N 62 70688

NASA TN D-114

NASA TN D-114

IN-08
381837

TECHNICAL NOTE
D-114

FLIGHT INVESTIGATION OF AN AUTOMATIC PITCHUP CONTROL

By George J. Hurt, Jr., and James B. Whitten

Langley Research Center
Langley Field, Va.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON August 1960

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

TECHNICAL NOTE D-114

FLIGHT INVESTIGATION OF AN AUTOMATIC PITCHUP CONTROL

By George J. Hurt, Jr., and James B. Whitten

L
6
7
9

SUMMARY

A flight investigation of an automatic pitchup control has been conducted by the National Aeronautics and Space Administration at the Langley Research Center. The pitching-moment characteristics of a transonic fighter airplane which was subject to pitchup were altered by driving the stabilizer in accordance with a signal that was a function of a combination of the measured angle of attack and the pitching velocity. An angle-of-attack threshold control was used to preset the angle of attack at which the automatic pitchup-control system would begin to drive the stabilizer. No threshold control as such existed for the pitching-velocity signal. A summing linkage in series with the pilot's longitudinal control allowed the automatic pitchup-control system to drive the stabilizer 13.5 percent of the total stabilizer travel independently of the pilot's control.

Tests were made at an altitude of 35,000 feet over a Mach number range of 0.80 to 0.90. Various gearings between the control and the sensing devices were investigated. The automatic system was capable of extending the region of positive stability for the test airplane to angles of attack above the basic-airplane pitchup threshold angle of attack. In most cases a limit-cycle oscillation about the airplane pitch axis occurred.

INTRODUCTION

A number of airplanes are limited in their range of maneuverability because of sharp decreases in longitudinal stability at high lift coefficients. The decrease in stability frequently results in pitchup with an accompanying loss of control by the pilot. In many instances the pitchup is of such a violent nature as to overstress the airplane to such an extent that major structural damage is sustained. The pitching motion associated with pitchup is usually so rapid that the pilot is unable to initiate proper preventive action.

The primary causes of pitchup are those associated with an unstable break in the wing pitching moment with increasing lift and those involving a loss of stabilizer effectiveness. The airplane used for the tests presented in this report normally pitched up because of an unstable break in the wing pitching moment with increasing lift. The stabilizer effectiveness of the test airplane remains relatively unchanged throughout the critical Mach number range (0.81 to 0.90, ref. 1).

Several methods of eliminating or alleviating pitchup have been
investigated. From the methods tested, several types of pitchup L
inhibitors have evolved. Aerodynamic fixes such as wing fences (ref. 2) 6
and vortex generators (ref. 3) have been used to raise the pitchup 7
boundary by preventing or delaying boundary-layer separation. Where 9
attempts to alleviate pitchup by aerodynamic fixes have met with limited success, a stick pusher (ref. 4) has been used in an attempt to prevent the pilot from entering the pitchup region. In contrast, the device used in the tests reported in this paper was intended as a means of extending the usable angle-of-attack range into the region of aerodynamic instability.

A simulator study (analog computer), unpublished, indicated that the pitchup of the test airplane could be alleviated by operating the stabilizer as a nonlinear function of angle of attack so as to offset the unstable break in the wing-pitching-moment curve of the basic airplane.

An automatic pitchup-control system was designed and incorporated in the test airplane. The system operated the horizontal stabilizer in a manner so as to counteract the destablizing effects which occurred with increasing angles of attack at high subsonic Mach numbers. The stabilizer was moved in accordance with a signal that was a function of a measured angle of attack and/or pitching velocity. Various gearings between the stabilizer and the sensing sources were investigated. This paper presents the results of the flight tests conducted and a discussion evaluating the automatic pitchup-control system investigated.

SYMBOLS

$\bar{c}$ mean aerodynamic chord, ft

C_m airplane pitching-moment coefficient about airplane center of gravity, $M_{cg}/qS\bar{c}$

C_m' apparent (to the pilot) pitching-moment coefficient

$C_m(\alpha)$	C_m as a function of angle of attack
C_{m_δ}	variation of C_m with stabilizer position, per deg
$C_{m_{\dot{\theta}}}$	variation of C_m with pitching velocity, per deg per sec
g	acceleration due to gravity, ft/sec^2
I_Y	moment of inertia, $slug\text{-}ft^2$
K_α	angle-of-attack gain, deg δ_e/deg α
$K_{\dot{\theta}}$	pitching-velocity gain, $\frac{\text{deg } \delta_e}{\text{deg/sec}}$
M	Mach number
M_{cg}	pitching moment about airplane center of gravity, ft-lb
$M(\alpha)$	airplane pitching moment as a function of angle of attack, ft-lb
$M_{\dot{\theta}}$	variation of airplane pitching moment with pitching velocity, $\frac{\text{ft-lb}}{\text{deg/sec}}$
M_δ	variation of airplane pitching moment with stabilizer position, ft-lb/deg
n	normal acceleration, g units
q	dynamic pressure, lb/sq ft
S	wing area, sq ft
α_T	angle-of-attack threshold for automatic pitchup alleviator, deg
α	vane-measured angle of attack, measured with respect to X-axis of airplane, deg
$\dot{\alpha}$	rate of change of angle of attack, radians/sec
β	angle of sideslip, deg

δ_e	stabilizer deflection, deg
$\dot{\delta}_e$	rate of change of stabilizer deflection, deg/sec
$\Delta\delta_{e,p}$	stabilizer deflection due to pilot's stick deflection, deg
$\delta_{e,PA}$	automatic pitchup-control stabilizer input, deg
$\dot{\theta}$	pitching velocity, radians/sec
$\ddot{\theta}$	pitching acceleration, radians/sec^2

L
6
7
9

TEST APPARATUS

Airplane

A single-engine, turbojet, low-wing, fighter airplane was used for these tests. (See figs. 1 and 2.) The wing had 35° sweepback along the 25-percent-chord line. The wing was equipped with automatic leading-edge slats which were spring loaded in the open position. The slats were normally closed by aerodynamic forces above an indicated airspeed of 180 to 190 knots in straight flight. The all-movable horizontal stabilizer was operated by an irreversible hydraulic actuator. The flow-stroke characteristics of the actuator were nonlinear (fig. 3). The stabilizer had 35° sweepback along the 25-percent-chord line.

The test airplane was equipped with a variable-slope stabilizer followup system. The followup was designed to minimize overcontrol during high-speed operation, yet allow adequate stabilizer movement with the allowable stick travel during slow-speed flight. The stabilizer settings required for the Mach number range tested for this report were within the linear range of the variable-slope followup.

Table I presents additional airplane specifications.

Instrumentation

Standard NASA instruments were installed in the test airplane to record airspeed, altitude, linear accelerations at the center of gravity, control positions, angle of attack, and angular velocities and accelerations. All recording instruments except the airspeed-altitude recorder were mounted in a modified rocket package. The internal rocket package (ground extendible for instrument service) afforded a nearly ideal location of the instruments in relation to the airplane center of gravity. The rigidity of the package structure and the location of the package were beneficial in minimizing the effects of vibration on the recording instruments during the buffet phase of the approach to the pitchup boundary. All instruments were mounted at 0° to the reference axes of the airplane.

Automatic Pitchup-Control System

A block diagram of the final configuration of the automatic pitchup-control system is shown in figure 4. The stabilizer was deflected in accordance with a signal which was a function of a combination of the measured angle of attack and the pitching velocity. An amplifier unit received the angle-of-attack and pitching-velocity signals and relayed them in the desired proportions to the automatic-system actuator. A bias signal was used to hold the actuator in the retracted (rearward) position so that the system was inoperative during minimum signal conditions. A threshold control was used to preset the conditions at which the automatic actuator would begin to drive. The threshold was preset so as to commence driving the actuator at or near the angle of attack at which the unstable break occurred in the wing pitching moment.

Two angle-of-attack vanes, one on each side of the fuselage, were used during the initial flight tests. The vanes were mounted at 0° incidence to the fuselage reference line. Figure 5 shows the vane position. Inasmuch as inspection of the records from the initial flight tests revealed that there was no appreciable difference due to sideslip ($\beta = \pm 10^{\circ}$) in the recordings of the two vanes (maximum difference, 0.25°), the left vane was discarded for the remaining flights. A position synchro was used with the angle-of-attack vane to generate a signal to the automatic pitchup-control system. Maximum gain K_{α} for the angle-of-attack signal was $0.98^{\circ}/\text{deg}$.

A rate gyro was used to measure and generate a signal proportional to the pitching velocity. Figure 6 shows the calibration of the rate gyro for various $K_{\dot{\theta}}$ settings. The data for figure 6 were obtained by mounting the rate gyro on a remote turntable and recording the stabilizer displacement for a given turntable rpm and cockpit dial setting. The angle-of-attack signal source was disconnected during the calibration of the rate gyro. Tests discussed in this paper were made with $K_{\dot{\theta}}$ set at the maximum position.

Either of the signal sources was capable of generating a signal of sufficient magnitude as to drive the automatic actuator from neutral to its maximum extended position.

Figure 7 shows a schematic diagram of the modified longitudinal control system. An electric-motor-driven actuator was linked in series with the pilot's longitudinal control. The series-summing linkage was designed so as to maintain the same stick-to-stabilizer static-deflection characteristics as the original system. The summing linkage allowed the automatic pitchup-control system to drive the stabilizer 13.5 percent of the total stabilizer travel independently of the pilot's control. The automatic system was capable of producing a stabilizer change of 3.1° in 0.3 second.

The stabilizer-actuator valve centering spring was modified from a 96-lb/in. spring to a 12.5-lb/in. spring. The original 3-pound breakout force of the valve and spring was maintained. A reduction in spring gradient was found to be necessary in order to allow the pitchup-control-system actuator to cause movement of the stabilizer actuator at the desired rate. The nonlinear flow-stroke characteristics of the stabilizer actuator (fig. 3) required nearly maximum valve displacement from the automatic-system actuator in order to obtain the required stabilizer displacement within the desired time.

L
6
7
9

TEST PROCEDURE

A windup turn was used as the approach to the pitchup boundary. This type of approach afforded some measure of control over the altitude range covered, the maximum normal acceleration attained, and the Mach number variation during the maneuver. The entry rate during the approach varied from 0.07 g/sec to 0.4 g/sec. The test Mach number range varied from 0.80 to 0.90. All tests recorded were from windup turns to the left.

The pitchup threshold for the particular airplane used for these tests was usually preceded by a high-frequency buffet which supplied ample warning to the pilot that he was approaching a dangerous attitude. If the airplane was allowed to remain in the buffet region, the Mach number would fall off very rapidly and no pitchup would occur. Unless otherwise stated, the afterburner was used to aid in maintaining airspeed. In order to obtain the test results presented in this paper, the test airplane was intentionally pulled into a pitchup maneuver.

Rapid roll-out as a result of the high angle of attack and stall was frequently experienced after the pitchup. From a windup turn to the left, the basic airplane would usually conclude the pitchup maneuver with a rapid roll to the right. The magnitude of the resulting rolling velocity would exceed the capabilities of the airplane roll control. No attempt was made to relieve the roll condition. Very little, if any, roll-control correction was required during the approach to the pitchup or during the first stages of the pitchup.

The pilot reported that there was no visible opening of the wing leading-edge slats during the tests.

In order to insure that the test airplane would not be inadvertently overstressed during the tests, all tests were made in the vicinity of 35,000 feet. At that altitude, the maximum normal acceleration attainable for the test airplane was 4.8g at $M = 1.0$. Maximum allowable load factor for the airplane in test configuration was 5.6g.

L679

RESULTS AND DISCUSSION

Basic Airplane

The airplane used for the tests discussed in this report pitched up because of an unstable break in the wing pitching moment with increasing lift (ref. 1). Data from the initial flight tests of the basic airplane placed the unstable break at an angle of attack of about 6° for the critical Mach number range of 0.81 to 0.90. Various combinations of entry rate and Mach number caused a variation of only $\pm1.0^{\circ}$ in the angle-of-attack pitchup threshold. Above or below the critical Mach number range, the test airplane was stable.

A typical time history of a pitchup of the basic airplane is shown in figure 8. The pitchup threshold was preceded by a high-frequency buffet. During this buffet phase and the following pitchup maneuver, the Mach number would fall off very rapidly unless maximum thrust (afterburner) was used. For this flight test the afterburner was not used.

Three types of longitudinal control motions were briefly investigated. In the first case, the pilot attempted to control the severity of the pitchup; for the second case, the pilot held the stick fixed when the pitchup occurred; and for the third case, the pilot made a continuous steady rearward motion of the control stick throughout the flight test. A comparison of the flight tests for each of the three control procedures revealed that there was no appreciable change in airplane response once the pitchup began. The worst cases were those in which the pilot attempted to control the pitchup. A 2° to 5° higher angle of attack occurred during the pitchup, and two to three more oscillations ensued before the pilot completed the recovery from the pitchup.

Angle-of-Attack Signal

The initial tests of the automatic pitchup-control system were made with an angle-of-attack measuring vane as the only signal source.

A typical maneuver with the system in operation is shown in figure 9. The automatic control settings for this test were $\alpha_T = 6^\circ$ and $K_\alpha = 0.62^\circ/\text{deg}$. A pitching oscillation began when the angle of attack approached the pitchup boundary and continued until the angle of attack was reduced to a value well below the pitchup boundary.

Above or below the critical Mach number range of 0.81 to 0.90, the test airplane had sufficient positive stability in pitch to prevent a pitchup. As may be seen in figure 9, at the time the airplane
approached the pitchup boundary the Mach number was slightly below the L
critical range and the airplane should have been stable in pitch. How- 6
ever, additional analysis of the data presented in figure 9 revealed 7
that there was a lag between the measured angle of attack and the 9
resulting stabilizer response. During the short-period oscillation, the phase lag between these quantities was 55°. With this condition imposed on the system, the airplane was neutrally stable in pitch and the oscillation did not cease until the pilot commanded an angle of attack which was below the α_T value preset into the automatic system.

Several combinations of α_T and K_α were tested. None of the combinations tested gave a satisfactory response.

Angle-of-Attack Signal Plus Pitching-Velocity Signal

When an angle-of-attack signal alone was used, the lags in the automatic control system and the stabilizer actuator resulted in a pitching-moment component in phase with $\dot{\alpha}$. By using $\dot{\alpha}$ as a negative feedback signal, the stability of the system could have been improved. Since it was more convenient to measure $\dot{\theta}$, and since at the frequencies at which the oscillation occurred $\dot{\alpha} \approx \dot{\theta}$, a pitching-velocity signal was used to compensate the lags in the system. The addition of the pitching-velocity signal effectively gave the automatic system an anticipation of the impending pitchup.

A typical maneuver made with both signal sources in operation is presented in figure 10. A comparison of figures 9 and 10 illustrates the lag improvement made by the addition of the pitching-velocity signal. The lag has been reduced to less than 10°. The pilot has approached a 10° angle of attack without encountering pitchup. Control settings of the automatic pitchup-control system for the test presented in figure 10 were as follows: $\alpha_T = 8^\circ$, $K_\alpha = 0.62^\circ/\text{deg}$, and $K_{\dot{\theta}}$ = Maximum.

A comparison of figures 10 and 11 indicates the effect of a change in the angle-of-attack gain. For figure 11, K_α was increased from

0.62°/deg to 0.98°/deg. (Again, $\alpha_T = 8^\circ$ and $K_{\dot{\theta}}$ = Maximum.) The frequency of the oscillation is approximately 31 percent lower for a 58 percent increase in K_α. The amplitude of the oscillation is nearly twice as large and the lag has increased.

The 10° residual lag indicated in figure 10 and the increase in lag shown in figure 11 might be attributed to the dead spot in the stabilizer-actuator valve. Figure 3, a plot of stabilizer rate against valve displacement, shows the valve overlap. The valve overlap was an intentional design feature of the basic airplane to minimize control hunting and chatter. Figures 3, 10, and 11 show that when the amplitude of the airplane oscillation is small, the effective stabilizer rate will be negligible and the phase lag will increase. The increase in lag will tend to build up the amplitude of the oscillation, while the pitching-velocity lead signal from the automatic system will attempt to reduce the amplitude. A limit-cycle oscillation will occur. An additional source of lag was the load imposed on the automatic-system actuator by the preload and friction in the stabilizer actuator valve.

Figure 12 is a representative time history of the flight tests made at automatic control settings of $\alpha_T = 10^\circ$, $K_\alpha = 0.45^\circ$/deg, and $K_{\dot{\theta}}$ = Maximum. These settings produced a response that the test pilot considered the optimum for the equipment. For these tests the pilot steadily increased the angle of attack until the basic-airplane (6°) pitchup threshold was exceeded. The airplane was then allowed to stabilize for 5 to 10 seconds with the stick held fixed. The pilot then continued the rearward stick motion to bring the airplane to a still higher angle of attack. The first stick position usually brought the airplane angle of attack to approximately 10°. Angles of attack of 18° to 22° were reached with the second stick motion. At angles of attack in the vicinity of 20°, the Mach number rapidly fell below the critical range and no pitchup could occur. The low-amplitude short-period oscillations usually persisted throughout the maneuver, but the pilot was able to control the airplane while in the critical Mach number range at angles of attack in the vicinity of 15° without encountering pitchup.

The short-period oscillation of the airplane was of such a frequency that the automatic pitchup-control drive was normally limited by the magnitude of the signal input to approximately 2° of stabilizer change. The 2° change was sufficient to stop the pitchup but was not fast enough nor in proper phase relation to damp out the oscillation once it had commenced.

The test pilot indicated that the short-period oscillation was preferable to the high-amplitude divergent motion of the pitchup. He felt that he had a considerably wider range of control over the modified airplane with the optimum automatic control settings than he had experienced during the initial flights with the basic airplane.

The apparent pitching moment as experienced by the test pilot may be seen in figure 13. The apparent pitching-moment coefficient was calculated as the sum of the basic pitching-moment coefficient plus the contribution of the automatic pitchup-control system. (See appendix.) There was, of course, no change in the basic pitching-moment characteristics of the test airplane by the addition of the automatic control system.

L
6
7
9

A favorable gain from the addition of the automatic pitchup-control system that was not readily apparent from the recorded data was the change in the roll-off occurring after the pitchup. The pilot did not consider the roll-off excessive for any test during which the automatic pitchup-control system was engaged. Apparently the action of the automatic system was sufficient to reduce the magnitude of the change in angle of attack so that the reduction in Mach number was gradual and no abrupt stall was encountered.

CONCLUDING REMARKS

The pitching-moment characteristics of a transonic fighter airplane which was subject to pitchup were altered by driving the stabilizer in accordance with a signal that was a function of a combination of the measured angle of attack and the pitching velocity. A summing linkage in series with the pilot's longitudinal control allowed the automatic pitchup-control system to drive the stabilizer 13.5 percent of the total stabilizer travel independently of the pilot's control.

Tests were made at an altitude of 35,000 feet over a Mach number range of 0.80 to 0.90. Various gearings between the control and the sensing devices were investigated. No attempt was made to alter the gains or threshold during a flight test.

The initial flight tests of the basic airplane indicated that the pilot was not able to prevent the pitchup if the angle-of-attack boundary of about 6° was exceeded; nor was he able to control the magnitude of the pitchup or the number of oscillations occurring during the pitchup.

The results of this investigation indicate the feasibility of avoiding pitchup by means of an automatic control system which operates the stabilizer as a function of angle of attack in a manner to offset the unstable pitching-moment variation of the basic airplane. The automatic system was capable of extending the region of positive stability for the test airplane to angles of attack of 15° to 20° which was considerably above the basic airplane pitchup threshold angle of attack of 6°.

Throughout the tests of the automatic control system, the results were characterized by a low-amplitude short-period oscillation. The oscillation resulted from the nonlinear flow-stroke characteristics of the stabilizer actuator valve combined with velocity limiting of the automatic-control-system actuator.

L
6 The test pilot was of the opinion that the short-period oscillation
7 was preferable to the high-amplitude divergent motion of the pitchup.
9 He felt that he had a considerably wider range of control over the modified airplane than he had experienced during the initial flights with the basic airplane.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., May 5, 1960.

APPENDIX

CALCULATION OF PITCHING MOMENT

The equation used to calculate basic values of pitching-moment coefficient for the test airplane was

$$I_Y\ddot{\theta} = M(\alpha) + M_{\dot{\theta}}\dot{\theta} + M_\delta \delta_e \tag{1}$$

L679

or

$$\frac{I_Y\ddot{\theta}}{qS\bar{c}} = C_m(\alpha) + C_{m_{\dot{\theta}}}\dot{\theta} + C_{m_\delta}\delta_e \tag{2}$$

By using instantaneous values of α, equation (2) becomes

$$C_m = \frac{I_Y\ddot{\theta}}{qS\bar{c}} - C_{m_{\dot{\theta}}}\dot{\theta} - C_{m_\delta}\delta_e \tag{3}$$

For the apparent (to the pilot) values of pitching-moment coefficient, the following equation was used:

$$C_m' = \frac{I_Y\ddot{\theta}}{qS\bar{c}} - C_{m_{\dot{\theta}}}\dot{\theta} - C_{m_\delta}\delta_e + C_{m_\delta}\delta_{e,PA} \tag{4}$$

The pitching-moment coefficient defined by equation (4) was a function of both angle of attack and pitching velocity. No attempt was made to separate $C_{m_\delta}\delta_{e,PA}$ into α and $\dot{\theta}$ components.

For the Mach number range covered in these tests, it was found that using a constant value of C_{m_δ} introduced a variation of only 0.4 percent in the C_m values calculated. For the test airplane at the maneuver entry Mach number of 0.86, C_{m_δ} was -0.022 per degree. A $C_{m_{\dot{\theta}}}$ of -0.0555 per degree per second was used.

REFERENCES

1. McFadden, Norman M., and Heinle, Donovan R.: Flight Investigation of the Effects of Horizontal-Tail Height, Moment of Inertia, and Control Effectiveness on the Pitch-Up Characteristics of a 35° Swept-Wing Fighter Airplane at High Subsonic Speeds. NACA RM A54F21, 1955.

2. Bray, Richard S.: The Effects of Fences on the High-Speed Longitudinal Stability of a Swept-Wing Airplane. NACA RM A53F23, 1953.

3. Miller, J. W.: Elimination of B-47 Pitch-Up by Means of Vortex Generators. Doc. No. D-12247 (Contract No. W33-038 ac-22413), Boeing Airplane Co., Sept. 26, 1951.

4. Sadoff, Melvin, and Stewart, John D.: 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. NACA RM A57K07, 1958.

L
6
7
9

TABLE I.- TEST-AIRPLANE SPECIFICATIONS

Wing:	
Total area (49.92 sq ft covered by fuselage), sq ft	287.90
Span, ft	37.12
M.A.C., ft	8.086
Aspect ratio	4.785
Incidence to fuselage reference line -	
Root chord, deg	1
Tip chord, deg	-1
Dihedral, deg	3.0
Sweepback, 25-percent-chord line, deg	35
Airfoil section -	
Root	NACA 0012-64 (modified)
Tip	NACA 0011-64 (modified)
Horizontal tail:	
Total area (14.89 sq ft covered by fuselage), sq ft	53.90
Span, ft	16.85
M.A.C., ft	3.47
Aspect ratio	5.1
Taper ratio	0.423
Dihedral of chord plane, deg	0
Airfoil section	NACA 64A010
Type	All-movable stabilizer
Root chord, ft	4.61
Travel -	
Leading edge up, deg	7
Leading edge down, deg	16
Tail length, ft	15.16
Sweepback, 25-percent-chord line, deg	35
Vertical tail:	
Area, sq ft	31.05
Span, ft	7.29
Aspect ratio	1.71
Airfoil section	NACA 0011-64 (modified)
Rudder area, sq ft	5.26
Travel, deg	±27.5
Fuselage:	
Length, ft	39.0
Depth (maximum), ft	5.71
Width (maximum), ft	5.0
Fineness ratio	7.05
Frontal area, sq ft	24.14
General:	
Turbojet engine with afterburner	GE J47
Maximum allowable load factor -	
Positive, g units	5.6
Negative, g units	2.0
Weight -	
Gross, lb	17,747
Empty, lb	13,067
Fuel -	
Internal tanks, gal	608
External tanks, gal	240
Moment of inertia, I_Y (without external tanks) -	
Empty, slug-ft^2	23,300
Full internal service (16,625 lb), slug-ft^2	29,775
Center of gravity (full internal service), percent $\bar{c}$	25.6

L-679

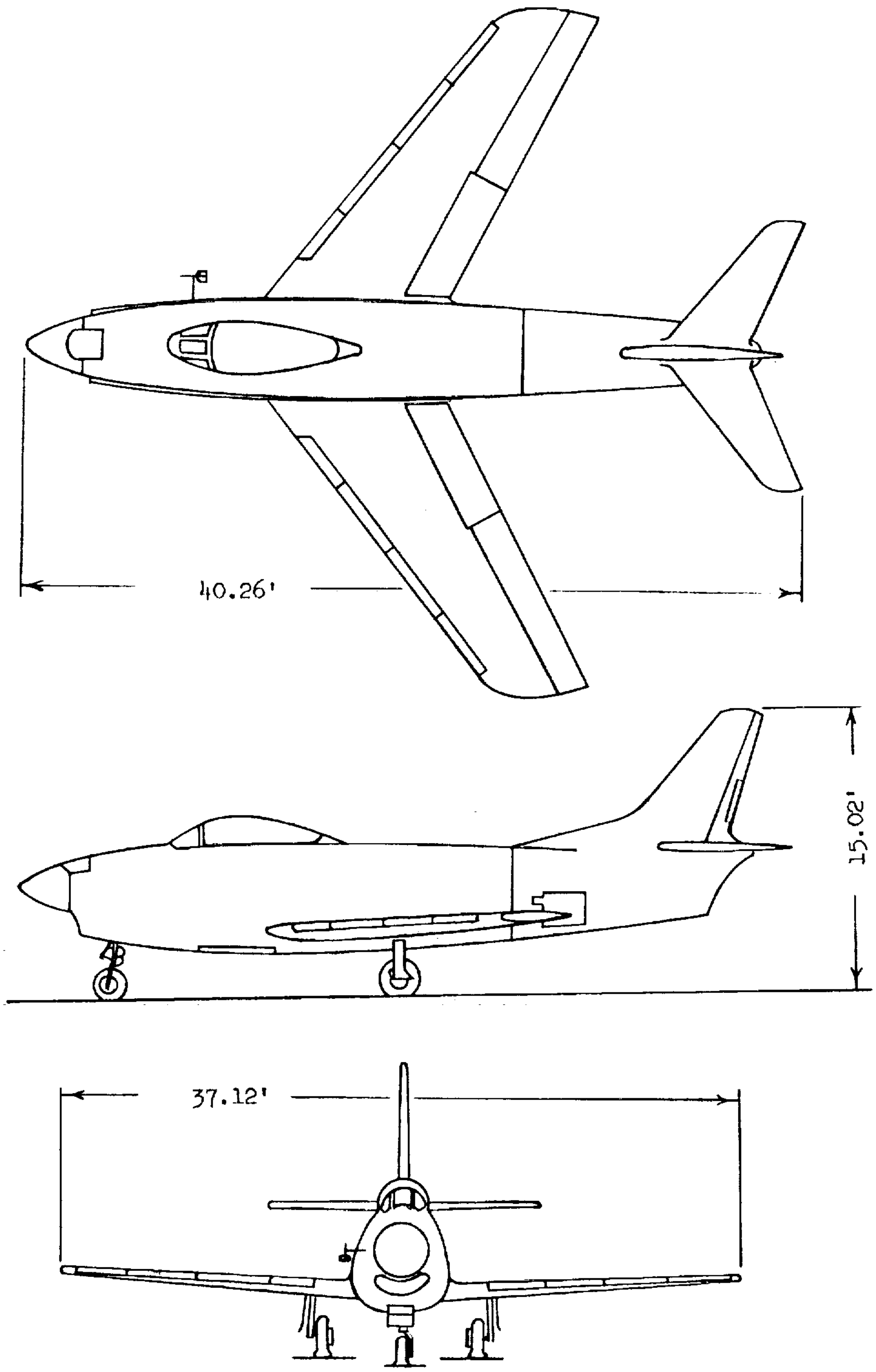

Figure 1.- Three-view drawing of the test airplane.

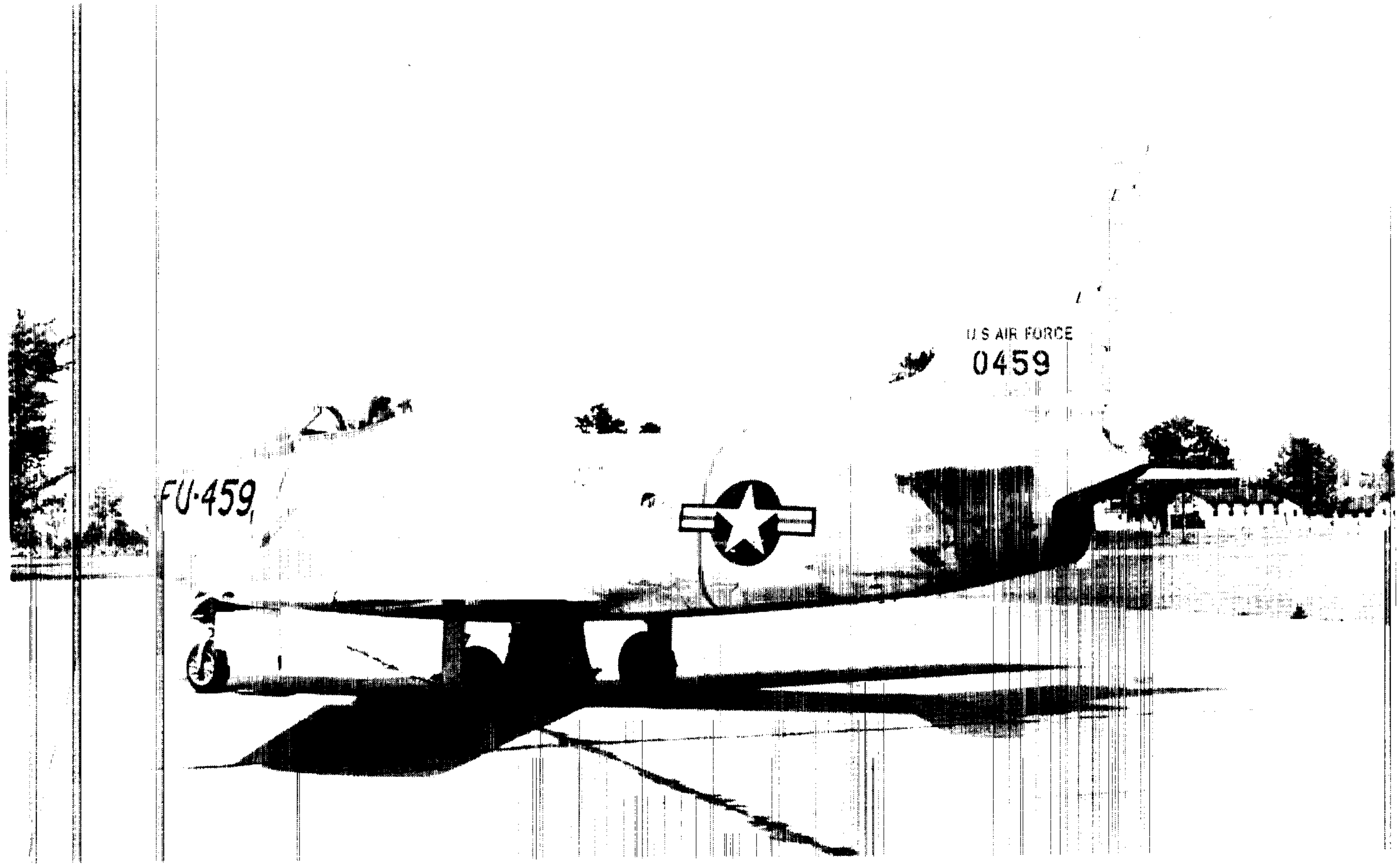

Figure 2.- Test airplane. L-89424

L-679

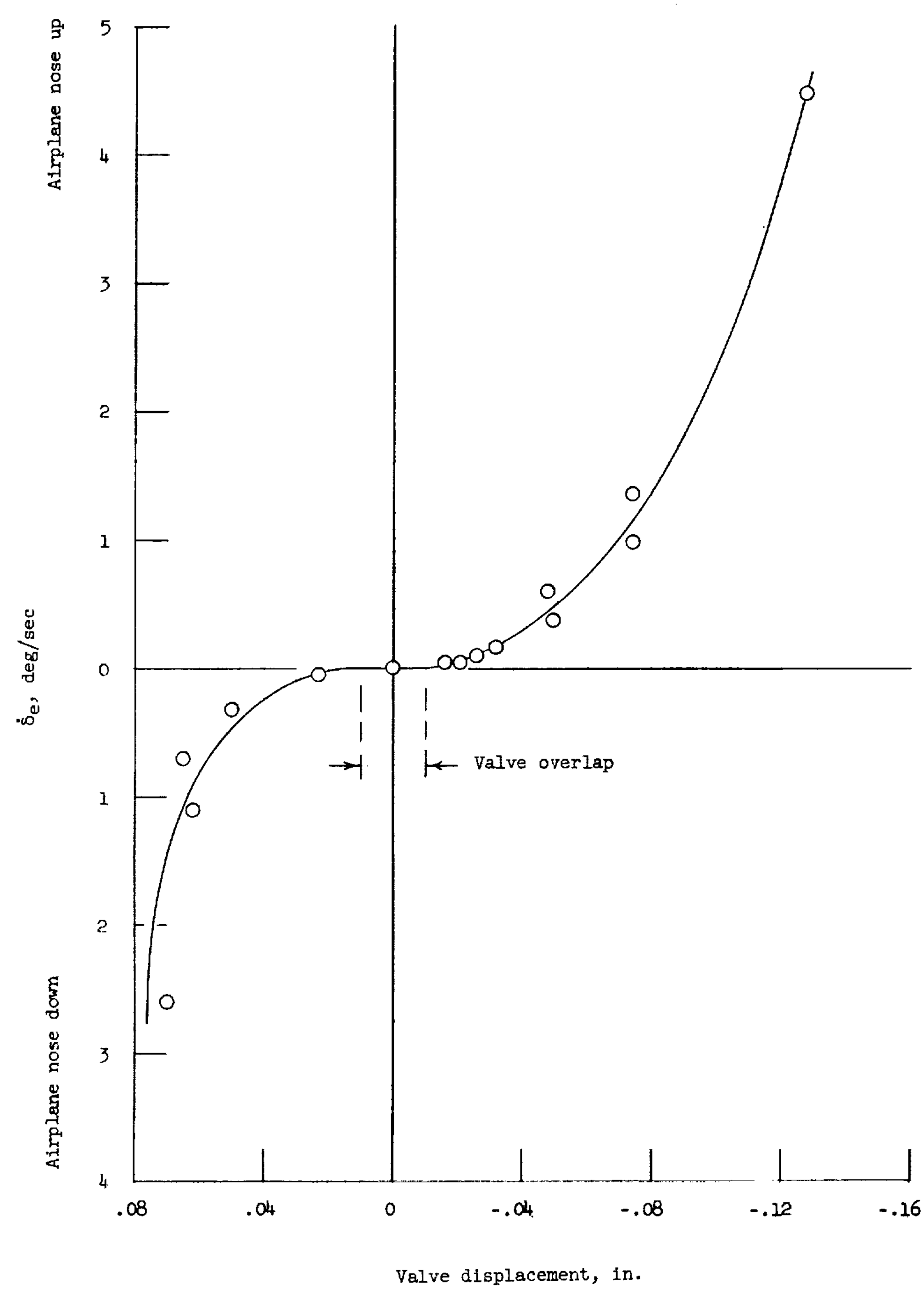

Figure 3.- The valve overlap and nonlinear flow-stroke characteristic of the stabilizer actuator valve. (Maximum valve displacement ≈ ±0.20 inch.)

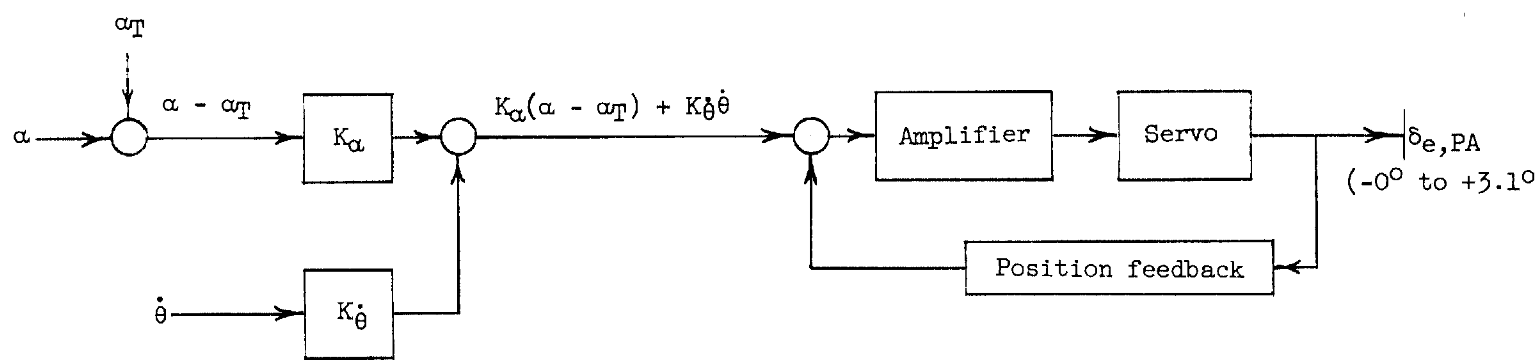

Figure 4.- A block diagram of the final automatic pitchup-control system. When $K_\alpha(\alpha - \alpha_T) + K_{\dot\theta}\dot\theta > 0$, $\delta_{e,PA} = K_\alpha(\alpha - \alpha_T) + K_{\dot\theta}\dot\theta$ and when $K_\alpha(\alpha - \alpha_T) + K_{\dot\theta}\dot\theta < 0$, $\delta_{e,PA} = 0$.

L-679

L-89428

Figure 5.- A view of the test airplane showing the location of the angle-of-attack measuring vane.

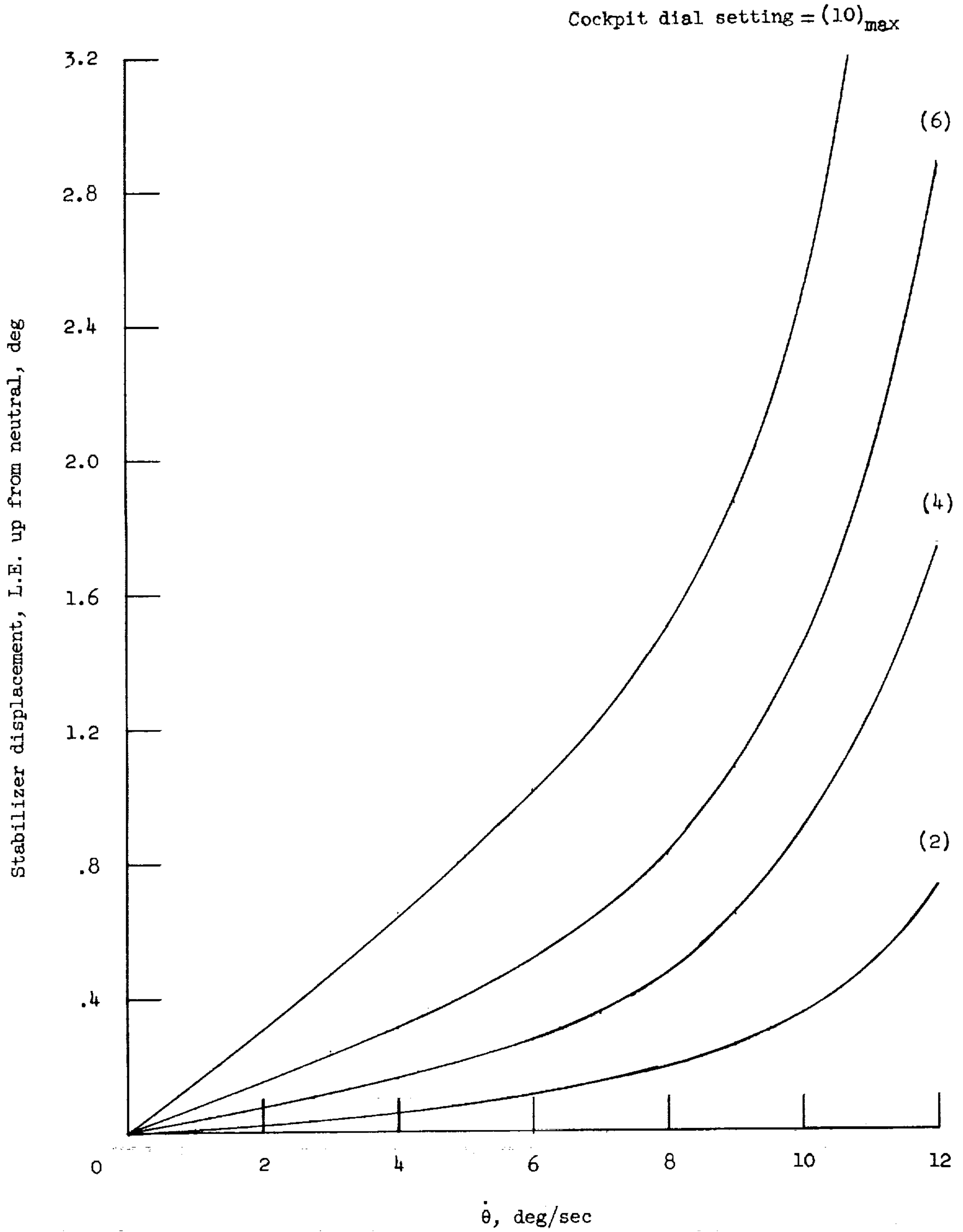

Figure 6.- The cockpit pitching-velocity control gain $K_{\dot{\theta}}$.

L-679

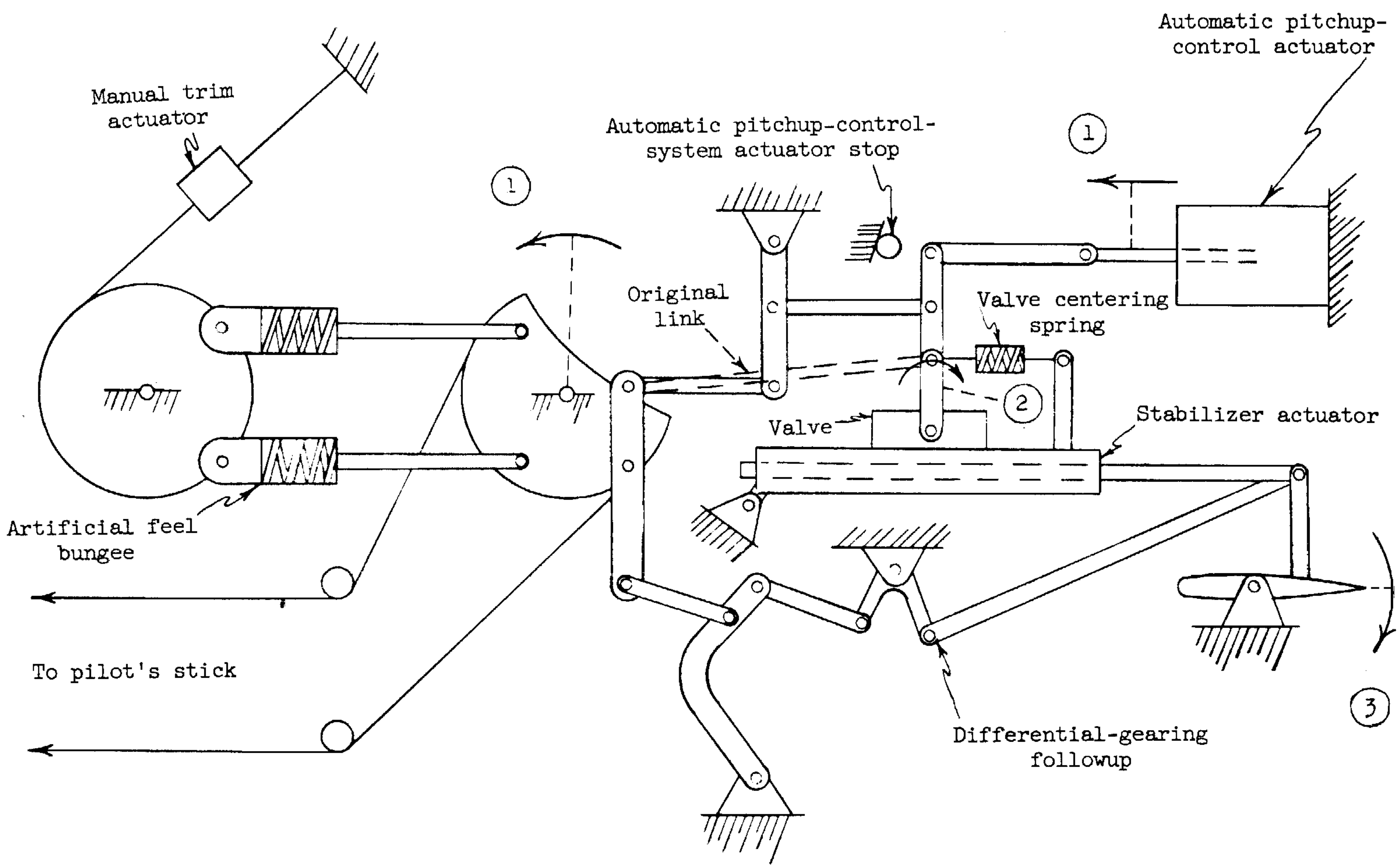

Figure 7.- A schematic diagram of the modified longitudinal control system. Circled numbers indicate sequence for either stick or automatic longitudinal control.

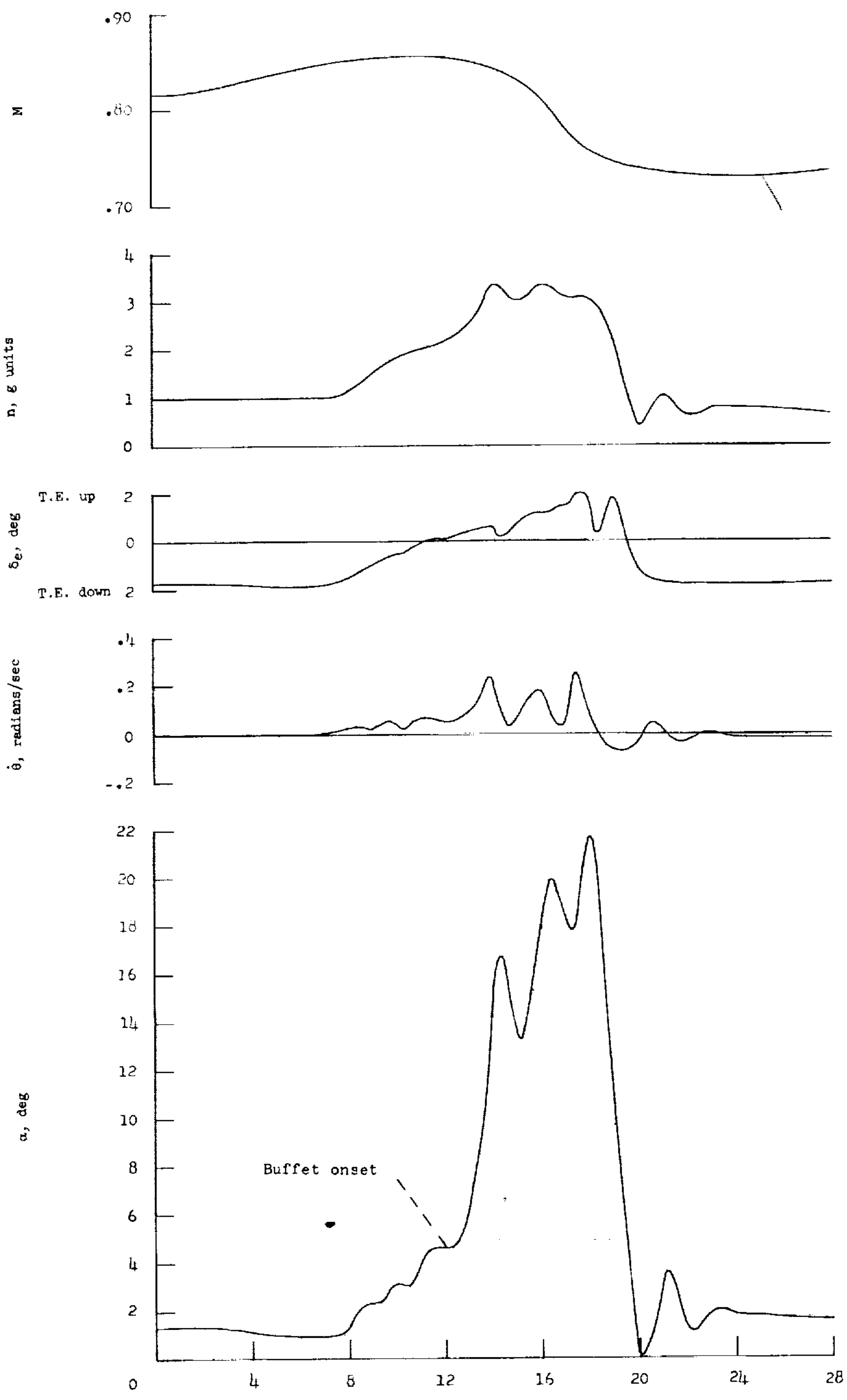

Figure 8.- A pitchup time history of the basic airplane.

L-679

L-679

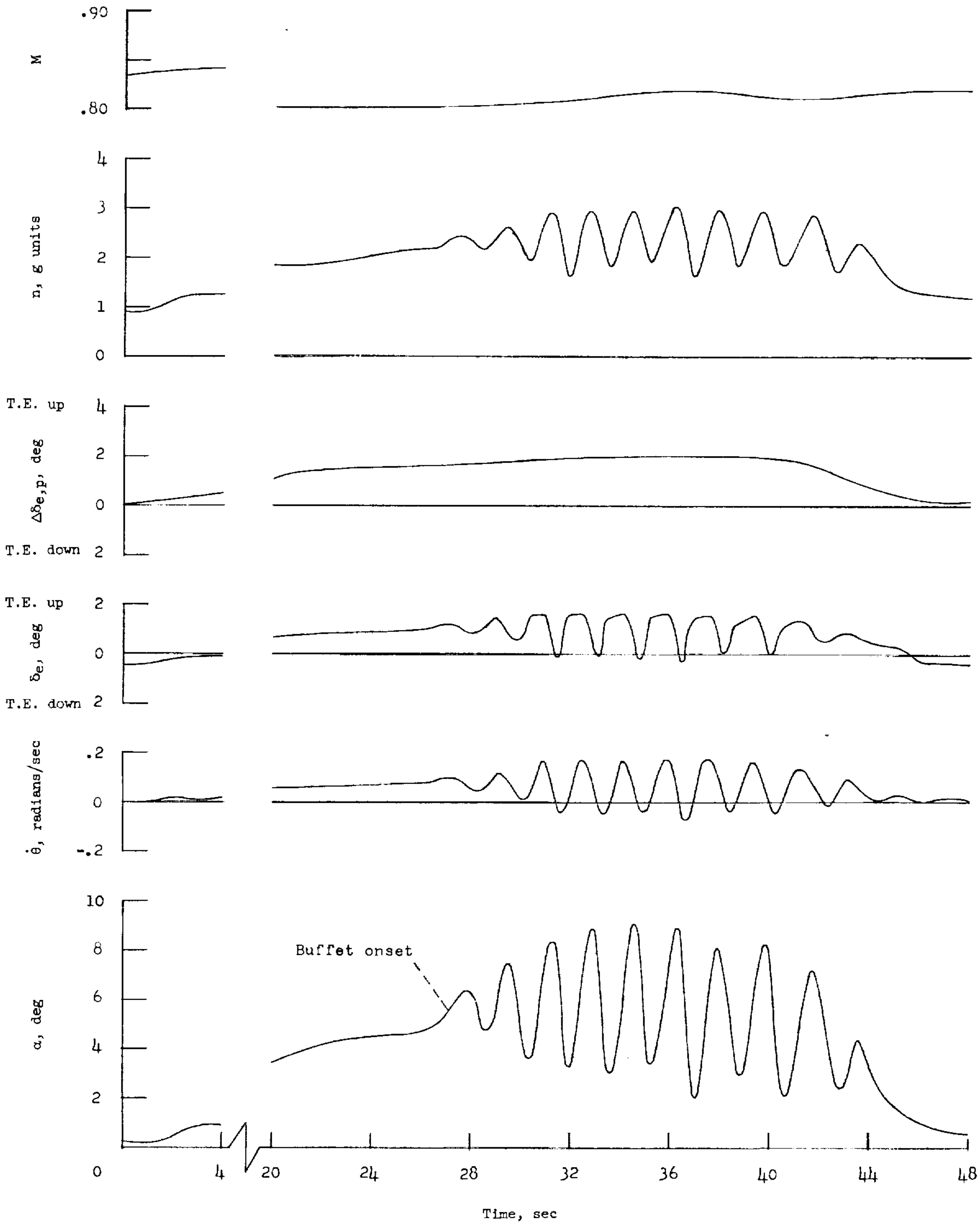

Figure 9.- A time history of the modified airplane with an angle-of-attack input signal only. $\alpha_T = 6^{\circ}$; $K_{\alpha} = 0.62^{\circ}/\text{deg}$.

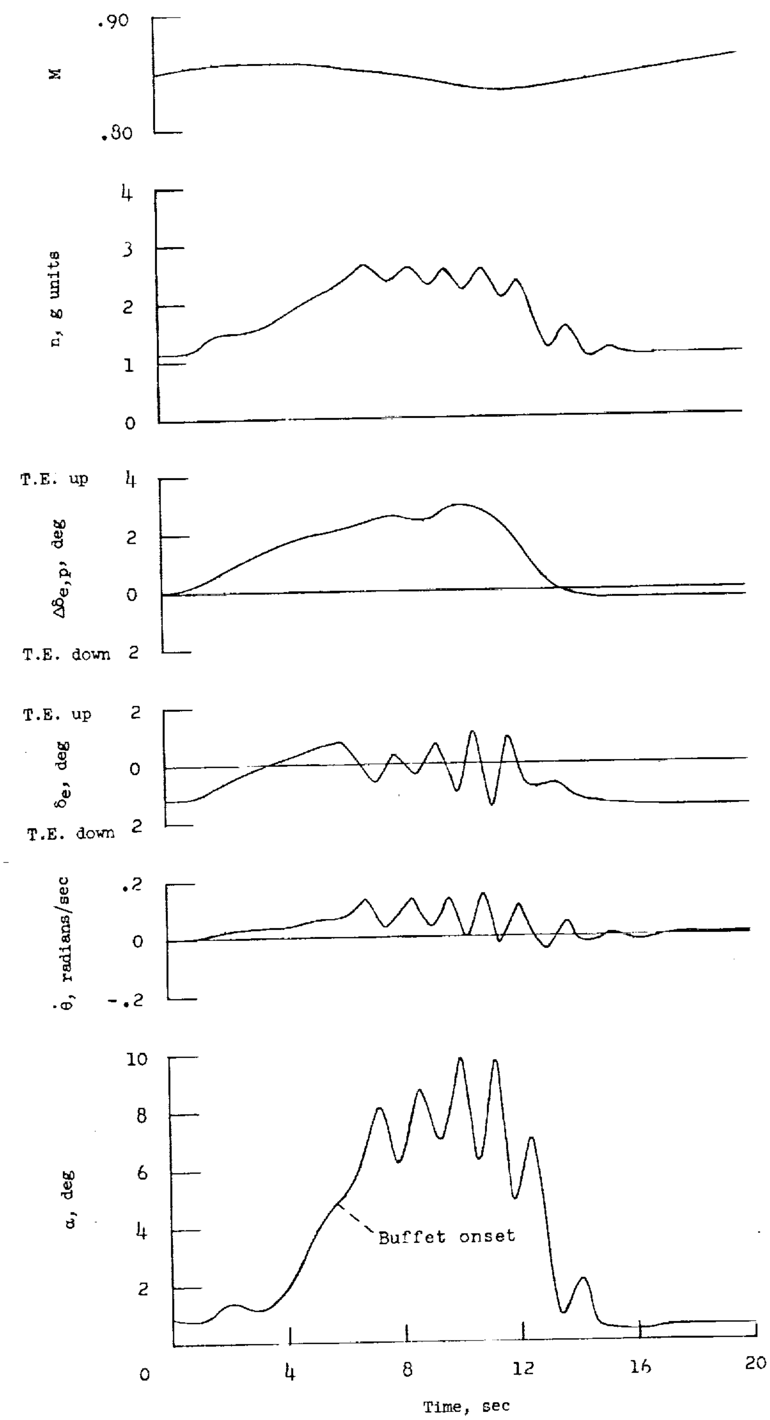

Figure 10.- A time history of the modified airplane with a dual signal input. $\alpha_T = 8^{\circ}$; $K_{\alpha} = 0.62^{\circ}/\text{deg}$; $K_{\dot{\theta}}$ = Maximum.

L-679

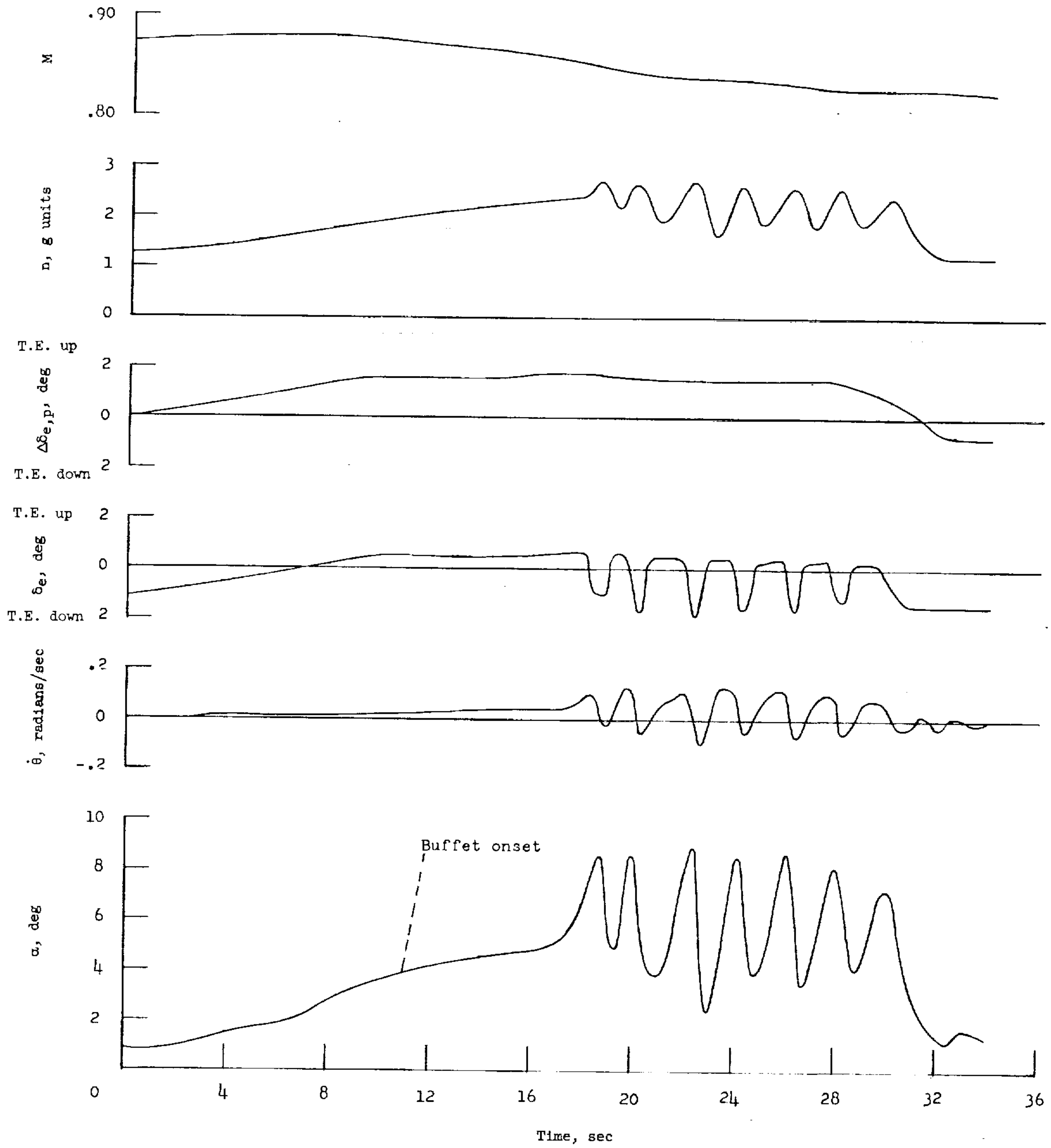

Figure 11.- A time history of the modified airplane illustrating the effect of an increase in K_α (compare with fig. 10). $\alpha_T = 8^\circ$; $K_\alpha = 0.98^\circ$/deg; $K_{\dot{\theta}}$ = Maximum.

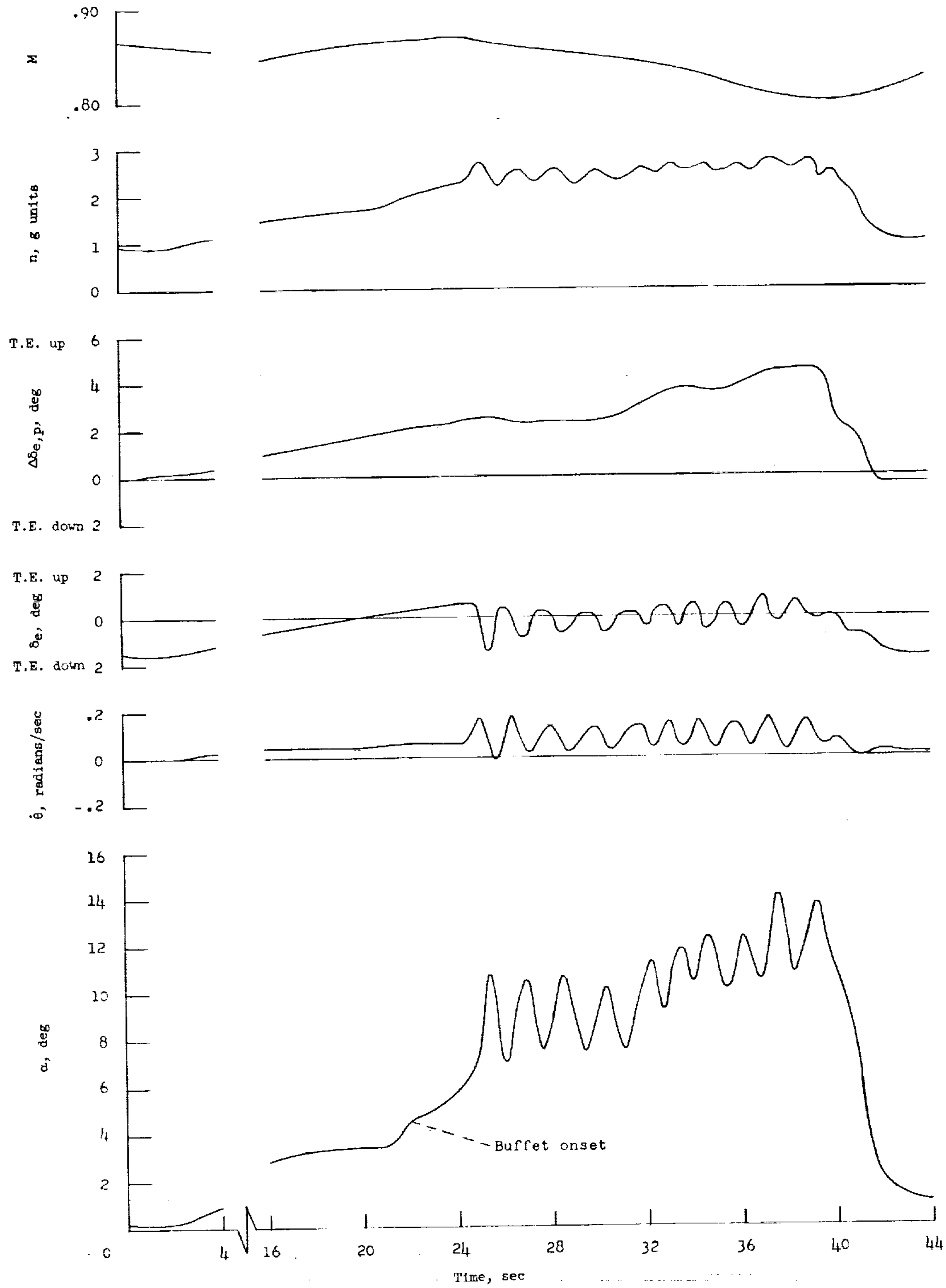

Figure 12.- A time history of the modified airplane with optimum cockpit-control settings. $\alpha_T = 10^\circ$; $K_\alpha = 0.45^\circ/\text{deg}$; $K_{\dot{\theta}}$ = Maximum.

L-679

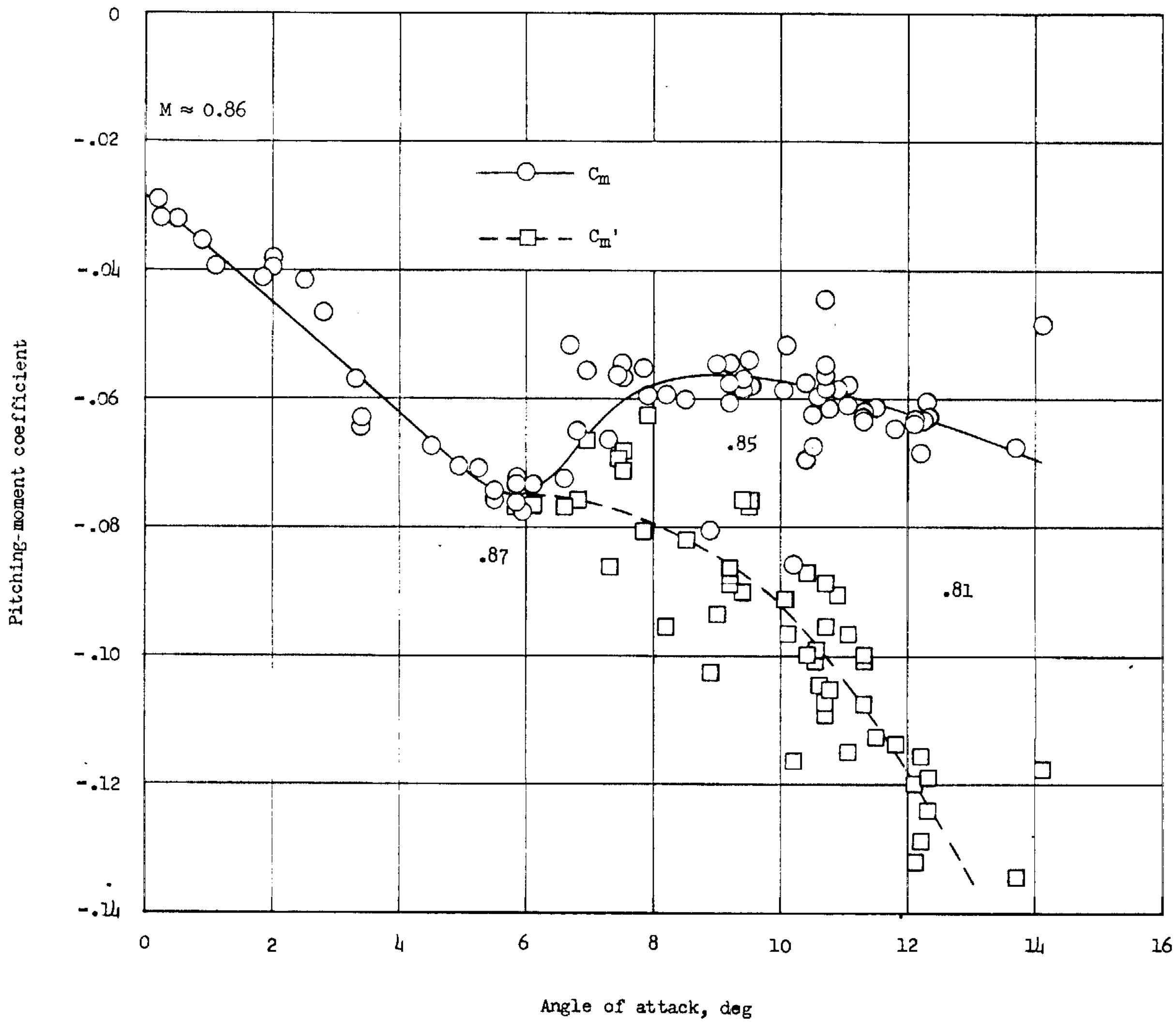

Figure 13.- An illustration of the contribution of the automatic pitchup-control system. $\alpha_T = 10^{\circ}$; $K_{\alpha} = 0.45^{\circ}/\text{deg}$; $K_{\dot{\theta}}$ = Maximum.