation 5 it can be seen that the bank-angle response at high frequency is L'_β / s^2. With this observation and the values of L'_β, τ_R and ω_{n_d} for the various configurations of Parts I and II of Reference F5, sketches of the ϕ/β_G amplitude ratio can be made in log-log form.

From these sketches it is seen that the response at all frequencies is proportional to the value of L'_β. The response at low frequency is inversely related to the Dutch roll frequency, ω_{n_d}, and to the roll inverse time constant, $1/\tau_R$. The Dutch roll damping ratio determines the amplification of the Dutch roll frequency.

The bank-angle frequency response to a random sideslip disturbance would, of course, be dependent on the power spectral density of the random disturbance as well as the airplane transfer function. We will assume a high-bandwidth gust spectrum.

To minimize the response to sideslip disturbances, $|L'_\beta|$ should be small, ω_{n_d} should be high, and ζ_d and $1/\tau_R$ should be high. When the Dutch roll damping is low, the bank angle response at the Dutch roll frequency will be dominant; in this case, the magnitude of the bank angle to sideslip ratio in the Dutch roll mode, $|\phi/\beta|_d$, is a good indicator of the response.

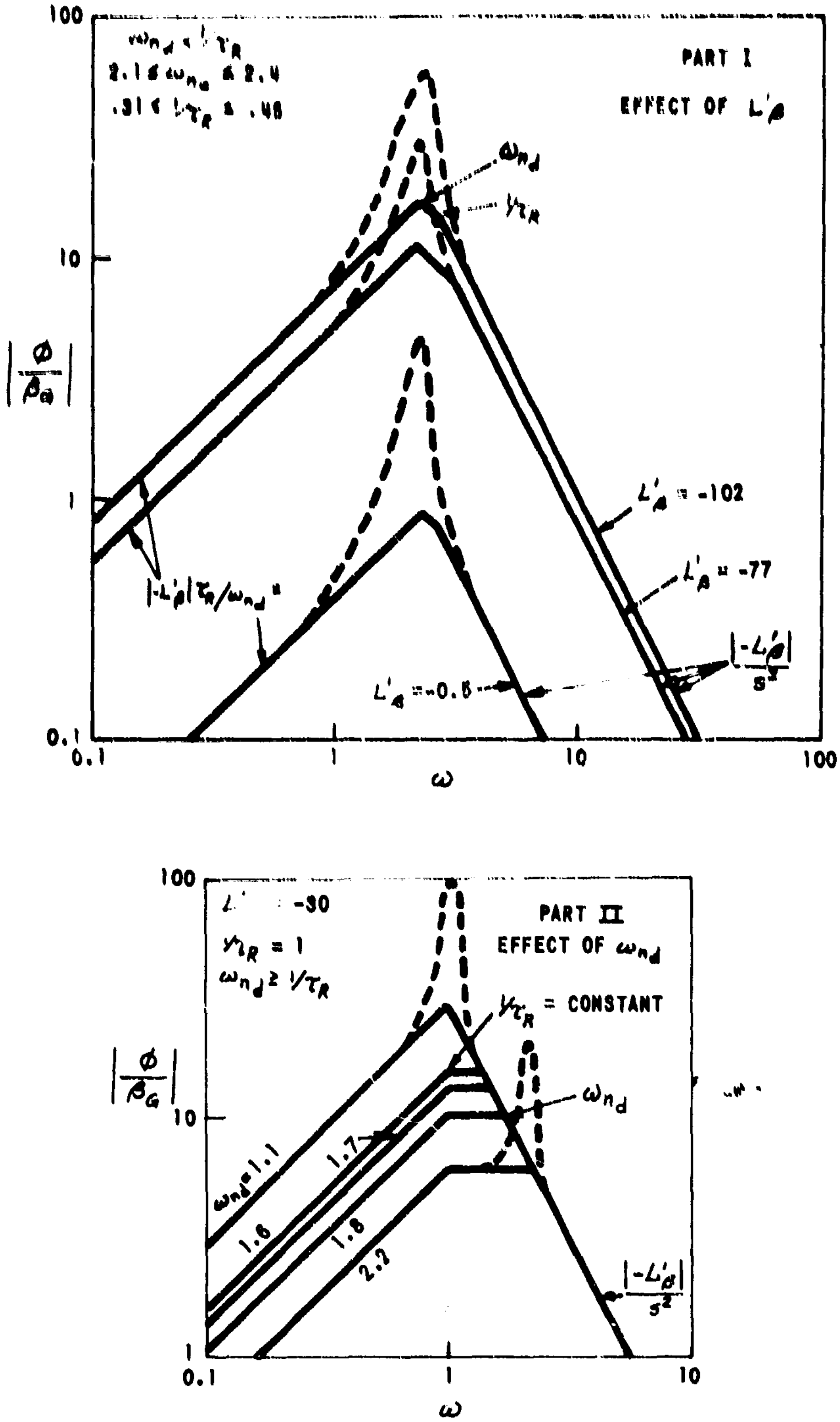
For a low-frequency roll-spiral mode, which was encountered with some of the Part III configurations in Reference F5, $L'_r N'_\beta - L'_\beta N'_r$ was still quite small. Neglecting the L'_β and $L'_r N'_\beta$ terms, the transfer function now becomes:

$$\frac{\phi}{\beta_G} \approx \frac{-L'_\beta s^2}{(s^2 + 2\zeta_{RS}\,\omega_{RS}\, s + \omega_{RS}^2)(s^2 + 2\zeta_d\, \omega_{n_d}\, s + \omega_{n_d}^2)} \qquad (7)$$

With the assumption that the roll-spiral mode is approximately cancelled by the s^2 term in the numerator, this reduces to:

$$\frac{\phi}{\beta_G} \approx \frac{-L'_\beta}{s^2 + 2\zeta_d\, \omega_{n_d} s + \omega_{n_d}^2} = \frac{-L'_\beta}{\omega_{n_d}^2} \; \frac{1}{\frac{s^2}{\omega_{n_d}^2} + \frac{2\zeta_d}{\omega_{n_d}} s + 1} \qquad (8)$$

When the roll-spiral frequency is not low enough to neglect the $(L'_r N'_\beta - L'_\beta N'_r)$ term, Equation 4 can be approximated as shown in Equation 9. The assumption is still made that L'_β and $L'_r N'_\beta$ can be neglected.



$|\phi/\beta_a|$ FREQUENCY-RESPONSE ASYMPTOTES, WITH EFFECT OF LIGHT DUTCH ROLL DAMPING INDICATED

$$\frac{\phi}{\beta_G} \approx \frac{-s(L'_\beta s + L'_r N'_\beta - L'_\beta N'_r)}{(s^2 + 2\zeta_{RS}\omega_{RS} s + \omega_{RS}^2)(s^2 + 2\zeta_d \omega_{n_d} s + \omega_{n_d}^2)} \qquad (9)$$

The constant term in the denominator can be expressed as follows: $\frac{g}{V_0}(N'_r L'_\beta - N'_\beta L'_r) = (\omega_{RS}\omega_{n_d})^2$. When this substitution is made in the numerator, Equation 9 becomes

$$\frac{\phi}{\beta_G} \approx \frac{-L'_\beta s(s + 1/\tau_\phi)}{(s^2 + 2\zeta_{RS}\omega_{RS} s + \omega_{RS}^2)(s^2 + 2\zeta_d \omega_{n_d} s + \omega_{n_d}^2)} \qquad (10)$$

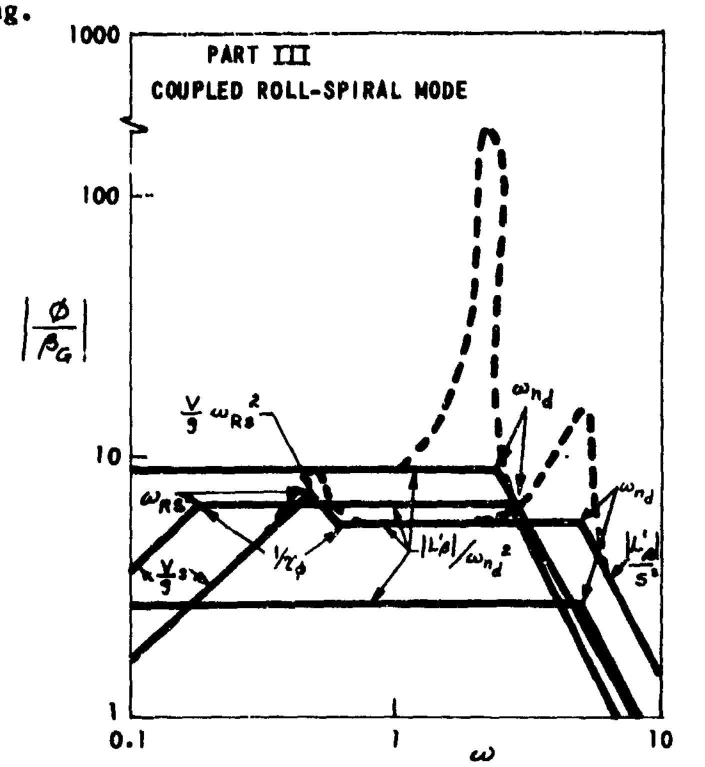
or

$$\approx \frac{V_0}{g} \frac{s(\tau_\phi s + 1)}{\left(\frac{s^2}{\omega_{RS}^2} + \frac{2\zeta_{RS}}{\omega_{RS}} s + 1\right)\left(\frac{s^2}{\omega_{n_d}^2} + \frac{2\zeta_d}{\omega_{n_d}} s + 1\right)} \qquad (11)$$

where

$$\tau_\phi = \frac{g}{V} \frac{L'_\beta}{\omega_{RS}^2 \omega_{n_d}^2}$$

The following sketch illustrates the bank angle response to sideslip disturbances for the configurations in Part III of Reference F5 which had very low roll damping.



From Equations 8 and 10 together with the above sketches, it is seen that the response at all frequencies is again proportional to the value of L'_β . Also the response at low frequency is inversely proportional to ω^2_{nd} and ω_{RS} . The Dutch roll damping ratio again determines the amplification at the Dutch roll frequency.

The same conclusions apply as for the Parts I and II configurations; that is, to minimize roll response to side gusts, $|L'_\beta|$ should be small, the Dutch roll frequency should be large and the Dutch roll and roll damping ratios should be large.

VB. MEASUREMENT OF LATERAL-DIRECTIONAL DYNAMIC PARAMETERS - TIME HISTORY ANALYSIS TECHNIQUES

1. INTRODUCTION

The roll rate response to a step aileron input is usually made up of three distinct modes: the roll mode, the spiral mode, and the Dutch roll mode. If linearity is assumed, the principle of superposition applies. Then any point on the roll rate trace at any given time must be the sum of these three modes at that time. Therefore, if the three modes can be identified on the roll rate trace, it is possible to extract the roll mode time constant, τ_R .

The $\frac{p}{\delta_a}$ response function for a step input can be written as

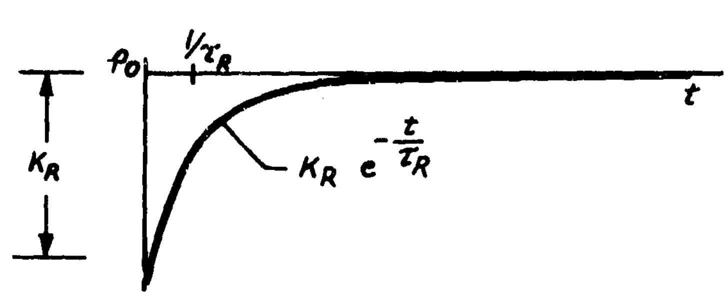
$$\left.\frac{p(s)}{\delta_a}\right|_{STEP} = \frac{K(s^2 + 2\zeta_\phi \omega_\phi s + \omega_\phi^2)}{(s + \frac{1}{\tau_s})(s + \frac{1}{\tau_R})(s^2 + 2\zeta_d \omega_{n_d} s + \omega_{n_d}^2)}$$

Transforming to the time domain, the roll rate time history following a step aileron input is given by:

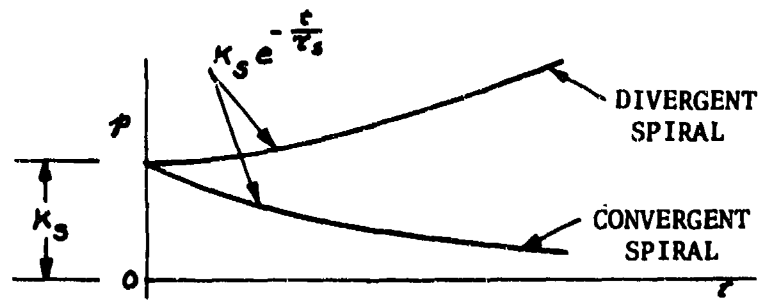
$$\left.\frac{p(t)}{\delta_a}\right|_{STEP} = K_S e^{-\frac{t}{\tau_s}} + K_R e^{-\frac{t}{\tau_R}} + K_d e^{-\zeta_d \omega_{n_d} t} \cos\left[\omega_{n_d}\sqrt{1-\zeta_d^2}\, t + \psi_p\right]$$

For a normal airplane, the characteristic modes take on the following forms following a step aileron input:

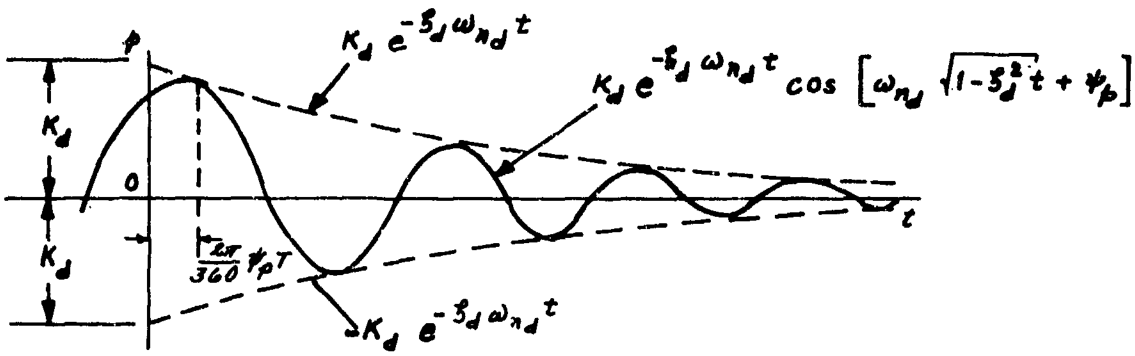
The roll mode, characterized by the first-order time constant, τ_R :



The spiral mode, characterized by the first-order time constant, which may be convergent or divergent:



The Dutch roll mode characterized by the second order parameters ζ_d and ω_{n_d} and the phase angle ψ_p. (The sketch below illustrates a negative phase angle.)



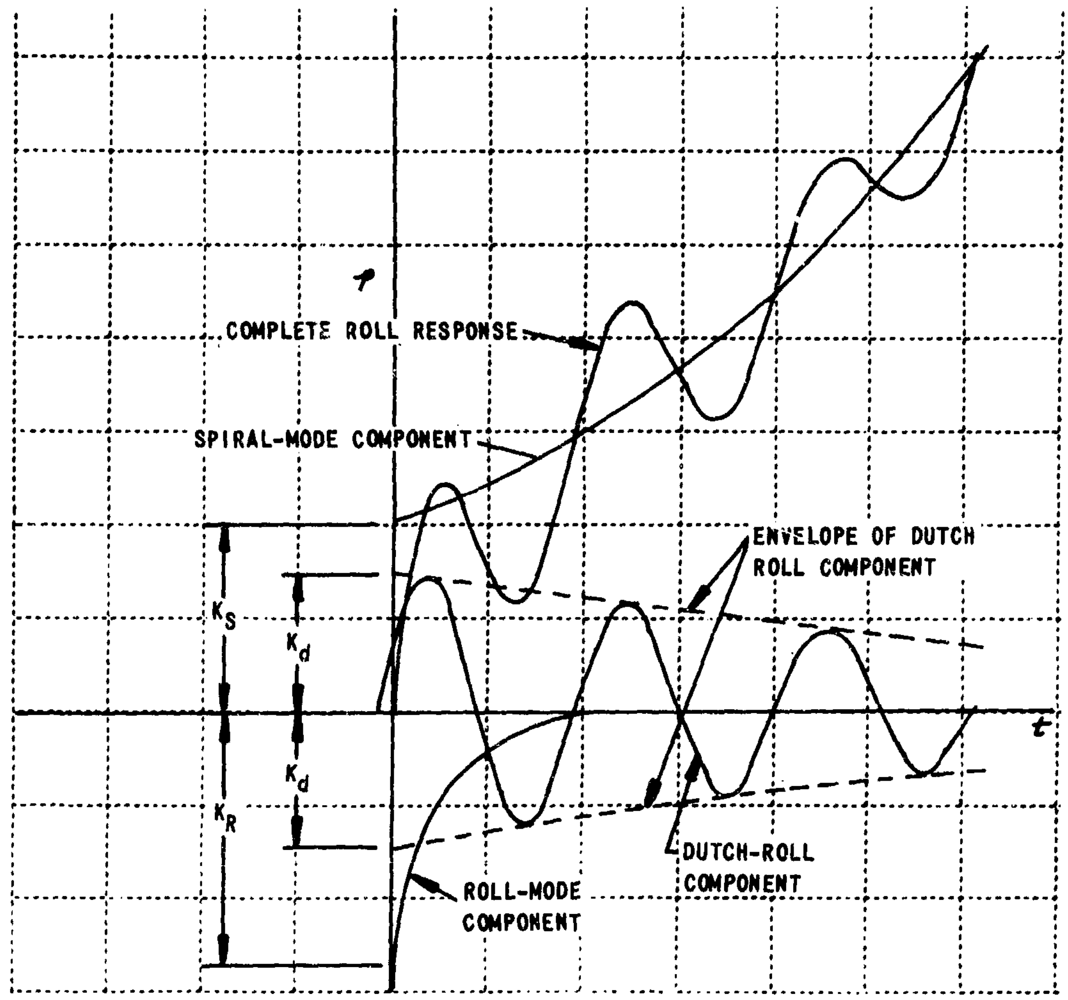
The individual responses are summed as sketched on the next page, to show the Dutch roll, roll, and spiral mode components of a roll rate time history following a step aileron input. Note, for this divergent-spiral case, that the composite peaks are displaced to the right, and valleys to the left, from those of the Dutch-roll-component trace.

The problem, however, is to analyze, not synthesize. Two methods of extracting the lateral-directional modes from a roll rate time history are graphical and analog matching. A graphical technique is presented in Section 2, and an analog-matching technique extracted from Reference B101 is presented in Section 3 of this appendix.

The roll-sideslip coupling requirements limit p_{osc}/p_{AV} and $\Delta\beta_{max}$ as functions of ψ_β, the sideslip-response parameter that corresponds to ψ_p in the roll response. Measurement of ψ_β may be difficult in two circumstances:

- When $\omega_\phi \approx \omega_{n_d}$ and $\zeta_\phi \approx \zeta_d$, there will be little Dutch-roll response to measure in the aileron-response time histories but then one would expect that the response would meet the requirements at the most critical value of ψ_β, so there should be no problem.

- When large-amplitude step commands are used, nonlinearities such as the variation of side force with $\cos\phi$ may obscure the phase relationships. Stability-augmentation nonlinearities can cause similar uncertainties. The requirements in large-amplitude rolls, 3.3.2.2 and 3.3.2.4, are intended to cover such situations.



2. GRAPHICAL TECHNIQUE FOR DETERMINING LATERAL-DIRECTIONAL MODAL PARAMETERS

In practice, this method assumes linearity, conventional modal characteristics, wide separation of the roll and spiral modes, light Dutch roll damping and an abrupt aileron input. However, good results have been obtained for conventional airplanes when the roll and spiral mode characteristics were in the ranges required by Reference A1 and the Dutch roll damping ratio was not greater than approximately 0.3.

Figure 1 shows the p trace resulting from a δ_a step for an aircraft with a short roll time constant, a lightly damped Dutch roll, and a highly divergent spiral mode. Under the assumption that the roll mode is fast-acting, its influence is small after the first few seconds, and the remainder of the response is essentially made up of only the spiral and the Dutch roll modes. (A first-order modal response reaches 95% of its final value in 3τ, and typically τ_R is of the order of 1 second.) So after the roll mode contribution is small:

$$\left.\frac{p(t)}{\delta_a}\right|_{STEP} = K_s e^{-\frac{t}{\tau_s}} + K_d e^{-\zeta_d \omega_{n_d} t} \cos\left[(\omega_{n_d} \sqrt{1-\zeta_d^2}\, t + \psi_p\right]$$

To obtain the spiral-mode contribution, the maximum and minimum points of the response from about two seconds and on are plotted on semilog paper as in Figure 2. Strictly speaking, the points of tangency of the Dutch roll envelope to the total response should be taken; however, using the actual peaks is well within the accuracy limits of the method. The first maximum and the first minimum are ignored because the roll mode is making a significant contribution during the initial part of the response. A smooth curve is drawn through the upper peaks and another one through the lower peaks. The upper curve is given by $K_s e^{-t/\tau_s} + K_d e^{-\zeta_d \omega_{n_d} t}$; the lower curve by $K_s e^{-t/\tau_s} - K_d e^{-\zeta_d \omega_{n_d} t}$. The curves define the envelope of the spiral and Dutch roll components. The numerical average between the two boundaries defines the spiral component, which must be a straight line on the semilog paper. The intersection at zero time is the spiral mode residue, K_s. The slope of this line may be checked if the spiral characteristics are known from another source: for example, another flight test; then, if necessary, the curves can be adjusted accordingly by eye. The time for the spiral to double amplitude, T_2 (for the divergent spiral), or the time to half amplitude, $T_{1/2}$ (for the convergent spiral), can be calculated and the corresponding slope plotted on the semilog paper for comparison with the previously determined slope. To determine the spiral-mode time constant from the plot, calculate

$$\tau_s = \frac{\Delta t}{\ln \frac{x_2}{x_1}}$$

The conversion between the time constant and the time to double or half amplitude is

$$T_2 \text{ or } T_{1/2} = .693\, \tau_s$$

Where there are a minimum number of usable peaks (two maximums and one minimum, or two minimums and one maximum) after the effect of the roll mode is small, the spiral component may still be extracted if the spiral characteristics are known. As before, the peaks are plotted on semilog paper and the best numerical average is selected. This value may be no more than one point of the spiral. Through this point or points, a line is drawn with a slope based on the known spiral characteristics. This line should define the spiral contribution to the roll response under investigation.

The next step is to plot the spiral component on linear graph paper and graphically subtract it from the roll rate response time history. The difference between the two curves will contain only the Dutch roll and roll mode components. All three curves (the original response, the spiral component and the difference between the two) are shown in Figure 3. Since we are only interested in the peaks of the Dutch roll component (as will be discussed in

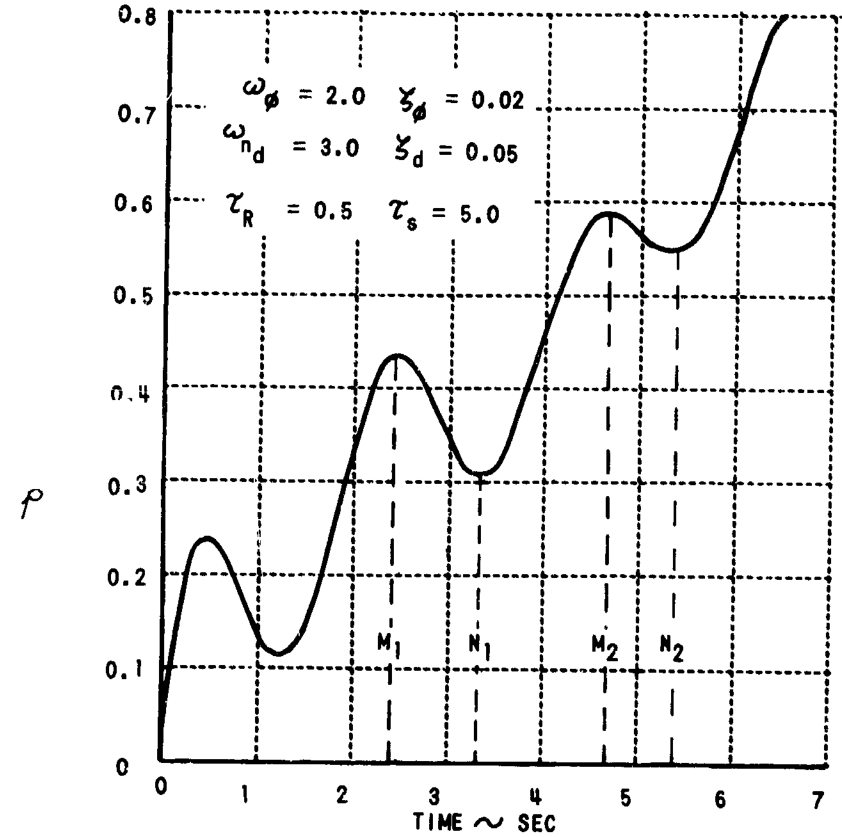


Figure 1 (APPENDIX VB)
ROLL TIME HISTORY

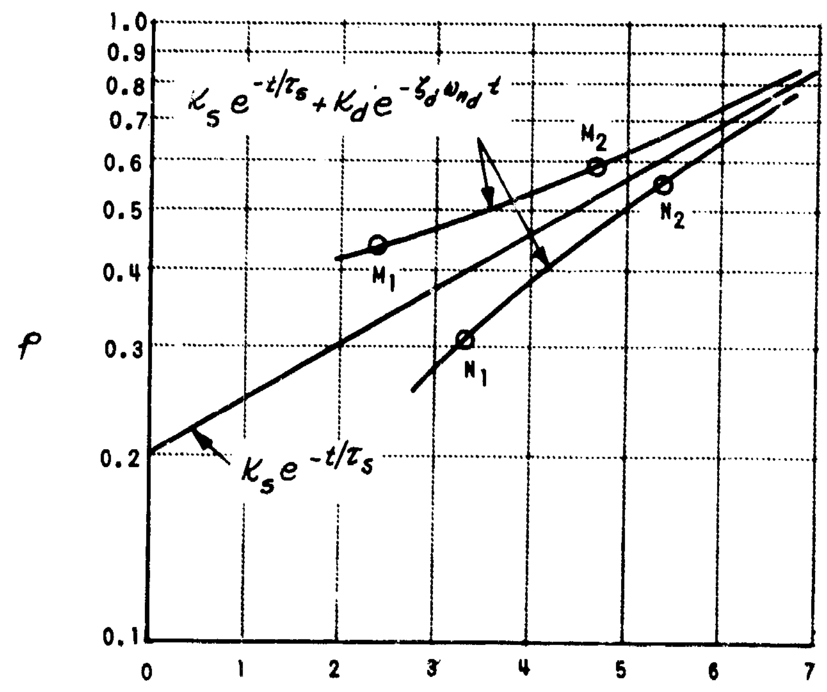


Figure 2 (APPENDIX VB)
SPIRAL-DUTCH ROLL ENVELOPE - SEMI-LOG PLOT

the following paragraph) and the initial roll response, it is only necessary to subtract the spiral component in the vicinity of the peaks and during the first second or so of the response. For completeness and clarity of presentation, however, the entire curve is shown.

To determine the Dutch roll envelope from the combined curve of Dutch roll and roll modes, the magnitudes of the later peaks (which occur after the time when the roll mode contribution is considered to be small) are plotted on semilog paper versus time. The data, as shown in Figure 4, will define an approximate Dutch roll envelope. If the Dutch roll damping ratio is known from another source, this step can be eliminated or the equivalent slope can be checked with the average slope obtained graphically. All this enables the best slope to be determined.

The approximate envelope is not precisely the Dutch roll envelope because the Dutch roll envelope does not touch the peaks, but rather touches the tangent points to the right of the peaks for the convergent oscillation. The displacement in degrees is equal to $\sin^{-1}\zeta$. This angle can be converted to time by the following equation:

$$\frac{T_d \sin^{-1}\zeta}{360}$$

where T_d is the period of the damped Dutch roll oscillation. For low damping ratios and short periods the displacement is small.

Next, the envelope is transferred from Figure 4 and drawn through the roll mode plus Dutch roll trace of Figure 3. This is shown with an enlarged scale in Figure 5. Since the roll mode contribution is insignificant after about two seconds for this example, the curve beyond this time defines the Dutch roll component alone. From this information, the period of the oscillation can be found and the initial part of the Dutch roll component can be drawn between the envelopes previously determined: locate the peaks and zero crossings of the undistorted Dutch roll oscillation on the time line and fair in the best damped sinusoid. Finally, the Dutch roll component for the first second or so is subtracted graphically from the Dutch roll plus roll mode trace, isolating the roll mode component as is shown in Figure 5.

An alternate method may be used to determine the initial Dutch roll component after the Dutch roll envelope has been obtained from Figure 4. This method is generally more accurate, since the determination of the initial part of the sinusoid is not dependent on the skill of fairing-in a damped oscillation. Instead, the magnitude of the Dutch roll component at any time is precisely determined from polar coordinate graph paper. First, a well-defined tangent point or zero crossing outside the influence of the roll mode is located. In the example, the tangent point at 2.45 seconds is selected. Since the damped Dutch roll period, T_d , is 2.10 seconds, the next tangent point is at .35 second, one period earlier. These tangent points, as determined from Figure 3 or 5, are plotted on the 360-degree radial in Figure 6. The phase angle, ψ_o , is calculated in this example by dividing the time at the first tangent point by the period and multiplying it by 360 degrees. A line drawn from the center along an angle of this magnitude to the left of the

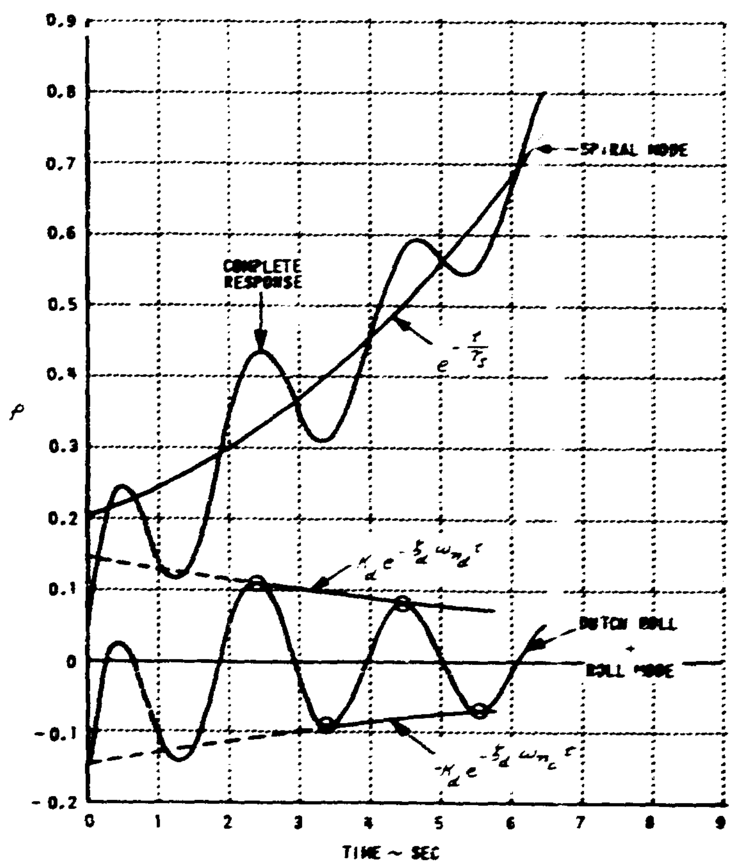


Figure 3 (APPENDIX VB
SEPARATION OF THE SPIRAL COMPONENT
FROM THE RESPONSE

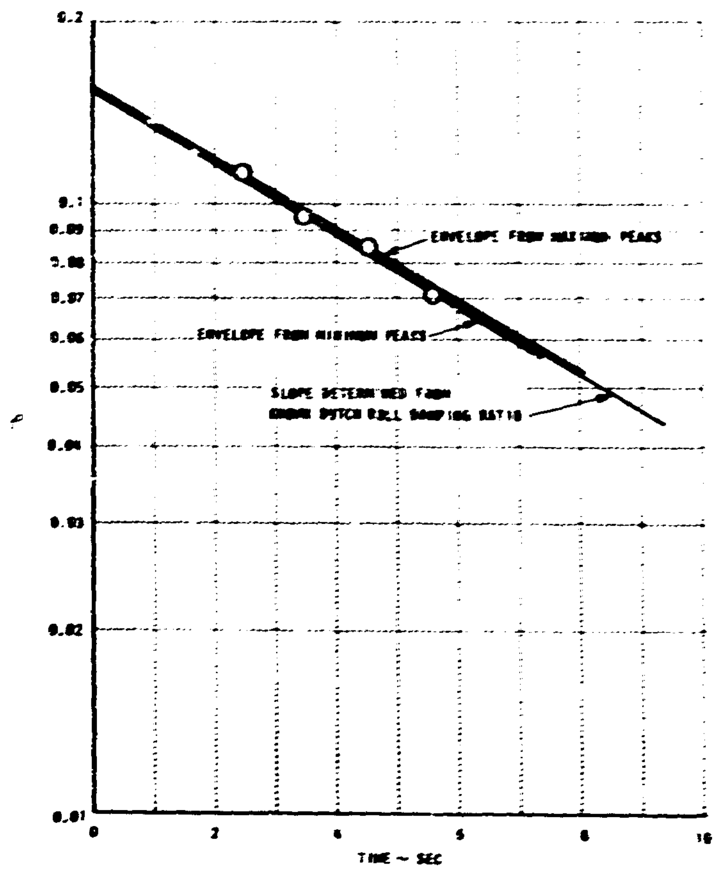


Figure 4 (APPENDIX VB)
DETERMINATION AND CHECK OF
DUTCH ROLL DAMPING RATIO

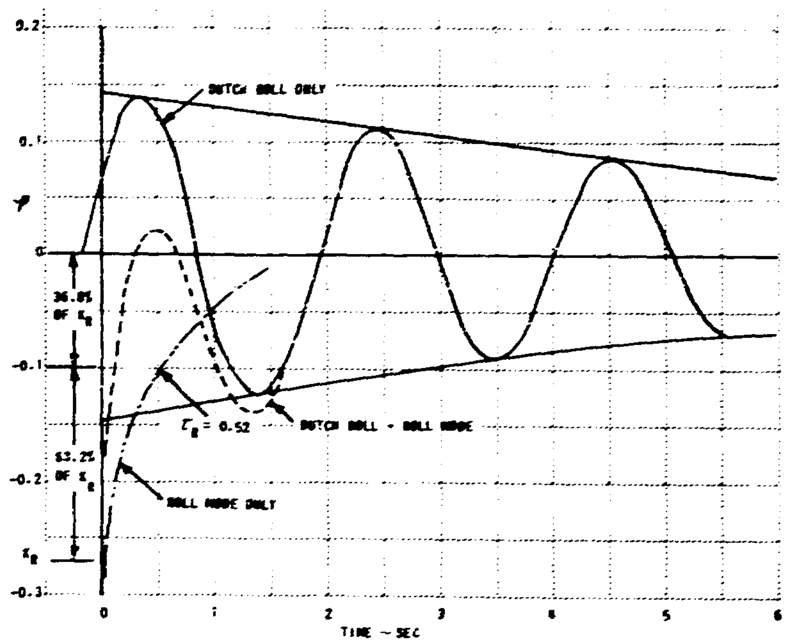


Figure 5 (APPENDIX VB)
ISOLATION OF ROLL COMPONENT

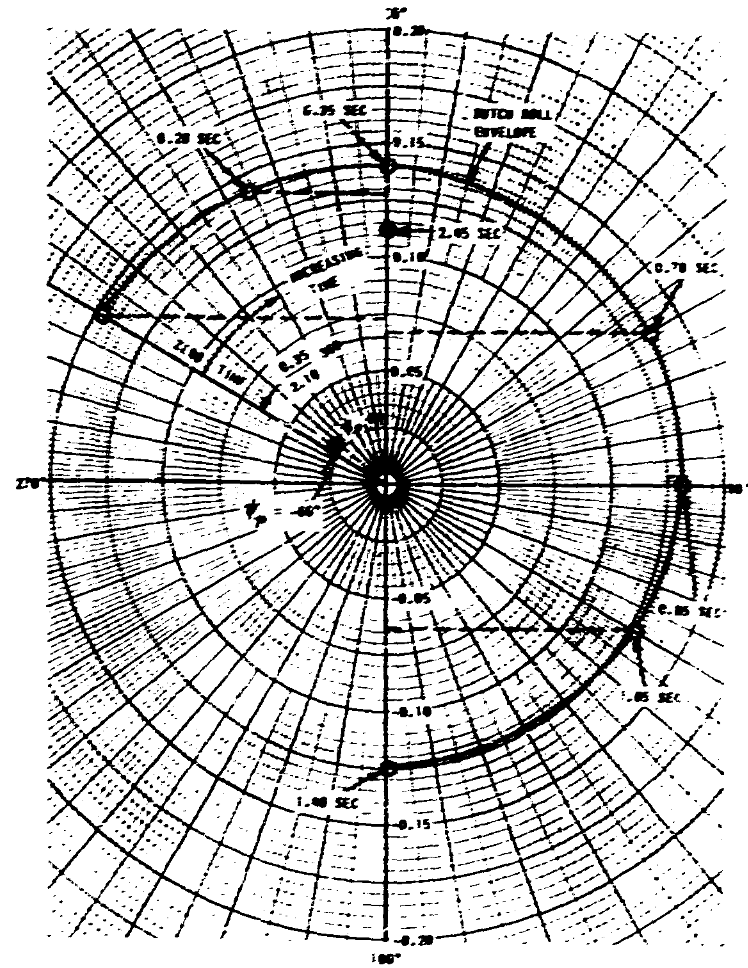
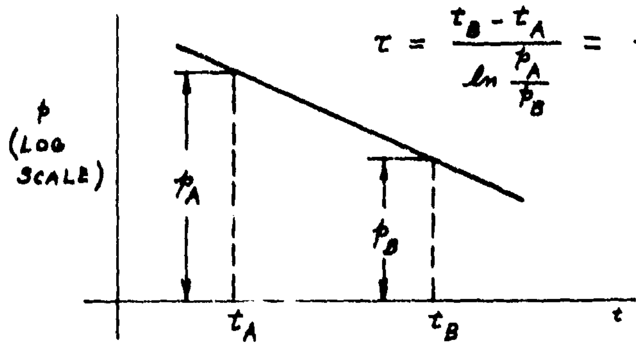


Figure 6 (APPENDIX VB)
RECONSTRUCTION OF INITIAL DUTCH ROLL MODE

360-degree line corresponds to zero time, as shown in Figure 6. Angles corresponding to fractions of a second can be located on the polar paper by converting them in a similar manner. The radial magnitude at any time is simply the magnitude of the Dutch roll envelope at that time. Points from the envelope in Figure 5 are plotted for every 30 or 40 degrees in Figure 6, and a curve is faired through the points to define the Dutch roll envelope on the polar paper. The Dutch roll contribution at any time is simply the vertical projection of the Dutch roll envelope at that time. To determine this from Figure 6, a horizontal line is drawn from the intersection of the time radial and the Dutch roll envelope. This projection on the vertical scale represents the magnitude of the Dutch roll at that instant. As before, when this value is subtracted from the Dutch-roll-plus-roll-mode curve (Figure 5), the roll-mode component is obtained. The result should plot as the "roll mode only" trace of Figure 5.

Since the roll mode time constant is the time for the roll response to settle to $e^{-1}\kappa_R$, $0.368\ \kappa_R$ is computed and a horizontal line is drawn at that value of ϕ in Figure 5. The intersection with the roll mode response gives the modal time constant. Another way to extract the time constant is to plot the roll mode values on semilog paper versus time, as in Figure 7. The result should be a straight line, and the intersection at zero time is κ_R. The time constant is the time corresponding to the intersection of a horizontal line drawn at $0.368\ \kappa_R$. Another way to obtain the time constant is to use the increment in time and the natural logarithm of the amplitude ratio as shown:

$$\tau = \frac{t_B - t_A}{\ln \frac{\phi_A}{\phi_B}} = \frac{t_B - t_A}{2.303 \log \frac{\phi_A}{\phi_B}}$$



Although a total of seven figures was used in the explanation of the graphical method, not all of these graphs are necessary to extract the roll mode. As a minimum, Figures 1, 2, 4, 6, and either 7 or the roll mode trace, alone, of Figure 5, are necessary to completely determine the time constant.

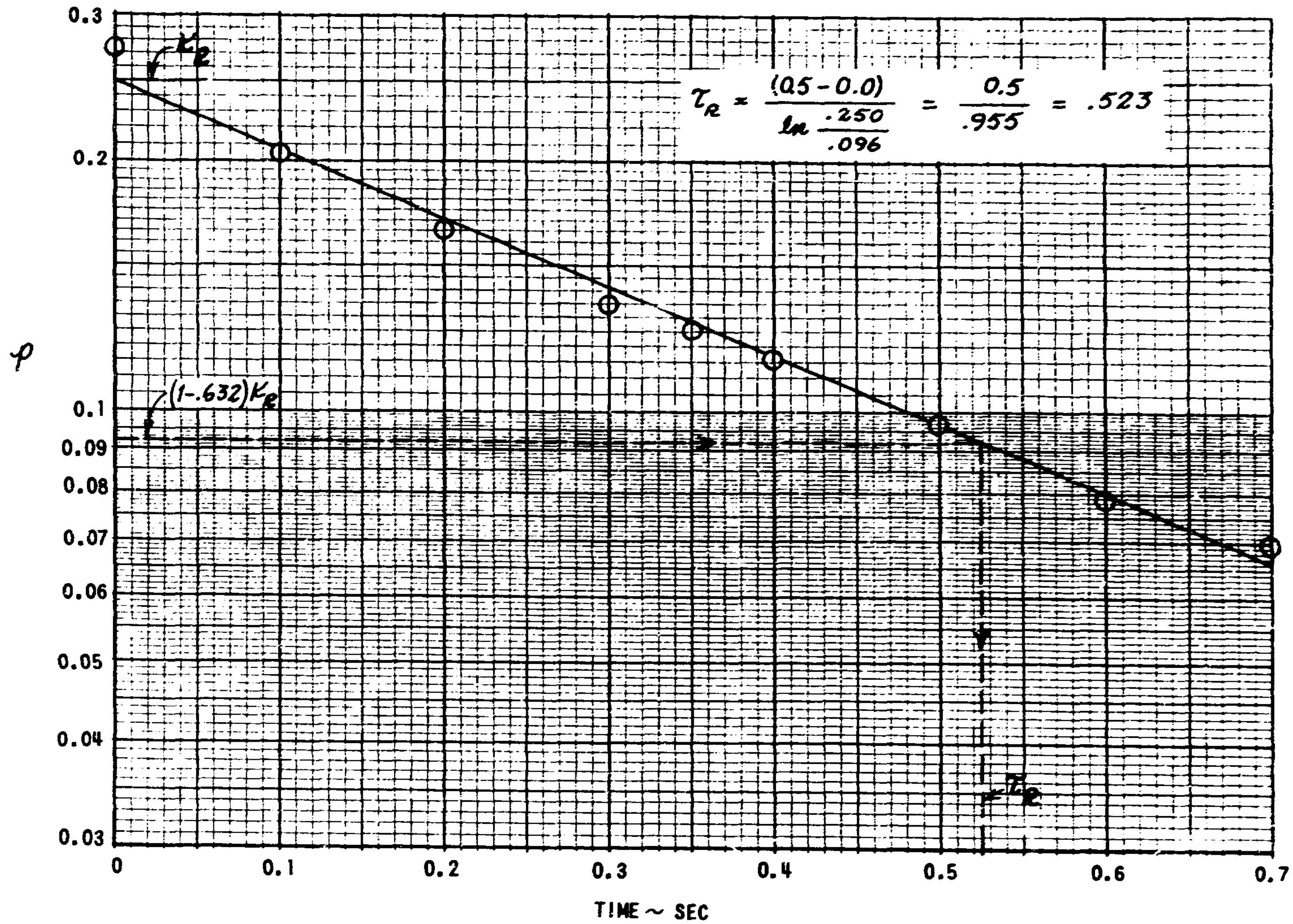


Figure 7 (APPENDIX VB)

3. ANALOG-MATCHING TECHNIQUE FOR DETERMINING LATERAL-DIRECTIONAL MODAL PARAMETERS

General

A method has been developed which allows the direct matching of flight test records with the output of an analog computer.

An analog computer program was developed that allows the matching of the $\frac{\beta}{\delta_a}$, $\frac{r}{\delta_a}$, $\frac{p}{\delta_a}$, or $\frac{\phi}{\delta_a}$ transfer-function factors by adjusting the analog time history to match a corresponding record obtained in flight.

The $\frac{\beta}{\delta_a}$ transfer function in terms of mode characteristics can be written as follows:

$$\frac{\beta}{\delta_a} = \frac{K_\beta\left(s+\frac{1}{\tau_\beta}\right)\left(s^2+2\zeta_\beta\omega_\beta s+\omega_\beta^2\right)}{\left(s+\frac{1}{\tau_s}\right)\left(s+\frac{1}{\tau_R}\right)\left(s^2+2\zeta_d\omega_{n_d}s+\omega_{n_d}^2\right)}$$

(The β/δ_a numerator may, instead, have the form $K_\beta\left(s+\frac{1}{\tau_{\beta_1}}\right)\left(s+\frac{1}{\tau_{\beta_2}}\right)\left(s+\frac{1}{\tau_{\beta_3}}\right)$, which can be written as $\left(s+1/\tau_{\beta_1}\right)\left[s^2+\left(\frac{1}{\tau_{\beta_2}}+\frac{1}{\tau_{\beta_3}}\right)s+\frac{1}{\tau_{\beta_2}}\frac{1}{\tau_{\beta_3}}\right]$.)

The $\frac{r}{\delta_a}$ transfer function can be written as:

$$\frac{r}{\delta_a} = \frac{K_r\left(s+\frac{1}{\tau_r}\right)\left(s^2+2\zeta_r\omega_r s+\omega_r^2\right)}{\left(s+\frac{1}{\tau_s}\right)\left(s+\frac{1}{\tau_R}\right)\left(s^2+2\zeta_d\omega_{n_d}s+\omega_{n_d}^2\right)}$$

The $\frac{p}{\delta_a}$ transfer function can be written as:

$$\frac{p}{\delta_a} = \frac{K_\phi s\left(s^2+2\zeta_\phi\omega_\phi s+\omega_\phi^2\right)}{\left(s+\frac{1}{\tau_s}\right)\left(s+\frac{1}{\tau_R}\right)\left(s^2+2\zeta_d\omega_{n_d}s+\omega_{n_d}^2\right)}$$

The $\frac{\phi}{\delta_a}$ transfer function can be written as:

$$\frac{\phi}{\delta_a} = \frac{K_\phi\left(s^2+2\zeta_\phi\omega_\phi s+\omega_\phi^2\right)}{\left(s+\frac{1}{\tau_s}\right)\left(s+\frac{1}{\tau_R}\right)\left(s^2+2\zeta\omega_{n_d}s+\omega_{n_d}^2\right)}$$

The only difference in form between the ϕ/δ_a and the β/δ_a transfer function is the term $\left(s+\frac{1}{\tau_\beta}\right)$ which has been replaced by 1 and, of course, the substitution of ζ_ϕ and ω_ϕ for ζ_β and ω_β.

The only difference in form between the $\frac{p}{\delta_a}$ and the $\frac{\beta}{\delta_a}$ transfer function is the term $\left(s+\frac{1}{\tau_\beta}\right)$ which has been replaced by s and, of course, the substitution of ζ_ϕ and ω_ϕ^β for ζ_β and ω_β.

Figure 8 is a general composite analog computer diagram from which each of the transfer functions can easily be set up. The diagram is presented without scale factors.

For the $\frac{\beta}{\delta_a}$ transfer function, use the diagram as shown where:

$$\tau_x = \tau_\beta, \; \zeta_x = \zeta_\beta, \; \omega_x = \omega_\beta$$

For the $\frac{r}{\delta_a}$ transfer function, use the diagram as shown where:

$$\tau_x = \tau_r, \; \zeta_x = \zeta_r, \; \omega_x = \omega_r$$

For the $\frac{p}{\delta_a}$ transfer function, set pot 3 or $\frac{1}{\tau_x} = 0$ and use:

$$\zeta_x = \zeta_\phi, \; \omega_x = \omega_\phi$$

For the $\frac{\phi}{\delta_a}$ transfer function, set pot 3 or $\frac{1}{\tau_x} = 1$ and remove line A where

$$\tau_x = 1, \; \zeta_x = \zeta_\phi, \; \omega_x = \omega_\phi$$

The value of K_x determines the scaling of the final trace and will have to be varied any time a change in denominator characteristics is made to keep the amplitude of the trace constant for a fixed δ_a input determined by K_i. Although a step input generally has been used, the form of δ_a can be made to match that of the flight record.

Analog Computer Matching Technique

The technique discussed here will be more directly applicable to the $\frac{p}{\delta_a}$ transfer function for a δ_a step input, since more experience has been gained in this particular area. However, the general technique should be applicable to the other three transfer functions.

The denominator characteristics ζ_d and ω_{n_d} can best be determined from another record, a rudder doublet record. These are relatively easy to measure and can be obtained with considerable accuracy. The damped Dutch roll frequency can be obtained by direct measurement, as described in Appendix III. The β trace is usually used since it is relatively free from the effects of the spiral mode and roll mode. Once ω_{n_d} and ζ_d have been obtained, the Dutch roll can be plotted in the s plane:

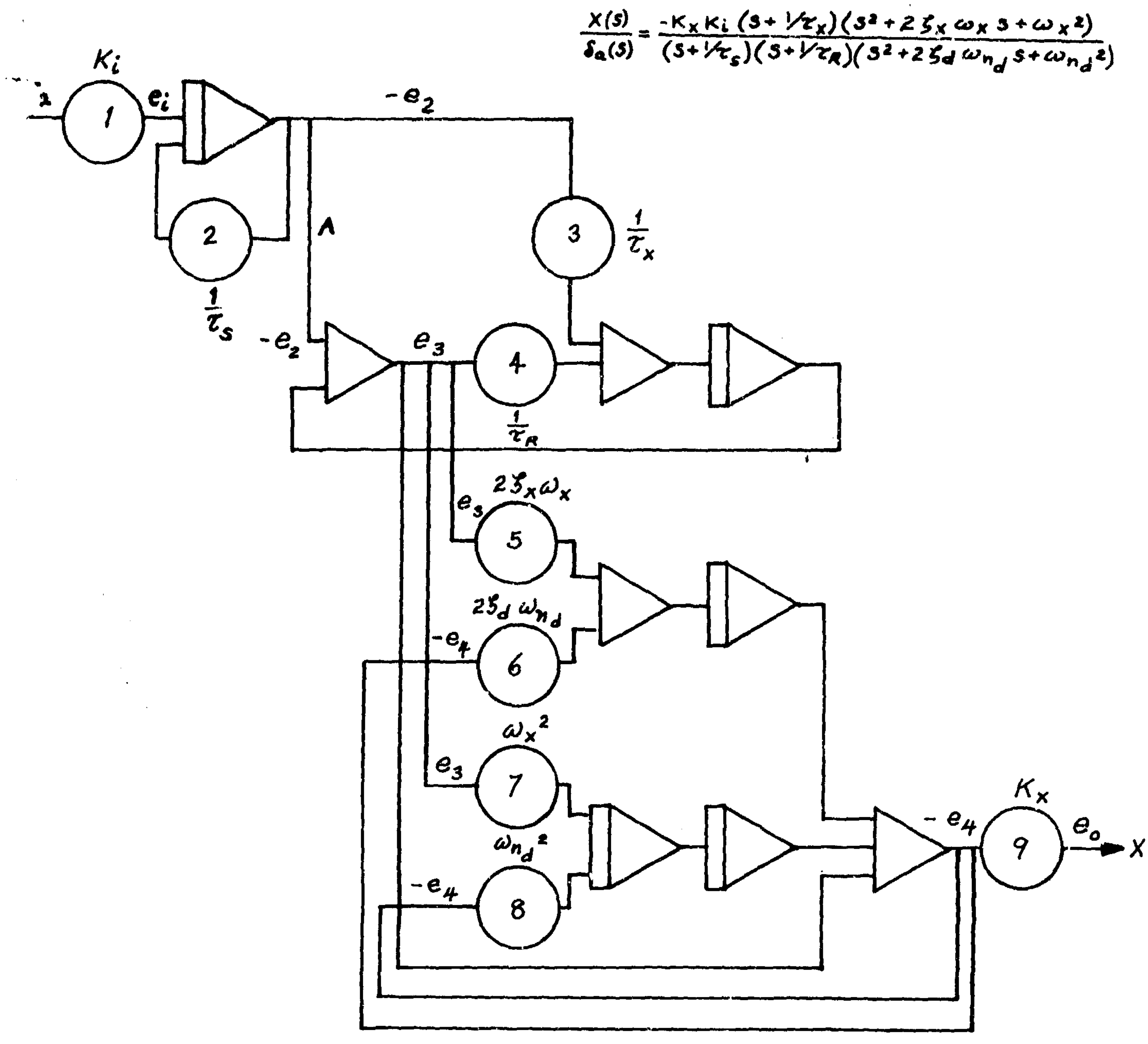
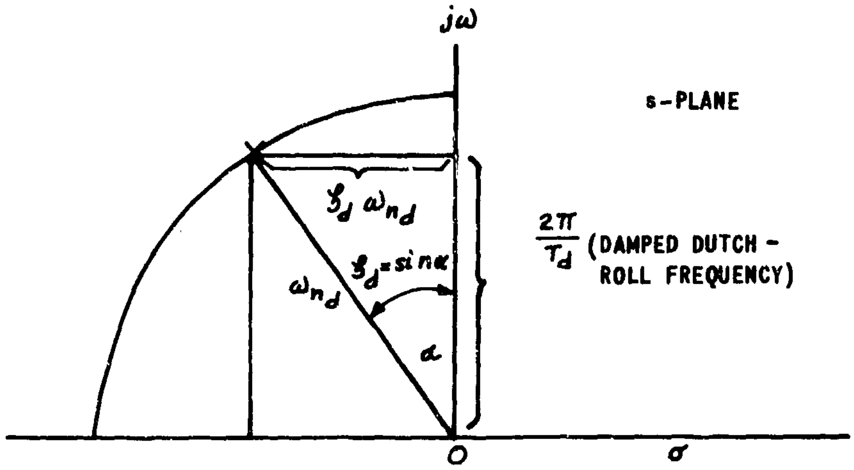
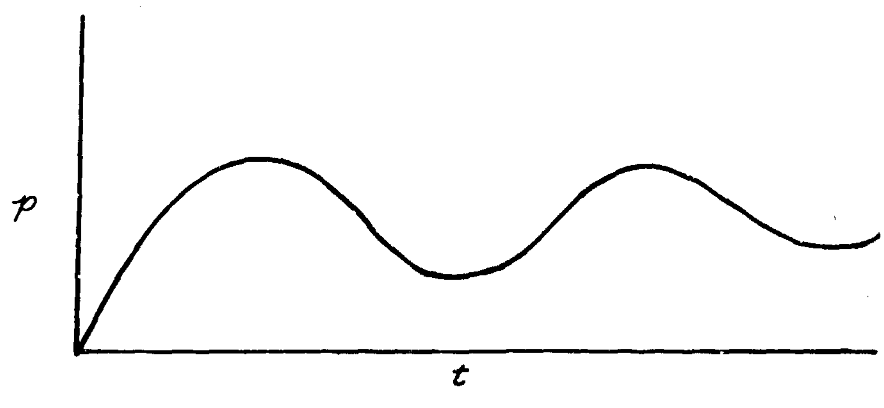


Figure 8 (APPENDIX VB)
GENERAL COMPOSITE ANALOG COMPUTER DIAGRAM

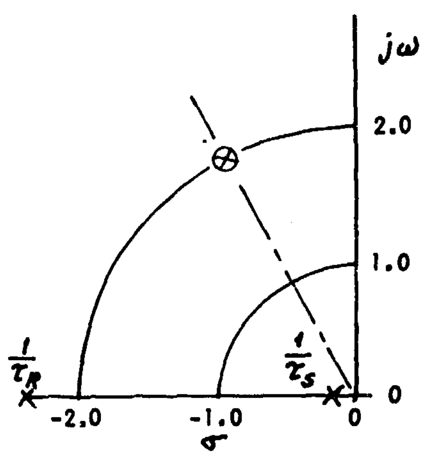


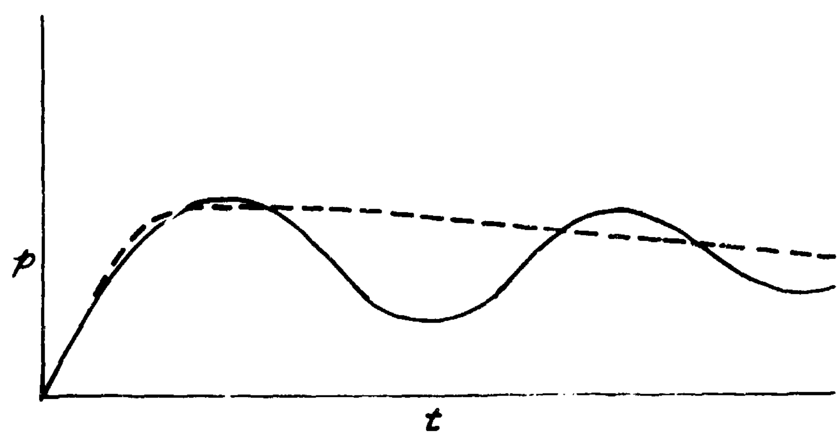
The sketch shows the relationships that exist in the s plane between ζ_d and ω_{n_d}. A line of constant radius about the origin is a line of constant ω_{n_d}, while a straight line drawn through the origin is a line of constant ζ_d. A quick determination of ζ_d can be made by drawing a curve of radius one about the origin, and reading the value of ζ_d directly as the horizontal displacement along the abscissa of the intersection of the line through the pole or zero and the curve of radius one.

In the analog-matching procedure, the response to be matched (example shown in the next sketch) is displayed on the analog computer output device, for example, an x-y plotter or an oscilloscope, with the x-coordinate driven proportional to time.



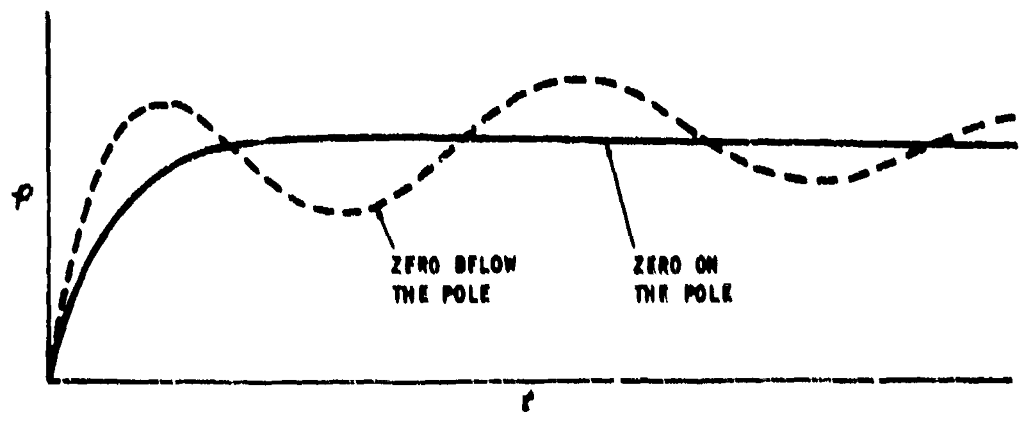
Next, set ω_{n_d} and ζ_d on the computer to the values of ω_{n_d} and ζ_d previously determined for the response. Also set $\omega_\phi = \omega_{n_d}$ and $\zeta_\phi = \zeta_d$. This allows a first guess at τ_R and possibly τ_S, depending on how much oscillation or Dutch roll appears on the trace to be matched. The resulting analog trace will not have any Dutch roll. The value of τ_R should be adjusted to closely approximate the initial slope of the curve to be matched. It may also be possible to get some indication of τ_S at this time from the general slope of the latter portion of the curve to be matched. With $\omega_\phi = \omega_{n_d}$ and $\zeta_\phi = \zeta_d$, the numerator zero will lie directly on the Dutch roll pole, and the dashed curve sketched will result when τ_R and τ_S are adjusted to their first approximations. The corresponding s-plane plot is shown alongside.



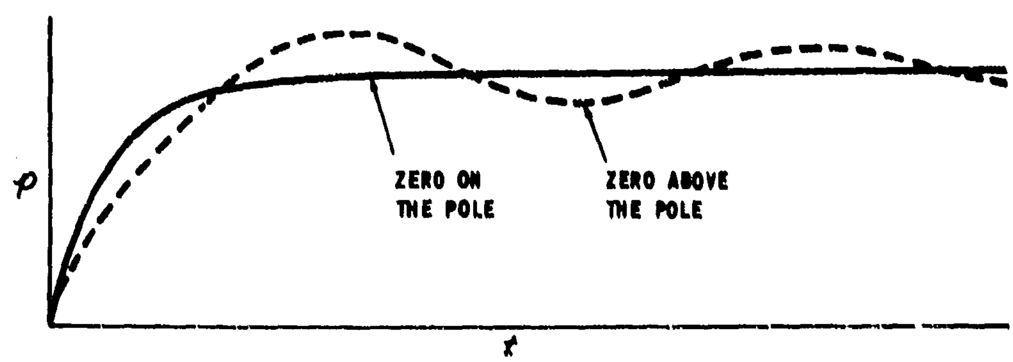


In the example shown, the spiral root is stable; however, this is not a necessary condition for a good match. It could just as well have been unstable; then the p trace would have had a slight upward trend. It is usually not necessary to put the roll-mode or spiral-mode poles on the s-plane plot, since they normally will not move very much during the matching procedure.

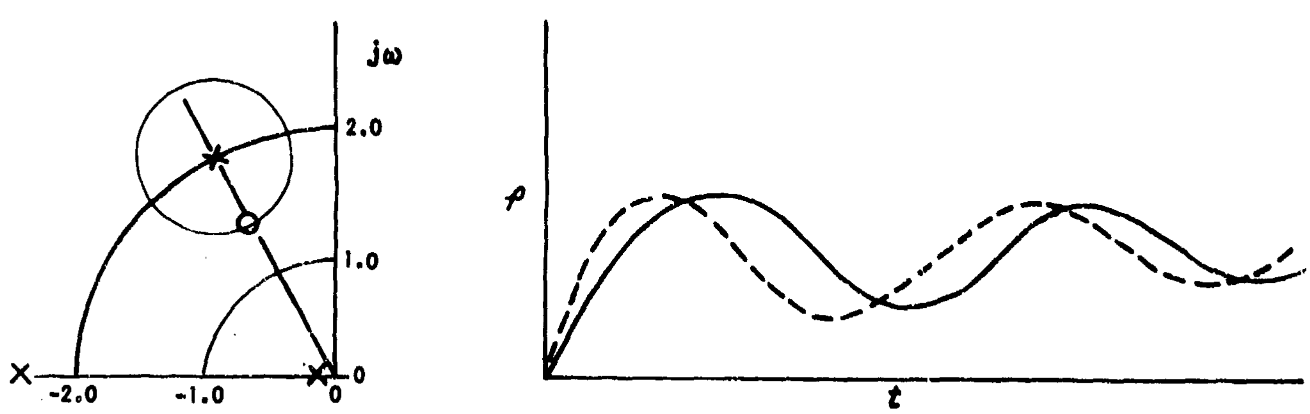
The next step is to displace the zero along the line of constant ζ toward or away from the origin. After a little experience, it will become obvious which displacement will give convergence. However, if the improper choice is made, the trace will be close to 180 degrees out of phase and this will be picked up in the next step. As a quick guide for the p trace: normally, for a zero below the pole, the time history will exhibit a character similar to the one shown immediately below, in which the Dutch roll oscillation will appear to shorten the roll mode time constant.



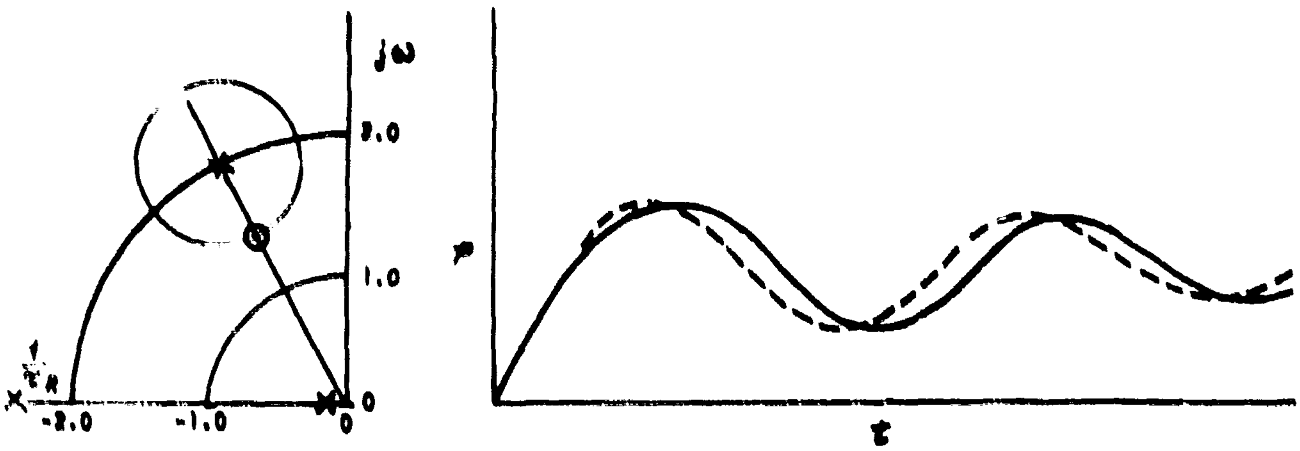
When the zero appears above the pole, the Dutch roll phase is shifted close to 180 degrees and will seem to lengthen the roll mode time constant. It will seem to have a delayed effect on the p trace, as shown next. These characteristics are shown in more detail in Figure 2 of the discussion of 3.3.2.2.



The amplitude of the Dutch roll motion is a function of the distance the numerator zero is displaced from the pole. To continue with the example, it will be assumed that the zero was displaced toward the origin and has reached the position where the amplitude of the Dutch roll closely approximates that of the trace to be matched. The result will be as shown next:



The curve that results will have the same amplitude of oscillation of Dutch roll, but it will probably not be in phase with the trace to be matched. The next step is to vary τ_R again to get the initial slope of the trace matched as closely as possible (little is to be gained by varying τ_s at this point):



What is now known is that the zero lies near a circle around the pole, with a radius that was determined by the displacement necessary to make the Dutch roll amplitudes equal.

The next step is to get the two traces in phase. This is accomplished as follows: Since the zero lies near the circle shown around the Dutch roll pole, and since the contributions to the phase angle of the slignt movements expected from the roll-mode and spiral-mode poles are small, the phase shift will be primarily a function of the rotation of the zero along the circle.

Two simple rules apply:

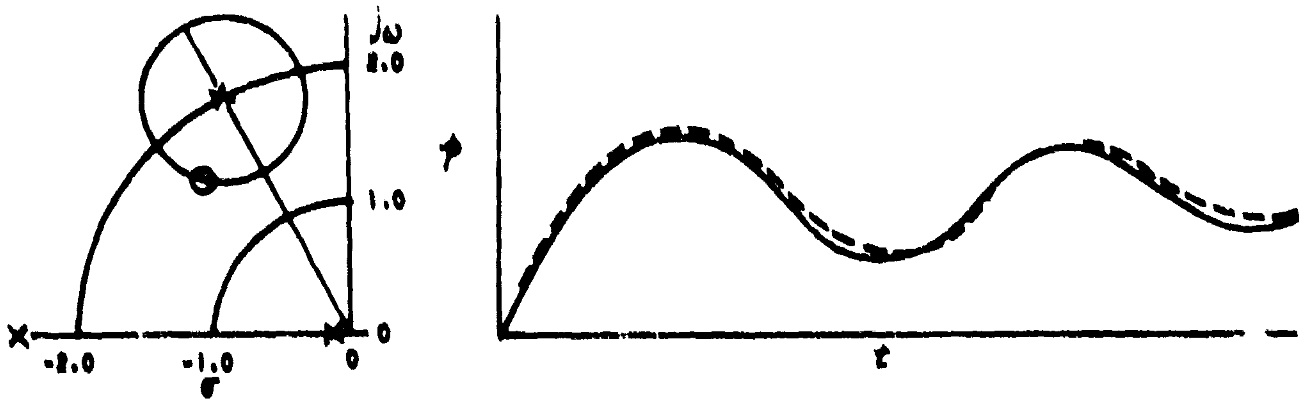
1. To move the analog trace to the <u>right</u> requires a <u>clockwise</u> rotation of the zero around the pole.

2. To move the analog trace to the <u>left</u> requires a <u>counterclockwise</u> rotation of the zero around the pole.

This characteristic can also be seen from Figure 2 of the discussion for 3.3.2.2.

The angle of rotation can be closely approximated by measuring the distance the trace has to be shifted in degrees. This is found by dividing the distance in time by the period of the trace and multiplying by 360 degrees.

This step should have the two traces pretty well matched. The example would now appear as shown below:



Small adjustments can now be made in τ_R and τ_S to get the analog solution to match the flight record as closely as possible.

Some general rules that can be followed are these:

1. Increasing τ_R (decreasing $\frac{1}{\tau_R}$) will cause the initial slope of the trace to become less steep and will move the phase slightly to the right.

2. For a stable spiral root, decreasing τ_S (increasing $\frac{1}{\tau_S}$) will lower the latter portion of the analog trace and have little effect on the phase.

3. For an unstable spiral root, decreasing τ_S (increasing $\frac{1}{\tau_S}$) will raise the latter portion of the analog trace and have little effect on the phase.

4. The converse of the above three rules is true.

<u>Planning</u>

A little advance planning can be a big help and can greatly increase the accuracy that can be obtained by the use of the analog computer matching technique.

In the example presented, it was assumed that τ_R was unknown and that the only trace available was the one to be matched. Although the accuracy here is good, movement of the roll-mode root and movement of the numerator zero both cause a change in the phase of the trace. Therefore, the match is somewhat of a compromise between the positions of these two factors. This is especially true when the numerator zero is located above and to the left of the pole. In this case, the magnitude of the Dutch roll is usually large and tends to cancel the effect of the roll mode, making the matching problem more difficult.

This problem can be avoided to a large extent if a series of records can be taken, one of which contains the desired configuration and another with the same denominator characteristics, but with the numerator characteristics varied until the zero is located as close as possible to the Dutch roll pole. As indicated, this will give minimum Dutch roll excitation and allow an accurate determination of both the roll mode pole and the spiral mode pole. Combination aileron and rudder inputs, or perhaps just rudder, might be used.

These denominator values should then be fixed, and an effort made to match the desired trace varying only the position of the numerator zero.

If possible, the series of records obtained in flight should include a rudder doublet, the desired trace and a trace with minimum Dutch roll excitation. This allows an accurate determination of both the denominator and numerator characteristics.

A scaled analog computer diagram is presented that will be useful within the following limits (see Figure 9):

$$\tau_s \approx 1.0$$

$$\tau_R \approx 0.1$$

$$\omega_{n_d} \approx \sqrt{10}$$

$$2\zeta_d \omega_{n_d} \approx 1$$

$$\tau_x \approx 0.1$$

$$\omega_x \approx \sqrt{10}$$

$$2\zeta_x \omega_x \approx 1$$

With one additional integrator and a pot, the x scale of the x-y plotter can be changed to compensate for changes that may occur in the time scale of the oscillograph records to be matched.

Theory

For a step input, the Laplace transform of $p(t)/\delta_a$ is obtained by multiplying the $p(s)/\delta_a(s)$ transfer function,

$$\frac{p(s)}{\delta_a(s)} = \frac{K_\phi s(s^2 + 2\zeta_\phi \omega_\phi s + \omega_\phi^2)}{(s-\lambda_s)(s-\lambda_R)(s^2 + 2\zeta_d \omega_{n_d} s + \omega_{n_d}^2)}$$

by the Laplace transform of a unit step, $1/s$.

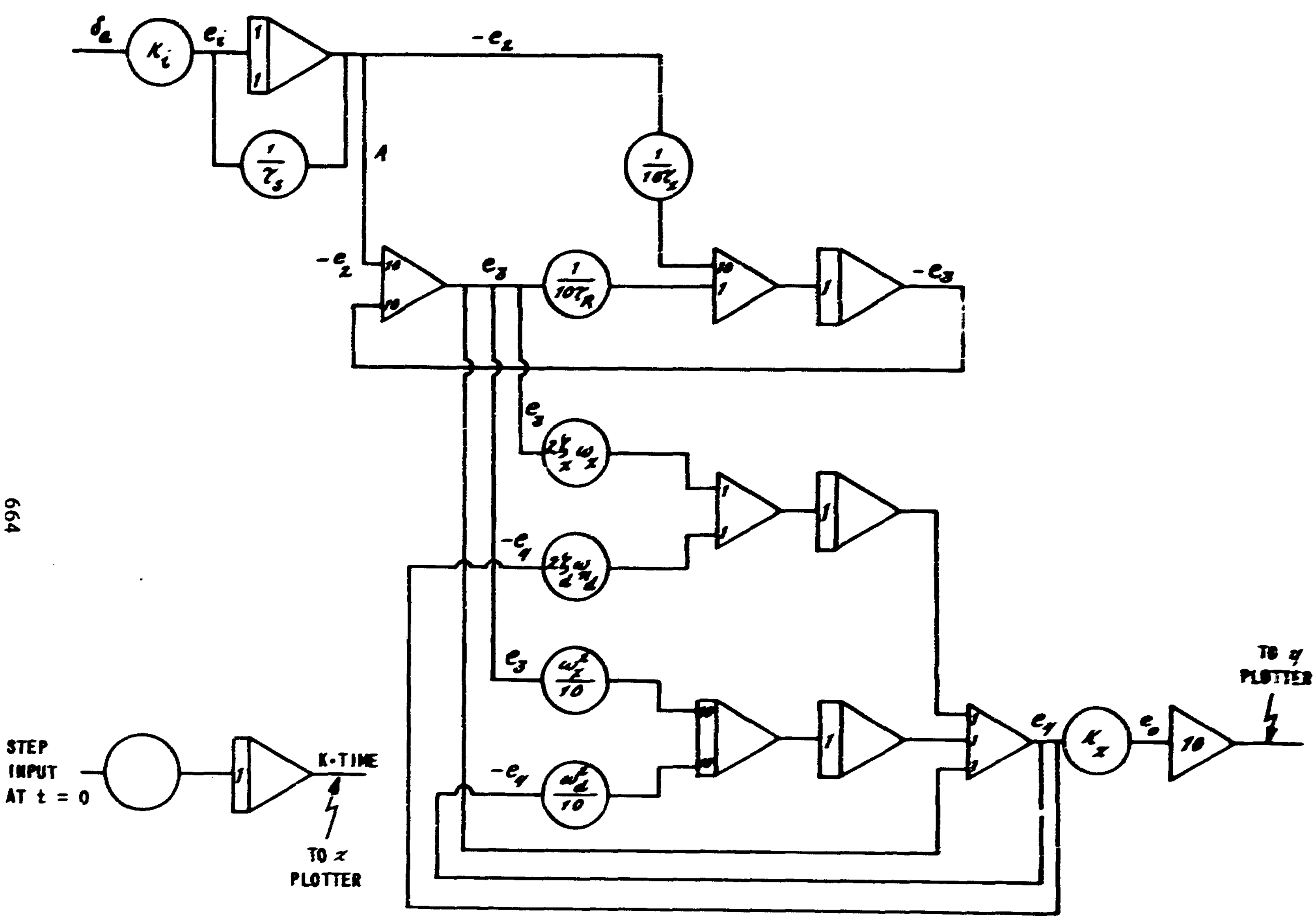
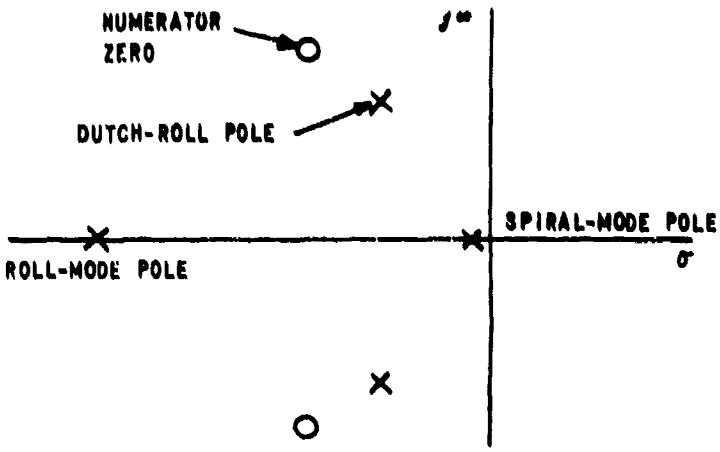


Figure 9 (APPENDIX VC) SCALED ANALOG COMPUTER DIAGRAM

The expression for the $\left.\frac{p}{\delta_a}\right|_{STEP}$ time history can be written as:

$$\frac{p}{\delta_a} = K_s e^{\lambda_s t} + K_R e^{\lambda_R t} + K_d e^{-\zeta_d \omega_{n_d} t} \cos(\omega_{n_d}\sqrt{1-\zeta_d^2}\, t + \psi_p)$$

A pole-zero plot for a normal airplane would be:



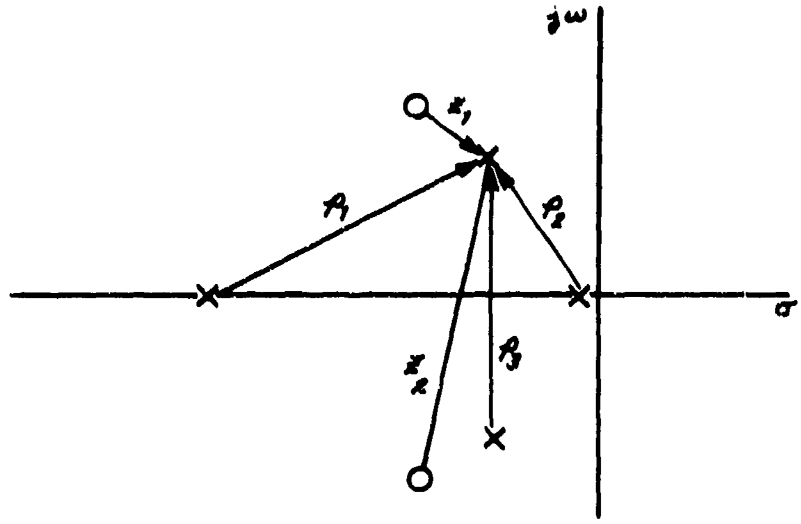
Of interest here are the magnitude and phase of the Dutch roll mode. As shown in Reference G10, the magnitude of the constant K associated with any sinusoidal component is twice the product of the lengths of the numerator vectors divided by the lengths of the denominator vectors drawn to a root of that factor:

$$|K_o| = \frac{2\,|z_1|\,|z_2|}{|p_1|\,|p_2|\,|p_3|}$$

The associated phase angle for a cosine representation, as above, is the sum of the angles of all the numerator vectors minus the sum of the angles of all the denominator vectors drawn to that pole:

$$\measuredangle K_o = \psi_{z_1} + \psi_{z_2} - \psi_{p_1} - \psi_{p_2} - \psi_{p_3}$$

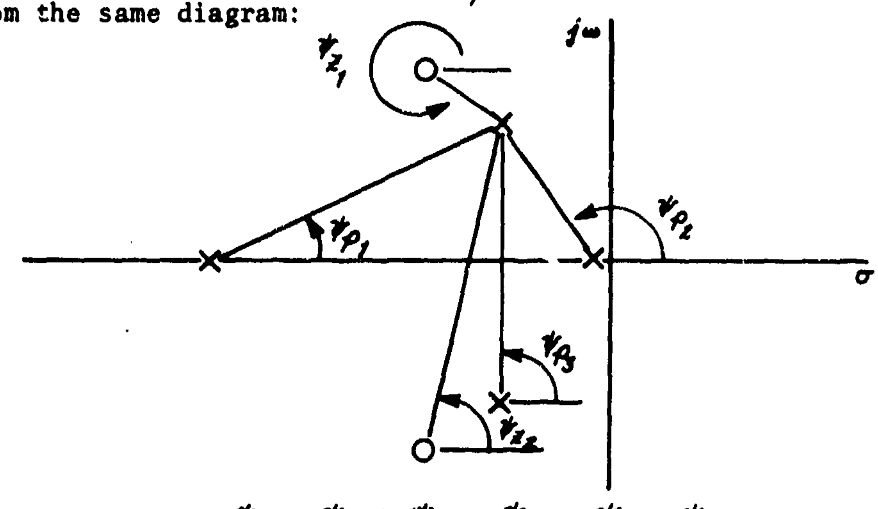
FOR THE DUTCH-ROLL POLE AT $s = -\zeta_d \omega_{n_d} + j\,\omega_{n_d}\sqrt{1-\zeta_d^2}$,



$$K_d = \frac{2|z_1||z_2|}{|p_1||p_2||p_3|} = \frac{|z_1||z_2|}{|p_1||p_2|\,\omega_{n_d}\sqrt{1-\zeta_d^2}} = \frac{|z_1||z_2|}{p_1\,\omega_{n_d}^2\sqrt{1-\zeta_d^2}}$$

From this expression and illustration it can be seen that the magnitude of the Dutch roll residue (K_d) is primarily a function of the separation of the numerator zeros from the denominator Dutch roll roots. Also, once the zero is separated from the pole, a small change in the roll-mode root location does have a small effect on the magnitude of K_d. Generally the spiral-mode pole is negligibly distant from the origin.

The phase angle in roll rate, ψ_p, of the Dutch roll pole can be calculated from the same diagram:



$$\psi_p = \psi_{z_1} + \psi_{z_2} - \psi_{p_1} - \psi_{p_1} - \psi_{p_3}$$

From the preceding diagram, it is easily seen that the major contribution to the phase angle will come from the rotation of the zero around the pole. It can also be seen that a movement of the roll-mode pole will cause some change in the phase angle.

VC. SIGNIFICANCE OF THE PHASE ANGLES ψ_β AND $\measuredangle p/\beta$ AND THE RATIO $|\phi/\beta|_d$

General Considerations

The discussions of roll-sideslip coupling (3.3.2 and subsidiary paragraphs) make extensive use of the phase angles ψ_1, ψ_p, ψ_β and $\measuredangle p/\beta$. First, the tolerable amount of coupling was found to be a function of ψ_1, the s-plane phase angle between the numerator and denominator oscillatory roots of the ϕ/δ_a transfer function. Next ψ_p (the phase of the roll-rate time response to a step aileron input) was investigated as a measure of ψ_1. But it was found that ψ_β (the phase of the sideslip time response to a step aileron input) was a better measure of ψ_1. Now, ψ_β is related to ψ_p through $\measuredangle p/\beta$. Reference A1 also uses $\measuredangle p/\beta$ as a discriminator of positive or negative dihedral, shifting the ψ_β scale 180° as dihedral goes from positive to negative. In Reference A1, para. 3.3.1.1, $|\phi/\beta|_d$ is a factor in determining the minimum allowable damping factor $\zeta_d \omega_{n_d}$. The purpose of this appendix is to derive and illustrate the use of these parameters.

Consider first a set of lateral-directional linearized, small-perturbation equations of motion, written in terms of primed stability derivatives (that include product-of-inertia effects) referenced to body axes with the x axis initially aligned with the relative wind in straight-and-level flight. Y_p, Y_r, $Y_{\dot\beta}$ and Y_{δ_a} will be assumed zero, and $p(t)$ will be taken to be $\dot\phi(t)$ $[\,p(s) = s\phi(s)\,]$.

$$\left.\begin{aligned}
&(-s + Y_\beta)\beta \quad - r \quad + \tfrac{g}{V}\phi = 0 \\
&(L'_{\dot\beta} s + L'_\beta)\beta \quad + L'_r r - (s^2 - L'_p s)\phi = L'_{\delta_a}\delta_a \\
&(N'_{\dot\beta} s + N'_\beta)\beta \quad - (s - N'_r) r \quad + N'_p s\phi = -N'_{\delta_a}\delta_a
\end{aligned}\right\} \tag{1}$$

Input-Dependent Parameters

For a step input, $\delta_a(s) = \Delta_a / s$ with Δ_a a constant. From these equations we find (e.g., Reference B73) the response functions

$$\left.\frac{p(s)}{\Delta_a}\right|_{STEP} = \frac{\phi(s)}{\delta_a(s)} = \frac{K_p\left(s^2 + 2\zeta_\phi\omega_\phi s + \omega_\phi^2\right)}{(s + 1/\tau_s)(s + 1/\tau_R)(s^2 + 2\zeta_d\omega_{n_d} s + \omega_{n_d}^2)} \triangleq \frac{N_{p\delta}(s)}{\Delta(s)} \tag{2}$$

$$\left.\frac{\beta(s)}{\Delta_a}\right|_{STEP} = \frac{1}{s}\,\frac{\beta(s)}{\delta_a(s)} = \frac{K_\beta\left[s^2 + (1/\tau_{\beta_1} + 1/\tau_{\beta_2})s + (1/\tau_{\beta_1})(1/\tau_{\beta_2})\right]}{s(s + 1/\tau_s)(s + 1/\tau_R)(s^2 + 2\zeta_d\omega_{n_d} s + \omega_{n_d}^2)} \triangleq \frac{N_{\beta\delta}(s)}{s\Delta(s)} \tag{3}$$

and time responses

$$\left.\frac{\rho(t)}{\Delta_a}\right|_{STEP} = K_S\, e^{-t/\tau_S} + K_R\, e^{-t/\tau_R} + K_d\, e^{-\zeta_d \omega_{n_d} t} \cos\left(\omega_{n_d}\sqrt{1-\zeta_d^2}\,t + \psi_\rho\right) \qquad (4)$$

$$\left.\frac{\beta(t)}{\Delta_a}\right|_{STEP} = C_0 + C_S\, e^{-t/\tau_S} + C_R\, e^{-t/\tau_R} + C_d\, e^{-\zeta_d \omega_{n_d} t} \cos\left(\omega_{n_d}\sqrt{1-\zeta_d^2}\,t + \psi_\beta\right) \qquad (5)$$

where, for example,

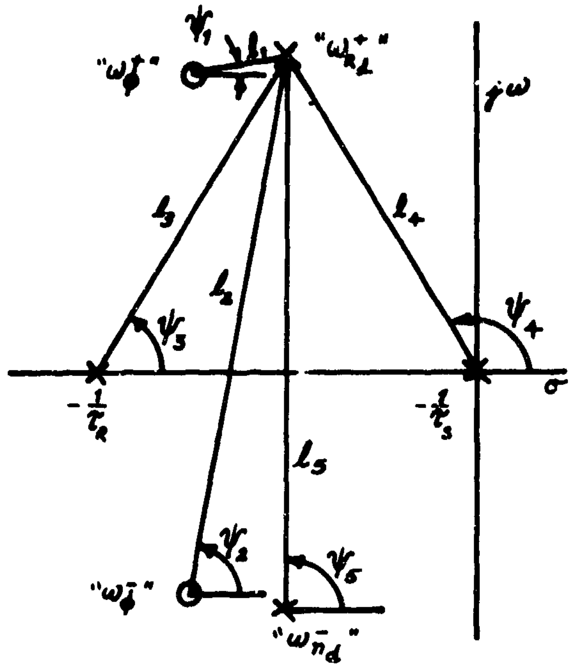
$$K_d = 2\left|\left(s + \zeta_d \omega_{n_d} - j\,\omega_{n_d}\sqrt{1-\zeta_d^2}\right) N_{\rho\delta}(s)/\Delta(s)\right|_{s=-\zeta_d \omega_{n_d} + j\omega_{n_d}\sqrt{1-\zeta_d^2}} \qquad (6)$$

$$\psi_\rho = \angle\left\{N_{\rho\delta}(s) - \angle\left[\Delta(s)/\left(s + \zeta_d \omega_{n_d} - j\,\omega_{n_d}\sqrt{1-\zeta_d^2}\right)\right]\right\}_{s=-\zeta_d \omega_{n_d} + j\omega_{n_d}\sqrt{1-\zeta_d^2}}$$

$$C_d = 2\left|\left(s + \zeta_d \omega_{n_d} - j\,\omega_{n_d}\sqrt{1-\zeta_d^2}\right) N_{\beta\delta}(s)/\Delta(s)\right|_{s=-\zeta_d \omega_{n_d} + j\omega_{n_d}\sqrt{1-\zeta_d^2}} \qquad (7)$$

$$\psi_\beta = \angle\left\{N_{\beta\delta}(s) - \angle\left[\Delta(s)/\left(s + \zeta_d \omega_{n_d} - j\,\omega_{n_d}\sqrt{1-\zeta_d^2}\right)\right]\right\}_{s=-\zeta_d \omega_{n_d} + j\omega_{n_d}\sqrt{1-\zeta_d^2}}$$

The modal response coefficients (K's, C's, ψ_ρ and ψ_β) can also be evaluated graphically, from residues of the poles and zeros plotted in the s plane. For example, consider the pole-zero plot of the above $\left.\frac{\rho(s)}{\delta_a}\right|_{STEP}$ response function, Sketch 1. The residues are evaluated as in Appendix VB.



SKETCH 1: POLE-ZERO PLOT OF $p(s)/\Delta_a \Big|_{\text{STEP}}$

$$K_s = \frac{\overline{(-1/\tau_s)(``\omega_\phi^+")}\cdot\overline{(-1/\tau_s)(``\omega_\phi^-")}}{\overline{(-1/\tau_s)(-1/\tau_R)}\cdot\overline{(-1/\tau_s)(``\omega_{n_d}^+")}\cdot\overline{(-1/\tau_s)(``\omega_{n_d}^-")}}$$

$$K_R = \frac{\overline{(-1/\tau_R)(``\omega_\phi^+")}\cdot\overline{(-1/\tau_R)(``\omega_\phi^-")}}{\overline{(-1/\tau_R)(-1/\tau_s)}\cdot\overline{(-1/\tau_R)(``\omega_{n_d}^+")}\cdot\overline{(-1/\tau_R)(``\omega_{n_d}^-")}}$$

where $\overline{(\;)(\;)}$ denotes the scaled distance between the two points.

$$K_d = \frac{2\,l_1\,l_2}{l_3\,l_4\,l_5}$$

$$\psi_p = \psi_1 + \psi_2 - \psi_3 - \psi_4 - \psi_5$$

$$\approx \psi_1 - \psi_3 - \psi_4 \quad \text{since } \psi_2 \approx \psi_5$$

When, as usual, $|1/\tau_s| \ll \omega_{n_d}$, the further approximation holds:

$$\psi_p \approx \psi_1 - \psi_3 - 90^\circ - \sin^{-1}\zeta_d \tag{8}$$

The relation of the oscillatory pole-zero pair determines ψ_1 :

$$\left(\frac{\omega_\phi}{\omega_{n_d}}\right)^2 \approx 1 - \frac{N'_{\delta_a}\,L'_\beta}{L'_{\delta_a}\,N'_\beta}$$

$$2\zeta_\phi\,\omega_\phi - 2\zeta_d\,\omega_{n_d} \approx \frac{L'_\beta}{N'_\beta}\left(N'_p - \frac{g}{V}\right) + \frac{N'_{\delta a}}{L'_{\delta_a}}\,L'_r \tag{9}$$

(See, for example, Reference B73.) Notice that both $(\omega_\phi/\omega_{n_d})^2$ and $2\zeta_\phi\omega_\phi - 2\zeta_d\omega_{n_d}$ depend directly upon L'_β . Thus there will not be significant Dutch-roll motion in roll for aileron inputs if L'_β is very small. Also, negative dihedral (positive L'_β) tends to shift both ψ_1 and ψ_p (but not ψ_β , as will be shown) by about 180°, thereby reversing the roll-sideslip coupling effects from those encountered with positive dihedral.

The discussion of 3.3.2.2 shows that the tolerable level of p_{osc}/p_{AV} depends on ψ_1 . It is apparent that the large possible variations in $1/\tau_R$ lead to imprecision of ψ_p as an indicator of ψ_1 . It will be found that ψ_β is a better measure. To determine ψ_β , a pole-zero plot can be drawn for $\beta(s)/\Delta_a\,|_{STEP}$, similar to the plot of $p(s)/\Delta_a\,|_{STEP}$ in Sketch 1; but in this appendix another approach is taken.

The Modal Parameters $p/\beta|_d$ and $\phi/\beta|_d$

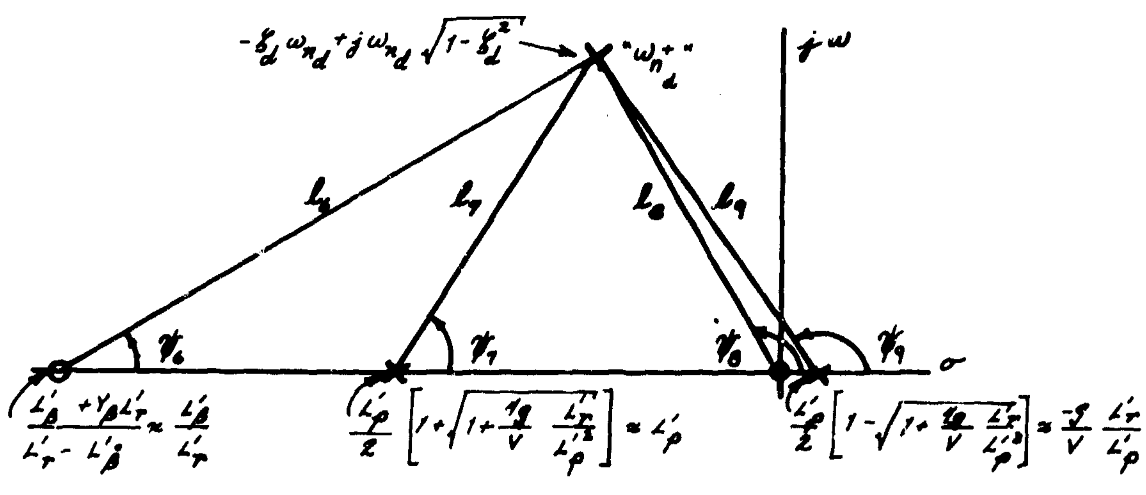
The parameter $p/\beta|_d = \frac{K_d}{C_d} e^{j(\psi_p - \psi_\beta)} = |p/\beta|_d \, e^{j \measuredangle p/\beta}$ is a modal characteristic, like ζ_d and ω_{n_d}. That is, the roll-sideslip amplitude ratio and phase difference of the Dutch-roll components of the motion are completely independent of the input (see, for example, Reference F29). A general solution for $p/\beta|_d$, valid for any input, can be derived from the equations of motion for a pure yawing-moment input:

$$\frac{p}{\beta}\bigg|_d = \frac{-s\left[(L'_r - L'_\beta)\,s - (L'_\beta + Y_\beta L'_r)\right]}{s^2 - L'_p s - \frac{g}{V} L'_r}\Bigg|_{s = -\zeta_d\omega_{n_d} \pm j\omega_{n_d}\sqrt{1-\zeta_d^2}}$$

$$= \frac{-(L'_r - L'_\beta)\,s\left[s - \frac{L'_\beta + Y_\beta L'_r}{L'_r - L'_\beta}\right]}{\left[s - \frac{L'_p}{2}\left(1 + \sqrt{1 + \frac{4g}{V}\frac{L'_r}{L'^2_p}}\right)\right]\left[s - \frac{L'_p}{2}\left(1 - \sqrt{1 + \frac{4g}{V}\frac{L'_r}{L'^2_p}}\right)\right]}\Bigg|_{s = -\zeta_d\omega_{n_d} \pm j\omega_{n_d}\sqrt{1-\zeta_d^2}} \qquad (10)$$

$$\approx \frac{-L'_r\, s\,(s - L'_\beta/L'_r)}{(s - L'_p)\left(s + \frac{g}{V}\frac{L'_r}{L'_p}\right)}\Bigg|_{s = -\zeta_d\omega_{n_d} \pm j\omega_{n_d}\sqrt{1-\zeta_d^2}}$$

This $p/\beta|_d$ response function is sketched below in the s plane, evaluated at the Dutch-roll root $s = -\zeta_d\omega_{n_d} + j\omega_{n_d}\sqrt{1-\zeta_d^2}$:



SKETCH 2: POLE-ZERO PLOT FOR DETERMINING $p/\beta|_d$

$$\left|\frac{p}{\beta}\right|_d = \frac{A_b A_g}{A_\gamma A_q} \quad , \quad or \quad \left|\frac{\phi}{\beta}\right|_d = \frac{A_b}{A_\gamma A_q}$$

$$\begin{aligned} \angle \frac{p}{\beta} &= \psi_b + \psi_g - \psi_\gamma - \psi_q + 180^\circ \\ &\approx \psi_b - \psi_\gamma + 180^\circ \end{aligned} \tag{11}$$

(The 180° is required because of the negative gain in Equation 10, assuming positive $L'_r - L'_{\dot\beta}$.) In the sketch, positive dihedral [negative $(L'_\beta + Y_\beta L'_r)$] has also been assumed. The approximation for $\angle\, p/\beta$ holds when, as usual, $1/\tau_s \approx 0$ and $|g L'_r / V L'_p| \ll 1$.

From Equation 10 and its development can be found

$$\left|\frac{\phi}{\beta}\right|_d = \left|\frac{(L'_r - L'_{\dot\beta})s - (L'_\beta + Y_\beta L'_r)}{s^2 - L'_p s - \frac{g}{V} L'_r}\right|_{s = -\zeta_d \omega_{n_d} \pm j\omega_{n_d}\sqrt{1-\zeta_d^2}} \tag{12}$$

$$= \left(\frac{(L'_r - L'_{\dot\beta})^2 \omega_{n_d}^2 + 2(L'_r - L'_{\dot\beta})(L'_\beta + Y_\beta L'_r)\zeta_d \omega_{n_d} + (L'_\beta + Y_\beta L'_r)^2}{\omega_{n_d}^4 + 2L'_p \zeta_d \omega_{n_d}^3 + \left[L'^2_p + \frac{2g}{V} L'_r (1 - 2\zeta_d^2)\right]\omega_{n_d}^2 - \frac{2g}{V} L'_r L'_p \zeta_d \omega_{n_d} + \left(\frac{g}{V} L'_r\right)^2} \right)^{1/2}$$

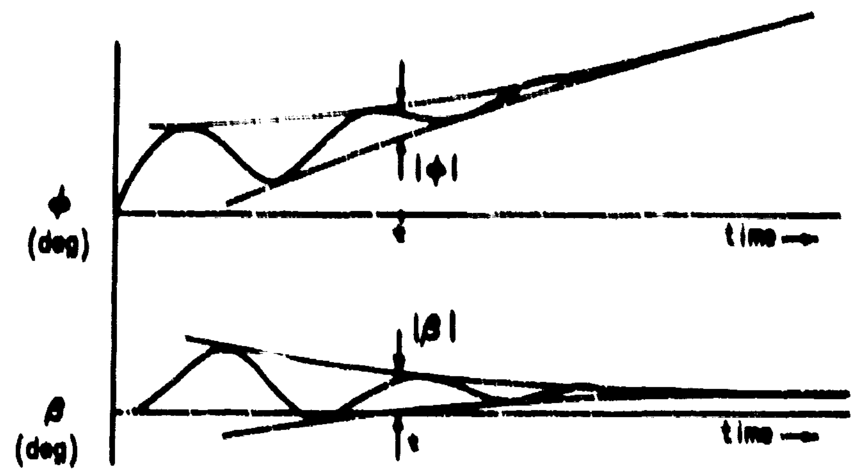
or approximately $$\left|\frac{\phi}{\beta}\right|_d \approx \left|\frac{L'_r (s - L'_\beta / L'_r)}{(s - L'_p)\left(s + \frac{g}{V}\frac{L'_r}{L'_p}\right)}\right|_{s = -\zeta_d \omega_{n_d} \pm j\omega_{n_d}\sqrt{1-\zeta_d^2}}$$

$$\approx \frac{L'_r}{\omega_{n_d}\sqrt{1-\zeta_d^2}} \sqrt{\frac{1 + \left(\frac{\zeta_d \omega_{n_d} + L'_\beta / L'_r}{\omega_{n_d}\sqrt{1-\zeta_d^2}}\right)^2}{\left[1 + \left(\frac{\zeta_d \omega_{n_d} + L'_p}{\omega_{n_d}\sqrt{1-\zeta_d^2}}\right)^2\right]\left[1 + \left(\frac{\zeta_d \omega_{n_d} - \frac{g}{V} L'_r / L'_p}{\omega_{n_d}\sqrt{1-\zeta_d^2}}\right)^2\right]}}$$

Additional approximations can be found in Reference B11. The simplest form, generally not very accurate, is

$$\left|\frac{\phi}{\beta}\right|_d \approx \frac{L'_\beta}{N'_\beta}$$

The parameter $|\phi/\beta|_d$ is one factor in determining the required damping factor $\zeta_d \omega_{n_d}$ (3.3.1.1). From flight or simulator time histories, $|\phi/\beta|_d$ can be determined simply (Sketch 3). Draw the envelopes of the Dutch-roll oscillations in ϕ and β. Then at any instant sufficiently far along in the response (Appendix III), measure $|\phi|$ and $|\beta|$. The ratio is $|\phi/\beta|_d$. At high ζ_d, $|\phi/\beta|_d$ becomes difficult to measure. Partially offsetting this difficulty, fortunately, is the beneficial effect on flying qualities: ζ_d tends to smooth the roll response. If $|\phi/\beta|_d$ cannot be determined from time histories, an analog-matching technique (Appendix VB and References B110-B112) might be used to find the Dutch-roll root and stability derivatives needed to evaluate $|\phi/\beta|_d$ analytically.



SKETCH 3: DEFINITION OF $|\phi/\beta|_d$

We have seen that large $|L'_\beta|$ is necessary to excite the Dutch roll in roll to a great degree with the aileron. It is apparent from Equation 10 and Sketch 2 that large $|L'_\beta|$ will also give large $|p/\beta|_d$ and $|\phi/\beta|_d$, so that the roll response will exhibit much of the Dutch-roll motion; while small $|L'_\beta|$ will tend to minimize $|\phi/\beta|_d$. For roll-sideslip coupling effects, then, interest is in moderate-to-large $|L'_\beta|$. (The limitations on this qualitative discussion will be found by analyzing cases.) Therefore, for conditions of interest, $|L'_\beta + Y_\beta L'_r| \approx |L'_\beta|$ is much larger than $|L'_r - L'_\beta| \approx |L'_r|$, which is usually small. As a result, the zero at $s = (L'_\beta + Y_\beta L'_r)/(L'_r - L'_\beta)$ will be large, so that, depending on the signs of L'_β and L'_r, ψ_ϕ will approach 0 or 180°.

For positive dihedral (negative L'_β), if L'_r is positive, $\psi_\phi \approx 0$; if L'_r is negative, $\psi_\phi \approx 180°$, but 180° must be subtracted from the expression for $\angle p/\beta$ (right side of Equation 11) because the gain of $p/\beta|_d$ (Equation 10) is no longer negative. Thus, for positive dihedral and small L'_r of either sign,

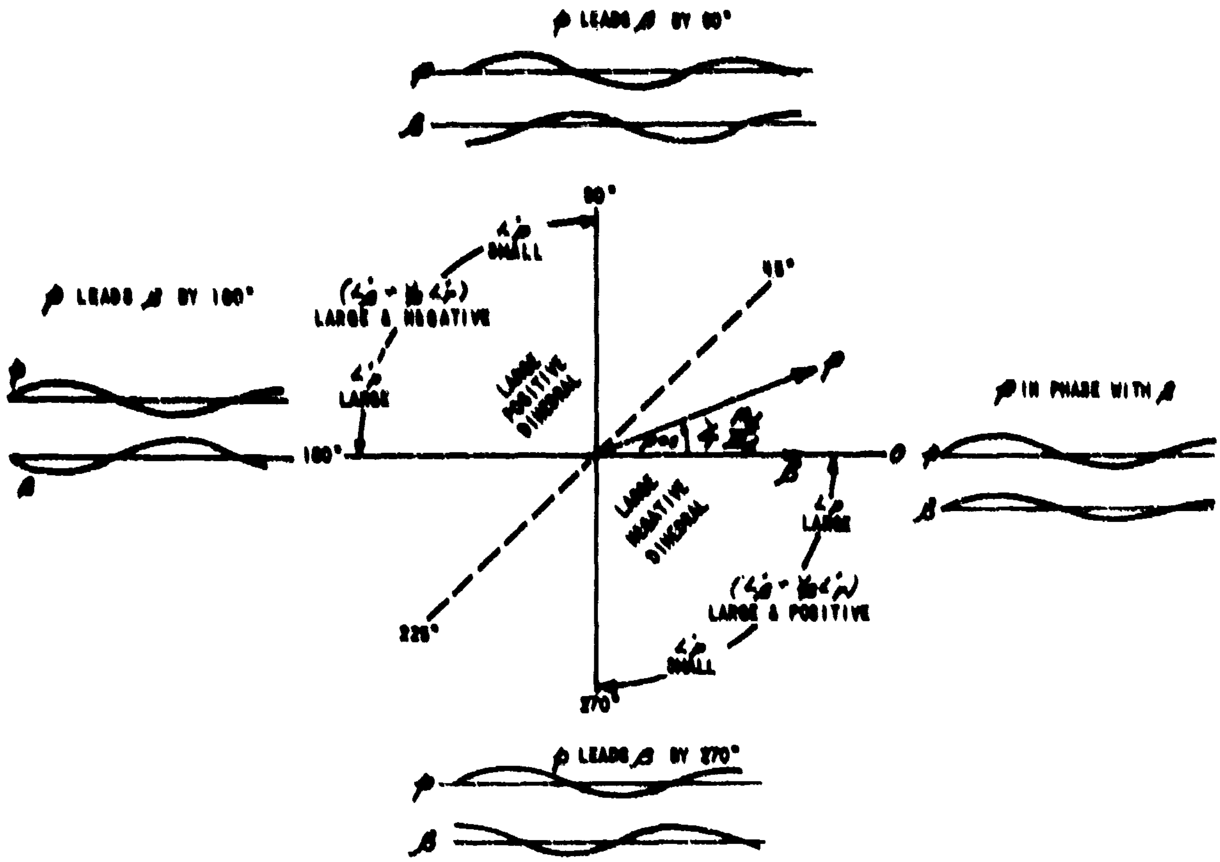
$$\angle \frac{p}{\beta} \approx 180° - \psi_7 \tag{13}$$

Similarly, for negative dihedral (positive $L'_\beta + Y_\beta L'_r \approx L'_\beta$), regardless of the sign of small $L'_r - L'_\beta \approx L'_r$,

$$\angle \frac{p}{\beta} \approx -\psi_7 \quad \text{or} \quad 360° - \psi_7 \tag{14}$$

Now, if $-L'_p$ is large, ψ_7 approaches zero; while if $-L'_p$ is small, $\psi_7 \approx 90°$. These arguments allow a graphical presentation of $\angle p/\beta$ as a function of

$(L'_\beta + Y_\beta L'_r)$ and L'_p, Sketch 4. The central, polar plot might be likened to a time vector diagram of ϕ, with β as a phase reference. The peripheral plots illustrate representative time histories for selected values of $\angle \phi/\beta$.



SKETCH 4: EFFECT OF DIHEDRAL AND ROLL DAMPING ON ROLL-SIDESLIP PHASING IN THE DUTCH ROLL MODE

Allowing 45° buffer zones for cases that violate our assumptions, we can state that:

$45^\circ < \measuredangle \frac{p}{\beta} < 225^\circ$ corresponds to positive dihedral

$225^\circ < \measuredangle \frac{p}{\beta} < 405^\circ$ (45°) corresponds to negative dihedral

Such discrimination will be needed in relating p_{osc}/p_{av} to ψ_β .

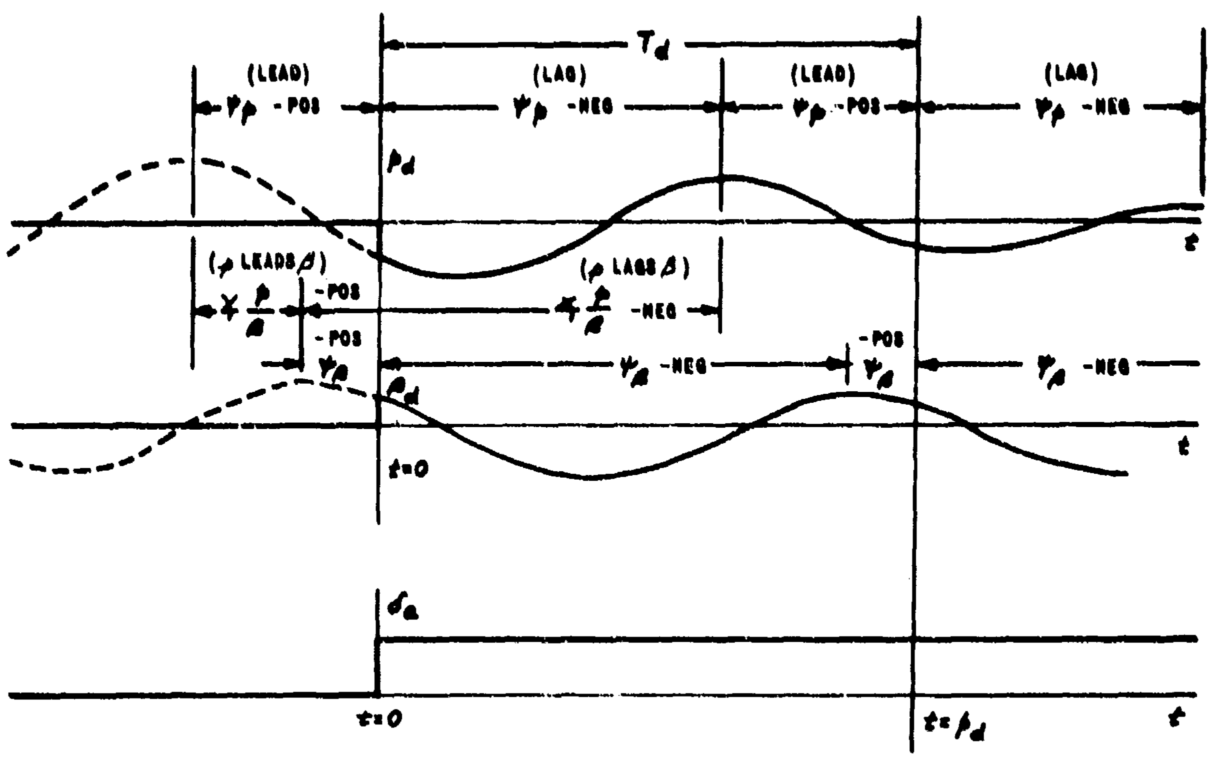
Relationship of ψ_β to ψ_p

Consider the Dutch-roll components of the motion:

$$\left.\frac{p_d}{\delta_a}\right|_{STEP} = K_d e^{-\zeta_d \omega_{n_d} t} \cos\left(\omega_{n_d}\sqrt{1-\zeta_d^2}\, t + \psi_p\right)$$

$$\left.\frac{\beta_d}{\delta_a}\right|_{STEP} = C_d e^{-\zeta_d \omega_{n_d} t} \cos\left(\omega_{n_d}\sqrt{1-\zeta_d^2}\, t + \psi_\beta\right)$$

These motion components are sketched below, with phase angles shown as equivalent time intervals $\left[t = \psi/\left(\omega_{n_d}\sqrt{1-\zeta_d^2}\right)\right]$.



SKETCH 5: TIME HISTORY OF p_d AND β_d FOR A STEP δ_a

From the sketch it can be seen that the signs of ψ_p and ψ_β are positive if the peaks of p_d and β_d are considered to lead the step input (or to lead reference points at t = integer multiples of T_d), while the signs are negative if the peaks of p_d and β_d are considered to lag the step input. Similarly it can be seen that $\angle p/\beta$ is positive if p is considered to lead β, negative if p is considered to lag β. A further relationship that is apparent in the sketch is

$$\psi_\beta = \psi_p - \angle \frac{p}{\beta} \tag{15}$$

Knowing ψ_p and $\angle p/\beta$, we can evaluate ψ_β. For positive dihedral, from Equations 8, 13 and 15,

$$\psi_\beta \approx (\psi_1 - \psi_3 - 90° - \sin^{-1}\zeta_d) - (180° - \psi_7)$$

Since usually $1/\tau_R \approx -L'_p$, we find from Sketches 1 and 2 that $\psi_3 \approx \psi_7$. Then

$$\psi_\beta \approx \psi_1 - \sin^{-1}\zeta_d + 90°; \quad 45° < \angle \frac{p}{\beta} < 225° \tag{16}$$

For negative dihedral (positive $L'_\beta + Y_\beta L'_r$), from Equations 8, 14 and 15 we find similarly:

$$\psi_\beta \approx \psi_1 - \sin^{-1}\zeta_d - 90°; \begin{cases} -135° < \angle \frac{p}{\beta} < 45° \\ (225° < \angle \frac{p}{\beta} < 405°) \end{cases} \tag{17}$$

(These expressions are, of course, modulo 360°). In the preceding section of this appendix we have established $\angle p/\beta$ as a discriminator of positive and negative dihedral. Thus, knowing ψ_β and $\angle p/\beta$, we can estimate ψ_1. Both ψ_β and $\angle p/\beta$ can be measured from flight-test and simulator records.

It was shown previously that a change from positive to negative dihedral shifts ψ_1 by about 180°. But, from Equations 16 and 17, ψ_β would be the same for the two cases. This conclusion verifies the need for two scales for p_{osc}/p_{AV}.

Verification

Since the analysis was based on several assumptions and used several approximations, a rough error analysis was performed to determine the validity of this relationship when individual assumptions were violated. The most significant parameter in these assumptions is L'_r. If L'_r is small, the errors introduced by the individual assumptions will be small. If L'_r is large, however, the errors introduced by the individual assumptions may no longer be small. The error analysis showed that in general, as the errors introduced by individual assumptions become appreciable, the signs of the individual errors were such that they tended to cancel one another.

To confirm this behavior, data from References G10, G11, F1, and F22, and C2 and from operational aircraft were analyzed.

Data from References G10 and G11 are presented in Figure 1 as ψ_1 vs ψ_β. It can be seen that Equation 16 describes the relationship between ψ_1 and ψ_β very accurately. Data from Reference F1 showed a similar behavior, even though in both these programs the roll mode was changed over large values. The Reference F22 data are shown in Figure 2 and can be seen to be described also by Equation 16.

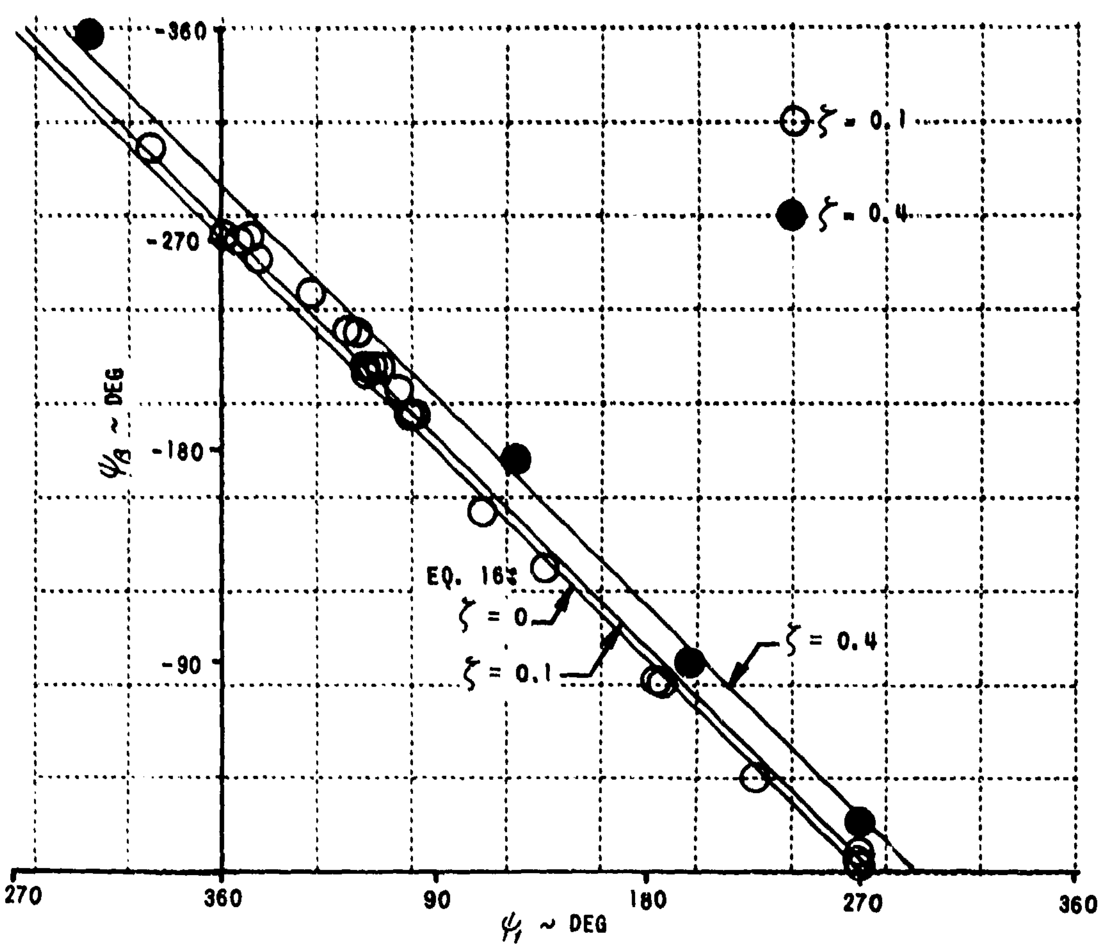


Figure 1 (APPENDIX XC)

RELATIONSHIP BETWEEN ψ_β AND ψ_1 FOR THE DATA OF REFERENCES G10 AND G11

The Reference C2 data show that, for the case where the $\left|\frac{\phi}{\beta}\right|_d$ response ratio is so low that degradation in flying qualities would be expected to arise from sideslip problems rather than from roll oscillations, ψ_β is out by 40°. This represents a very extreme case, outside the limits in which the $\frac{p_{osc}}{p_{AV}}$ criteria need apply or should be expected to apply.

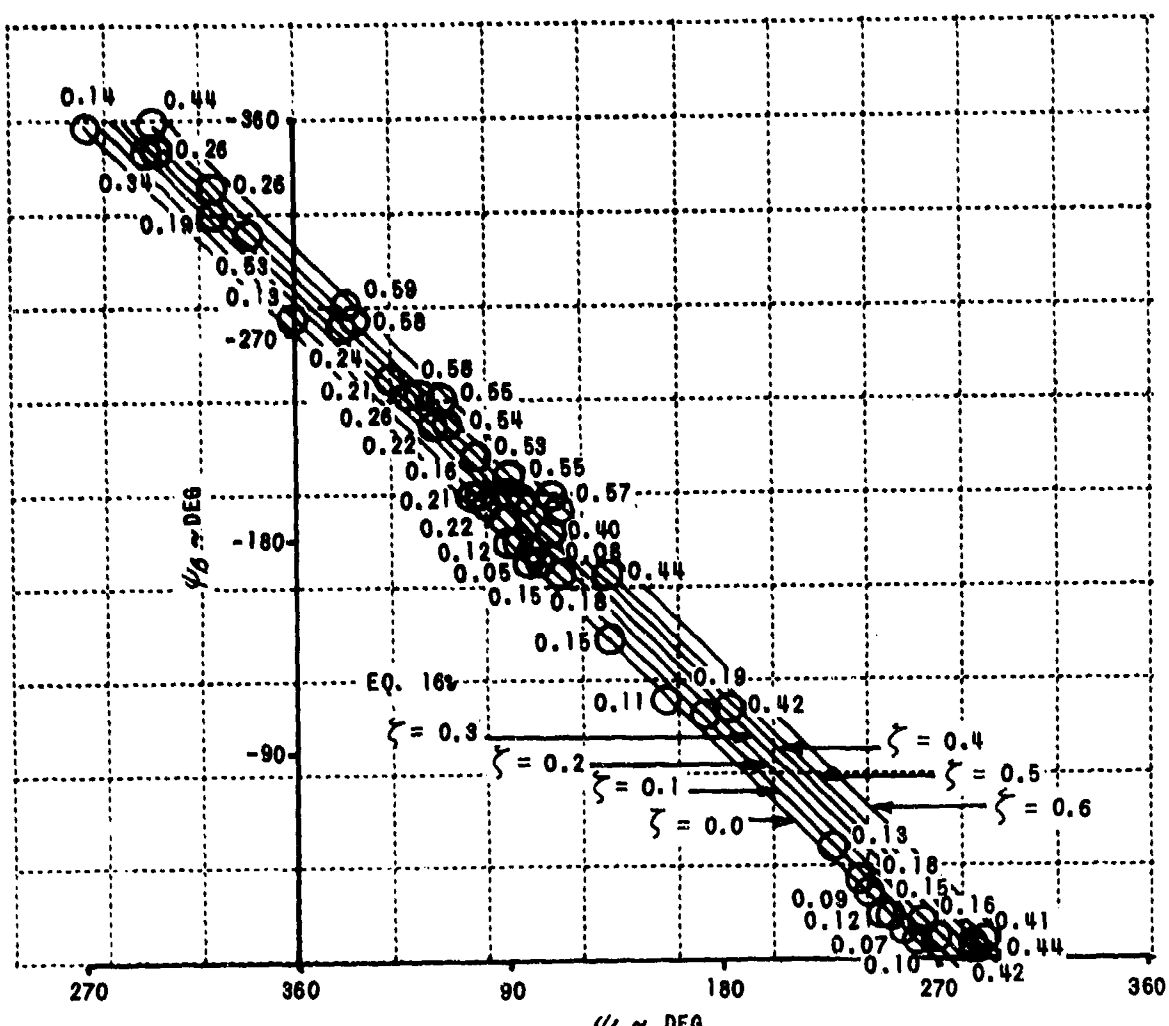


Figure 2 (APPENDIX XC)

RELATIONSHIP BETWEEN ψ_β AND ψ_1 FOR THE DATA OF REFERENCE F22

Finally, the characteristics for a number of existing airplanes are plotted in Figure 3. It can be seen that Equations 16 and 17 are good approximations of the relationship between ψ_β and ψ_1 for existing operational aircraft.

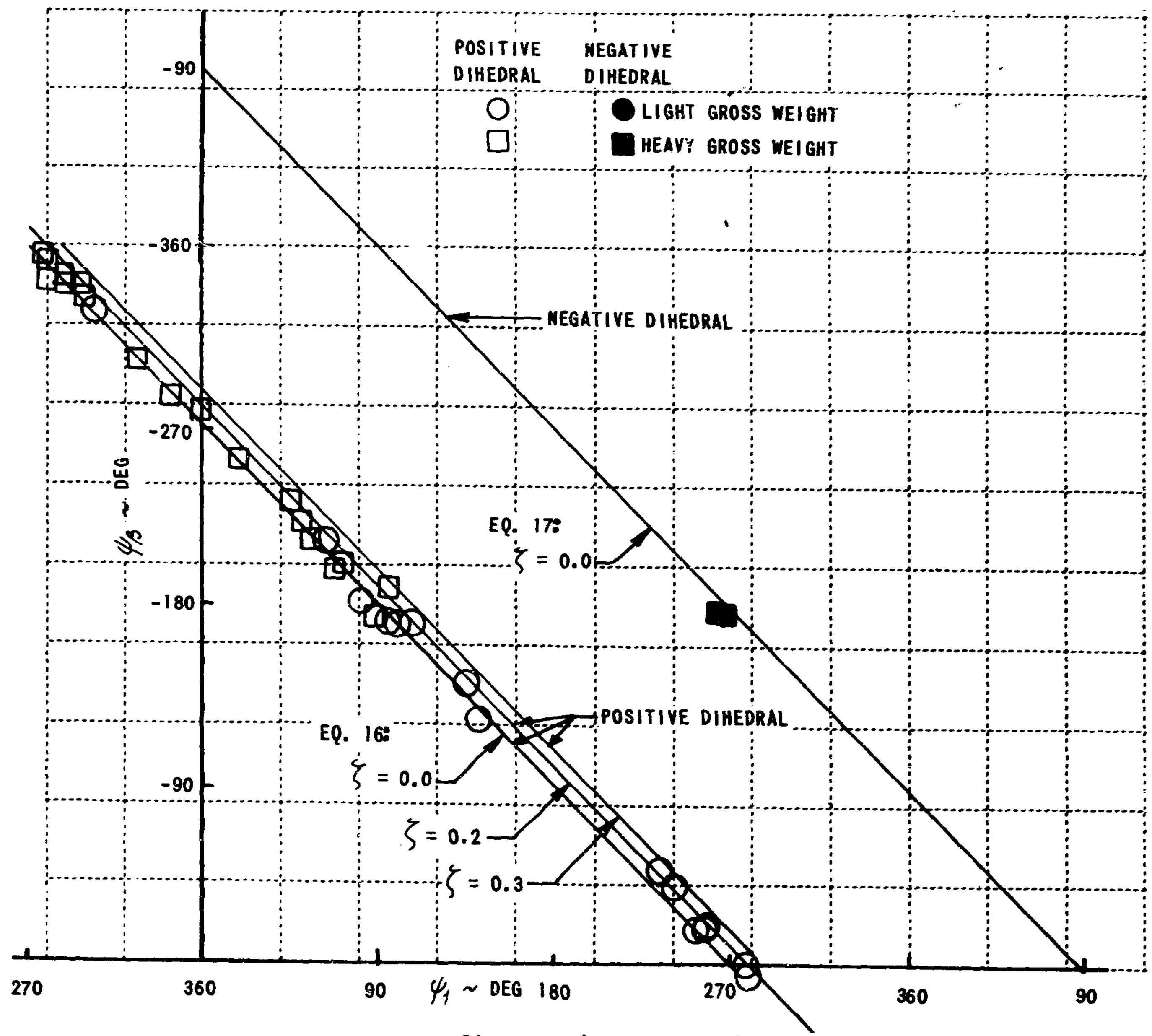


Figure 3 (APPENDIX VC)
RELATIONSHIP BETWEEN ψ_β AND ψ_1 FOR SOME EXISTING AIRPLANES

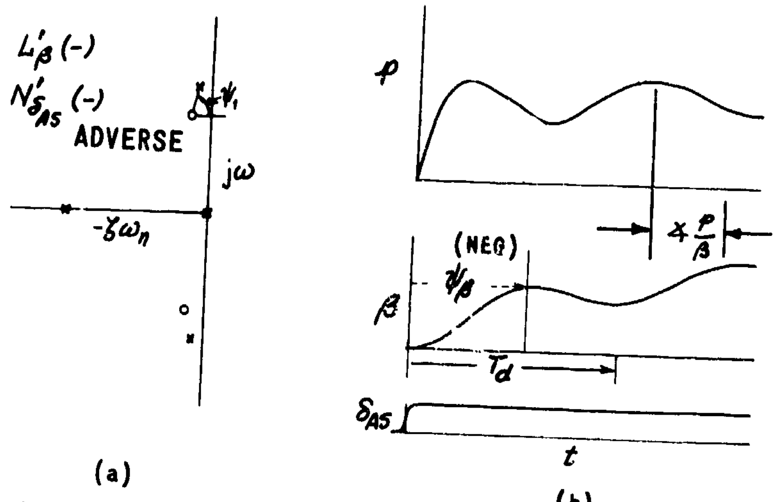
In conclusion, although it has not been proved rigorously, it seems evident that in general when $|L'_\beta|$ is so large that roll-sideslip coupling can lead to bank-angle tracking problems, ψ_β rather uniquely defines the relative position of the ϕ/δ_a transfer-function zero with respect to the Dutch-roll pole. In fact, from examination of a wide variety of data (mostly for negative L'_β) it appears that this relationship holds even for relatively small $|L'_\beta|$. This implies that, even though $(L'_\beta - Y_\beta L'_r)/(L'_r - L'_\beta)$ may range from very large values (where $\psi_\phi \approx 0$) to smaller values (where ψ_ϕ may be quite large), a given phase angle of the Dutch roll in sideslip relatively uniquely describes the trend of pilots' closed-loop tracking problems. In other words, ψ_β is a good measure of ψ_1; and for a given ψ_1, a pilot's aileron inputs proportional to bank-angle error will have the same type of effect on the tolerable level of p_{osc}/p_{AV} even though L'_β ranges from large negative to fairly small negative values. (For zero L'_β there is little or no effect, since then $\zeta_\phi \approx \zeta_d$ and $\omega_\phi \approx \omega_{n_d}$, so roll control does not excite the Dutch roll in the roll response; the sideslip response is another story.)

Relationship Between ψ_β and Rudder Pedal Coordination Characteristics

As well as reflecting lateral-directional closed-loop stability characteristics of an airplane, ψ_β also reflects the difficulty a pilot will experience in attempting to coordinate a turn entry. Consider Sketches 6 through 9, which present sideslip and roll rate time histories following an abrupt aileron input for several values of the coupling parameters $N'_{\delta_{AS}}$ and $[N'_p - g/V]$.

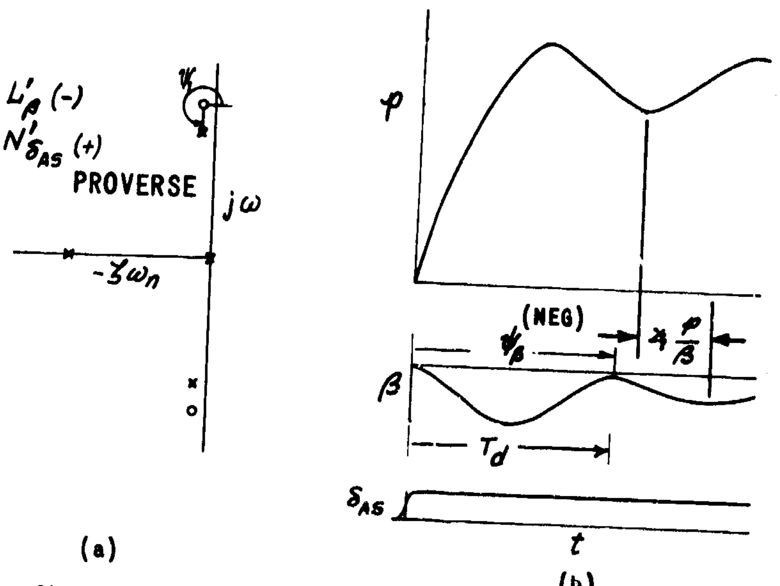
Sketch 6 shows the response for adverse yaw-due-to-aileron ($N'_{\delta_{AS}}$ negative) and Sketch 8 shows the response when the yaw-due-to-roll rate parameter $[N'_p - \frac{g}{V}]$ is in the adverse sense ($[N'_p - \frac{g}{V}]$ negative). Analysis of pilot comments associated with these characteristics reveals that, even though fairly large sideslip angles build up for rudder-pedal-free turn entries, pilots are able to coordinate with rudder pedals quite well. The ψ_β associated with the phasing of these adverse yawing moments is approximately $-180° \geq \psi_\beta \geq -270°$. Examination of the sideslip response from Sketches 6 and 8 indicate that in order to coordinate airplanes with these characteristics, rudder inputs would have to be in the same sense as the aileron input and would have to be phased with, or slightly lag, the aileron inputs. This, apparently, pilots find natural.

Sketch 7 shows the response for proverse yaw-due-to-aileron ($N'_{\delta_{AS}}$ positive) and Sketch 9 shows the response when the yaw-due-to-roll-rate parameter, $[N'_p - g/V]$, is in the proverse sense ($[N'_p - g/V]$ positive). Analysis of pilot comments associated with these characteristics reveals that coordination during turn entries is so difficult that pilots either do not attempt to coordinate or, if they do, often aggravate the situation. The ψ_β associated with the phasing of these proverse yawing moments is approximately $0° \geq \psi_\beta \geq -90°$. Examination of the sideslip response for Sketches 7 and 9 indicates that, in order to coordinate airplanes with these characteristics, rudder inputs would have to be in the opposite sense to aileron inputs; in other words, the pilot would have to cross control. Pilots find this most unnatural.



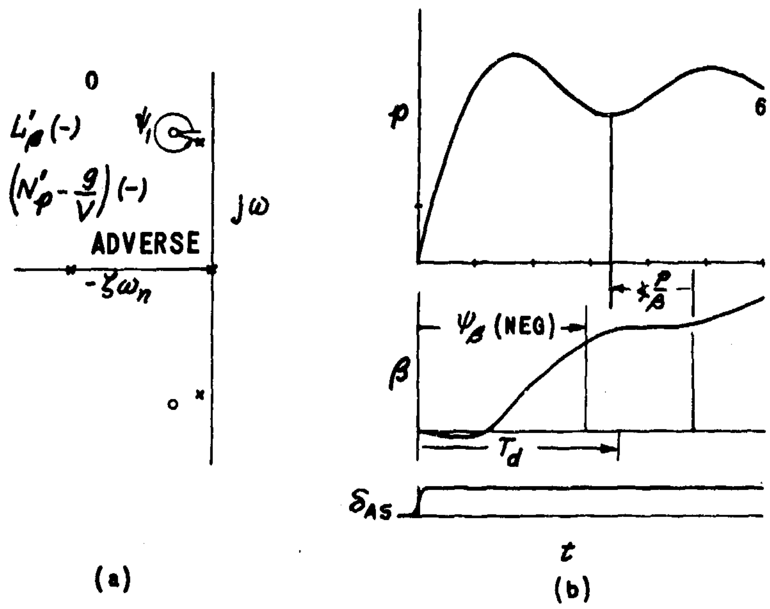
Sketch 6 (a) ϕ/δ_{AS} TRANSFER FUNCTION FACTORS

(b) TIME HISTORY RESPONSES OF p AND β FOR δ_{AS} STEP INPUT, $N'_{\delta_{AS}}$ ~ADVERSE



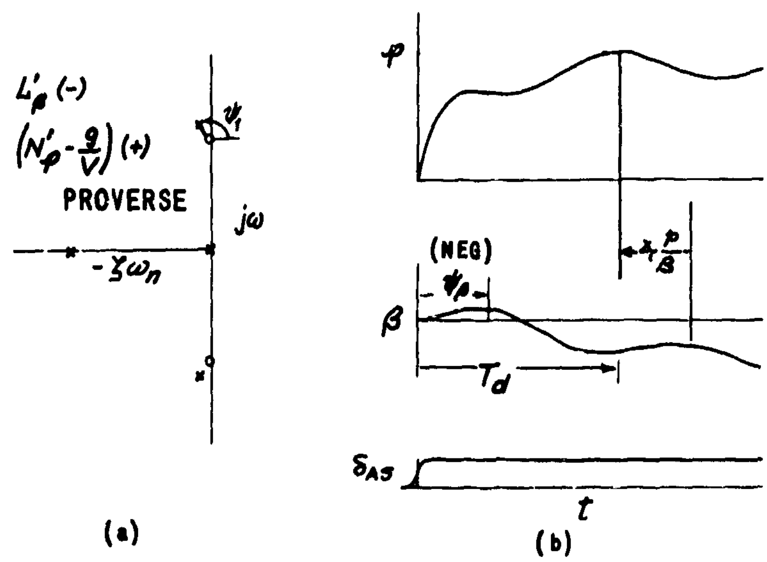
Sketch 7 (a) ϕ/δ_{AS} TRANSFER FUNCTION FACTORS

(b) TIME HISTORY RESPONSES OF p AND β FOR δ_{AS} STEP INPUT $N'_{\delta_{AS}}$ ~ PROVERSE



Sketch 8 (a) ϕ/δ_{AS} TRANSFER FUNCTION FACTORS

(b) TIME HISTORY RESPONSES OF p AND β FOR δ_{AS} STEP INPUT, $N'_p - \frac{g}{V} \sim$ ADVERSE



Sketch 9 (a) ϕ/δ_{AS} TRANSFER FUNCTION FACTORS

(b) TIME HISTORY RESPONSES OF p AND β FOR δ_{AS} STEP INPUT, $N'_p - \frac{g}{V} \sim$ PROVERSE

For airplanes exhibiting yaw-due-to aileron in one direction and yaw-due-to the roll rate parameter $[N'_p - \frac{g}{V}]$ in the other, rudder coordination ranges from relatively easy to extremely difficult depending on the magnitude and sense of the yawing moments. For proverse $N'_{\delta_{AS}}$ and adverse $[N'_p - \frac{g}{V}]$, ψ_β is approximately in the range $-270° \geq \psi_\beta \geq -360°$; and for adverse $N'_{\delta_{AS}}$ and proverse $[N'_p - \frac{g}{V}]$, ψ_β is approximately in the range of $-90° \geq \psi_\beta \geq -180°$.

Comments on the ψ_β Requirements

From the previous discussions it can be seen that the parameter ψ_β describes such seemingly diverse and nebulous factors as closed-loop stability characteristics that are related to lateral-directional coupling derivatives and the ability of a pilot to coordinate turn entries with rudder pedals. Moreover, for an airplane with positive dihedral, it seems that the range of ψ_β for which the closed-loop stability characteristics (rudder pedals free) are favorable, is also the range in which rudder pedal coordination can be readily effected. Conversely, the range of ψ_β for which the closed-loop stability characteristics are unfavorable coincides with the range of ψ_β in which rudder pedal coordination is difficult.

For airplanes with negative dihedral, the relationship between ψ_β and rudder coordination characteristics is the same as for positive dihedral, but the relationship between ψ_β and the closed-loop stability characteristics shifts by 180°. Since there are almost no data relating to negative dihedral, and since for strong negative dihedral the closed-loop stability characteristics were considered to be of paramount importance in roll, ψ_β was shifted by 180° in the roll rate and bank angle oscillation requirements for negative dihedral. Although this clearly does not take into account coordination characteristics or other effects that may be peculiar to negative dihedral, it does address the problem and should not have to be applied often. Since for the sideslip requirements rudder coordination considerations predominate, a single ψ_β scale is applicable for both positive and negative dihedral.

VD. (FROM REFERENCES P1 - P52) NOTES ON ROLL PERFORMANCE OF CURRENT AIRPLANES AND ADEQUACY OF MIL-F-8785 AND REFERENCE A1 REQUIREMENTS

Class I

L-28A - SUPER COURIER

"Lateral control was satisfactory during normal landing approaches, bad during approaches at speeds below 40 knots IAS, and in gusty wind conditions, lateral control was inadequate, particularly during flare and touchdown."

L-23D

"handling characteristics are satisfactory"

L-27A (U-3)

"roll rates are adequate"

Class II

JET COMMANDER - MODEL 1121

"These figures show that the roll capabilities did not comply with MIL-F-8785(ASG) specification except in cruise configuration at high altitude. This is very misleading since both the roll capability and response were more than adequate for this type of aircraft."

Did not meet MIL-F-8785 but would meet Reference A1, B_1 and C_1 requirements if $\tau_R \approx .5-.7$, so strongly supports Reference A1.

LEAR JET - MODEL 23

"...the roll capability did not comply with MIL-F-8785(ASG) specification. This is very misleading since both the roll rate capability and response were greater than necessary for this type of aircraft."

Did not meet MIL-F-8785 requirements but exceeds Reference A1, B_1 (almost meets A_1) and C_1 requirements, so strongly supports Reference A1.

McDONNELL MODEL 119A

Airplane meets MIL-F-8785 and Reference A1 requirements for coordinated rolls with spoilers but does not meet them without spoilers. Report states roll performance was excellent with spoilers but unsatisfactory without; therefore supports Reference A1.

DO-28 Maximum Weight 5400 lb., maximum g - 3.00

Does not quite meet MIL-F-8785 Class I requirements or Reference A1 C_1 requirement, yet stated, "lateral control at low speed was excellent, there being at least 20 degrees of roll per second in the landing configuration at 30 knots."

Shows that more work required for small and low speed aircraft. Supports Reference A1 if this twin engine airplane considered as Class II.

NORATLAS

Did not meet MIL-F-8785 requirement but was considered acceptable. Just met Reference A1 Class II C_2 requirements. Report recommended increased aileron effectiveness to $\frac{pb}{2V}$ = 0.07 or 15 degrees/second. This aircraft would seem to fit Reference A1 Class III requirement better than Class II. Therefore, Class II requirements perhaps a little stringent for large Class II airplanes.

YAC-1DH-CARIBOU

Even though MIL-F-8785 requirements almost met at low speeds, the lateral control was judged to be inadequate. This clearly shows the $\frac{pb}{2V}$ requirements to be inadequate. Roll performance lies just above Reference A1 controllability boundary for Class II so checks with comments. Therefore supports Reference A1 controllability boundary.

CL-329 - JETSTAR

Does not meet MIL-F-8785 in cruise or PA, yet rolling performance is rated excellent. Exceeds Reference A1 C_1 requirement and meets A_1 requirement. Therefore strongly support Reference A1.

C-140 - JETSTAR

Roll rate adequate yet does not meet MIL-F-8785.

YC-134

Low load factor, 62,000 lb. aircraft. Meets MIL-F-8785 yet would not meet Reference A1 Class II B_1 or C_1 requirements. Do not know how roll performance was. Therefore, inconclusive but maybe should be Class III.

TC-131B

Adequate roll rates though forces heavy.

C-123B	54,000 lb.

Meets MIL-F-8785 $\frac{p\phi}{\beta V}$ requirements and rated satisfactory. Fits Reference A1 Class III requirements better than Class II.

B-66B	Design T.O. Weight = 78,000 lb.

Report states that roll rates are excellent and that the lateral control characteristics are far superior at all weights and airspeeds to any other operational bomber.

MIL-F-8785 $\frac{p\phi}{\beta V}$ requirements are just met over the entire speed range.

Exceeds Reference A1 C_1 requirement and almost meets Reference A1 A_1 requirement.

Thus, strongly supports Reference A1.

B-57A

Low roll rates seriously penalize the maneuverability. Did not meet MIL-F-8785 and would probably not meet Reference A1 B_2 requirements. Falls between C_1 and C_2 in PA, but does not comment.

Therefore supports revision.

Class III

C-135B

Roll capability adequate. Would just meet Reference A1 C_1 requirement if τ_R = 1.4 second. Therefore supports Reference A1.

C-141A

MIL-F-8785 $\frac{p\phi}{\beta V}$ requirements not met in cruise or PA. Seems to have marginal roll performance and unacceptable below 150 knots. Recommended minimum acceptable limits check with Reference A1 B_2 and C_2 requirements. Therefore supports Reference A1.

C-133A

Meets MIL-F-8785. Probably falls between Reference A1 Level 1 and 2 requirements. Has adequate roll performance.

Therefore supports Reference A1.

C-130A

Generally met MIL-F-8785 and Reference A1 Class III B_1 and C_1 to C_2 requirements. Does not meet Class II requirements. Therefore, this aircraft belongs in Class III if its roll performance is considered satisfactory. To meet Class II requirements, its roll performance would have to be significantly increased.

C-130B

MIL-F-8785 $\frac{pb}{2V}$ requirement generally met. Seems to only meet Reference A1 Level 2 requirements for Class III. Don't know how roll performance is rated.

YB-58A

Seems to meet Reference A1 B_1 and exceed the C_1 requirements. Fell well below MIL-F-8785 requirements. Lateral control is rated satisfactory so supports Reference A1.

B-52H

Does not meet MIL-F-8785 or Reference A1 requirements but was considered adequate. Reference A1 B_2 requirements not even met under certain flight conditions. Therefore generally supports revision but suggest for very large aircraft requirements could be a bit stringent.

In PA, roll performance is not adequate and probably would not meet Reference A1 C_2 requirement, yet MIL-F-8785 $\frac{pb}{2V}$ requirements are almost met. Therefore supports Reference A1 Flight Category C requirements.

YDB-47E

Seems to meet Reference A1 C_1 requirements. Considered adequate, so supports Reference A1.

Class IV

F-105D

Restricted to 80 degrees/second with pylons yet was considered satisfactory. Would meet B_1 requirement if τ_R = .7 second. Probably is in this range at attack speed. Therefore supports Reference A1.

F-106A

Does not meet MIL-F-8785 $t_{\phi o}$ = 1 second requirement but roll capabilities are considered satisfactory. Roll performance falls between A_1 and B_1 requirements so supports Reference A1.

TF-102A

Does not meet MIL-F-8785 except in PA but considered adequate for mission. Meets Reference A1 PA and up-and-away requirements. Therefore supports Reference A1.

YAT-28B

Generally acceptable although does not meet MIL-F-8785 or Reference A1 requirements everywhere.

OV-10A

Poor roll performance. Did not meet MIL-F-8785 or Reference A1 requirements. Reference A1 Class I or II requirements look about right, but Class IV requirements may be too stringent. It did not even meet Class II requirements. Probably Class I requirements would be appropriate for this type of airplane.

F-100D

In PA $\frac{pb}{2V}$ = .03 and t_{30} = 1 second. So did not meet MIL-F-8785 requirements but meets Reference A1 t_{30} requirement. No adverse comments on roll performance so assume it was acceptable. Therefore supports Reference A1. Met both MIL-F-8785 and Reference A1 requirements in cruise (M = .85).

F-104B

t_{30} varied from 1.0 to 1.7 seconds at speeds from M = .7 to 1.8. Therefore did not meet MIL-F-8785 but met Reference A1 B_1 requirement. Roll performance was rated satisfactory. Therefore supports Reference A1.

F-86H

t_{30} = 1.5 seconds at 310 kts. Does not meet MIL-F-8785 but lies between Reference A1 A_1 and B_1 requirement. Was rated satisfactory. Therefore supports Reference A1.

F-4C

Does not quite meet MIL-F-8785 or Reference A1 requirements everywhere - primarily high supersonic. Rated acceptable in cruise. In PA did not meet MIL-F-8785 or Reference A1 requirements at low end of speed range yet rated satisfactory. Came closer to meeting Reference A1 requirement than MIL-F-8785 requirement. Therefore supports Reference A1.

F-5

Does not meet MIL-F-8785 or Reference A1 requirements nor does the roll performance seem to be very good. Has potential of fitting Reference A1 requirement far better than MIL-F-8785. Supports Reference A1 but underlines problem of external stores.

A-37

With external stores does not come close to meeting MIL-F-8785 requirements. Comes much closer to meeting Reference A1 requirements in 3.3.4.1.1, therefore supports Reference A1.

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13. ABSTRACT
This document is published in support of Military Specification MIL-F-8785B, "Flying Qualities of Piloted Airplanes." It was compiled after an extensive literature review and many meetings and discussions with personnel from essentially all concerned civilian and governmental organizations. The primary purpose is to explain the concept and philosophy underlying MIL-F-8785B and to present some of the data and arguments upon which the requirements were based.

A secondary purpose is to present what are believed to be the important governing variables in the field of flying qualities and to define their significance and relationship to each other. The significance of such mission-oriented factors as airplane class, flight phase, flight condition, loading and configuration is discussed, as is the treatment of failure states. The document should also, to a degree, serve as a summary of the state of the flying qualities art as determined from operational experience, flight test, experiment, analysis and theory.

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