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Thomas T. Myers, Donald E. Johnston, and Duane McRuer

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SYMBOLS

a_y or A_y	Lateral acceleration
A_h	High-frequency gain, altitude numerator
A_θ	High-frequency gain, pitch attitude numerator
c.g.	Center of gravity
C	Cost function, Equation 17
CAP′	See Equation 24
CAP_e'	See Equation 25
$C_{\ell_{\delta_a}}$	Aerodynamic coefficient for roll due to aileron deflection
F	Force
g	Gravitational constant, 32.2 ft/sec
G_{HOS}	HOS transfer function
G_i	Inner forward loop transfer function, Equation 4
G_{LOES}	LOES transfer function
G_o	Outer forward loop transfer function, Equation 7
h	Altitude
h_p	Altitude at pilot location
$j\omega$	Imaginary part of Laplace operator
K_p	Pilot gain
K_{p_h}	Pilot gain, altitude loop
$K_{p\theta}$	Pilot gain, pitch loop, Equation 13
$K_{\dot\theta}$	Outer-loop gain (GDQ), Figure II-2
ℓ_p	Distance from c.g. to pilot, positive forward
$\Delta\ell_p$	$\ell_p - X_{ICR}$

L	Vehicle length
L_δ	Rolling acceleration due to surface deflection
m	Vehicle mass
M	Pitching moment; Mach number
M_q	Pitch damping, $(1/I_y)(\partial M/\partial q)$
M_w	Pitching moment due to heave, $(1/I_y)(\partial M/\partial w)$
M_α	Pitch acceleration due to angle of attack $(1/I_y)$ $(\partial M/\partial \alpha)$
M_{δ_e}	Elevator pitch effectiveness, $(1/I_y)$ $(\partial M/\partial \delta_e)$
n	Normal load factor
n_y	Lateral acceleration
n_z	z-axis load factor
$N_{\delta_e}^{\dot{h}_p}$	Altitude rate, elevator numerator
$N_{\delta_e}^{\theta}$	Pitch attitude numerator
p	Roll rate
q	Pitch rate
Δ_{q_1}, Δ_{q_2}	Peak magnitudes, Fig. II-25
$\bar{q}$	Dynamic pressure
r	Yaw rate
s	Laplace operator, $\sigma \pm j\omega$
t	Time
t_1, t_2	References times, Figure II-25
T	Time constant, sec; torque, in.-lb
T_1	Time constant, rate command LOES form, Equation 16
$1/T_{h_1}$, $1/T_{h_2}$, $1/T_{h_3}$	Altitude numerator zeroes

$1/T_{sp_1}$, $/T_{sp_2}$	(Real) short-period poles
$1/T_{\theta_A}$	Augmented pitch attitude zero, Equation 9
$1/T_{\theta_1}$, $1/T_{\theta_2}$	Pitch attitude numerator zeros
V_T	Total speed
w_k	kth weighting factor, Equation II-17
W	Aircraft weight
x	RHC displacement
$\bar{X}$, $\bar{Y}$, $\bar{Z}$	Distance from aircraft c.g. along the X, Y, Z axes
X_{ICR}	Distance from c.g. to ICR, positive forward
y	Electrical command from RHC displacement
Y_c	Controlled element transfer function
Y_p	Pilot describing function
Y_{p_h}	Pilot describing function, altitude loop, Figure II-1
Y_{p_θ}	Pilot describing function, pitch loop, Figure II-1
Z_w	z-acceleration due to heave, $(1/m)\ (\partial Z/\partial w)$
Z_α	z-acceleration due to angle of attack, $(1/m)(\partial Z/\partial \alpha)$
Z_{δ_e}	z-acceleration due to elevator, $(1/m)(\partial Z/\partial \delta_e)$
Z	z-force
α	Angle of attack
β	Sideslip angle
γ	Flight path angle
δ	Generic surface deflection
δ_c	Inner-loop error signal, Figure II-2
δ_{cc}	Inner-loop command signal, Figure II-2

δ_e	Elevator (elevon) deflection
Δ	Characteristic polynomial
ζ	Damping ratio
ζ_A	Augmented damping ratio, Equation 9
ζ_{h_p}	Damping ratio, pilot's altitude numerator
ζ_p''	Augmented phugoid damping ratio, Equation 8
ζ_{sp}	Short-period damping ratio
θ	Euler pitch angle
$\dot{\theta}_c$	Pitch rate command, Figure II-2
$\ddot{\theta}_{max\ HOS}$	Maximum pitch acceleration, Fig. II-26
σ	Standard deviation; real part of Laplace operator
τ	Time delay
τ_e	Effective time delay
ϕ	Euler roll angle
ψ	Euler yaw angle
ω	Natural frequency
ω_A	Augmented natural frequency, Equation II-9
ω_c	Crossover frequency
ω_e	Inverse time constant, elevator position feedback, Figure II-4
ω_{h_p}	Natural frequency, pilot's altitude numerator
ω_k	kth matching frequency, Equation 17
ω_p''	Augmented phugoid natural frequency, Equation 8
ω_{sp}	Short-period natural frequency

Subscripts

a	Aileron
c	Command
d	Dutch roll
L	Prefilter lag
ps	Pilot station
R	Roll subsidence mode
s	Sidestick
S	Stability axis; spiral mode
SR	Coupled spiral-roll subsidence mode
W	Wheel

Operational Symbols

$\measuredangle$	Phase angle
$(\dot{\ })$	Time derivative
Δ	Increment
$\lvert\ \rvert$	Absolute value

ACRONYMS AND ABBREVIATIONS

A/L or AL	Approach and landing
ALT	Approach and landing test
AR	Amplitude ratio, db
CAP	Control anticipation parameter
c.g.	Aircraft center of gravity
CH	Cooper-Harper (rating scale)
CSS	Control stick steering
dB	Decibel
DA	Design assessment
DOF	Degree of freedom
ELERROR	Filter name, Figure II-2
EM	Elastic mode
FCS	Flight control system
FF5	Free flight 5
FQ	Flying qualities
GRTLS	Glide return to launch site
HAC	Heading alignment cylinder
HOS	Higher order system
HQR	Handling quality rating
ICR	Instantaneous center of rotation
KEAS	Knots equivalent airspeed
LAHOS	Landing and approach higher order system
LOES	Lower order equivalent system
LRU	Line replaceable unit

LVAR	Lateral variation (off-nominal aerodynamic coefficients)
MMLE	Modified maximum likelihood estimator
N	Nominal flight configuration
ND	Not determined
OFT	Orbiter, flight test
PIO	Pilot-induced oscillation
RHC	Rotational hand controller
RI	Rockwell International
SCAS	Stability/command augmentation system
SCR	Supersonic cruise research
SDM	Shuttle design manual
SSCT	Super cruise transport
SST	Supersonic transport
STS	Space Transportation System
TAEM	Terminal area energy management
TIFS	Total in-flight simulator

SECTION I

INTRODUCTION

The purpose of this program is to assess the suitability of existing and proposed flying quality and flight control system criteria for application to the space shuttle, to help define optimum use of flight data in the development of flying quality criteria for space shuttle craft, and to assist the program definition of an Orbiter Experiment for flying qualities and flight control system design criteria. The technical effort was divided into the six major tasks:

I) Review current and proposed flying quality specifications and design guides

II) Review flight control system specifications and design guides

III) Assess applicability of I and II to the flying qualities of space shuttle class vehicles throughout entry, approach, and landing

IV) Recommend critical deficiencies

V) Identify technical areas for flight data to improve flying qualities criteria and design guides

VI) Outline program to develop flying qualities and flight control system criteria and design guides

Succeeding sections will first cover the longitudinal flying qualities review and assessment. This will be followed by a lateral directional flying qualities review and assessment. A summary of flying quality shortcomings will then be presented. Next, the flight control systems specification will then be reviewed and assessed, along with the rotational hand controller review. Finally, a summary of overall findings will be presented.

SECTION II

LONGITUDINAL FLYING QUALITIES CRITERIA REVIEW AND ASSESSMENT

A. INTRODUCTION

There are a number of unique characteristics of the Space Shuttle which have a fundamental impact on its flying qualities requirements. Some derive from the Shuttle's characteristic as an unpowered glider with complex constraints to insure proper energy management during entry. The extreme range of Mach number and altitude during entry result in great variations in aerodynamic characteristics that are as yet uncertain in many areas and require rapid changes in augmentation structure and gains to provide good flying qualities. The longitudinal tasks for either automatic or manual entry include precise control of flight path angle, γ, and speed, V_T, as well as direct control of angle-of-attack, α, for thermal control in the early entry phase. The primary lateral-directional tasks are banking maneuvers for range control.

In the STS-1 flight, entry was flown by the automatic system until the mission commander took control at M=5 and 115,000 ft to fly the final two bank maneuvers. These manual maneuvers (which were planned only shortly before the first flight) was made using the attitude indicator. Before and after each manual bank maneuver, the roll control system was switched to auto and remained in this mode until manual approach was begun at 35,000 ft with a 1.3 g left turn around the Heading Alignment Cylinder (HAC). Speed brakes were used during this maneuver. The final approach was flown at 280 kt on a 20 deg glide slope to the preflare pullup at 1750 ft. Airspeed was then bled off to about 187 kts at touchdown.

B. ORGANIZATION OF SECTION II

The discussion of longitudinal flying qualities begins in Subsection C with some theoretical background for pitch attitude control and explains uncertain unconventional characteristics of the Shuttle due to pitch control augmentation. Section D reviews the Shuttle time domain pitch control specification to assess this specification as well as the Shuttle's pitch control flying qualities. Comparisons are made to recent flying qualities experimental data and data for a number of actual aircraft.

Section E discusses the use of Lower Order Equivalent System models for flying qualities specification as a prelude to the consideration of alternative specifications in Section F and G. Section F considers relevant sections of the present U.S. Military Specification for Flying Qualities -- in particular the treatment of effective time delay. Other alternative specification approaches, either proposed or in development, are considered in Section G. Path control is examined using manual control theory in Section H to explain pilot location problems for the Shuttle which are not adequately treated with conventional flying qualities parameters. Finally, a summary of conclusions regarding longitudinal flying qualities is given in Section I.

C. THEORETICAL OVERVIEW OF SHUTTLE LONGITUDINAL MANUAL CONTROL

1. Introduction

Despite the many unique features of the Shuttle and its operation, it is a "conventional" aircraft in the sense that the pilot controls the flight path (altitude and/or flight path angle) by modulating angle-of-attack with an inner pitch attitude loop. As is common for conventional aircraft, the pilot may be expected to use a series loop structure as shown in Fig. II-1*. The "vehicle dynamics" represented by

*Further support for this will be given in Subsection H.

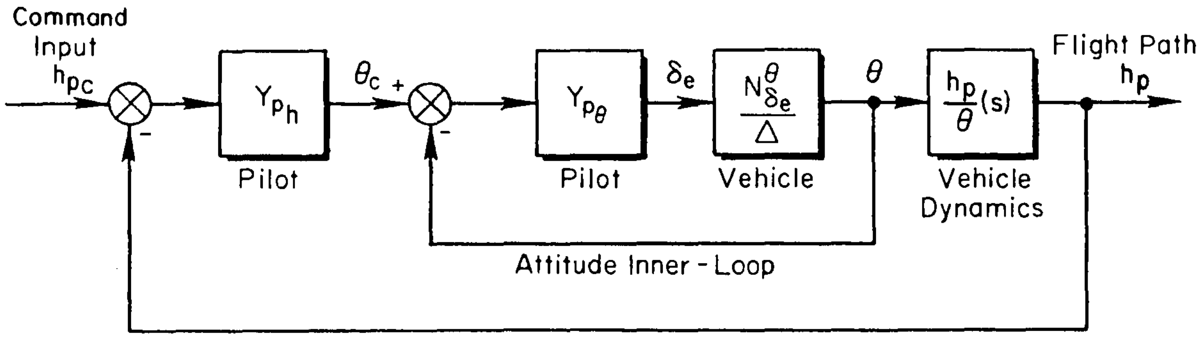


Figure II-1. A Model for Manual Control of Flight Path with an Inner Attitude Loop (series closure)

$$\frac{h_p}{\theta}(s) = \frac{(1/s)N^{\dot{h}_p}_{\delta_e}}{N^{\theta}_{\delta_e}} = \frac{A_h(1/T_{h_1})(1/T_{h_2})(1/T_{h_3})}{A_\theta s(1/T_{\theta_1})(1/T_{\theta_2})}\ ^{*} \tag{1}$$

are those of the bare airframe and will not be changed by any augmentation (i.e., feedbacks to the elevator). Since the airframe parameters which determine h_p/θ are now essentially fixed for the Shuttle, the only real possibilities for modifying longitudinal flying qualities would be in changes to the (augmented) pitch attitude dynamics. These may come about from changes in the digital software and hardware of the flight control system as the Shuttle evolves.

Under the short period (constant speed) approximation, Ref. 1, the vehicle path dynamics simplify to

$$\frac{h_p}{\theta}(s) \doteq \frac{A_h(1/T_{h_2})(1/T_{h_3})}{A_\theta s(1/T_{\theta_2})} \tag{2}$$

---

*Transfer function notation

$$(a) \equiv (s + a)$$

$$[\zeta, \omega] \equiv [s^2 + 2\zeta\omega s + \omega^2]$$

and the (airframe) pitch attitude response to elevator is

$$\frac{\dot{\theta}}{\delta_e}(s) = \frac{A_\theta(1/T_{\theta_2})}{[\zeta_{SP}, \omega_{SP}]} \tag{3}$$

Thus, for a conventional aircraft without significant pitch augmentation, the attitude numerator zero is $(1/T_{\theta_2})$ and is therefore intrinsically the same as the path mode inverse time constant in Eq. 2. It will be shown next, that this identity does not necessarily hold for a highly augmented aircraft such as the Shuttle and that this creates special problems in using the existing flying qualities data base to assess the Shuttle.

2. Approximation of the Augmented Pitch Response

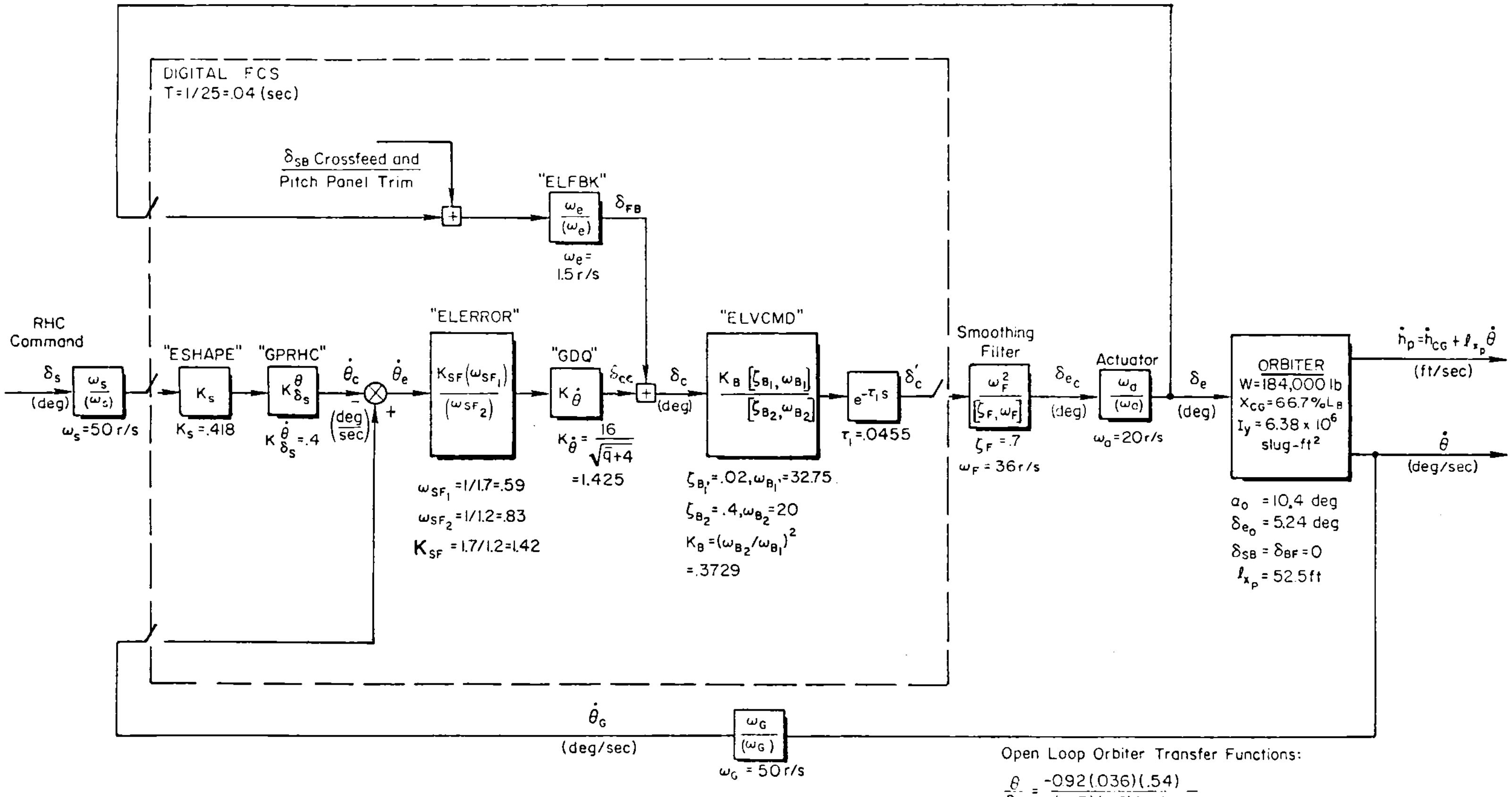
The portion of the Shuttle longitudinal flight control system of primary interest is the pitch control channel. Figure II-2 shows the pitch channel for the OFT in a typical approach flight condition (h = 2420 ft, V_T = 190 KEAS, $\bar{q}$ = 122 psf) taken from Ref. 2. The gains and even the loop structure vary with flight condition and the present Shuttle FCS has had a number of detail changes; however, this system is representative of the basic concept.

The inner feedback loop ($\delta_e \to \delta_c$) is employed to provide an integral path in the forward loop. The forward loop of this part of the system consists of elements having bandwidths much higher than the equalization (ELERROR) and airframe modal frequencies. With the inner loop closed

$$\frac{\delta_e}{\delta_{cc}} = \frac{G_i(s)}{1 - \left(\frac{\omega_e}{s + \omega_e}\right)G_i(s)} = \frac{(s + \omega_e)G_i(s)}{s + \omega_e\left(1 - G_i(s)\right)} \tag{4}$$

where the (inner) forward loop transfer function, $G_i(s)$, is

$$G_i(s) = \frac{.3729[.02, 32.75]}{[.4, 20]} \cdot \frac{36.0^2}{[.7, 36.0]} \cdot \frac{20}{(20)} \cdot e^{-.0455s}$$



Open Loop Orbiter Transfer Functions:

$$\frac{\theta}{\delta_e} = \frac{-.092(.036)(.54)}{(-.17)(.85)[.42,.161]}$$

$$\frac{h_p}{\theta} = \frac{-0.251(.0013)(3.16)(-3.93)}{s(.036)(.54)}$$

Note: $(\omega_i) \equiv (s+\omega_i)$

$[\zeta_i;\omega_i] \equiv [s^2 + 2(\zeta_i)(\omega_i)s + (\omega_i)^2]$

Figure II-2. Linearized OFT Orbiter/DFCS Pitch Channel Landing Approach Condition: h=2420 ft, V=190 kt EAS, $\bar{q}$=122 psf Small Inputs: $|\delta_s| \leq 5$ deg (Ref. 4)

At frequencies well below the $G_i(s)$ break frequencies ($|s| \ll 20$ rad/sec) $|G_i(s)| \doteq 1.0$ and

$$\frac{\delta_e}{\delta_{cc}} \doteq \left(\frac{s + \omega_e}{s}\right) G_i(s) \tag{5}$$

This approximation shows that the inner loop essentially provides a low frequency ($|s| < \omega_e$) integrator for the outer pitch rate loop.

A system survey of the outer loop closure is shown in Fig. II-3 with the bare airframe pitch rate response to elevator given by

$$\frac{\dot{\theta}}{\delta_e}(s) = \frac{A_\theta s\left(1/T_{\theta_1}\right)\left(1/T_{\theta_2}\right)}{\left(1/T_{SP_1}\right)\left(1/T_{SP_2}\right)\left[\zeta_p, \omega_p\right]} \tag{6}$$

$$= \frac{-0.92(0)(.036)(.54)}{(-.17)(.85)[.42,.16]} \quad \frac{\text{deg/sec}}{\text{deg}}$$

The short period mode has real roots due to the unstable static margin. Figure II-3 shows the Space Shuttle to be a "highly augmented" vehicle in that, at the nominal loop gain of $K_{\dot{\theta}} = 1.425$ deg/deg sec^{-1}, the airframe poles are greatly modified. Specifically, the "short period" (aperiodic) poles are driven into the numerator zeroes and effectively canceled while the phugoid is driven to higher frequency to become the dominant closed loop mode (effective short period).

The major influences determining this dominant mode may be seen from an approximation in the crossover (0 dB) region. Making asymptotic approximations around ω_e for the open loop transfer function $G_o(s)$

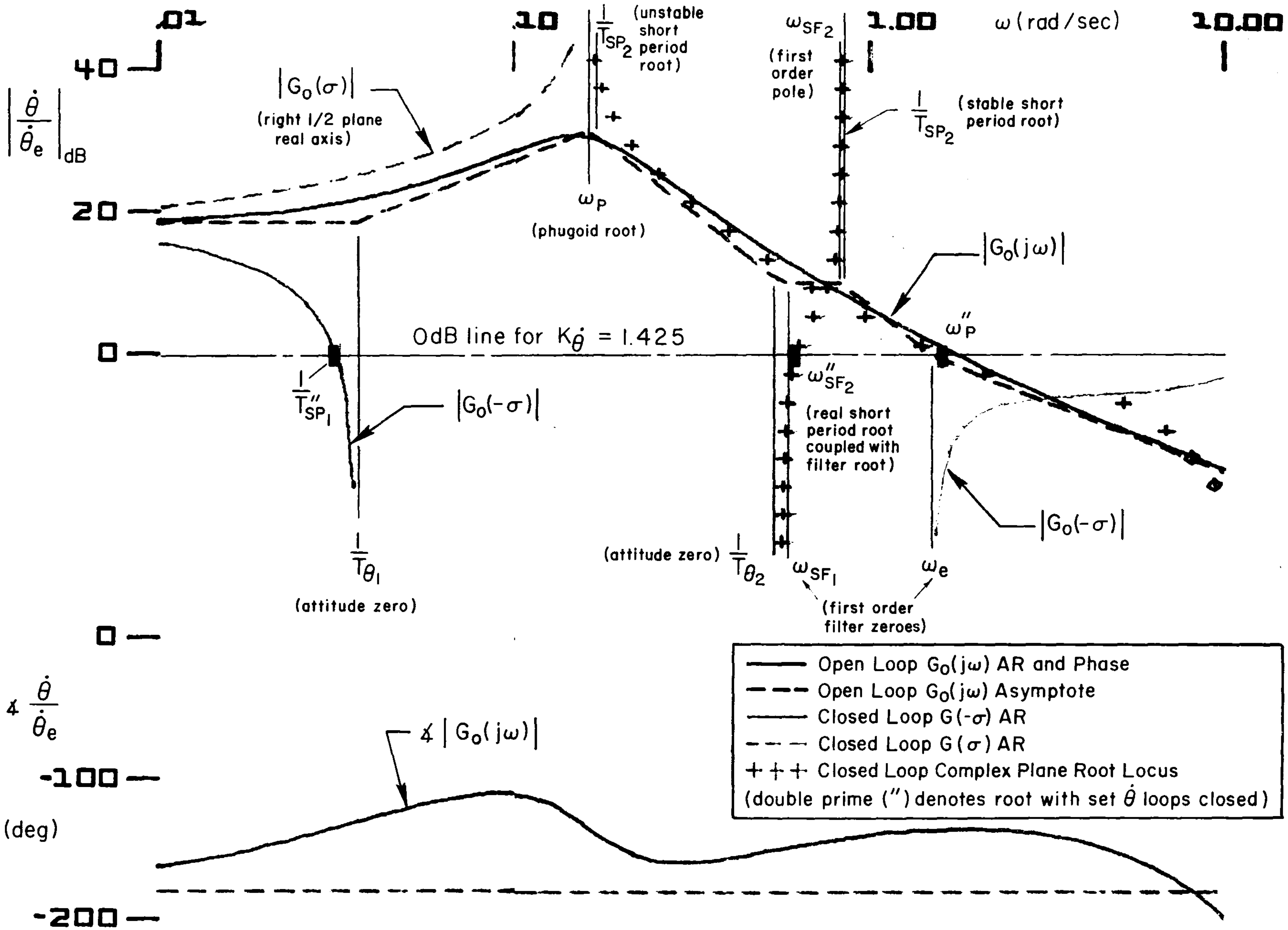


Figure II-3. Closure of Pitch Rate Loop, OFT

$$G_o(s) = \left.\frac{\theta}{\delta_e}\right|_{O.L} \doteq \frac{M_{\delta_e} K_{\dot{\theta}}(s + \omega_e)}{s^2} \tag{7}$$

$$= \frac{-.92 \times 1.425(1.5)}{(0)^2} \quad \frac{\text{deg/sec}}{\text{deg/sec}}$$

and thus the dominant closed loop pole is

$$[\zeta_p'', \omega_p''] \doteq s^2 - M_{\delta_e} K_{\dot{\theta}} s - M_{\delta_e} K_{\dot{\theta}} \omega_e$$

$$= \left[\frac{1}{2}\sqrt{\frac{-M_{\delta_e} K_{\dot{\theta}}}{\omega_e}}, \sqrt{-M_{\delta_e} K_{\theta} \omega_e}\right] \tag{8}$$

$$= [.468, 1.40]$$

These approximate values should be compared to the exact values of

$$[\zeta_p'', \omega_p''] = [.625, 1.60]$$

from Ref. 2. It should be noted that the equalization element "ELERROR" has little effect and that the high frequency elements may again be taken "out of the loop." Thus, with the outer loop closed

$$\frac{\dot{\theta}''}{\dot{\theta}_c} \doteq \frac{-M_{\delta_e} K_{\dot{\theta}}(\omega_e)}{\left[\frac{1}{2}\sqrt{\frac{-M_{\delta_e} K_{\dot{\theta}}}{\omega_e}}, \sqrt{-M_{\delta_e} K_{\dot{\theta}} \omega_e}\right]} \cdot e^{-\tau_e s} \tag{9}$$

$$\equiv \frac{K_{\dot{\theta}}(1/T_{\theta_A})}{[\zeta_A, \omega_A]} e^{-\tau_e s}$$

where the equivalent time delay, $\tau_e = 0.174$ sec, is a low frequency approximation to $G_o(s)$ i.e.,

$$
\begin{aligned}
G_o(s) &= |G_o(s)|e^{-j\angle G_o(s)} \\
&\doteq e^{-.174s} \quad \text{for } |s| \ll 20 \text{ rad/sec}
\end{aligned}
\tag{10}
$$

For a conventional aircraft with stable static margin, the attitude zero is $(1/T_{\theta_2})$ where $1/T_{\theta_2} \doteq -Z_w + M_w Z_{\delta_e}/M_{\delta_e}$ and exists as a consequence of the coupling of γ into the pitching moment equation through the M_α term. When $M_\alpha \doteq 0$, as for the Shuttle, this attitude zero is cancelled by a corresponding "short period" (aperiodic) pole, $1/T_{SP_2} \doteq -Z_w$; thus there will be no attitude zero unless one is introduced "artificially" as in Eqs. 7 and 9. Some additional perspective may be gained by considering the pitching moment equation with augmentation where the inner loop is represented by a low frequency approximation of Eq. 11

$$
\begin{aligned}
\ddot{\theta} - M_q\dot{\theta} - M_\alpha \alpha &= M_{\delta_e}\delta_e \\
&\doteq M_{\delta_e} K_{\dot{\theta}} \frac{(s + \omega_e)}{s}\left(\dot{\theta} - \dot{\theta}_c\right)
\end{aligned}
\tag{11}
$$

If M_α and M_q are negligible, Eq. 11 leads approximately to Eq. 9.

An important implication of the attitude zero should be noted. Whether the zero is due to γ coupling into the pitching moment equation in the conventional case or to an augmentation artifact as for the Shuttle, the attitude response may be written in the form of

$$
\begin{aligned}
\frac{\dot{\theta}}{\delta} &= \frac{A_\theta\left(1/T_{\theta_2}\right)}{\left[\zeta_{SP}, \omega_{SP}\right]} \\
&= \frac{A_\theta s}{\left[\zeta_{SP}, \omega_{SP}\right]} + \frac{A_\theta/T_{\theta_2}}{\left[\zeta_{SP}, \omega_{SP}\right]}
\end{aligned}
\tag{12}
$$

Equation 12 shows that the pitch rate response is the sum of two terms -- the first being the derivative of T_{θ_2} times the second. This is shown graphically in Fig. II-4 which reveals that the $(1/T_{\theta_2})$ zero may produce a large $\dot{\theta}$ overshoot even when the short period is well damped. Thus, the $(1/T_{\theta_2})$ zero can have a significant influence on rise time and effective time delay.

In summary, Eq. 9 shows the Shuttle FCS to be a pitch rate command system with a bandwidth of about 1.5 rad/sec for this approach flight condition. This characteristic should be compared to the classical short period approximation in Eq. 3 to see that while the <u>form</u> is the same ("first over second order") the classical roots are different from those of the augmented vehicle which are independent of aerodynamics except for elevator pitch effectiveness, M_{δ_e}. It should be noted that the pitch augmentation will have no direct effect on path response but rather h_p will be influenced only by the effect on θ. Thus, for the Shuttle OFT we have an unconventional situation in which the numerator lead in $\dot{\theta}/\dot{\theta}_c$ normally given by $1/T_{\theta_2}$ is instead approximated by $\omega_e = 1.5\ \sec^{-1}$ which is greatly different from the path angle inverse time constant $1/T_{\theta_2} = 0.54\ \sec^{-1}$.

3. Manual Control of the Augmented Pitch Dynamics

The unconventional difference between the attitude numerator and the path angle inverse time constant just noted implies that special consideration must be given to assessing Shuttle flying qualities on the basis of the empirical flying qualities data base which is derived largely from experience with conventional airframes. This issue will be discussed further in Subsections D and E. However, the manual control theory originally developed for conventional aircraft is general enough to be directly applicable to the Shuttle. This is well summarized in Refs. 3, 4 and 5 which show how $1/T_{\theta_2}$ and ω_{SP} (or now more generally $1/T_{\theta_A}$ and ω_A) largely determine the pilot model form for pitch attitude control. This theory was used in the Ref. 2 analysis of the Shuttle ALT landing and it will be briefly reviewed here as background for the review of specifications in Subsections D, F, and G and in preparation

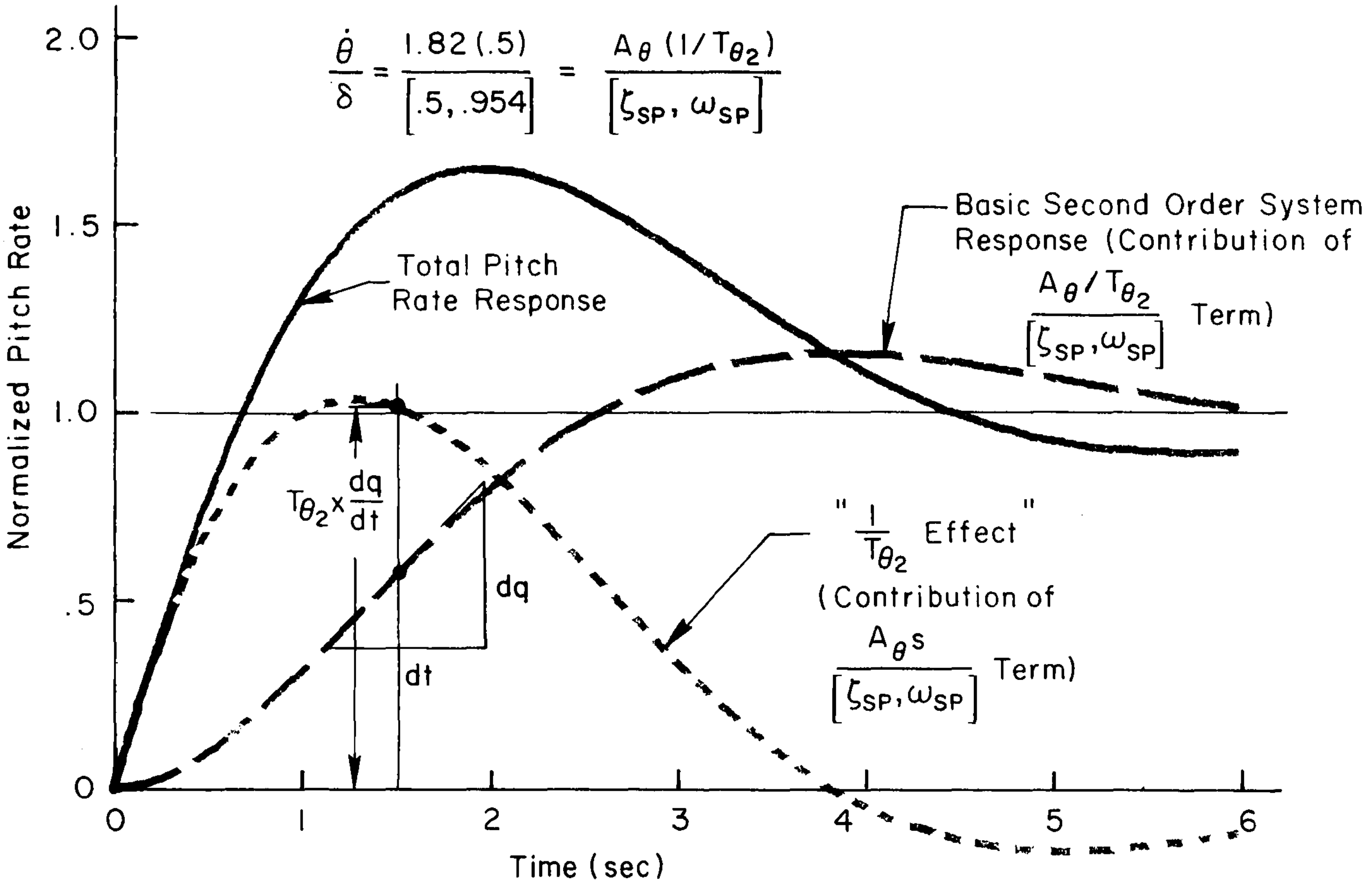


Figure II-4. Effect of $1/T_{\theta_2}$ Zero on Pitch Rate Step Response for the discussion of path (or equivalently altitude) control in Subsection H.

For manual attitude control of the Shuttle OFT the controlled element, Y_c, may be simply represented by the integral of Eq. 9. A pilot model, Y_{p_θ}, was defined for the Shuttle ALT in Ref. 2 using the pilot modeling rules of Ref. 5. In the same manner a pilot model for the OFT may be formulated as

$$Y_{p_\theta}(s) \quad = \quad K_{p_\theta}\left(\frac{s}{1.91} + 1\right) e^{-.26s} \;^{*} \tag{13}$$

Thus

$$Y_{p_\theta}Y_c \quad \doteq \quad \frac{.686K_{p_\theta}(1.91)(1.5)}{(0)[.468,\ 1.40]}\, e^{-.434s} \tag{14}$$

for which the attitude loop closure is shown in Fig. II-5 with the time delay approximated by a first-order Padé form as

$$e^{-.434s} \quad \doteq \quad -\frac{(s - 4.61)}{(s + 4.61)} \tag{15}$$

The system survey of Fig. II-5 indicates that the highest crossover frequency that the pilot could achieve would be about $\omega_c \doteq 1.4\ \text{sec}^{-1}$ (based on a 45 deg phase margin requirement). The 0.434 sec effective time delay (40 percent of which comes from the controlled element) is a primary factor in this bandwidth limit since it reduces the phase angle by $\tau\omega = 34.8^\circ$ at $\omega = 1.4\ \text{sec}^{-1}$. This dominant effect of time delay in pitch attitude control will be a recurring theme in the following sections.

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*This model is based on the "rate command" approximation of Y_c given in Table 3 and discussed in Subsection E.

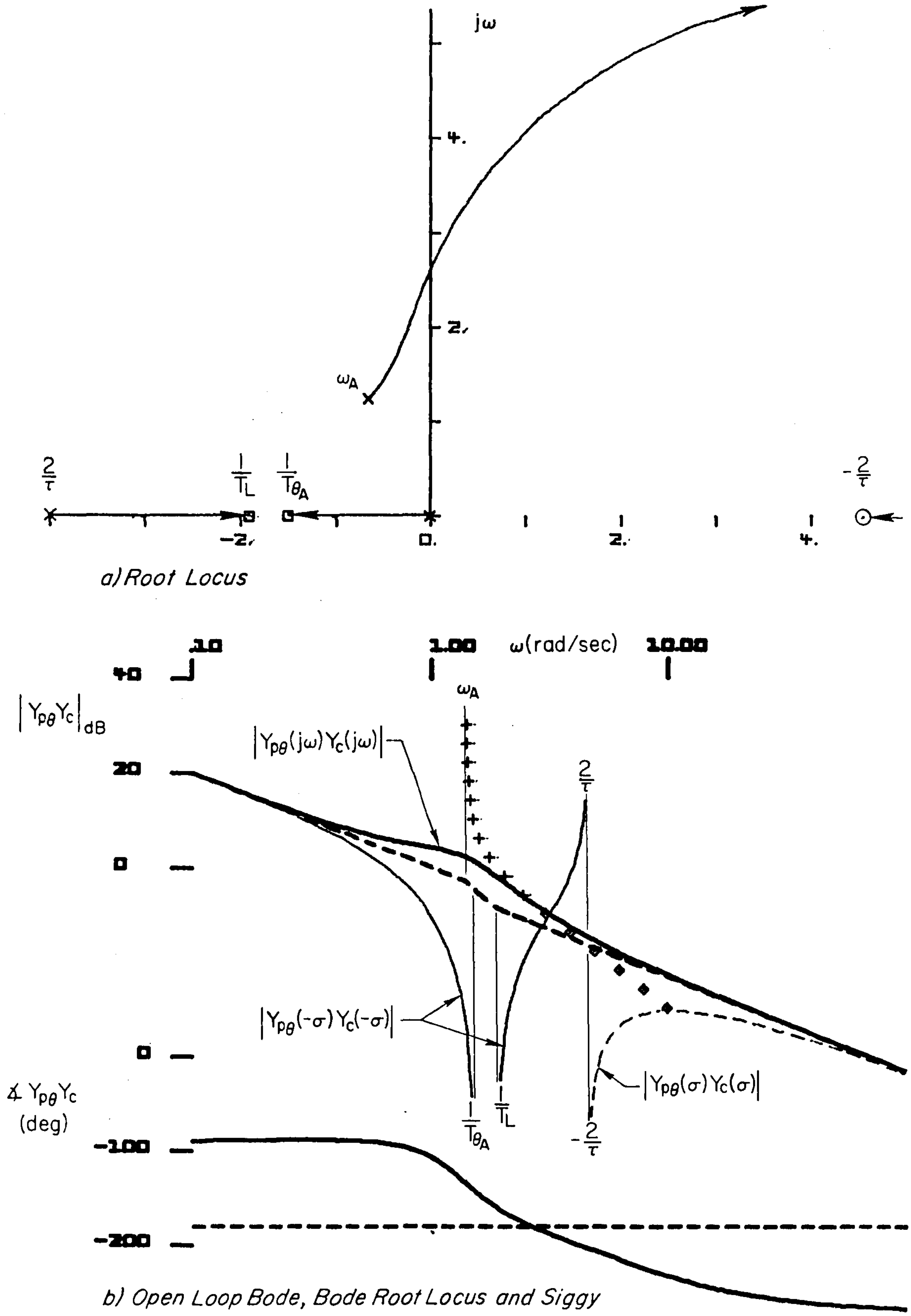


Figure II-5. System Survey of Pilot's Attitude
Loop Closure, Shuttle OFT

D. REVIEW OF THE SPACE SHUTTLE PITCH CONTROL SPECIFICATION

The review of the Space Shuttle pitch control specifications began with the 1973 Shuttle flying qualities requirements of Ref. 6. In this document, the primary specifications on pitch attitude control for (unpowered) aerodynamic flight during entry are

a) Paragraph 3.4.2.1 "Rate Command" which specifies that the FCS shall provide a proportional pitch rate for RHC inputs. The means by which this has been implemented is clear from the previous discussion of the Shuttle FCS pitch channel.

b) Paragraph 3.4.3.1 "Response to Command" specifies the maximum and minimum steady state pitch rates that may be commanded and provides a time domain specification for transient response.

The time domain transient response specification is the most complex specification item and received most of the attention in the review. Consequently it will be the topic of this section. The primary issues are the definition of the time response criterion boundary (the boundary has evolved during Shuttle development and continues to change) and whether the present boundaries are adequate.

1. Pitch Rate Time Response Boundaries

Figure II-6 shows the pitch rate response boundaries (which are also used for roll rate) for step inputs given in the 1973 Shuttle flying qualities specification, Ref. 6. It may be seen that the high supersonic boundary (solid lines) was the least restrictive, the low supersonic boundary (dashed lines) was more restrictive, and finally that the subsonic boundaries had the tightest requirement on time delay and rise time. All boundaries, however, limit the allowable overshoot to 25 percent.

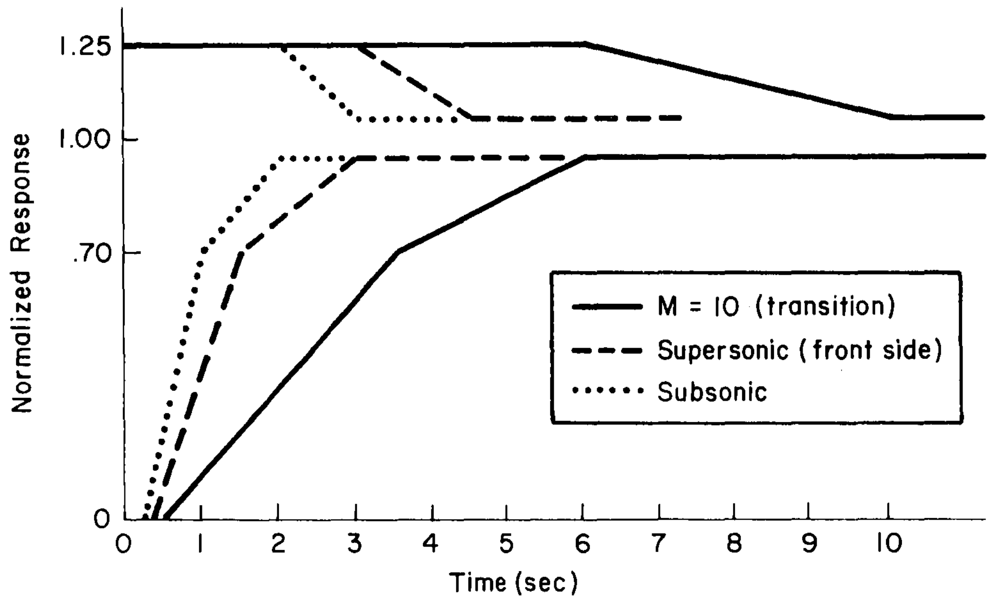


Figure II-6. 1973 Shuttle Pitch Rate Response Boundaries, Ref. 6

2. Evaluation of the Transient Response Boundary Against Recent Flying Qualities Experiments

The first assessment of the adequacy of the Shuttle time domain transient response specification was made by comparing it with data from recent flying qualities experiments. A number of experiments were reviewed and two were found to be the most useful. These were the Landing and Approach Higher Order System (LAHOS) Study of Ref. 7, and the "Neal-Smith experiment" of Ref. 8. Both of these experiments were performed using the Calspan variable stability NT-33 aircraft, and both were designed to examine the effect of higher order control system elements which produce unconventional response modes. The basic approach to evaluation was to apply the Shuttle pitch rate specification to these data to see if the specification could discriminate between good and bad flying qualities. There is a conceptual problem with this approach, however. Although the Calspan T-33 uses pitch feedbacks to augment the short period, the attitude numerator zero remains conventional $\left(1/T_{\theta_2}\right)$ since the airframe has significant static margin and there is no forward loop integrator equivalent to that in the Shuttle. Thus, in principle, if the Shuttle specification fails to distinguish good from bad LAHOS and Neal and Smith configurations, it might be because it is "not applicable" rather than a bad specification. As a practical matter, however, there are no appropriate experimental data (with pilot ratings). Furthermore, there is no evidence that the issue of the Shuttle's unconventional pitch attitude control was considered in formulating the Ref. 6 specification.

The LAHOS study was built around an approach and landing (Category C) task in which the evaluation flights were made through touchdown. A large number of configurations were evaluated, usually by two pilots, with repeat evaluations being made randomly for many of the configurations. Six LAHOS configurations were picked (Table II-1), on the basis of their basic response form, as being particularly relevant to the Shuttle flying qualities assessment.

The Neal-Smith study was performed before LAHOS and was similar in concept. However, the flight scenario was Category B (up-and-away flight) and the task was rather general with the pilots instructed to "rapidly acquire and track distant air and ground targets." It is felt that this task is analogous to the Shuttle tasks in the initial part of Terminal Area Energy Management (TAEM). Eighteen Neal-Smith configurations (Table II-2) were picked as being particularly relevant to the Shuttle. Of these, most were evaluated by two pilots with a number of repeat runs.

3. LAHOS Comparison

To assess the Shuttle pitch rate response boundary the six selected LAHOS configurations were divided into two groups. The first group, shown in Fig. II-7, consists of those configurations which exceded the Shuttle boundary. This group consists of three configurations which all have overall pilot ratings equal to or better than 3-1/2, i.e., Level 1 flying qualities. In the LAHOS experiment, pilots gave a Cooper-Harper pilot rating for the approach task alone and a second Cooper-Harper pilot rating for the overall task of approach and landing through touchdown. The pilot ratings shown are two-pilot averages.

TABLE II-1

SUMMARY OF SELECTED LAHOS CONFIGURATIONS (REF. 7)

CONFIGURATION NUMBER (Ref. 3)	NORMALIZED PITCH ATTITUDE TO STICK FORCE TRANSFER FUNCTION	OVERALL COOPER-HARPER PILOT RATING		AVERAGE
		PILOT A	PILOT B	
CONFIGURATIONS NOT MEETING THE SHUTTLE REQUIREMENT				
2-1	$\frac{7.41(.714)}{s[.57,\ 2.3]}$	2	2	2
3-C	$\frac{13.6(.714)(5.0)}{s(10.)[.25,\ 2.2]}$	2	5	3.5
4-C	$\frac{11.2(.714)(5.0)}{s(10.)[1.06,\ 2.0]}$	3	3	3
CONFIGURATIONS MEETING SHUTTLE REQUIREMENT				
4-0	$\frac{6.19(.714)}{s(1.08)(4.09)}$	6	--	6
4-3	$\frac{22.4(.714)}{s(1.42)(2.82)(4.0)}$	5.7	8	6.5
4-4	$\frac{11.2(.714)}{s(1.42)(2.0)(2.82)}$	7	6	6.5

TABLE II-2

SUMMARY OF SELECTED NEAL AND SMITH CONFIGURATIONS (REF. 8)

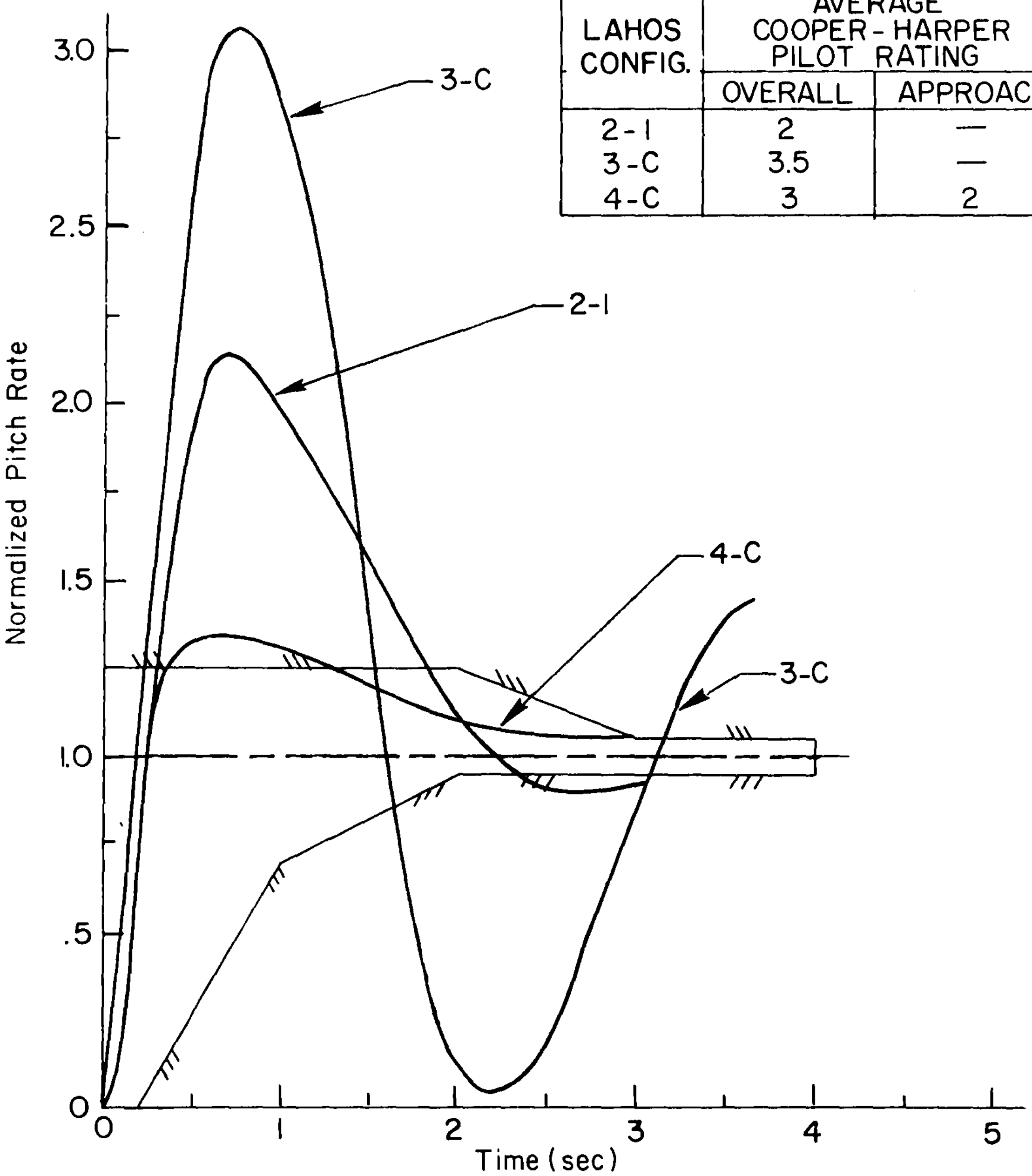
CONFIGURATION NUMBER (Ref. 2)	NORMALIZED PITCH ATTITUDE TO STICK FORCE TRANSFER FUNCTION	OVERALL COOPER-HARPER PILOT RATING		AVERAGE
		PILOT M	PILOT W	
CONFIGURATIONS NOT MEETING THE SHUTTLE REQUIREMENT				
1B	$\frac{38{,}420.(1.25)(2.0)}{s(5.0)[.69,\ 2.2][.75,\ 63.]}$	3.5	3	3.5
2D	$\frac{108{,}045.(1.25)}{s[.70,\ 4.9][.67,\ 75.]}$	3, 2.5	2.5	2.5
3C	$\frac{1{,}493{,}800.(1.25)}{s(5.0)[.63,\ 9.7][.75,\ 63.]}$	4	3	3.5
6B	$\frac{46{,}345.(2.4)(3.3)}{s(8.)[.67,\ 3.4][.75,\ 63.]}$	2.5, 1	4	2.5
7A	$\frac{213{,}644.(2.4)(3.3)}{s(8.)[.73,\ 7.3][.75,\ 63.]}$	5, 4	2	3.5
7C	$\frac{124{,}898.(2.4)}{s[.73,\ 7.3][.67,\ 75.]}$	3, 3	4, 1.5	3.0
8C	$\frac{3{,}601{,}870.(2.4)}{s(8.)[.69,\ 16.5][.75,\ 63.]}$	3.5	3	3.5
8D	$\frac{1{,}485{,}770.(2.4)}{s(3.3)[.69,\ 16.5][.75,\ 63.]}$	2	4	3
8E	$\frac{360{,}187.(2.4)}{s(0.8)[.69,\ 16.5][.75,\ 63.]}$	2.5	5	3.5

TABLE II-2 (Concluded)

CONFIGURATION NUMBER (Ref. 2)	NORMALIZED PITCH ATTITUDE TO STICK FORCE TRANSFER FUNCTION	OVERALL COOPER-HARPER PILOT RATING		AVERAGE
		PILOT M	PILOT W	
CONFIGURATIONS MEETING SHUTTLE REQUIREMENT				
1F	$\frac{30,736.(1.25)}{s(2.)[.69,\ 2.2][.75,\ 63.]}$	8	8	8
2H	$\frac{152,473.(1.25)}{s(2.0)[.70,\ 4.9][.75,\ 63.]}$	5, 6	5.5	5.5
6C	$\frac{27,094.(2.4)}{s[.67,\ 3.4][.67,\ 75.]}$	4	2.5, 5	4
6E	$\frac{63,087.(2.4)}{s(3.3)[.67,\ 3.4][.75,\ 63.]}$	5.5, 8.5	7	7
7F	$\frac{290,824.(2.4)}{s(3.3)[.73,\ 7.3][.75,\ 63.]}$	3,4, 4	7, 7	5
7G	$\frac{176,257.(2.4)}{s(2.0)[.73,\ 7.3][.75,\ 63.]}$	5	6	5.5
9	$\frac{23,798.(1.25)}{s(.748)(7.07)[.67,\ 75.]}$	5[a] 6[b]	--	
10	$\frac{23,690.(1.25)}{s(1.23)(4.28)[.67,\ 75.]}$	4[a] 4[b]	--	
11	$\frac{48,804.(1.25)}{s(2.11)(5.14)[.67,\ 7.75]}$	2.5[a] 3 [b]	--	

[a]Flown with force commands.

[b]Flown with position commands.



LAHOS CONFIG.	AVERAGE COOPER-HARPER PILOT RATING	
	OVERALL	APPROACH
2-1	2	—
3-C	3.5	—
4-C	3	2

Figure II-7. LAHOS Configurations Which Do Not Meet the Shuttle Pitch Rate Requirement

A qualification must be made for the rating of the very oscillatory configuration 3-C. Here, while the overall average pilot rating was 3-1/2, one pilot evaluated the configuration as a 2 and the other as a 5. It is felt that this inconsistency in rating arises from pilots assessing different aspects of the response and thus it cannot reliably be said that Configuration 3-C has Level 1 flying qualities. However, for the other two configurations the pilot ratings were consistent and both are Level 1. Thus, there are two configurations with good flying qualities which exceeded the Shuttle upper boundary due to their significant overshoot characteristics. A related characteristic of the responses are the relatively rapid rise times and minimal time delays.*

Figure II-8 shows the three LAHOS configurations in the second group, i.e., those that do meet the Shuttle response criterion. All have overall pilot ratings between 6 and 7, i.e., flying qualities between Level 2 and Level 3. It should be noted that Configuration 4-0 which is the most rapidly responding of the three was rated by only one pilot and thus the rating cannot be considered as reliable as those for the other two configurations. Configurations 4-3 and 4-4 can be compared to the previous group in terms of the overall group characteristics by noting that 4-3 and 4-4 have less overshoot, longer rise time and greater effective time delay.

Thus in summary the LAHOS data indicates that, for six configurations relevant to the Shuttle, two configurations which did not meet the Shuttle response criteria had good (Level 1) flying qualities and two which did meet the criterion had poor (Level 2 to 3) flying qualities.

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*Various definitions of rise time and time delay are used in the literature. In the time domain a tangent to the response curve at the steepest point is often drawn with time delay and rise time defined by the intersection of this tangent with the time axis and the steady state response level respectively.

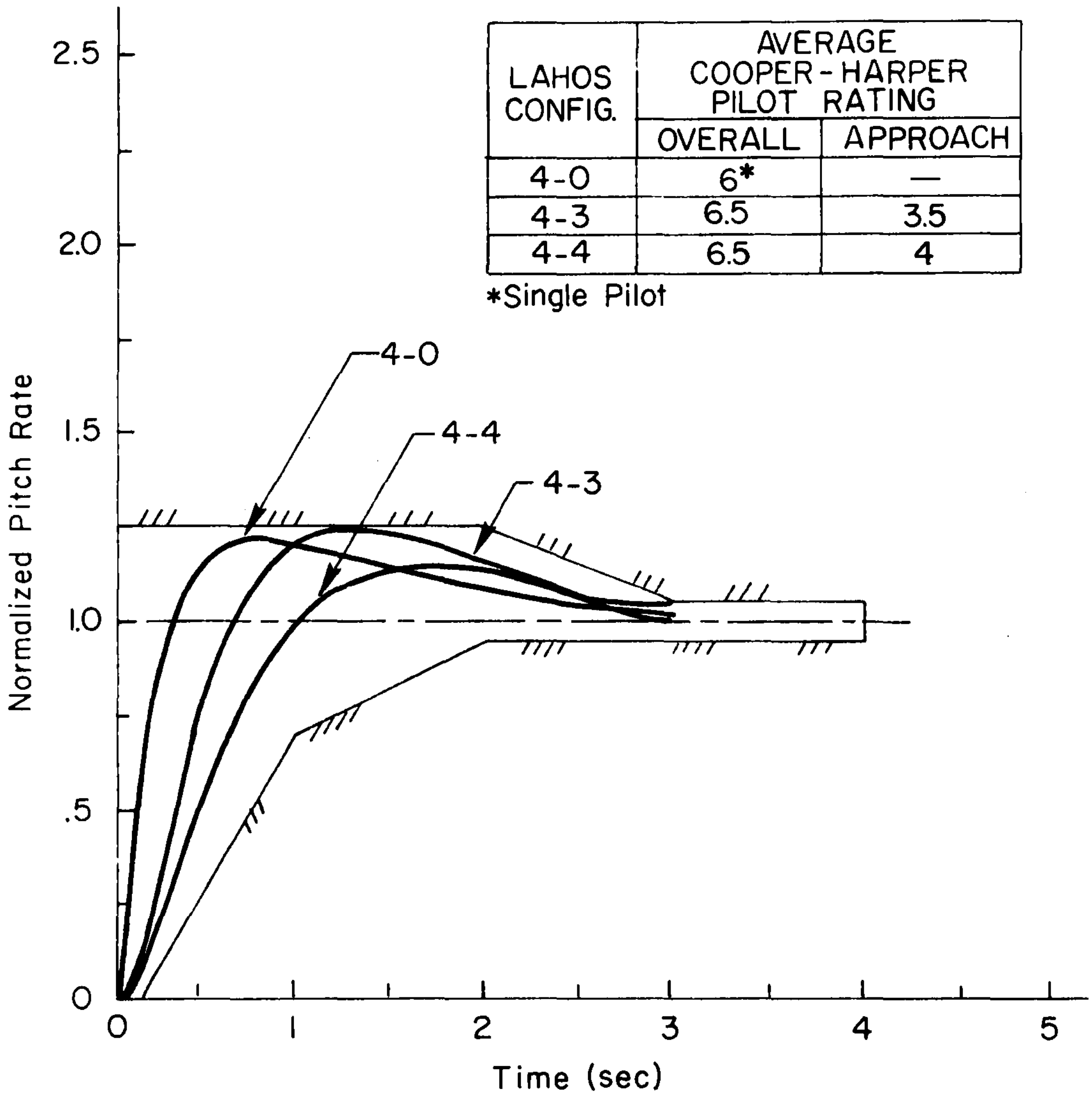


Figure II-8. LAHOS Configurations Satisfying the Shuttle Pitch Rate Requirement

4. Neal and Smith Comparison

The Neal-Smith data were used in the same way as the LAHOS data and gave similar results. Figure II-9 shows seven Neal-Smith configurations which do not meet the Shuttle pitch rate response boundary and one configuration which barely meets it. It may be seen that the average pilot rating for each of these configurations is 3-1/2 or better, i.e., Level 1 flying qualities. As a group these configurations may be characterized in general as having overshoot which exceeds the Shuttle upper boundary (one exception) and an associated rapid rise time and minimal time delay.

The second group of Neal-Smith configurations (shown in Fig. II-10) consist of those that meet the Shuttle criteria, plus two configurations, 9 and 8e, which just exceed the lower boundary. It may be seen that all these configurations have average pilot ratings of 4 or worse indicating Level 2 or in some cases Level 3 flying qualities. As a group, these configurations show minimal overshoot, greater rise time and larger effective time delay than those of Fig. II-9.

Thus the conclusion drawn from the Neal-Smith data is consistent with that from the LAHOS data; i.e., configurations with good flying qualities may be found which exceed the Shuttle pitch rate boundary, and conversely, configurations with poor flying qualities may be found which meet the criterion. The more difficult question is whether the LAHOS and Neal-Smith data, obtained in a variable stability fighter-type aircraft, is relevant to the larger Shuttle with its unconventional relationship between altitude and path dynamics.

5. Evaluation of the Transient Response Boundary Against Six Aircraft

To provide further assessment of the Shuttle pitch rate response boundaries, the responses of six aircraft in approach flight conditions were compared to the boundaries. All six aircraft had some stability augmentation incorporated in their flight control systems. The six aircraft included the Shuttle ALT and OFT configurations, the YF-12 and

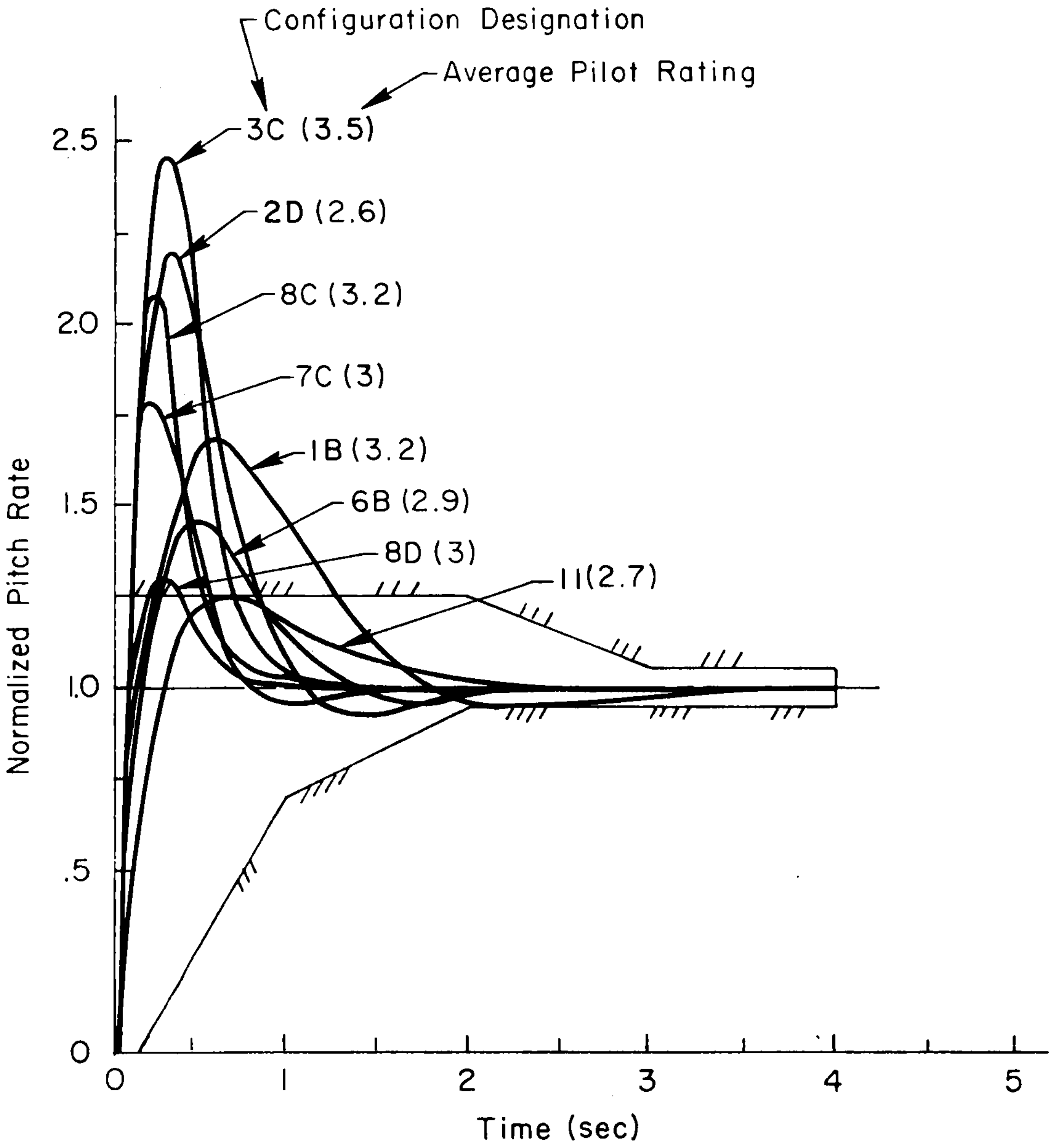


Figure II-9. Neal and Smith Configurations Which Do Not Meet the Shuttle Pitch Rate Requirement

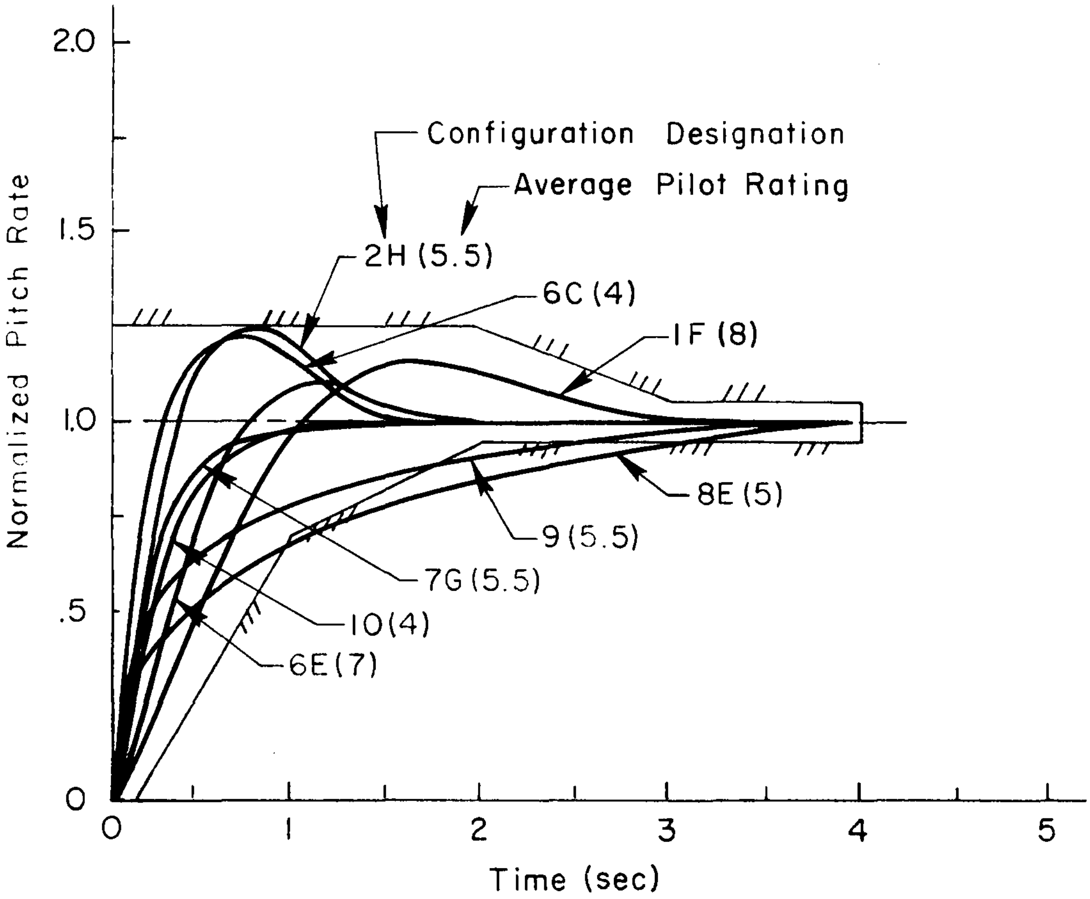


Figure II-10. Neal and Smith Configurations Satisfying the Shuttle Pitch Rate Requirement

YF-17 and two modern highly augmented aircraft refered to as "A" and "B". The comparisons for the Shuttle ALT and OFT provide an assessment of Shuttle pitch control flying qualities (as well as the Shuttle specification) relative to the other aircraft. Since the fighters A and B have advanced flight control systems they offer the chance to assess the Shuttle criteria against some unconventional aircraft.

a. <u>Shuttle ALT and OFT</u>

Figure II-11 shows a comparison of the Shuttle Approach and Landing test (ALT) configuration and the Orbital Flight Test (OFT) vehicle responses compared to the pitch rate boundaries. These responses were generated using linearized models for approach flight conditions (which are slightly different between the ALT and OFT). It may be seen that the ALT configuration satisfies the requirements and that the OFT configuration exceeds the upper limit by a small amount. Compared to the responses examined previously, the Shuttle shows somewhat lower rise time and significant effective time delay.

Because the pitch rate responses shown in Fig. II-11 were generated from a linearized model, it is useful to compare these to responses for equivalent inputs generated from a nonlinear digital simulation program (SIMEX) by Honeywell, Inc. for the Shuttle flight control analytic verification tests (Ref. 9). The Shuttle pitch rate response to a step rotational hand control (RHC) input shown in Fig. II-12 (second response from the top) is compared to the Shuttle verification criterion boundary. It may be seen that this response is quite similar to the linearized OFT response shown in the previous figure, i.e., there is an overshoot which exceeds the upper boundary with a peak at approximately two seconds, and an effective time delay of several tenths of a second. However, the elevator response (second from the bottom) shows rate limiting which contributes to the effective time delay and rise time.

b. <u>YF-12 and YF-17</u>

Figure II-13 shows the responses of two aircraft considered to have good flying qualities, the YF-12 and YF-17, compared to the Shuttle

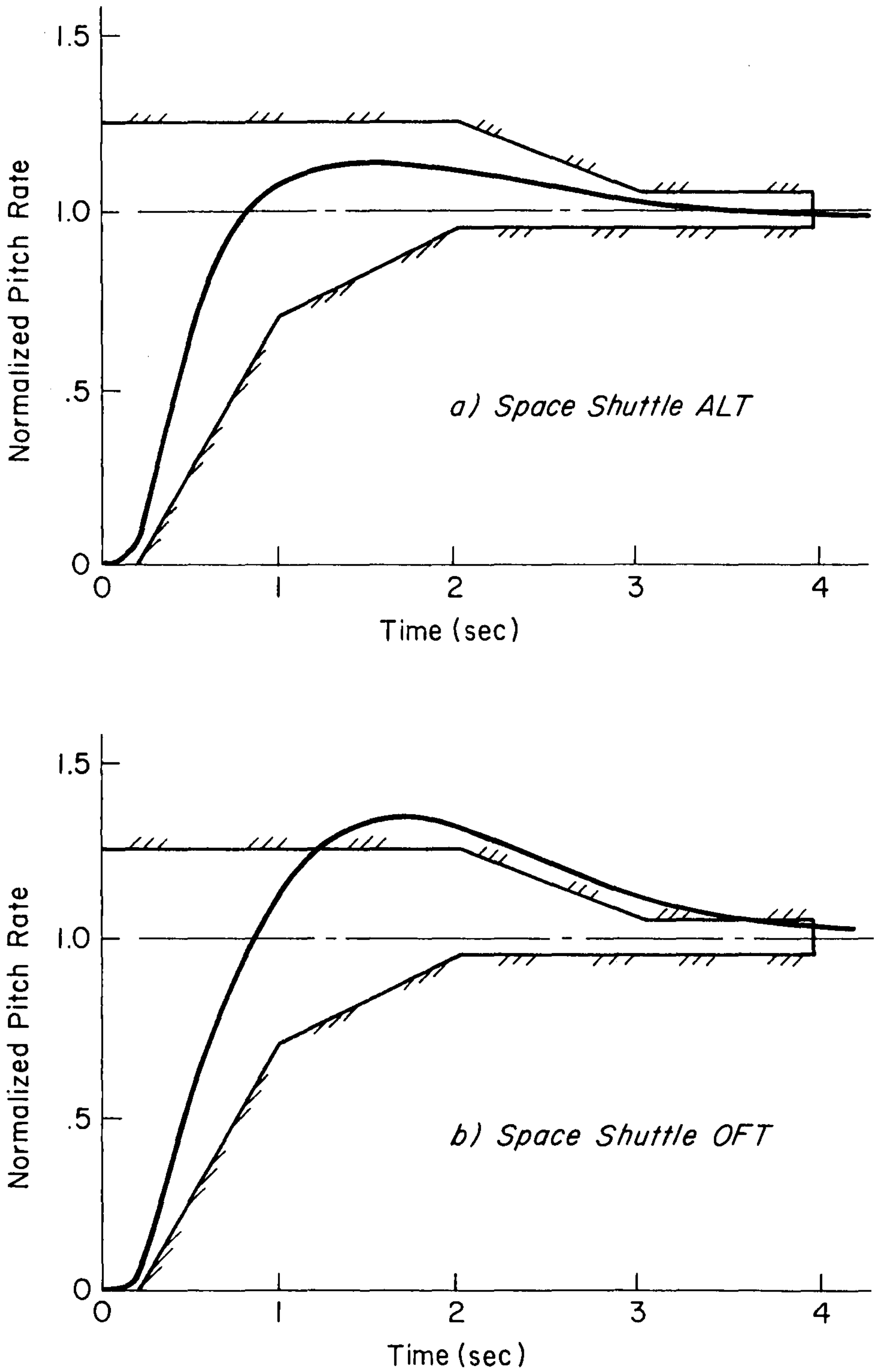


Figure II-11. Comparison of the Shuttle Pitch Rate Response to the Shuttle Requirement

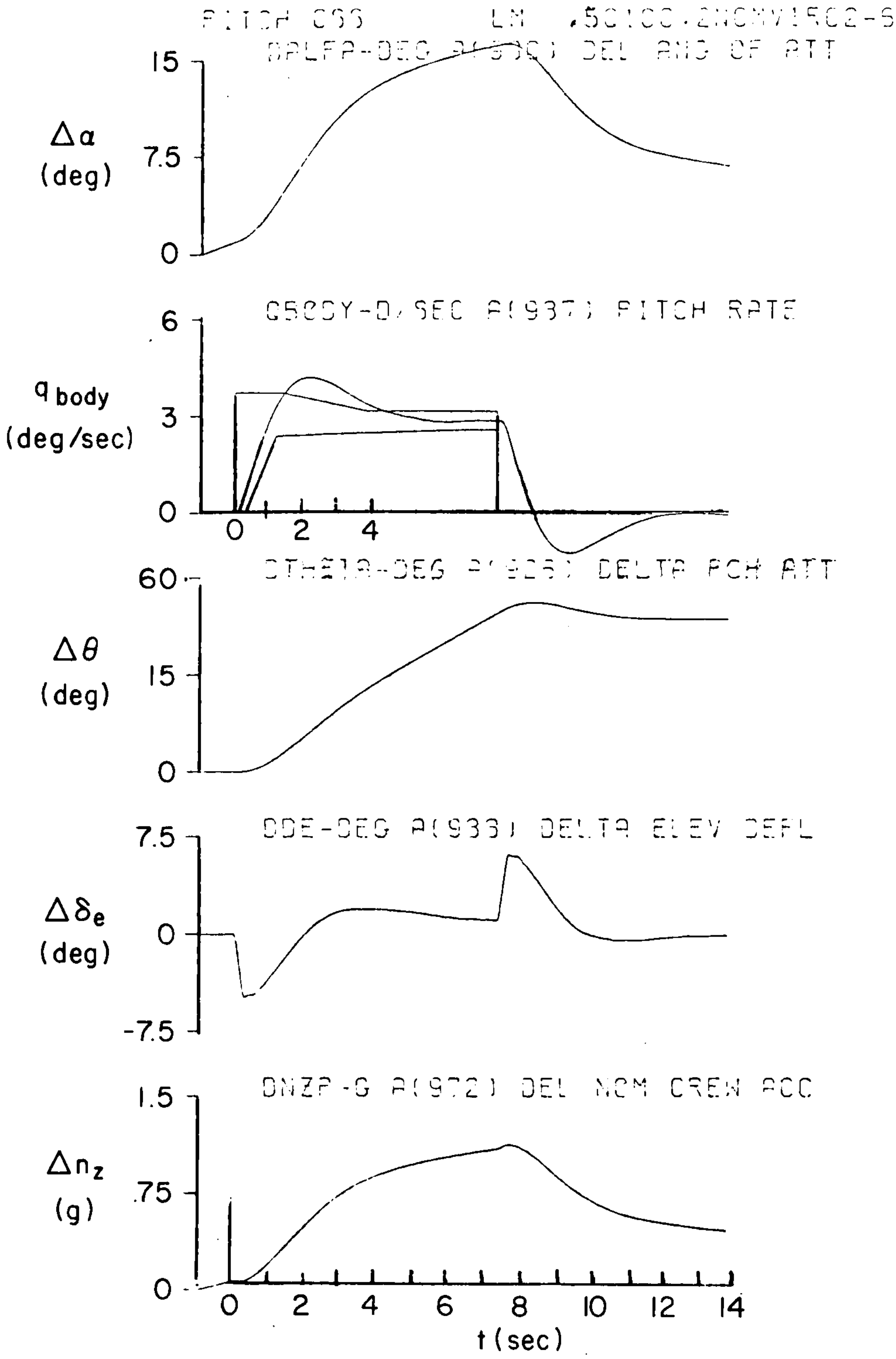


Figure II-12. Shuttle Responses from the SIMEX Nonlinear Digital Simulation, Landing Approach (Ref. 9)

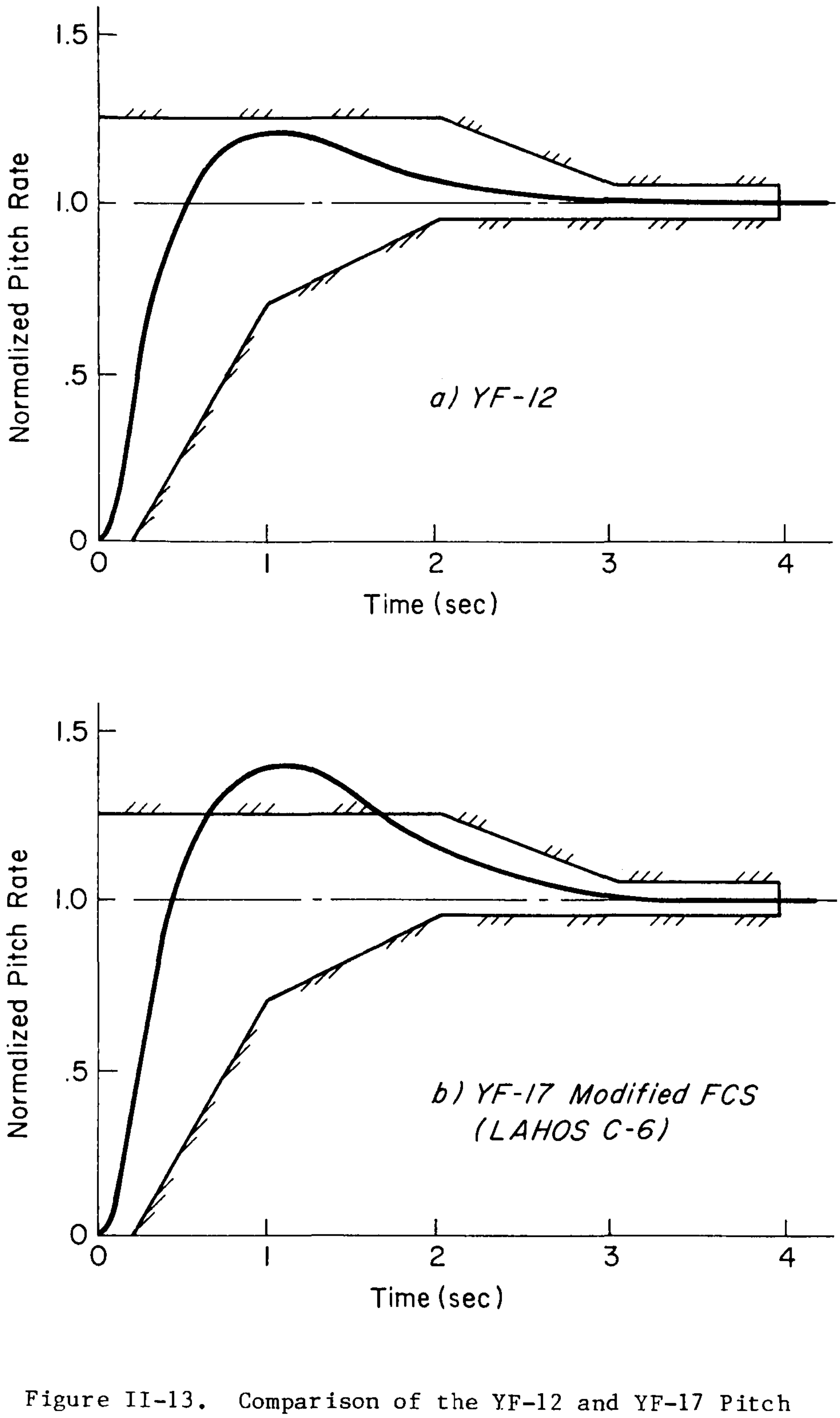


Figure II-13. Comparison of the YF-12 and YF-17 Pitch Rate Responses to the Shuttle Requirements boundary. It may be seen that the YF-12 response meets the Shuttle boundary while the YF-17 response (which was obtained from an in-flight simulation of the YF-17 modified flight control system in the LAHOS program) exceeds the upper boundary. However, these two configurations may be categorized as both having fairly rapid rise time and minimal time delay due in part to the absence of digital components in their flight control systems.

c. Modern Fighters A and B

Figure II-14 shows the comparison for two fighter aircraft known to have flying qualities problems in landing. Both aircraft exceed the upper boundary and remain above it for the specification period of 4 sec. In addition, these aircraft have larger effective time delays than the YF-12 and YF-17 shown in the previous figure. For Modern Fighter A, the source of the delay is primarily high order elements within the analog control system whereas for Modern Fighter B, a significant part of the time delay comes from computational delays in the digital flight control system. Landing problems were noted in the flight test program of Modern Fighter A, and also by experienced pilots in its initial operation; however, the plane is used operationally with changes in training procedures for new pilots. The characteristics shown for Modern Fighter B have since been modified by reducing the digital time delay.

6. Present Status of the Pitch Rate Transient Response Boundaries

A comparison of the pitch rate response criterion boundaries for the Ref. 6 1973 flying qualities specification (dashed lines) and the response specification given in the 1977 flight control system specification (Ref. 10) as shown in Fig. II-15. These latter boundaries are presently being used for Shuttle verification as in the Ref. 9 effort. The subsonic boundaries shown on previous figures correspond to the dashed boundaries in the lower figure. It may be seen that the subsonic boundary presently in use has been relaxed from these original (1973) boundaries -- in particular, there is some relaxation in the time delay

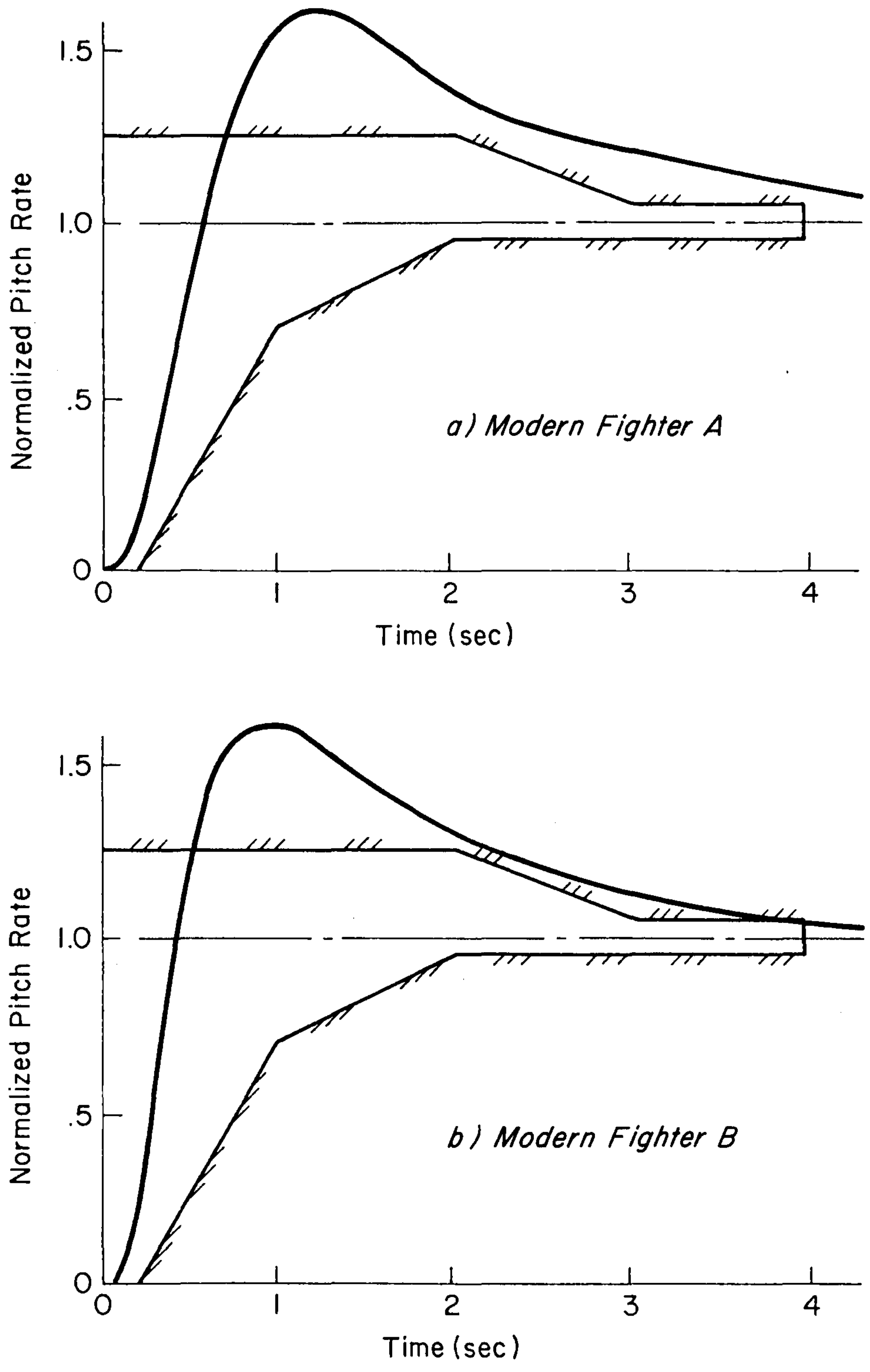


Figure II-14. Comparison of the Fighter A and B Pitch Rate Responses to the Shuttle Requirement

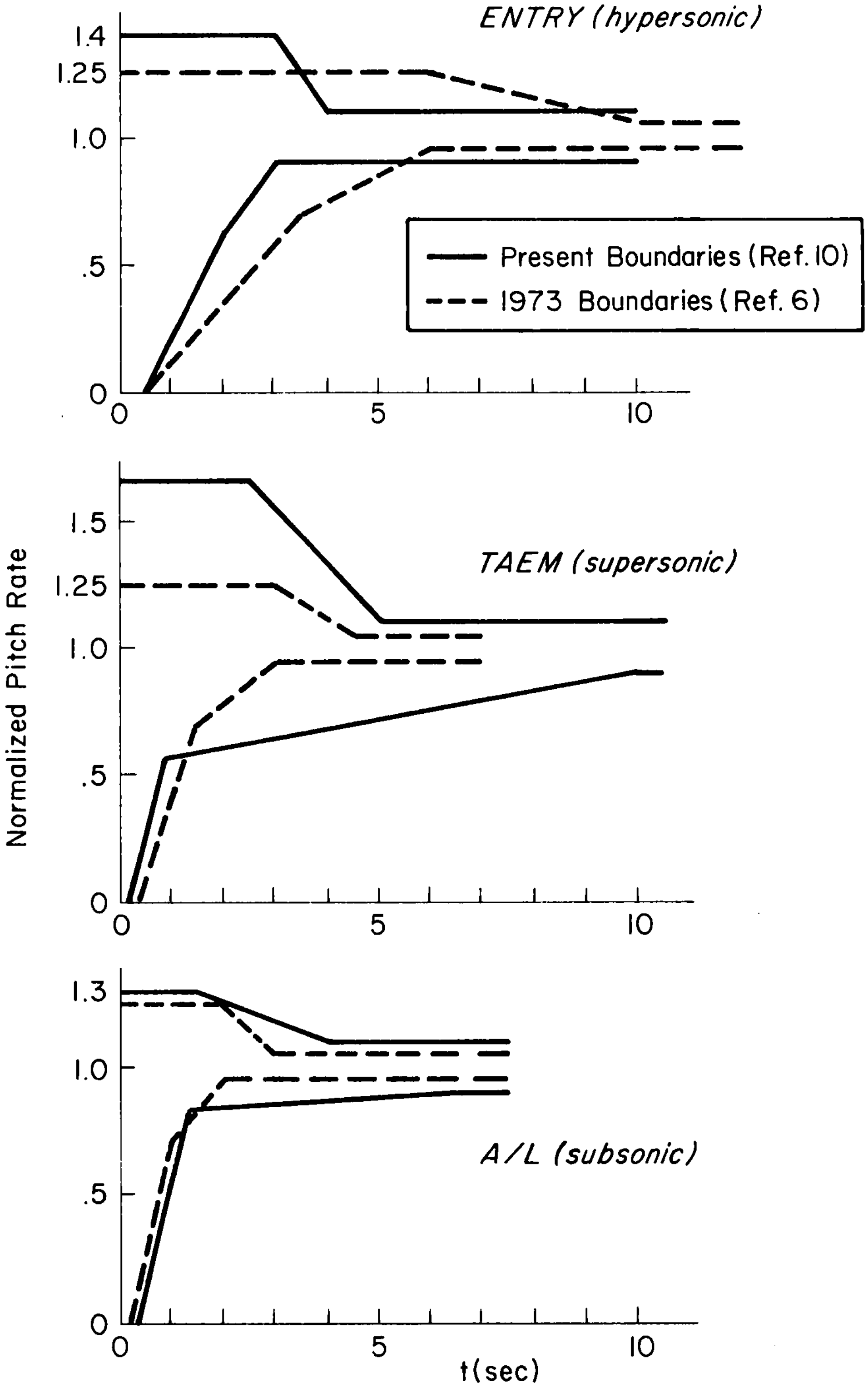


Figure II-15. Comparison of the 1973 and Present (1977) Pitch Rate Boundaries requirement and also a very small increase in the upper limit. However, these modifications would not significantly change the conclusions drawn previously.

The supersonic TAEM boundary shown in the center figure indicates a tightening of the time delay requirements (actually beyond those for the subsonic case) and a significant increase in the allowable overshoot. Finally, the hypersonic entry boundary has been revised to reduce maximum rise time without changing the maximum allowable time delay and also to increase the allowable initial overshoot.

The origins of the various boundaries are not well documented. The 1973 boundaries were drawn primarily on the basis of simulator studies made at the Johnson Space Flight Center. The revisions have occured as the Shuttle program evolves, although the detailed rationale has not been uncovered.

7. Supersonic Data

As noted previously, the data available for assessment in supersonic flight is extremely limited. However, some supersonic pitch rate responses generated with the Honeywell SIMEX nonlinear digital simulation, are available from the Ref. 9 verification study. A SIMEX time response set is shown in Fig. II-16 for the M=2.4 flight condition (nominal trajectory, Control Stick Steering pitch mode and off-nominal aerodynamics). The second time response from the top shows the pitch rate response to a step RHC input and it may be seen that the overshoot exceeds the upper boundary which, as indicated in the previous figure, has been increased significantly from the 1973 upper boundary. The effective time delay is large for this response, although the lower boundary is not exceeded. The slow initial response is partially due to the elevator rate limiting evidenced in the elevator time response trace (second from the bottom).

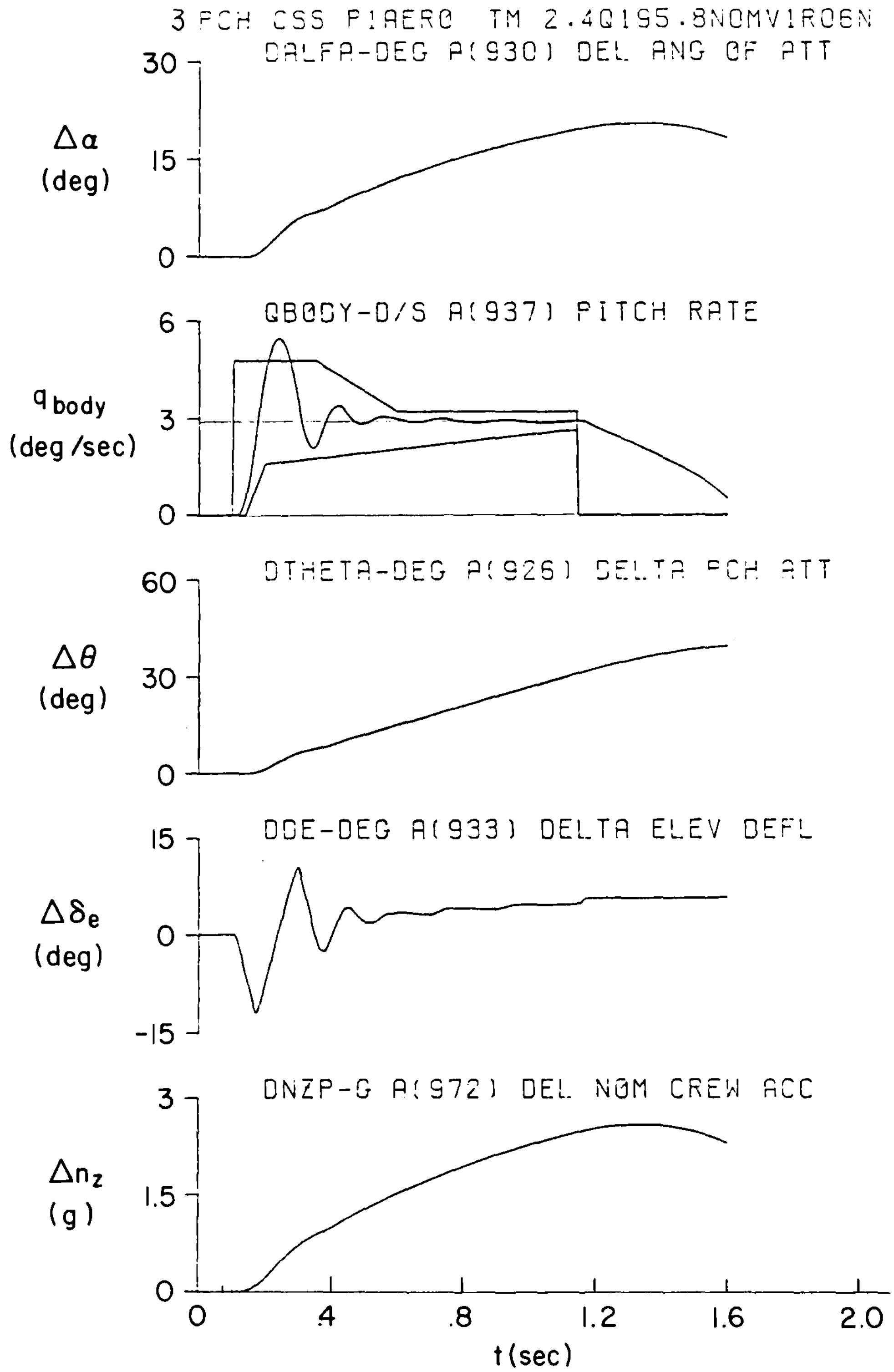


Figure II-16. Shuttle Response from the SIMEX Non-Linear Digital Simulation, M = 2.4, $\bar{q}$ = 195.8 psf (Ref. 9)

8. Summary

The preceding review has indicated a potential problem for the Shuttle time domain pitch rate boundary — namely that it may allow configurations with poor flying qualities while possibly excluding some with good flying qualities. In particular, it may unduly restrict overshoot but not sufficiently restrict effective time delay — at least in the subsonic region. However, these conclusions must be qualified in light of the Shuttle's unconventional attitude/path dynamics. Finally, further data are needed for proper validation in the supersonic regime.

E. LOWER ORDER EQUIVALENT SYSTEM MODELS FOR FLYING QUALITIES SPECIFICATION

1. Alternative Specifications

In addition to the problems noted above, the time domain transient response specifications have other problems. For instance, graphical determination of "effective time delay" values pertinent for piloted control from a time history is more difficult practically than from a frequency response (Bode) plot (because of the difficulty in accurately defining the steepest tangent to a step response). A more fundamental problem with the Shuttle specification is that it considers pitch attitude control — only without explicit regard for flight path control.

These considerations led to an examination of other forms of specification. This effort began with the present U.S. Military specification, MIL-F-8785C, (Ref. 11) not because it was considered to represent a superior approach but rather because it is probably the most widely used and well established flying qualities specification. It therefore codifies much of the specification data and lore of flying qualities research and concepts.

Following the review of the MIL-Spec a number of other specification forms were considered. Many of these specifications are formulated in the frequency domain and involve use of the lower-order equivalent system concept. Thus a discussion of the frequency domain treatment of higher order systems (HOS) using the lower order equivalent systems (LOES) modeling approach is given in the next section.

2. Considerations in LOES Modeling

The use of lower order equivalent system models has been a topic of considerable interest in flying qualities research in recent years (Refs. 12 and 13). LOES models are formulated by fitting a low order form to a high order transfer function numerically with a digital computer program. The LOES form is specified a priori with variable parameters to be adjusted to a best fit of the HOS. The LOES form is generally taken as that for a classical unaugmented airframe with the idea that at least some of the flying qualities data accumulated over the years for conventional aircraft could thereby be extended to highly augmented aircraft. By extension of this reasoning, the flying qualities specification formats developed for conventional aircraft could also be used for some highly augmented aircraft.

For longitudinal pitch attitude control, the LOES model is usually based on the short period approximation, Eq. 3. This lead to a controversy over whether $(1/T_{\theta_2})$ should be allowed to vary in the numerical fitting process (the "galloping L_α" issue) or be fixed at the classical value of $1/T_{\theta_2} \doteq -Z_w$. The generally accepted present view seems to be that noted in Ref. 13 — namely that the pitch attitude and path angle HOS transfer functions should be fitted simultaneously. For fairly conventional aircraft this produces a $1/T_{\theta_2}$ near the classical airframe value.

This LOES modeling concept based on numerical fitting to a "classical" form is fundamentally different than the analytical approach used in Subsection C to derive the literal approximation to $\dot{\theta}/\dot{\theta}_c$ given in Eq. 9. Instead, the analytical form evolves as part of the approximation process. In particular it should be noted that the significant difference between $1/T_{\theta_A} \doteq \omega_e = 1.5 \text{ sec}^{-1}$ and the inverse time constant of h_p/θ, i.e., $1/T_{\theta_2} = .54 \text{ sec}^{-1}$ found in Subsection C would not occur with the use of the numerical LOES modeling procedures noted above.

These considerations were important in formulating an approach to the use of LOES modeling in this study. While the analytical modeling of Subsection C gives an accurate and insightful representation of the

Shuttle's augmented pitch dynamics there is a further consideration -- namely that the form of the LOES model must be reconciled not only with the augmented vehicle being considered but also with the data base to be used. Thus, given a data base not entirely appropriate to the Shuttle, the best that can be done is to use a conventional model which best approximates the unconventional Shuttle — but there is no guarantee that this approach will be adequate.

3. Summary of LOES Models

Several approaches were considered in formulating LOES models in this study. Since the Shuttle and some other highly augmented aircraft have "rate command" pitch control systems, the HOS transfer functions were first fitted with the "rate command" form shown in Eq. 16 below.

$$\frac{\theta}{\delta_s} = \frac{A_\theta e^{-\tau s}}{s(T_1 s + 1)} \tag{16}$$

It was later found that the short-period approximation (Eq. 3) gave equal or better fits when made with $1/T_{\theta_2}$ constrained to the airframe value ($\doteq Zw$). A summary of LOES models developed in this study is given in Table II-3. The steady state gains of the LOES models have been normalized to unity to be consistent with the form of the Shuttle pitch rate specification.

4. Numerical Methods and an Example

The LOES approximations were obtained by use of the STI Multi-Frequency Parameter Identification (MFP) computer program. In this program a numerical search routine is used to minimize a cost function related to the difference between the HOS and LOES models at 15 discrete frequency, ω, points (0.1 rad/sec $\leq \omega \leq 10.0$ rad/ sec). The cost function, C, is defined as:

TABLE II-3. SUMMARY OF PITCH ATTITUDE CHARACTERISTICS FOR AUGMENTED AIRCRAFT

AIRCRAFT	HIGH-ORDER PITCH ATTITUDE TRANSFER FUNCTIONS	LOWER-ORDER EQUIVALENT SYSTEM TRANSFER FUNCTIONS	
		RATE COMMAND	SHORT PERIOD
Space Shuttle ALT (Approach) (Ref. 4)	$\frac{\theta}{\delta_s} = \frac{.114 \times 10^6(.042)(.72)(1.5)(1.8)(50.)[.02,32.75]}{s(.041)(.87)(10)(51.2)[.80,1.40][.99,7.67][.49,21.][.71,36.4]}$	$\frac{1.27e^{-.232s}}{s(2.90)}$	$\frac{1.17(.72)e^{-.219s}}{s[.952,1.48]}$
Space Shuttle OFT (Approach) (Ref. 4)	$\frac{\theta}{\delta_s} = \frac{1.50 \times 10^5(.036)(.537)(.590)(1.50)(-42.9)[.02,32.75]}{s(.031)(14.2)[.97,.620][.63,1.59][.39,20.6][.68,37.1][.99,50.2]}$	$\frac{.371e^{-.227s}}{s(1.905)}$	$\frac{.344(.54)e^{-.213s}}{s[.728,1.104)}$
YF-12 (Approach) (Ref. 4)	$\frac{\theta}{\delta_s} = \frac{7726.(.0376)(.79)(4.)[.047,9.65][.7,50.5]}{(1.13)(28.6)(38.3)[.28,.054][.796,3.10][.05,20.][.744,39.6]}$	$\frac{2.91e^{-.093s}}{s(3.5)}$	$\frac{1.69(.79)e^{-.025s}}{s[.746,1.36]}$
Neal and Smith Configuration 10 Calspan T-33 (Up and Away) (Ref. 2)	$\frac{\theta}{\delta_s} \doteq \frac{23690.(1.25)e^{-.083s}}{s(1.23)(4.28)[.67,75.]}$ Time delay approximation to feel system and actuator	$\frac{4.03e^{-.097s}}{s(4.03)}$	—
YF-17/LAHOS Configuration 6-2 Calspan T-33 (Approach and Landings (Ref. 3)	$\frac{\theta}{\delta_s} \doteq \frac{.5864(.714)(2.0)(2.33)(16.7)}{s(.91)(5.)(10.)[.65,1.9]} e^{-.065s}$ Time delay as above	$\frac{4.80e^{-.106s}}{s(4.08)}$	$\frac{4.04(.714)e^{-.080s}}{s[.827,1.728]}$
Modern Fighter A (Approach)	$\frac{\theta}{F} = \frac{.4336 \times 10^8(.0481)(.490)(1.0)(5.0)(10.0)(12.)(50.)}{(8.3)(9.8)(45.9)(145.5)[.0307,.202][.767,.80]} \times [.561,3.05][.842,16.5][.751,73.8]$	—	$\frac{.72(.49)3^{-.219s}}{s(.854)(2.24)}$
Modern Fighter B (Approach)	$\frac{\theta}{F} = \frac{.320(.175)(.394)[.387,.702](1.0)^2(2.31)(2.5)(2.9)(5.88)(10.)(20.)}{(.002)(.115)[.412,.737](.794)[.858,.796](2.42)(2.87)} \times (4.13)[.774,4.22](5.86)(10.55)(16.8)$	—	$\frac{.0458(.435)e^{-.187s}}{s(.994)(2.27)}$

$$C = \sum_{k=1}^{15} w_k^2(j\omega_k)\left|G_{HOS}(j\omega_k) - G_{LOES}(j\omega_k)\right|^2 \tag{17}$$

where

ω_k is the kth matching frequency

w_k is the kth weighting factor

$G_{HOS}(s)$ is the HOS transfer function

$G_{LOES}(s)$ is the LOES transfer function

Some experimentation with the weighting factors was required to produce satisfactory fits in the frequency domain of interest. Initially all matching frequencies were weighted equally, which was found to give matches that diverged with increasing frequency. This is because, as the response amplitude decreases with frequency, the HOS/LOES difference vector tends to decrease in magnitude as may be seen in Fig. II-17. This problem was solved by using

$$\begin{aligned} w_{k_{dB}} &= 20 \log G_{HOS}^{-1}(j\omega_k) \\ &= - G_{HOS}(j\omega_k)_{dB} \end{aligned} \tag{18}$$

which essentially imposes the same penalty on magnitude (in dB) mismatches at all frequencies.

Figure II-18 shows a frequency domain comparison of the high and lower order (short period form) systems for the OFT transfer functions.

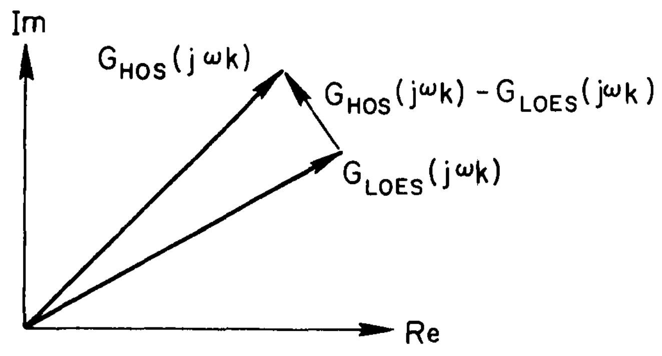


Figure II-17. Definition of a Cost Function at a Frequency ω_k

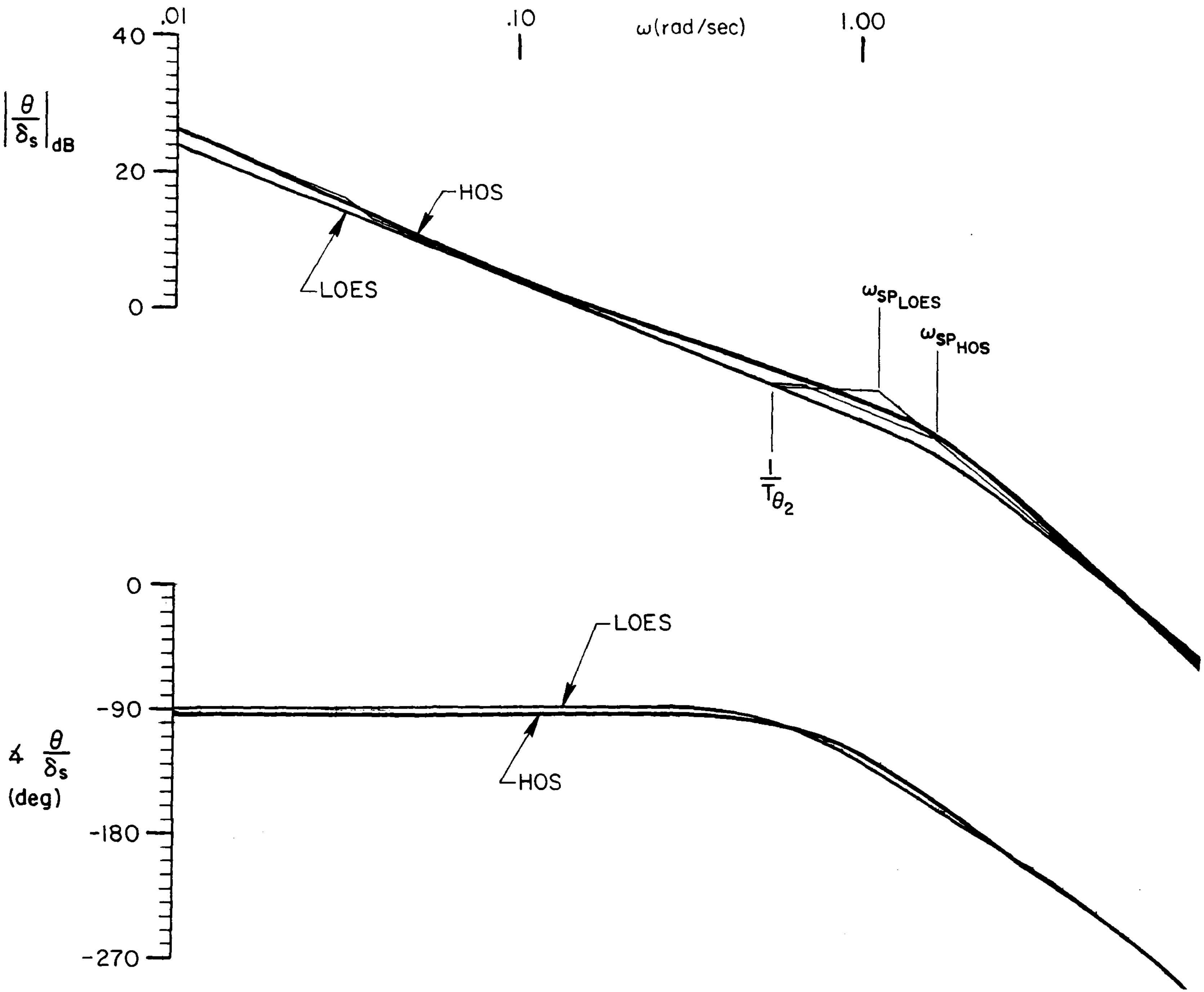


Figure II-18. Frequency Domain Comparison of HOS and LOES Models of Shuttle OFT

The comparison shows negligible differences between the higher and lower order systems in the fitting region. However, the equivalent short period frequency has been reduced from the HOS value of 1.6 rad/sec to a LOES value of approximately 1.1 rad/sec.

Figure II-19 shows the time domain comparison between the higher and lower order system responses for the OFT. The lower order system shows somewhat higher response from 1 sec to approximately 4-1/2 sec. However, the initial responses are essentially identical down to the time delay region.

F. COMPARISONS WITH THE US MILITARY FLYING QUALITIES SPECIFICATION

The present US military flying qualities specification, MIL-F-8785C, Ref. 11 contains three requirements which are conceivably relevant to Shuttle pitch control: "Short-period frequency and acceleration sensitivity" (Section 3.2.2.1.1); "Short-period damping" (Section 3.2.2.1.2); and "Dynamic characteristics" (Section 3.5.3, Table XIV).

1. The Control Anticipation Parameter

The short-period frequency and acceleration sensitivity requirement relates pitch attitude and path (normal acceleration) response through the Control Anticipation Parameter, CAP (Ref. 14). CAP was developed by Birhle, Ref. 15, from evidence that for longitudinal maneuvering pilots were initially concerned with pitch acceleration but ultimately with steady state load factor. Steady state in this context means after the short period transient but before the phugoid response. Thus Birhle defined CAP as

$$\mathrm{CAP} \equiv \frac{\ddot{\theta}(0)}{n_{ss}} \tag{19}$$

Using the short-period approximation and assuming $Z_{\delta_e} \doteq 0$.

$$\mathrm{CAP} = \frac{\ddot{\theta}(0)}{\frac{V_{T_o}}{g}\dot{\gamma}_{ss}} = \frac{\ddot{\theta}(0)}{\frac{V_{T_o}}{g}\dot{\theta}_{ss}} \tag{20}$$

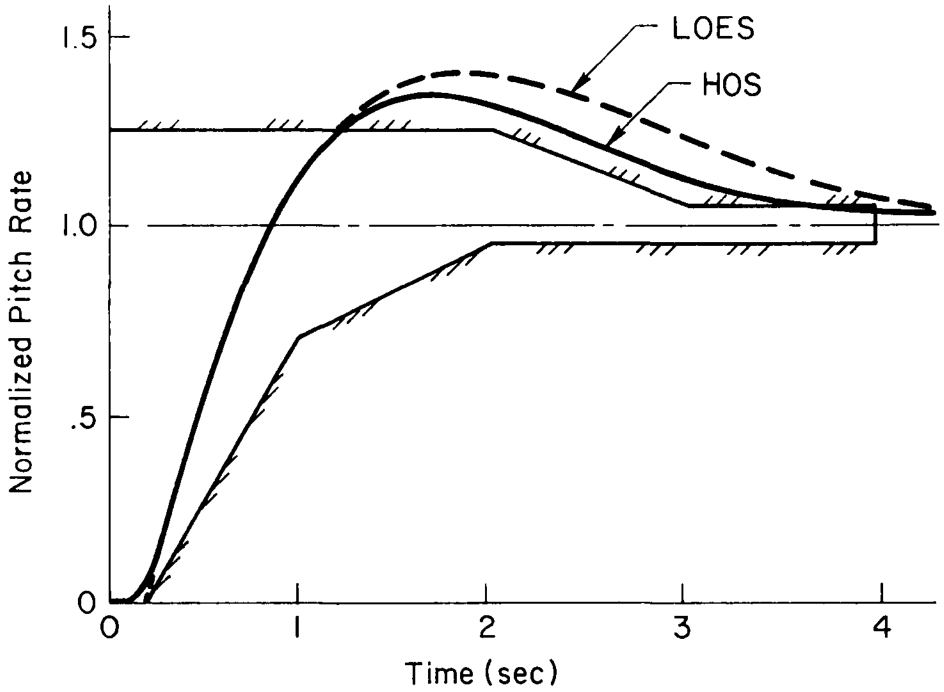


Figure II-19. Time Domain Comparison of HOS and LOES Models of Shuttle OFT

from which it may be seen that CAP is proportional to the ratio of the high and low frequency gains of the short period $\dot{\theta}/\delta_e$ (Eq. 3), i.e.,

$$\text{CAP} = \frac{A_\theta}{\dfrac{V_{T_o}}{g} A_\theta \dfrac{\frac{1}{T_{\theta_2}}}{\omega_{SP}^2}} = \frac{\omega_{SP}^2}{\dfrac{V_{T_o}}{g} \dfrac{1}{T_{\theta_2}}} \tag{21}$$

where $1/T_{\theta_2} \doteq -Z_w$. Since

$$\frac{n_{ss}}{\alpha} = \frac{V_{T_o} \dot{\gamma}_{ss}}{g\alpha} \doteq \frac{-Z_\alpha}{g} \doteq \frac{V_{T_o}(-Z_w)}{g} \doteq \frac{V_{T_o}}{g} \frac{1}{T_{\theta_2}} \tag{22}$$

$$\text{CAP} = \frac{\omega_{SP}^2}{n/\alpha} \tag{23}$$

2. Short Period Frequency and Acceleration Sensitivity

The Shuttle was compared to the MIL-F-8785C Category C short period requirements on two bases. First, in terms of the HOS short period characteristics as is shown in Fig. II-20 and secondly in terms of the LOES characteristics as will be shown in Fig. II-21. In addition to the Shuttle response shown in Fig. II-20, the responses of a number of other highly augmented aircraft are also shown. It may be seen that all aircraft shown fall within the Level 1 boundaries for Category C flight. The Shuttle OFT and ALT configurations are comparable to the B-70 bomber (Ref. 16) and somewhat lower in frequency compared to the other smaller aircraft as a consequence of their lower pitch moment of inertia.

In Fig. II-21 the Shuttle OFT and ALT, the YF-12 and YF-17 are compared with the MIL-F-8758C short period requirement on the basis of their LOES characteristics. It may be seen that while these aircraft are in somewhat different positions with respect to the previous figure, all four aircraft are still Level 1. For an additional comparison, three large aircraft with more conventional flight control systems: the C5A, (Ref. 16) the Concorde (with damper off, Ref. 3), and the Boeing 747 (Ref. 16) are also shown. As a group, these larger aircraft have reduced short period frequency and are borderline Level 1/Level 2 (or even Level 3 in the case of the Concorde). However, these aircraft are not generally considered to have flying qualities problems associated with short period characteristics. Thus, on the basis of this comparison there is no evidence to indicate a problem with the short period characteristics of the Shuttle as defined by the existing military specification. However, for supersonic flight conditions, the data both for assessment and the data base on which the original specification was founded are very limited and thus further data would be needed for good substantiation.

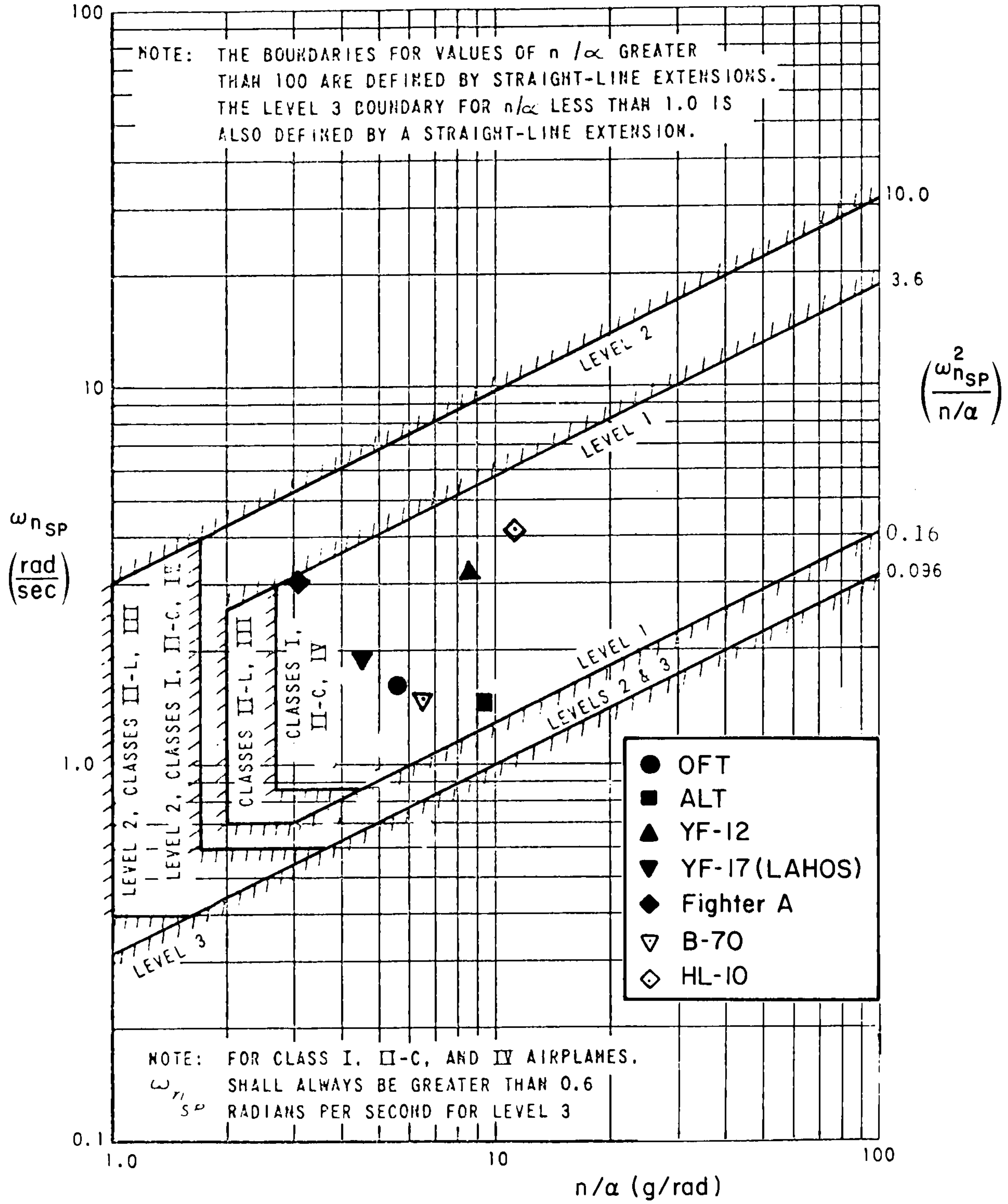


Figure II-20. Comparison of Aircraft to MIL-F-8785C Short-Period Frequency Requirements - Category C Based on HOS Short Period

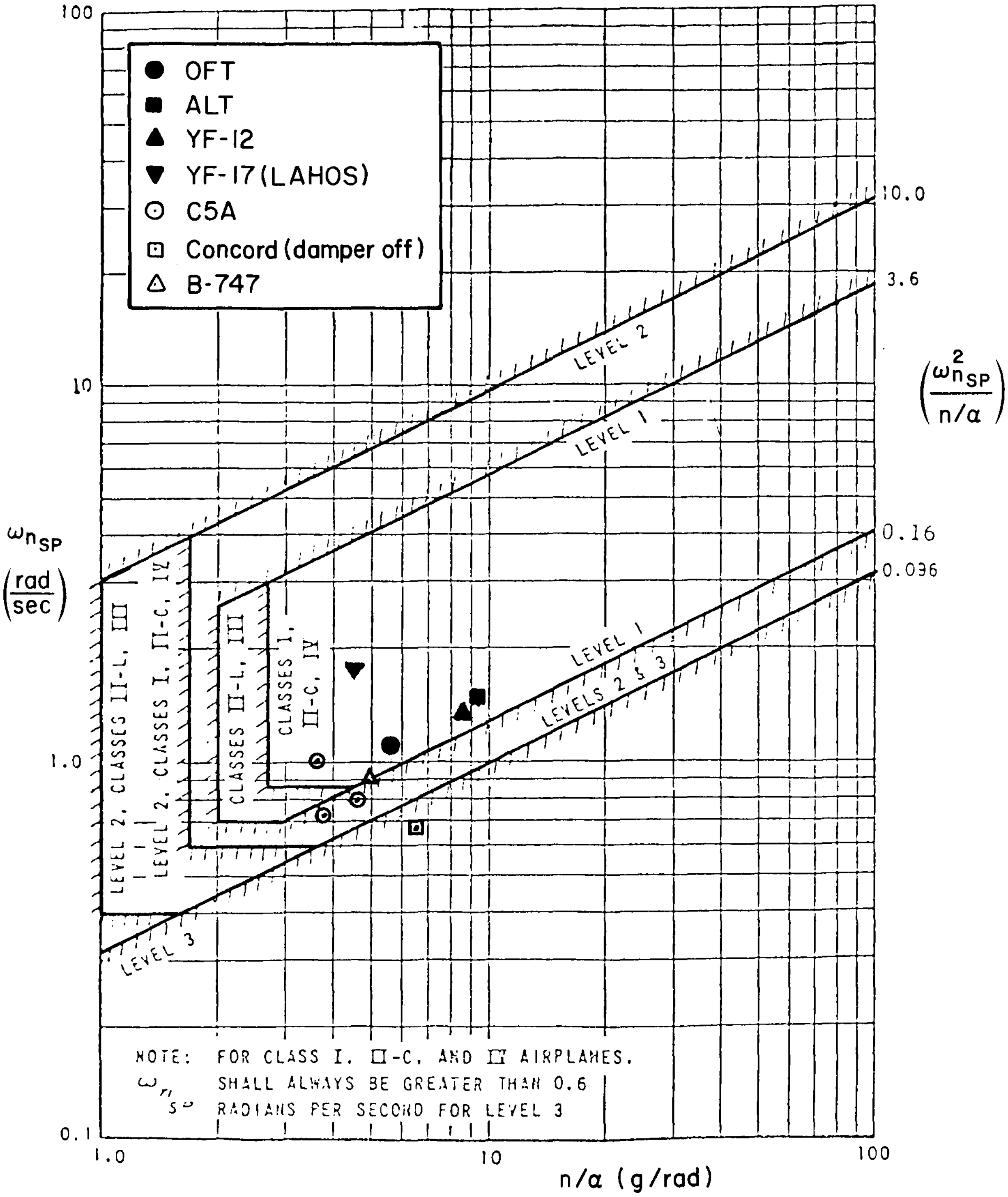


Figure II-21. Comparison of Aircraft to MIL-F-8785C Short-Period Frequency Requirements - Category C Based on LOES Short Period

3. Short Period Damping

The Shuttle and the other aircraft examined in the previous subsection easily meet the MIL-F-8785C short period damping requirements and, consequently, this specification is not considered further.

4. Effective Time Delay ("Dynamic Characteristics")

Figure II-22 shows the effective time delay (τ) compared to the time delay requirements of MIL-F-8785C for the Shuttle ALT and OFT and four aircraft examined previously. It may be readily seen that the two aircraft earlier identified as having good flying qualities, the YF-12 and YF-17, have the lowest effective time delay and both are well within the Level 1 requirements of 8785C. The Shuttle configurations both have much larger time delays which fall into the Level 3 region and are comparable to Modern Fighters A and B which were earlier noted as having landing problems.

A comparison of the MIL-F-8785C time delay requirements with data from an experiment on the NASA-Dryden F-8 aircraft (Ref. 18) is shown in Fig. II-23. The F-8 data indicates a threshhold on pilot rating degradation due to time delay above which pilot rating is a function of the stressfulness of the task. The origins of the time delay requirements of MIL-F-8785C, are not well documented and the comparison shown here indicates that they may be overly restrictive.

The conclusions of this examination are that the Shuttle is Level 3 on MIL-F-8785C and that the effective time delay of the Shuttle is comparable to several aircraft with known flying qualities problems in landing. However, a question remains concerning the validity of the criterion for large aircraft and higher altitudes typical of the Shuttle in TAEM.

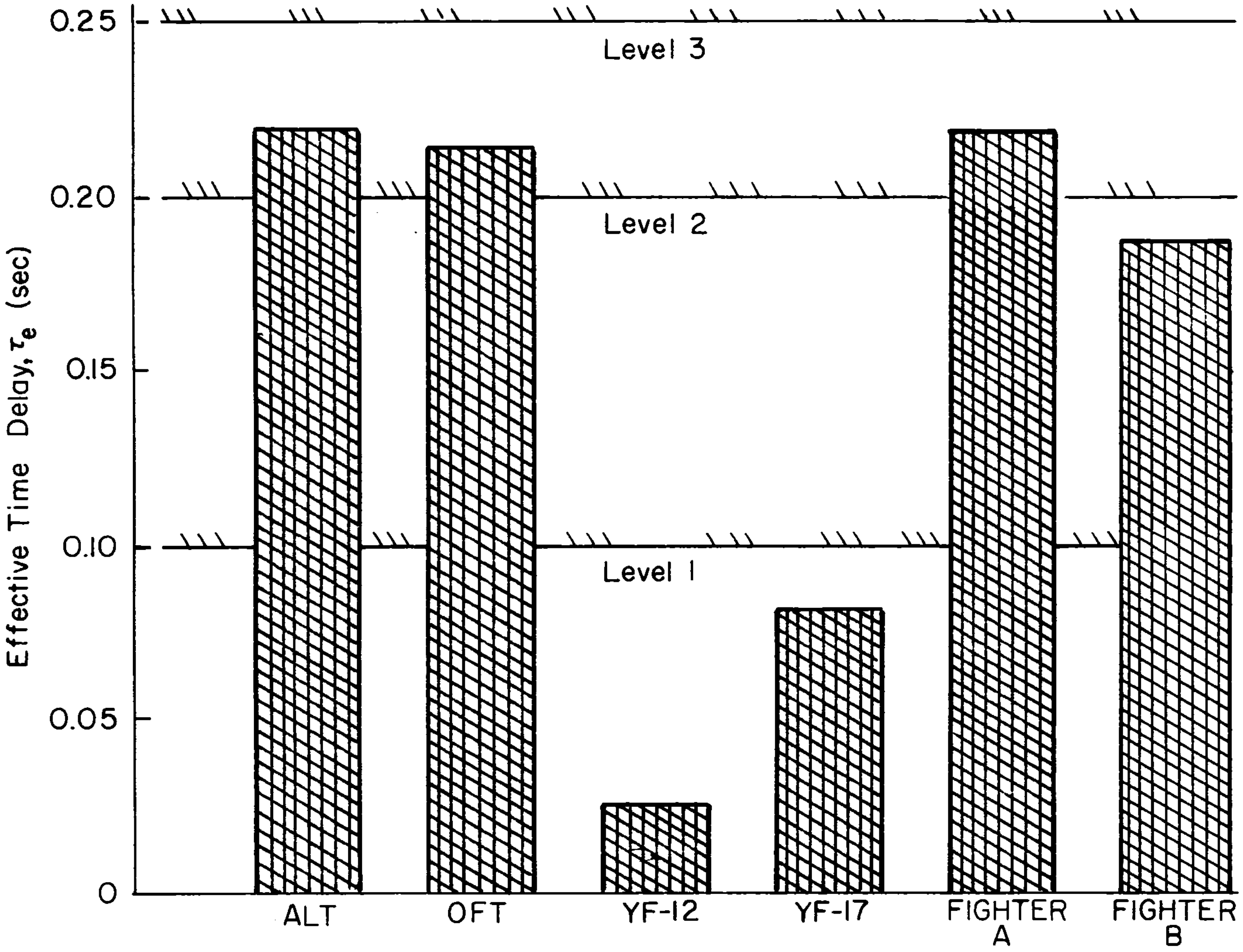


Figure II-22. Comparison of 6 Aircraft with MIL-F-8785C Time Delay Requirement (based on short-period LOES)

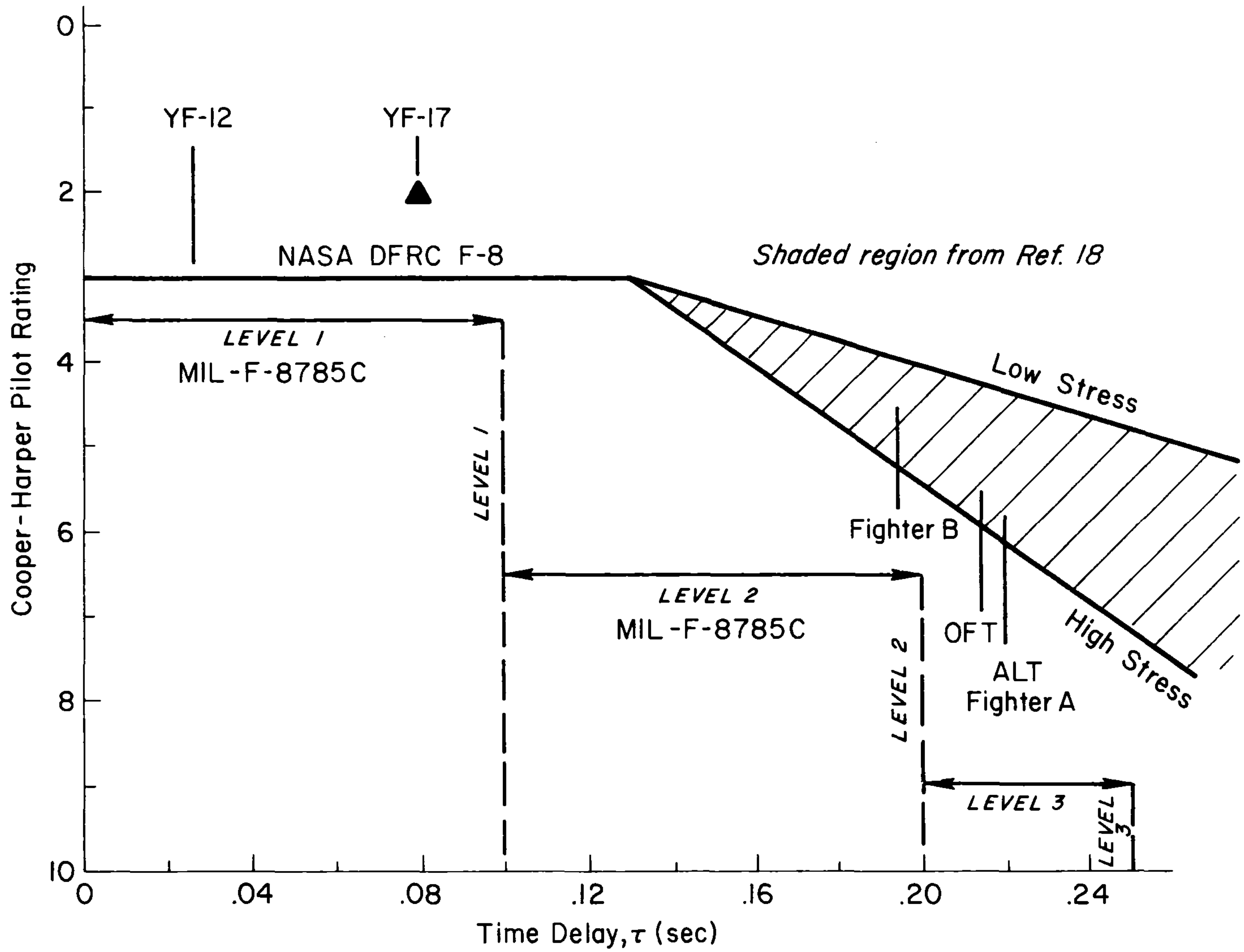


Figure II-23. Comparison of MIL-F-8785C Time Delay Requirements with Data from Ref. 18

G. COMPARISONS WITH OTHER PROPOSED SPECIFICATIONS

1. Bandwidth/Time Delay Criterion

A new specification being considered for the Military Standard, in progress at Systems Technology, Inc. (STI) (Ref. 19) is shown in Fig. II-24. In this criterion, the effective time delay, determined from a LOES model, is plotted against bandwidth — in this case, the pitch attitude bandwidth — defined as the maximum frequency with at least a 6 dB gain margin and at least a 45 deg phase margin. The bandwidth is computed including the effect of time delay. The OFT and ALT are shown along with the aircraft previously examined and it may be seen that the results are similar to the MIL Spec time delay comparison, i.e., the Shuttle OFT and ALT configurations and the Modern Fighter A are Level 3. The Modern Fighter B is again Level 2, however the YF-17 is Level 2 and the YF-12 is borderline Level 1/Level 2 whereas these latter two aircraft are both Level 1 in terms of MIL-F-8785C.

2. Supersonic Cruise Research Vehicle Specification

Figure II-25 shows the application, to the Shuttle ALT, of the pitch attitude response specification developed for the Supersonic Cruise Research Vehicle by Chalk, Ref. 17. Chalk's criteria consists of three separate requirements. First, there is a limit on the peak ratio, such that $\Delta q_2/\Delta q_1$ is less than or equal to 0.30 for Level 1 flying qualities. Response times t_1 and t_2 are defined by drawing a tangent to the curve at the steepest slope, defining t_1 as the intersection of the tangent with the time axis, and defining t_2 as the time of intersection of the tangent with the steady state level (1.0). Rise time is defined as $t_2 - t_1$ and required to be less than 0.48 seconds for Level 1. The time delay is defined to be T_1 and is required to be less than 0.12 sec for Level 1. On the basis of these requirements the ALT would be Level 1 in terms of peak ratio and rise time but would be Level 3 ($t_1 > 0.21$ sec) in time delay.

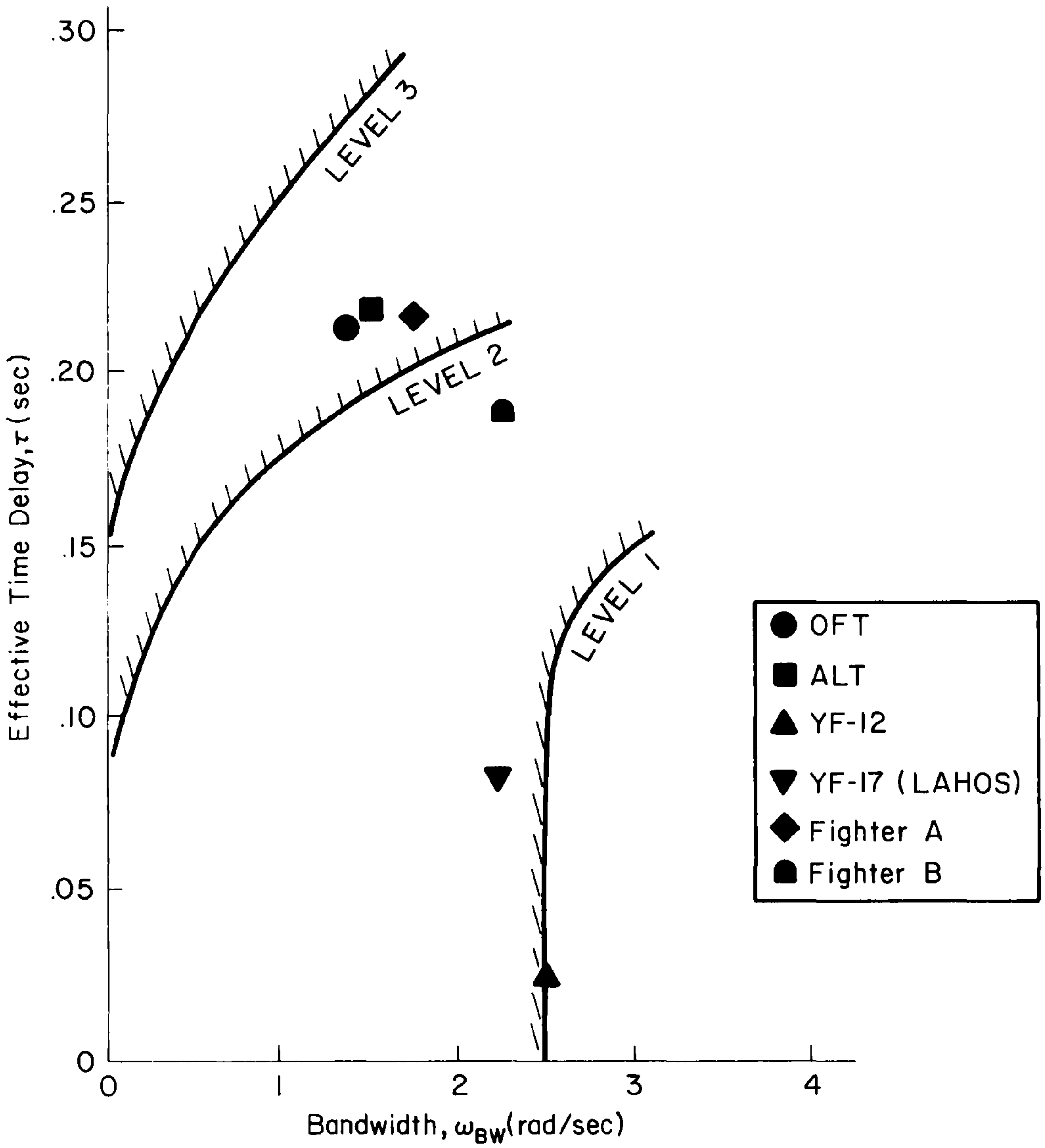


Figure II-24. Comparison of Six Aircraft With the Proposed Specification of Ref. 19

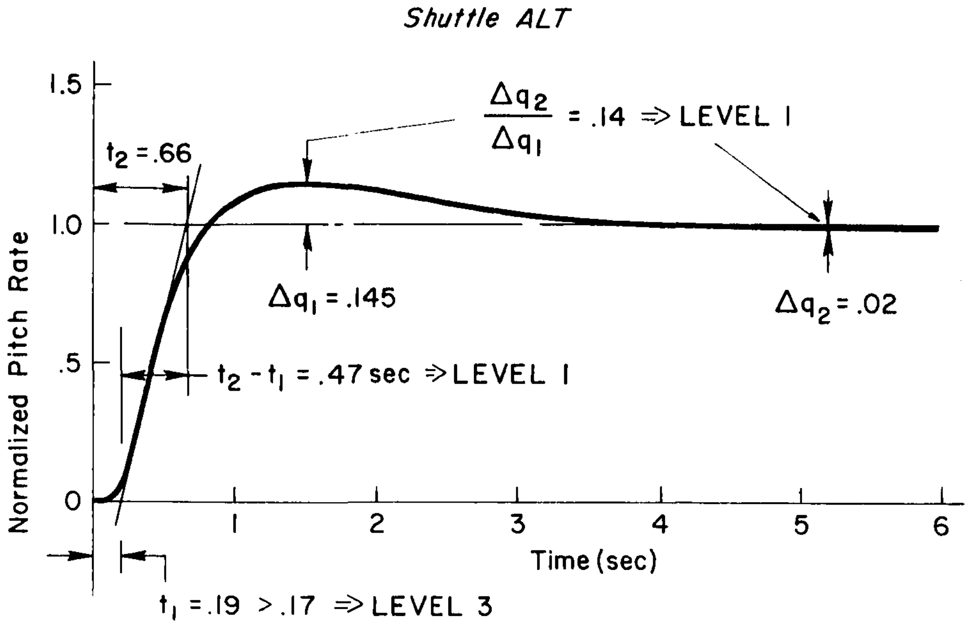


Figure II-25. Application of the Supersonic Cruise Research Vehicle Specification to the Shuttle ALT

3. The CAP′ Specification

As noted recently by Bischoff, Ref. 20, the control anticipation parameter must be redefined for aircraft with effective time delay since $\ddot{\theta}(0) = 0$ in this case. Following DiFranco, Ref. 21, Bischoff defines, on the basis of a unit step stick force input, a more general control anticipation parameter, CAP′, as

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

where the maximum pitch acceleration, $\ddot{\theta}_{\max_{\text{HOS}}}$, will occur sometime after the force input as shown in Fig. II-26 for the ALT. CAP′ is further extended to the short period lower order equivalent system model by defining

$$\text{CAP}_e' \equiv \left(\frac{\omega_n^2}{n/\alpha}\right)_e \frac{\ddot{\theta}_{\max_{\text{HOS}}}}{\ddot{\theta}_{\text{LOES}}(\tau)} \tag{25}$$

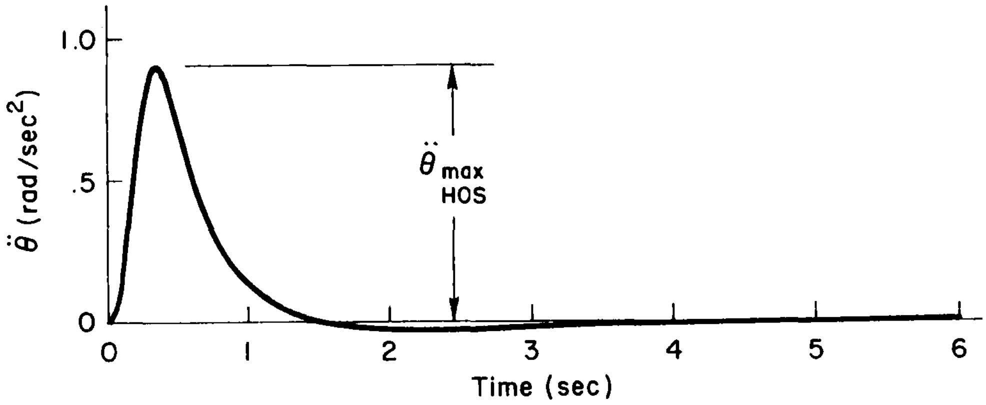


Figure II-26. Pitch Acceleration Response to a Unit Step RHC Input, Shuttle ALT

where the first factor in parentheses is defined from the LOES parameters. This factor alone does not give a good approximation to CAP′ because the short period LOES model will not generally be accurate in the high frequency region which largely determines the initial pitch acceleration history. Thus the second factor is required where $\ddot{\theta}_{LOS}(\tau) = A_\theta$ (from the LOES model) and $\ddot{\theta}_{max_{HOS}}$ is determined numerically from the HOS response (such as in Fig. II-26).

Reference 20 does not explicitly address the question of the validity of extending CAP to systems with significant effective time delay. For instance, the question of whether a pilot might consider a rapid pitch acceleration following a time delay "too abrupt" whereas the same initial pitch acceleration without time delay might be "desirably responsive" is not addressed. Bischoff does account for time delay explicitly by defining flying qualities levels in the CAP′ – τ plane (see Fig. II-27). The boundaries shown for each flying quality level were defined by correlations of data from DiFranco (Ref. 21), Neal and Smith (Ref. 8), and the LAHOS study (Ref. 7). These boundaries do seem to correlate the data somewhat better than is achieved using the present MIL-spec requirement based on CAP, however, the "τ-bandwidth" specification discussed in G-1 shown previously appears to do an even better job. In particular, the use of CAP′–τ admits Neal and Smith and LAHOS configurations with pilot ratings greater (poorer) than 3.5 to the Level 1 region whereas the τ-bandwidth specification properly classifies them.

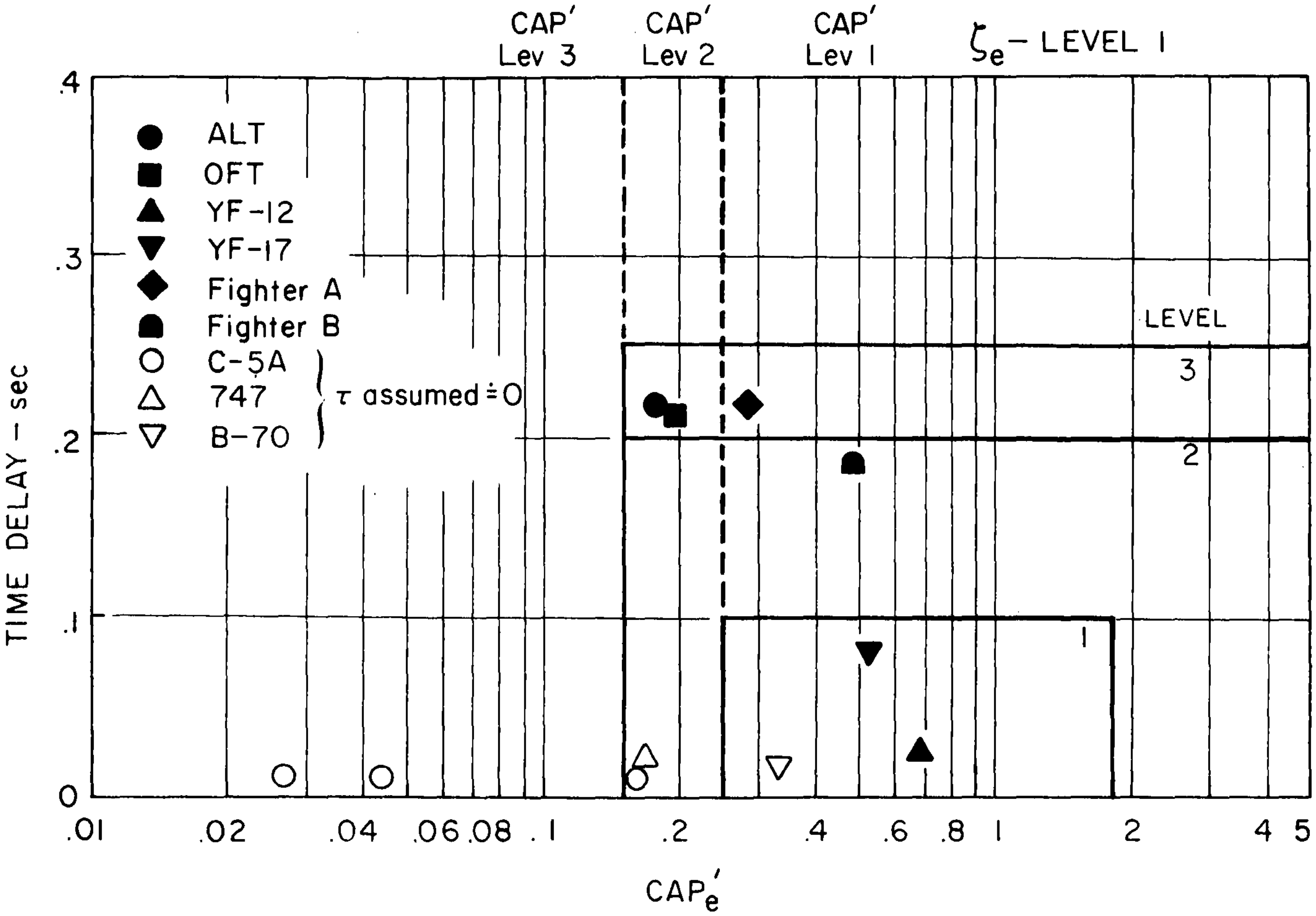


Figure II-27. Comparison of Eight Aircraft with the Ref. 20 CAP′ Criterion

However, despite these questions, it does appear that if CAP is to be extended to systems with significant HOS effects then some accounting of the effect of time delay on CAP must be made. To this end, the aircraft previously compared to the MIL-F-8785C short period boundaries are compared to the Ref. 20 boundaries in the CAP_e' – τ plane in Fig. II-27. The Shuttle OFT and ALT are Level 2 whereas they were shown to be Level 1 per MIL-F-8785C. This change is not directly due to the use of CAP′ but rather due to the redefinition of the lower boundary in Ref. 20 (i.e., Level 2 is $0.15 < 0.25$ versus $0.096 < 0.16$ for Category C in MIL-F-8785C). It should be noted that while the definition of the levels vary with flight category in the MIL-spec they are apparently the same for all levels in Ref. 20.

A further point should be noted regarding the YF-12. The YF-12 value of $CAP_e = (\omega_{SP}^2/n/\alpha)_e$ based on the short period LOES model is comparable to the Shuttle and several times smaller than that for the other three fighters. The YF-12's Level 1 CAP′ was traced to a large $\ddot{\theta}_{max_{HOS}}$ resulting from the lightly damped first body bending mode. This flexibility effect is noted in Ref. 22 and represents a difficulty for the CAP concept (at least for larger, flexible aircraft), since it seems questionable to expect an aircraft's flying qualities to be improved by increased fuselage flexibility.

4. Sources of Time Delay in the Shuttle Pitch Control System

The primary conclusion reached from the survey of three alternative criteria is consistent with that of Subsection F — effective time delay is the primary potential problem for Shuttle flying qualities. Figure II-28 shows a comparison of the various sources of effective time delay in the Shuttle pitch control system (Fig. II-2) and the hypothetical effect of removing each one. It may be seen that the largest single time delay is due to the actuator followed by digital computational delays. However, the net effect of the bending and smoothing filters is larger than either of these. While these sources effect time delay directly, they also have a smaller effect on the effective pitch attitude bandwidth.

H. PATH CONTROL AND PILOT LOCATION EFFECTS

To this point path control has not been analyzed directly but rather only treated through the concept of CAP. The Shuttle has unconventional characteristics (beyond those of pitch augmentation discussed earlier) with regard to the pilot's location with respect to the instantaneous center of rotation (ICR). This characteristic has potentially adverse consequences for path control. Furthermore, this effect reduces the validity of the concept of CAP compared to a conventional airframe as will be shown shortly.

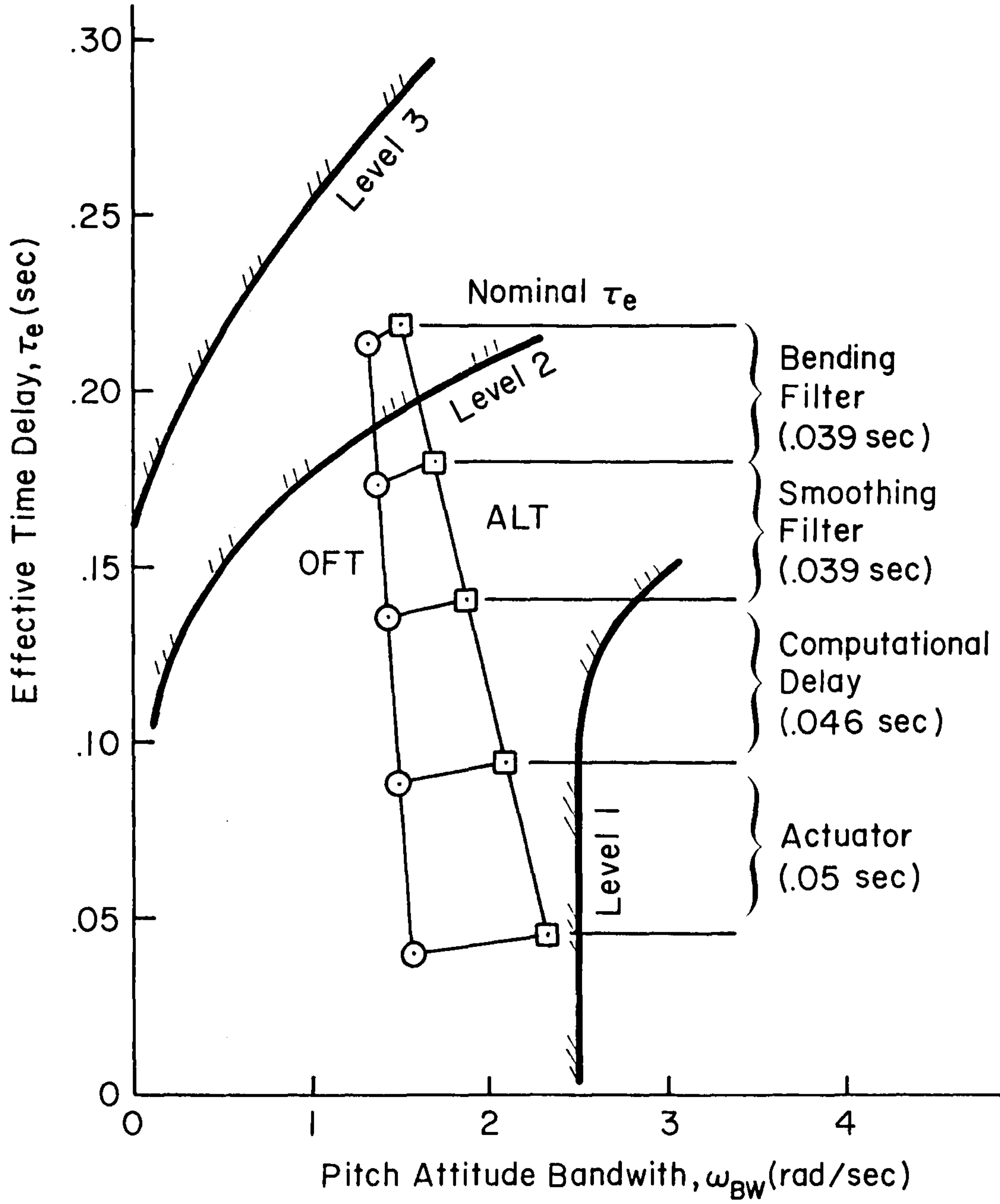


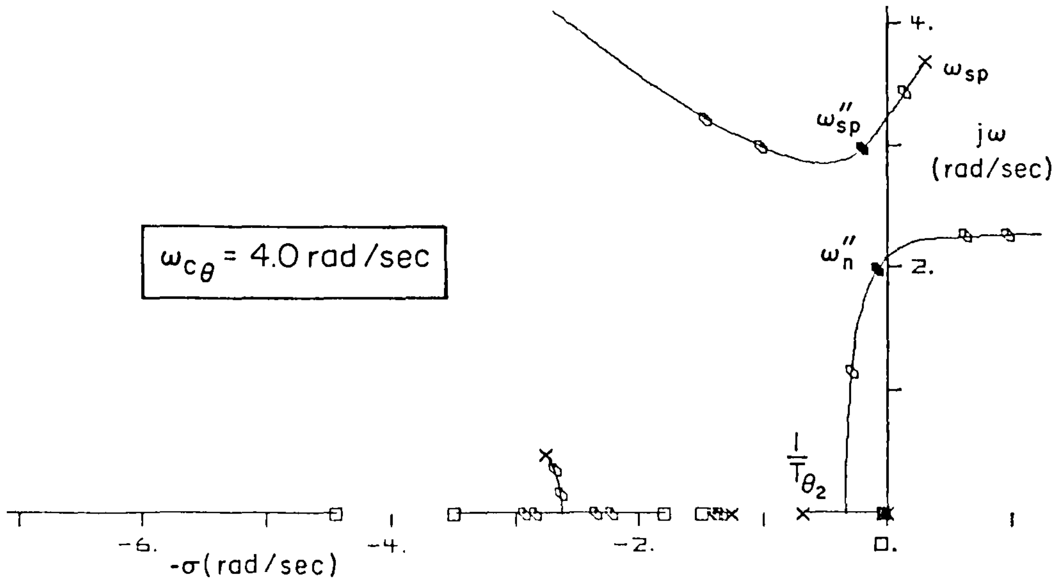
Figure II-28. Contribution of Shuttle Pitch Control System Elements to the Effective Time Delay

1. Previous Studies of Path Control

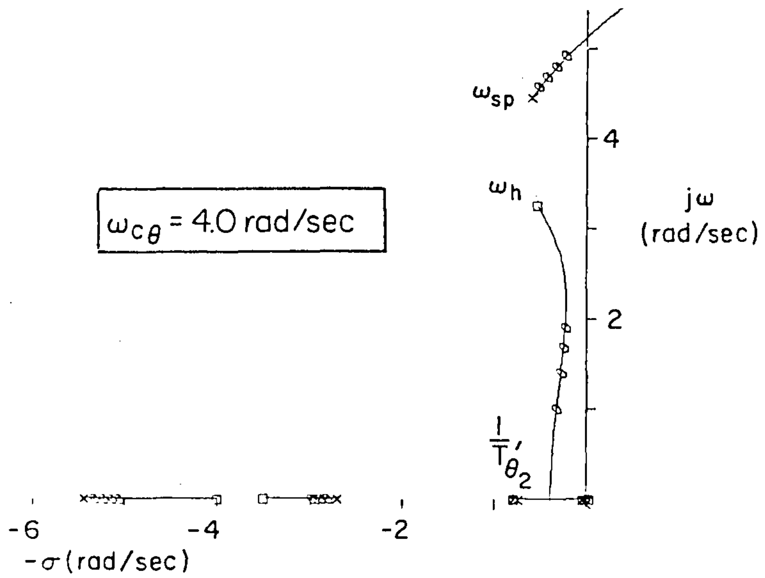
Manual control of the Shuttle in landing was analyzed theoretically and through simulation in the Ref. 2 study to investigate the Shuttle ALT PIO in Free Flight Five (FF5). Much of this study was based on comparison of the Shuttle with the YF-12 which is considered to have adequate flying qualities in approach. This study showed the Shuttle to be deficient in bandwidth both for the pitch attitude inner loop and the outer path loop. The pitch control deficiencies have been seen in the previous discussions of effective time delay in the pitch channel. The path control problems stem largely from problems of pilot location. To begin an examination of this the pilot's closures of the outer path loop for the Shuttle ALT (Fig. II-29a) and the YF-12 (Fig. II-29b) may be compared for an inner loop closure of 4.0 rad/sec. These root locus plots (from Ref. 2) were constructed from complete 3-DOF transfer functions including all flight control system elements. The complex loci which comes from the $1/T_{\theta_2}$ pole in each case gives rise to the "path mode" which becomes unstable for the Shuttle at approximately 2 rad/sec. For the YF-12 this mode remains stable largely because of the position of the altitude (path) numerator zero $[\zeta_{h_p}, \omega_{h_p}]$. The differences in this zero between the Shuttle and the YF-12 (and other aircraft such as the Boeing 747) significantly contributes to the Shuttle's path control problems. The reasons for the unusual characteristics of the altitude numerator may be seen from short period (θ, γ) approximations when the effects of lift-due-to-elevator (i.e. Z_{δ_e}) are considered. This will be done in the following subsection.

2. Pilot Location Relative to the Instantaneous Center of Rotation

The primary parameter affecting the numerator of altitude perceived by the pilot is the location of the pilot with respect to the instantaneous center of rotation, ICR, for elevator inputs. The ICR is the point at which the normal acceleration due to $Z_{\delta_e}\delta_e$ is just canceled by the pitching component $X_{ICR}M_{\delta_e}\delta_e$ so that



a) Shuttle ALT



b) YF-12

Figure II-29. Closure of Outer Path Loop by Pilot (from Ref. 2)

$$X_{ICR} = \frac{Z_{\delta_e}}{M_{\delta_e}} = -\frac{k_y^2}{\ell_{\delta_e}} \quad \text{(ft, positive forward from c.g.)} \tag{26}$$

For a HOS system with effective time delay, the ICR is undefined during the time delay period and X_{ICR} is given by the above equation at $t = \tau$.

It should be noted that in the derivation of CAP and CAP′; L_{δ_e} is assumed zero implying that the ICR is at the c.g. ($X_{ICR} = 0$). Table II-4 indicates that, with respect to the pilot, this is a reasonable approximation for large transport aircraft such as the C-5A and 747 due to their large effective tail lengths. Even for fighters, with more closely coupled elevators, the pilot is well ahead of the ICR. The information of Table II-4 is shown graphically in Fig. II-30 where the vehicle half length, L/2, is used as a non-dimensionalizing parameter. The extreme position of the Shuttle's $X_{ICR}/(L/2)$ value indicates it is the instant center location rather than the pilot position per se which is unusual. This is attributable to the large radius of gyration from the engines mounted behind (a presently) empty payload bay combined with short effective tail lengths for the Shuttle elevons. The segregation of aircraft along the ordinate which separates transport/bomber types from smaller "fighter" types stems largely from the pilot being located relatively further forward in large aircraft.

3. Manual Path Control

The effect of pilot location on VFR altitude control (which is of most interest for approach and landing) may be seen from pilot/vehicle analysis of the pilot's closure of the outer altitude loop. This requires including the "lift-due-to-elevator" in the normal acceleration equation and consideration of altitude rate, $\dot{h}_p$, at the pilot location

where

$$\dot{h}_p = \dot{h}_{cg} + \ell_p\dot{\theta}$$

TABLE II-4

COMPARISON OF PILOT POSITION WITH RESPECT TO THE INITIAL INSTANTANEOUS CENTER OF ROTATION FOR EIGHT AIRCRAFT

AIRCRAFT	W lb × 10^{-3}	LENGTH ft	V_T	ALTITUDE ft	Z_{δ_e} ft/sec^2	M_{δ_e} sec^{-2}	x_{ICR} ft (pos fwd)	PILOT LOCATION, ℓ_p ft(pos fwd)	ℓ_p/x_{ICR}	REFERENCE
OFT	184.	114.	197	2420	-61.55	-0.9202	67.1	52.5	0.782	2
C-5A	581.	228.8	146	SL	-9.53	-0.686	13.9	81.68	5.88	16
747	564.	226.8	165	SL	-9.73	-0.574	16.95	86.	5.07	16
B-70	300.	177.2	205	SL	-43.8	-0.836	52.4	99.	1.89	16
HL-10	6.5	20.8	327	3000	-212.	-28.	7.57	6.52	0.86	16
X-15	156.	49.2	331	SL	-160.	-13.8	11.6	18.8	1.62	16
NT-33A	11.8	38.6	135	SL	-13.4	-4.19	3.20	6.50	2.03	16
YF-12	--	93.0	786	25,000	-141.5	-6.084	23.3	50.	2.14	22
YF-12	--	93.0	Landing		-60.16	-2.53	23.8	50.	2.10	2

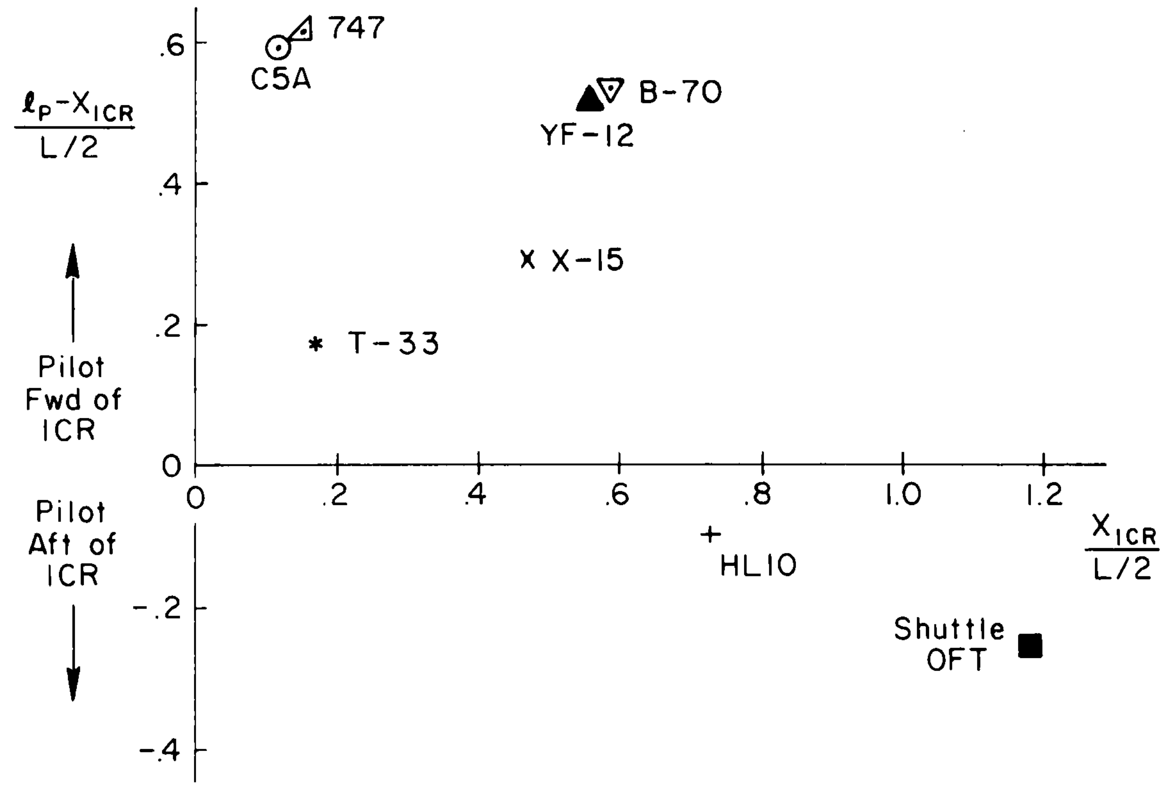


Figure II-30. Comparison of Pilot Location with Respect to the Instantaneous Center of Rotation for Eight Aircraft

As shown in the Appendix this leads to

$$N_{\delta_e}^{\dot{h}_p} = Z_{\delta_e}\left(U_o M_{\dot{w}} + M_q - Z_w + \frac{Z_{\delta_e}}{M_{\delta_e}} M_w\right)s + M_{\delta_e}U_o\left(-Z_w + \frac{Z_{\delta_e}}{M_{\delta_e}} M_w\right)$$

$$+ \left(\frac{\Delta \ell_p}{X_{ICR}}\right) Z_{\delta_e} s \left(s - Z_w + \frac{Z_{\delta_e}}{M_{\delta_e}} M_w\right)$$

$$= A_{h_p}\ (1/T_{h_2})(1/T_{h_3}) \quad \text{or} \quad A_{h_p}[\zeta_{h_p}, \omega_{h_p}] \tag{27}$$

The above equation may be used to construct a root locus for $N_{\delta_e}^{\dot{h}_p}$ (sketched in Fig. II-31) as a function of relative pilot position, $\Delta \ell_p / X_{ICR}$, where $\Delta \ell_p = \ell_p - X_{ICR}$ is the distance from the ICR to the pilot. Equation 27 is derived in the Appendix under the assumption of conventional pitch attitude dynamics. The Shuttle pitch augmentation issues discussed in Subsection C will have an effect on $N_{\delta_e}^{\dot{h}_p}$. However, this effect is secondary to the pilot location effect and thus use of Eq. 27 reveals the basic considerations.

It was stated in Subsection C that the pilot would use the series loop structure shown in Fig. II-1 for path control. In Ref. 4 it is noted that the use of a parallel structure is also conceivable, however, it would appear feasible only if pilot lag compensation were adopted in the attitude loop. Since we have shown in Subsection C that pilot lead compensation is to be expected in the inner loop, the usual assumption of a series closure will be made here. The distinction is important in that it determines the manner in which the pilot's inner loop equalization effects the outer loop.

It is useful to consider first the conventional situation (pilot forward of the ICR with aft elevator) where the $N_{\delta_e}^{h_p}$ roots will be complex and in the left half plane as the pilot is moved forward (or as the ICR is moved toward the c.g.). If for simplicity we assume that the pilot is controlling purely altitude, $Y_{p_h} Y_c$ for the outer loop becomes

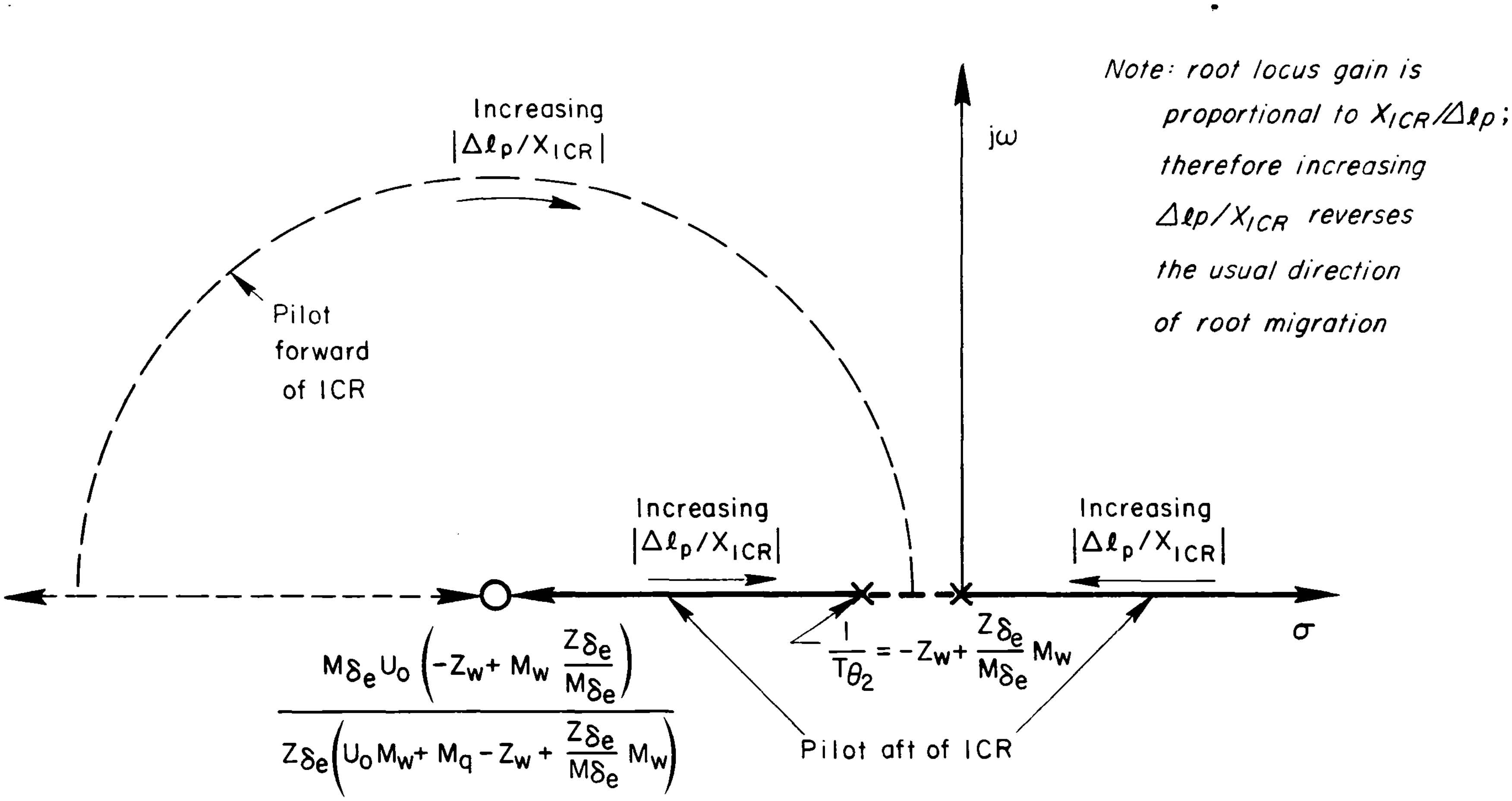


Figure II-31. Sketch of Root Locus for Variations in Relative Pilot Position, $\Delta\ell_p/X_{ICR}$. For Aft Elevator Control Typical of Shuttle in Landing Approach.

$$Y_{P_h} Y_c = K_{P_h} \cdot \frac{\theta'}{\theta_c} \cdot \frac{A_{h_p}[\zeta_{h_p}, \omega_{h_p}]}{A_\theta s(1/T_{\theta_2})} \tag{28}$$

where the prime denotes closure of the pilot's inner loop and $Y_{P_h} = K_{P_h}$ is a pure gain.

For the limiting case representing the "conventional transport"* situation ($\Delta\ell_p/X_{ICR}$ large)

$$[\zeta_{h_p}, \omega_{h_p}] \qquad (1/T_{h_2})(1/T_{h_3}) \qquad (0)(1/T_{\theta_2})$$

(see Appendix) and

$$Y_{P_h} Y_c \rightarrow K_{P_h} \cdot \frac{A_h}{A_\theta} \cdot \frac{\theta'}{\theta_c} \tag{29}$$

Thus, the dynamics of the altitude closure are essentially the same as the attitude dynamics. In this situation CAP — which is proportional to the ratio of the high and low frequency gains of the airframe attitude transfer function — may adequately account for the dynamics relevant to the altitude closure if the attitude augmentation is not too unconventional.

The situation is different for the more general case when $|\Delta\ell_p/X_{ICR}|$ is of the order of 1.0. The two cases of interest are: the (conventional) Case A where the pilot is forward of the ICR (such as for the YF-12 in Fig. II-29b) and Case B where the pilot is aft of the ICR (such as for the Shuttle in Fig. II-29a). The details of the inner attitude closure will affect the outer loop closure; however, to clarify the primary effect of the altitude zeroes the same, idealized inner loop will be assumed for Case A and B. Thus,

$$\frac{\theta'}{\theta_c} = \frac{Y_{P_\theta} N^\theta_{\delta_e}}{\Delta + Y_{P_\theta} N^\theta_{\delta_e}} \rightarrow \frac{\omega^2_{SP}}{[\zeta_{SP}, \omega_{SP}]} \tag{30}$$

and

$$Y_{P_h} Y_c = K_{P_h} \frac{A_{h_p}[\zeta_{h_p}, \omega_{h_p}]}{A_\theta s(1/T_{\theta_2})} \cdot \frac{\omega_{SP}^2}{[\zeta_{SP}, \omega_{SP}]} \quad (31)$$

for Case A and

$$Y_{P_h} Y_c = K_{P_h} \frac{A_h(1/T_{h_2})(1/T_{h_3})}{A_\theta s(1/T_{\theta_2})} \cdot \frac{\omega_{SP}^2}{[\zeta_{SP}, \omega_{SP}]} \quad (32)$$

for Case B.

The system survey for the outer loop closure in Case A is sketched in Fig. II-32a. When the pilot closes this loop the $1/T_{\theta_2}$ pole and pole at the origin (kinematic integrator) couple and migrate toward the $[\zeta_{h_p}, \omega_{h_p}]$ zero to form the path mode. The short period (attitude) mode is generally driven to higher frequencies. This situation should be compared to the YF-12 in Fig. II-29b.

The "Shuttle" situation is sketched in Fig. II-32b. The presence of the non-minimum phase $1/T_{h_3}$ zero causes the path mode to be driven unstable at relatively low gains, thus, greatly limiting path bandwidth compared to Case A. As may be seen from the asymptotic Bode plot sketches in Fig. II-32b, this may be attributed to the effective cancellation of the phase lead from the real h_p zeroes in Case B. This situation should be compared to Fig. II-29a.

It should be noted that these numerator effects will not be reflected in CAP and thus CAP can not distinguish between Case A and B. A final view of this situation is provided in Fig. II-33. It may be seen that, immediately after a step elevator input for a pullup, a pilot at B will "go down before he goes up" as does the c.g. (and also the main landing gear, the altitude of which is of primary concern for landing). The pilot at A will immediately go up i.e., he will lead the c.g. motion.

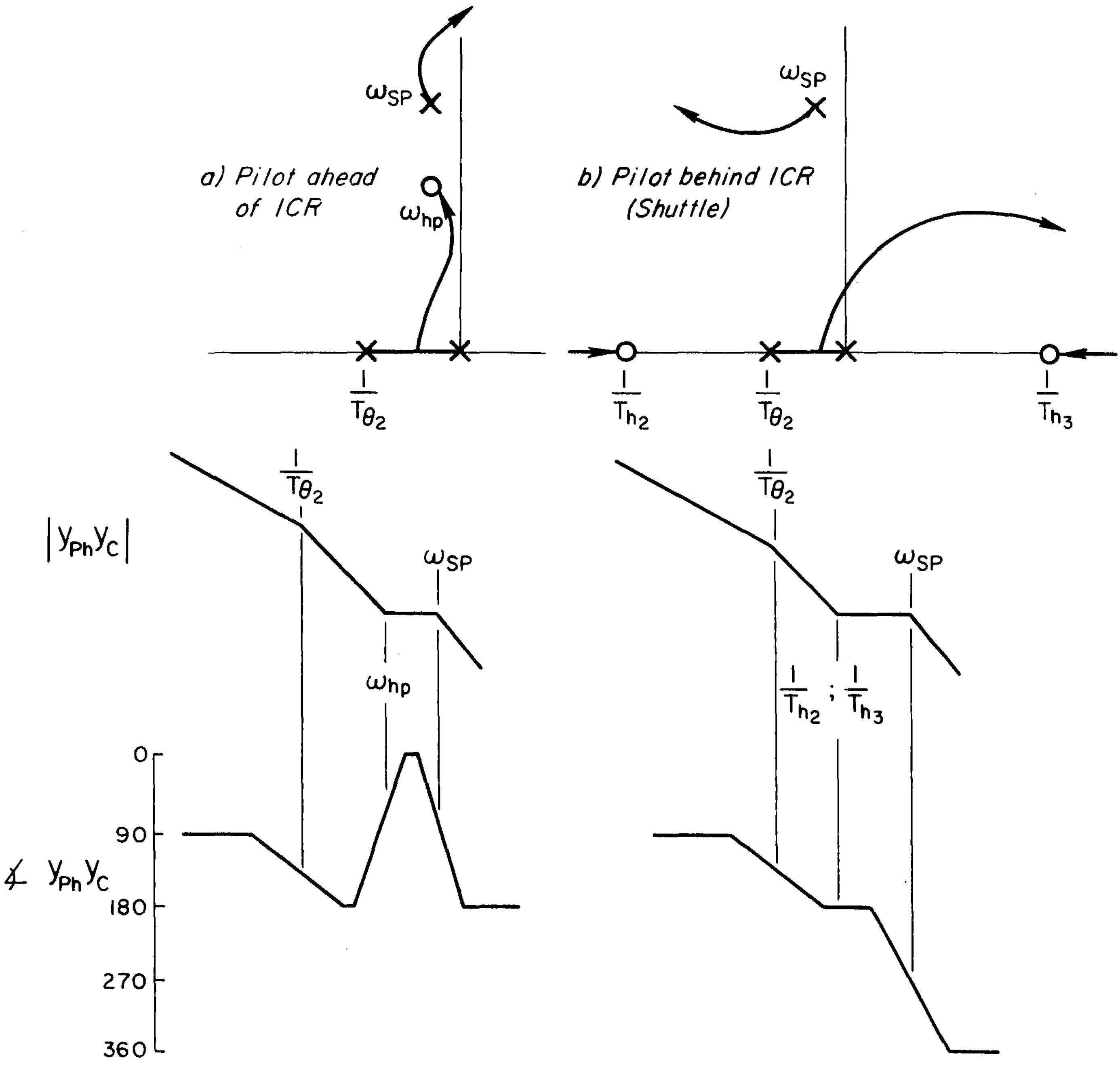


Figure II-32. System Survey Sketches of the Pilot's VFR Altitude Loop Closure for Two Pilot Locations

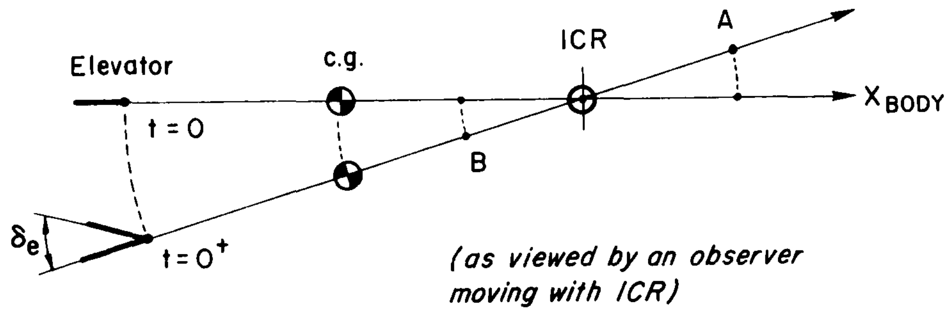


Figure II-33. Illustration of Initial Motion Following a Step (Aft) Elevator Input at 2 Pilot Positions

I. PILOT/AUTOMATIC FLIGHT CONTROL INTERFACE CONSIDERATIONS

The previous subsections have focused on manual flying quality specifications and criteria. However, the Shuttle Orbiter is designed to operate primarily in automatic modes with crewmembers supervising overall vehicle, navigation, and control system performance. Thus, consideration must also be given to workload assessment involving pilot monitoring, scanning, and anticipation for action in the event direct intervention is required. Pilot-vehicle-supervision-control interface workload is highest at the point where it is necessary for the pilot to intervene and take over manual control of some function. At this point manual control and monitoring/scanning workload combine. Under ideal conditions the supervisory workload should not be so high that active intervention into vehicle control will exceed 100 percent of his capacity.

Based on conventional flight considerations, one would expect the highest workload to be associated with terminal area maneuvering and energy management. Unfortunately no data are available as to pilot scan pattern during Shuttle terminal area operations and, until the second orbital flight and entry, no information was available concerning pilot workload during manual or automatic flight. On STS-2 TAEM the workload summation involving PTI inputs, system and flight supervision, and a last-minute change in landing runway due to high crosswind conditions (which necessitated manual intervention during at least a portion of the heading alignment maneuver) apparently came very close to saturating both crew members. At this crucial point the pilot was able to intervene and accomplish the manual control task; however, it appears that supervision of other system functions deteriorated and resulted in a significant energy loss with landing short of the target touchdown point. This tends to indicate either that the supervisory workload is too high or that the vehicle handling qualities may be deficient or both. It then serves as a workload saturation benchmark for further analysis.

Past flying quality experiments have shown a direct relationship between Cooper-Harper flying quality ratings (or levels of flying qualities) and pilot attention level required. For example, Fig. II-34 from Ref. 36 derives from a pitch attitude control primary task and a first-order cross-coupled instability as a secondary task. The analytic predictions of the preceding flying quality criteria (e.g., time delay, bandwidth, CAP, etc.) have shown the Shuttle Orbiter longitudinal flying qualities during the terminal phase of flight to be at best about Level 2 (i.e., a CH rating of, say, 4.5). From Fig. II-34 it may be seen that this imposes an attentional workload of 40 percent or greater for the longitudinal control task. Thus, 60 percent or less excess capacity remains to be devoted to other attentional requirements. Of course, the task from which the Fig. II-34 data derives was performed by a single pilot whereas the Orbiter is operated by a crew of two. The resulting implication is that both crewmembers were workload saturated in the above-noted STS-2 terminal operation. Again this tends to indicate that the combined crewmember supervisory workload may be too high, the aircraft flying qualities may be deficient, or both. Additional study of this interchange is warranted.

J. CONCLUSIONS

1. The Shuttle pitch control system combined with the low (unstable) static margin of the airframe produces an unconventional pitch attitude numerator zero $\left(1/T_{\theta_A}\right)$ which is significantly different than

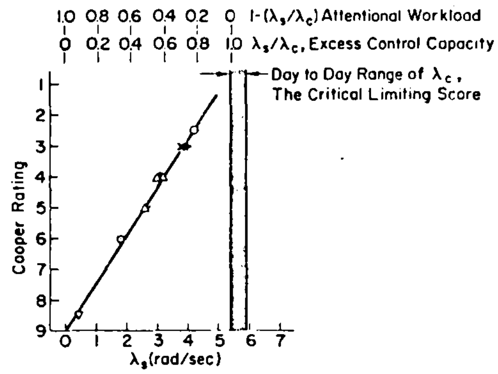


Figure II-34. Subjective Pilot Rating Versus First-Order Cross-Coupled Instability Score

the path mode inverse time constant $\left(1/T_{\theta_2}\right)$. This requires special consideration in the use of empirical data (which is largely based on conventional aircraft) but does not hamper the use of conventional manual control theory (Subsection C).

2. The pitch rate time response boundaries presently in use may unduly restrict overshoot characteristics, and conversely, may not sufficiently restrict effective time delay. However, this conclusion must be qualified in light of the unconventional attitude dynamics of the Shuttle noted above (Subsection D). There is a need for further validation of the pitch rate time response boundaries for large aircraft and for the supersonic flight regime.

3. The assessment of short period dynamics indicated that the Shuttle is apparently adequate, i.e., Level 1, on the basis of the conventional MIL-spec short period requirements for subsonic flight. However, for supersonic flight there is a need for further substantiation of the requirements since the data base is primarily subsonic (Subsection F).

4. Effective time delay appears to be the longitudinal flying qualities parameter of most concern. The Shuttle appears to be Level 3 based on existing time delay specifications and is comparable to aircraft with known landing problems (Subsections F and G).

5. The Shuttle is unconventional in that the pilot is located slightly aft of the instantaneous center of rotation for elevator inputs due to unusual inertial properties. Manual control theory and previous simulation studies indicate that this can degrade flying qualities by limiting the bandwidth of the pilots path angle control loop. Conventional parameters such as the control anticipation parameter (CAP) are not adequate for assessing this problem (Subsection H).

SECTION III

LATERAL DIRECTIONAL FLYING QUALITIES CRITERIA REVIEW AND ASSESSMENT

A review of the applicable literature revealed some nine key references (Refs. 6, 9-11, 23-27) relevant to lateral-directional flying qualities of the Space Shuttle Orbiter. From these documents, five parameters were selected as most pertinente to this highly augmented vehicle. The five are roll rate time response, lateral acceleration at the pilot station, roll command prefilter, lateral response time delay, and time to roll to a specific bank angle. Each of these is addressed in a following subsection. The first presents a sequence of entry flight phase roll rate time response boundary criteria that have evolved during the design/development cycle for the Shuttle Orbiter. These have been the principal flying quality criteria during the program and show a gradual relaxation of performance requirements from those more representative of Class IV (fighter) type vehicles to those more representative of Class II or III (large transport) type vehicle as the Orbiter design has matured. Typical roll rate time response to step roll rate commands are presented for several subsonic and supersonic flight conditions and show the response falls within the boundaries with considerable margin to spare.

The relationship between such open loop time responses and closed loop bank angle control is then examined using one low supersonic ($M = 1.5$) flight condition. The achievable roll attitude control closed loop bandwidth is found to be far less than that considered ideal or even acceptable in past pilot/vehicle flying qualities investigations. This raises serious questions regarding adequacy of the supersonic roll rate time response boundary criteria. A key factor in limiting closed loop roll control bandwidth is the large time delay due to computational, filter, etc., lags. The subsonic roll rate time response boundary criteria is tested through application to roll responses obtained in ground based and in-flight landing simulation of a hypothetical supersonic cruise transport configuration. These results show the

Orbiter time response boundaries accommodate vehicle responses rated acceptable in fixed-base simulation but unacceptable in in-flight landing simulation. A major contributor to the latter was the lateral acceleration at the pilot location during the rolling maneuver.

Lateral acceleration at the crew station is not addressed by the Shuttle Orbiter flying quality specification. Two criteria which have been presented in the literature are reviewed along with acceleration levels obtained from the supersonic cruise transport simulation and typical Shuttle Orbiter roll maneuvers. It is shown the criteria produce inconsistent predictions as to the acceptability of the lateral accelerations magnitudes involved.

Results from investigations of command prefilter lag influence on flying quality ratings in moving-base and in-flight landing/approach simulations are reviewed and a comparison is made between the Orbiter prefilter break frequency and those of the latest fighter aircraft. It is shown that the Orbiter prefilter break frequency rates favorably in both instances.

Time delay criteria from two flying quality specifications are compared with rating degradation obtained from recent landing simulation (ground and in-flight) involving increasing time delay. Results show difference in sensitivity to time delay due, possibly, to difference in aircraft type (size), task stress level, or both. The large but presumably acceptable time delay values for the Shuttle Orbiter are shown to be incompatible with the criteria and experimental results.

The last parameter to be examined is subsonic roll performance. The Orbiter time to roll through 30 deg is compared with that of other large transport aircraft and with the current military flying quality specification criteria. It is shown the Orbiter compares favorably with other transport aircraft but does not meet requirements of the military flying qualities specification. A principal reason for the latter is the significant time delay incurred before roll begins.

The section is concluded with a summary of potentially weak areas in the Shuttle Orbiter flying qualities specification.

A. ROLL RATE TIME RESPONSE

1. Time Response Boundary Criteria

Roll rate time response requirements for the Shuttle Orbiter are to be found in three separate documents, Refs. 6, 9, 10. Reference 6 is the initial (1973) Shuttle Orbiter flying qualities specification. Reference 10 is the circa 1977 flight control system specification and Ref. 9 contains some of the 1980 verification tests. Roll rate time response boundaries from the Ref. 6 flying qualities and Ref. 10 flight control specifications are presented in Fig. III-1. The dashed boundaries are from the flying qualities specification and the solid boundaries from the flight control system specification. Three separate flight regimes are identified. The upper sets cover the hypersonic portion of entry (24 > M > 2.5) for which Ref. 6 has a single set of response bounds while Ref. 10 has separate boundaries for M > 10 and < 10. We are unsure about the specific rationale for this distinction. The middle set of boundaries is applicable to the Terminal Area Energy Management (TAEM) portion of the flight. This starts at about 2.5M and is therefore identified as the supersonic region. The bottom set of boundaries apply to subsonic flight. Note that in all cases the boundaries have been relaxed between the flying quality specification (1973) and the flight control system specification (1977).

Normalized roll rate boundaries from the latest Rockwell International verification test report (Ref. 9), are shown in Fig. III-2. Two sets of boundaries are shown, one for supersonic, the other for subsonic flight. Comparison of the boundaries in Fig. III-2 with those of Fig. III-1 reveal the lower boundaries are essentially the same as for the subsonic and supersonic flight regimes from the flight control systems specification. The major difference is the upper boundaries which have been increased for the verification tests. Thus there is a continuing relaxation of the roll rate boundaries and these apparently are still not firm.

There are no other sources of normalized roll rate response boundaries for comparison. However, for subsonic and low supersonic flight

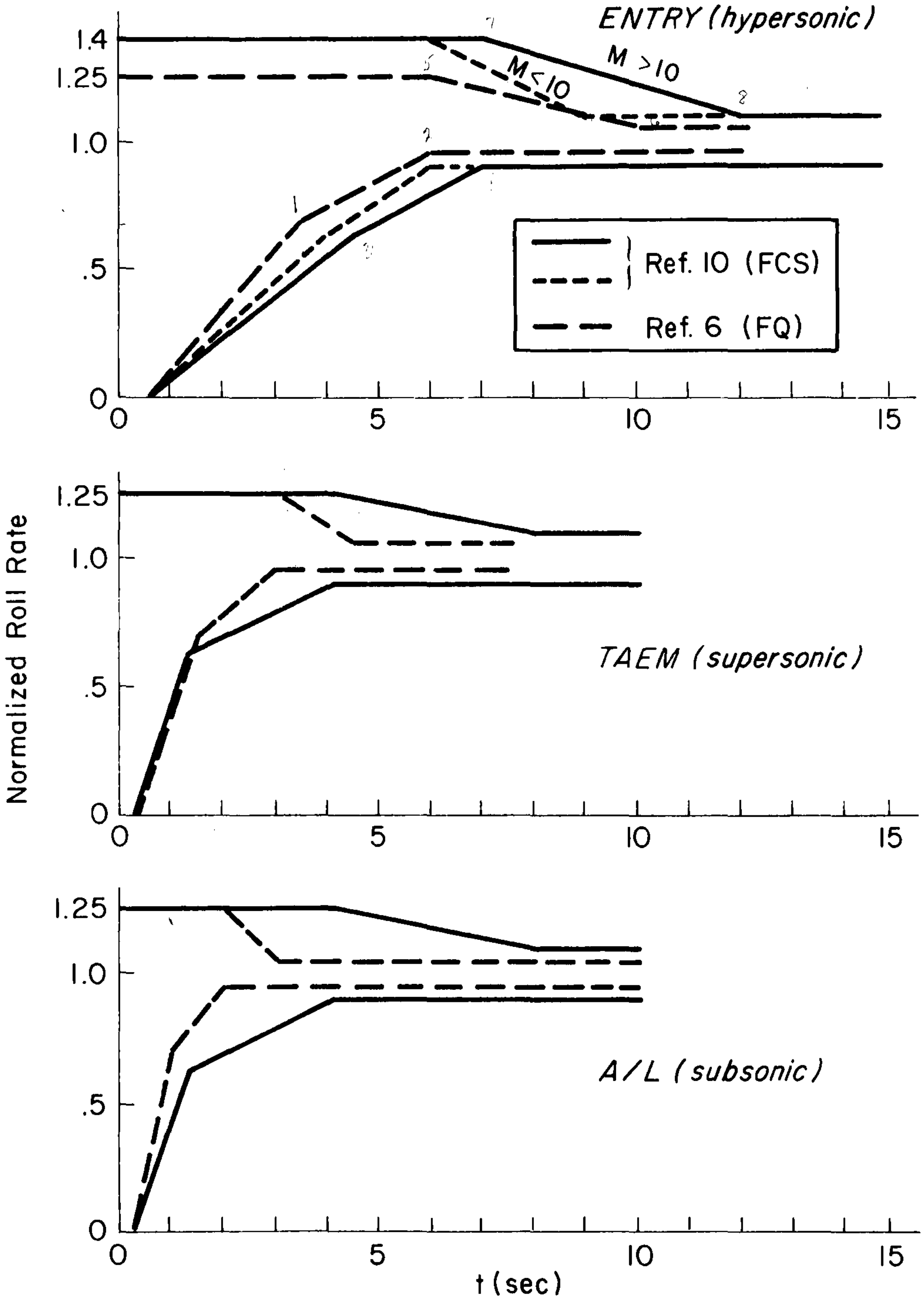


Figure III-1. Normalized Roll Rate Response Boundaries from References 6 and 10

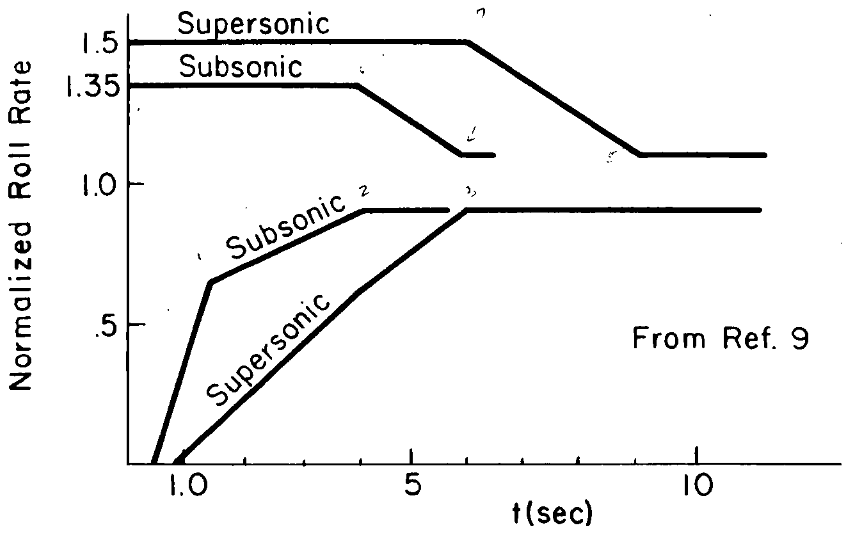


Figure III-2. Normalized Roll Rate Response Boundaries
from the Lateral Verification Tests

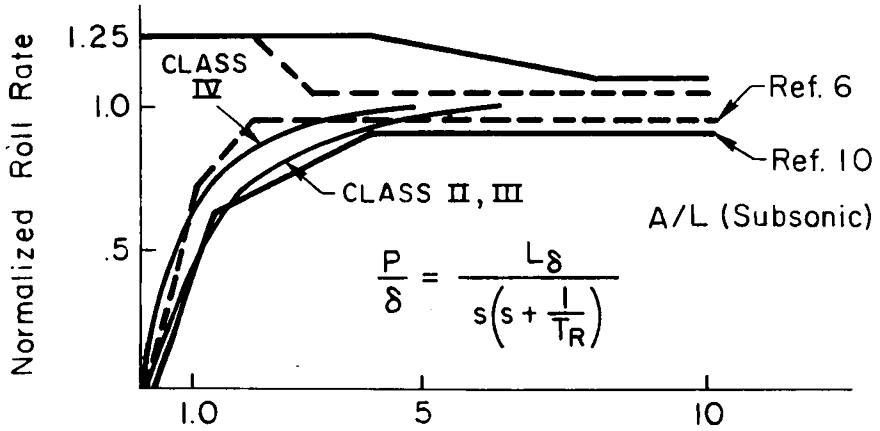


Figure III-3. Comparison of Shuttle Orbiter Roll Rate Response
Boundaries with MIL-F-8785C Roll Mode Requirements

the Ref. 11 roll subsidence mode time constant criteria can be applied if we assume a simple 1 deg of freedom response. Figure III-3 shows such a comparison. For Class IV aircraft, 8785C requires the roll time constant to not exceed 1 sec maximum. This is approximately the same as the Shuttle flying quality specification boundary. For Class II and III aircraft, 8785C requires a roll subsidence time constant of less than 1.4 sec maximum. This is approximately the same as the flight control system specification boundary. Thus, the Shuttle flying quality system specification is comparable to the military specification for fighter (Class IV) type aircraft while the Shuttle flight control specification is comparable to the military specification for medium (Class II) to large (Class III) transport and bomber type aircraft. This might indicate that the Ref. 6 Orbiter requirements were patterned after fighter aircraft while those of Ref. 10 were based upon large aircraft considerations. The latter is probably more appropriate.

2. Typical Vehicle Responses

An example time trace from the Ref. 9 verification tests (Fig. III-4) shows a typical response to a 3 sec roll rate "pulse" command in subsonic flight. This specific test is based on nominal non-dimensional aerodynamic coefficients for 0.7M but with a -6σ deviation in dynamic pressure which results in a speed approximating that for landing. Since the aerodynamic coefficients are essentially constant from 0.7 Mach on down, these flight traces are fairly representative of the landing configuration. The trace shows stability axis roll rate response to a step

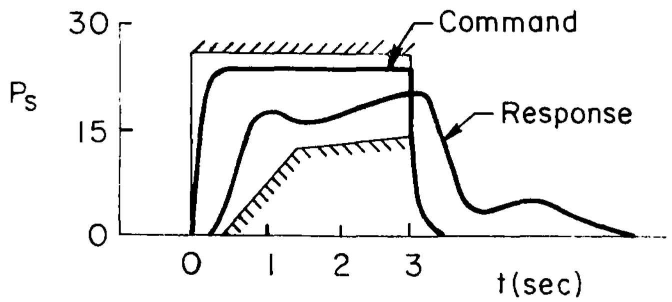


Figure III-4. Typical Orbiter Subsonic Roll Rate Response (from Ref. 9, page A2-203) command input. The verification boundaries are also shown. The initial roll rate response easily satisfies the boundary limits. The relatively large time delay of approximately one third second also easily falls within the boundaries.

Figure III-5 is a similar set of traces for a Mach 1.1 flight condition. In this case, the supersonic roll rate response boundaries are applied. Again, a relatively long time delay and what would normally be considered a poor roll rate response easily meets the boundaries. Figure III-6 presents a 1.5M normalized roll rate response from an STI 3 deg of freedom analysis which includes all system lags and computational delays. This shows an effective time delay of approximately 450 msec which is somewhat longer than the time delay obtained in the verification run of Fig. III-4 but consistent with Fig. III-5. Figure III-7 is a 3 M normalized roll rate response, again from the STI 3 deg of freedom analysis; however, this time the computational delays and the higher order bending filters were not included. This trace is presented to emphasize the influence of roll numerator right half plane zeroes which occur in this flight regime. Note the initial roll rate response is opposite to the final response. This is caused by a non-minimum phase (right half plane) zeros. The reversal produces an effective time delay of approximately a quarter of a second to which the computational delays, filter lags, etc., must be added. Figure III-8 presents a time response trace from the Ref. 9 verification test at 3.4M with nominal airframe aerodynamics but with a malfunction of the yaw jets such that they fire only on the one side. This produces the high

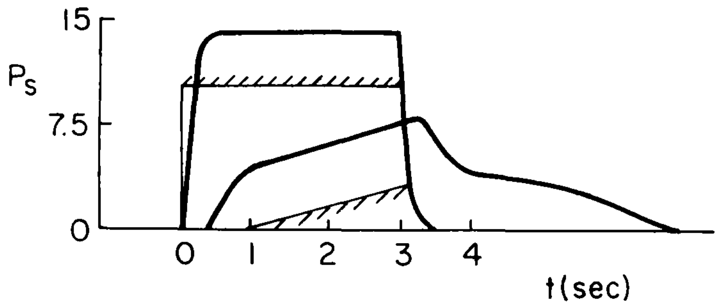


Figure III-5. Orbiter Roll Rate Response at 1.1M
(for Ref. 9, page A2-207)

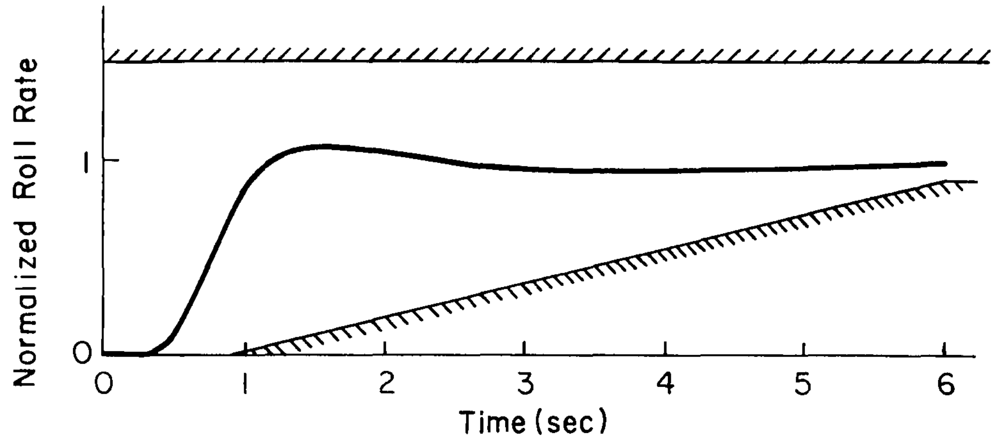


Figure III-6. 3 DOF Model Response, M = 1.5

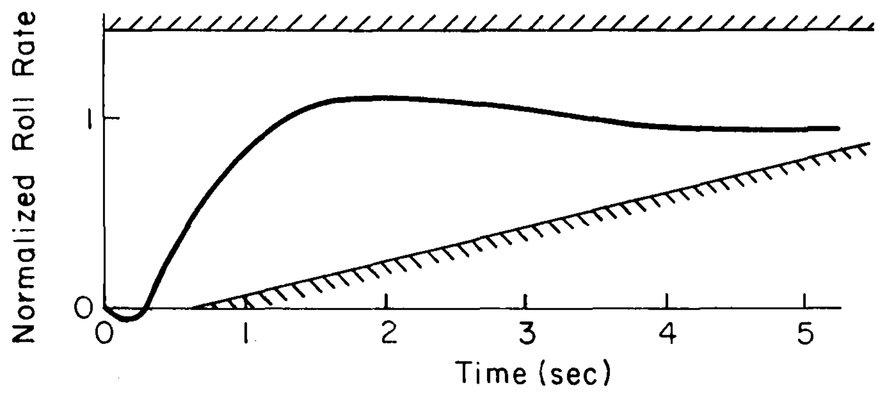


Figure III-7. 3 DOF Linearized Model Roll Response, M = 3.0

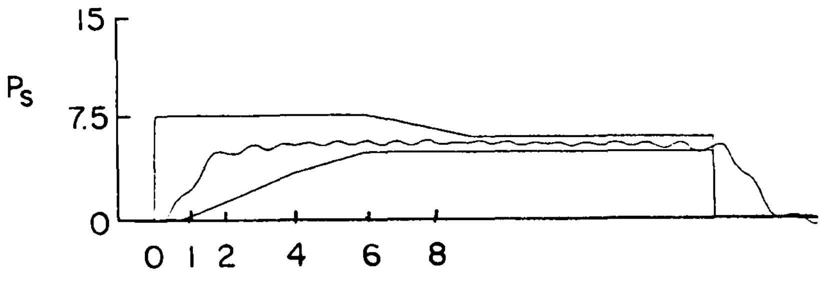


Figure III-8. Orbiter Roll Rate Response at 3.4M
(from Ref. 9, page A1-83)

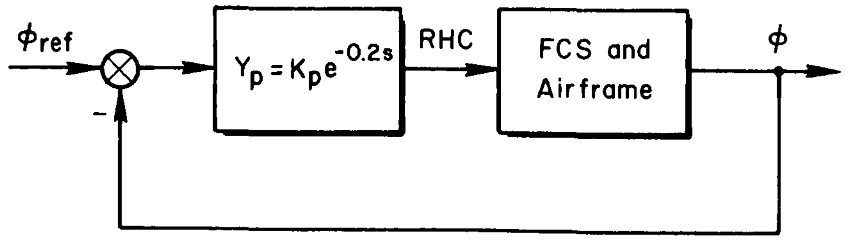
frequency ripple seen on the trace. The important aspect is that the roll rate response time delay appears to be approximately 0.4 second which is slightly greater than that obtained with our 3 DOF linearized model (Fig. III-7) and is consistent with adding computational lags to the Fig. III-7 non-minimum phase lag. Again, the Shuttle Orbiter easily meets the very loose normalized roll rate boundaries for supersonic flight.

3. Open Loop Time Response Boundaries Versus Closed Loop Bandwidth

It is pertinent at this point to include some results from a critical flight condition investigation (Ref. 29) currently being conducted for Rockwell International. This involves the neutrally damped or possibly divergent 0.25 Hz oscillation that developed in the STS-1 flight at about 1.5M during a period when the yaw jets were not firing. It appeared that jet firing temporarily halted the divergence. Aerodynamic coefficients extracted from the flight data using MMLE techniques were supplied by R.I. These data were modified slightly in order to reproduce the 0.25 Hz closed loop oscillation. This modification consisted solely of a 30 percent reduction in roll control power, $C_{\ell_{\delta_a}}$.

The normalized roll rate response to a step rotational hand controller (RHC) input is shown in Fig. III-9 for a case where the aileron and rudder loops are closed but the yaw jet loop is open (jets not firing). In this case the effective time delay almost exceeds the allowable delay and the roll rate reversal does exceed the response boundary after 4 sec. The time to achieve 63 percent of commanded roll rate is 1.5 sec.

The roll control bandwidth that this vehicle response will permit can be predicted by assuming the simple pilot/vehicle loop structure shown below.



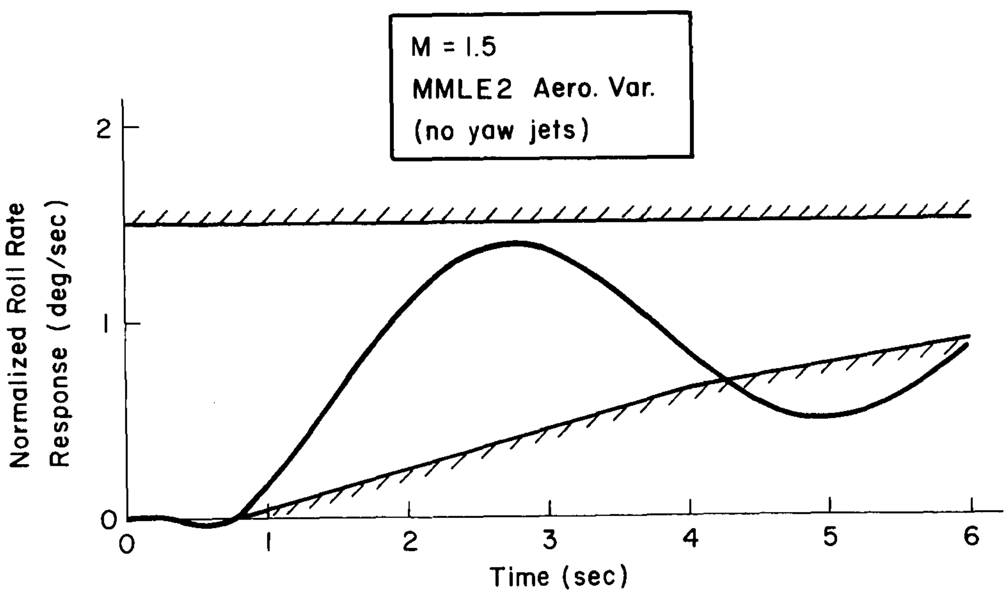


Figure III-9. Normalized Roll Rate Response to Step Command; Orbiter Aerodynamic Coefficients Derived from STS-1

System survey plots for this closure are presented in Fig. III-10. The upper plot is the conventional root locus and the lower plots are the $j\omega$ and $-\sigma$ bodies or Bode-root locus. Note that the 1.4 rad/sec (0.25 Hz) mode is rapidly destabilized by the ϕ loop closure (via manual or automatic control means). The option of lead equalization by the pilot in an attempt to stabilize this mode would be futile because of the very sharp phase drop off shown on the Bode phase curve. Thus, a pure gain closure is the only means of controlling roll. Conventional pilot closure criteria are 35 deg phase margin or 6 db gain margin, whichever gives the higher bandwidth. In this case the closure line which intersects the amplitude peak at or above 1.4 rad/sec will result in instability. A 6 db gain margin then places the crossover, ω_c, at about 0.13 rad/sec. This is totally unacceptable for roll attitude control -- the nominal bandwidth criteria is about 2 rad/sec. Any attempt to increase gain and bandwidth will drive the 0.25 Hz mode unstable. This is a basic PIO situation if the pilot attempts to control roll attitude.

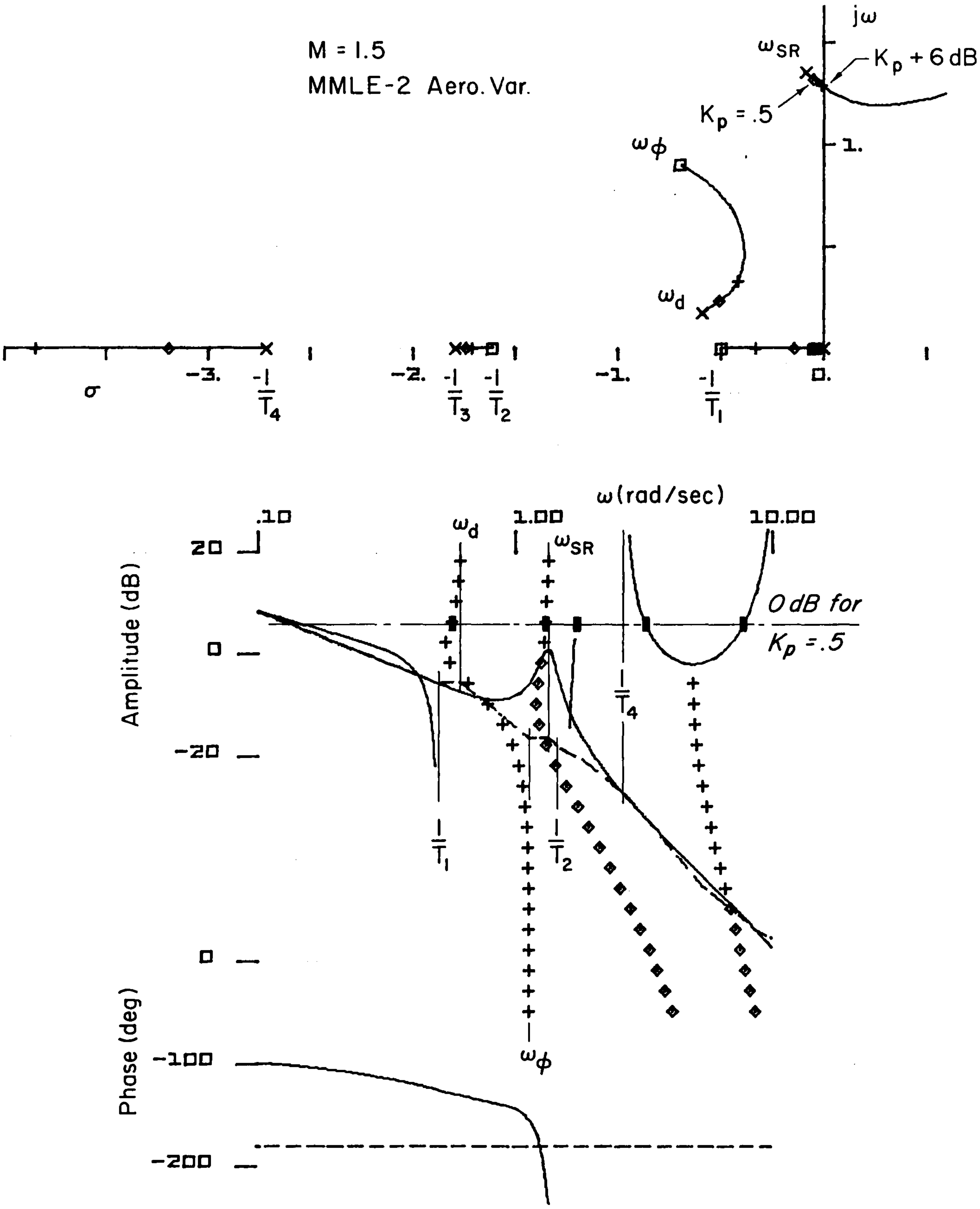


Figure III-10. System Survey for Pilot Control of Bank Angle, Yaw Jet Loop Open

With the yaw jet loop closed (jets firing) the normalized roll rate response to a step RHC input is as shown in Fig. III-11. In this instance the response stays within the boundaries except for the small initial roll reversal. The effective time delay due to computational lags and the reversal is about 0.65 sec. The time to reach 63 percent of the commanded roll rate is about 1 sec. System survey plots for manual roll control of this vehicle are presented in Fig. III-12. Again, pilot closure of the roll loop destabilizes the high frequency mode and creates a low frequency oscillatory mode. The 6 db gain margin closure criteria gives a crossover $\omega_c \doteq 0.45$ which is only slightly better than with the yaw jets off. However, the Bode phase cut-off is not so sharp in this case and the pilot might be able to increase the bandwidth to approximately 0.6 rad/sec by adopting a first order lead in the vicinity of 1 to 1.5 rad/sec and increasing his gain about 2 db. This would require considerable concentration (increase workload) and possibly further degrade any handling quality assessment. The lead is of very limited help because of the very large phase lag contribution of the

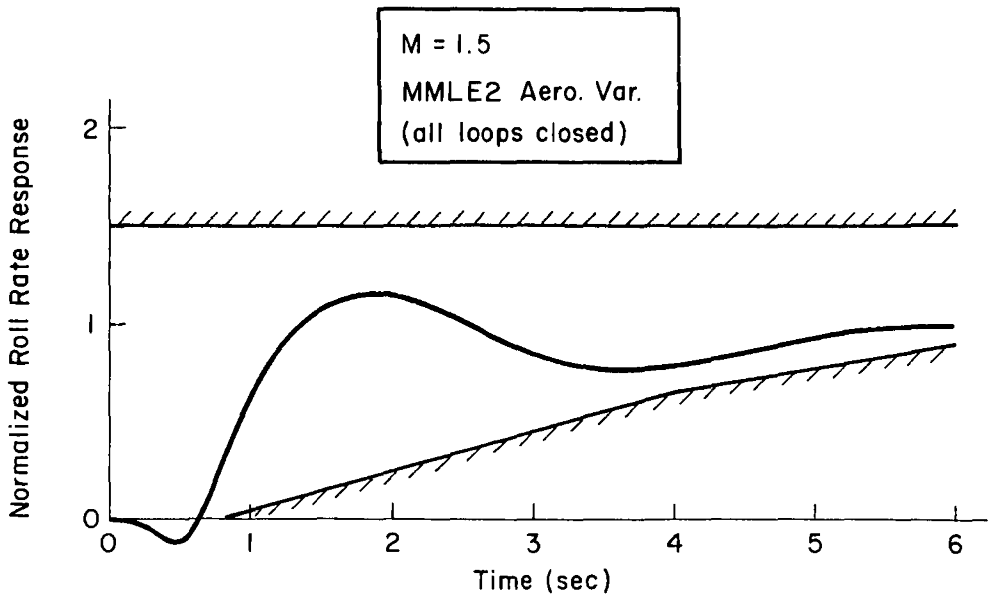


Figure III-11. Normalized Roll Rate Response to Step Command; STS-1 Aerodynamic Coefficients, Yaw Jets Operating

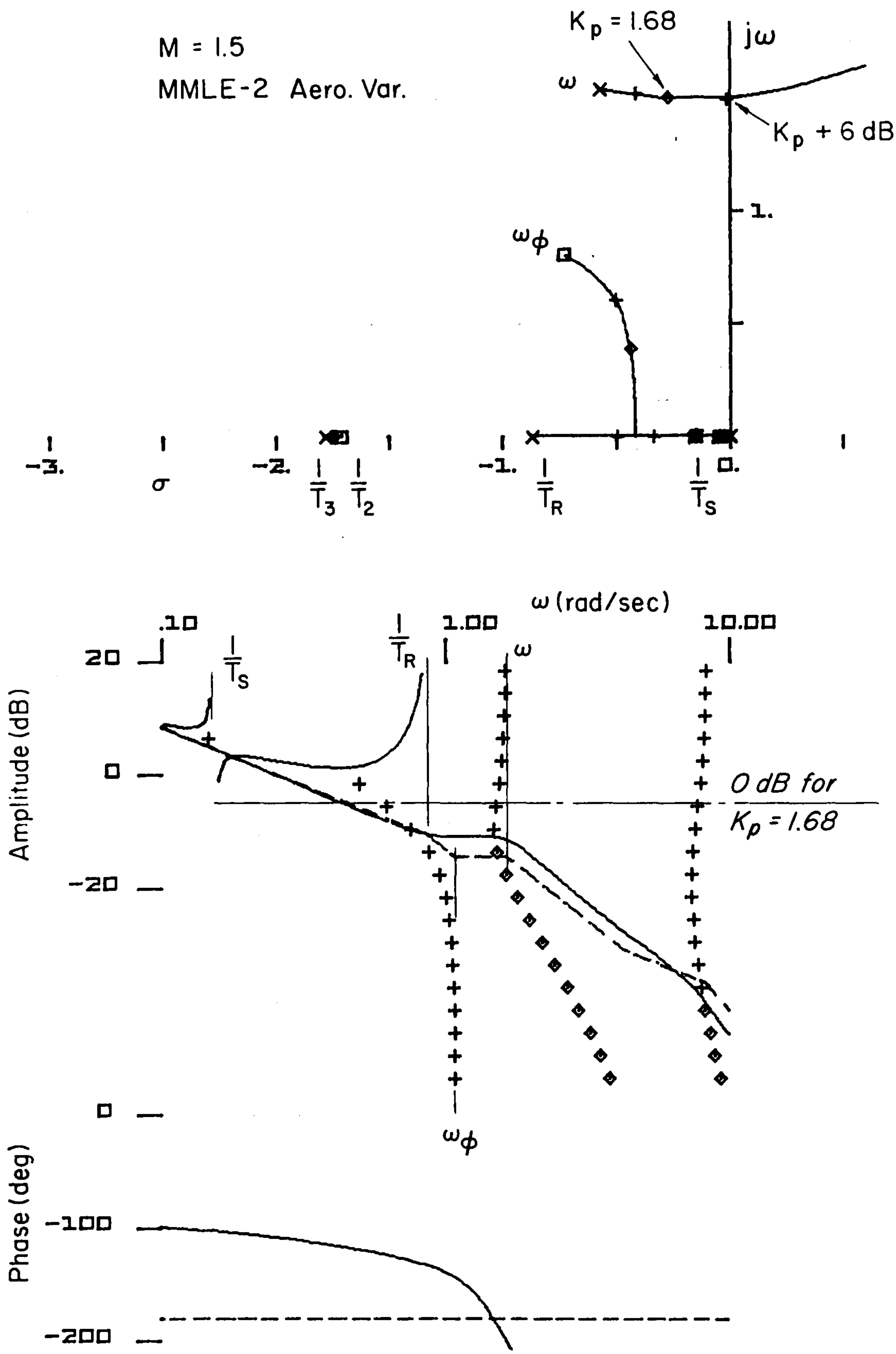


Figure III-12. System Survey for Pilot Control of Bank Angle; Yaw Jets Operating computational time delay. Any attempt to increase bandwidth by further increasing gain would cause the high frequency mode to become quite lowly damped. We would, therefore, predict a poor handling quality rating due to a combination of low bandwidth, lowly damped nuisance mode oscillation, and high workload.

Interestingly, the augmented vehicle dynamic characteristics with the yaw jets firing meet the MIL-F-8785C oscillatory mode damping requirements. It is not known for certain whether this mode derives from the unaugmented airframe dutch roll or lateral phugoid modes. But whichever it is, MIL-F-8785C requires for Level 1:

$$\zeta_d > 0.08 \qquad (\zeta\omega)_{RS} > 0.5$$

The values for the oscillatory mode are:

$$\zeta = 0.347 \qquad \zeta\omega = 0.57$$

In summary, this analysis has shown that:

- Meeting the supersonic normalized roll rate response boundaries does not assure acceptable closed loop roll control bandwidth or precision
- The large effective time delays at this flight condition (which has already been noted at other conditions) restrict the closed loop roll control to bandwidths far below that normally considered acceptable.
- For the current flight control system feedback loop structure, equalization, and gains, it is essential to lateral stability and controllability that the yaw jets remain active.

4. Other Considerations

Example time traces from the Ref. 24 Langley Research Center simulation of a Supersonic Cruise Transport (SSCT) approach and landing are shown in Fig. III-13. The traces on the left show the step control wheel input and the roll rate, sideslip, and roll attitude responses. The traces on the right show lateral acceleration at the cg and at the pilot station which is located some 145 ft ahead of the cg. Lateral components due to the yaw and roll angular accelerations at the pilot

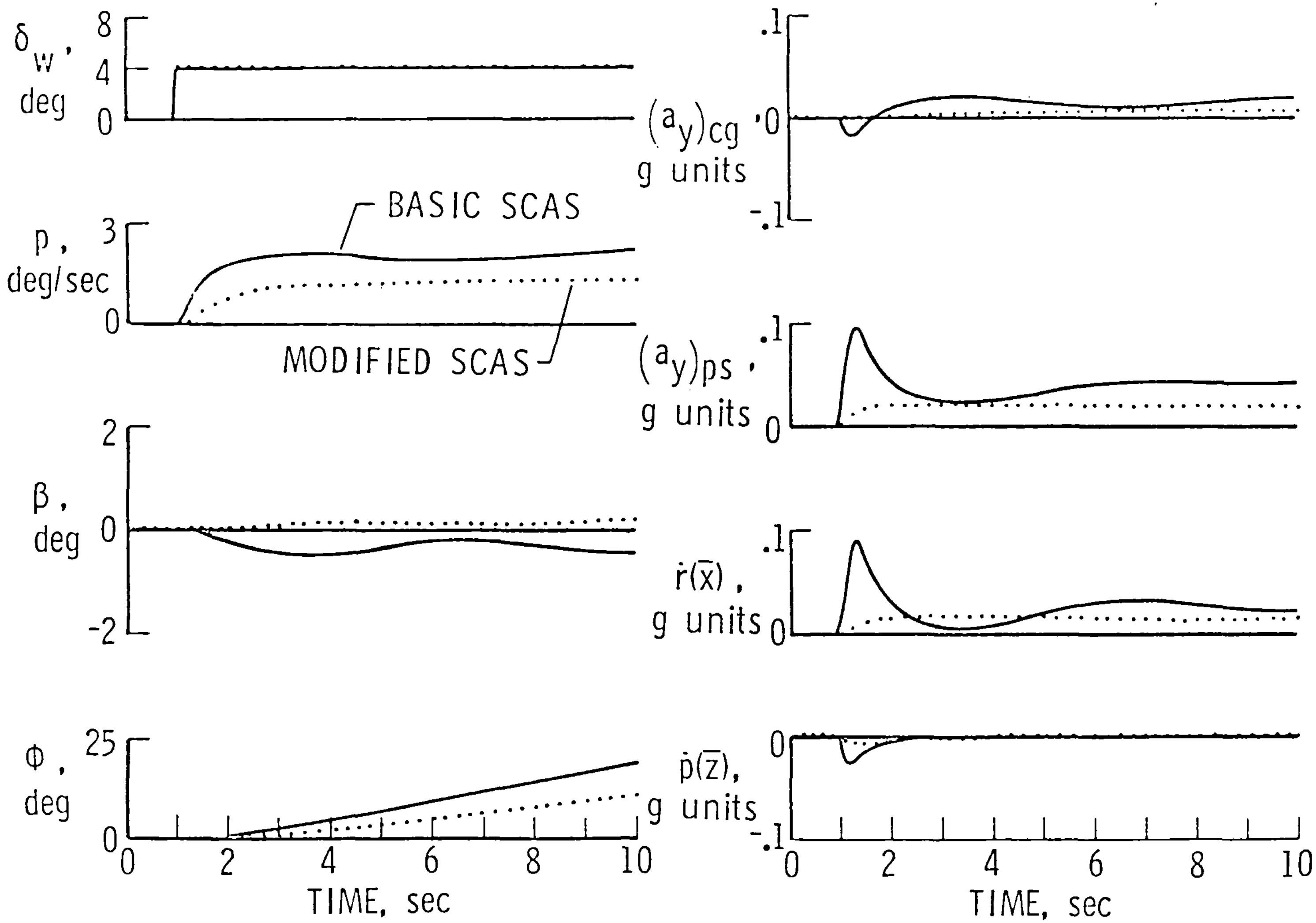


Figure III-13. Comparison of Lateral Response to a Wheel Step Input for SCAS and Modified SCAS (Ref. 24) station are also shown separately. The aircraft response with the basic stability/command augmentation system (SCAS) received a handling quality rating (HQR) of 2 for landing approach maneuvers on the fixed base simulator. However, when this same vehicle and control system was simulated on the Calspan Total In-Flight Simulator (TIFS) the lateral flying qualities were rated unacceptable (HQR > 4) primarily due to the large lateral acceleration at the pilot station. The SCAS roll rate command algorithm was then modified to decrease the command gain and incorporate a first order lag prefilter:

$$\text{Basic:} \qquad P_c = 0.5\delta_w$$

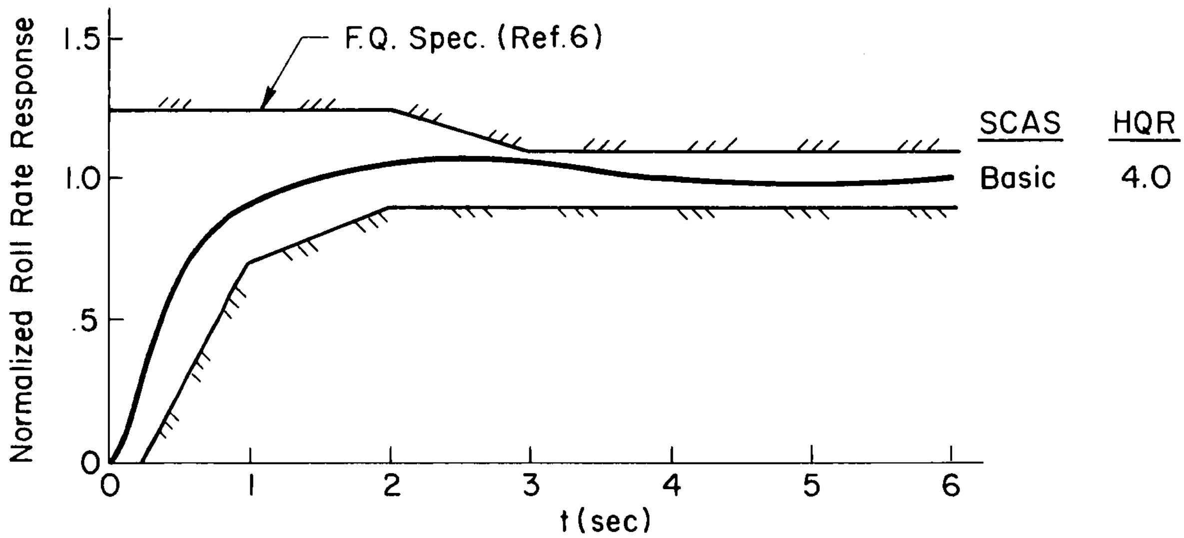
$$\text{Modified:} \qquad P_c = \frac{0.35\ \delta_w}{(0.7\ s + 1)}$$

The resulting lateral response received a HQR of 3 on the TIFS.

The two roll rate responses of Fig. III-13 are fitted in Fig. III-14 with normalized subsonic time response boundaries from the Ref. 6 and 10 Shuttle Orbiter specifications. It is important to note that either SCAS meets the Orbiter specification boundaries but the faster responding vehicle is rated unacceptable by the pilot. Thus, this roll rate requirement is, by itself, insufficient to produce satisfactory flying qualities. These results also tend to support the relationship previously noted in Fig. III-3 between the Orbiter flight control system specification boundary and the roll rate time response of large (Class II and III) aircraft.

B. LATERAL ACCELERATION AT THE PILOT

Figure III-13 shows that the modified SCAS totally eliminated the large (0.1 g) initial lateral acceleration pulse at the pilot. The result is a lateral acceleration buildup very similar to that of the roll rate. Thus the ratio of lateral acceleration to roll rate is relatively constant. Both of these aspects relate to criteria proposed in Refs. 25 and 26. The effect of pilot location relative to the aircraft center of gravity is shown in Fig. III-15 (taken from Ref. 24). For the supersonic transport configuration simulated, the pilot station was



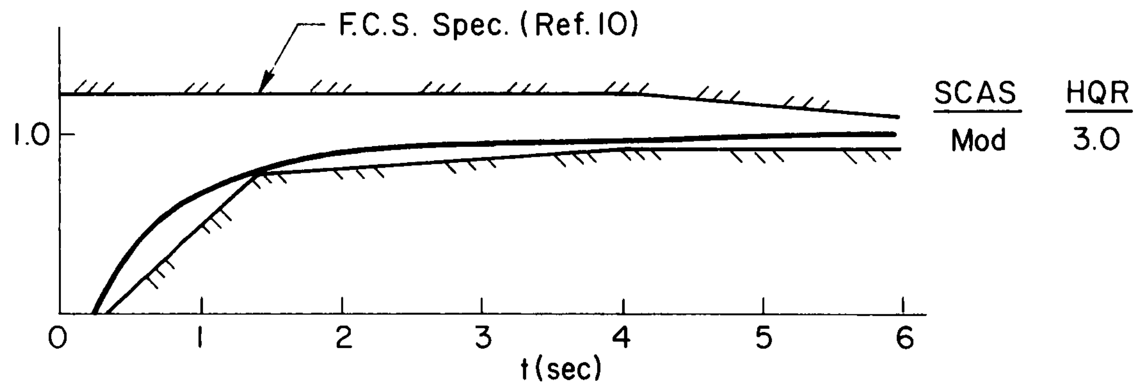


Figure III-14. Normalized Roll Rate Responses -- Landing Supersonic Cruise Transport In-Flight Simulation (TIFS)

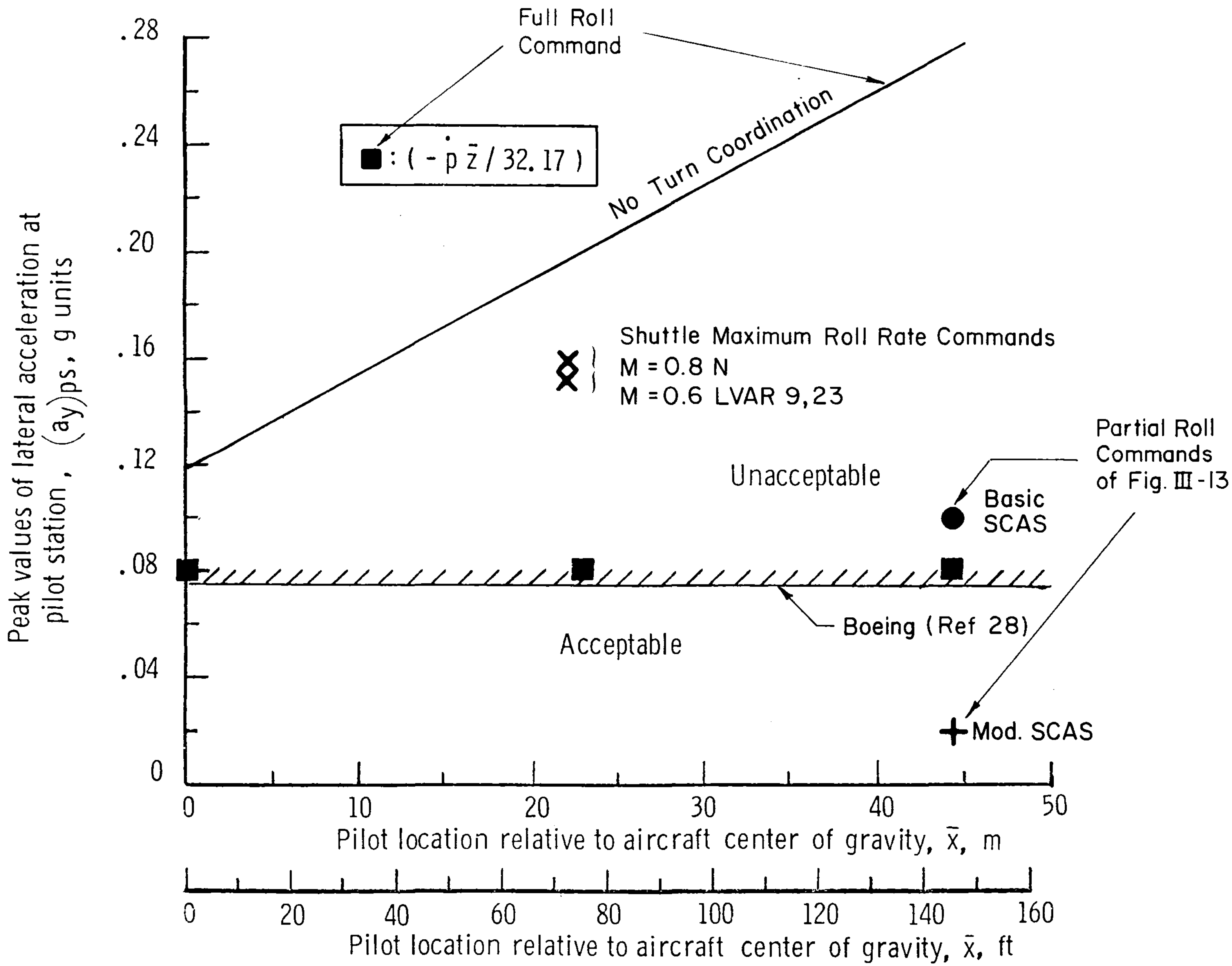


Figure III-15. Peak Values of $A_{y_{ps}}$ Compared With Criterion of Ref. 24

located 16 ft above and 145 ft ahead of the aircraft c.g. The squares in Fig. III-15 reflect the lateral acceleration component due only to a pilot location 16 ft above the aircraft c.g. Thus, if the pilot were located longitudinally at the aircraft c.g., but above it a distance of 16 ft he would experience 0.08 g lateral acceleration due a full roll command input. In addition to this roll acceleration component, the c.g. would experience a lateral acceleration of roughly 0.04 g's, bringing the total to 0.12 g's. If the control system contained no turn coordination features and if the pilot were to be moved forward of the c.g., the lateral acceleration experienced for this same input would be as shown by the upper line. This indicates approximately 0.28 lateral g's when the pilot is located 145 ft ahead of the c.g. Thus, without turn coordination features, a pilot location ahead of and above the c.g. can experience quite large lateral accelerations in rolling maneuvers. Turn coordination effectively moves the instantaneous center of rotation for roll control inputs and therefore can be tuned to reduce lateral acceleration at the pilot (but at the expense of increased a_y at the aft sections of the aircraft). The peak lateral accelerations obtained from Fig. III-13 partial roll commands for the two different SCAS configurations are shown by the solid circle and the cross on Fig. III-15. The solid circle reflects the basic SCAS which had a peak lateral acceleration of 0.1 g's. The cross represents the modified SCAS with a maximum lateral acceleration of 0.02 g's. It should be noted that these control configurations included turn coordination features which were also changed in the modified SCAS to help reduce the lateral accelerations at the pilot.

The hashed boundary in Fig. III-15 is a criterion proposed by Boeing (Ref. 28) based upon their commercial transport experience. This criterion states in part:

> "Lateral acceleration at the pilot station shall not exceed a level of ± 0.075g peak, and the critical passenger station shall not exceed ± 0.05g peak. These levels shall be met for all normal maneuvers including 30 degree bank and capture using an average roll rate of 5°/sec in cruise and 10°/sec at landing.

> If unpiloted time studies are conducted, the wheel input should be a 0.5-second ramp of magnitude sufficient to produce the specified average roll rates."

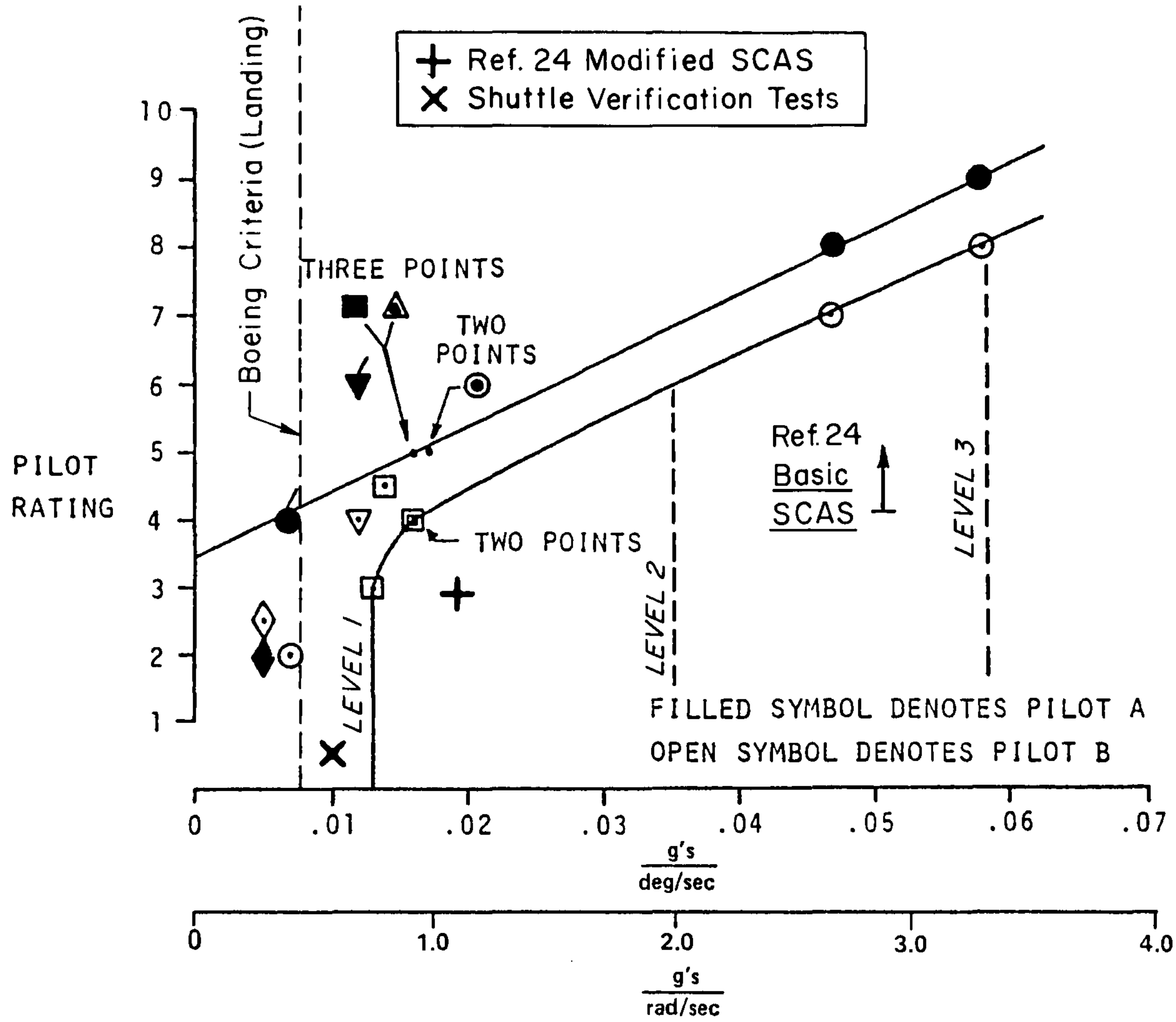
Note the Langley SST simulation with the modified flight control system lies in the Boeing acceptable region while the basic command augmentation system is above the acceptable boundary. However, if the maneuver were increased to achieve 10 deg/sec roll rate it would appear that both configurations would exceed the Boeing criteria.

Limits on lateral acceleration at the pilot station are not included in any of the Shuttle specifications although the problem was given early exposure in Ref. 35. The pilot station acceleration from two Shuttle Orbiter subsonic verification tests are shown by the X's on Fig. III-15. These show the pilot station 68 ft ahead of the vehicle c.g. experiences about 0.15 g peak acceleration for a roll rate of 20 deg/sec. The 0.8M case was obtained with nominal aerodynamic coefficients, the 0.6M cases identified as LVAR 9 and 23 have off-nominal cases having proverse aileron yaw, which could influence the lateral acceleration somewhat. It is apparent that these responses would just meet the Boeing criteria if the roll rate command were reduced appropriately.

One problem with the boundary and requirements of Fig. III-15 is that the severity of maneuver is not adequately taken into account. A second criterion was proposed in Ref. 23 with the allowable lateral acceleration at the pilot ratioed to the maximum roll rate developed in the maneuver. The input is a step roll rate command and peak lateral acceleration and peak roll rate are measured within the first 2-1/2 secs of response. Figure III-16 presents a plot of the response ratio versus pilot rating obtained from experiments reported in Ref. 23. Results of these experiments showed a steady degradation in handling quality ratings as the parameter values increase. The criteria proposed in Ref. 23 were Level 1 for values less than 0.012 g per degree/second; Level 2 for values from 0.012 to 0.035 g per degree/second; and a Level 3 boundary at approximately 0.058 g per degree/second. The appropriate response ratios from the Langley configurations of Fig. III-14 and the Shuttle configuration shown in the Fig. III-15 are spotted in Fig. III-16. The basic SCAS (which received a rating of unacceptable) has a parameter value of 0.05 and is found to lie between Level 2 and Level 3 boundaries in the Ref. 23 criteria. The modified SCAS (solid cross) is seen to lie between Level 1 and Level 2 but was given a HQR 3 in the TIFS. The X identifies parameter values for the two Shuttle configurations from Fig. III-15. No pilot ratings are available but with this criterion they lie within the Level 1 region whereas in Fig. III-15 they were outside the acceptable regions but would scale down to meet the Boeing criteria. Obviously, there is significant difference between the criteria of Figs. III-15 and III-16. The only consistent prediction between the two is the basic SCAS which is judged unacceptable by both. The problem is which is the more appropriate for the Shuttle Orbiter. The data points of Fig. III-16 were reported to be obtained using the Calspan TIFS aircraft and presumably simulating the SCR vehicle. Thus, the results should be representative of transport size aircraft with their high inertia and relatively low roll rates. At a given lateral acceleration, the criterion of Fig. III-16 indicates the flying quality rating can be improved by increasing the peak roll rate. That is, the criteria of Fig. III-16 reflect the need for some measure of harmony between lateral acceleration at the pilot and roll rate. The problem remains as to which criteria may be the more appropriate for Shuttle type vehicles.

C. COMMAND PREFILTER

As indicated previously, command prefiltering can reduce the rolling acceleration but will introduce phase lag. Figure III-17 presents curves of Cooper-Harper rating versus break frequency of a first order lag command filter obtained from Refs. 25 and 26. The Ref. 25 data reflect actual landing flare maneuvers in the NT-33. Two roll mode time constants were simulated. The squares reflect a roll time constant of 0.3 sec and the circles a roll mode time constant of 0.8 sec. These data indicate that the flying quality rating degrades rapidly as the prefilter lag inverse time constant moves from 10 to 1 rad/sec. It also indicates that the vehicle with the larger roll mode time constant is



$$\left.\frac{n_{y_{pilot_{max}}}}{p_{max}}\right|_{\substack{\text{Step Input} \\ t \leq 2.5 \text{ sec}}}$$

NOTES:

1. Flagged points are configurations specifically downgraded by Pilot A due to poor Dutch roll damping - not lateral acceleration.
2. The lines indicate degredation in pilot rating to be expected because of ride qualities for an airplane with otherwise satisfactory flying qualities parameters.

Figure III-16. Lateral Acceleration Criterion Versus Pilot Rating from Reference 23

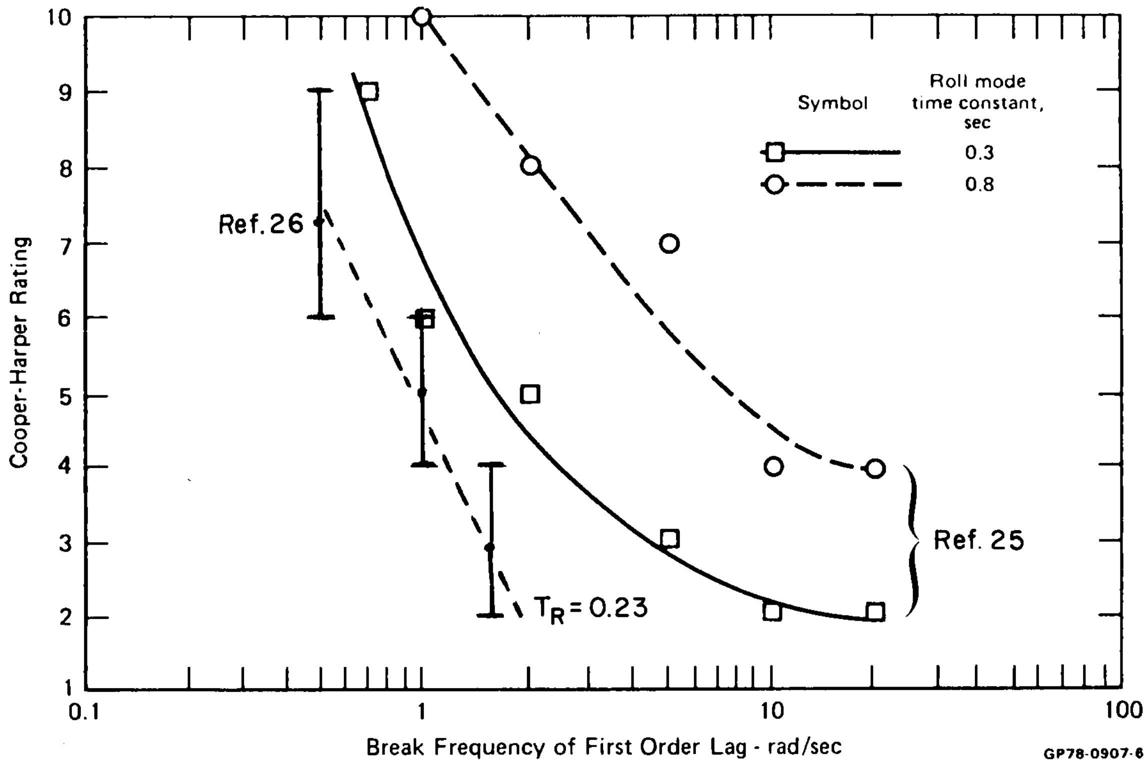


Figure III-17. Effect on Pilot Rating of First Order Lag Roll Rate Command Prefilter

more sensitive to the lag introduced by the prefilter. An additional set of data from Ref. 26 reflects moving base, ground simulation of a large twin engine transport aircraft with a roll mode time constant of 0.23 sec. The average ratings from three pilots is indicated by the dashed line on the left of Fig. III-17. The rating spread between the pilots is indicated by the bar. These two sets of data are consistent in two aspects. First, reducing the prefilter break frequency rapidly degrades flying qualities. Second, as the roll mode time constant increases, the prefilter frequency must also be increased to achieve acceptable flying qualities. Since both of the parameters of concern involve lag in the roll rate response to a command, the two are summed to reflect the total roll response lag to a roll rate command and plotted in Fig. III-18 versus the Cooper-Harper rating from Fig. III-17. The NT-33 results coalesce somewhat. With the exception of three data points, the ratings can be fitted by a single curve such as the solid

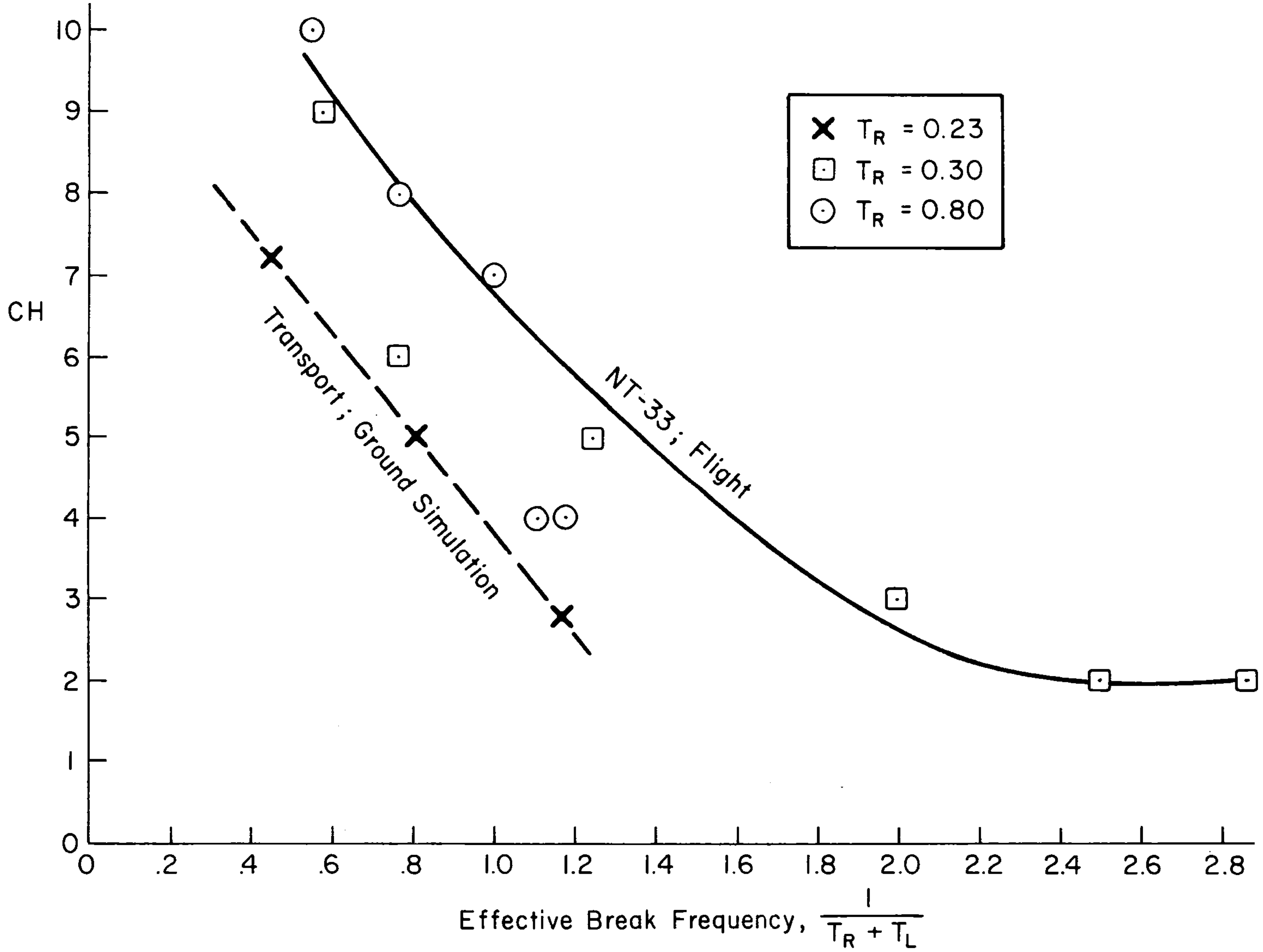


Figure III-18. Degradation in Pilot Rating with Total Effective Roll Lag; Approach and Landing line in Fig. III-18. The data points from the large twin engine transport moving base simulation remain separated from most of the NT-33 data. The spread between the two sets of data could be due to aircraft size or to simulation artifacts such as display or stress level. That is, the NT-33 in actual landing flare could be considered a high stress situation, whereas the ground based simulation approach and landing would be low stress. This has been suggested previously to explain differences in flying quality ratings obtained in simulation versus actual flight. Although there is insufficient data to draw conclusions, it might also be argued that pilots generally expect lower or more sluggish response of large aircraft and therefore the lower effective break frequency would be more acceptable. Thus one may project that the differences in the two curves reflect vehicle class or size consideration.

Prefilter break frequencies for two of the latest fly-by-wire fighter aircraft, the Shuttle Orbiter, and the SST simulated at Langley are summarized in Fig. III-19. Both fighters have the roll rate prefilter at 5 rad/sec and the pitch-rate at somewhat higher frequency. This probably reflects the desire for higher bandwidth control in pitch. From Fig. III-17 it may be observed that a 5 rad/sec prefilter may degrade the flying quality rating (neither of these aircraft are considered to have excellent flying qualities in the landing approach). The Shuttle Orbiter roll rate prefilter is at 10 rad/sec and from Fig. III-17 would indicate little or no flying qualities degradation due to the filter. On the other hand, the Langley SSCT simulator had the prefilter at 1.43 rad/sec. Upon first glance at Fig. III-17 one might expect this would degrade the flying quality rating rather than improve it. But it might also be noted that this break frequency is the same as one which produced Cooper-Harper ratings as high as 2 and 3 in the Ref. 26 transport aircraft simulation. Thus again, aircraft size may be involved.

D. LATERAL RESPONSE TIME DELAY

As noted previously, the Shuttle Orbiter flying quality and flight control specifications do not include a criterion for time delay between

AIRCRAFT	LONGITUDINAL	LATERAL
SHUTTLE	NONE	$\frac{1}{(s+10)}$
FIGHTER A	$\frac{1}{(s+8.3)}$	$\frac{1}{(s+5)}$
FIGHTER B	$\frac{1}{(s+6.67)}$	$\frac{1}{(s+5)}$
LRC/SSCT SIMULATOR	NONE	$\frac{1}{(s+1.43)}$

Figure III-19. Typical Prefilter Lags

the command input and vehicle response. Such criteria are contained in the new MIL-8785C and the SCR criteria of Ref. 23. The time delay criteria of these two documents are plotted as boundaries against the Cooper-Harper scale in Fig. III-20. The 8785C Level 1, 2 and 3 boundaries are shown in Fig. III-20 by the solid lines. A diagonal from the origin shows the time delay values selected coincide with Cooper-Harper 3.5, 6.5 and 8.5 rating points, respectively. The Level 1, 2, and 3 criteria of Ref. 23 are shown in Fig. III-20 as the dashed lines. A diagonal intersecting these lines at 3.5, 6.5, and 8.5 intersects the time delay scale in the vicinity of 0.1 sec. Thus, the Ref. 23 requirements appear to allow approximately 100 msec before the onset of flying quality rating degradation.

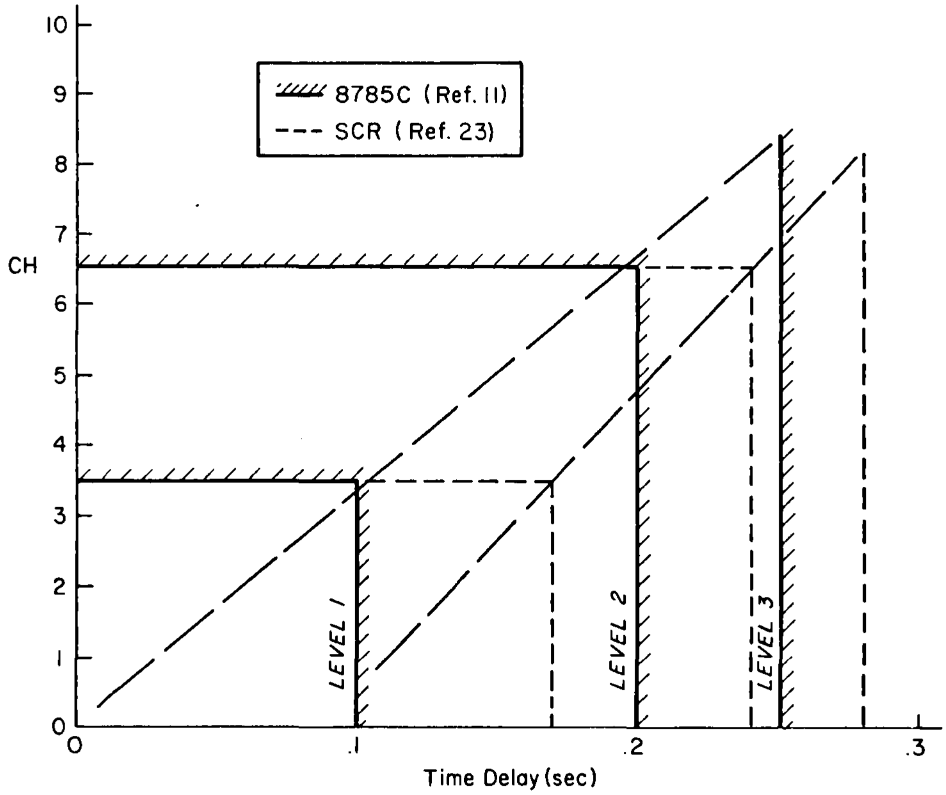


Figure III-20. Lateral Response Time Delay Criteria Comparison

Only two sets of lateral time delay experimental data were found and these tend to support some time delay threshold before a flying quality rating degradation is incurred. The two data sets are reflected in Fig. III-21. Again, these are from Refs. 25 and 26. As before, the squares represent the 0.3 sec roll mode time constant and the circles the 0.8 sec time constant. The other curve in Fig. III-21 is the Ref. 26 transport aircraft in the moving base approach and landing simulation. Interestingly, separate straight lines fitted to the larger time delay data points would intersect the time delay axis in the vicinity of 0.13 to 0.15 sec. The sensitivity to increasing time delay appears to be

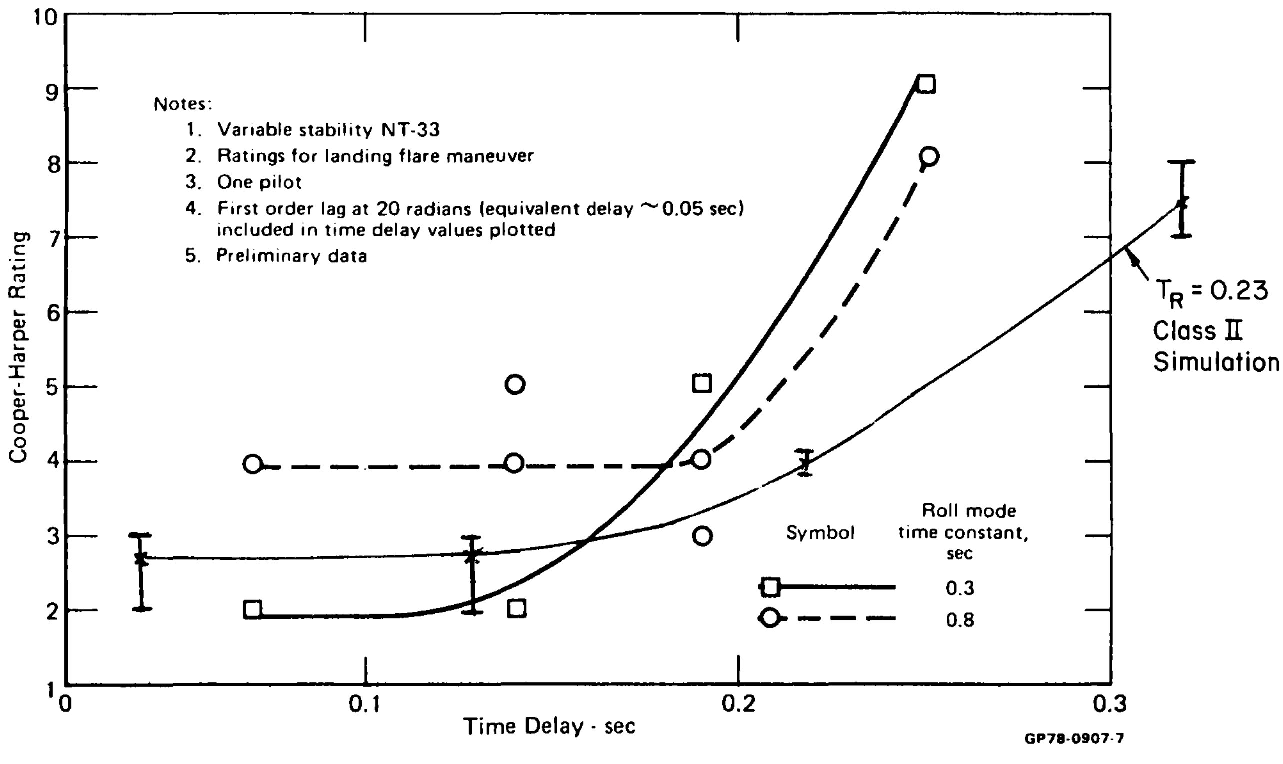


Figure III-21. Effect on Pilot Rating of Time Delay in Lateral Response considerably greater for the NT-33 (actual flight maneuver) than for the same task in the transport (ground simulation). Again, this could be due to a difference in stress level between flight and moving base simulation as was noted for the longitudinal axis in Fig. II-23. Nevertheless the two sets of data tend to agree on an allowable threshhold on the order of 140-150 msec which is about mid-way between the two Level 1 criteria of Fig. III-20.

The lateral time delay degradation effects are compared to longitudinal in Fig. III-22. Both sets of data are from the NT-33 and indicate very similar degradation in pilot rating with increasing time delay, including approximately the same threshold effects.

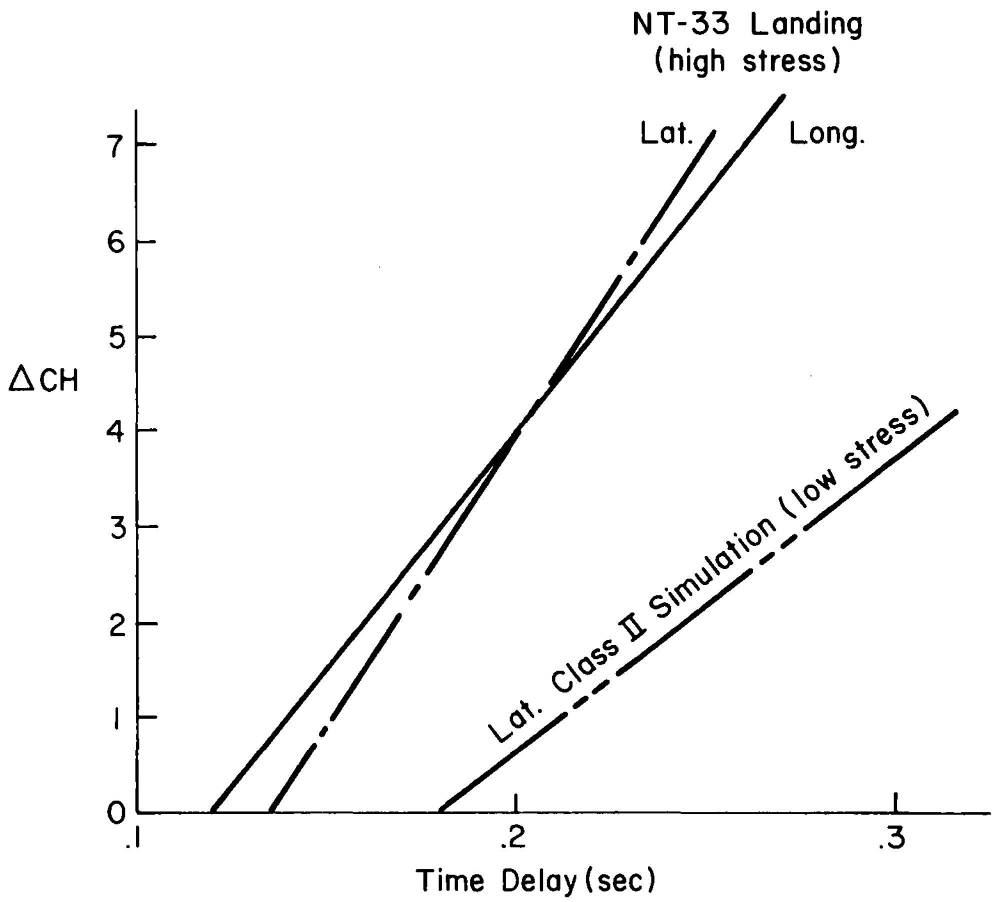


Figure III-22. Comparison of Effect of Time Delay on Pilot Ratings for Lateral and Longitudinal Control Tasks

Interestingly, the effective time delays shown in the Shuttle verification tests (Figs. III-4 through III-8) are all of the order of 0.3 sec or greater. Furthermore, the Shuttle roll rate response criteria of Figs. III-1 and III-2 allow effective time delays up to about 0.8 sec. Such delays are totally unacceptable on the basis of Fig. III-21 data. Since there is no available evidence that the Shuttle Orbiter flying qualities in roll control are totally unacceptable, then one must question the criteria of Refs. 11 and 23 or their applicability to Shuttle-like vehicles.

E. TIME TO ROLL

Typical Shuttle Orbiter time traces for a roll maneuver in the landing condition are shown in Fig. III-23. These are from the verification tests of Ref. 9. The roll rate command is about 20 deg/sec; the time to bank 30 deg is 2.2 sec. The traces also show a time delay of approximately 500 msec before the roll begins to respond. The fourth trace from the top indicates a peak lateral acceleration at the pilot to be 0.15 g's at about the time the bank angle begins to change. The lateral acceleration/roll rate ratio that was plotted previously in Fig. III-16 was obtained from these traces. The bottom time trace shows the lateral acceleration at the c.g. to be roughly half that at the pilot's station.

The Shuttle flight control system specification contains a criterion for time to roll 30 deg as a function of Mach number. This is shown in Fig. III-24. For Mach numbers less than 0.6, the allowable time is 2.5 sec. For Mach numbers above 1.5, the time should not exceed 7 sec. Spotted on this plot are three data points from the Shuttle verification tests. These verify that the Shuttle meets its specification. Also, six data points for the supersonic Concorde (extracted from Ref. 23) show that between 0.4 and 1.8 Mach, the Concorde also meets this Shuttle specification. The Shuttle and Concorde roll performance appear quite consistent.

The 8785C roll performance requirements for Class 3 aircraft are presented in Fig. III-25, with the requirements for time to bank 30 sec in non-terminal flight phases plotted versus the roll subsidence time

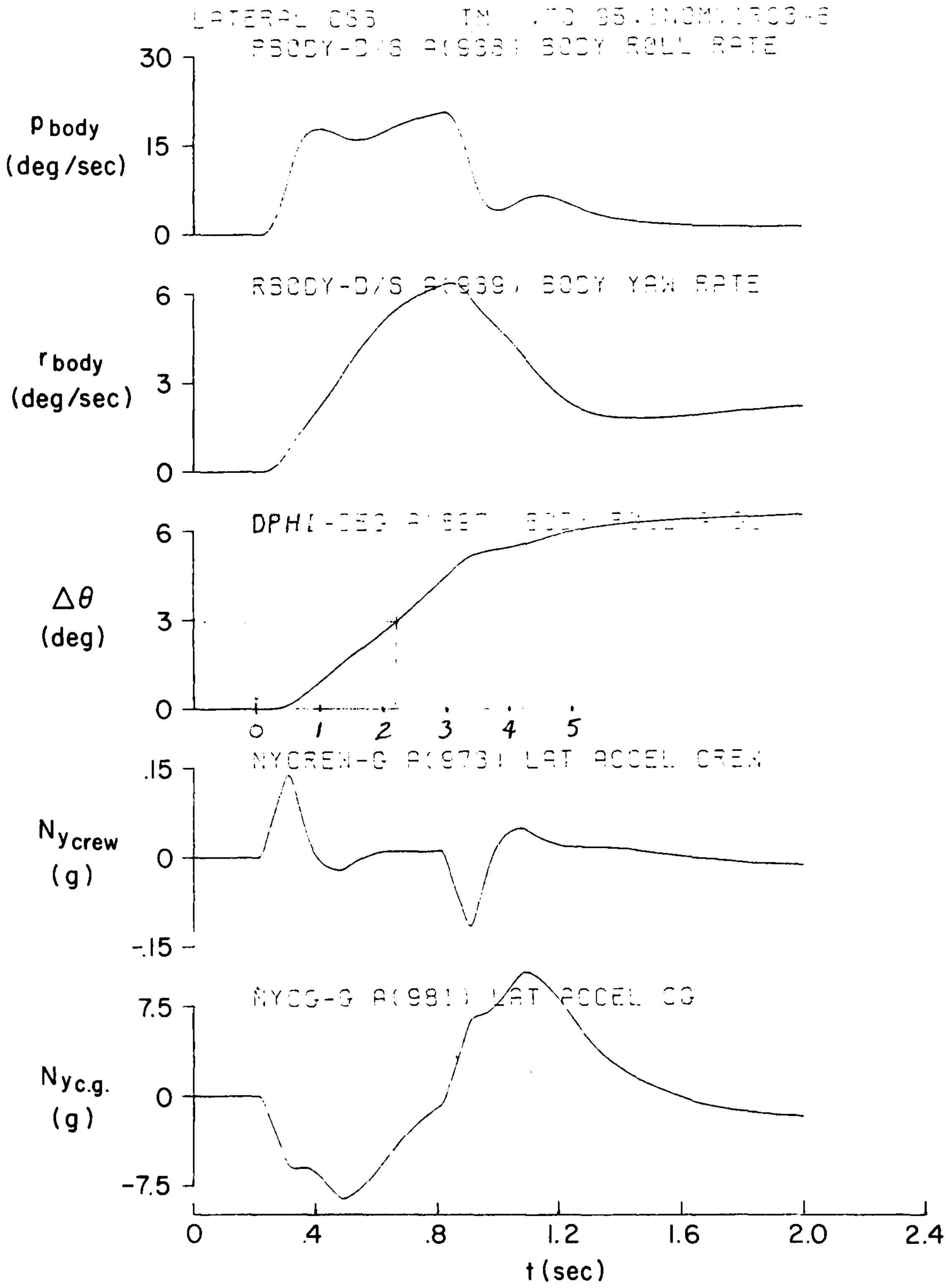


Figure III-23. Orbiter Lateral Response to Step Roll Rate Command; Landing Flight Condition

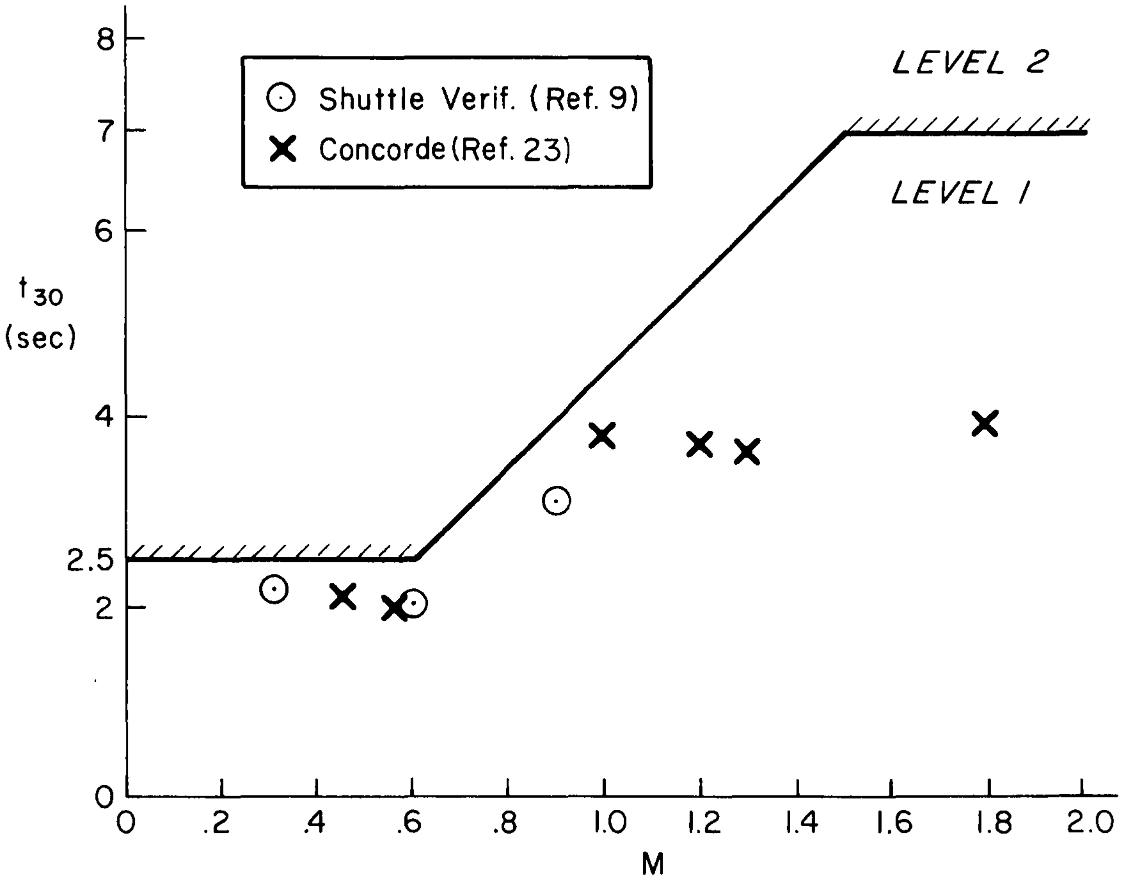


Figure III-24. Orbiter F.C.S. Specification Time to Bank 30 Degrees

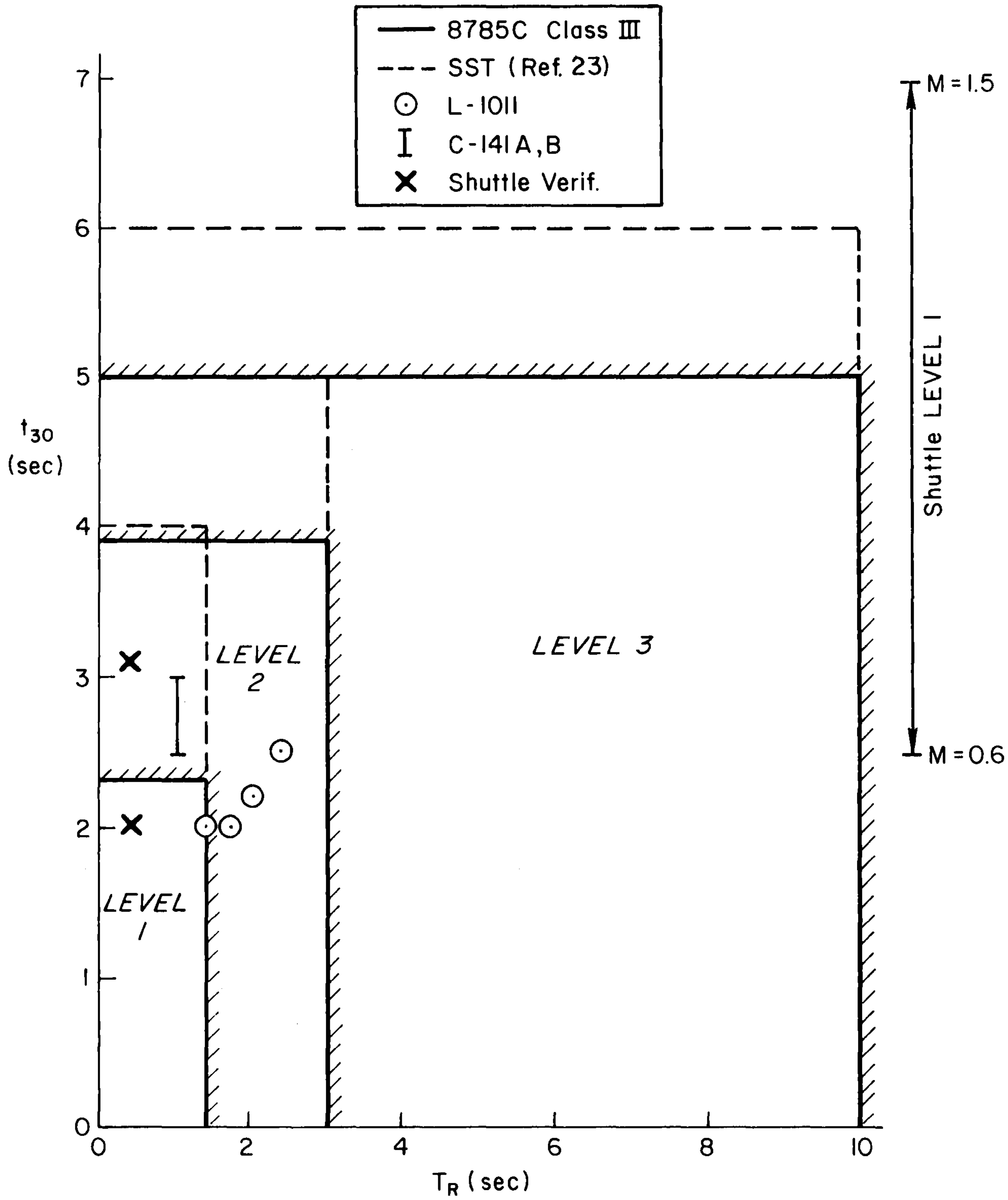


Figure III-25. Roll Response Criteria; Non-Terminal Flight Phases constant boundaries. The rectangles define regions of Level 1, 2, and 3 flying qualities. The dashed lines define expanded regions for the same levels as proposed for the supersonic transport in Ref. 23. Spotted on this plot are the results from flight test of the Lockheed L1011 and C141A/B aircraft. All three of these aircraft fall within the Level 2 boundaries of 8785C. None of the data points are actually within Level 1 requirements. This suggests that the 8785C requirements may be too strict for Class 3 (very large) aircraft. Also spotted on the plot are the results from two of the Shuttle verification simulations. One falls within the Level 1, the other falls within Level 2. Both are consistent with the range of time to bank results obtained with the L1011 and C141 aircraft. These also suggest that the time requirements to bank 30 deg might be relaxed somewhat for the Shuttle.

Similar roll response criteria for the Category C or landing flight phase is shown in Fig. III-26. In this plot the 8785C boundaries separating Levels 1 and 2 are shown for Class 2 and Class 3 aircraft. The Ref. 26 roll rate response parameter $t_{63\%}$ is used in place of the roll mode time constant as in Fig. III-25. The data points are from two transport aircraft landing simulations. The lower plot indicates Trials 1-4 and the numbers in parentheses beside each of the data points is the pilot rating assigned for the rolling maneuver response. In trials T1-T4 the maximum roll rate response was limited at 36 deg/sec. The upper data points, R1-R4, reflect a roll rate limit of 18 deg/sec. Data points T1, T2, R1, and R2, tend to indicate via the pilot ratings that the Level 1 roll rate response boundary ($t_{63\%}$ or T_R) should be in the vicinity of 0.7 sec rather than 1.4 sec. These results are not consistent with those from the L-1011 and C-141A/B flight tests (Fig. III-25).

The data of Fig. III-26 do indicate the time to bank distinction between Class II and III aircraft may be unnecessary.

In all, it appears from Fig. III-25 and III-26, that additional data are needed for defining the roll response criteria for large aircraft.

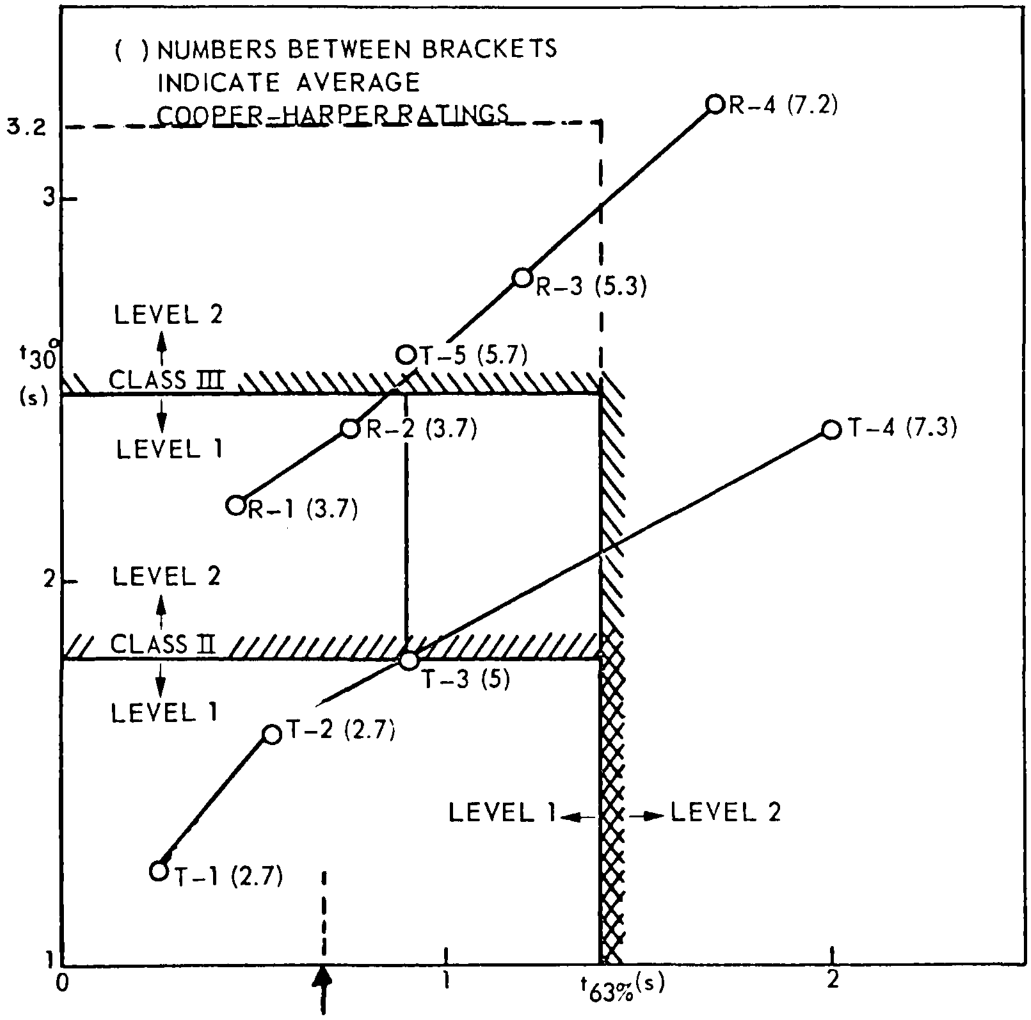


Figure III-26. Roll Response Criteria; Landing Phase

F. SUMMARY

In summary, this preliminary review of the literature and available data has identified four lateral flying qualities areas that may be tentatively identified as weak. The first is the roll rate response boundaries as currently set forth in the Shuttle Orbiter flying qualities and flight control system specifications. These boundaries appear to be rather broad and really could stand refining and narrowing.

There is indication that there may be a tradeoff between flying quality and ride quality requirements. A major problem occurs for vehicles which roll about the velocity vector and are flown at large angles of attack and/or with the pilot located far in front of the vehicle c.g. Configurations that meet the current roll rate response boundaries may be found unacceptable because of large lateral accelerations at the pilot station. Lateral acceleration at the crew station is currently unspecified for the Shuttle Orbiter. Two ways of specifying the allowable lateral acceleration have been examined and neither appears to be completely satisfactory. Certainly additional tests and data are required to pin this down.

The equivalent time delay between pilot command of a maneuver and the actual vehicle response also is currently unspecified in any of the Shuttle Orbiter documents although it may be inferred from the roll rate response boundaries. There are two criteria presently available in the literature, one of which allows some time delay threshhold, the other does not. There also is evidence that the criteria might be related to either the vehicle size or to a particular task. The current criteria are questionable and again there is need for additional data and refining of the requirements. It should be noted that roll rate response and equivalent time delay combine to form a limit to achievable closed loop bandwidth in both roll and path control. That is, increasing equivalent time delay and/or decreasing roll rate response reduce the closed loop bandwidth control that the pilot can obtain in either the roll or path control tasks. Further experiments should address this interrelationship.

SECTION IV

FLIGHT CONTROL SYSTEM CRITERIA REVIEW AND ASSESSMENT

Several key flight control system criteria have already been covered in the preceding flying qualities review. This subsection will be devoted to additional items which influence closed loop stability, handling, and ride qualities. It will focus on six performance related requirements or criteria of the Ref. 10 Shuttle Orbiter flight control system specification. These are: stability margins of the various control system feedback loops, automatic turn coordination performance, residual oscillations, failure transients, gust sensitivity, and control sensitivity. These will be treated in each of the subsections to follow.

There is little hard data against which the Ref. 10 requirements or criteria can be compared. The major source for similar criteria is Ref. 29, the U.S. Air Force Flight Control System Specification, MIL-F-9490D. This specification treats augmentation systems but does not cover complete fly-by-wire type control. The U.S. Navy flight control system specification (Ref. 30) is totally outdated and has no criteria pertinent to highly augmented or fly-by-wire aircraft. At this time both of these specifications are undergoing updating; however, neither revision has progressed to the point of providing additional criteria for this review. Criteria and design guides pertinent to sidestick type controllers were found in Refs. 31 and 32. These sources are augmented wherever possible by available information from specific flight control systems.

Before getting into the Ref. 10 requirements, it is pertinent to comment on the performance levels identified in that specification and used in the preflight system verification tests. These are summarized in Fig. IV-1. Level 1, 2, and design assessment performance levels are designated. Note that in a quad redundant, fly-by-wire system, degraded performance should only result from multiple, similar, control system

"LEVEL 1"

- SPECIFIED STABILITY MARGINS (TYPICALLY 6 dB, 30 DEG) & RESPONSE CRITERIA
- PILOT RATING (COOPER-HARPER) - 3 OR LESS

"LEVEL 2"

- DEGRADED STABILITY MARGINS (4 dB, 20 DEG)
- LARGE SIGNAL OPERATION - STABLE
- DEGRADED TURN COORDINATION (RELAXED LATERAL ACCELERATION & SIDESLIP CRITERIA)
- PILOT RATING - 6 OR LESS

"DESIGN ASSESSMENT"

- NO LOSS OF VEHICLE

Figure IV-1. Shuttle Orbiter Performance Levels out-of-tolerance conditions, or from the vehicle aerodynamic characteristics being considerably different from those predicted. Reference 9 shows the difference between Levels 1 and 2 to reside primarily in no degradation versus FCS component 3σ out-of-tolerance buildup combined with large aerodynamic variations. Any lesser tolerance buildup still must meet Level 1 performance requirements. An additional Level 1 requirement is that the flying qualities be rated 3 or better on the Cooper-Harper scale while the maximum component out-of-tolerance buildup and aerodynamic uncertainty case still must achieve a Cooper-Harper flying quality rating of 6 or better. Thus, each level requires a specific loop-by-loop stability margin within the flight control system for all automatic functions <u>and</u> an additional flying quality consideration where the pilot interacts. This is considerably different from the Ref. 11 specification which relates Level 1 and 2 flying quality rating requirements with flight control system failure states. In the case of the Shuttle Orbiter, flight control system failure (i.e., multiple, similar, control system component failures) would undoubtedly result in loss of the vehicle and this is legislated against by the "design assessment" performance level.

A. STABILITY MARGIN

Closed loop stability margin requirements for the Orbiter (SDM Para. 3.4.1.9.3.2) and MIL-F-9490 (Para. 3.1.3.6.) are shown in Fig. IV-2. These specifications are quite similar but the military specification may be somewhat the tighter. For instance, the Orbiter SDM Level 1 requirements for frequencies less than 6 Hz are -6 dB amplitude and +30 deg phase module. For frequencies above 6 Hz, it only requires -6 dB amplitude margin. For Level 1, the phugoid should not have a time to double amplitude less than 55 sec, or a spiral with time to double amplitude less than 12 sec. Level 2 requires stability margin of -4 dB amplitude and +20 deg phase margin for all frequencies. The Mil specification requires frequencies less than the first elastic mode to have 6 dB gain margin and 45 deg phase margin. For frequencies above the first elastic mode it requires 8 dB and 60 deg phase margin. There is no Level 2 requirement given in 9490.

SDM: 3.4.1.9.3.2		9490: 3.1.3.6	
Level 1		Level 1	
$\omega < 6$ Hz	-6 db +30 deg	$\omega <$ 1st EM	-6 db +45 deg
$\omega > 6$ Hz	-6 db		
phugoid	$T_2 > 55$ sec	$\omega >$ 1st EM	-8 db +60 deg
spiral	$T_2 > 12$ sec		
Level 2		Level 2	
all ω	-4 db +20 deg	?	
Design Assessment			
No loss of vehicle			

Figure IV-2. Stability Margin (Closed Loop)

B. AUTOMATIC TURN COORDINATION

Figure IV-3 indicates the automatic turn coordination requirements of the Orbiter SDM. These are separated into two flight regimes. The first covers the initial entry, hypersonic, and supersonic regions down to about 2.5M. Only Level 1 is specified and that requires maintaining sideslip within 1 deg. For the terminal area energy management and approach and landing regions, two levels are given and Level 1 requirements are further separated into steady turning (at zero roll rate) and steady rolling conditions. Note that lateral acceleration levels are specified for the c.g. and not at the pilot or crew station. Referring back to the Fig. III-23 roll maneuver time traces, a lateral acceleration at the c.g. of less than 0.1 g can result in 0.15 g at the crew station.

MIL-F-9490D has no requirement for automatic turn coordination. It refers to MIL-F-8785 where the requirement is in terms of manual coordination (e.g., pedal force).

C. RESIDUAL OSCILLATIONS

Figure IV-4 provides a comparison between the Orbiter and Air Force specification for residual oscillations at the pilot station. Both specifications have a Level 1 requirement but there is no indication of a Level 2 requirement. The specifications are quite similar except that 9490 refers to MIL-F-8785 for pitch oscillation limits which, in turn, are set at ±3 Mils (0.17 deg) for flight phases requiring precise attitude control.

D. FAILURE TRANSIENTS

Figure IV-5 has a comparison of the Orbiter and Air Force specifications for failure transients. Here there are several differences between the two specifications. The main consideration is that the Orbiter has the same requirements following the first <u>and</u> second failures. This is consistent with a quad-redundant fly-by-wire type

SDM: 3.4.1.9.2	9490: No Requirement
<u>ENTRY</u>	(8785C manual coordination within rudder pedal force criteria)
<u>Level 1:</u> $\beta < 1^0$	
<u>TAEM & AL</u>	
<u>Level 1:</u>	
$p = 0 \quad \begin{cases} a_{y_{cg}} < 0.03\ g \\ \beta < 1^0 \end{cases}$	
$\Delta\phi = 45^0$, $p = 20\ ^0/sec$, $\bar{q} > 125\ psf \quad \begin{cases} a_{y_{cg}} < 0.1\ g \\ \beta < 2^0 \end{cases}$	
$\bar{q} < 125\ psf \quad \begin{cases} a_{y_{cg}} < 0.15\ g \\ \beta < 3.5^0 \end{cases}$	
<u>Level 2:</u>	
$a_{y_{cg}} < 0.1\ g$	
β < 6° adverse	
< 2° proverse	

Figure IV-3. Automatic Turn Coordination

SDM: 3.4.1.9.4	9490: 3.1.3.8
Level 1	**Level 1**
$\theta < \pm 0.25^0$	$\theta < \pm 3$ mil (0.17°)
$\phi < \pm 0.5^0$	$\phi < \pm 0.6^0$
	$\psi < \pm 0.6^0$
$n_z < \pm 0.05g$	$n_z < \pm 0.04\,g$
$a_y < \pm 0.02g$	$a_y < \pm 0.02\,g$
Level 2	**Level 2**
?	?

Figure IV-4. Residual Oscillations at Pilot Station

SDM: 3.4.1.5.2	9490: 3.1.3.3.4
For $t \ \overline{<} \ 2$ sec following 1st <u>and</u> 2nd failures:	$t \ \overline{<} \ \underline{ND}^{*}$ sec
	<u>Level 1 & 2:</u>
$\dot{\theta} < \pm 0.005 \ \bar{q} \quad ^{o}/sec$	$n_{z_{cg}}$ or $a_{y_{cg}} < \pm 0.5$ g
$\dot{\phi} < \pm 0.01 \ \bar{q} \quad ^{o}/sec$	$\dot{\phi} < \pm 10 \ ^{o}/sec$
$\beta < \pm 2^{0}$	
	<u>Level 3:</u>
	$n_{z_{cg}} < 75\% \quad n_{z_{limit}}$
	<u>or</u>
	$\Delta \ n_{z_{cg}} < \pm 1.5$ g
	*9490 refers to 8785 8785 gives no specific time

Figure IV-5. Failure Transients

system. The main considerations are on sideslip excursions and the pitching and rolling rates of the vehicle. The latter are tied to the dynamic pressure at which the vehicle is operating. Even at dynamic pressures of 200-300 psf the allowable transients are very mild. All of these are further time limited to less than 2 sec following the failure. The Air Force specification (9490) refers to the flying qualities specification (8785) for time limitation and, in turn, 8785 lists none. 9490 does identify three performance levels and these relate to accelerations at the c.g., either normal or lateral. The Level 3 requirement is to prevent breaking of the airframe.

E. GUST SENSITIVITY

The Orbiter flying quality specification (Ref. 6) refers to a gust sensitivity requirement to be determined by the Contractor. No further indication of a gust sensitivity requirement has been found except for a briefing chart prepared by Rockwell International in support of verification tests performed on the simulator. The actual source of the requirement has not been determined. However, the chart indicates that the gust induced motion at the pilot station shall not exceed a normal acceleration of 0.03 g's rms per ft/sec gust or a lateral acceleration of 0.015 g's rms per ft/sec applied gust.

F. CONTROLLER COMMAND SENSITIVITY

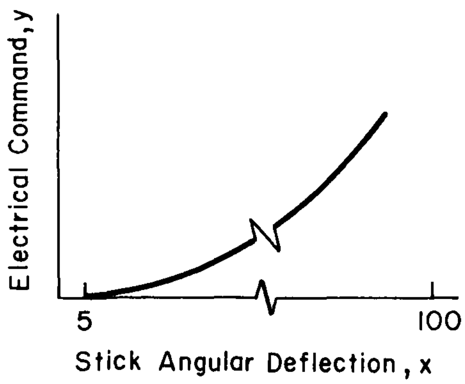
Figure IV-6 summarizes the Ref. 10 control sensitivity requirement for RHC software command in response to RHC angular deflection. This in effect specifies a threshold of 5 percent of full deflection before an output results and, after exceeding the 5 percent threshold, the output shall be a quadratic function of input. The force-displacement and roll rate command characteristics as mechanized in the RHC and flight control system software is shown in Fig. IV-7. On the left, the force-displacement characteristics are shown to have a constant force gradient up to approximately 2/3 of the controller displacement. At this point an intermediate or "soft" stop is reached. Further application of force will allow the final 1/3 stick displacement. The electrical signal out

SDM: 3.4.1.9.5

RHC software command in response to angular deflection (x):

$$y = 0 \quad ; \quad x < 5\%$$

$$y = A(x-5) + B(x-5)^2 \quad ; \quad x > 5\%$$



No similar requirement found

Figure IV-6. Control Sensitivity

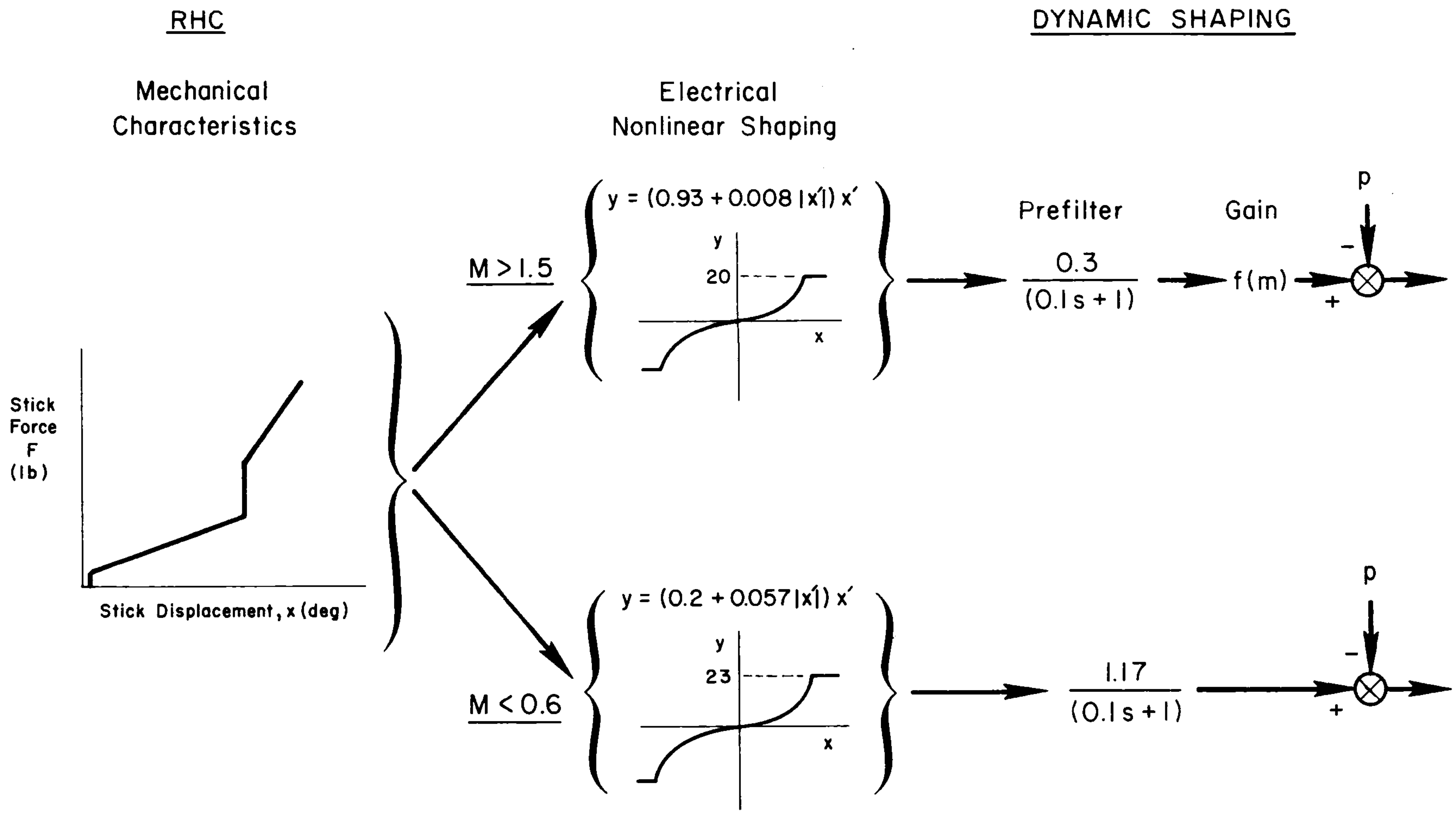


Figure IV-7. Roll Rate Command Characteristics of the RHC is proportional to stick deflection. The stick signal is then modified in the FCS to produce the Fig. IV-6 nonlinear shaping (note $x' = x - 1.1$ deg). One shaping is used for supersonic regimes and another is used for low subsonic. Between these, the coefficients are obtained using straight line interpolation as a function of Mach. The output of the shaping algorithm then is passed through a limit which is again a function of Mach. It then goes through the first order lag prefilter and another Mach gain before being compared with the roll rate and other feedback responses. The pitch/rate command is similar to that shown in Fig. IV-7 except that the force gradient is about 30 percent lower.

There is little data on sidestick controller characteristics against which the Orbiter mechanization can be compared. The most complete reference material derives from an investigation of force-deflection characteristics for side-stick controllers conducted in the Calspan NT-33 and reported in Refs. 31 and 32. The purposes of that investigation were to determine if it is necessary or desirable for side-stick controllers to have motion (displacement) for good flying qualities and, if so, to determine the amount of motion desired for different flight phases and piloting tasks. The tasks employed encompassed Category A (formation, air-to-air tracking, and aerobatic maneuvering) and Category C (ILS approach and touch and go landings). All tasks were flown in relatively calm air. Pilot commentary and flying quality ratings were given for both lateral and longitudinal tasks. The test matrix encompassed four force/command-response gradients (low, medium, high, and very high) and three stick displacement levels (fixed, small, and large). Unfortunately, the specific force, displacement, and electrical gain values employed did not provide a sufficiently large range of controller characteristics to allow complete assessment of or comparison with the Orbiter sidestick. That is, some Orbiter RHC characteristics fall outside those investigated. In addition, only roll rate command characteristics can be directly compared because the longitudinal input in the NT-33 commanded normal acceleration whereas pitch rate is commanded in the Orbiter.

The upper plot of Fig. IV-8 presents lateral sidestick force/command sensitivity characteristics in terms of torque (T_s) applied about the sidestick pivot point. The solid lines reflect the characteristics investigated in the NT-33 for approach and landing tasks. The Orbiter RHC subsonic torque/command curve is shown out to the soft stop and is seen to be somewhat higher than the torque/command gradient classified "Very High" in the NT-33 experiments.

The lower plot of Fig. IV-8 is a cross-plot of the initial torque/command gradient (L, M, H, VH) and the stick displacement to torque ratios (fixed, small, large) used in the NT-33 experiments. The combinations tested are identified by the circles. The numbers by the circles are the average Cooper-Harper handling quality ratings (HQR) given by two pilots. These indicate that in the NT-33 approach and landing tasks the light and medium force gradients were acceptable while the high and very high gradients were unacceptable.

While the Orbiter RHC torque/response feel is higher than the highest curve of the NT-33, its torque to displacement ratio is much smaller than that tested in the NT-33. Thus the Orbiter RHC displacement/force characteristic is relatively "free" and essentially the opposite of a "fixed" stick. Tracking experiments reported in Ref. 34 showed that a "free" sidestick induced more high-frequency phase lag on the part of the pilot and rms error were larger than for "fixed" sidestick configurations. This additional source of closed-loop lag has not been considered previously and could be a significant contribution to PIO tendencies in situations where large stick deflection is employed. The longitudinal torque/displacement ratio is also shown in Fig.IV-8 to indicate the "free" stick-induced latency may be even more significant in pitch control.

The F-16 sidestick development (Ref. 33) provides additional background information. But again, the applicability to the Orbiter is somewhat limited because the F-16 sidestick is a fixed (rigid) configuration whereas the Orbiter RHC has ± 19.5 deg deflection to the soft stops.

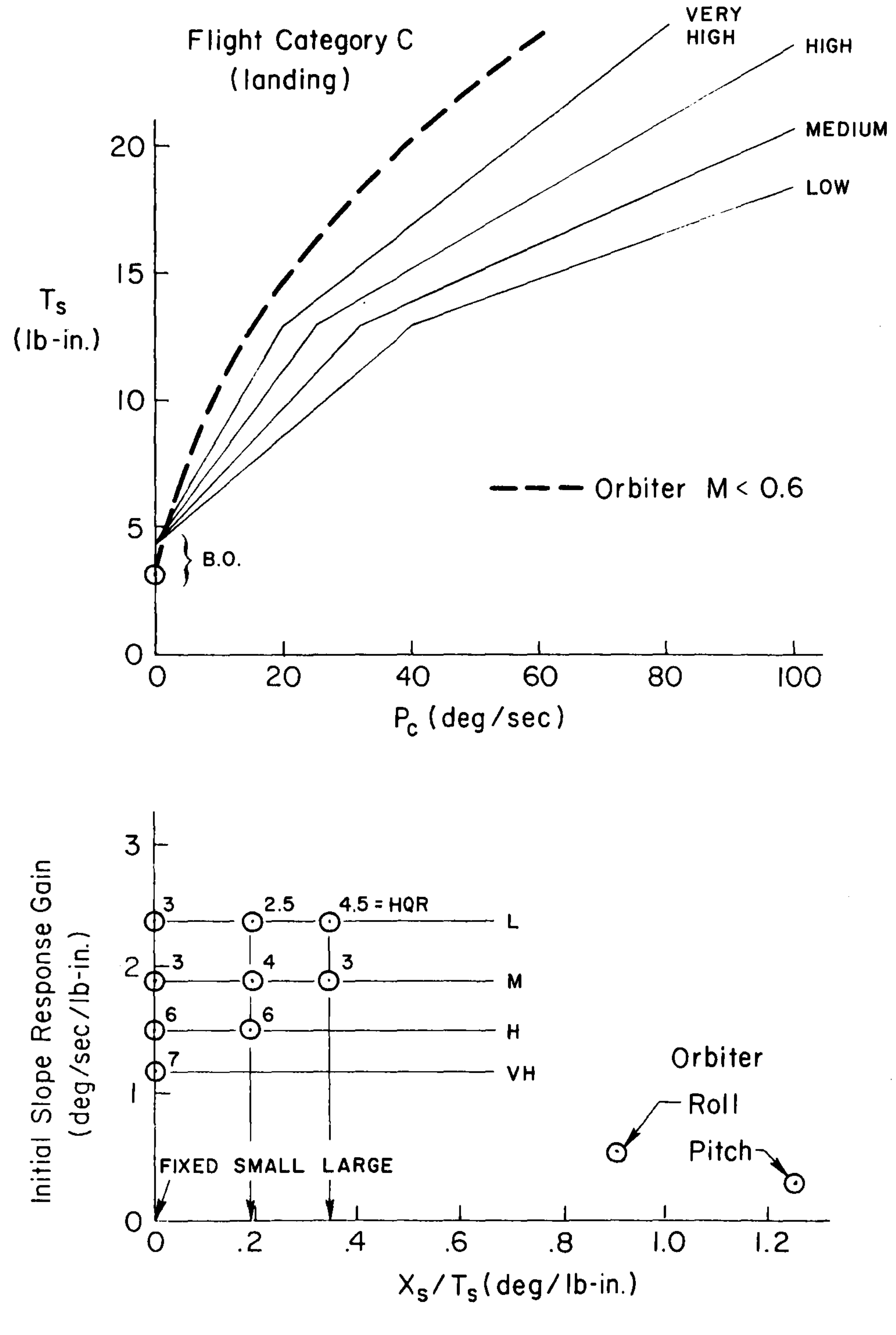


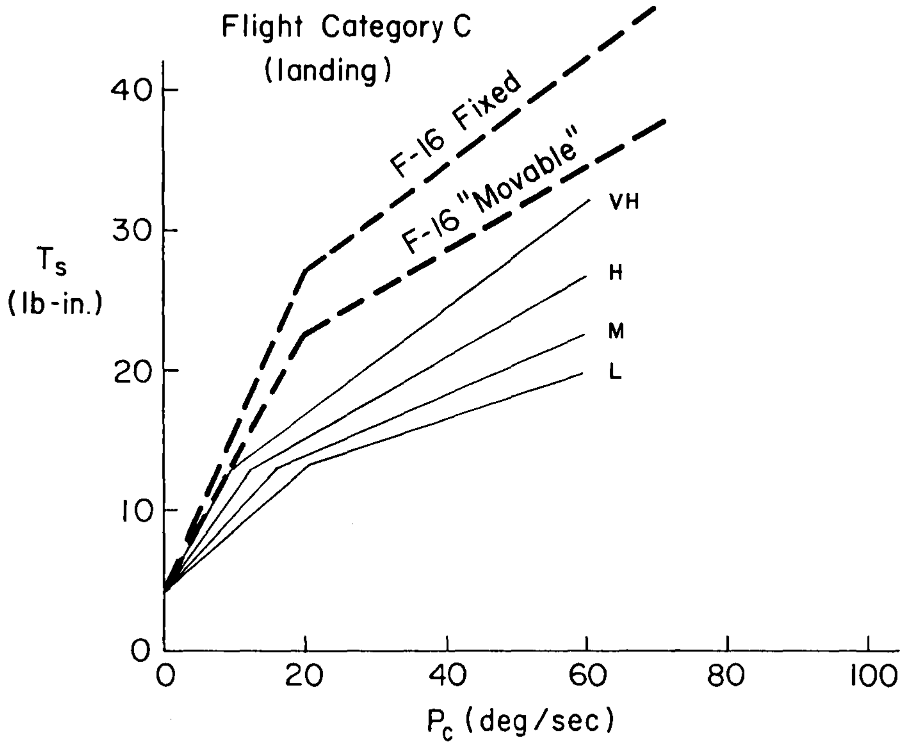
Figure IV-8. Shuttle Orbiter Lateral RHC Feel/Command Sensitivity Characteristics in Landing

Figure IV-9 shows two F-16 sidestick configurations, the original fixed (0.046 in. maximum displacement) and the later "movable" (0.122 in. maximum displacement), compared to the NT-33 data base. Note that the F-16 and Orbiter torque/command gradients are very similar. The original configurations were considered to be excessively tiring and lacking in cues to indicate when the command limit had been reached. The modified stick is rated improved in that the cue for command limit is better but is still has heavier forces than desired. The F-16 results thus support the NT-33 findings.

The Shuttle Orbiter RHC feel/command sensitivity characteristics are compared with the NT-33 results for up-and-away flight tasks in Fig. IV-10. Two sets of feel/sensitivity characteristics are represented. One is for low subsonic flight representative of approach lineup, the other for the supersonic beginning of the terminal area energy management regime. The RHC subsonic characteristics again exceed the NT-33 very high feel ratings and the supersonic regime RHC feel/sensitivity gradient is an order of magnitude from that considered very high in the NT-33.

The F-16 feel/sensitivity gradients are plotted against the NT-33 up-and-away results in Fig. IV-11. This shows the F-16 again to have very high force characteristics which are reported to be tiring to the pilot in gross maneuvering tasks.

Comparison of the upper plots of Figs. IV-10 and IV-11 show the F-16 sidestick characteristics to be almost identical to the Orbiter subsonic stick characteristics. This would tend to indicate the Orbiter RHC feel/sensitivity gradient in all flight regimes might also be tiring. However, under normal conditions the Orbiter attitude and maneuvering control task is so benign that the high force-to-command gradient may not be a fatigue factor. It would appear, however, that the large change in command sensitivity with Mach number might lead to pilot adaption problem and possible overcontrol (PIO) if the pilot were continuously controlling attitude during rapid deceleration from $M = 2$ to $M = 0.6$.



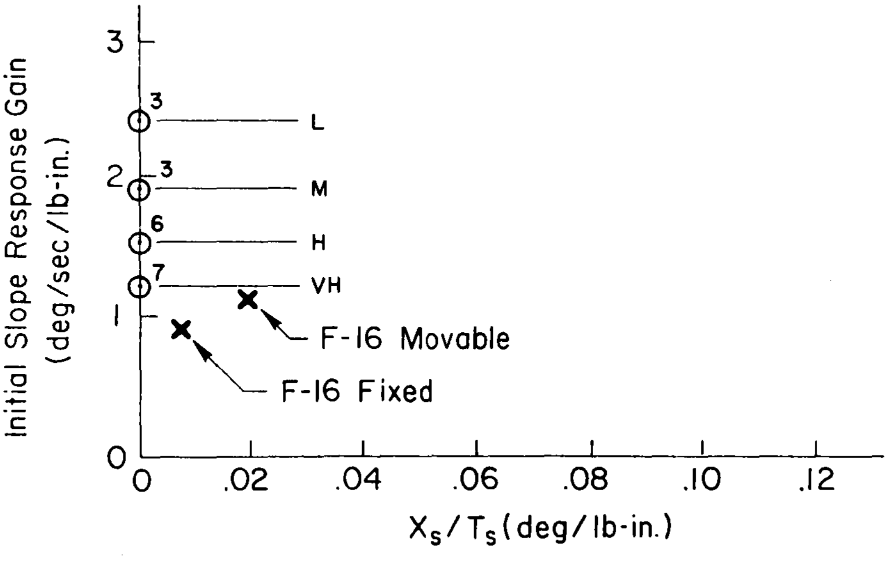


Figure IV-9. F-16 Sidestick Feel/Command Sensitivity Characteristics in Landing

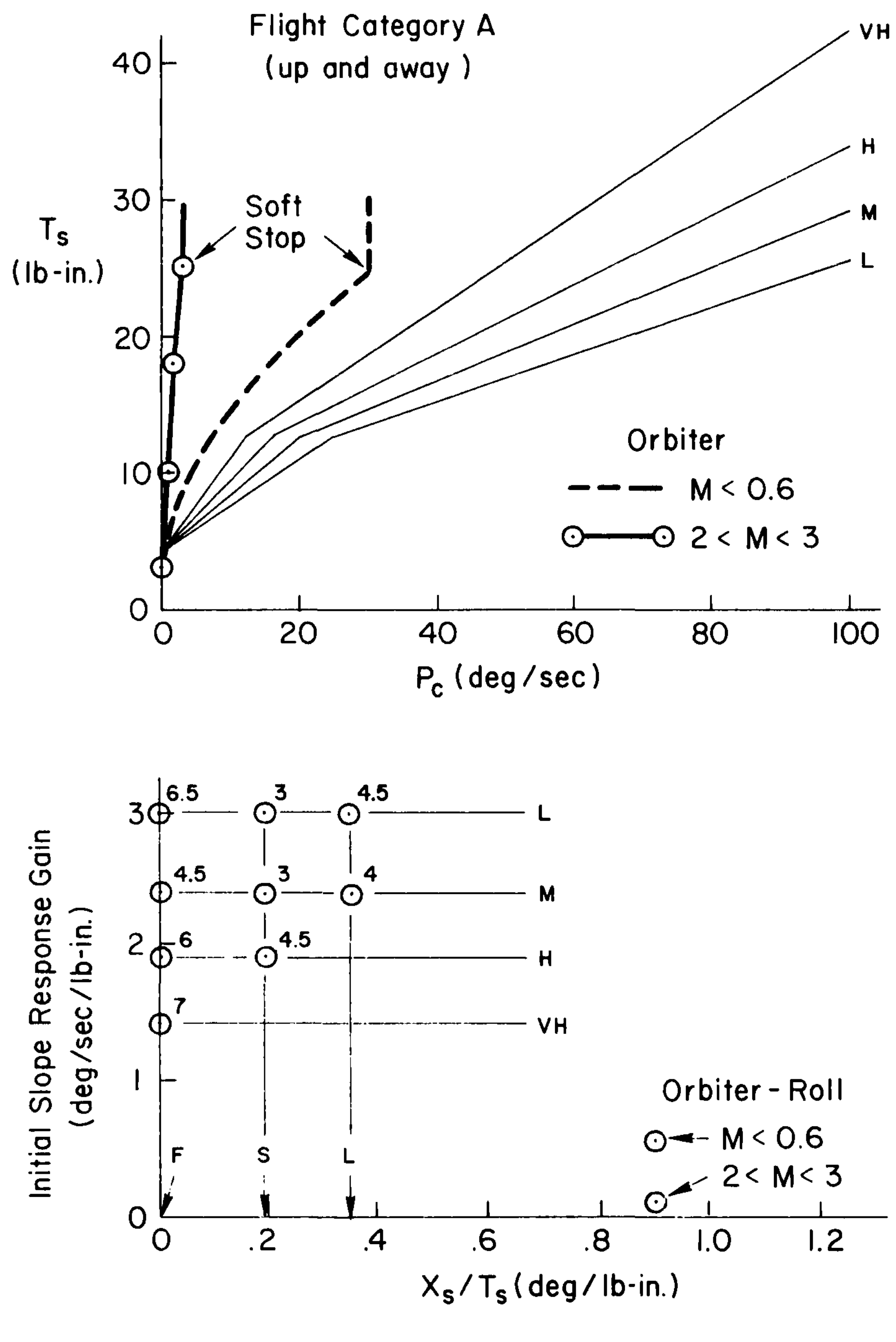
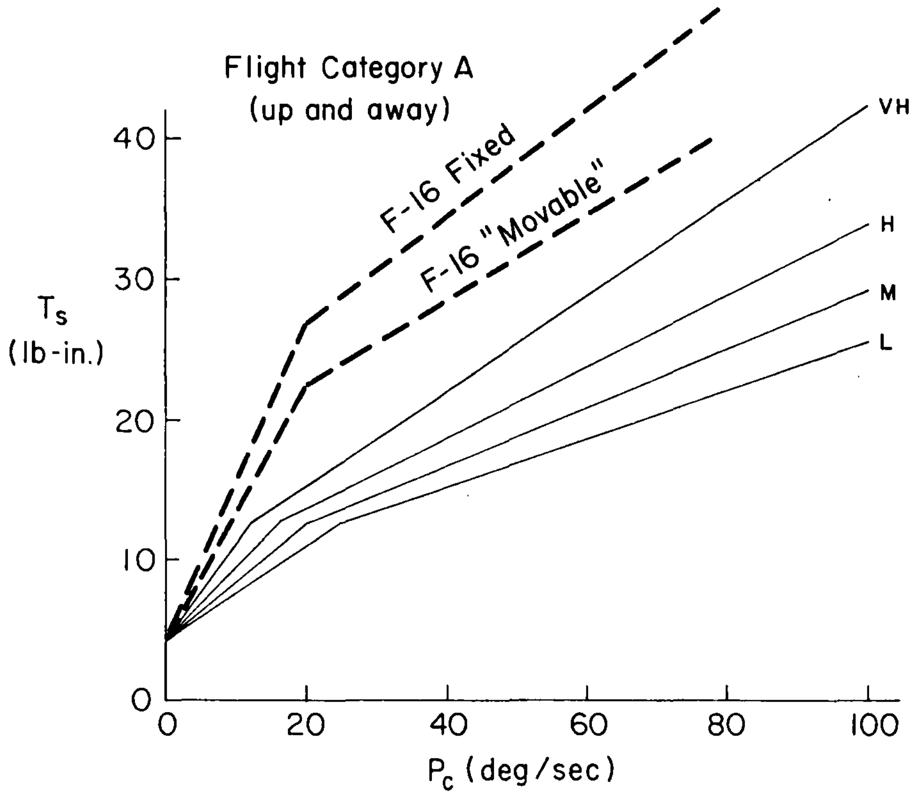


Figure IV-10. Shuttle Orbiter Lateral RHC Feel/Command Sensitivity Characteristics for Up-and-Away Flight Tasks



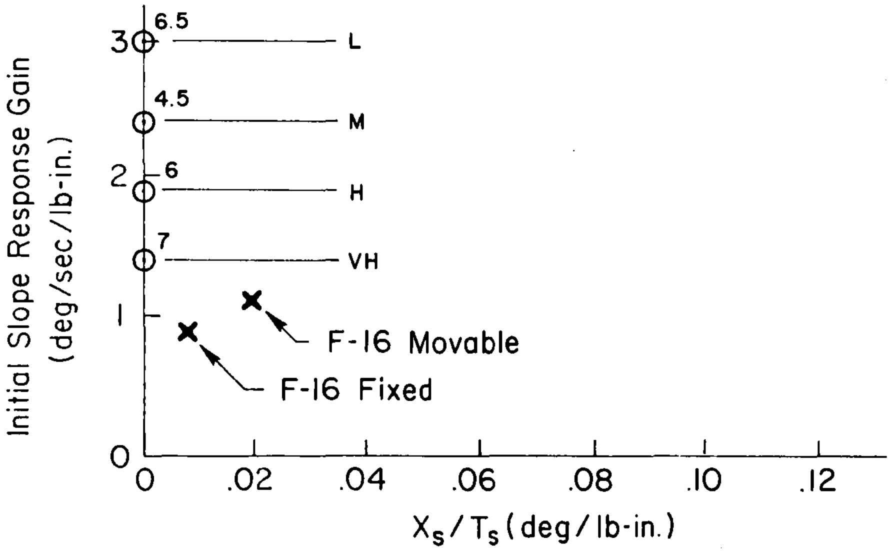


Figure IV-11. F-16 Sidestick Feel/Sensitivity Characteristics in Up-and-Away Flight

Overall, the Shuttle Orbiter RHC feel and roll rate command characteristics differ appreciably from other sidesticks which have been formally evaluated. There are no firm data to say the Orbiter feel/command characteristics are either desirable or undesirable. However, there is enough evidence of less-than-optimal parameters to indicate the combined mechanical and electrical characteristics might be a target for future flying quality investigation.

SECTION V

CONCLUSIONS AND RECOMMENDATIONS

Extensive review of the Shuttle Orbiter flying quality and control system requirements and comparison of these with other flying quality requirements and data have revealed several areas of disagreement and possible deficiencies in the Orbiter requirements. Limited closed-loop analysis has indicated several likely or existing stability or controllability problem areas. Five items or areas of concern are associated with longitudinal flying qualities, three additional aspects involve lateral-directional flying qualities, and one more is associated with the pilot's control feel characteristics.

The Shuttle pitch control system combined with the low (unstable) static margin of the airframe produces an unconventional pitch attitude numerator zero ($1/T_{\theta_A}$) which is significantly different from the path mode inverse time constant ($1/T_{\theta_2}$). This requires special consideration in the use of empirical data (which is largely based on conventional aircraft) but does not hamper the use of conventional manual control theory.

The pitch rate time response boundaries presently in use may unduly restrict overshoot characteristics and, conversely, may not sufficiently restrict effective time delay. There is a need for further validation of the pitch rate time response boundaries for large aircraft and for the supersonic flight regime. However, this conclusion must be considered in the light of the unconventional attitude dynamics of the Shuttle noted above.

The assessment of short-period dynamics indicated that the Shuttle is apparently adequate, i.e., Level 1, on the basis of the conventional MIL Spec short-period requirements for subsonic flight. However, for supersonic flight there is a need for further substantiation of the requirements since the data base is primarily subsonic.

Effective time delay appears to be the longitudinal flying qualities parameter of most concern. The Shuttle appears to be Level 3 based on existing time delay specifications and is comparable to aircraft with known landing problems. However, there is still uncertainty as to whether the time delay criteria are relevant to aircraft such as the Shuttle.

The Shuttle is unconventional in that the pilot is located slightly aft of the instantaneous center of rotation for elevator inputs due to unusual inertial properties. Manual control theory and previous simulation studies indicate that this can degrade flying qualities by limiting the bandwidth of the pilot's path angle control loop. Conventional parameters such as the control anticipation parameter (CAP) are not adequate for assessing this problem.

The roll rate response boundaries as currently set forth in the Shuttle Orbiter flying quality and control system criteria appear to be very broad and unrestrictive. There is evidence that large aircraft that meet the subsonic response boundaries have been rated unacceptable. There is need for further validation of the roll rate time response boundaries for large aircraft is both the subsonic and supersonic flight regimes.

There is indication that there may be a tradeoff between flying quality and ride quality requirements for large aircraft such as the Orbiter. A problem occurs for vehicles that roll about the velocity vector and are flown at large angles of attack and/or with the pilot located far above and in front of the vehicle c.g. Configurations that meet the current roll rate response boundaries may be found unacceptable because of large lateral accelerations at the pilot station. Lateral acceleration at the crew station is currently unspecified for the Shuttle Orbiter. Two ways of specifying the allowable lateral acceleration have been examined and neither appears to be completely satisfactory. Additional tests and data are required in order to develop an appropriate criterion.

The equivalent time delay between pilot command of a maneuver and the actual vehicle response also is currently unspecified in any of the Shuttle Orbiter documents although it may be inferred from the roll rate response boundaries. There are two criteria presently available in the literature, one of which allows some time delay threshold, the other does not. The current open-loop criteria are questionable. Equivalent time delay and roll rate rise time interact to limit achievable closed-loop bandwidth in both roll and lateral path control. Increasing equivalent time delay and/or decreasing roll rate response reduce the closed-loop bandwidth control that the pilot can obtain in either the roll or path control tasks. Further experiments should address this interrelationship.

The Orbiter rotational hand control (RHC) feel and roll rate command characteristics differ appreciably from other sidesticks that have been formally evaluated. The RHC displacement/force/electrical command combined characteristics possibly result in larger pilot control latencies (due to near isotonic properties). This can affect the control bandwidth and contribute to control difficulties in urgent tasks.

It is recommended that each of the above deficiencies be addressed in future (OEX) Flying Qualities and Flight Control System Design Criteria Experiments. These are areas where flight data will make the greatest impact in helping to provide needed criteria and design guides. However, an integrated program of simulation, pre-experiment analysis, and post-experiment interpretation and analysis should precede the flight test to thoroughly explore the estimated critical and surrounding conditions. Flight validation can then be accomplished on a one or two point basis.

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APPENDIX A

DERIVATION OF EXPRESSIONS FOR THE ALTITUDE NUMERATOR, $N_{\delta_e}^{\dot{h}_p}$ AT THE PILOT'S POSITION

The short period (constant speed) approximations for the angle-of-attack and pitch attitude elevator transfer functions are (see Ref. 1, pg. 307).

$$\frac{\alpha}{\delta_e}(s) = \frac{s\left[Z_{\delta_e}s + \left(U_oM_{\delta_e} - Z_{\delta_e}M_q\right)\right]}{U_o\Delta} \qquad \text{A-1}$$

$$\frac{\theta}{\delta_e}(s) = \frac{\left(M_{\delta_e} + Z_{\delta_e}M_{\dot{w}}\right)s + \left(Z_{\delta_e}M_w - M_{\delta_e}Z_w\right)}{\Delta} \qquad \text{A-2}$$

where

$$\Delta = s\left[s^2 - \left(U_oM_{\dot{w}} + Z_w + M_q\right)s + \left(M_qZ_w - U_oM_w\right)\right]$$

The altitude rate at the c.g. is

$$h = U_o r \qquad \text{A-3}$$

and at the pilot's position

$$h_{\dot{p}} = \dot{h} + \ell_p\dot{\theta} = U_o(\theta - \alpha) + \ell_p s\theta \qquad \text{A-4}$$

Thus

$$N_{\delta_e}^{\dot{h}_p} = -Z_{\delta_e}\left(s^2 - \left(U_oM_{\dot{w}} + M_q\right)s - M_\alpha + \frac{M_{\delta_e}}{Z_{\delta_e}}Z_\alpha\right) + \ell_p s\left(sM_{\delta_e} + \frac{Z_{\delta_e}M_\alpha}{U_o} - \frac{Z_\alpha M_{\delta_e}}{U_o}\right) \qquad \text{A-5}$$

Defining $\Delta\ell_p$ as $\Delta\ell_p \equiv \ell_p - X_{ICR}$

$$N_{\delta_e}^{\dot{h}_p} = Z_{\delta_e}\left(U_o M_{\dot{w}} + M_q - Z_w + \frac{Z_{\delta_e}}{M_{\delta_e}} M_w\right)s + M_{\delta_e} U_o\left(-Z_w + \frac{Z_{\delta_e}}{M_{\delta_e}} M_w\right)$$
$$+ \left(\frac{\Delta\ell_p}{X_{ICR}}\right) Z_{\delta_e} s\left(s - Z_w + \frac{Z_{\delta_e}}{M_{\delta_e}} M_w\right) \qquad \text{A-6}$$

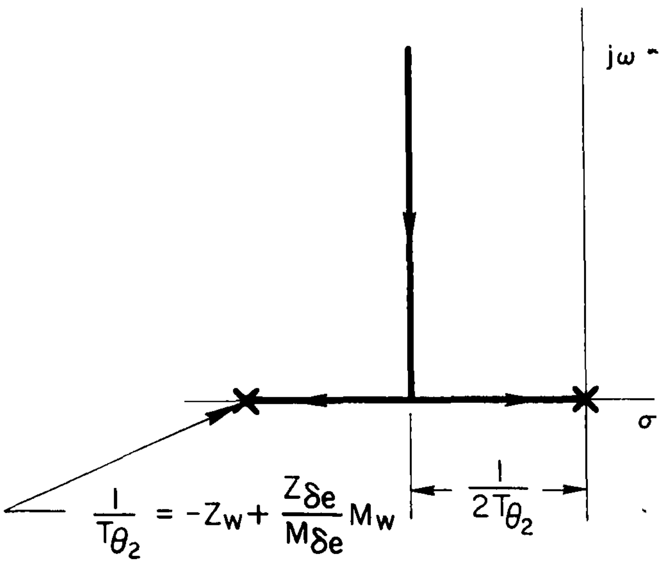
$$= A_{h_p}(1/T_{h_2})(1/T_{h_3}) \text{ or } A_{h_p}[\zeta_{hp}, \omega_{hp}] \qquad \text{A-7}$$

The variation of the roots with $(\Delta\ell_p/X_{ICR})$ is shown in Fig. II-31.

When the pilot is relatively for forward $\Delta\ell_p/X_{ICR}$ is large and

$$N_{\delta_e}^{\dot{h}_p} = \left(\frac{\Delta\ell_p}{X_{ICR}}\right) Z_{\delta_e} s\left(s - Z_w + \frac{Z_{\delta_e}}{M_{\delta_e}} M_w\right) \qquad \text{A-8}$$

which shows that the root locus simplifies to the sketch below



Thus in the limit with the pilot far forward

$$(1/T_{h_2})(1/T_{h_3}) \rightarrow (0)\ (1/T_{\theta_2})$$

With the pilot at the ICR, one root "goes to infinity" and

$$\frac{1}{T_{h_p}} = \frac{M_{\delta_e} U_o\left(-Z_w + M_w Z_{\delta_e}/M_{\delta_e}\right)}{Z_{\delta_e}\left(U_o M_{\dot{w}} + M_q - Z_w + Z_{\delta_e}/M_{\delta_e} M_w\right)} \qquad \text{(A-11}$$

As the pilot is moved aft of the ICR, there are always two real roots, one of which always has non-minimum phase (i.e., is in the right half plane). In this case

$$N_{\delta_e}^{\dot{h}p} = A_{\dot{h}p}(1/T_{h_2})(1/T_{h_3}) \qquad \text{A-12a}$$

where

$$A_{\dot{h}p} = \Delta\ell_p M_{\delta_e} = -Z_{\delta_e} + \ell_p M_{\delta_e} \qquad \text{A-12b}$$

$$\frac{1}{T_{h_2}} \cdot \frac{1}{T_{h_3}} = \frac{-M_{\delta_e} Z_\alpha - M_\alpha Z_{\delta_e}}{\Delta\ell_p M_{\delta_e}} \qquad \text{A-12c}$$

$$\frac{1}{T_{h_2}} + \frac{1}{T_{h_3}} \doteq \frac{1}{\Delta\ell_p}\left[\frac{-Z_{\delta_e}}{M_{\delta_e}}\left(U_o M_{\dot{w}} + M_q\right) + \ell_p\left(-\ Z_w + \frac{Z_{\delta_e}}{M_{\delta_e}} M_w\right)\right] \qquad \text{A-12d}$$

for the typical situation in which $|Z_{\delta_e} M_\alpha| << |M_{\delta_e} Z_\alpha|$

$$\frac{1}{T_{h_2}} \cdot \frac{1}{T_{h_3}} \doteq -Z_\alpha/\Delta\ell_p \qquad \text{A-12e}$$

and

$$\frac{1}{T_{h_2}} + \frac{1}{T_{h_3}} \doteq \frac{X_{ICR}}{\Delta \ell_p}\left(U_o M_{\dot{w}} + M_q\right) - \frac{\ell_p}{\Delta \ell_p} Z_w \qquad \text{A-12f}$$

The adequacy of these 2-DOF (θ and γ) approximations may be checked by comparison to the complete 3-DOF (θ, γ and u) transfer functions given in Ref. 2.

	2-DOF APPROXIMATION	COMPLETE 3-DOF
$A_{\dot{h}p}$	0.234	0.231
$\frac{1}{T_{h_2}} \cdot \frac{1}{T_{h_3}}$	-12.2	-12.4
$\frac{1}{T_{h_2}} + \frac{1}{T_{h_3}}$	-0.60	-0.77

APPENDIX B

OEX ON SHUTTLE FLYING QUALITIES

PURPOSE

To develop improved useful flying quality and flight control system criteria and design guides for Space Shuttle craft by combining analytical, simulation, and Shuttle orbiter flight test results.

BACKGROUND

1. Extensive review of Shuttle flying quality and control system requirements and comparison of these requirements with other flying quality requirements and data reveal several areas of disagreement and possible deficiencies in the Shuttle requirements. Most important of these are:

   a. Pitch rate requirements — Shuttle time response upper boundary specification may be misplaced, being too tight on pitch rate overshoot allowable (or even desirable). Alternatively, the existing flying qualities data, and perhaps parameters, appear inappropriate for heavily augmented, relaxed-static-stability, aircraft (e.g., F-16, F-18)!

   b. Allowable dead time on the Shuttle time response spec for pitch rate and roll rate is probably too large

2. Comparison of Shuttle closed-loop dynamic characteristics with existing flying quality criteria, data, and design guides (all developed since the Shuttle specifications were finalized years ago) indicates several <u>likely</u> or <u>existing</u> problem areas:

   a. Large longitudinal effective time delay

      1) Consequent lowered effective vehicle bandwidth and hence reduced pilot-vehicle and autopilot-vehicle attainable closed-loop bandwidth in path control functions

      2) Tendency for PIO under high stress, precise control conditions

   b. Large lateral effective time delay

      1) Lowered effective vehicle bandwidth and thus reduced pilot-vehicle and autopilot-vehicle attainable closed-loop bandwidth in rolling and path control functions

      2) Tendency for PIO under high stress, precise control situations

3) Increased time to bank (ϕ_{30} is 8785C Level 2, due entirely to the lateral effective time delay).

c. Controllability of lateral coupled roll subsidence-spiral oscillation (lateral phugoid)

1) In the 1.5 > M > 1.2 regime an effective lateral phugoid exists (1/4 Hz)

a) Divergent oscillation, yaw jets off
b) Stable, yaw jets firing

2) Damping (effective $[\zeta\omega]_{RS}$) is 8785C marginal with jets on, unsatisfactory with jets off

d. Pilot location effects -- while well ahead of the c.g., the pilot is aft of the center of instantaneous rotation for longitudinal control inputs (whereas on most large aircraft the pilot is ahead of the CIR). This location has consequences on:

1) Longitudinal path control -- possibly quite unfavorable for precise control situations

2) Lateral acceleration at the pilot station which is possibly deleterious

e. The RHC displacement/force/electrical command combined characteristics possibly result in larger pilot control latencies (due to near isotonic properties). This can affect the control bandwidth and contribute to control difficulties in urgent tasks.

3. Comparison of possible or conceivable Shuttle dynamic characteristics with analyses, limited data, and tentative design guides focuses attention on several <u>conceivable</u> problem areas:

a. Possibly marginal bank angle control in the 3.5 > M > 2.5 area <u>if</u> some aerodynamic characteristics approach the extremes of critical variation sets

b. Coordination in rolling maneuvers and sideslip trimming characteristics for "bent" airframe and laterally off-center c.g. effects -- especially above M = 3.5 (where rudder is inactive and yaw jets provide coordination and trim)

c. Reduced surface rates with 2 failed APUs

1) Possible deficient control with crosswind, runway landings

2) Increased PIO potential with such landings

INITIAL EMPHASIS

1. Settle, with flight and backup simulation and analysis data, those areas where known discrepancies exist, i.e.,

   2a, 2b -- large effective τ

   2e -- pilot contribution to effective τ

   2c -- controllability of lateral coupled roll subsidence-spiral

   2d -- pilot location effects

   (CATEGORY 1)

2. Explore further, with simulator and analysis, those areas where potential problems are possible but only in unlikely circumstances, i.e.,

   3b -- coordination with "bent" airframe and off-center c.g.

   3c -- reduced surface rates

   (CATEGORY 2)

3. Postpone consideration of conceivable, but not yet demonstrated, and highly unlikely phenomena, e.g., 3a

APPROACH

1. For flight experiments (CAT 1) an integrated program of simulation, pre-experiment analysis, and post-experiment interpretation and analysis to thoroughly explore the estimated critical and surrounding conditions. Flight validation on a 1 or 2 point basis <u>unless</u> the expected results do not occur in flight.

2. For (CAT 2) experiments, again an integrated program of simulation, pre- and post-experiment analysis, etc.

3. Tools and techniques:

   a. Identification of the effective vehicle

      1) Analysis -- use best current estimates of the linearized conditions

      2) Simulation:
         small perturbation checks with linear analysis data
         frequency sweeps and subsequent effective transfer characteristics reduced via FFT

      3) Flight -- use best estimates available

b. Identification of pilot dynamics/behavior

1) Flight and simulator -- use non-intrusive pilot identification routine

2) Simulator alone -- use, for limited check cases, sums of sinusoid disturbance and FFT

c. Task-tailored pilot questionnaires/ratings scales

1) For each of the CAT 1 critical areas a specialized questionnaire and adjectival-phrase-based rating scales will be evolved for use in both the simulations and flight

2) Debriefing questionnaire -- for the experimenter/debriefer. Will be structured to expand, further explore, and clarify interpretations

3) Those questionnaire/scales will be indicative of ease of control, workload, response qualities of primary and secondary motions, flight performance, flight safety, etc.

4. Flight tasks/maneuvers

a. The unpowered glider nature of the Shuttle precludes the use of a test matrix in which a number of evaluations can be made in succession at a particular flight condition. Instead evaluations must be made on the run at particular spots along the entry trajectory. As a practical matter even a "spot" cannot be relied on for a particular entry, so the flight condition can only be approximated within a region around a desired nominal. (Consequently simulation support activities may be conducted both pre- and post-flight to assure that the region is adequately covered, with at least one simulation point which has a corresponding flight point.)

b. Safety of flight and mission priority considerations probably imply that acceptable flight maneuvers be relatively modest. It is anticipated that rapid pushovers or pullups and lateral offset maneuvers with subsequent precision path control simulating final adjustments in approach and landing will suffice for most of the Category 1 tests associated with large τ and pilot location effects. These can be conducted at altitudes which permit safe recovery and can possibly use a flight director with special signals inserted. Ideally these would be conducted during both TAEM and APPROACH flight phases.

The lateral controllability tests for the coupled roll subsidence-spiral will need to be a modest roll maneuver and subsequent heading regulation task at $M \doteq 1.2$.

c. Special instructions and procedures for the flight maneuvers will be required to establish pilot "set" at a high stress level, and to create or adequately simulate a constrained very high precision control environment (e.g., akin to landing on a short narrow runway at a specific point). Pilot gain and skill utilization must be at maximum levels to get the ultimate in control precision from the pilot-vehicle closed-loop system.

5. Instrumentation

a. Ideally the instrumentation available should be sufficient to provide all control inputs (including jets) as total quantities, pilot commands, and a complete set of orbiter output responses. Time references for any pieces of unsynchronized data must be available.

b. The FCS modes, switch status, etc., must be available in a form which is time synchronized relative to the response and control input data.

c. Ideally the use of onboard recording on to a tape would permit ready modification of the data to formats suitable for NIPIP and similar analysis. Time trace and corresponding flight tape segments or punch cards would also be suitable.

OEX PRELIMINARY NOTES ADDENDUM

Basic issues to investigate are influence of τ_{eff}, heavy θ/δ_e augmentation for relaxed static stability, and near isotonic manipulators (larger pilot τ) on longitudinal, lateral, and combined urgent tasks.

Aside from full-scale flight test evaluations on the Shuttle itself there must be a series of analyses and experiments to block out the area of convern. OEX flight tests will then be used (in a limited way) to verify basic analysis and simulation data.

Possible simulation tools include:

DFRC Fixed-Base Simulation

ARC FSAA

ARC VMS

TIFS

STA (Grumman Gulfstream)

DFRC F-8

F-16, F-18

The fighter-type aircraft (above) are included because they share, or can be made to approximate (F-8), the delays and heavy θ/δ_e augmentation similar to those found on the Shuttle. Presumably, limited flying under forced "urgent" conditions could reveal universal heavy augmentation and time-delay-induced control problems.

Additionally, some of the older TIFS, VMS, etc., data should be reviewed for possible applicability. Such review should establish the fidelity with which the Orbiter was simulated and, if possible, a quantitative measure of the "urgency" demonstrated in the simulators. The latter point refers to the pilot gain or bandwidth adopted which could then serve as the basis for closed-loop predictive analyses.

Although the simulations would hopefully be performed for real-life "urgent" tasks, some artificially induced urgency will probably be necessary. This could take the form of a time constraint (as in the STI fixed-base simulations of Shuttle PIO) or of additional workload, e.g., coupled side task. The simulations should also cover flight situations that could be duplicated on the OEX, e.g., simulated urgent "landings" at 5000 ft AGL.

The "urgent" situations pertinent to Shuttle operations deserve some assessment. This could conceivably be addressed by eliciting astronaut consensus as to the worst (tightest) situations they would expect to encounter. Of course, we already have a very pertinent example in the FF5 PIO record.

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15. Supplementary Notes

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16. Abstract

This report documents an analytic effort to assess the suitability of existing and proposed flying quality and flight control system criteria for application to the space shuttle orbiter during atmospheric flight phases and to assist the program definition of an orbiter experiment for flying qualities and flight control system design criteria. The assessment includes orbiter longitudinal and lateral-directional flying characteristics, flight control system lag and time delay considerations, and flight control manipulator characteristics. The study shows that much of the data obtained from conventional aircraft may be inappropriate for application to the shuttle orbiter and there is a dearth of flying quality data for this class of vehicle.

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