NACA

RESEARCH MEMORANDUM

AN ANALYTICAL EVALUATION OF THE EFFECTS OF AN AERODYNAMIC MODIFICATION AND OF STABILITY AUGMENTERS ON THE PITCH-UP BEHAVIOR AND PROBABLE PILOT OPINION OF TWO CURRENT FIGHTER AIRPLANES

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Moffett Field, Calif.

NATIONAL ADVISORY COMMITTEE
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AN ANALYTICAL EVALUATION OF THE EFFECTS OF AN AERODYNAMIC MODIFICATION AND OF STABILITY AUGMENTERS ON THE PITCH-UP BEHAVIOR AND PROBABLE PILOT OPINION OF TWO CURRENT FIGHTER AIRPLANES

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SUMMARY

The effects of a wing modification and of stability augmentation on the computed longitudinal behavior in the pitch-up region and probable pilot opinion of the pitch-up characteristics of two current fighter airplanes are presented.

The computations indicated that the addition of a wing leading-edge extension to one of these airplanes would (1) reduce the peak and overshoot angles of attack for all flight conditions investigated, with the exception of those at take-off speeds, and (2) should improve probable pilot opinion of the pitch-up behavior from unacceptable to unsatisfactory - the category of the elevator-controlled F-84F and F-86A airplanes. Added pitch damping provided by a simple pitch damper with constant gain did not materially reduce the response overshoots and would not be expected to improve pilot opinion on this airplane. One beneficial effect attributable to the pitch damper, however, was a reduction in the peak positive maneuvering tail-load increments.

For the other fighter airplane, the computations indicated that added pitch damping provided by a simple pitch damper with nonlinear gain reduced the peak and overshoot values of angle of attack significantly for all flight conditions considered but the landing approach. This should improve the probable pilot opinion of the pitch-up behavior for this airplane from unacceptable or unsatisfactory to unsatisfactory but acceptable - the rating category for the YF-86D and F-86F airplanes. The results for this airplane equipped with a stick pusher, which is a device intended to prevent pitch-up altogether, indicated that the pitch-up region was generally avoided with this device operating, even for extremely rapid maneuvers. Comparison of two versions of this device

showed that care must be taken in the design of such systems to minimize reductions in the maneuvering capabilities of the airplane.

INTRODUCTION

Design considerations for supersonic fighter airplanes may, in some cases, lead to configurations which would be expected to have a severe pitch-up problem at high angles of attack. Three possible approaches to this problem are the use of aerodynamic modifications, stability augmenters, and devices for preventing entry into the pitch-up region. Generally, the approach selected would be based on careful weighing of several important factors, such as possible performance losses due to extensive aerodynamic modification, the complexity of the stability augmenters or pitch-up preventers required, and the magnitude of the basic pitch-up problem.

In reference 1 an analytical study of the comparative pitch-up behavior of several airplanes is presented and the computed results are correlated with documented pilot opinion. By comparison of the computed pitch-up characteristics of new airplane designs with the corresponding results from reference 1, the probable relative severity of pitch-up and the associated probable pilot opinion of pitch-up may be determined prior to actual flight experience. Applied in this manner, the method is also useful for investigating the effectiveness of aerodynamic modifications or of automatic control devices in altering the severity of pitch-up on existing airplanes. This method for estimating probable relative severity of pitch-up is based on only the longitudinal dynamic behavior in the pitch-up region. The effects of other modes of motion such as roll-off, directional divergence, and spin entry are not considered. Although beyond the scope of the present study, some consideration should be given to the possible adverse effects on pilot opinion of these other modes of motion; this could be based on an inspection of wind-tunnel rolling- and yawing-moment data in the pitch-up region.

In the present report the three approaches to the pitch-up problem noted above, that is, the use of aerodynamic modifications, stability augmenters, and devices for preventing entry into the pitch-up region, are assessed in the light of their specific application to two example supersonic airplanes - the F-101A and F-104A airplanes. The methods of references 1 and 2 are used to evaluate the effects on the pitch-up behavior and on probable pilot opinion of pitch-up of a wing leading-edge modification and of added pitch damping on the F-101A airplane and of added pitch damping on the F-104A airplane. Also presented are the results of an analog study of the effectiveness of a limiting device, referred to as a stick pusher, for preventing pitch-up altogether on the F-104A airplane.

DESCRIPTION OF AIRPLANES

Basic Airplanes

The two example supersonic fighter airplanes considered in the present study are the McDonnell F-101A and the Lockheed F-104A airplanes. Two-view drawings of these airplanes, hereinafter referred to as airplanes A and B, respectively, and their pertinent physical characteristics are presented in figure 1 and table I, respectively.

Wing Modification

In addition to the basic airplane A, a configuration with the wing leading-edge extension modification shown in figure 2 was also studied.

Stability Augmenters

Pitch dampers.- Two types of pitch dampers were considered in the present study. For airplane A, a pitch damper was considered which increased the total pitch damping to five times that of the basic airplane. In this case, it was assumed the damper was operative at all times with a gain constant of $5l_t/V$. (See Appendix A for definition of symbols.) A block diagram of this linear pitch damper is presented in figure 3. For airplane B, a pitch damper with a gain of $5l_t/V$ was again assumed. For this case, however, it was assumed the damper becomes operative only in the pitch-up region. The set-up assumed for the analog computations for this nonlinear damper is shown in block-diagram form in figure 4.

Stick pusher.- The stick pusher is a device designed by Lockheed Aircraft Corporation to limit the attainable airplane angle of attack to values below those at which pitch-up occurs. The assumed operational envelope for the stick pusher, provided by Lockheed, is shown in figure 5. When the combined signals from sensors sensitive to pitching velocity and angle of attack reach a predetermined value established by this envelope, an action signal is transmitted to the solenoid valve actuator assembly which in turn moves the stabilizer and cockpit stick in unison to the trim position (i.e., the position for zero stick force in steady level flight). If the pilot does not attempt to override the device, the stabilizer returns to trim at the rate of about 20° per second. A block diagram, assumed to represent the stick pusher for the analog computations, is shown in figure 6.

METHOD AND PROCEDURE

The dynamic behavior of airplanes A and B in the pitch-up region was computed by means of the evaluation maneuver and the basic equations of motion presented in reference 1. A representative time history of the evaluation maneuver from reference 1 is reproduced in figure 7. For the present study, since wind-tunnel data indicated a large decrease in control effectiveness in the pitch-up region for these two airplanes, these equations of motion were modified to include this effect. Also, the equations were rewritten in terms of absolute rather than incremental (from $n = 1g$) values of $C_L(\alpha)$ and $C_m(\alpha)$ since it was desired to record absolute values of α and n. The modified equations are:

$$-mV(\dot{\theta} - \dot{\alpha}) = \left[-C_L(\alpha) + C_{m_{\delta_s}}(\alpha)\,\frac{\bar{c}}{l_t}\,\delta_s\right] qS + W \qquad (1)$$

and

$$I_y\ddot{\theta} = \left[C_m(\alpha) + \left(C_{m_{\dot{\theta}}} + C_{m_{\dot{\alpha}}}\right)\dot{\alpha} + K\,\frac{l_t}{V}\,C_{m_{\delta_s}}(\alpha)\dot{\gamma} + C_{m_{\delta_s}}(\alpha)\delta_s\right] qS\bar{c} \qquad (2)$$

Several flight conditions were selected for analysis for each airplane. These are presented in the table below.

Airplane	Mach number	Altitude, ft	Configuration	Center of gravity location, $\bar{c}$
A	0.25	Sea level	Take-off	0.286
	.85	35,000	Clean	.286
	1.20	35,000	Clean	.286
B	.23	Sea level	Landing approach	.20
	.80	35,000	Clean	.07
	.975	35,000	Clean	.07

The basic aerodynamic data for these flight conditions were obtained from wind-tunnel measurements provided by Lockheed and McDonnell and are presented in figure 8 and in table I.

RESULTS AND DISCUSSION

This section is divided into two main subsections. In the first, the computed pitch-up behavior of the two example airplanes for all the flight conditions considered is discussed in some detail. Particular emphasis is placed on the effectiveness of the wing modification and of the linear pitch damper on airplane A and of the nonlinear pitch damper on airplane B in minimizing the peak angles of attack and the wing and tail loads in pitch-up maneuvers. Also, the effectiveness of a preliminary version of the Lockheed stick pusher for preventing entry into the pitch-up region is assessed. In the second section, the effects of the wing modification and of the pitch dampers on probable pilot opinion of the pitch-up behavior of airplanes A and B are presented and discussed.

With the exception of the take-off and landing-approach conditions, all results are presented for initial stick-deflection ramps corresponding to an average load-factor entry rate into the pitch-up region of approximately 0.5g per second. For the low-speed flight conditions, the initial stick-deflection ramps were programmed to provide a gradual stall entry comparable to that used by Ames pilots in evaluating 1g stall characteristics.

Computed Pitch-Up Behavior

The results of the computations are presented in figures 9 through 17. Computed pitch-up time histories for the two example airplanes for the various flight conditions considered are shown in figures 9 and 12. Figures 10, 11, 13, and 14 show the variations with recovery-control rate of the overshoots in airplane angle of attack and load factor and of the maneuvering tail-load increment for airplanes A and B. In figures 15 through 17, the effect of stick-pusher operation on the maneuvering boundaries of airplane B is shown. Figures 15 and 16 also show the boundaries for the onset of buffeting and lateral unsteadiness provided by Lockheed from flight-test results.

<u>Effect of wing modification</u>.- The effects on pitch-up of the wing modification, as applied to airplane A, may be determined by comparing the basic and modified airplane A results in figures 9, 10, and 11. The

results indicate some improvement in the behavior of the airplane at Mach numbers of 0.85 and 1.20. The peak angles of attack and overshoots are lowered about 20 to 30 percent at the lower recovery-control rates. This order of improvement may be sufficient to reduce the possibility of inadvertent spin entries at moderate subsonic Mach numbers. At a Mach number of 0.25 at sea level (fig. 9(c)), no improvement is observed, since no recovery occurs either for the basic or the modified airplane, even at the maximum recovery-control rate of 30° per second, within the available data limits. An appreciable increase in stability just prior to the unstable break (fig. 8(c)) may, however, provide sufficient warning to prevent inadvertent pitch-ups at low speeds.

The results for a Mach number of 1.20 at 35,000 feet shown in figure 9(b) also indicate that critical wing loads may be experienced during supersonic pitch-up maneuvers for both the basic and modified airplanes. Peak airplane load factors of 8 to 9g, corresponding to overshoots of 3 and 4g, are shown for this flight condition (see figs. 9(b) and 11(a)).

The effects of the wing modification on the maneuvering tail-load increments are relatively small. Peak values of about 10,000 pounds at a Mach number of 0.85 and 22,000 pounds at a Mach number of 1.20 are shown in figures 10(b) and 11(b). It should be recognized that these values refer only to the out-of-balance portion of the total maneuvering tail load. They do not include the tail loads required for balance. Additional information is presented in Appendix B relative to maximum positive and negative total maneuvering tail loads in pitch-up maneuvers for two of the flight conditions considered for airplane A in the present study.

Effects of linear pitch damper.- The effects of increased pitch damping on the pitch-up behavior of airplane A were first investigated by arbitrarily increasing the pitch damping $M_{\dot{\theta}}$ to five times the normal damping at low angles of attack. The actual decrease in control effectiveness was not taken into account in this initial study. The results obtained (figs. 10 and 11) indicated a sufficient order of improvement in the pitch-up characteristics to warrant further investigation of a pitch damper which realistically reflects the large decrease in control effectiveness of the airplane at high angles of attack (fig. 8). The type of pitch damper assumed for the analog computations is shown in block-diagram form in figure 3. The computed results with this damper operating (figs. 9, 10, and 11) show that relatively small reductions in the peak and overshoot angles of attack and load factors were realized with this type of damper.

Beneficial effects attributable to normal operation of the linear pitch damper are a reduction in the severity of the recovery transients (lower negative peak $\ddot{\theta}$) and an associated reduction in peak maneuvering tail-load increments to about 50 percent of the values for the basic airplane. (See figs. 10(b) and 11(b).)

Effect of nonlinear pitch damper.- Initially, a pitch damper with constant gain similar to the type assumed in the analysis on airplane A was investigated on airplane B. However, since no appreciable reduction in the overshoots was noted with this type of damper, it was decided to try the nonlinear type shown in block-diagram form in figure 4. It may be seen from the inset figures that the gain was nonlinear, varying from 0 for $\alpha < \alpha^*$ to $5l_t/V$ for $\alpha > \alpha^*$. The pitch damper thus becomes operative when two conditions are satisfied, that is, α exceeds some critical value [1]α^* (in the present case α at $\ddot{\theta}_{th}$) and $\ddot{\theta}$ exceeds $\ddot{\theta}_{th}$. The damper remains in operation until either $\ddot{\theta}$ attains a predetermined large negative value of the order of 5 radians per second squared or α decreases below the critical value α^*. The effects of this damper on the pitch-up behavior of airplane B are shown in figures 12, 13, and 14. These results show that the nonlinear pitch damper effects an appreciable improvement in the pitch-up behavior for all flight conditions with the exception of the landing approach. For example, at a Mach number of 0.80 at 35,000 feet, the peak angles of attack were reduced from values somewhat in excess of 28° to about 24° at low recovery-control rates (fig. 12(a)). The overshoots and maneuvering tail-load increments (fig. 13) are roughly only 50 percent of comparable values for the basic airplane. At a Mach number of 0.975, the peak angle of attack and airplane load factor were reduced from about 32° to 27° and from 7.2g to 6.6g, respectively, at a recovery-control rate of 20° per second (fig. 12(b)). The results in figures 13 and 14 indicate that the effectiveness of the pitch damper apparently increases as the recovery-control rate is reduced below 20° per second. Also shown in figures 13 and 14 is the beneficial effect of the pitch damper in reducing the maneuvering tail-load increment from about 12,000 to 6,000 pounds.

It should be pointed out that the pitch damper system illustrated in figure 4 was designed to investigate only the principle of switching on the damper as a function of both α and $\ddot{\theta}$. Questions of system reliability, servo authority, and methods of mechanization were not examined. Also, the indicated improvement in the pitch-up behavior of airplane B due to the pitch damper would be realized only if the reductions in peak angles of attack were sufficient to prevent or minimize the possibility of spin entry or other uncontrollable motions in roll and yaw. This would have to be established by referring to wind-tunnel data on the rolling and yawing derivatives at high angles of attack for this airplane.

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[1]In the present study, it was convenient to select α^* as the value of α at which $\ddot{\theta}_{th}$ was attained for an $\dot{n}$ of 0.5g per second. However, other values for α^* could be selected (e.g., values corresponding to a C_{m_α} of zero at various Mach numbers). It is only necessary to select α^* at a level sufficiently high that the damper does not become operative during abrupt maneuvers below the pitch-up region.

Effect of stick pusher.- As noted previously, the stick pusher is a device designed to limit the attainable airplane angle of attack below those at which pitch-up occurs. This device in block-diagram form is shown in figure 6. In an attempt to simulate partial overriding of the device by the pilot, a recovery-control rate of 15° per second, in addition to a maximum recovery-control rate of 25° per second, was assumed. Also, various values of control-system time constant τ from 0 to 0.5 seconds were considered. In addition to the normal maneuver rate of 0.5g per second,[2] rates of 1 and 3g per second were investigated to check the operation of the device in rapid maneuvers. With the exception of the case at 0.80 Mach number and 35,000 feet, for which a peak angle of attack of 17° was attained (for assumed values of $\dot{n}$, τ, and $\delta_{s_{rec}}$ of 3g per second, 0.2, and 15° per second, respectively), no significant effect was noted with variations of these quantities within the limits investigated. The following discussion is, therefore, primarily concerned with the results obtained for values of $\dot{n}$, τ, and $\delta_{s_{rec}}$ of 0.5g per second, 0, and 25° per second, respectively.

Figure 12 presents comparative time histories for airplane B illustrating the operation and effect of the stick pusher. It is apparent that the device performs its function well; that is, the maximum values of α attained are well below those where pitch-up is experienced. However, some loss in the maneuvering capabilities of airplane B is indicated as shown by the results in figures 15 and 16. In these figures, the maximum values of α, C_L, and n, attainable for the airplane with the stick-pusher system operating, are compared with values corresponding to a C_{m_α} of zero and the maximum values attained with the basic airplane in pitch-up maneuvers for a recovery-control rate of 20° per second. The permissible steady-state maneuvering acceleration boundary as a function of Mach number, based on the zero C_{m_α} boundary furnished by Lockheed, is shown in figure 17. Also shown in figure 17, are two boundaries which represent limitations imposed in gradual maneuvers ($\dot{n} \approx 0$) by two versions of the stick pusher. The lower curve represents results of the present study for an early version of the stick pusher for which an operational envelope furnished by the manufacturer (fig. 5) was assumed. Subsequent to the completion of the present investigation, an improved version of the stick pusher was developed by Lockheed which reduced the maneuverability loss to that indicated by the upper stick-pusher boundary in figure 17. This improvement was effected primarily by the incorporation of a so-called washout circuit which reduces the magnitude of the $\dot{\theta}$ contribution to the stick-pusher activation signal in gradual maneuvers (see ref. 3).

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[2]For the landing-approach conditions (M = 0.23 at sea level), the stabilizer entry rate was programmed to provide a gradual stall entry rate similar to that used by Ames research pilots in evaluating low-speed or 1g stall characteristics.

Probable Pilot Opinion

In this section a qualitative assessment is presented of the effect of a wing modification and of pitch dampers on probable over-all pilot opinion of the pitch-up behavior of two example supersonic airplanes. The prediction is based on a comparison of the computed angle-of-attack and load-factor overshoots for these airplanes with the corresponding values from reference 1 for six reference airplanes for which pilot opinion was obtained. The flight conditions selected for the evaluation are 0.85 Mach number at 35,000 feet for airplane A and 0.80 Mach number at 35,000 feet for airplane B, since these flight conditions corresponded closely to those for which pilot opinion was provided for the six reference airplanes (i.e., M = 0.90 at 35,000 ft).

It should be pointed out that the procedure outlined in reference 1 for estimating probable pilot opinion of pitch-up is based only on the longitudinal dynamic behavior in the pitch-up region. The effects of other modes of motion, such as roll-off, directional divergence, or spin entry, are not considered. If these other modes of motion are suspected to be important in a given case, comparisons, such as those presented in the following sections, should serve as a preliminary guide to the probable relative severity of pitch-up. Although beyond the scope of the present study, a supplementary analysis of the importance of these other modes of motion should be made based on wind-tunnel rolling and yawing-moment data at high angles of attack. Reference 4 presents such data for a model similar in configuration to that of airplane B.

Effect of wing modification.- Figure 18 presents a comparison of the overshoots at 0.85 Mach number and at 35,000 feet for several configurations of airplane A with values taken from reference 1 for the six reference airplanes. These results indicate that the attitude overshoots for the basic airplane (fig. 18(a)) are as much as 50 percent greater than those for any of the reference airplanes and would probably result in an over-all pilot opinion of pitch-up (based on question V of table II) of unacceptable, particularly if the airplane were precipitated into an inadvertent high-speed stall or spin. A description of the pilot rating schedule used is presented in table III. The effect of the wing leading-edge extension (fig. 2) was to reduce the angle-of-attack overshoots to values comparable to those for the F-84F airplane at low recovery-control rates and to values between those for the F-84F and F-100A airplanes at the higher recovery-control rates. Since there is some evidence presented in reference 1 that the pilot forms his opinions on the basis of an airplane's behavior at low recovery-control rates, the pilot opinion for this configuration (airplane A) should be comparable to that for the elevator-controlled F-84F airplane, particularly if an inadvertent spin entry or other unsymmetrical maneuver is avoided. Although this indicated improvement in pilot opinion from unacceptable to unsatisfactory may not be considered significant on the example airplane which has a severe pitch-up problem, these results should not rule out the use of relatively simple aerodynamic modifications (such as that considered here) in other cases with basically less serious pitch-up problems. For example, flight-test results presented in reference 5 indicated that the addition of a wing leading-edge extension eliminated pitch-up entirely over most of the speed range of an airplane with a moderately severe pitch-up problem.

The results in figure 18(b) also indicate that the load-factor overshoots for both the basic and modified airplane A are moderate and comparable to values computed for the F-100A airplane. This is due principally to the higher wing loading and lower lift-curve slope in the pitch-up region for airplane A.

Effect of linear pitch damper.- Figure 18 presents a comparison of the angle-of-attack and load-factor overshoots, with and without the linear pitch damper operating on airplane A, with corresponding values for the six airplanes investigated in reference 1. The effect of the linear pitch damper on the pitch-up behavior of airplane A was to reduce the α overshoots about 10 to 20 percent (fig. 18(a)). It is apparent that these relatively small reductions in overshoot would not be expected to materially improve probable pilot opinion over that for the basic airplane. Despite the lack of effectiveness of the type of pitch damper assumed in the present calculations, it is felt further consideration of some form of pitch damper is warranted, particularly in view of the more encouraging results obtained with the nonlinear type of pitch damper assumed for airplane B.

It should be pointed out that while the linear pitch damper was relatively ineffective on the example airplane which has a severe pitch-up problem, a similar approach on airplanes with milder pitch-up tendencies may prove somewhat more effective and should be considered as one possible approach to the problem.

Effect of nonlinear pitch damper.- Figure 19 presents a comparison of the angle-of-attack and load-factor overshoots, with and without the nonlinear pitch damper for airplane B, with corresponding values for the six airplanes considered in reference 1. In view of the large attitude overshoots and the associated extreme angles of attack ($\alpha_{max} > 28.6^{\circ}$) indicated for the basic airplane B at the lower recovery-control rates, probable pilot opinion should range from unsatisfactory to unacceptable (see table III).

Comparison of the overshoots for the nonlinear pitch-damper configuration with those for the basic airplane B in figure 19(a) indicates that the damper reduces the α overshoots to values comparable to those experienced by the YF-86D airplane. The probable pilot opinion corresponding to these overshoots should be unsatisfactory but acceptable. It should be mentioned in connection with the interpretation of these results that the peak angles of attack even with the pitch damper operating are still of the order of 24° to 26° at the lower recovery-control rates (fig. 12) because of the relatively high angles of attack associated with onset of pitch-up. Poor controllability in roll and yaw at these high angles of attack may affect pilot opinion adversely and cannot be accounted for by the type of comparison shown in figure 19.

For airplanes with somewhat milder pitch-up tendencies than those considered in the present study, this approach should prove effective in minimizing the pitch-up problem to an acceptable degree.

CONCLUSIONS

An analytical evaluation has been made of the effects of a wing modification and of stability augmentation on the pitch-up behavior and probable pilot opinion of the pitch-up characteristics of two fighter airplanes with severe pitch-up tendencies. Pitch-up behavior was computed for several flight conditions, while pilot opinion was estimated for these two airplanes for a Mach number of about 0.9 at 35,000 feet. The results of this analytical study indicated the following:

1. The addition of the wing leading-edge extension to airplane A reduced the peak and overshoot angles of attack for the lower recovery-control rates about 20 to 30 percent at Mach numbers of 0.85 and 1.20. This order of improvement may be sufficient to reduce the possibility of inadvertent spin entries at the lower Mach number. At the higher Mach number, an increase in the maneuvering capability before onset of pitch-up of about 1g with no penalty in increased load factor was obtained due to a decrease in overshoot load factor.

2. Probable pilot opinion of the pitch-up behavior of the modified airplane A at a Mach number of 0.85 at 35,000 feet should be improved from unacceptable to unsatisfactory - the category of the elevator-controlled F-84F and F-86A airplanes.

3. The effect of added pitch damping, provided by a simple pitch damper with constant gain, did not materially reduce the overshoots on airplane A and would not be expected to improve pilot opinion appreciably.

4. For airplane B, added pitch damping, provided by a pitch damper with nonlinear gain, reduced the peak and overshoot values of angle of attack significantly for all flight conditions considered except the landing approach.

5. Probable pilot opinion of the pitch-up behavior of airplane B with added pitch damping at a Mach number of 0.80 at 35,000 feet should be improved from unacceptable or unsatisfactory to unsatisfactory but acceptable - the rating for the YF-86D and F-86F airplanes.

6. The effect of the stick pusher was to prevent the attainment of angles of attack at which pitch-up occurs for airplane B. Results for two versions of this system indicated that care must be taken to insure that the maneuvering capabilities of the airplane are not compromised.

Ames Aeronautical Laboratory
National Advisory Committee for Aeronautics
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APPENDIX A

SYMBOLS

C_L	airplane lift coefficient, $\frac{L}{qS}$
C_m	airplane pitching-moment coefficient about airplane center of gravity, $\frac{M_{cg}}{qS\bar{c}}$
$\bar{c}$	wing mean aerodynamic chord, ft
g	acceleration due to gravity, 32.2 ft/sec^2
I_y	airplane pitching moment of inertia, slug-ft^2
h_p	pressure altitude, ft
K	parameter denoting the ratio of airplane damping to horizontal-tail damping
K_1, K_2	constants defining operational boundary for Lockheed stick pusher
l_t	distance from airplane center of gravity to aerodynamic center of horizontal tail, ft
L	airplane lift, lb
L_t	horizontal-tail lift, lb
$\Delta L_{t_{\ddot{\theta}}}$	maneuvering tail-load increment, $\frac{I_y\ddot{\theta}}{l_t}$
m	airplane mass, $\frac{W}{g}$, slugs
M_{cg}	pitching moment about airplane center of gravity, ft-lb
N	airplane normal force, assumed equivalent to $\left[C_L(\alpha) - C_{m_{\delta_s}}(\alpha)\,\frac{\bar{c}}{l_t}\,\delta_s\right] qS$, lb
n	airplane normal load factor, $\frac{N}{W}$

n_{over}	increment in n from value at pitching-acceleration threshold to n_{max}
n'	airplane normal load factor due to α, assumed equivalent to $\frac{C_L(\alpha)qS}{W}$
n'_{over}	increment in n' from value at pitching-acceleration threshold to n'_{max}
$\dot{n}$	load-factor rate at entry into pitch-up region, g/sec
q	dynamic pressure, $\frac{\rho V^2}{2}$, lb/sq ft
s	differential operator
S	wing area, sq ft
S_t	horizontal-tail area, sq ft
t	time, sec
V	airplane velocity, ft/sec
W	airplane weight, lb
α	airplane angle of attack, deg or radians
α_{over}	increment in α from α^* to α_{max}
α^*	value of α at pitching-acceleration threshold, deg
γ	flight-path angle, radians
δ_s	stabilizer angle, deg or radians
δ_{s_P}	stabilizer angle due to pilot stick deflection, deg or radians
δ_{s_D}	stabilizer angle due to pitch damper operation, deg or radians
θ	angle of pitch, radians
ρ	mass density of air, slugs/ft^3

ζ	airplane longitudinal short-period damping ratio
τ	control system time constant, sec
ω_n	airplane longitudinal short-period frequency
$C_L(\alpha)$	curve defining variation of airplane lift coefficient with α
$C_m(\alpha)$	curve defining variation of airplane pitching-moment coefficient with α
$C_{m_{\delta_s}}(\alpha)$	curve defining variation of stabilizer effectiveness with α
$C_{m_{\dot{\theta}_t}}(\alpha)$	$C_{m_{\delta_s}}(\alpha)\,\frac{l_t}{V}$
$C_{m_{\dot{\theta}}}(\alpha)$	$KC_{m_{\dot{\theta}_t}}(\alpha)$
$C_{m_{\dot{\theta}_D}}$	pitch damping derivative due to pitch damper, $\frac{5l_t}{V}\,C_{m_{\delta_s}}(\alpha)$ or $\frac{5l_t}{V}\left(C_{m_{\delta_s}}\right)_{\alpha=0}$, as noted
$(\dot{\ })$	$\frac{d(\ )}{dt}$
$\ddot{\theta}$	equivalent notation for $\frac{d^2\theta}{dt^2}$

Subscripts

th	threshold value at which pilot is first cognizant of pitch-up
rec	recovery
over	overshoot
max	maximum value

APPENDIX B

HORIZONTAL-TAIL LOADS IN PITCH-UP MANEUVERS

In a previous section of this report the maneuvering tail-load increment $\Delta L_{t_{\ddot{\theta}_{max}}}$ was presented as a function of recovery-control rate for airplanes A and B. It was noted that this represents only a portion of the total aerodynamic load on the tail; that is, the balancing load was not included. In this section a more complete analysis which includes the balancing tail load is presented for airplane A for Mach numbers of 0.85 and 1.20 at 35,000 feet.

Figure 20 shows the variation of the total aerodynamic load and of the maneuvering and balancing components with recovery-control rate. Also shown in figure 20 are the values of angle of attack at which these loads occur. It will be noted that these loads are referred to as "first-peak" and "second-peak" loads. The first-peak total load is a negative (down) load for stable tail-off configurations and occurs shortly after the onset of pitch-up at the time the peak positive pitching acceleration is attained. The second-peak total load may be either negative or positive and occurs at the time the peak negative pitching acceleration is reached during the recovery phase of the pitch-up maneuver. The maneuvering component of this load was previously presented in figures 10 and 11. The first-peak total loads in figures 20(a) and 20(c) were determined by adding the maneuvering component, $-I_y(\ddot{\theta}_{max})/l_t$, to the balancing component. Values of $\ddot{\theta}_{max}$ are presented in figure 21 as a function of recovery-control rate, and the balancing tail load was determined from the tail-off pitching-moment curves in figure 22 at the values of angle of attack at which $\ddot{\theta}_{max}$ occurred (fig. 20). In a similar manner, the second-peak total loads in figures 20(b) and 20(d) were determined.

These results indicate that the first-peak loads are critical in supersonic pitch-up maneuvers where the pilot does not attempt to check the pitch-up; that is, zero recovery control rate in figure 20(c). At subsonic speeds, the second-peak loads are generally critical in a positive sense where the pilot attempts to check the pitch-up by applying rapid recovery control (fig. 20(b)).

[redacted]

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3. Holleman, Euclid C., and Boslaugh, David L.: A Simulator Investigation of Factors Affecting the Design and Utilization of a Stick Pusher for the Prevention of Airplane Pitch-Up. NACA RM H57J30, 1957.

4. Tinling, Bruce E.: Subsonic Aerodynamic Characteristics Up To Extreme Angles of Attack of an Airplane Model Having a Low-Aspect-Ratio Wing and a High Horizontal Tail. NACA RM A57K05, 1958.

5. Matteson, Frederick H., and Van Dyke, Rudolph D., Jr.: Flight Investigation of the Effects of a Partial-Span Leading-Edge Chord Extension on the Aerodynamic Characteristics of a 35° Swept-Wing Fighter Airplane. NACA RM A54B26, 1954.

TABLE I.- AIRPLANE CHARACTERISTICS USED IN ANALYSIS

(a) Airplane A

Airplane weight, lb	36,800
Airplane mass, slugs	1,142
Airplane pitching moment of inertia, slug-ft^2	142,500
Wing area, sq ft	368
Wing mean aerodynamic chord, ft	10.24
Center-of-gravity position, percent $\bar{c}$	28.60
Horizontal-tail length, ft	28.80
Horizontal tail	
Deflection limits, deg	-20,+10
Maximum deflection rate, deg/sec	30
Mach number, 0.85 at 35,000 feet	
Dynamic pressure, lb/sq ft	252
True velocity, ft/sec	827
$C_{m_{\dot{\theta}}} + C_{m_{\dot{\alpha}}}$ (assumed invariant with α), per radian per sec	-0.144
$C_{m_{\dot{\theta}}}$, per radian per sec	$\frac{1.25 l_t}{V} C_{m_{\delta_s}}(\alpha)$
Mach number, 1.20 at 35,000 ft	
Dynamic pressure, lb/sq ft	502
True velocity, ft/sec	1,168
$C_{m_{\dot{\theta}}} + C_{m_{\dot{\alpha}}}$ (assumed invariant with α), per radian per sec	-0.09
$C_{m_{\dot{\theta}}}$, per radian per sec	$\frac{1.25 l_t}{V} C_{m_{\delta_s}}(\alpha)$
Mach number, 0.25 at sea level	
Dynamic pressure, lb/sq ft	98
True velocity, ft/sec	287
$C_{m_{\dot{\theta}}} + C_{m_{\dot{\alpha}}}$ (assumed invariant with α), per radian per sec	-0.267
$C_{m_{\dot{\theta}}}$, per radian per sec	$\frac{1.25 l_t}{V} C_{m_{\delta_s}}(\alpha)$

TABLE I.- AIRPLANE CHARACTERISTICS USED IN ANALYSIS - Concluded

(b) Airplane B

For design combat configuration:	
Airplane weight, W, lb	15,200
Airplane mass, m, slugs	472
Airplane pitching moment of inertia, I_y, slug-ft^2	56,655
Center-of-gravity position, percent $\bar{c}$	7
Horizontal-tail length, l_t, ft	20.4
Mach number, 0.80 at 35,000 ft	
Dynamic pressure, q, lb/sq ft	222
True velocity, V, ft/sec	778
$C_{m_{\dot{\theta}}} + C_{m_{\dot{\alpha}}}$ (assumed invariant with α), per radian per sec	-0.081
$C_{m_{\dot{\theta}}}$, per radian per sec	$\frac{1.25 l_t}{V} C_{m_{\delta_s}}(\alpha)$
Mach number, 0.975 at 35,000 ft	
Dynamic pressure, q, lb/sq ft	330
True velocity, V, ft/sec	949
$C_{m_{\dot{\theta}}} + C_{m_{\dot{\alpha}}}$ (assumed invariant with α), per radian per sec	-0.093
$C_{m_{\dot{\theta}}}$, per radian per sec	$\frac{1.25 l_t}{V} C_{m_{\delta_s}}(\alpha)$
Minimum landing weight configuration:	
Airplane weight, W, lb	12,360
Airplane mass, m, slugs	384
Airplane pitching moment of inertia, I_y, slug-ft^2	52,752
Center-of-gravity position, percent $\bar{c}$	20
Horizontal-tail length, l_t, ft	19.16
Mach number, 0.23 at sea level	
Dynamic pressure, q, lb/sq ft	78
True velocity, V, ft/sec	256
$C_{m_{\dot{\theta}}} + C_{m_{\dot{\alpha}}}$ (assumed invariant with α), per radian per sec	-0.145
$C_{m_{\dot{\theta}}}$, per radian per sec	$\frac{1.25 l_t}{V} C_{m_{\delta_s}}(\alpha)$
Horizontal tail	
Deflection limits, deg	-17.5,+5
Maximum deflection rate, deg/sec	25
Wing area, S, sq ft	196.1
Wing mean aerodynamic chord, $\bar{c}$, ft	9.54

TABLE II.- QUESTIONNAIRE FOR PILOT PITCH-UP RATING

I Is pitch-up region useful at all for maneuvering? Yes or No.

II Consider the following situations:

A. If you are tracking a target airplane and enter the pitch-up region, what is your assigned rating of your ability to return to or remain on the correct flight path to continue the tracking?

B. If you have entered the pitch-up region during a gunnery run, what rating would you give the airplane as a gun platform in the pitch-up region?

C. If rating for A and B is poor, is reason other than insufficient or inadequate controllability?

D. How would you rate this airplane with regard to the tendency for a pilot to apply rapid and perhaps excessive control during pitch-up recoveries?

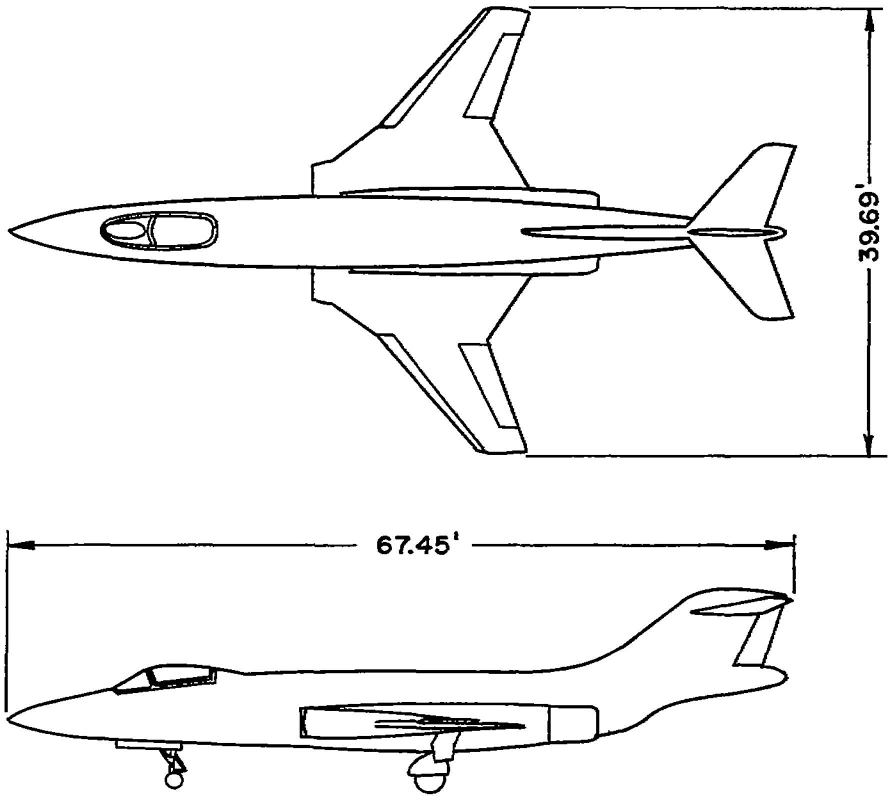
III Rate the pitch-up according to abruptness. (What is response quantity which you feel is related to the abruptness of pitch-up?)

IV Rate the pitch-up according to overshoot load factor. (What is your definition of overshoot load factor?)

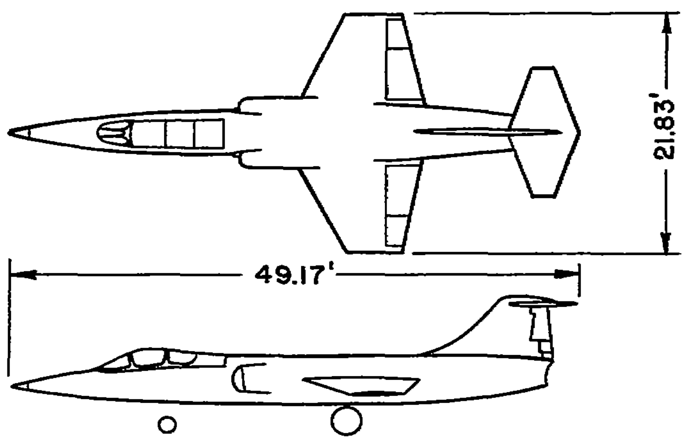
V What rating do you assign the airplane with regard to how much pitch-up restricts or limits maneuverability of the airplane?

TABLE III.- PILOT RATING SCHEDULE FOR PITCH-UP

Rating	Category	Description
0	Satisfactory	Satisfies stability and control requirements.
1 2	Marginally Satisfactory	Pitch-up barely perceptible. Does not appreciably diminish usefulness of the airplane in performing a desired task. Abruptness of airplane response and overshoot in attitude or load factor during pitch-up not much increased over comparable satisfactory airplane. Little tendency for the pilot to apply rapid and excessive corrective control.
3 4 5	Unsatisfactory but Acceptable	Pitch-up is more apparent. More or less difficulty experienced in performing the desired task. Abruptness of airplane motion and overshoot in attitude or load factor during pitch-up considerably increased over that for marginally satisfactory airplane. There may be some tendency for the pilot to apply rapid and perhaps excessive corrective control.
6 7 8	Unsatisfactory	Pitch-up severe ranging from controllable only with the greatest difficulty to practically uncontrollable. Abruptness of airplane motions during pitch-up approaching degree where pilot feels he has little or no control over the overshoots in attitude or load factor, which are relatively large. Increased tendency for the pilot to apply rapid and excessive corrective control.
9 10	Unacceptable	Pitch-up so severe that airplane is uncontrollable. The abruptness of the airplane motions and the magnitude of the overshoots are so extreme, even at high altitude, that the pilot would not consider approaching the pitch-up boundary because of concern for the structural integrity of the airplane. Some possibility of entering into a spin or other unusual maneuver from which recovery may be difficult or impossible.



(a) Airplane A.



(b) Airplane B.

Figure 1.- Two-view drawings of the two example airplanes.

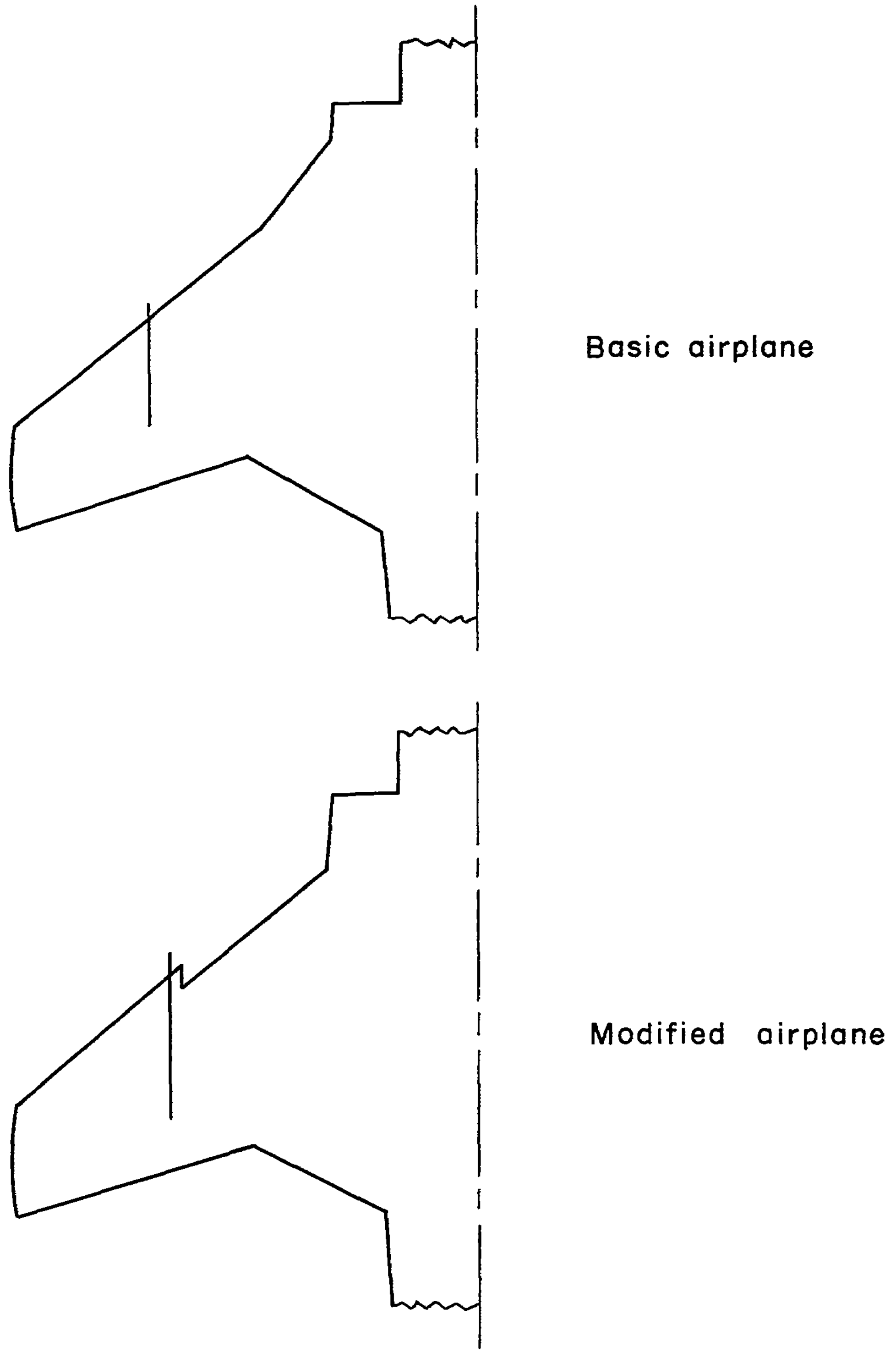


Figure 2.- Wing plan forms of airplane A.

[redacted]

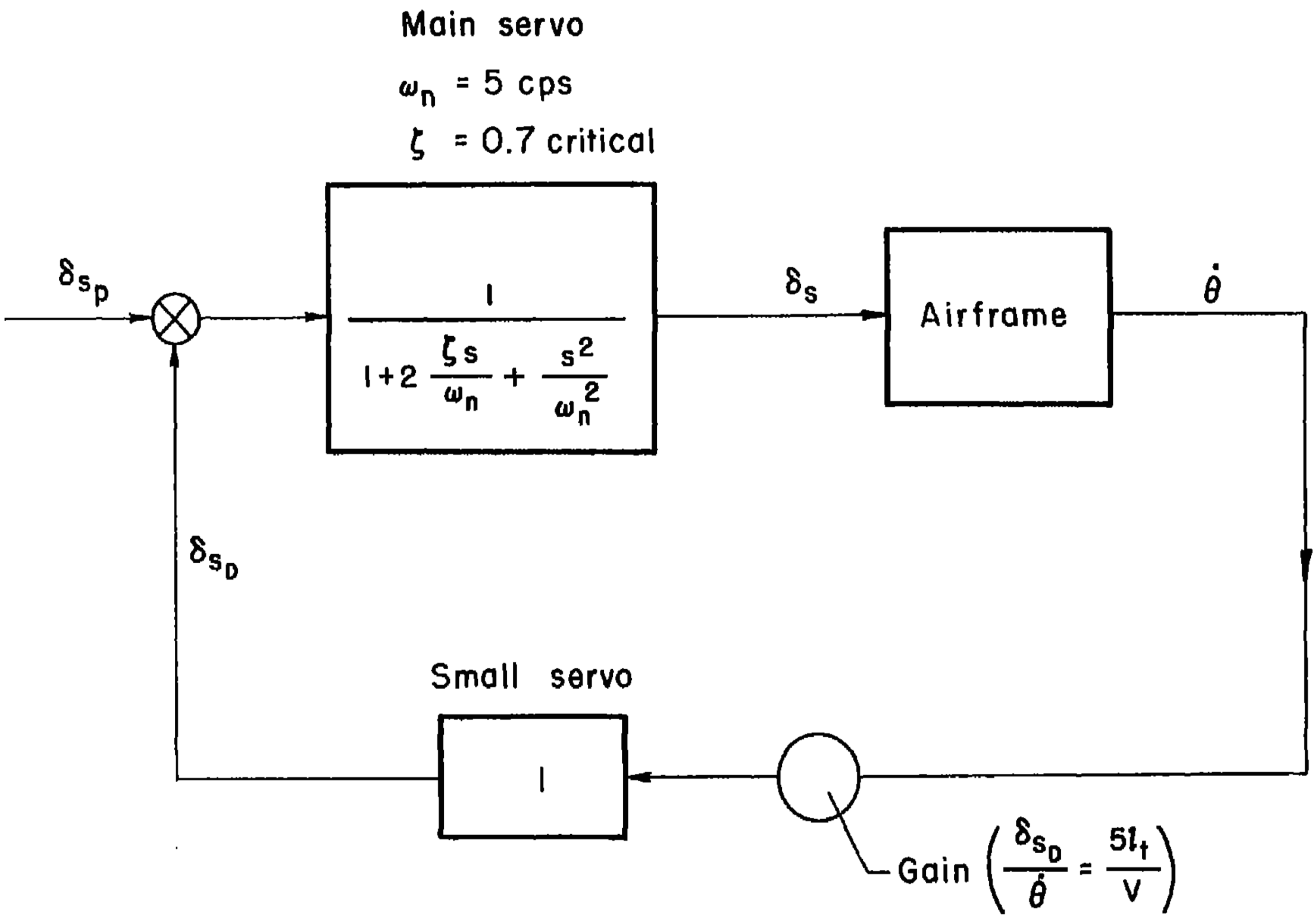


Figure 3.- Block diagram of linear pitch damper.

[redacted]

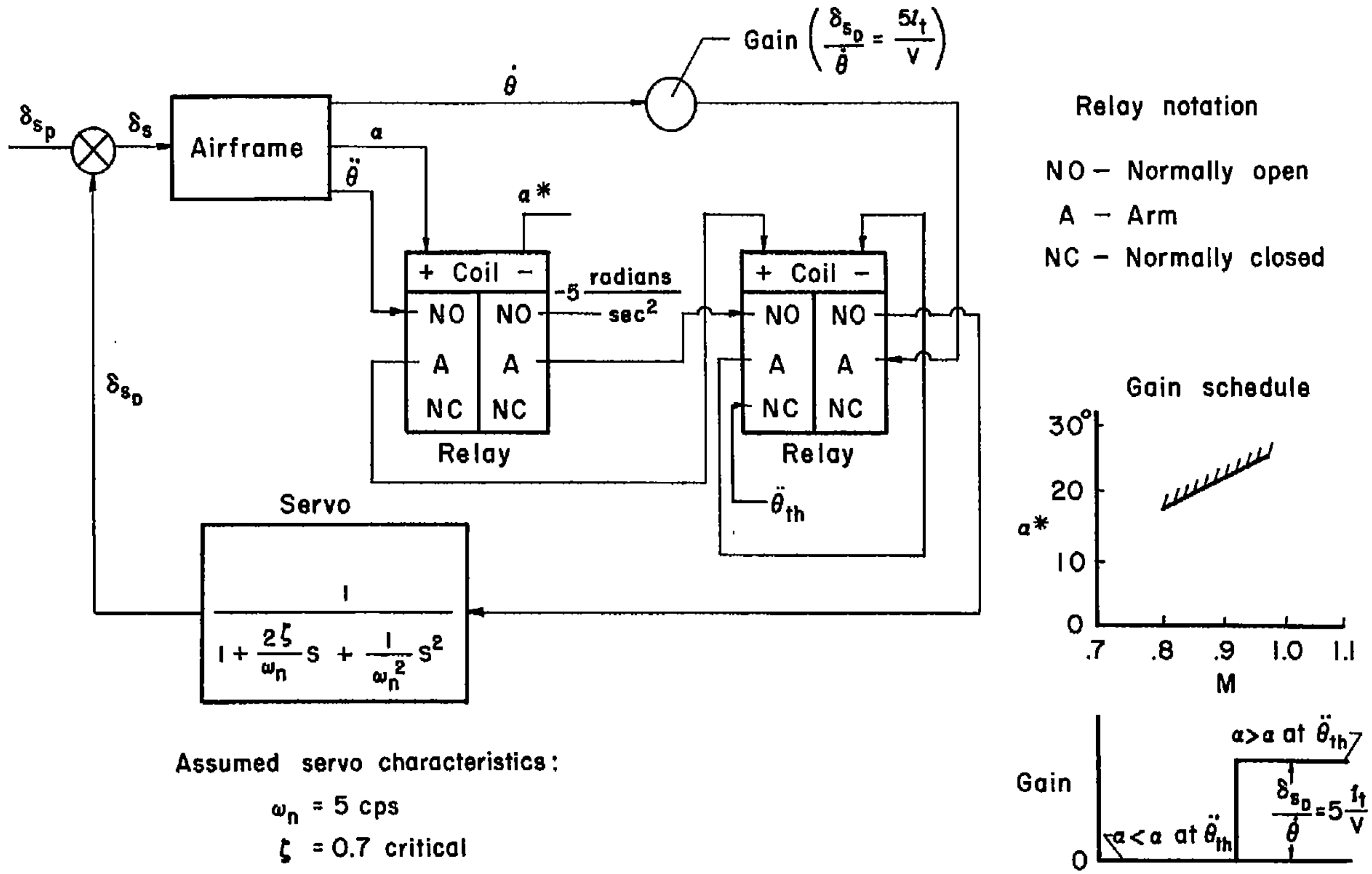


Figure 4.- Block diagram of nonlinear pitch damper.

[redacted]

[redacted]

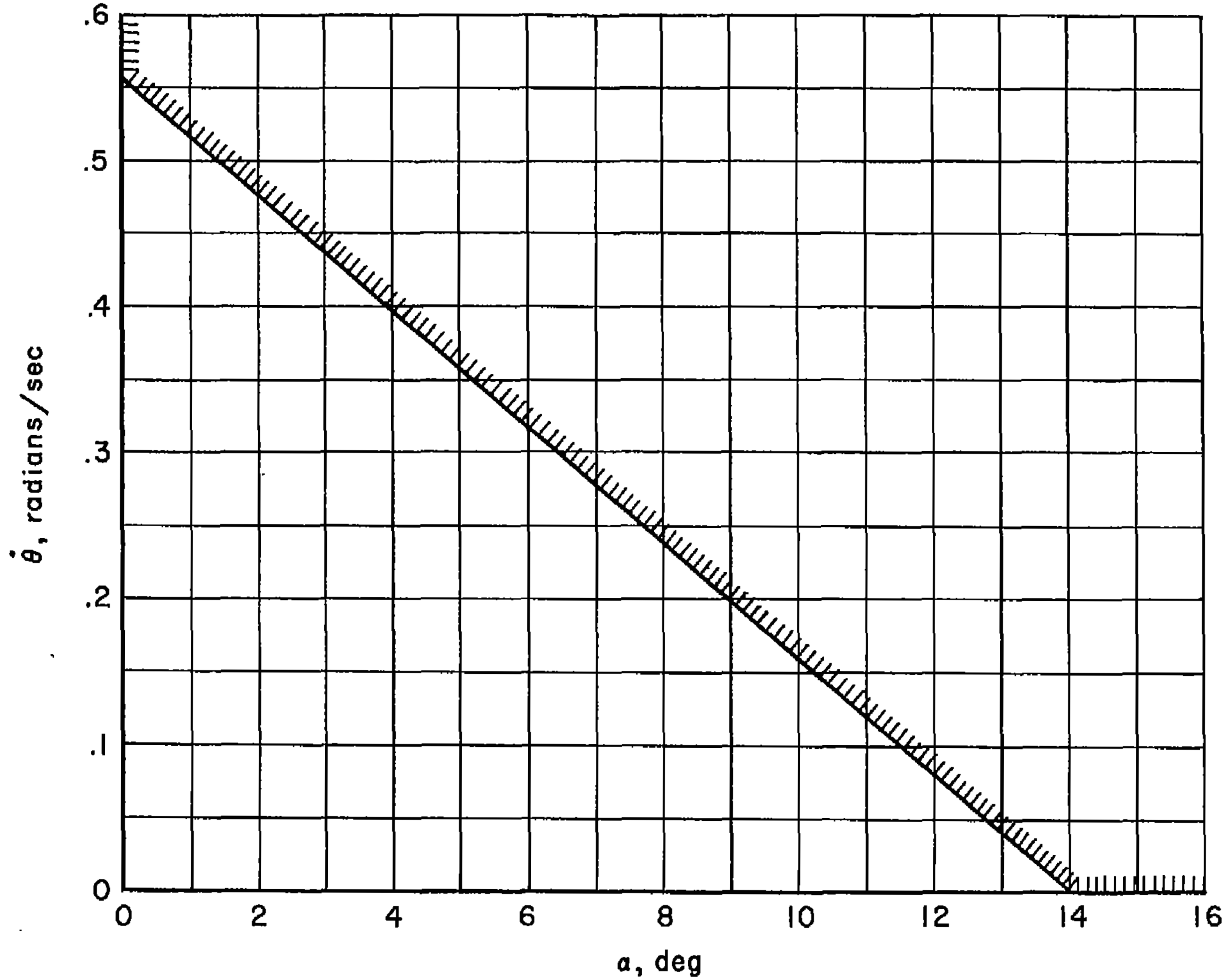
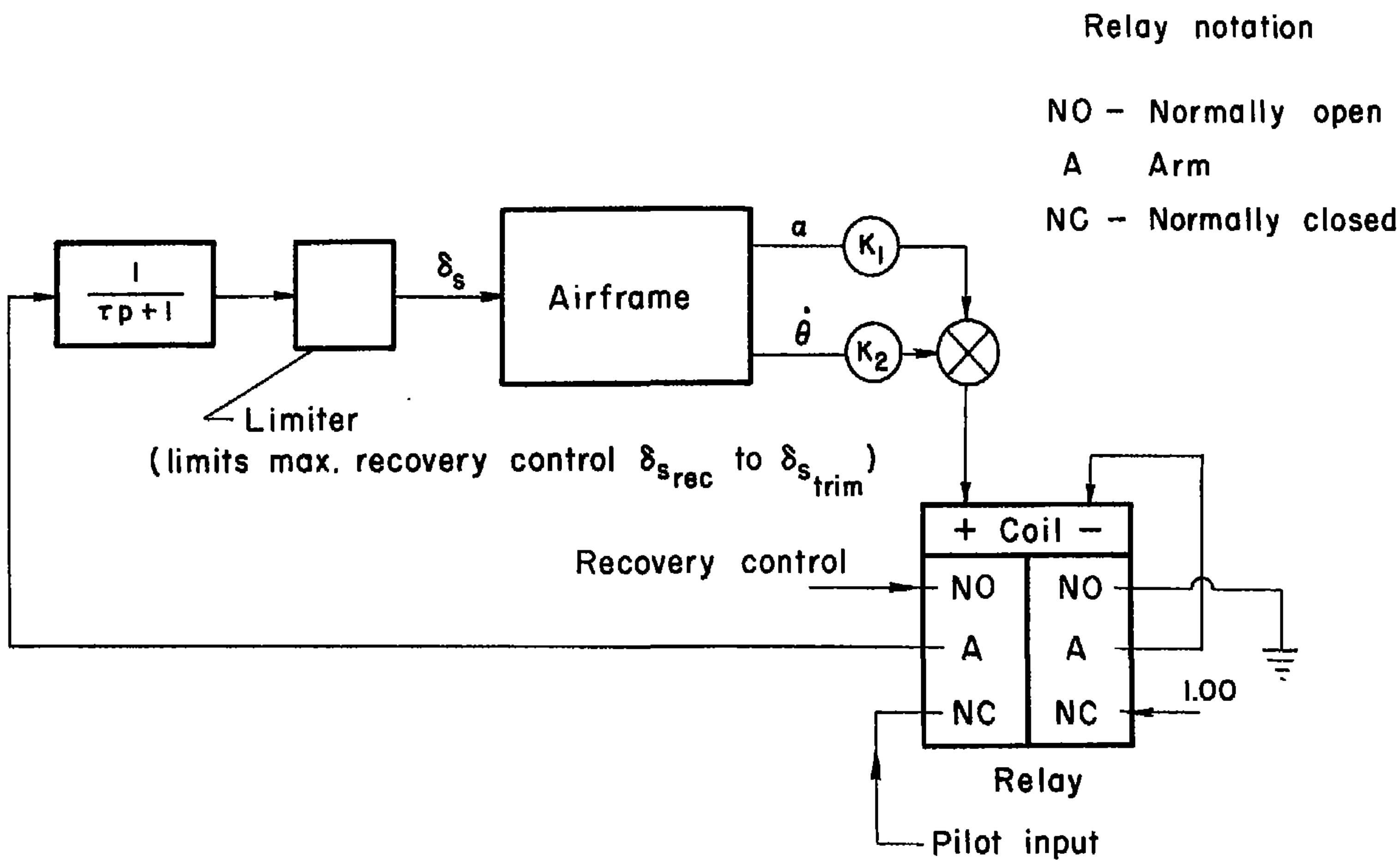


Figure 5.- Typical operational envelope for stick pusher.

[redacted]

[redacted]



$K_1\alpha + K_2\dot{\theta} = 1$ where: $K_1 = 0.0714$; $K_2 = 1.79$

(α given in deg, $\dot{\theta}$ in radians/sec)

Figure 6.- Block diagram representing stick pusher for analog study.

[redacted]

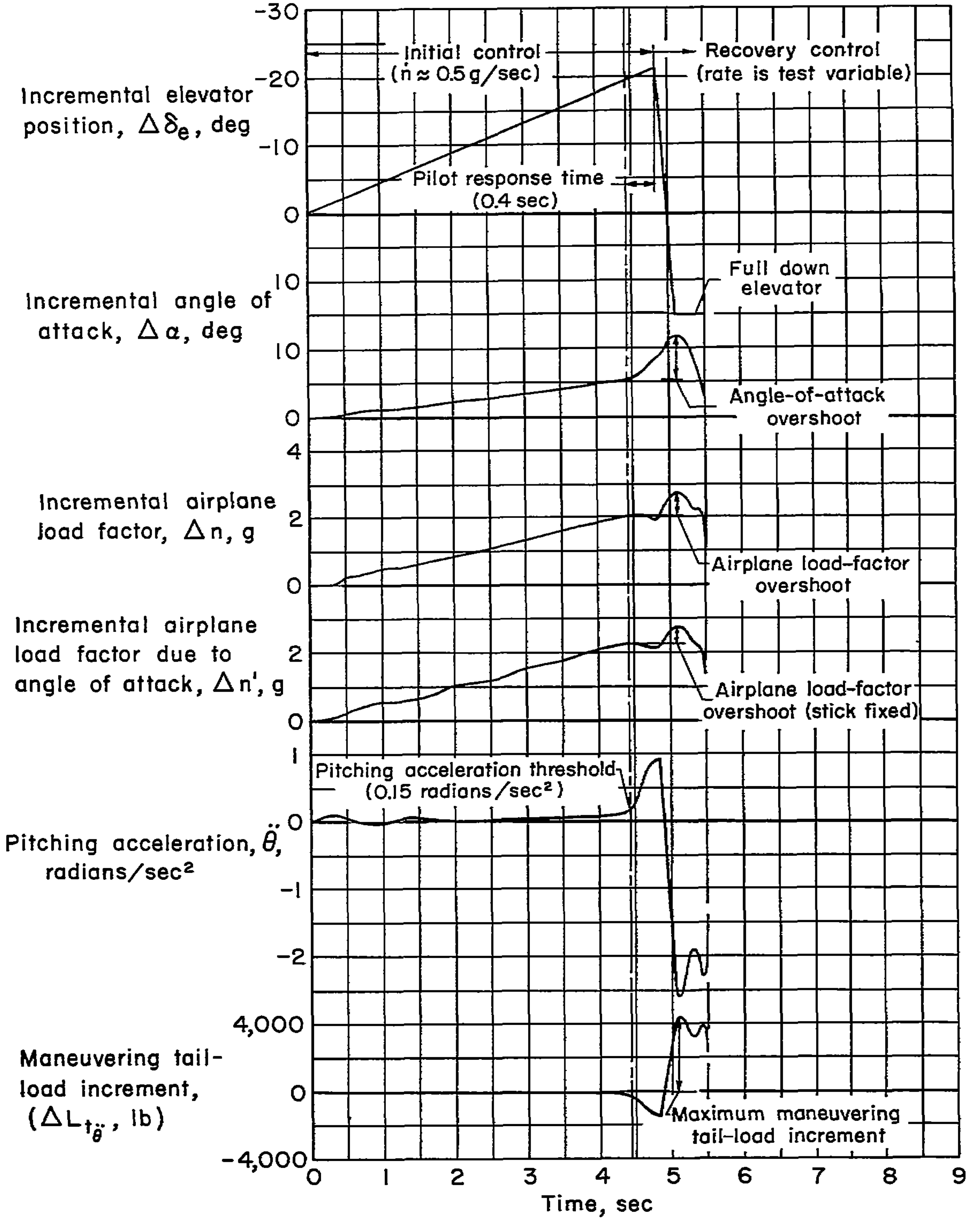
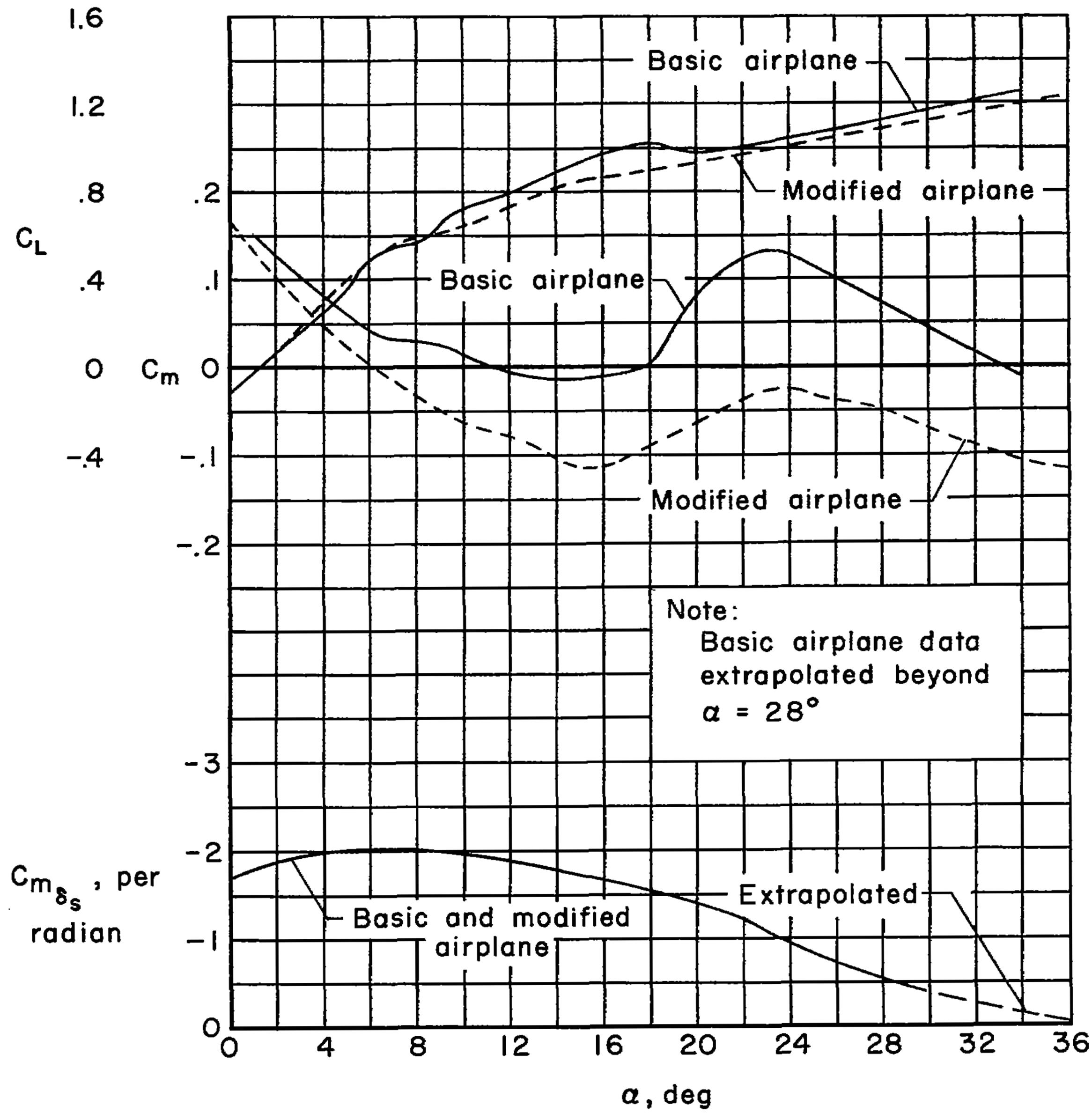
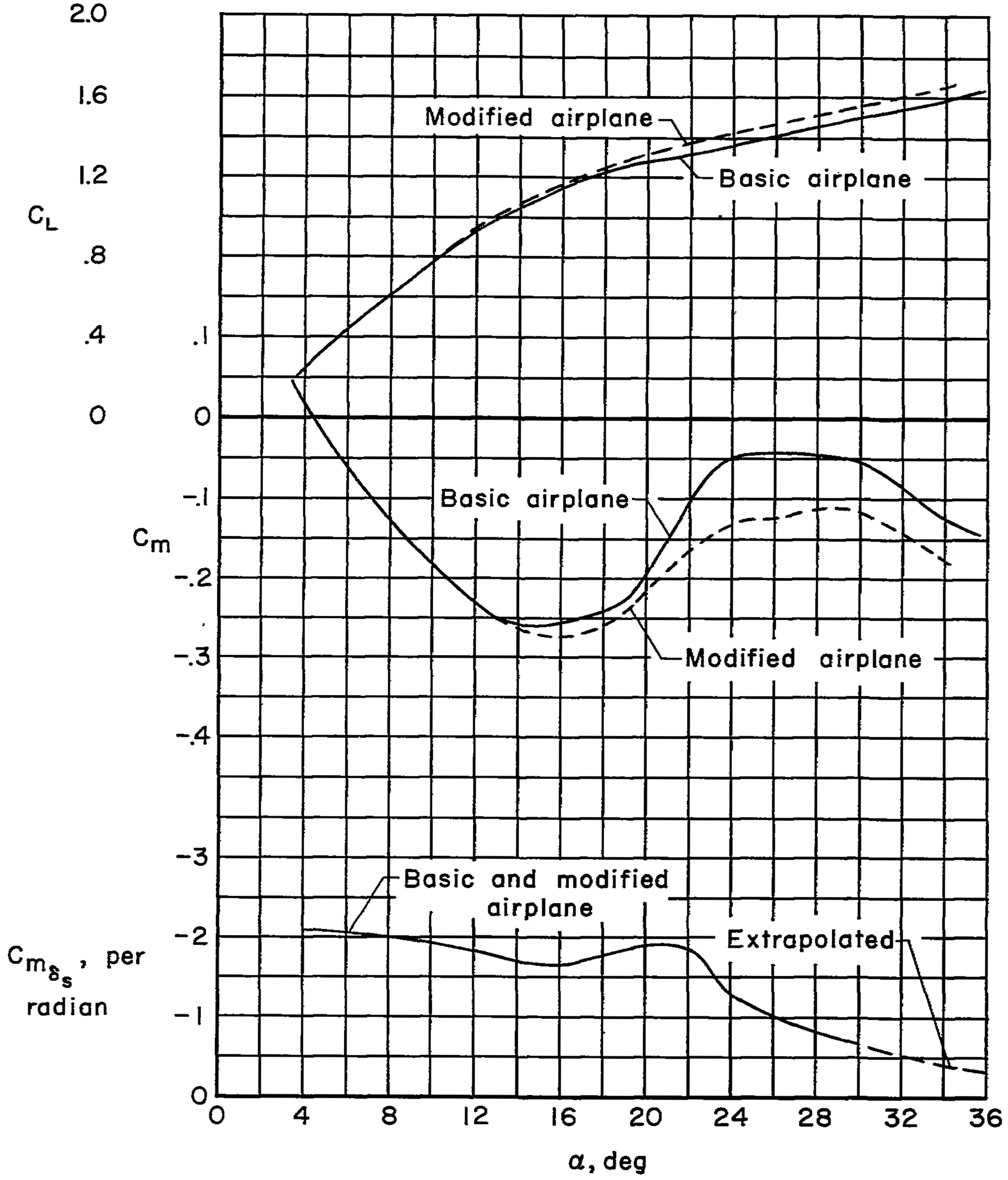


Figure 7.- Time history of standard pitch-up evaluation maneuver.



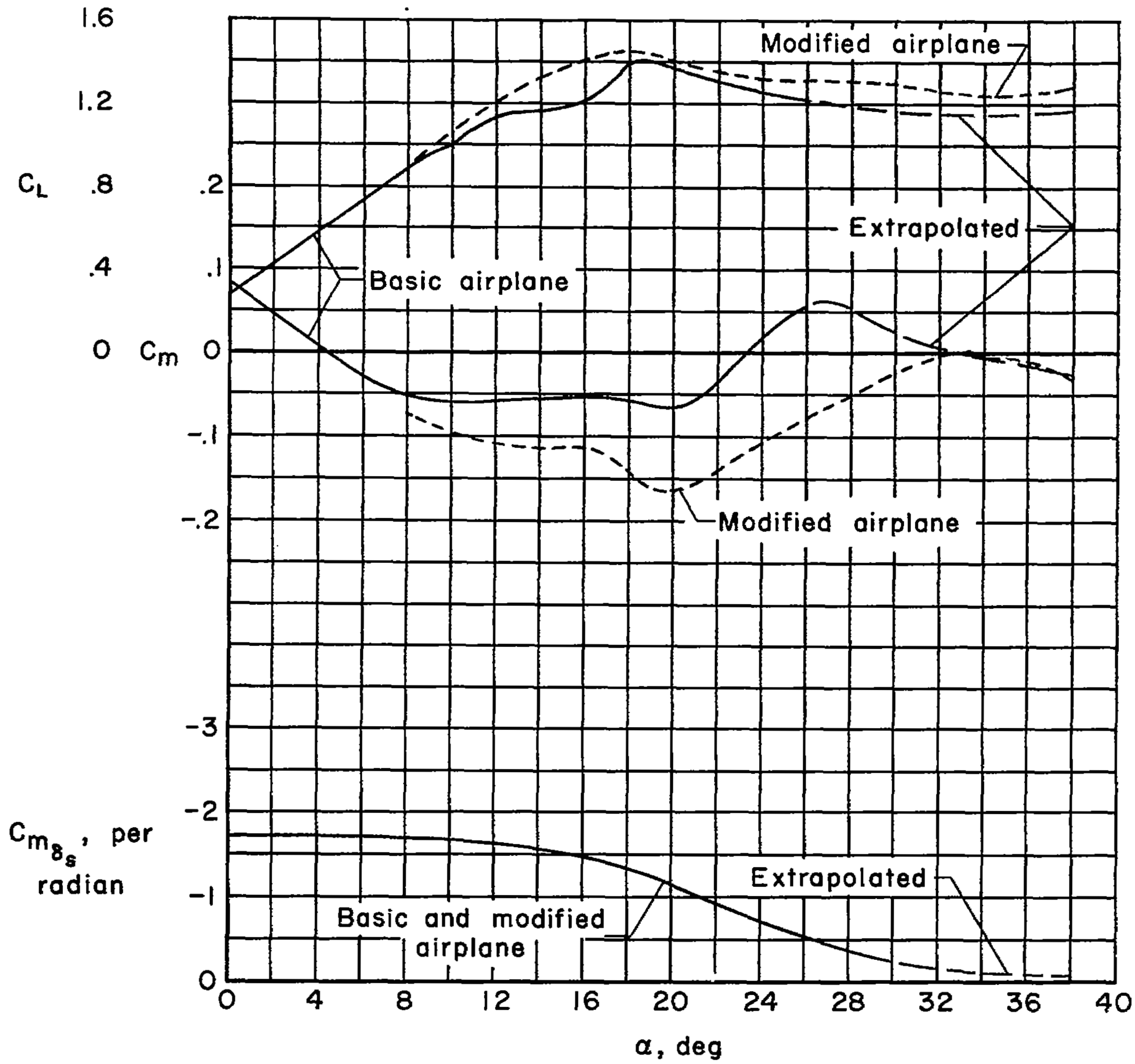
(a) Airplane A; δ_s = -4°; M = 0.85; h_p = 35,000 feet.

Figure 8.- Aerodynamic characteristics of the two fighter airplanes obtained from wind-tunnel measurements.



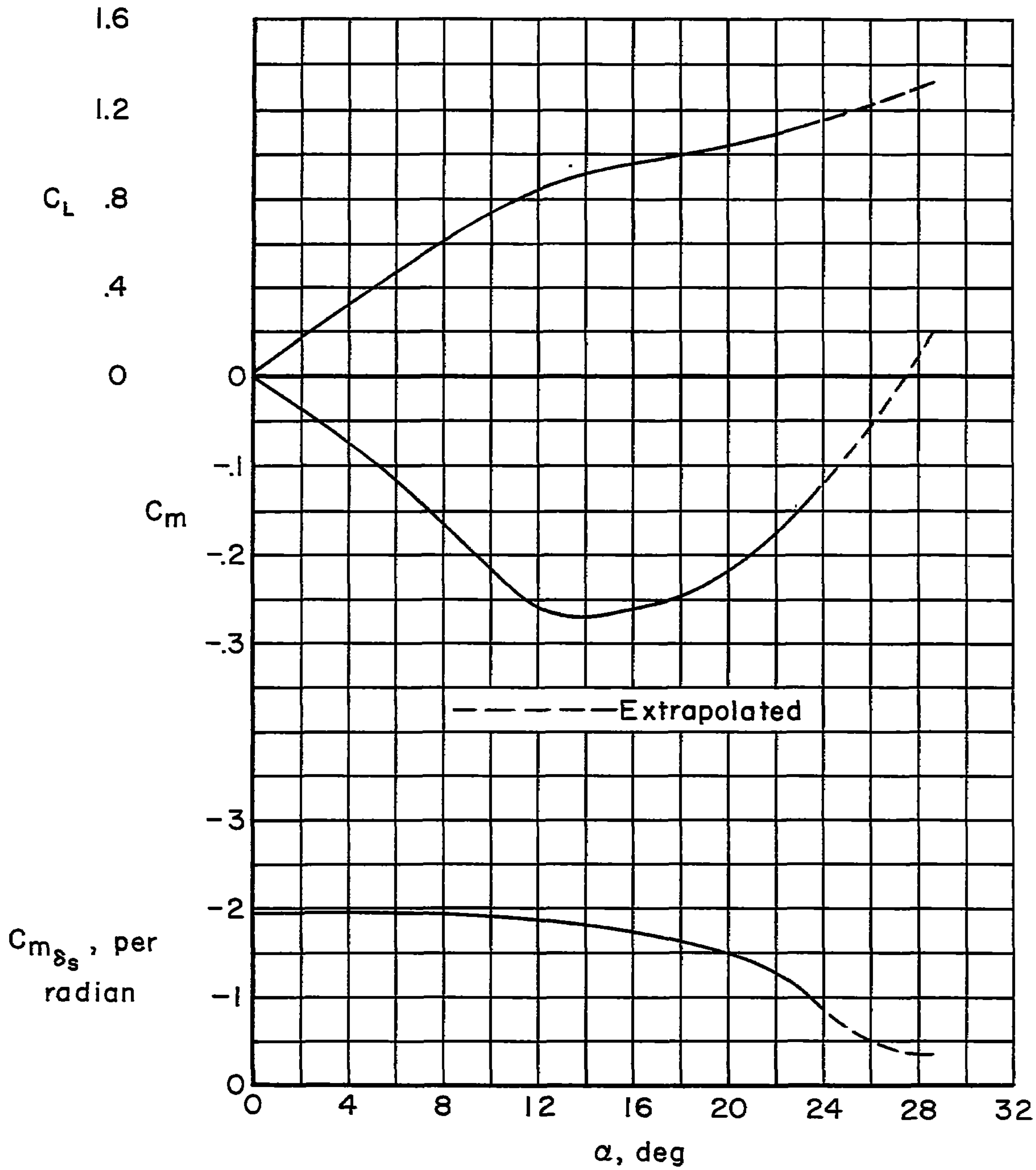
(b) Airplane A; δ_s = -4°; M = 1.20; h_p = 35,000 feet.

Figure 8.- Continued.



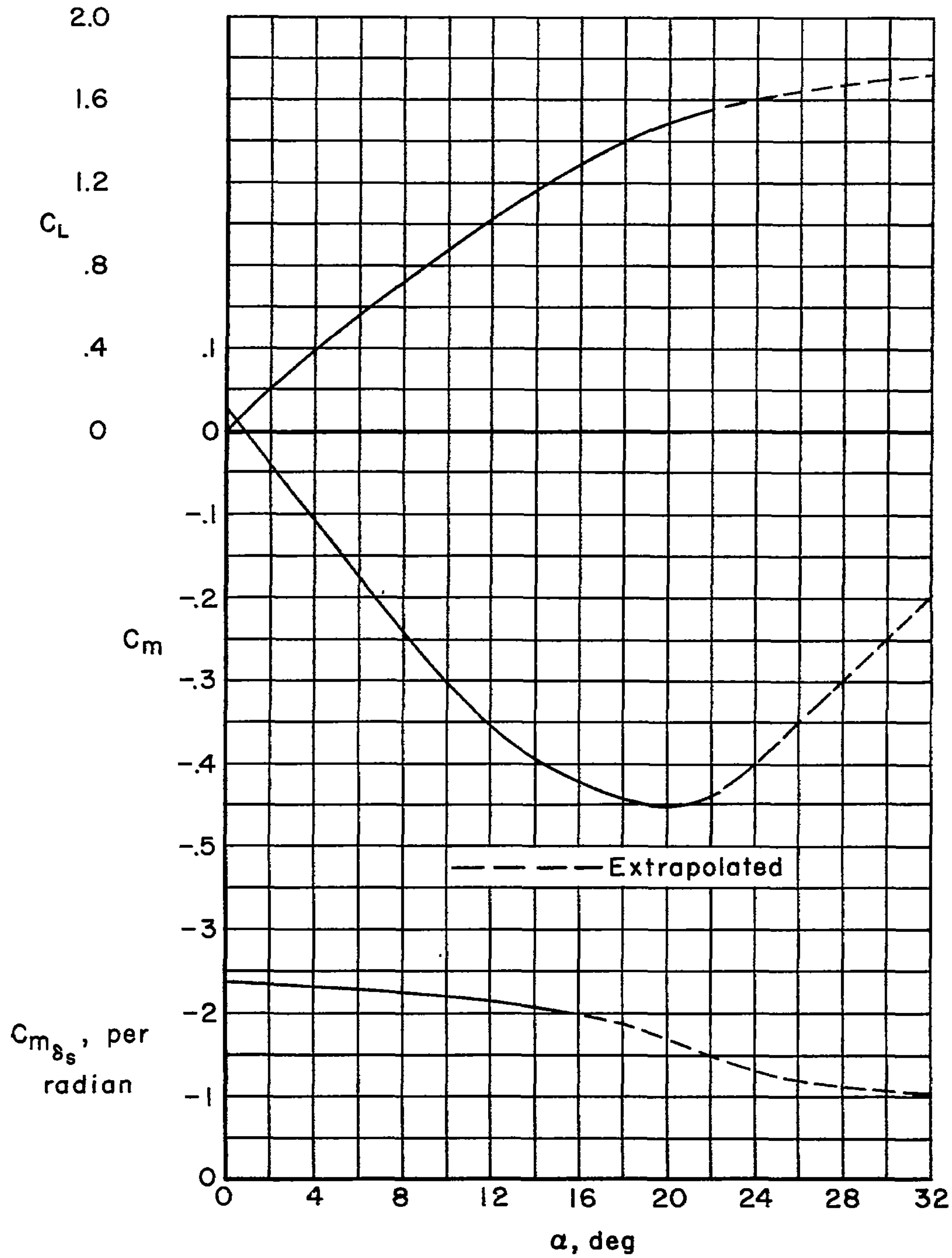
(c) Airplane A; $\delta_s = -4^{\circ}$; M = 0.25; h_p = sea level.

Figure 8.- Continued.



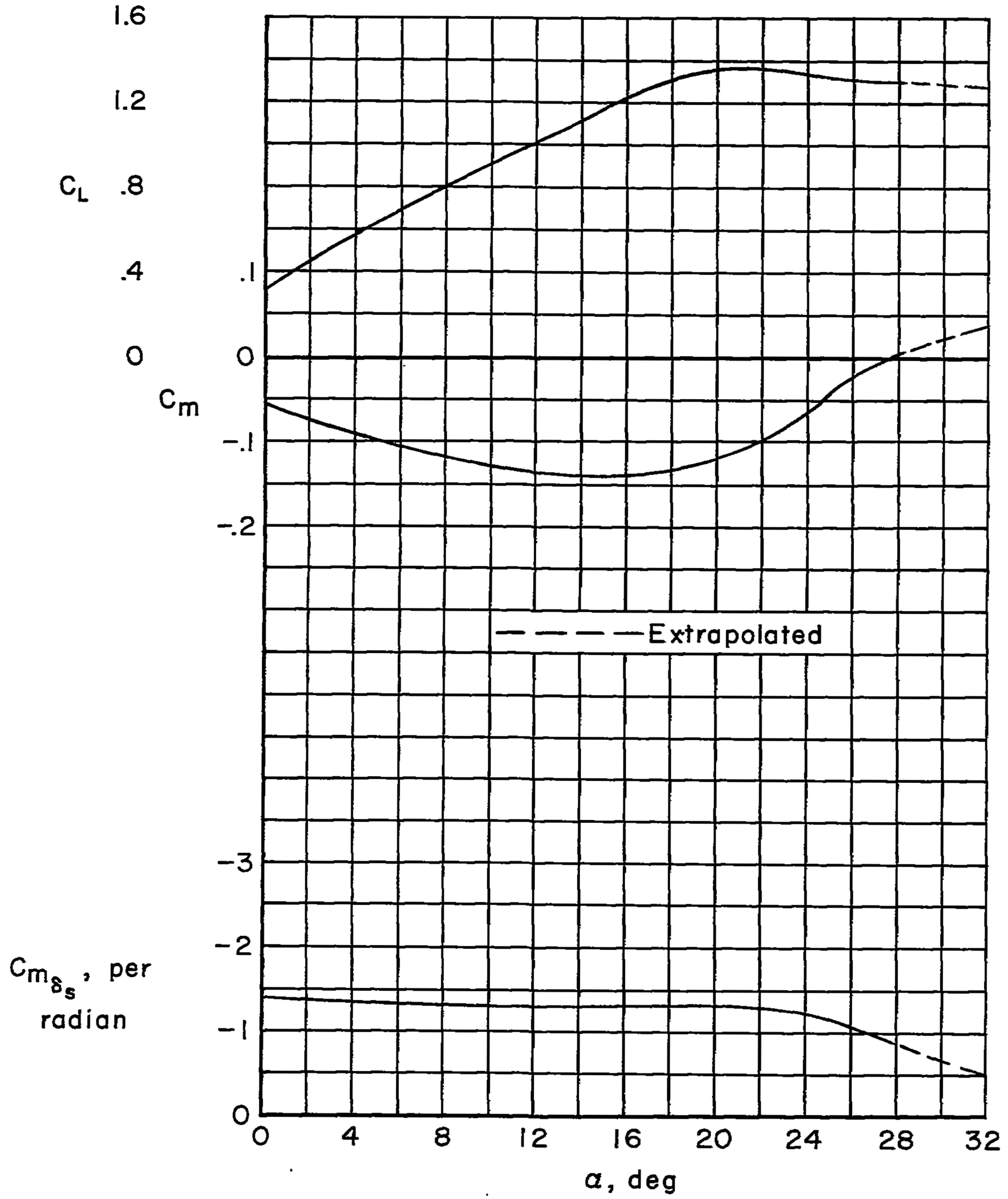
(d) Airplane B; $\delta_s = 0^o$; M = 0.80; h_p = 35,000 feet.

Figure 8.- Continued.



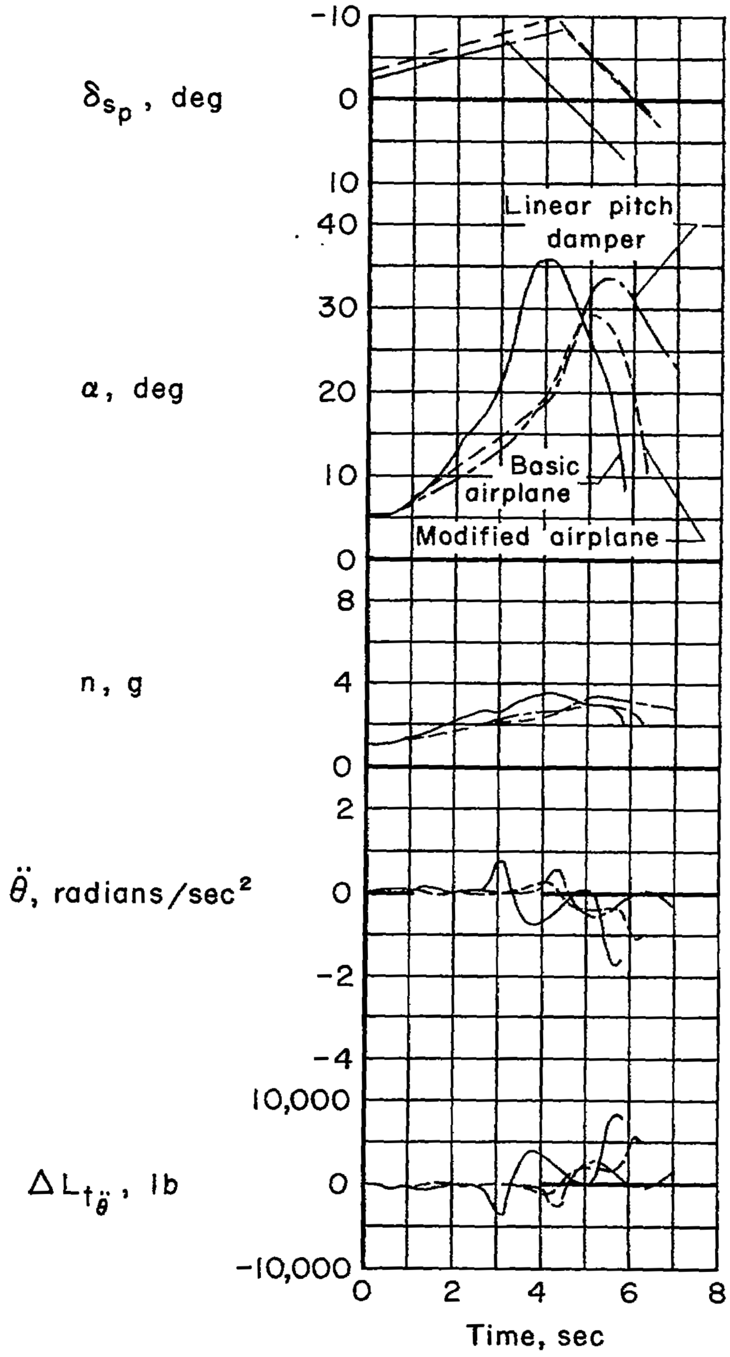
(e) Airplane B; $\delta_s = 0^{\circ}$; M = 0.975; h_p = 35,000 feet.

Figure 8.- Continued.



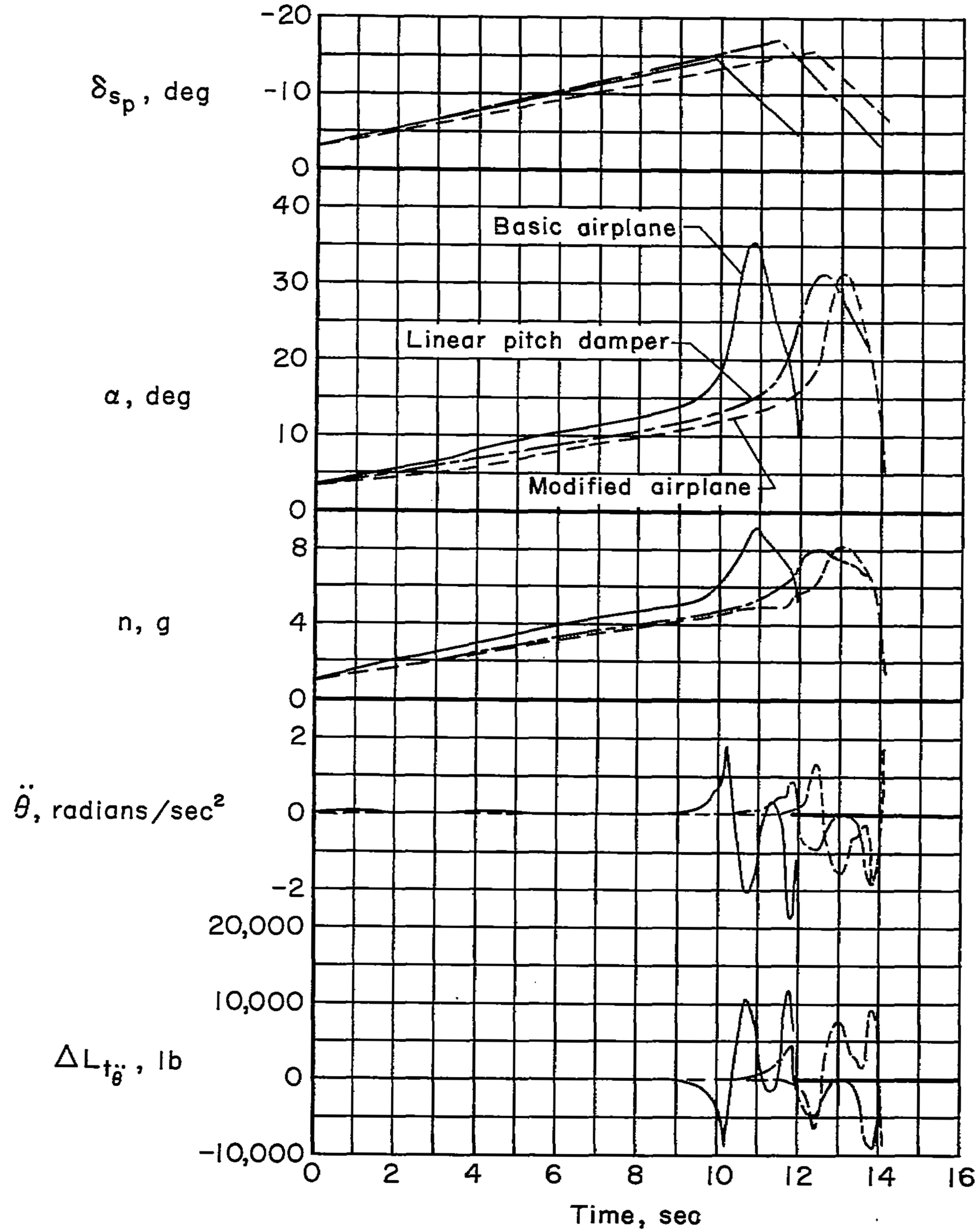
(f) Airplane B; $\delta_s = 0^\circ$; M = 0.23; h_p = sea level.

Figure 8.- Concluded.



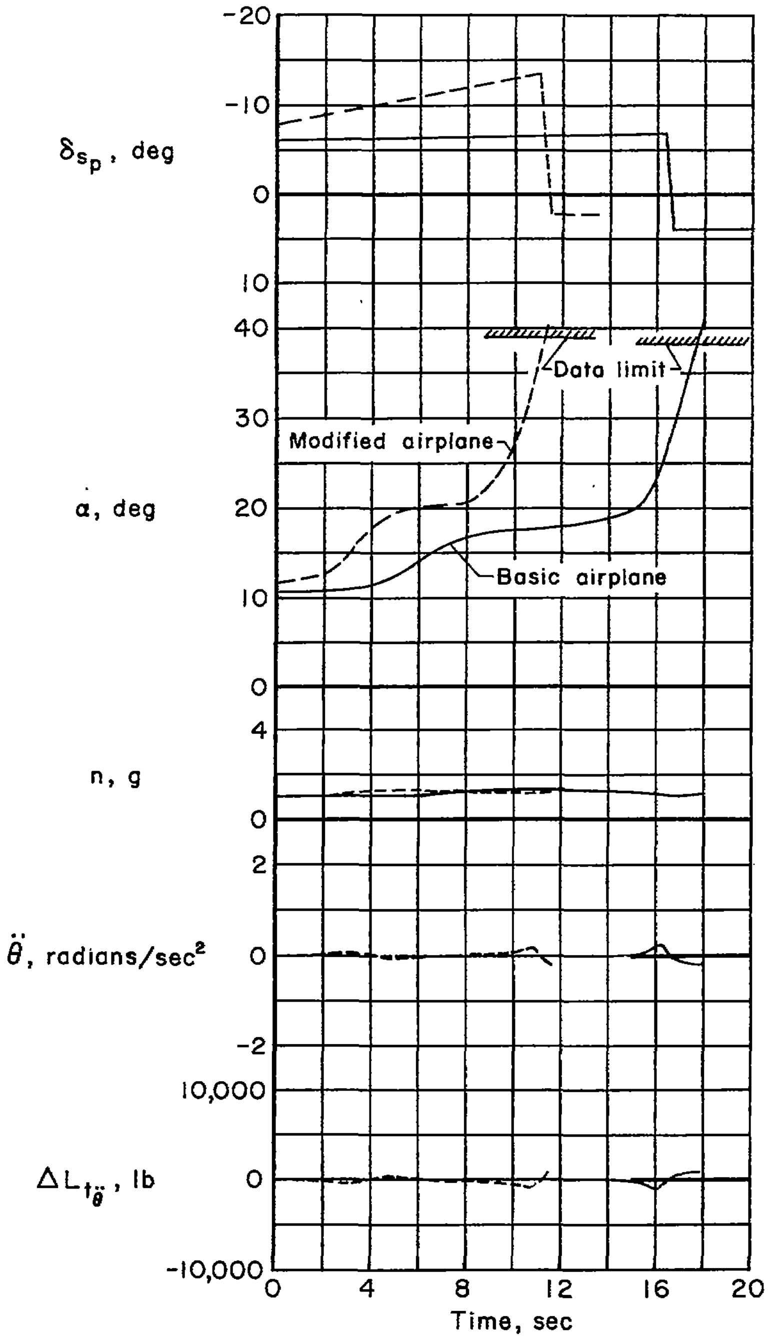
(a) M = 0.85 at 35,000 feet ($\dot{n} \approx 0.5$ g/sec).

Figure 9.- Typical time histories of computed pitch-up maneuvers; airplane A.



(b) M = 1.20 at 35,000 feet ($\dot{n} \approx 0.5$ g/sec).

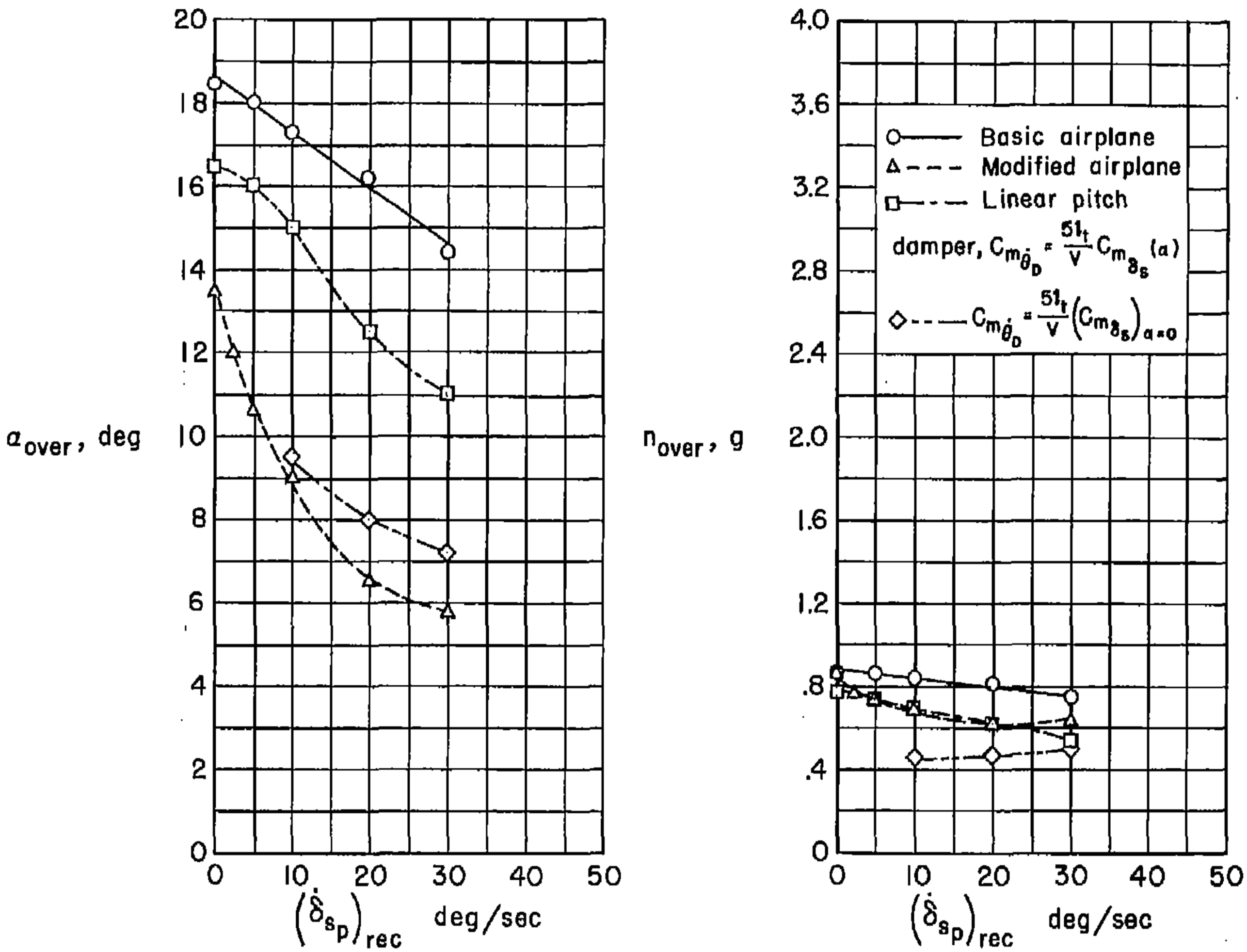
Figure 9.- Continued.



(c) M = 0.25 at sea level; initial $\dot{\delta}_{s_p}$ programmed to provide gradual low-speed stall entry.

Figure 9.- Concluded.

[redacted]

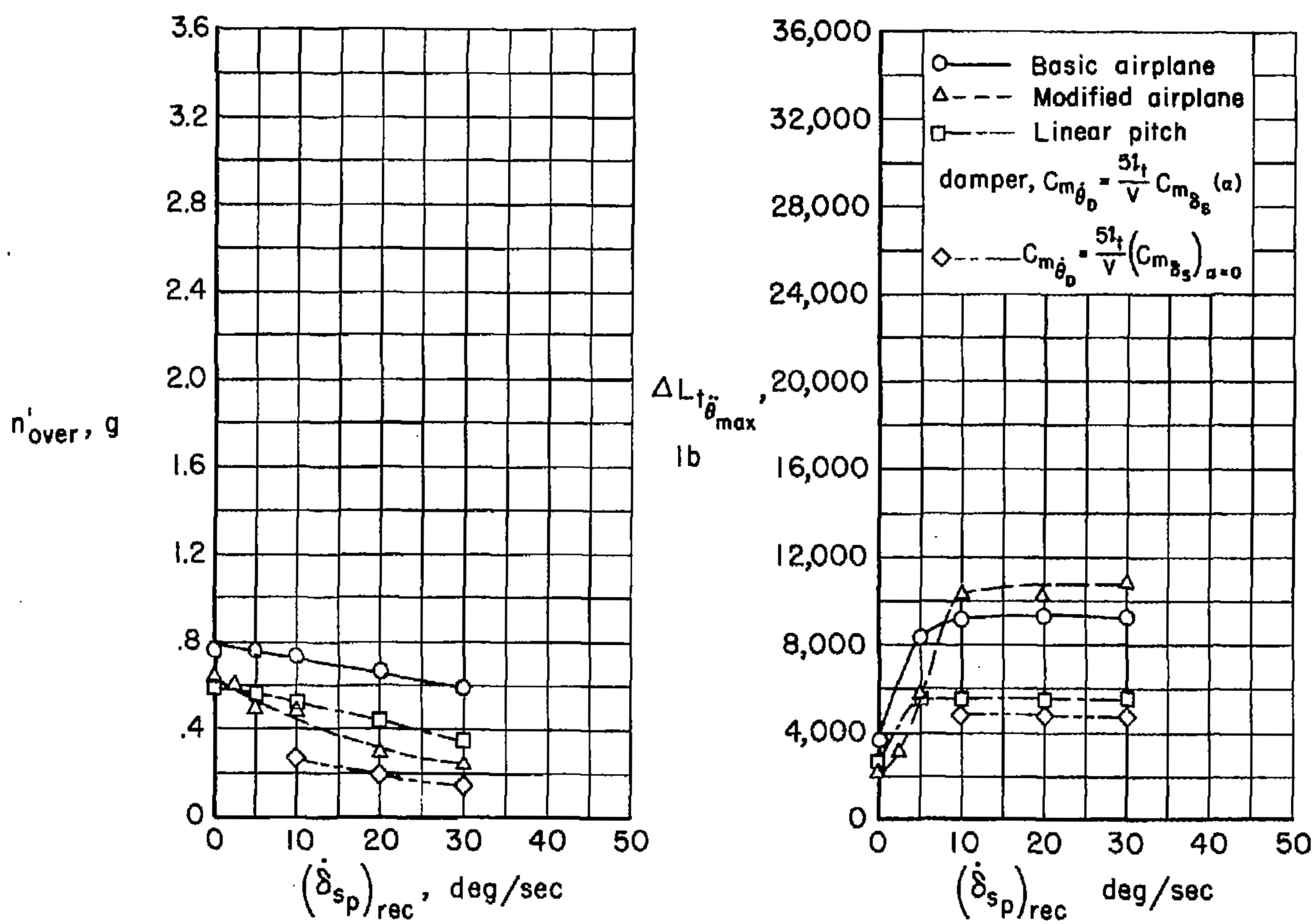


(a) α_{over} and n_{over}

Figure 10.- Variation of several response quantities with recovery-control rate; airplane A; M = 0.85; h_p = 35,000 feet.

[redacted]

[redacted]

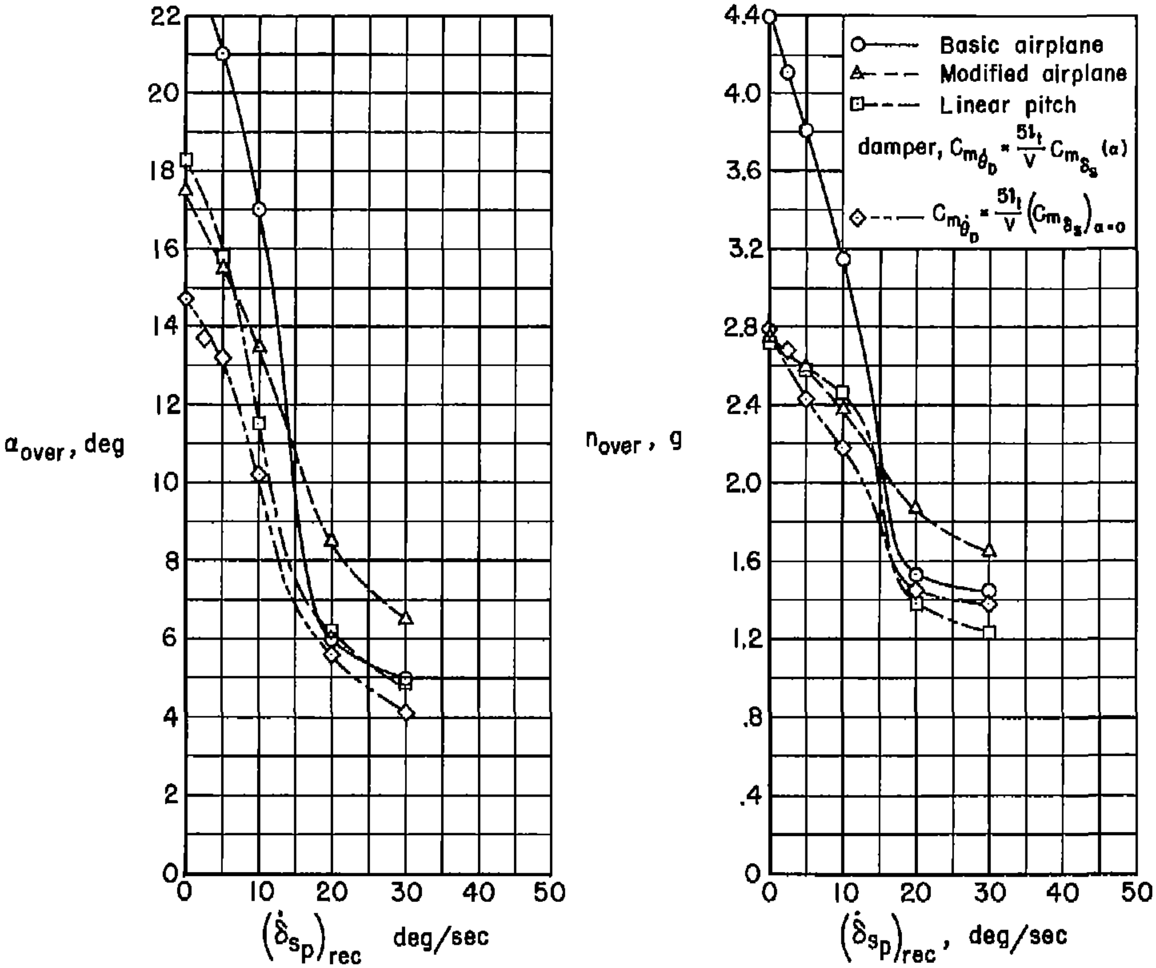


(b) n'_{over} and $\Delta L_{t_{\ddot{\theta}_{max}}}$

Figure 10.- Concluded.

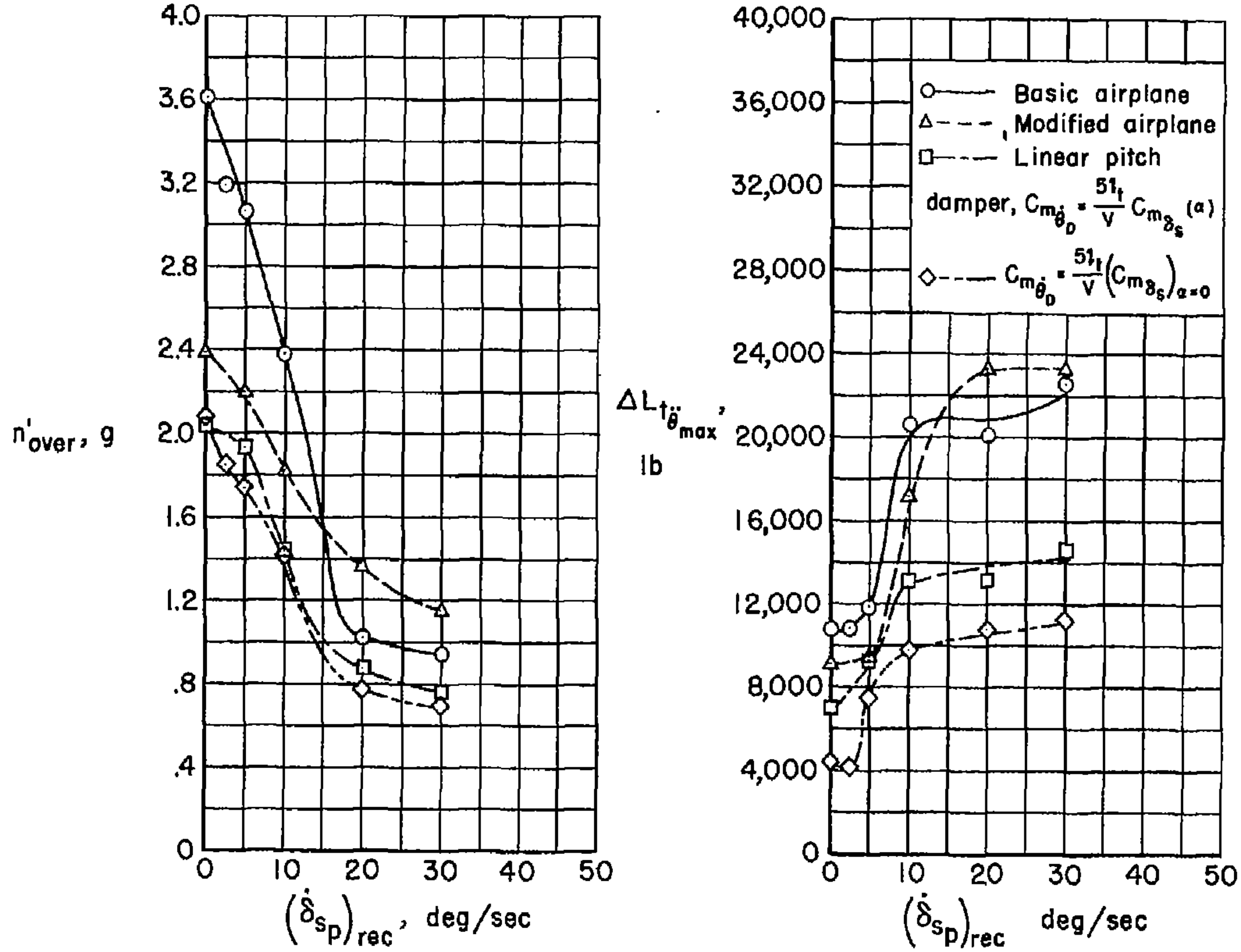
[redacted]

[redacted]



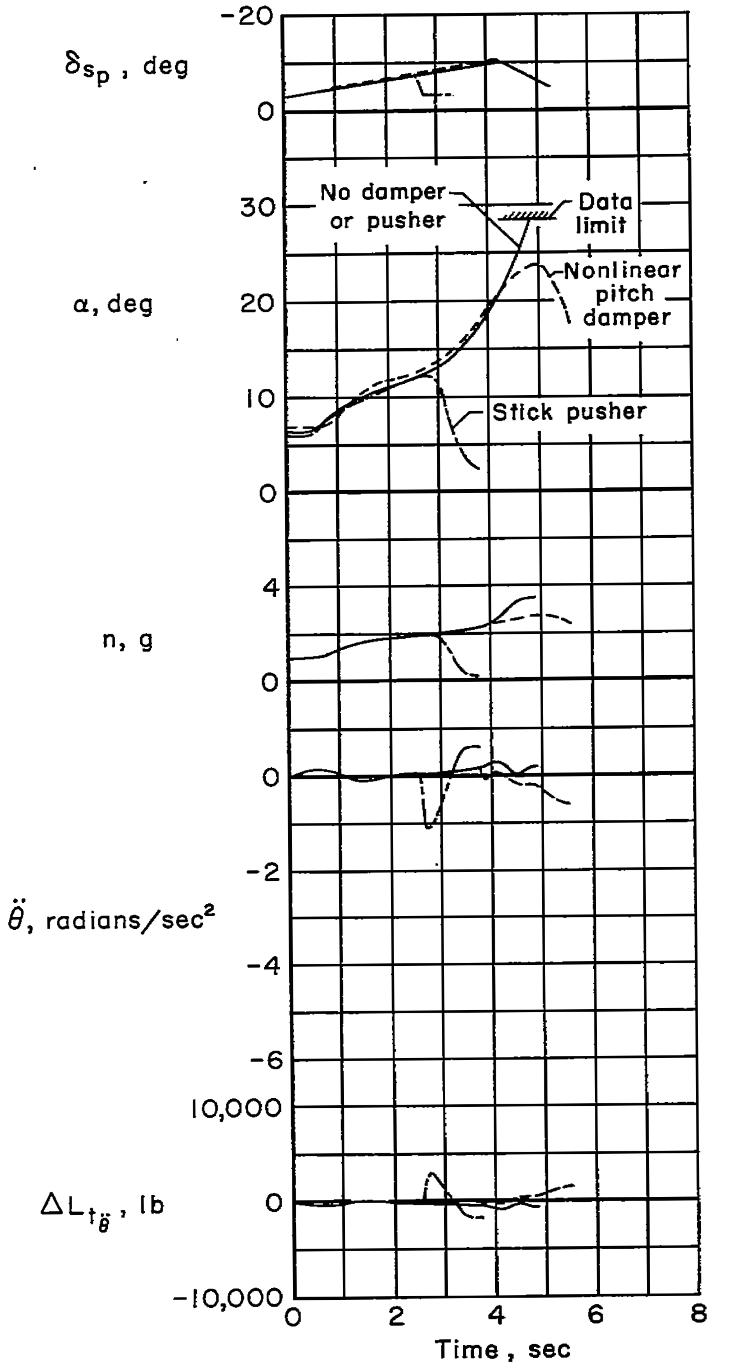
(a) α_{over} and n_{over}

Figure 11.- Variation of several response quantities with recovery-control rate; airplane A; $M = 1.20$; $h_p = 35,000$ feet.



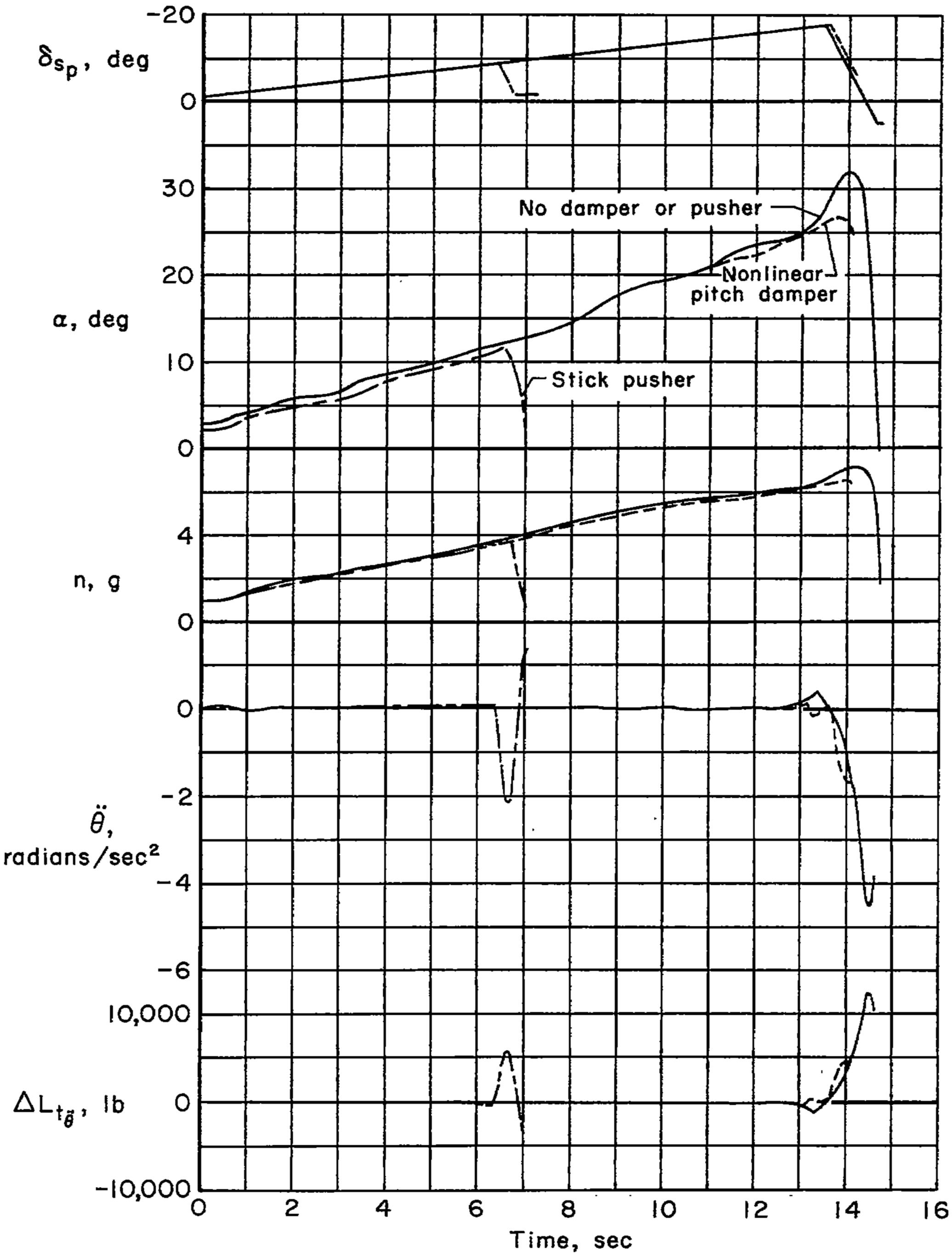
(b) n'_{over} and $\Delta L_{t_{\ddot{\theta}_{max}}}$

Figure 11.- Concluded.



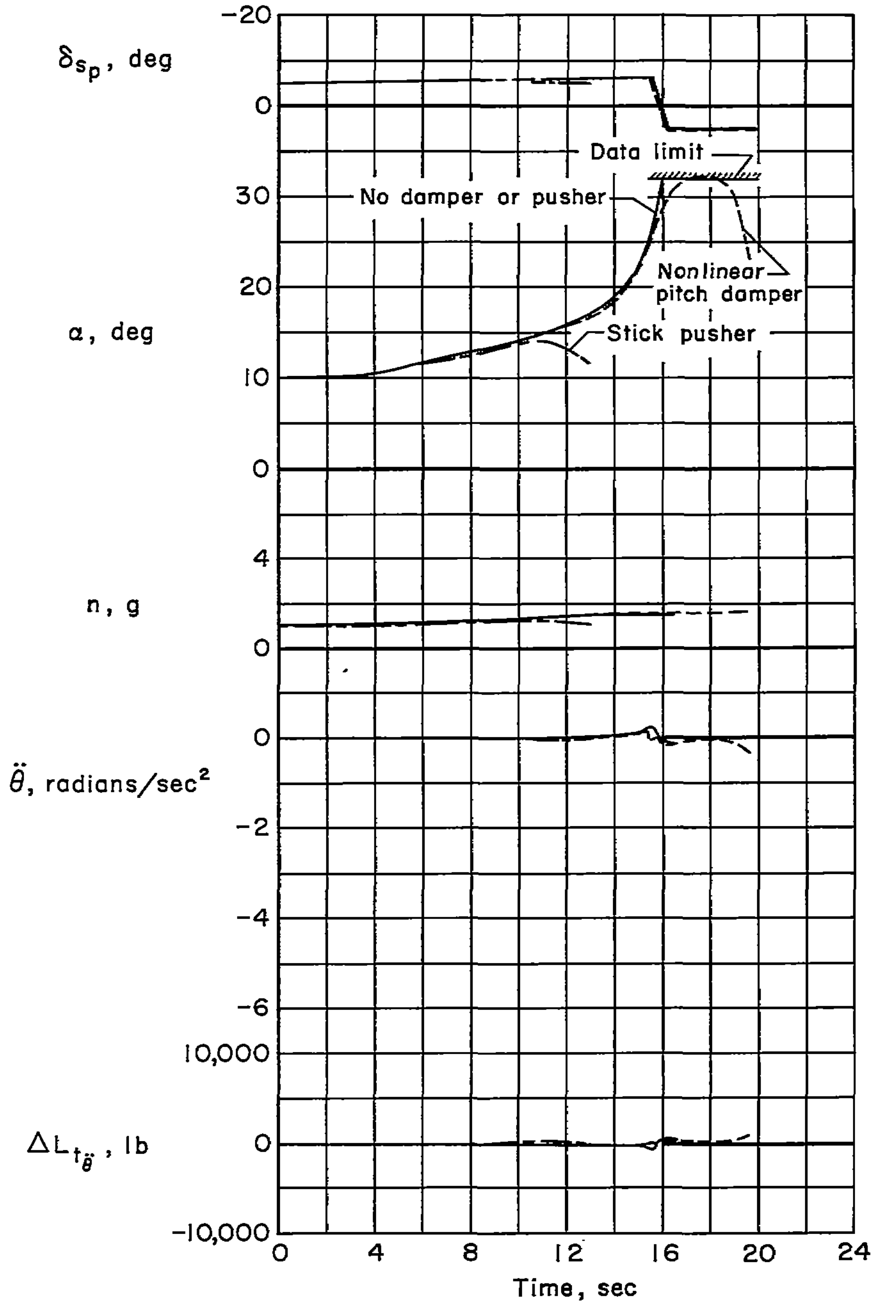
(a) M = 0.80 at h_p = 35,000 feet ($\dot{n} \approx 0.5$).

Figure 12.- Typical time histories of computed pitch-up maneuvers; airplane B.



(b) M = 0.975 at h_p = 35,000 feet ($\dot{n} \approx 0.5$).

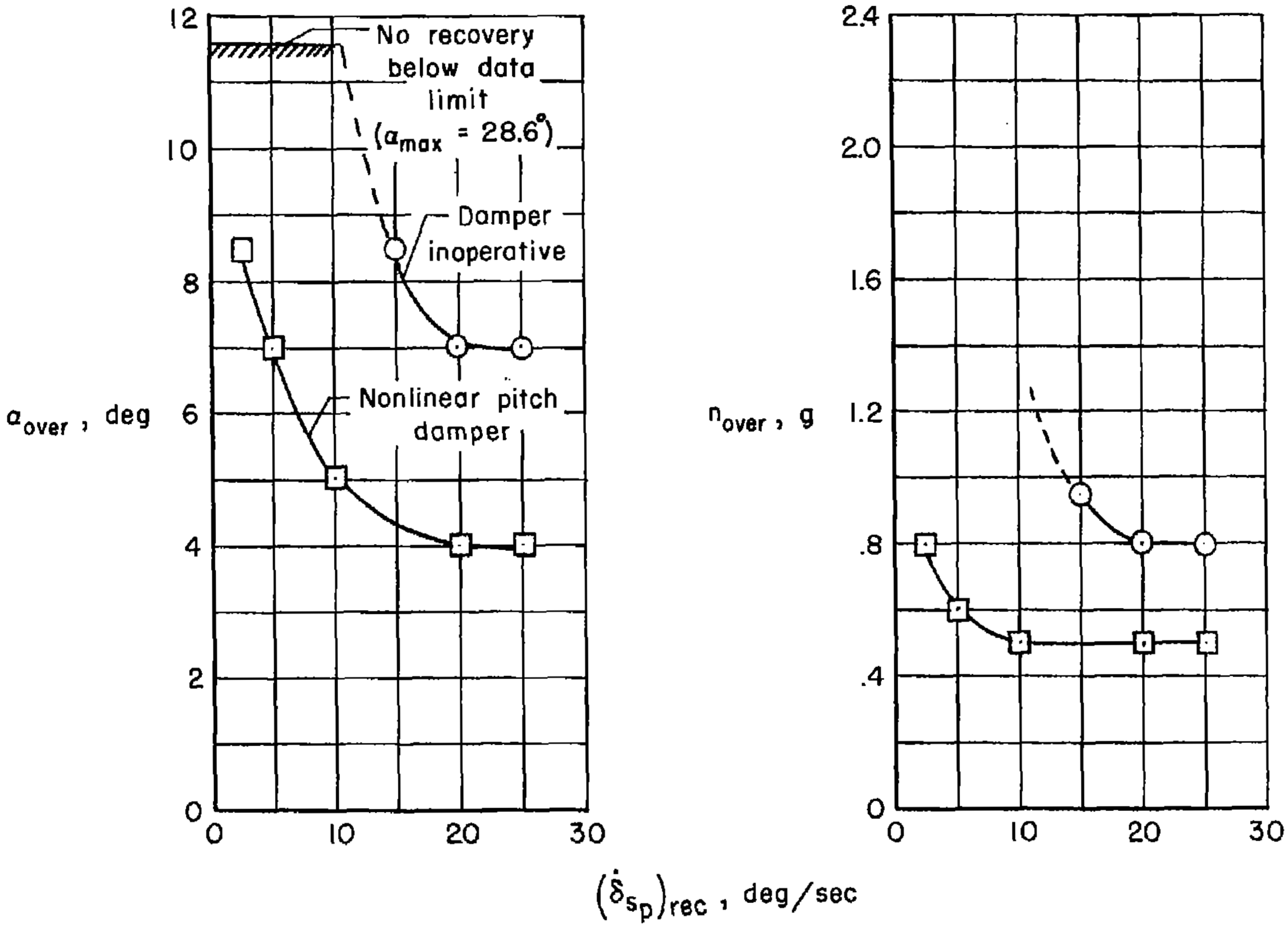
Figure 12.- Continued.



(c) $M = 0.23$ at h_p = sea level; initial $\dot{\delta}_{s_p}$ programmed to provide gradual low-speed stall entry.

Figure 12.- Concluded.

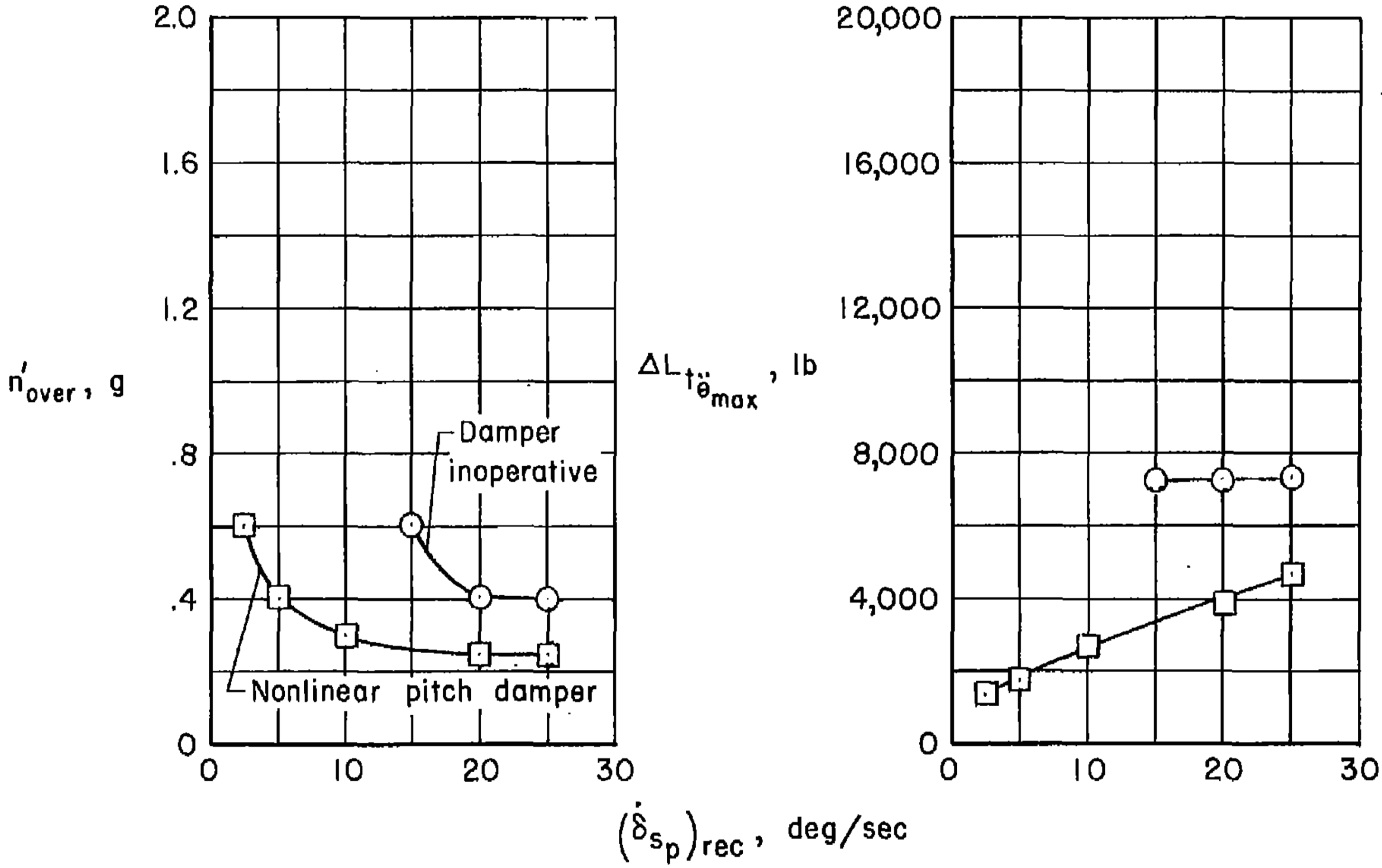
[redacted]



(a) α_{over} and n_{over}

Figure 13.- Variation of several response quantities with recovery-control rates; airplane B; $M = 0.80$; $h_p = 35{,}000$ feet.

[redacted]

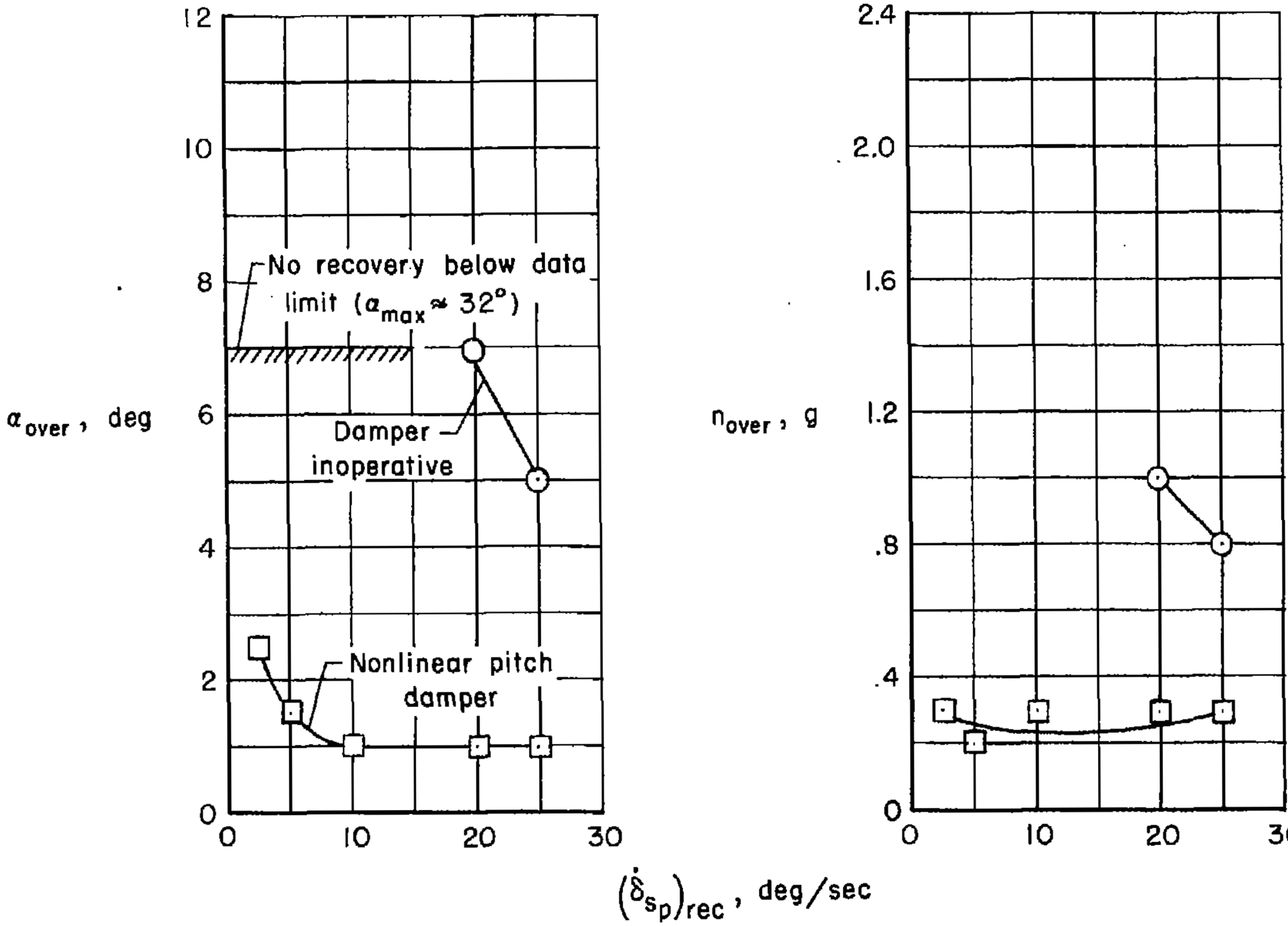


(b) n'_{over} and $\Delta L_{t\ddot{\theta}_{max}}$

Figure 13.- Concluded.

[redacted]

[redacted]

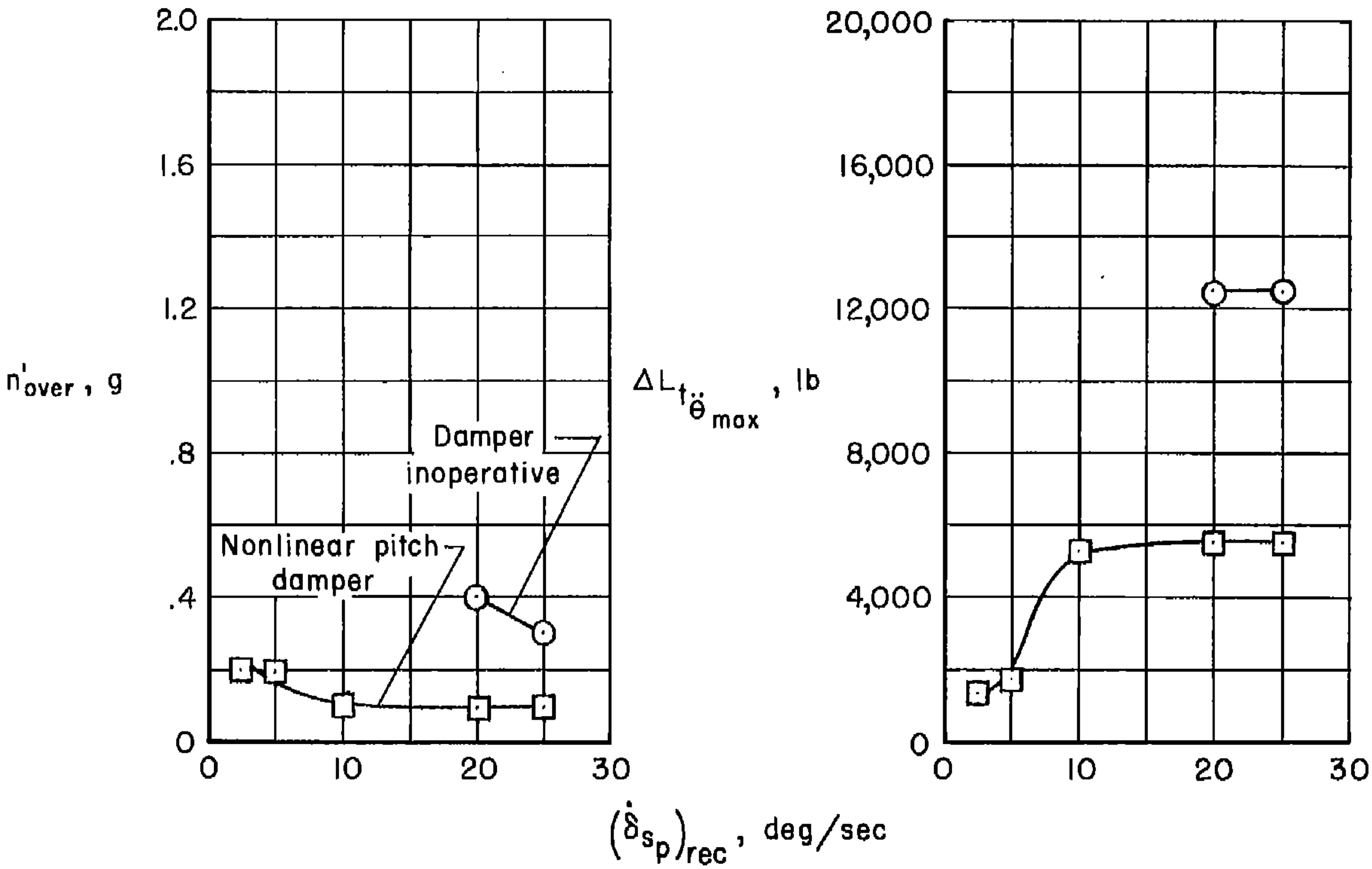


(a) α_{over} and n_{over}

Figure 14.- Variation with recovery-control rate of several response quantities; airplane B; $M = 0.975$, $h_p = 35{,}000$ feet.

[redacted]

[redacted]



(b) n'_{over} and $\Delta L_{t_{\ddot{\theta}_{max}}}$

Figure 14.- Concluded.

[redacted]

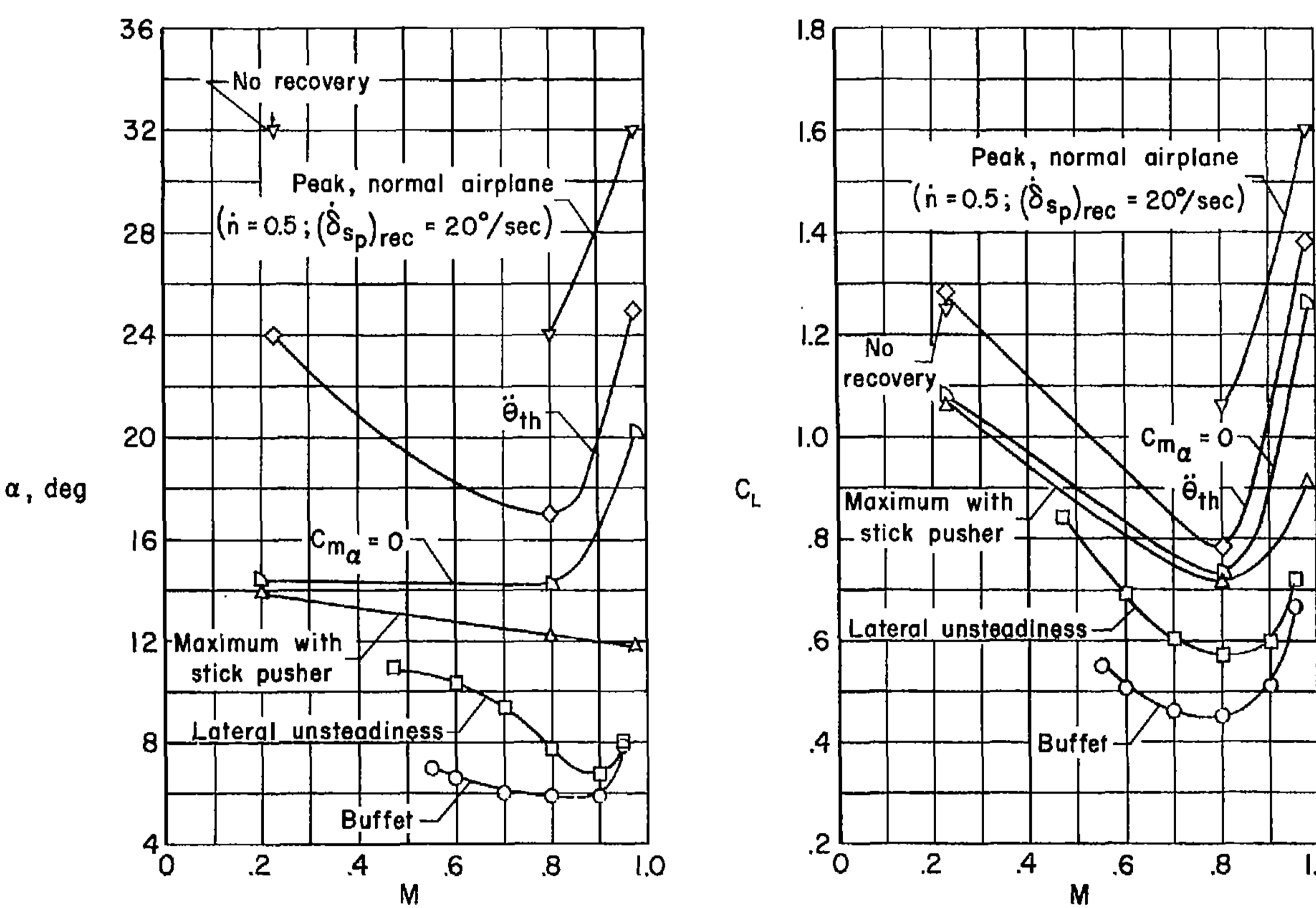


Figure 15.- Maximum angles of attack and lift coefficients attainable for unimpeded stick-pusher operation and their relationship to pitch-up warning and pitch-up boundaries; h_p = 35,000 feet.

[redacted]

[redacted]

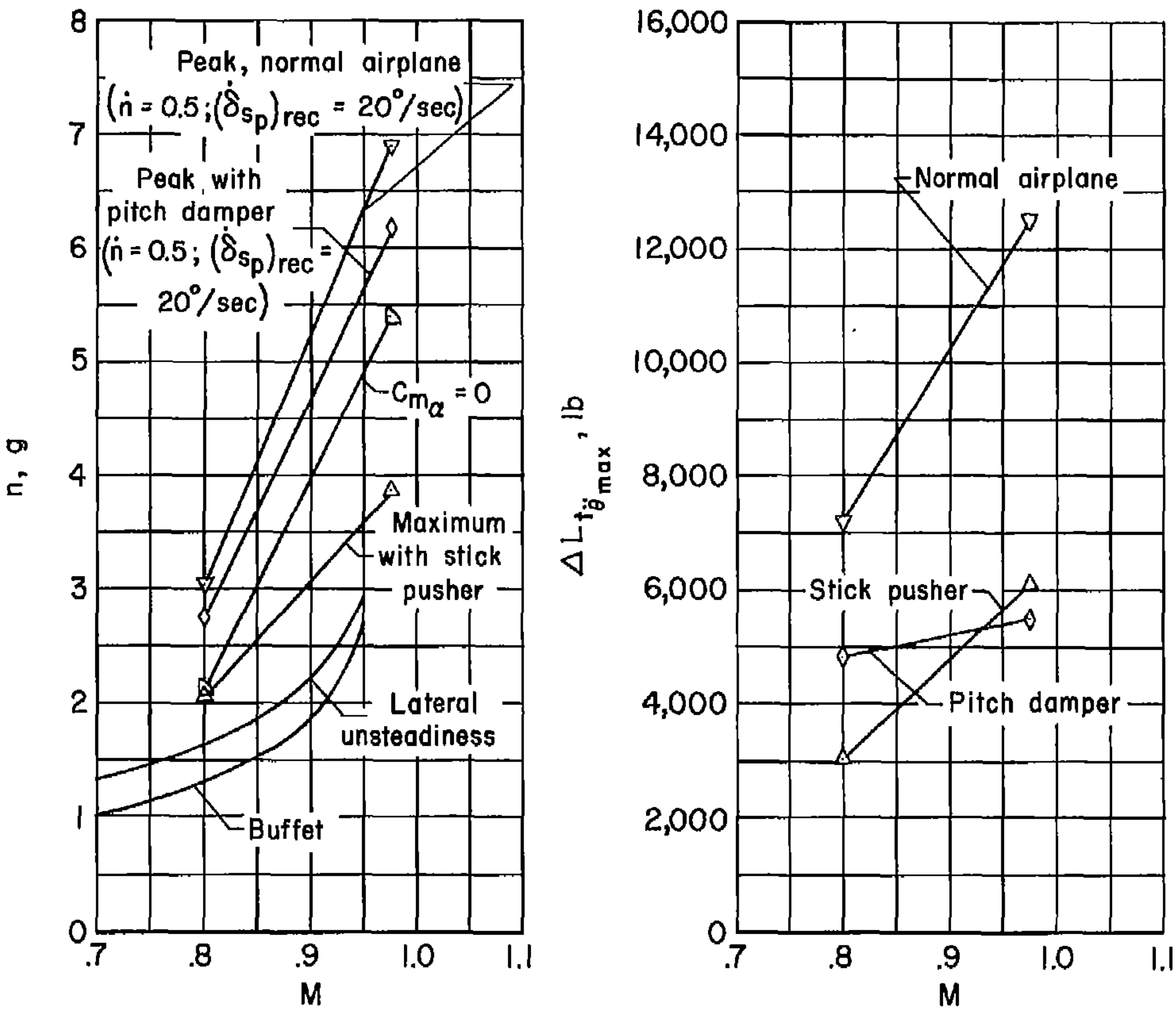


Figure 16.- Maximum load factors and maneuvering tail-load increments attainable for unimpeded stick-pusher operation and comparison with results for airplane with and without pitch damper; h_p = 35,000 feet.

[redacted]

[redacted]

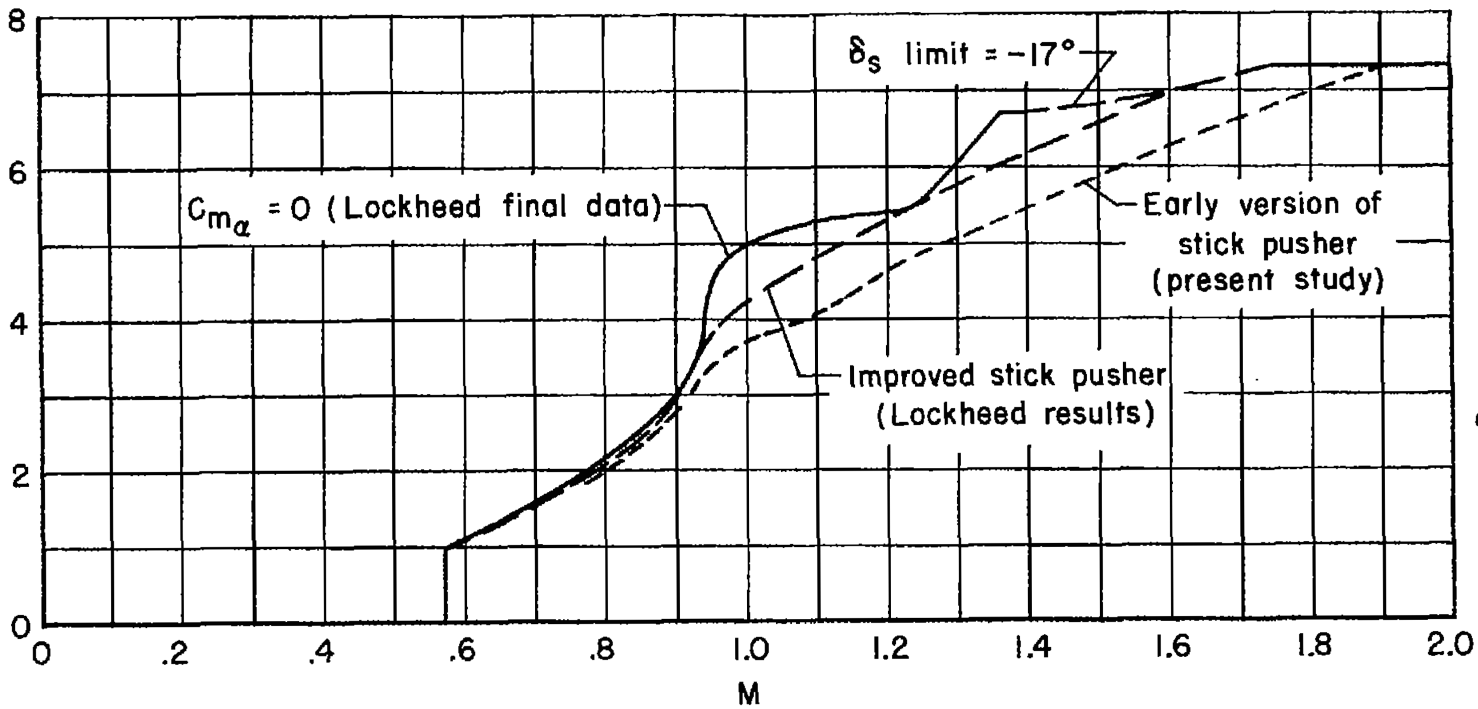
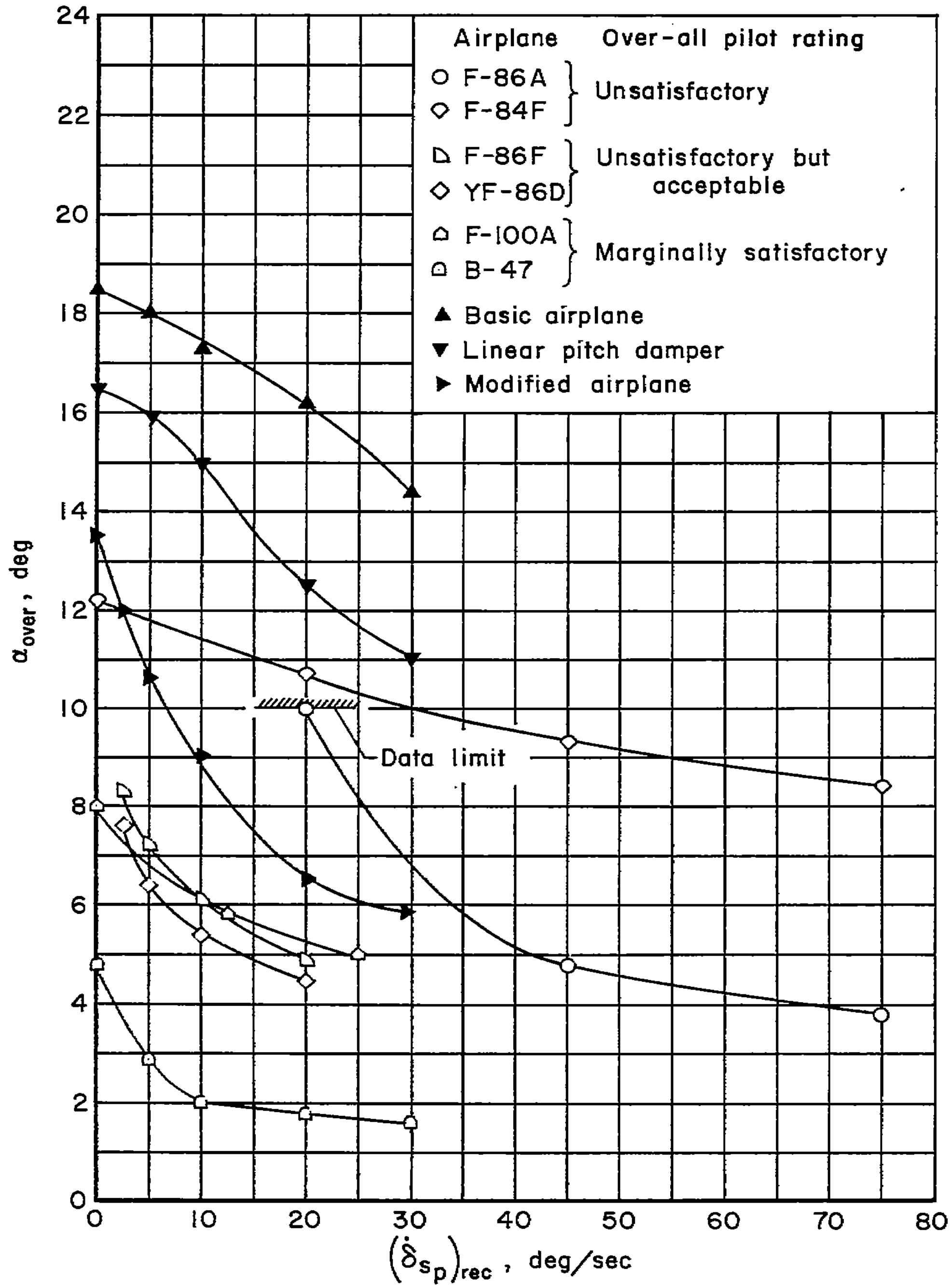


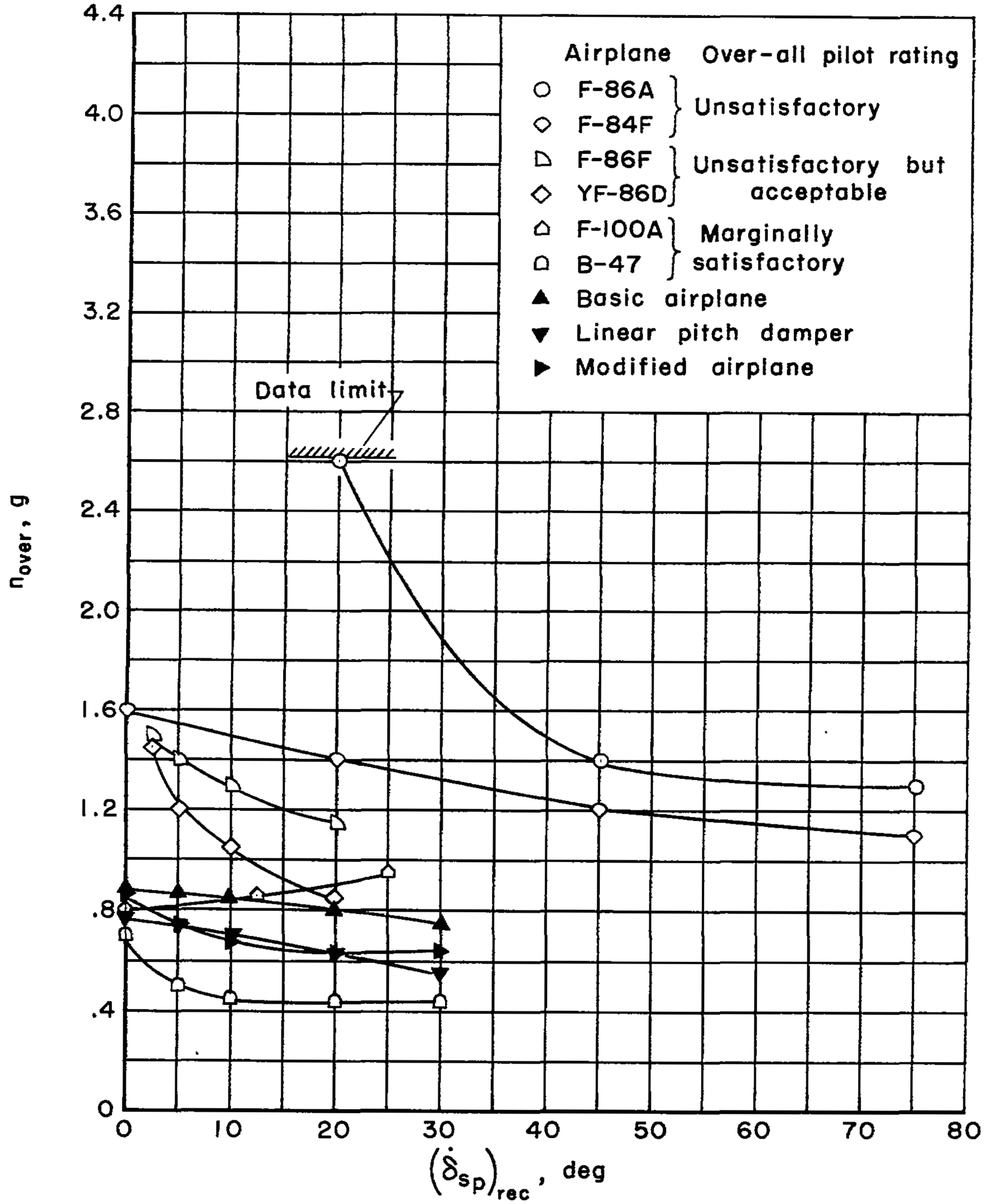
Figure 17.- Maneuvering capabilities of airplane B; h_p = 35,000 feet.

[redacted]



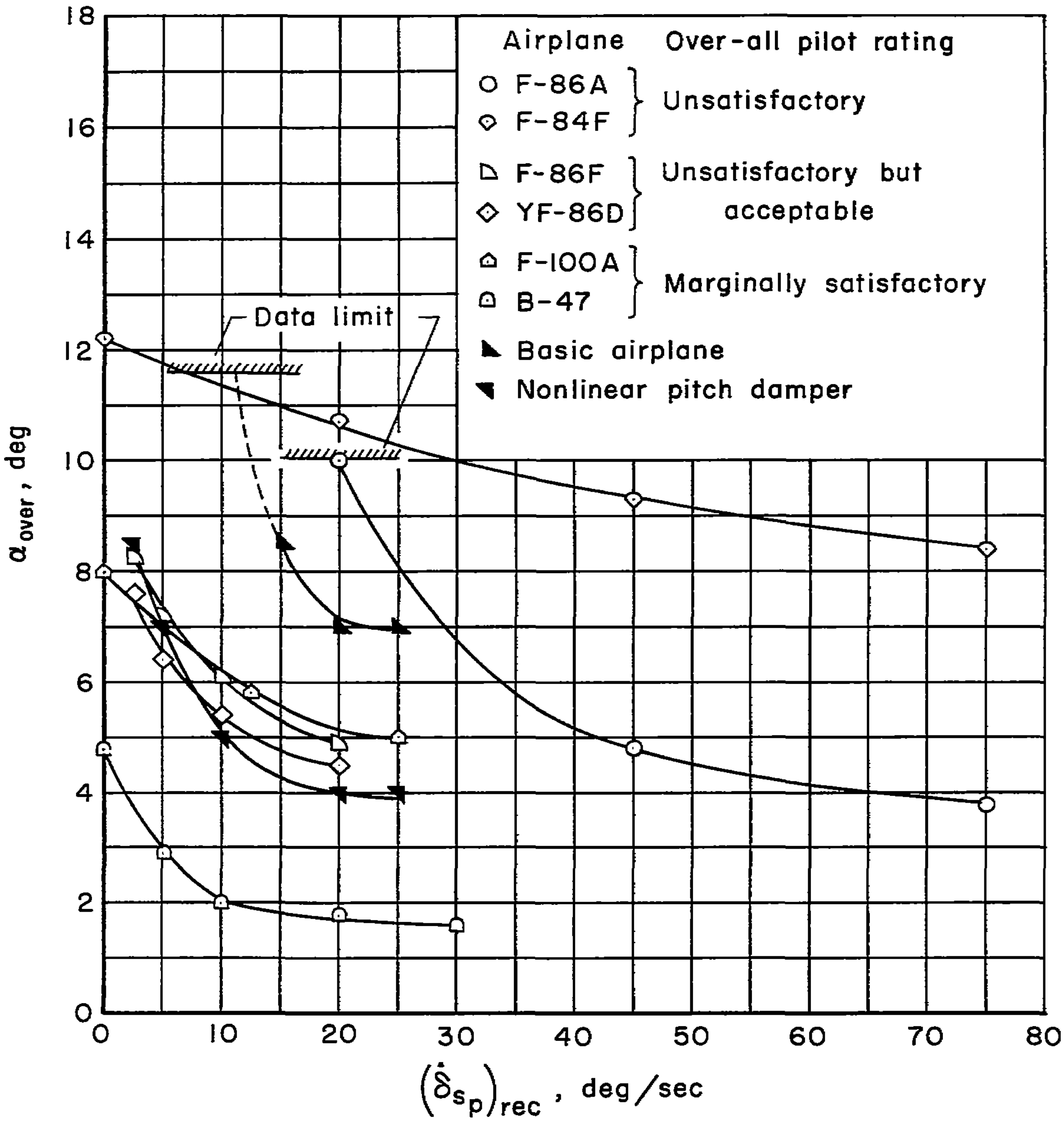
(a) α_{over}

Figure 18.- Comparison of the angle-of-attack and load-factor overshoots for airplane A with results for the six reference airplanes.



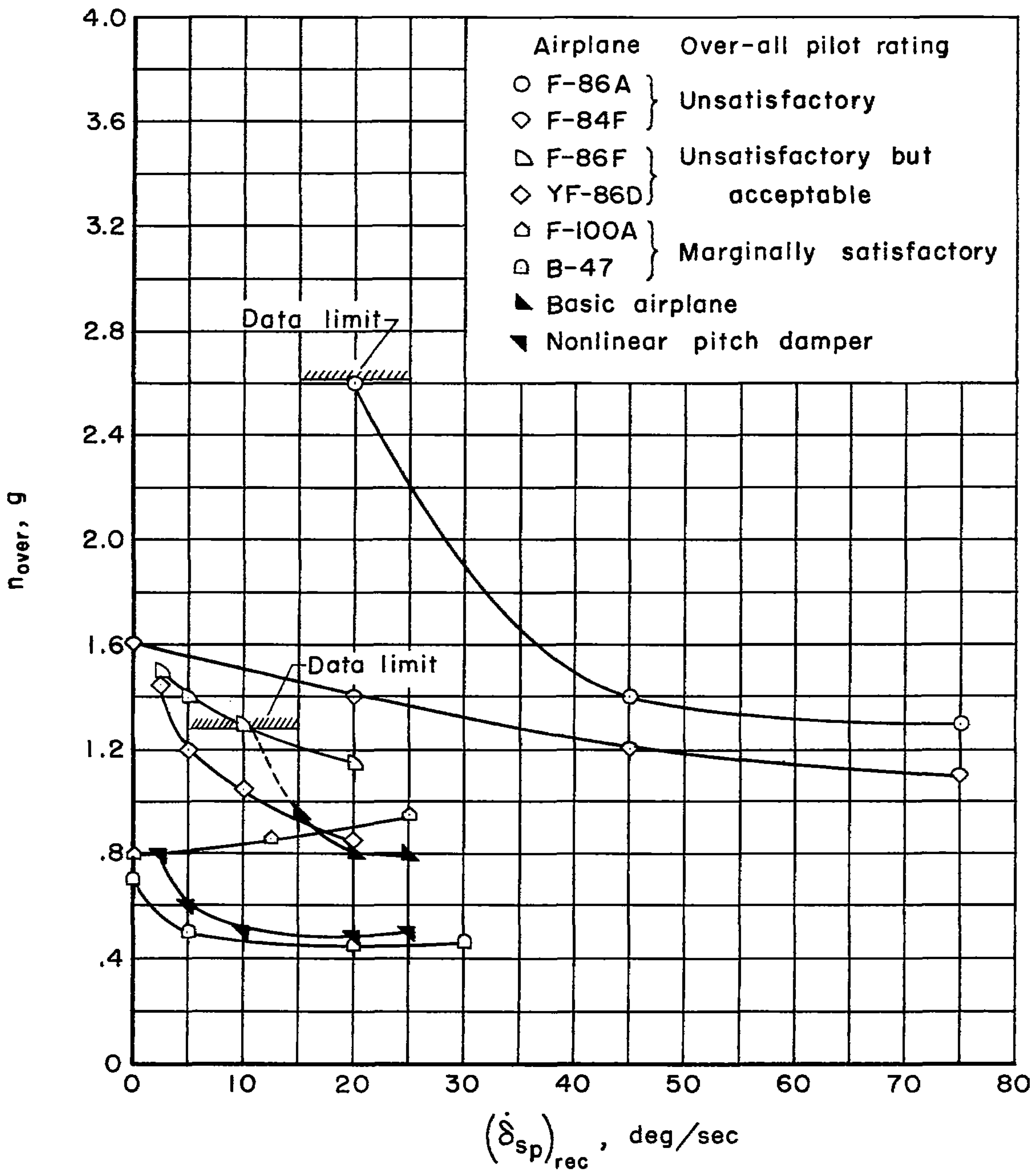
(b) n_{over}

Figure 18.- Concluded.



(a) α_{over}

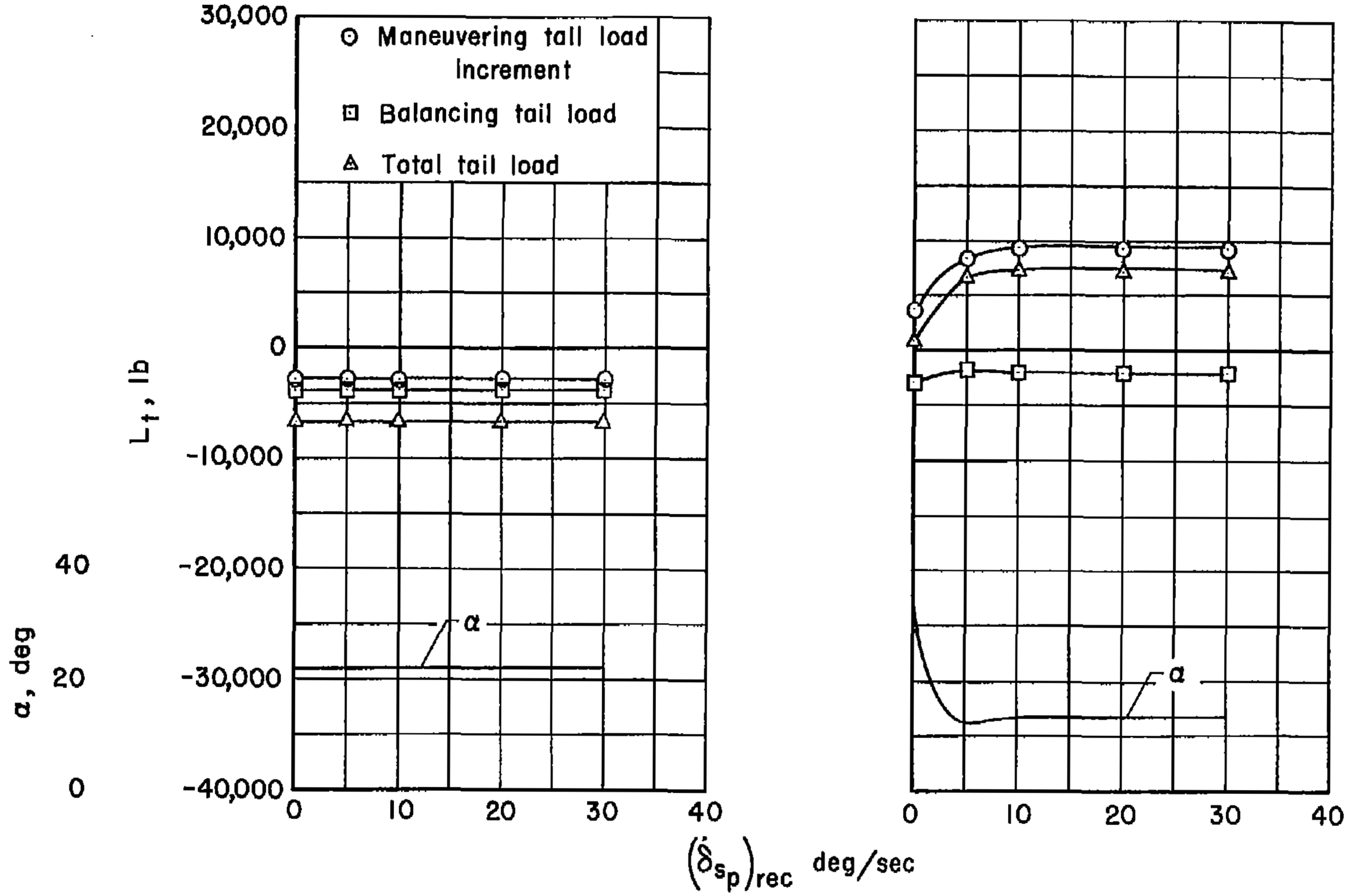
Figure 19.- Comparison of the angle-of-attack and load-factor overshoots for airplane B with results for the six reference airplanes.



(b) n_{over}

Figure 19.- Concluded.

[redacted]



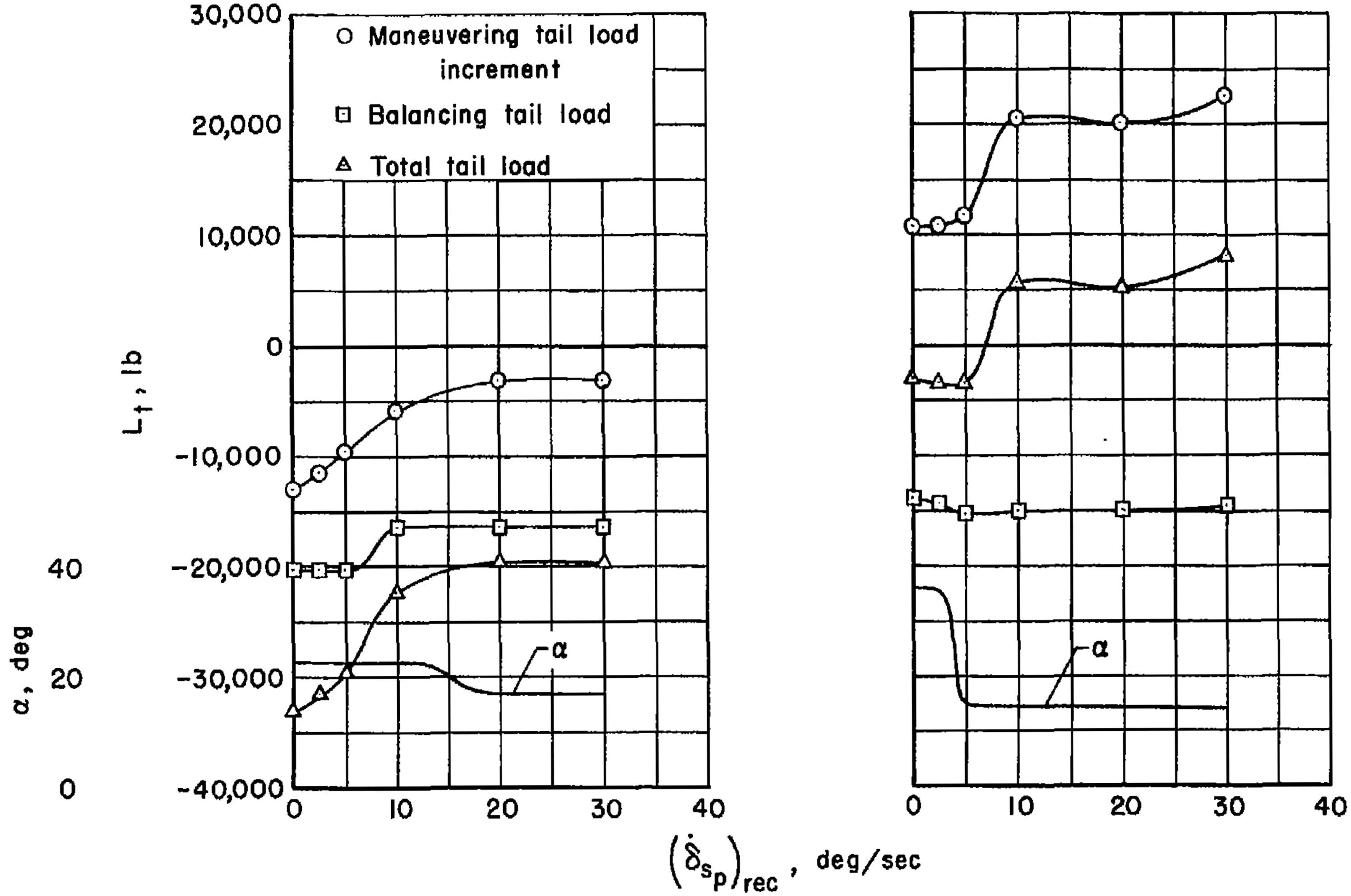
(a) First peak, M = 0.85

(b) Second peak, M = 0.85

Figure 20.- Variation with recovery-control rate of peak horizontal-tail loads and the angle of attack at which these loads occur; airplane A; h_p = 35,000 feet.

[redacted]

[redacted]



(c) First peak, M = 1.20

(d) Second peak, M = 1.20

Figure 20.- Concluded.

[redacted]

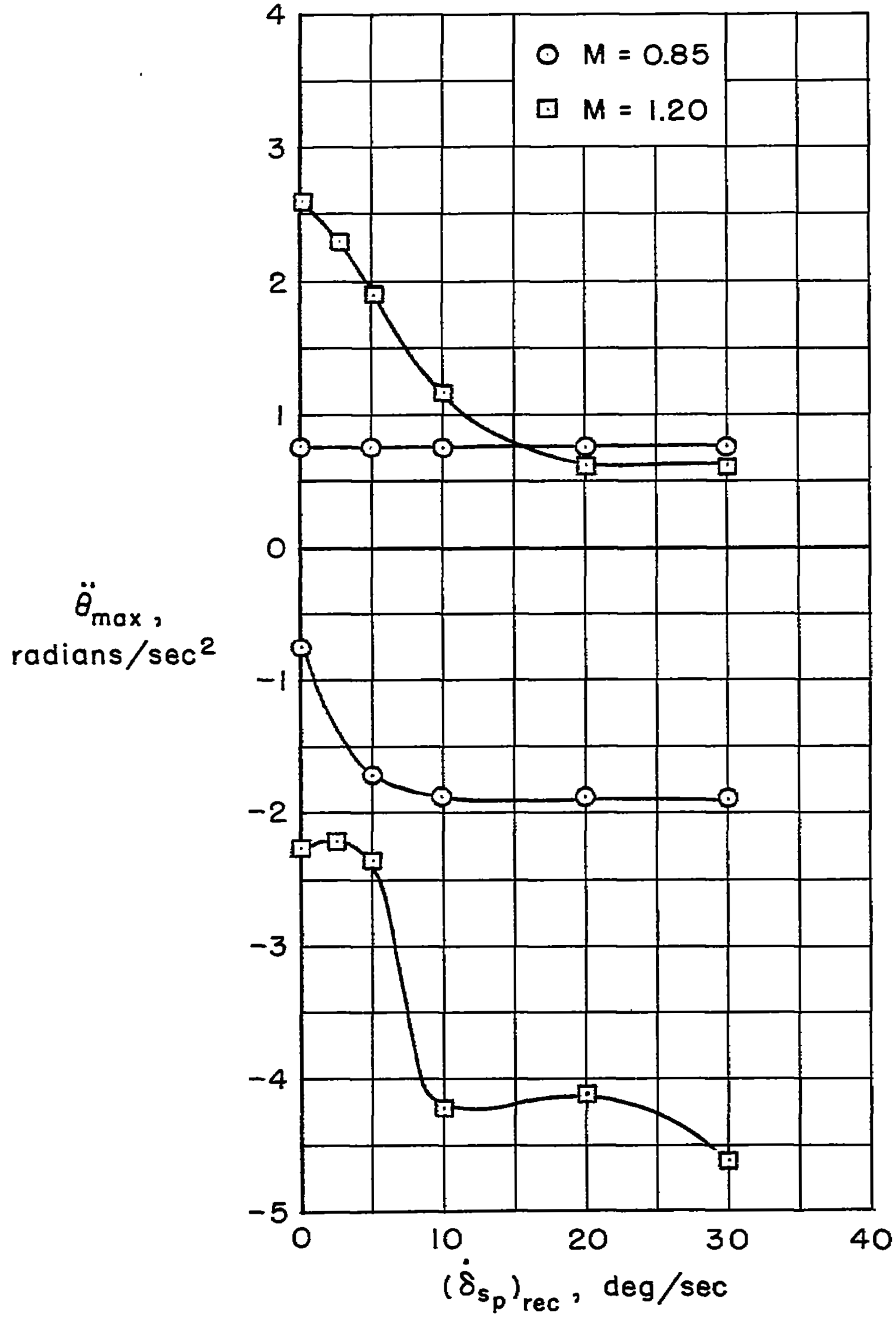
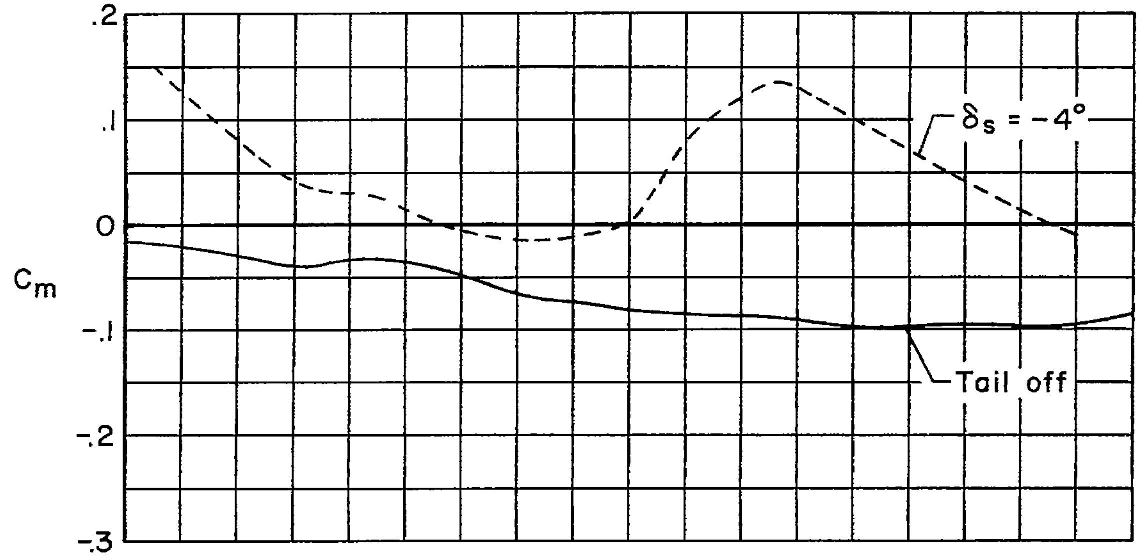
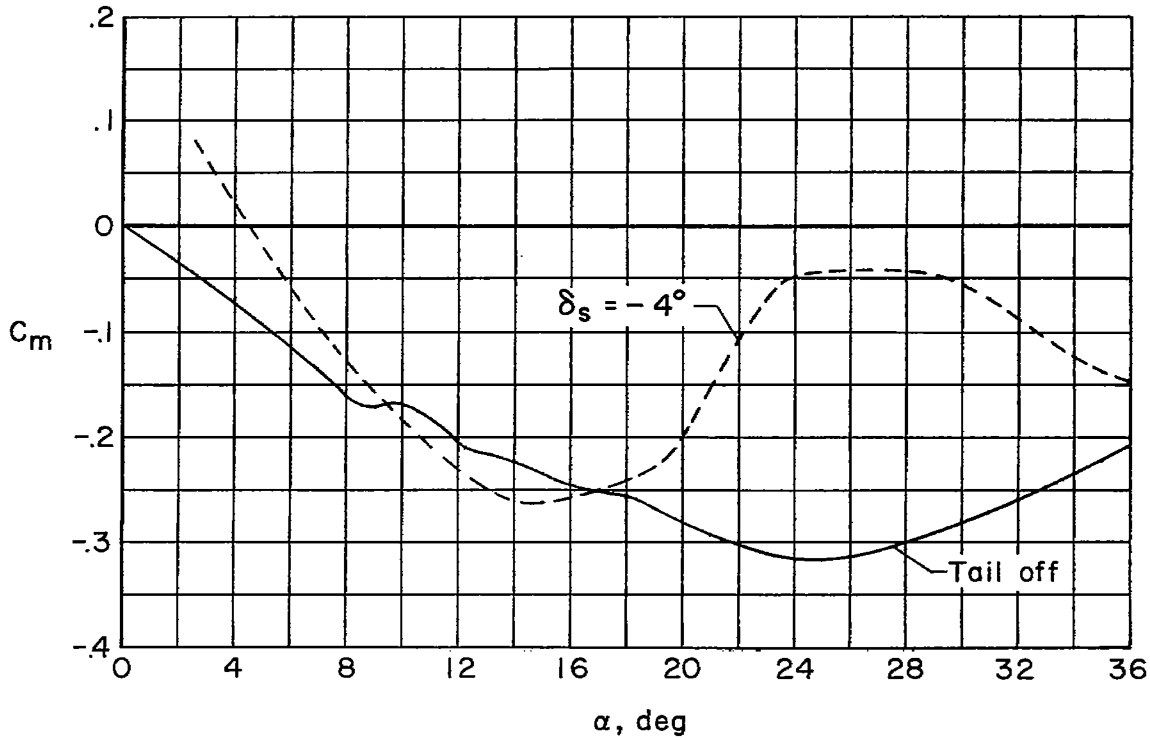


Figure 21.- Variation with recovery-control rate of the maximum positive and negative pitching accelerations; airplane A, h_p = 35,000 feet.



(a) M = 0.85



(b) M = 1.20

Figure 22.- Wind-tunnel measurements of the variation of pitching moment with angle of attack; airplane A.

[redacted]

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