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FLIGHT INVESTIGATION OF A NORMAL-ACCELERATION AUTOMATIC LONGITUDINAL CONTROL SYSTEM IN A FIGHTER AIRPLANE

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SUMMARY

A flight investigation was made to obtain experimental information on the handling qualities of a normal-acceleration type of automatic longitudinal control system. The control system was installed in a subsonic fighter-type airplane. In hands-off (stick-free) flight the normal-acceleration control system attempted to regulate the normal acceleration to a constant value which is dependent on the automatic-control-system trim setting. In maneuvering flight a given pilot's stick deflection produced a proportional change in normal acceleration, the change in acceleration being independent of flight condition. A small side-located controller stick was used by the pilot to introduce signals into the automatic control system. In the flight program emphasis was placed on the acceleration-limiting capabilities of the control system.

The handling qualities were investigated in maneuvers such as slow and rapid pull-ups and turns and also in flight operations such as cruising, stalls, landings, aerobatics, and air-to-air tracking. Good acceleration limiting was obtained with the normal-acceleration control system by limiting the magnitude of the input signal that the pilot could introduce into the control system. The same values of control-system gain settings could be used from an acceleration-limiting standpoint at both 10,000 and 30,000 feet for the complete speed range of the airplane. The response characteristics of the airplane-control system combination were also satisfactory at both high and low altitude with these same values of control-system gain setting. In the pilot's opinion, the normal-acceleration control system provided good stability and control characteristics in flight operations such as cruising, stalls, landings, aerobatics, and air-to-air tracking.

*Title, Unclassified.

INTRODUCTION

This paper describes results obtained in a flight investigation of a normal-acceleration type of automatic longitudinal control system. The control system was installed in a subsonic fighter airplane. In hands-off (stick-free) flight, the normal-acceleration control system attempts to regulate the airplane normal acceleration to a constant value which is dependent on the trim setting. In maneuvering flight, a given pilot's stick deflection produces a proportional change in steady normal acceleration, the value of acceleration being independent of airplane flight condition.

The general objectives of the flight program were to obtain experimental information on the handling qualities of an airplane when it is controlled through a normal-acceleration control system and also to determine the advantages that might be associated with a control system of this type. A more specific objective was to obtain information on the acceleration-limiting characteristics of the control system.

The pilot's flight controller used in the present program is the same side-located controller used in the investigation described in reference 1. The flight investigation of the normal-acceleration automatic control system reported herein is a part of a general program in which various types of automatic control systems are being investigated. References 2 and 3 report results obtained with rate and attitude types of automatic control systems.

SYMBOLS

a_n — normal acceleration, g units

h_p — pressure altitude, ft

K_{f_e} — elevator servo feedback gain, volts per radian of servo drum rotation

K_{a_n} — normal accelerometer gain, volts/g

$K_{\dot{\theta}}$ — pitch-rate-gyro gain, volts/radian/sec

M — Mach number

V — true airspeed, ft/sec

V_i indicated airspeed, knots

δ_s conventional control-system or rate control-system stick deflection fore and aft, deg

δ_{a_r} right aileron deflection, deg

δ_{c_l} side-located control stick deflection, lateral, deg

δ_{cp} side-located control stick deflection, fore and aft, deg

δ_e elevator deflection, deg

δ_r rudder deflection, deg

θ angle of pitch, deg

φ angle of bank, deg

ψ angle of yaw, deg

ω circular frequency, radians/sec

α angle of attack, deg

β angle of sideslip, deg

g acceleration due to gravity, ft/sec^2

A dot placed over a symbol indicates differentiation with respect to time.

DESCRIPTION OF AIRPLANE AND CONTROL SYSTEM

Airplane

The airplane used was a fighter-type airplane with an unswept wing and a turbojet engine. A photograph of the airplane is presented in figure 1 and a two-view drawing of the airplane is shown in figure 2. General dimensions and characteristics of the airplane are listed in table I. The wing-tip fuel tanks were on the airplane for all flights but no fuel was carried in them. A hydraulic booster system, which provides a boost ratio of approximately 37:1, is incorporated in the aileron control system of the airplane and a spring tab is used in the elevator control system. The rudder control system is of the conventional manual type.

Control Systems

The normal-acceleration automatic control system was all-electric in operation. Except for the servo actuator which operated on direct current, the components of the control system operated on alternating current. Frequency-response and speed-torque data for the servo actuator are presented in reference 3. The roll-rate control system described in the investigation of reference 2 was used in conjuction with the normal-acceleration control system. Also, the yaw control system was the same as that used in the investigations of references 2 and 3.

A block diagram of the norman-acceleration control system is presented in figure 3. In order to increase airplane damping, pitch rate is used as an inner-loop feedback in addition to the outer-loop normal-acceleration feedback. The function of the canceler system is to reduce steady-state pitch-rate-gyro and servo follow-up signals to zero. The time constant of the canceler system is about 1.4 seconds for all data presented in this report. Since steady-state pitch-rate-gyro and servo follow-up signals are eliminated, the static sensitivity between normal acceleration and controller deflection is independent of airplane speed, altitude, or center-of-gravity position. Steady normal-acceleration limiting, then, can be provided by limiting the maximum available pilot's input signal so that he cannot command an acceleration greater than the desired maximum. For adequate acceleration limiting, it is, of course, also necessary that little or no overshoot of normal acceleration above the steady-state value should result from rapid stick motions. The pitch-rate-gyro, normal accelerometer, stick controller, and servo follow-up gains and the canceler-system time constant were adjustable but no automatic gain changing was incorporated in the system.

The normal accelerometer was mounted 5 feet forward of the center of gravity of the airplane and approximately on the center line of the airplane laterally and vertically. The natural frequency of the accelerometer was about 54 cycles per second, the damping ratio was about 1, and the range was ±5g. The output-signal linearity was better than 0.5 percent. An adjustable gain linear amplifier with a maximum gain of 24 was used to preamplify the accelerometer output signal.

Except for the accelerometer the components of the automatic control system were General Electric G-3 automatic pilot components. Data on the natural frequencies and damping ratios and general information on the G-3 automatic pilot components are given in references 2, 3, and 4.

An accelerometer control system of the type tested is unstable with stick fixed in climbing flight and for nose-up disturbances from level flight. This instability arises because the normal-accelerometer "reading" decreases for straight flight paths as the climb angle increases. For example, if the airplane is trimmed for 1g in level flight and is then disturbed in a nose-up direction, an accelerometer reading of less than 1 g will cause the elevator to be deflected up which will cause the airplane to pitch up further. The degree of instability is a function of the climb angle since for straight flight paths the accelerometer reading is proportional to the cosine of the climb angle. The instability will be very small at small climb angles and larger at large climb angles. The effects of this instability on the handling qualities are discussed in the section entitled "Tests, Results, and Discussion."

The small side-located control stick described in reference 1 was used with the normal-acceleration control system. Figure 4 is a photograph of the side control stick installation. The variation of longitudinal stick force with stick deflection is shown in figure 5. The data presented in figure 5 are from ground measurements and were obtained as the stick was moved slowly. As noted in the figure, the forces are for a 2.75-inch moment arm which corresponds approximately to the point at which the pilot held the stick. The lateral stick-force—stick deflection variation is the same as that presented in reference 1.

The electrical servo actuators of the automatic control system were installed in parallel with the airplane primary control systems. The conventional control stick and rudder pedals therefore followed the control surface motions. For zero load, about 5° of servo-actuator drum rotation produced 1° of elevator deflection. In flight, flexibility of the control system decreased the ratio of elevator deflection to servo actuator drum rotation.

INSTRUMENTATION

NACA recording instruments, which measured the following quantities, were installed in the airplane: normal, longitudinal, and transverse accelerations; pitching, rolling, and yawing velocities and accelerations; airspeed and altitude; elevator, aileron, and rudder positions; aileron and rudder servo positions; angle of attack and sideslip angle; pitch and bank attitude angles; and longitudinal and lateral side-controller stick positions.

The recording accelerometer was located about 12 feet forward of the center of gravity. No corrections for angular accelerations have been applied to the normal acceleration time histories presented in this paper. The airspeed head, which was used to measure airspeed and altitude, was mounted on a boom which extended out of the nose of the airplane. (See fig. 1.) No calibration was made of the airspeed installation and therefore the airspeed and altitude data presented in this paper have not been corrected for position error. It is estimated that the error in the measured static pressure due to the fuselage pressure field is about 2 percent of the impact pressure at low angles of attack. The airplane angle of attack and sideslip angle were measured with vanes which also were mounted on the nose boom. No corrections have been made to the angle-of-attack or sideslip data presented in this paper.

TESTS, RESULTS, AND DISCUSSION

The response characteristics and handling qualities of the combination of the airplane and normal acceleration control system were evaluated in various maneuvers such as rapid and gradual pull-ups and turns, and in various flight operations such as cruising, stalls, landings, and air-to-air tracking. A total of 35 flights were made with the normal-acceleration control system. Nine pilots flew with the system and one pilot made 20 of the flights. The range of flight conditions covered in the test program was from landing speeds at sea level to a maximum Mach number of about 0.80 at an altitude of about 30,000 feet.

Response Characteristics

Transient response.- Time histories of the longitudinal response characteristics in pitch for near step side controller inputs are presented in figure 6. A chain, one end fixed to the airframe and the other end fixed to the top of the side controller, served as a stop for the step inputs. Figures 6(a) to 6(e) were obtained at an altitude of 10,000 feet and are for the speed range from $V_i = 150$ knots to $V_i = 384$ knots $(M = 0.69)$. Figures 6(f) to 6(h) are for an altitude of 30,000 feet and Mach numbers of 0.6, 0.7, and 0.76. The automatic control-system gains used for each run are given in the figures. For all the runs at an altitude of 10,000 feet (figs. 6(a) to 6(e)) and also at a Mach number of 0.7 at 30,000 feet (fig. 6(g)), all the control-system gains were the same. At a Mach number of 0.6 and an altitude of 30,000 feet (fig. 6(f)), the controller sensitivity was slightly less than that for the runs at 10,000 feet. This slightly lower controller sensitivity would have no effect on the dynamic response characteristics of the control-system—airplane combination but would change the static sensitivity between the controller deflection and normal acceleration. At a Mach number of 0.76 and an altitude of 30,000 feet (fig. 6(h)), both the accelerometer and controller gains were slightly lower than those for the runs at 10,000 feet. These slightly lower gains were not necessary from a response or flying-qualities standpoint but the run is presented because no suitable step input records were obtained at this flight condition with the same gains as used for the runs at an altitude of 10,000 feet.

Even though the servo follow-up gain setting was the same for all runs, flexibility in the control cables between the servo actuator and the elevator and the spring tab in the primary longitudinal control system caused the elevator deflection for a given error signal to decrease as the dynamic pressure increased. This effect is equivalent to a reduction in loop gain with an increase in dynamic pressure. As was previously noted, the static sensitivity between elevator deflection and servo actuator drum rotation was about 0.2 for the zero-load condition. At indicated airspeeds of 135 knots and 420 knots, the static sensitivities between elevator deflection and servo drum rotation were, respectively, about 0.17 and 0.071.

At an altitude of 10,000 feet (figs. 6(a) to 6(e)), there is no overshoot of normal acceleration at any speed. At the slowest speed, $V_i = 150$ knots in both the clean and landing-approach conditions (figs. 6(a) and 6(b)), the response of the airplane is slow but still about as fast as for the airplane alone. In the pilot's opinion, the response was satisfactory. No attempt was made to determine the best gains for low-speed flight in order to reduce the airplane response times. As the speed is increased at 10,000 feet, the airplane response becomes much more rapid and at a Mach number of 0.7 (fig. 6(e)) the response time of normal acceleration is about 0.4 to 0.5 second. The response time for the basic airplane at this flight condition is about twice this value. The pilot was of the opinion that the airplane response at the high speeds at an altitude of 10,000 feet was tolerable but he was of the opinion that a slower response would be more desirable. This opinion is based on precision-flying and comfort considerations rather than on any safety considerations as no tendency for pilot-induced oscillations was present and the airplane damping was good. Inspection of the elevator-position traces for the runs at Mach numbers of 0.6 and 0.7 at an altitude of 10,000 feet (figs. 6(d) and 6(e)) shows that large elevator pulses result from the step controller motions. This, of course, is the reason for the rapid airplane response. One method of obtaining a slower airplane response to pilot inputs at the higher speed would be to have a lag network operating on the pilot's input signal. However, it would be necessary to vary the time constant of the lag network in order to provide satisfactory airplane response at lower speeds and at high altitudes. With this method of slowing the airplane response to pilot inputs, a tight control for other disturbances, such as, for example, airplane configuration changes, is retained.

In establishing optimum gains for the higher speed conditions, the main consideration was that there should be little overshoot of normal acceleration above its steady-state value for near step controller inputs. This is a necessary condition if the normal-acceleration control system is to function as an acceleration limiter. However, for step controller inputs, some overshoot is acceptable because, when approaching the limit load factor, the pilot would not be likely to use such rapid inputs.

In the runs at an altitude of 30,000 feet (figs. 6(f), 6(g), and 6(h)) there is moderate overshoot of normal acceleration at Mach numbers of 0.6 and 0.7, and at a Mach number of 0.76 (the accelerometer gain was lower for this run), the normal-acceleration response appears to be overdamped. However, the pilot considered the response characteristics for all of these runs to be satisfactory. The rough normal-acceleration trace in figure 6(h) is caused by buffeting.

The variation of steady normal acceleration with side controller deflection, for constant values of control-system gain, has been determined from runs such as presented in figure 6 and other runs at various speeds and altitudes. These data are presented in figure 7. The variation of normal acceleration with stick deflection is seen to be linear and the static sensitivity between side controller deflection and normal acceleration is about 8° per g. Using this value of controller deflection per g and referring to the stick force-stick deflection curve (fig. 5), the stick force per unit acceleration is found to be about $2\frac{1}{4}$ pounds per g. In the pilot's opinion, the values of both the stick deflection per g and the stick force per g were in the satisfactory range.

In order to evaluate the acceleration limiting capabilities of the control system further, larger amplitude pilot controller inputs were also used. Figure 8 shows time histories of near step inputs to steady-state normal accelerations of about 2g and 4g. These runs were made at a Mach number of 0.6 and an altitude of 15,000 feet. The figure shows that increasing the magnitude of the pilot's input did not cause the normal acceleration to overshoot its steady-state value and the airplane response is very similar to that resulting from the smaller amplitude input. Another maneuver to illustrate the acceleration-limiting capabilities of the system at larger accelerations is shown in figure 9. Figure 9 is a time history of a maneuver at a Mach number of 0.67 and an altitude of 15,000 feet in which the pilot when flying in a turn at an acceleration of about 3.8g rapidly pulsed the stick fore and aft. A stop was used to limit the stick deflection to about 20°. This maneuver well illustrates the acceleration-limiting capabilities of the control system as any overshoots are of very small magnitude. The acceleration-limiting characteristics for the condition of changing airspeed are shown in figure 10. Figure 10 is a time history of a controller fixed turn in which the Mach number decreased from 0.67 to 0.46. The acceleration-control system was able to maintain an almost constant acceleration. The elevator deflection required to maintain the acceleration increased rapidly near the end of the run and a small decrease in the normal acceleration resulted. The wiggles in the normal acceleration trace at times between about 9 and 15 seconds result from rough air.

As previously mentioned in the discussion of the normal-acceleration control system, the airplane-control-system combination was unstable for straight flight paths in climbing flight. The degree of instability is proportional to $1 - \cos\theta$. For flight operations involving small airplane attitude angles (near level flight), such as cruising and landings, the instability was not noticeable to the pilot. Any tendency for the airplane to diverge was extremely slow and also the divergence tendency was masked by other factors such as inexact stick centering. The inexact stick centering resulting from friction could cause an acceleration increment of about 0.25g from the original trim setting. In order to explore the effects of the instability further, the pilot made maneuvers in which he trimmed in level flight and then made pull-ups to various climb angles, the maximum climb angle being about 15°. Again any divergence tendency of the airplane was extremely slow and was not considered by the pilot to be objectionable. For airplanes which are capable of climbing at very steep climb angles, the attitude-angle instability in climbs might be more objectionable. In order to give an idea of the magnitude of the pitch-divergence rates at various climb angles, the time to double the amplitude of a small disturbance from the trim climb angle has been estimated. The method used in making the estimates is presented in the appendix. The results for a true airspeed of 600 feet per second are as follows:

Trim climb angle, deg	Time for small disturbance from trim climb angle to double in amplitude, sec
5	148
15	49.9
30	25.8
45	18.3
60	14.9
75	13.4
90	12.9

For a given trim climb angle, the time to double the amplitude of a disturbance is directly proportional to the true airspeed. It should be noted that the instability is different from aerodynamic static instability in that the divergence would be in attitude, the normal acceleration being nearly constant. On the other hand, aerodynamic instability would result in a divergence in normal acceleration. Although the attitude angle instability was not objectionable to the pilot in the present case, if a fixed attitude or altitude were to be held for a long period, a suitable long-period reference would be required. Since the divergence rates are very slow at small attitude angles, the possibility of providing long-period stabilization by attitude or altitude hold appears to be practicable.

At the inauguration of the flight research program with the normal-acceleration control system, the possibility that the normal accelerometer might be excited by structural vibrations and thus cause elevator oscillations was recognized and this possibility was explored very thoroughly in the early flights. Flying was done under conditions of heavy buffeting. In addition, very rapid controller pulses and step inputs were used at various flight conditions to see whether any coupling existed between structural motion and elevator motions. At no time in the flight program was any coupling noted. The natural frequency of the servo actuator used in the normal-acceleration control system is between 2 and 3 cycles per second and this frequency is considerably lower than the natural frequency of any airplane structural mode. This may explain why structural vibrations had no apparent adverse effects on the control system. No gun firing has been done in the present program and any effects of noise arising from this source have not been determined.

Frequency response.- Frequency analyses were made of transient responses such as presented in figure 6 in order to obtain frequency-response data. The method of reference 5 was used in performing the frequency analyses. Figure 11 presents frequency-response data for the normal-acceleration control system. For comparison purposes, frequency-response data are shown also for the same airplane having the rate control system described in reference 1 and for the airplane with its manual control system. All the data are for a Mach number of approximately 0.6 and an altitude of 10,000 feet. The longitudinal output response quantities are normal acceleration and pitching velocity and the input quantity is side controller motion or stick motion. In order to make the data for the three control systems more directly comparable, the amplitude ratio curves for each control system have been normalized with respect to its value at zero frequency. The normal-acceleration frequency-response data are presented in figure 11(a). With the normal-acceleration control system, the peak in the amplitude-ratio curve occurs at a considerably higher frequency than with either the rate or airplane manual control systems. Also, the amplitude ratio curve for the acceleration control system is flat out to the peak amplitude ratio. Both of these characteristics are indicative of the rapid and well-damped transient response previously shown for this flight condition.

In general, the phase lags with the acceleration control system are less than those with the rate or manual control systems at frequencies below the frequency for peak amplitude ratio. At frequencies greater than the frequency for peak amplitude ratio, the normal-acceleration system has larger phase lags. This condition indicates a greater tendency for the acceleration system to couple with higher modes and thus cause higher frequency instabilities.

The pitching-velocity frequency-response data are presented in figure 11(b). The amplitude-ratio curve for the normal-acceleration control system is markedly different from those for the other two control systems in that the peak amplitude ratio is much larger and occurs at a higher frequency. This type of pitching-velocity amplitude-ratio curve results in an overshoot in the pitching-velocity response for a step command in normal acceleration. As was previously mentioned, the normal-acceleration response time was short (about 0.4 to 0.5 second at this flight condition) and in the pilot's opinion the response was tolerable but he would prefer a slower response. The phase lags with the normal-acceleration control system are again smaller at frequencies less than the frequency for the peak amplitude ratio and larger at frequencies greater than the frequency for the peak amplitude ratio.

Flight Operations

The characteristics of the airplane normal-acceleration control system were further evaluated in various flight operations such as cruising, stalls, landings, and air-to-air tracking.

Cruising characteristics.- The cruising characteristics of the normal-acceleration control system were investigated in a flight from Langley Field, Virginia, to Charleston, West Virginia, and return. In the pilot's opinion the normal-acceleration control system was very satisfactory for cruising flight. Also, the roll and yaw channels of the automatic control system decreased the tendency for any spiral divergence of the airplane when the pilot's attention was diverted. Short periods of instrument flight were also performed during that flight and the pilot also considered the instrument flight characteristics to be satisfactory. For hands-off automatic flight some positional stability (attitude or altitude hold) would be required.

Stalling characteristics.- The stalling characteristics were investigated in 1g stalls in the power-approach and clean conditions (figs. 12 and 13) and in accelerated flight in the clean condition (fig. 14). In the power-approach stall (fig. 12), the approach to the stall was started at about 130 knots but in order to keep the figure to a reasonable size, only that part of the run in which the airspeed is below 108 knots is presented. The controller motions (both longitudinal and lateral) used by the pilot during the stall approach and stall are very small and the controller was near neutral until the stall recovery was initiated. The pilot noticed little or no stall warning and at the stall the airplane started oscillating about all three axes. The pitching oscillation is probably caused by the wing alternately stalling and unstalling. When the wing stalls the airplane pitches nose down and the normal-acceleration control system, in the absence of any controller input, moves the elevator up (and thus pitches the airplane up) to try to maintain an acceleration of 1 g. The rolling oscillation was quite small in amplitude, the maximum bank angle being about 15°. The roll-rate control system caused considerable aileron deflection to be used in trying to maintain the rolling velocity at zero. Recovery from the stall was accomplished simply by moving the controller forward. Except for the lack of stall warning, the pilot considered the stalling characteristics in the power-approach condition to be satisfactory because the motions of the airplane at the stall were mild, particularly the initial airplane motions due to stalling, and the recovery from the stall could be accomplished rapidly and simply. In the pilot's opinion the normal-acceleration control system had no adverse effects on the stalling characteristics. The pilot had no objections to the essentially neutral stick-fixed stability present during the stall approach. Neutral stick-fixed speed stability is also a characteristic of the rate system described in references 1 and 2. Some of the pilots who have flown with the rate system thought that the lack of positive stick-fixed speed stability was undesirable during stall approaches.

In the 1 g clean condition stall (fig. 13), there was again very little stall warning. At the stall the airplane pitched nose down and rolled slightly. The pilot initiated the recovery as soon as uncontrolled for motions occurred and recovery was accomplished easily.

An accelerated flight stall is shown in figure 14. The principal airplane motion at the stall is a pitching oscillation with very little rolling or yawing. In the pilot's opinion, the stalling characteristics present with the normal-acceleration control system were at least as good as those of the basic airplane. Also, as previously mentioned, stall buffeting had no apparent adverse effect on the normal-acceleration control system.

Landing characteristics.- A time history of a landing made when using the normal-acceleration automatic control system is shown in figure 15. The pilot reported no difficulty in making landings with the control system and figure 15 shows no unusual characteristics. The pilot pumps the controller fore and aft in the same manner as is usual with conventional control systems. Although the pitch attitude angle of the airplane just prior to contact is about 11°, the pilot did not notice any tendency for the airplane to diverge in the nose-up direction due to the nose-up attitude-angle instability previously discussed.

Air-to-air tracking characteristics.- Tracking runs on a target airplane were made to evaluate the precision control characteristics of the normal-acceleration control system. For comparison purposes, tracking runs were also made when the pilot was controlling the airplane with the conventional control system. The rudder channel of the automatic control system was used for all tracking runs. The tracking was done at Mach numbers between 0.6 and 0.7, an altitude of about 20,000 feet, and at a range of about 500 yards. A fixed optical gunsight was used for the tracking and a 16-mm camera was used to photograph the gunsight presentation. The gunsight camera records were evaluated in terms of the standard deviations of the pitch and yaw sighting errors.

The following maneuvers were used in the air-to-air tracking: nonmaneuvering tail chase, 2g and 4g steady turns. Table II shows a comparison of the tracking errors when using the normal-acceleration and the conventional control systems. The tracking errors with the automatic control system are generally somewhat greater than those obtained with the conventional control system but the differences are not believed to be significant. The pilot was of the opinion that he could not do a significantly better job with one system than with the other. It should be noted that the flying qualities of the basic airplane with the yaw damper were very good and the pilot was able to do excellent tracking with the conventional control system.

Comparison of Normal-Acceleration and Pitch-Rate Longitudinal Control Systems

In this section of the paper some characteristics of the pitch-rate longitudinal control system described in reference 2 are compared with the characteristics of the normal-acceleration longitudinal control system.

With the pitch-rate control system the "best" gains were selected on the basis that the pitching angular velocity response to step controller inputs should be rapid but have little or no overshoot. With the normal-acceleration control system the "best" gains were selected on the basis that the normal-acceleration response to step inputs should be rapid but have little or no overshoot. This resulted in an overshoot in the pitching-velocity response for the acceleration control system, the peak amplitude of pitching velocity being considerably greater than the steady-state value. Correspondingly, the normal-acceleration response was considerably more rapid with the acceleration system than with the rate system. For a maneuvering control the pilots had little or no preference for one type of control system over the other.

With either the rate or acceleration control system, the trim changes resulting from flaps, gear, and so forth, are automatically trimmed out but in a somewhat different manner. During the transient, the rate system tries to maintain the airplane at the initial attitude and the acceleration system tries to maintain the airplane at the initial flight-path angle. The pilots considered the automatic trimming with either system to be highly desirable but had a slight preference for the type of trimming provided by the acceleration system. The reason for this preference was that, with the rate system when the flap position was changed during the final landing approach, the airplane tended to diverge farther from the initial flight path than with the acceleration system.

In steady rolling maneuvers the longitudinal and directional motions which occur are considerably different for the rate and acceleration control systems. Figure 16 shows time histories of aileron rolls made with the pitch-rate and normal-acceleration control systems. The lateral controller input is almost identical in both rolls and therefore only one curve is shown for lateral-controller position. The longitudinal controller motion was small in both cases. The rolls were continued through about two complete revolutions and the bank angles at various times are noted on the figure. As expected, the maximum pitching velocities occurring during the rolls are smaller for the rate-control system and the maximum normal accelerations are smaller for the normal-acceleration control system. Also, it should be noted that the maximum sideslip angles are smaller for the rate-control system. In the pilot's opinion, the differences in the longitudinal motions of the airplane that occurred are not of particular importance at moderate rolling velocities, such as in the maneuvers of figure 16, because the pilot was of the opinion that he could control the motions. At higher rolling velocities this is probably not the case.

In flight through turbulent air, the normal-acceleration and pitch-rate control systems would be expected to provide somewhat different airplane responses. The normal-acceleration system would attempt to maintain a constant acceleration by pitching the airplane to compensate for the gust and the pitch-rate system would attempt to maintain a constant airplane attitude. An analytical study of normal-acceleration automatic control systems presented in reference 6 indicates that some alleviation of low-frequency variations in normal acceleration would be obtained in flight through rough air, but at the same time an appreciable increase in the pitching response of the airplane would occur. In the present program, when flying in moderate to heavy turbulence, the pilot did not notice any difference in the airplane response when flying the basic airplane, the airplane with the pitch-rate control system, or the airplane with the normal-acceleration control system.

CONCLUSIONS

A flight investigation was made to obtain experimental information on the handling qualities of a subsonic fighter airplane which the pilot controlled by supplying signals to a normal-acceleration type of automatic longitudinal control system. A roll-rate automatic control system for lateral control and a yaw damper for directional control were used in conjunction with the normal-acceleration control system. The pilot introduced signals into the automatic control system through a small side-located stick controller. The range of flight conditions covered in the test program was from landing speeds at sea level to a maximum Mach number of about 0.8 at an altitude of 30,000 feet. The main conclusions reached as a result of the flight program are as follows:

1. Good normal-acceleration limiting was obtained with the normal-acceleration control system by restricting the magnitude of the signal that the pilot could introduce into the control system.

2. Satisfactory (but not necessarily optimum) airplane response was obtained with the control-system gain settings held constant throughout the range of flight conditions covered in the flight program. At the flight conditions for the highest dynamic pressure (Mach number of 0.7 to 0.75, altitude of 10,000 feet), the airplane response was very rapid (normal-acceleration response time of 0.4 to 0.5 second) and the pilot would have preferred a slower response.

3. For the climb angle range of which the test airplane was capable, the attitude-angle instability inherent in the normal-acceleration control system was not noticeable to the pilot.

4. The values of steady stick force per unit acceleration of about $2\frac{1}{4}$ pounds per g and steady stick deflection per unit acceleration of 8° per g present with the side controller at moderate accelerations were in the pilot's opinion satisfactory.

5. In the pilot's opinion and also as indicated by the data obtained, the normal-acceleration control system provided good stability and control characteristics in flight operations such as cruising, stalls, landings, aerobatics, and air-to-air tracking.

6. No difficulty was experienced in the flight program from the accelerometer being excited by structural vibrations probably because the structural modes were considerably higher than the airplane control modes.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., August 7, 1958.

APPENDIX

ESTIMATION OF DIVERGENCE RATES FOR NORMAL-ACCELERATION CONTROL SYSTEM

For a straight flight path in a climb

$$a_n = \cos \theta_o$$

where a_n is the accelerometer reading at the trim climb attitude angle θ_o.

Making the assumption that the acceleration control system maintains the acceleration constant as the attitude angle θ changes

$$\cos \theta_o = \cos\left(\theta_o + \Delta\theta\right) + \frac{V}{g}\frac{d\theta}{dt}$$

where $\Delta\theta$ is a small change from the trim climb angle θ_o. The $\cos\left(\theta_o + \Delta\theta\right)$ can be written as $\cos \theta_o \cos \Delta\theta - \sin \theta_o \sin \Delta\theta$. Since $\Delta\theta$ is small, the assumptions are made that $\cos \Delta\theta = 1$ and $\sin \Delta\theta = \Delta\theta_o$. Then

$$\cos\left(\theta_o + \Delta\theta\right) = \cos \theta_o - \sin \theta_o \, \Delta\theta$$

and

$$\frac{V}{g}\frac{d\theta}{dt} = \sin \theta_o \, \Delta\theta$$

The solution of this equation is

$$\Delta\theta = e^{g/V \sin \theta_o t}$$

The time to double amplitude t_2 is then

$$t_2 = \frac{V}{g \sin \theta_o} \log_e 2$$

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2. Russell, Walter R., Sjoberg, S. A., and Alford, William L.: A Flight Investigation of the Handling Characteristics of a Fighter Airplane Controlled Through a Rate Type of Automatic Control System. NACA RM L56F06, 1956.

3. Sjoberg, S. A., Russell, Walter R., and Alford, William L.: A Flight Investigation of the Handling Characteristics of a Fighter Airplane Controlled Through an Attitude Type of Automatic Pilot. NACA RM L56A12, 1956.

4. Anon: Handbook of Operation and Service Instructions for G-3 Automatic Pilot (General Electric). Rep. AN 05-45FB-1, BuAer and Air Materiel Command, June 15, 1951.

5. Huss, Carl R., and Donegan, James J.: Tables for the Numerical Determination of the Fourier Transform of a Function of Time and the Inverse Fourier Transform of a Function of Frequency, With Some Applications to Operational Calculus Methods. NACA TN 4073, 1957.

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TABLE I

GENERAL AIRPLANE DATA

Length (excluding nose boom), ft	38.13
Weight, take-off (tip tanks empty), lb	14,460
Center-of-gravity position, take-off, percent M.A.C.	26.5
Center-of-gravity position, landing (1,000 lb fuel), percent M.A.C.	28.4
Engine	J42-P-8
Wing:	
Span (with tip tanks), ft	37.99
Span (without tip tanks), ft	35.35
Area (without tip tanks), sq ft	250
Airfoil section	NACA 64_1A012
Aspect ratio (without tip tanks)	4.97
Taper ratio	0.46
Incidence, deg	0
Dihedral, deg	4
Twist, deg	0
Sweep of 27-percent chord line, deg	0
Mean aerodynamic chord (M.A.C.), in.	89.45
Total aileron area, sq ft	18.44
Aileron travel, deg	19 up; 14 down
Horizontal tail:	
Span, ft	17.21
Area (including elevator), sq ft	66.20
Elevator area, sq ft	19.20
Elevator travel, deg	18 up; 15 down
Tail length, 25-percent M.A.C. of wing to elevator hinge line, ft	18.45
Vertical tail:	
Area (not including dorsal fin), sq ft	36.02
Rudder area, sq ft	8.54
Rudder travel, deg	±26

TABLE II

STANDARD DEVIATIONS OF TRACKING ERRORS WITH

NORMAL-ACCELERATION CONTROL SYSTEM AND

CONVENTIONAL CONTROL SYSTEM

Maneuver	Pitch error, mils*, for -		Yaw error, mils*, for -	
	Normal-acceleration control system	Conventional control system	Normal-acceleration control system	Conventional control system
Nonmaneuvering tail chase	2.3	1.0	1.6	1.2
Turns, 2g to 4g	3.7	3.1	4.5	2.5

*6,400 mils = 360°.

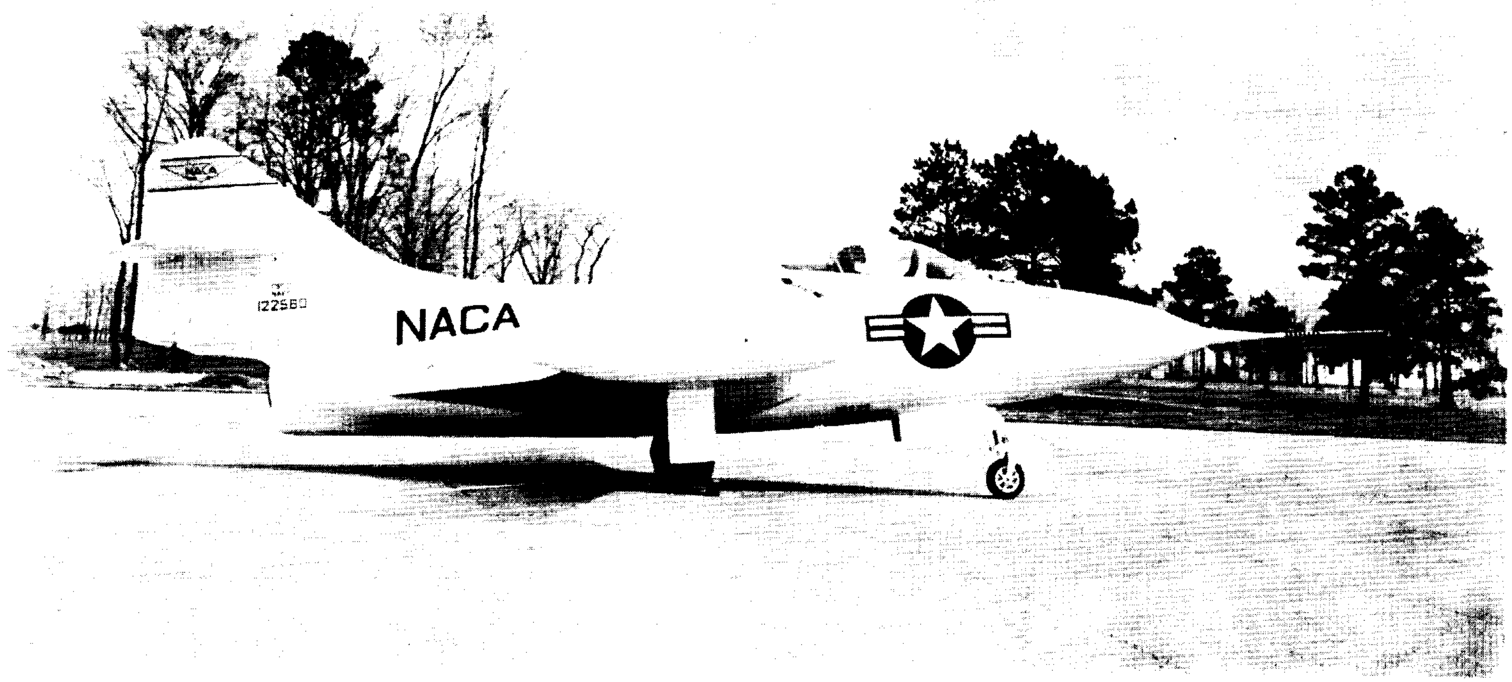


Figure 1.- Side view of airplane. L-93054

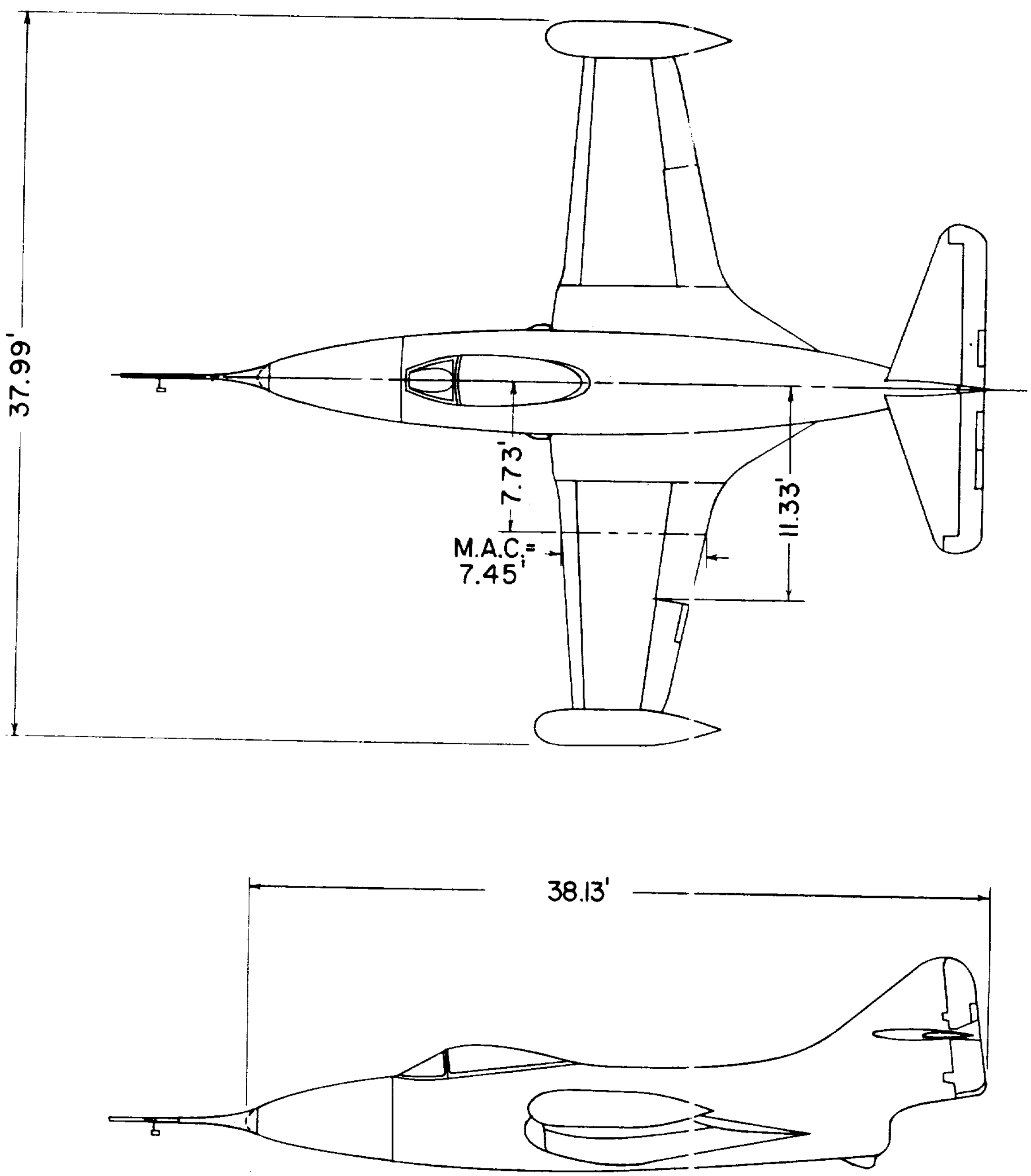


Figure 2.- Two-view drawing of airplane.

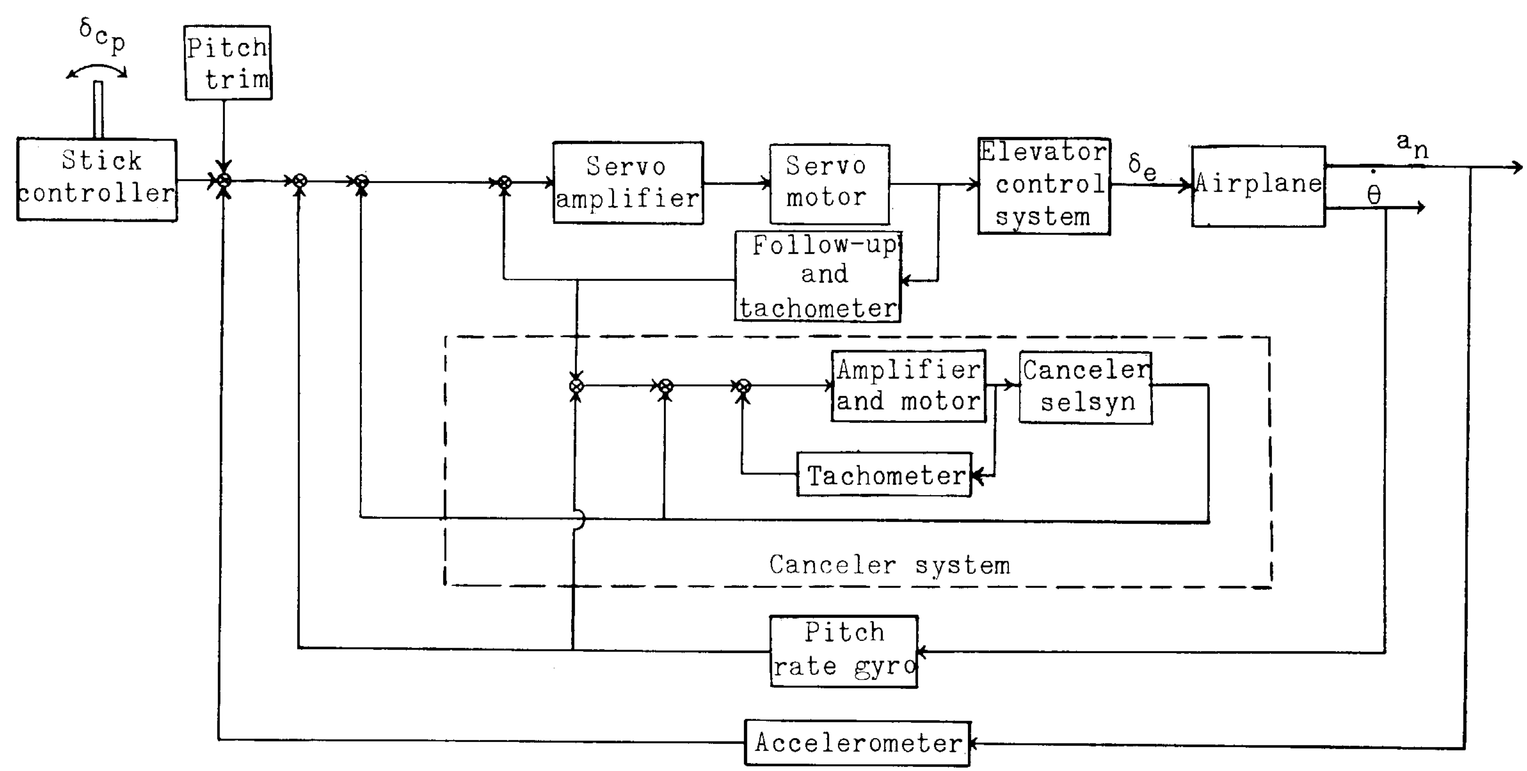


Figure 3.- Block diagram of normal-acceleration control system.

L-94832.1

Figure 4.- Photograph of side-located control stick.

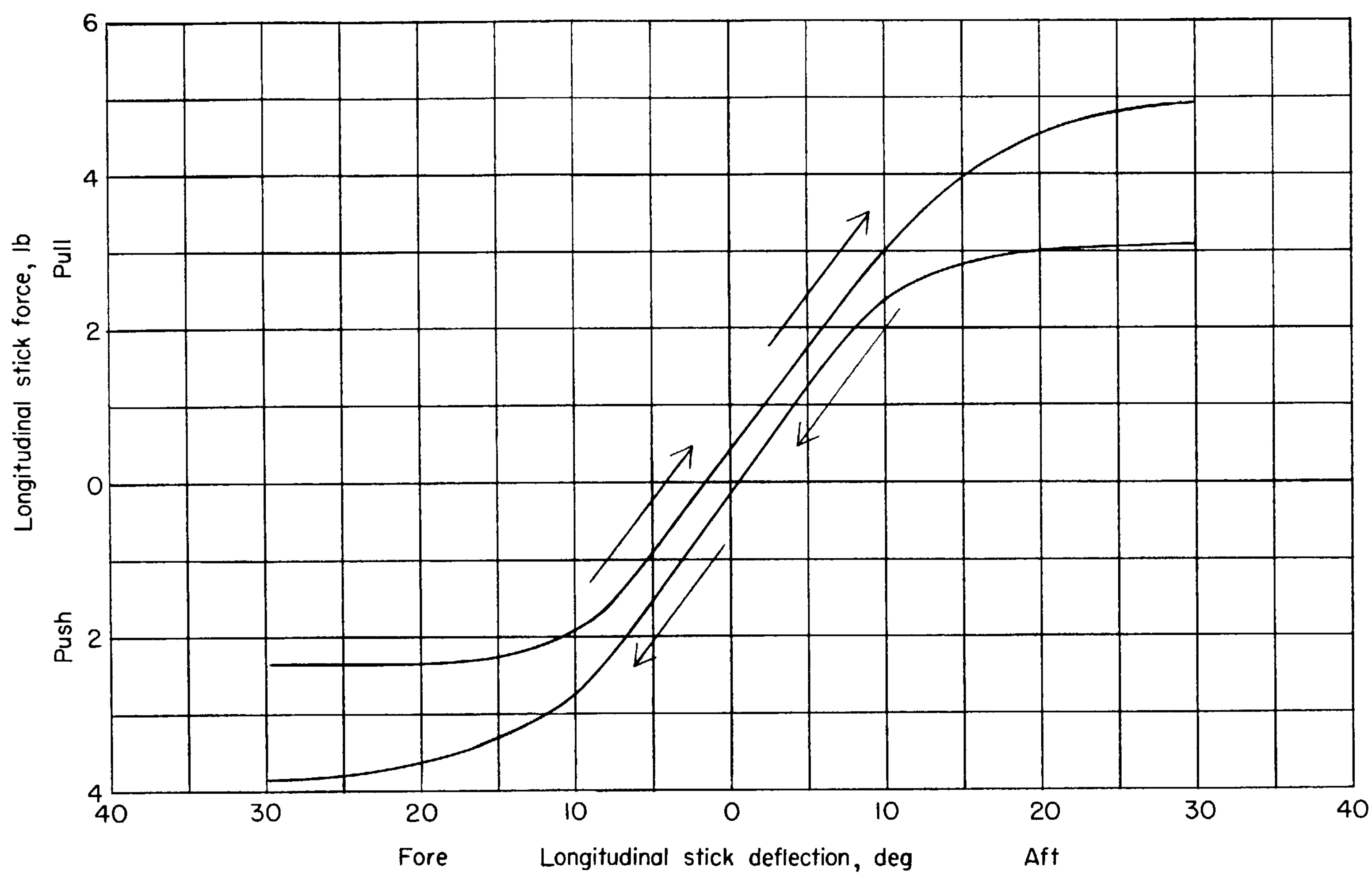
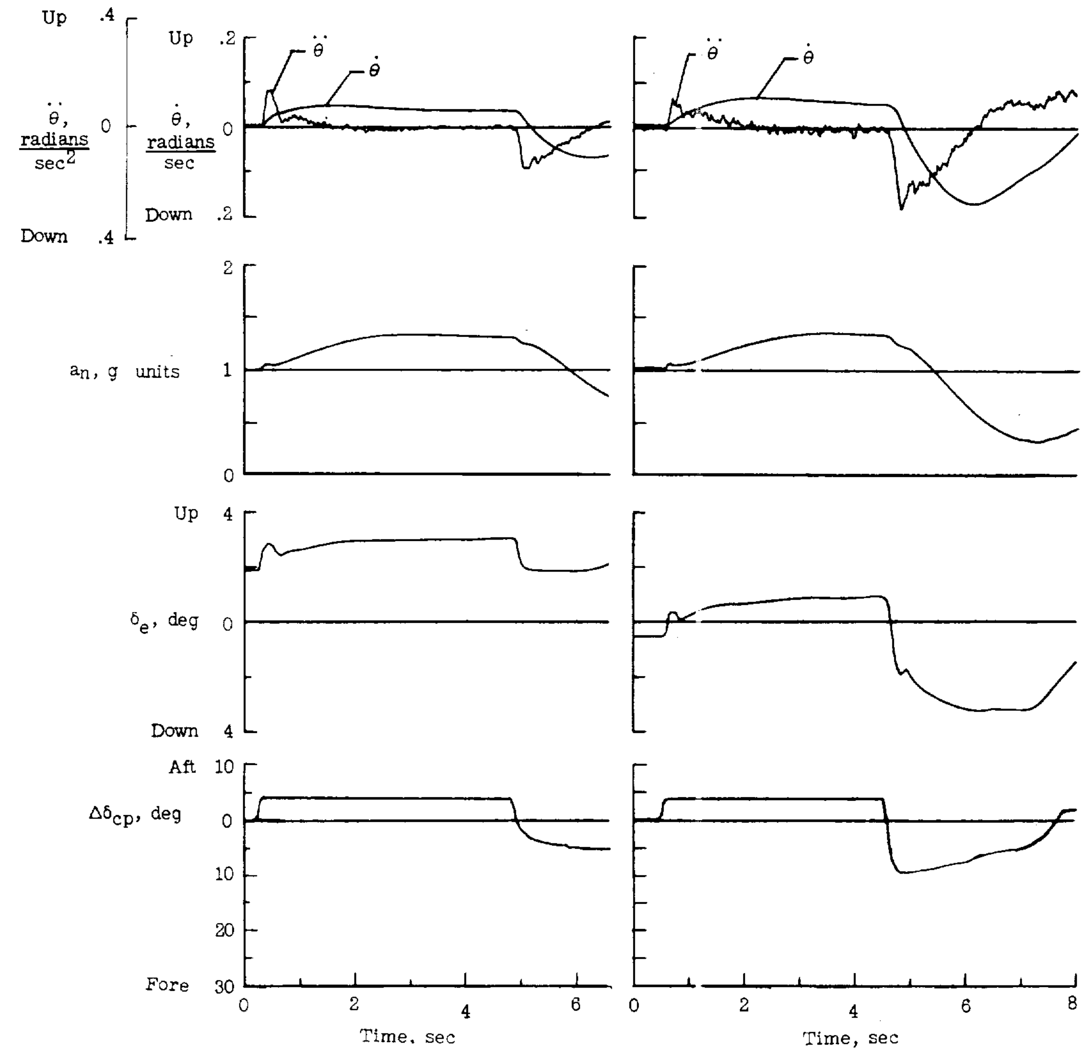


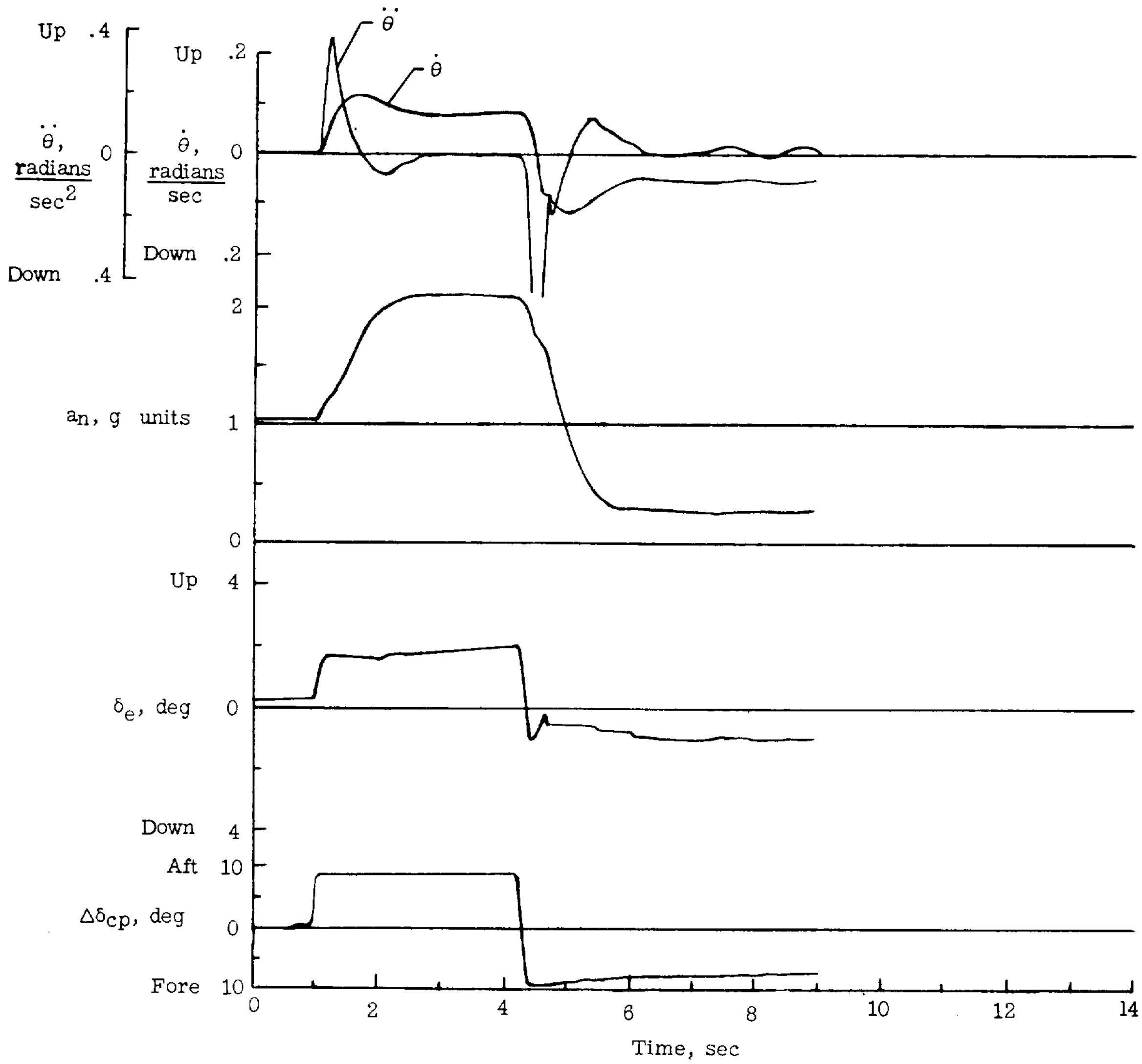
Figure 5.- Force-deflection characteristics of side-located control stick. Force is based on a 2.75-inch moment arm.



(a) Clean condition:
V_i = 157 knots;
h_p = 10,000 feet;
K_{a_n} = 1.35 volts/g;
$K_{\dot{\theta}}$ = 6.5 volts/radian/sec;
$K_{\delta_{cp}}$ = 0.62 volt/deg;
K_{f_e} = 7 volts/radian.

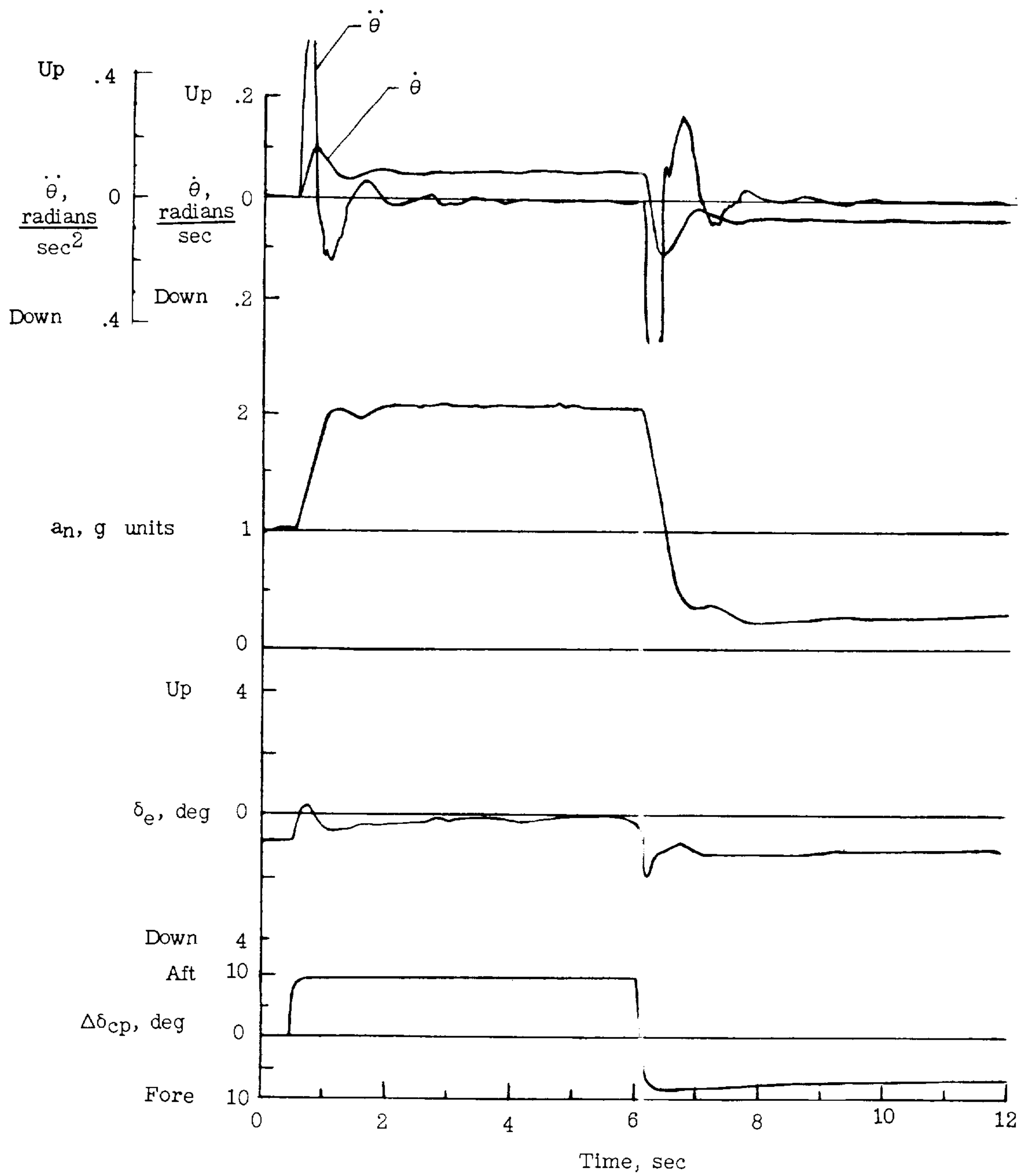
(b) Landing-approach condition:
V_i = 148 knots;
h_p = 10,000 feet;
K_{a_n} = 1.35 volts/g;
$K_{\dot{\theta}}$ = 6.5 volts/radian/sec;
$K_{\delta_{cp}}$ = 0.162 volt/deg;
K_{f_e} = 7 volts/radian.

Figure 6.- Transient responses in pitch of airplane normal-acceleration control system combination. Center of gravity approximately 27.5 percent of mean aerodynamic chord.



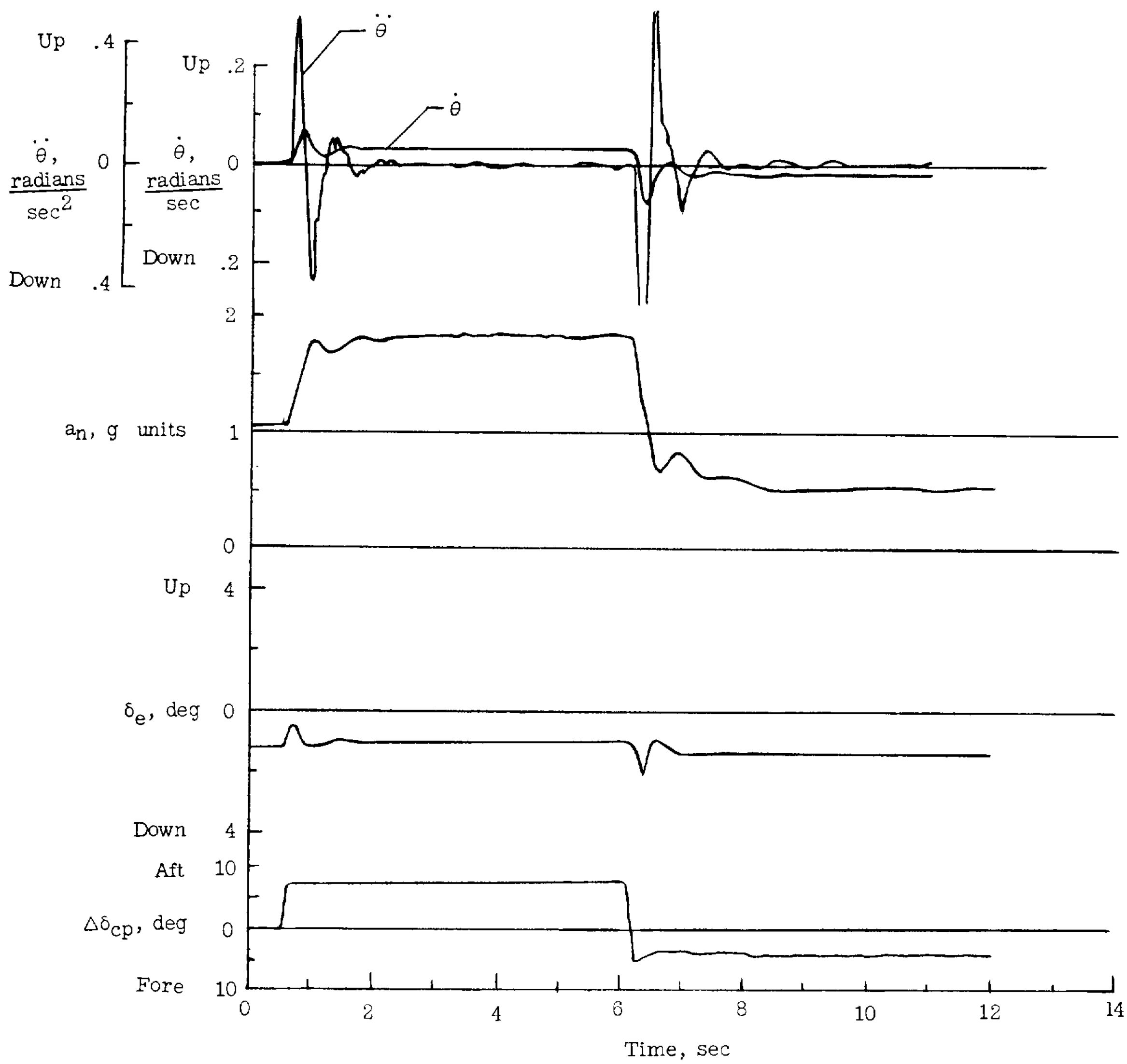
(c) Clean condition: V_i = 216 knots; M = 0.39; h_p = 10,000 feet; K_{a_n} = 1.35 volts/g; $K_{\dot{\theta}}$ = 6.5 volts/radian/sec; $K_{\delta_{cp}}$ = 0.162 volt/deg; K_{f_e} = 7 volts/radian.

Figure 6.- Continued.



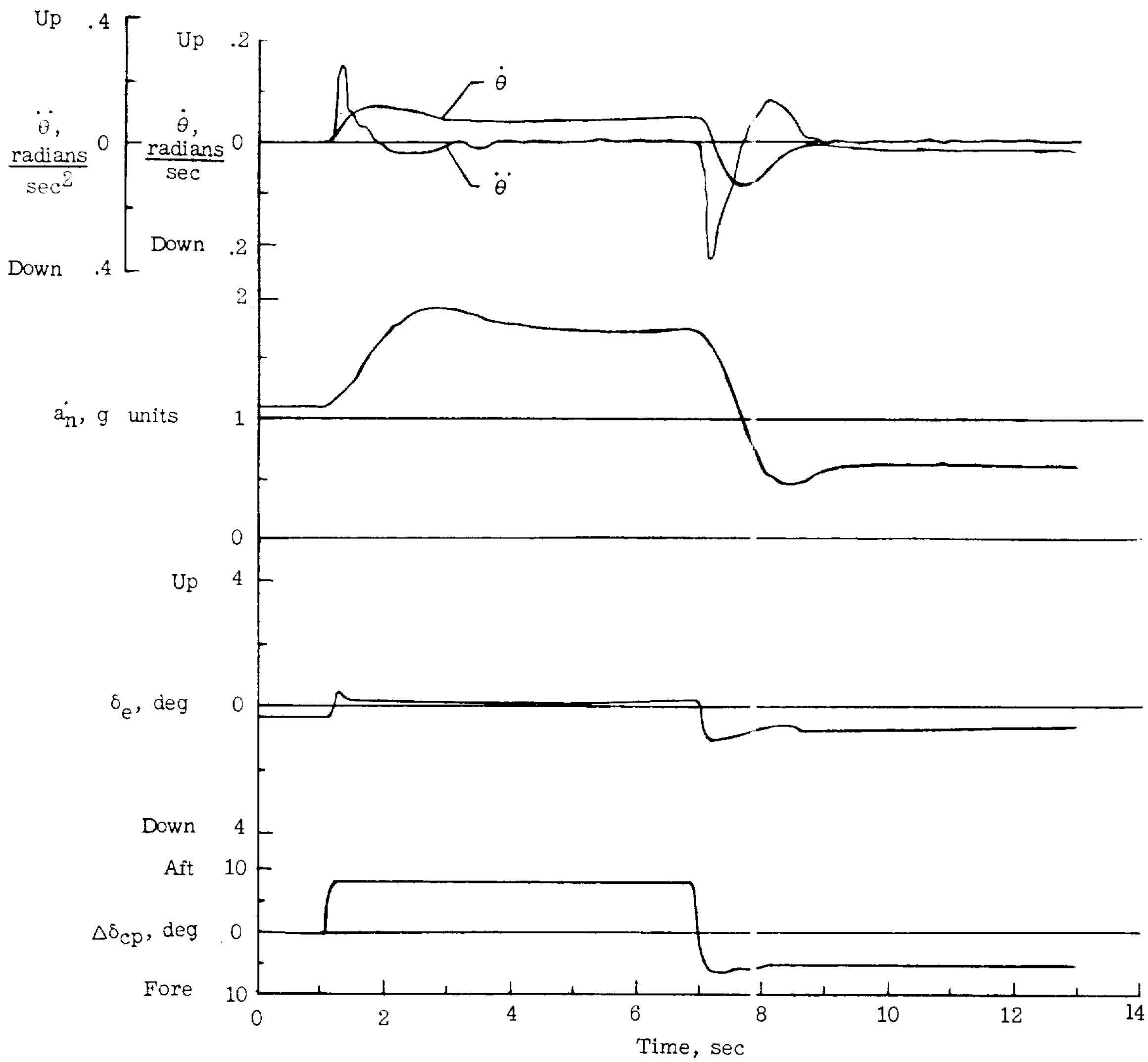
(d) Clean condition: V_i = 327 knots; M = 0.6; h_p = 10,000 feet; K_{a_n} = 1.35 volts/g; $K_{\dot{\theta}}$ = 6.5 volts/radian/sec; $K_{\delta_{cp}}$ = 0.162 volt/deg; K_{f_e} = 7 volts/radian.

Figure 6.- Continued.



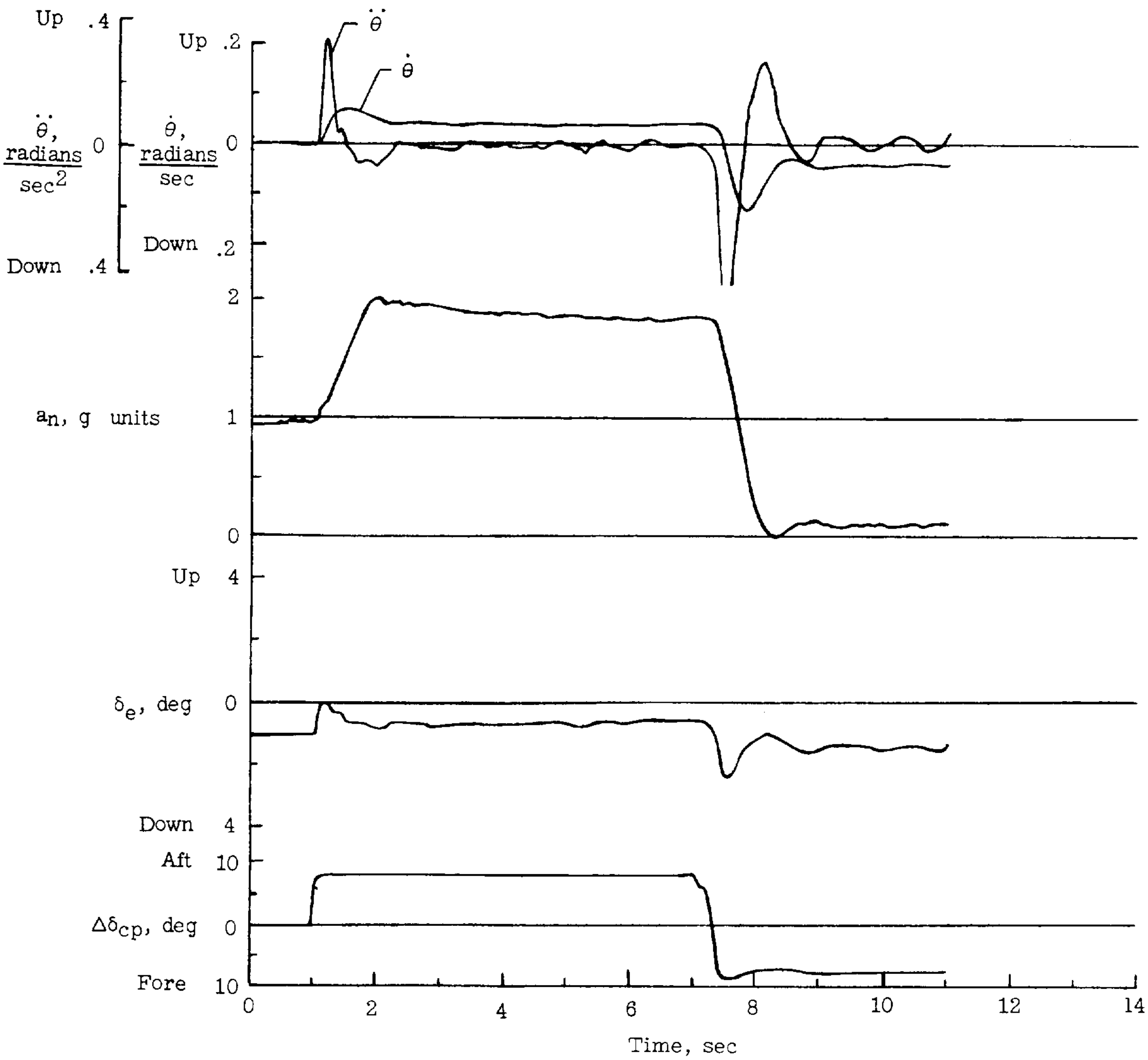
(e) Clean condition: V_i = 384 knots; M = 0.69; h_p = 10,000 feet; K_{a_n} = 1.35 volts/g; $K_{\dot{\theta}}$ = 6.5 volts/radian/sec; $K_{\delta_{cp}}$ = 0.162 volt/deg; K_{f_e} = 7 volts/radian.

Figure 6.- Continued.



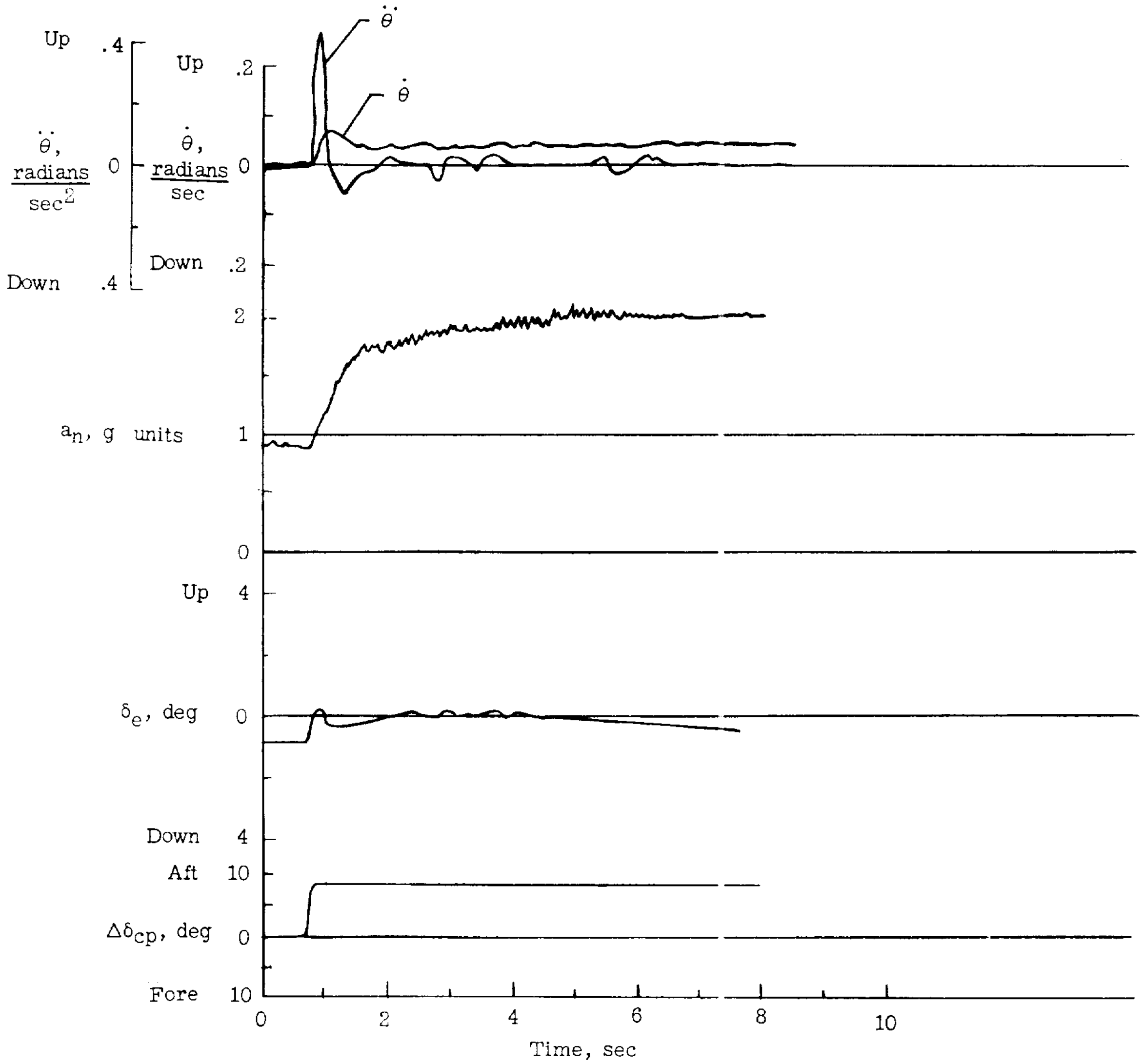
(f) Clean condition: V_i = 227 knots; M = 0.6; h_p = 30,000 feet; K_{a_n} = 1.35 volts/g; $K_{\dot{\theta}}$ = 6.5 volts/radian/sec; $K_{\delta_{cp}}$ = 0.127 volt/deg; K_{f_e} = 7 volts/radian.

Figure 6.- Continued.



(g) Clean condition: V_i = 270 knots; M = 0.7; h_p = 30,000 feet; K_{a_n} = 1.35 volts/g; $K_{\dot{\theta}}$ = 6.5 volts/radian/sec; $K_{\delta_{cp}}$ = 0.162 volt/deg; K_{f_e} = 7 volts/radian.

Figure 6.- Continued.



(h) Clean condition: V_i = 289 knots; M = 0.76; h_p = 30,000 feet; K_{a_n} = 1.13 volts/g; $K_{\dot{\theta}}$ = 6.5 volts/radian/sec; $K_{\delta_{cp}}$ = 0.127 volt/deg; K_{f_e} = 7 volts/radian.

Figure 6.- Concluded.

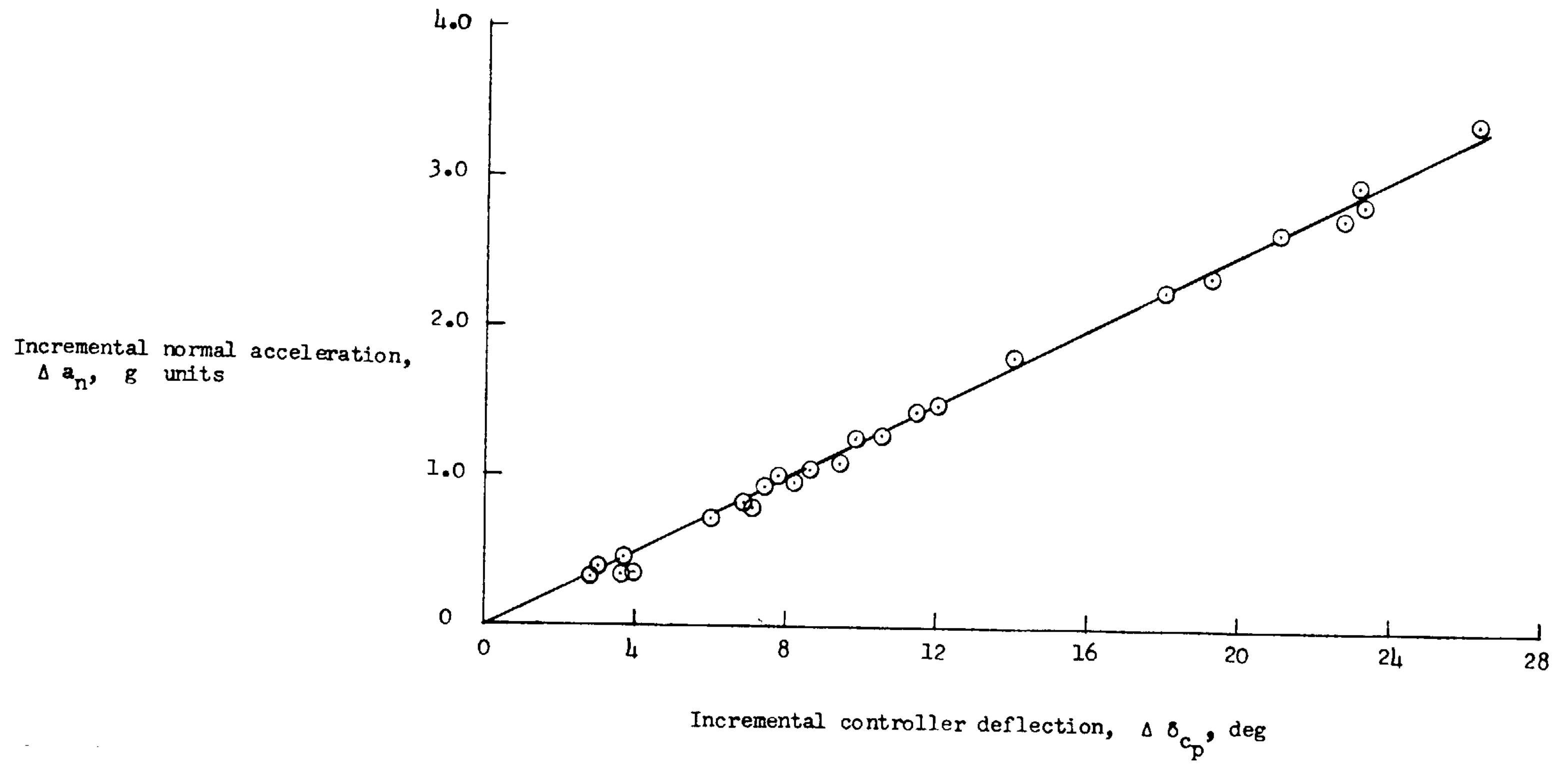
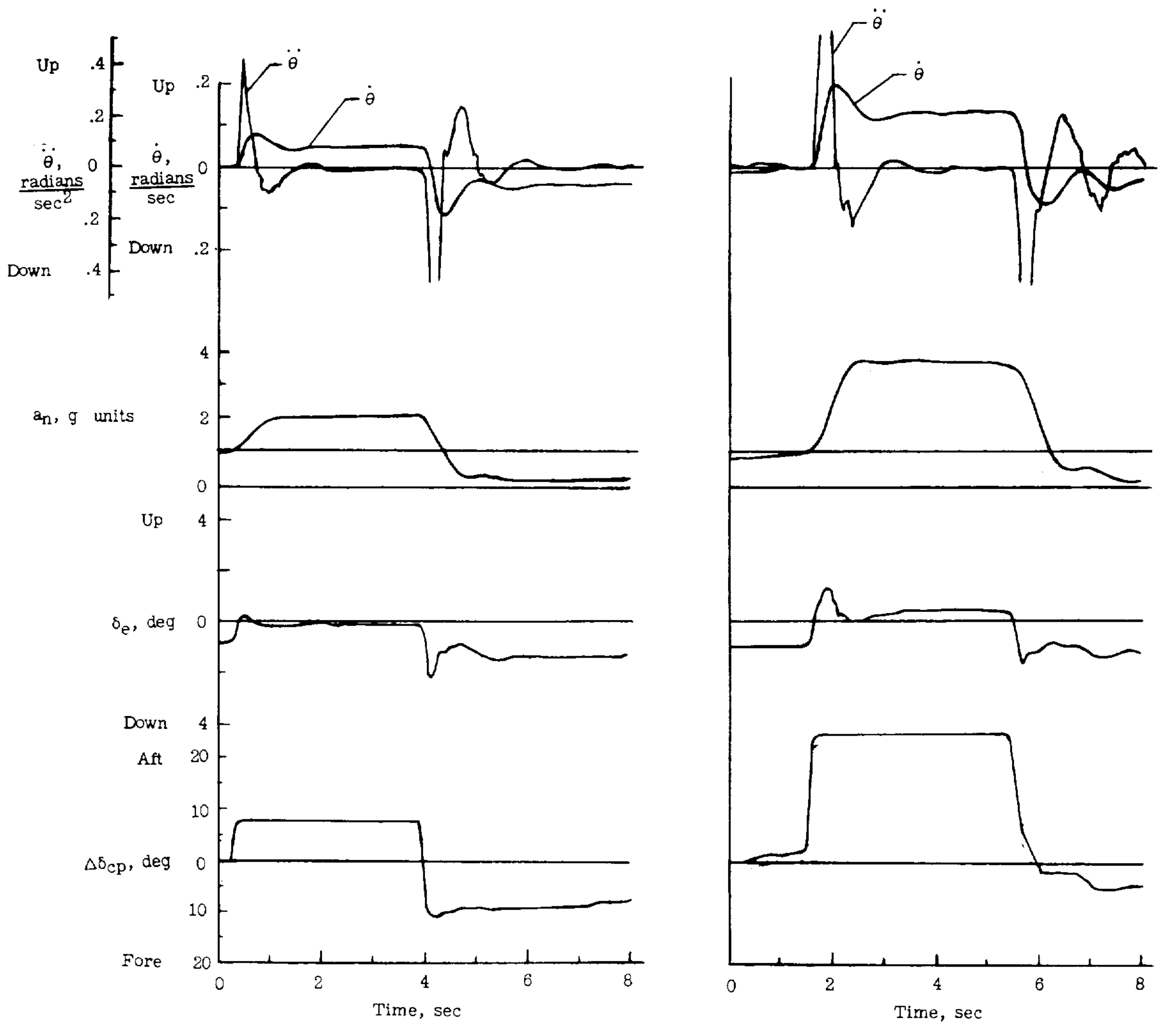


Figure 7.- Variation of steady normal acceleration with side controller deflection. Center of gravity approximately 27.5 percent mean aerodynamic chord. Normal-acceleration control system gains: K_{a_n} = 1.35 volts/g; $K_{\dot{\theta}}$ = 6.5 volts/radian/sec; $K_{\delta_{cp}}$ = 0.162 volt/deg; K_{f_e} = 7 volts/radian.



(a) Small amplitude input.
V_i = 285 knots; M = 0.57.

(b) Large amplitude input.
V_i = 308 knots; M = 0.61.

Figure 8.- Transient responses in pitch of airplane normal-acceleration control system combination for two magnitudes of input signal. Clean condition: h_p = 15,000 feet, center of gravity approximately 27.5 percent of mean aerodynamic chord; K_{a_n} = 1.35 volts/g; $K_{\dot{\theta}}$ = 6.5 volts/radian/sec; $K_{\delta_{cp}}$ = 0.162 volt/degree; K_{f_e} = 7 volts/radian.

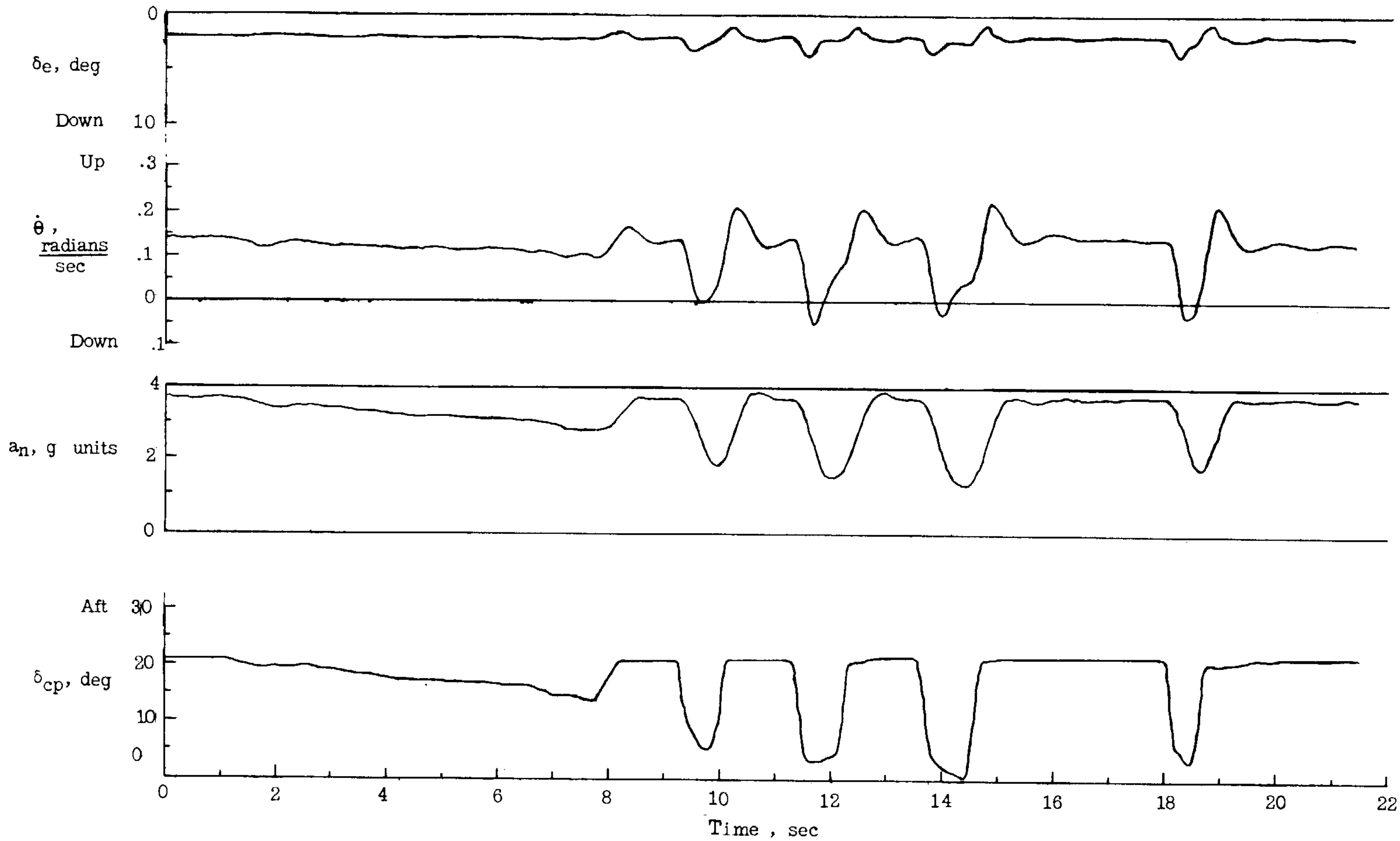


Figure 9.- Time history showing normal acceleration limiting during a turn. Clean condition: V_i = 340 knots; M = 0.67; h_p = 15,000 feet; center of gravity approximately 27.5 percent mean aerodynamic chord; K_{a_n} = 1.35 volts/g; $K_{\dot{\theta}}$ = 6.5 volts/radian/sec; $K_{\delta_{cp}}$ = 0.162 volt/deg; K_{f_e} = 7 volts/radian.

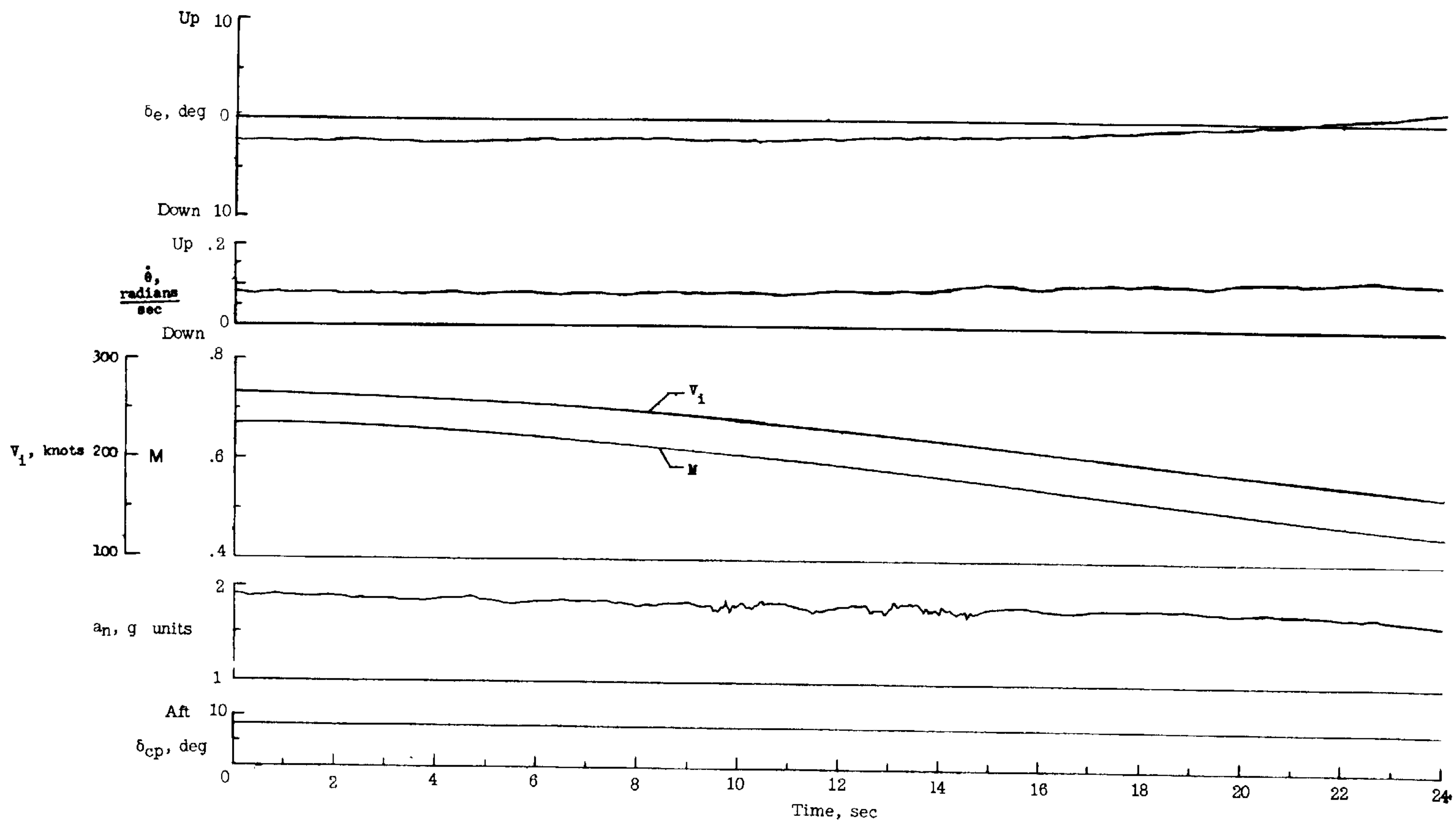
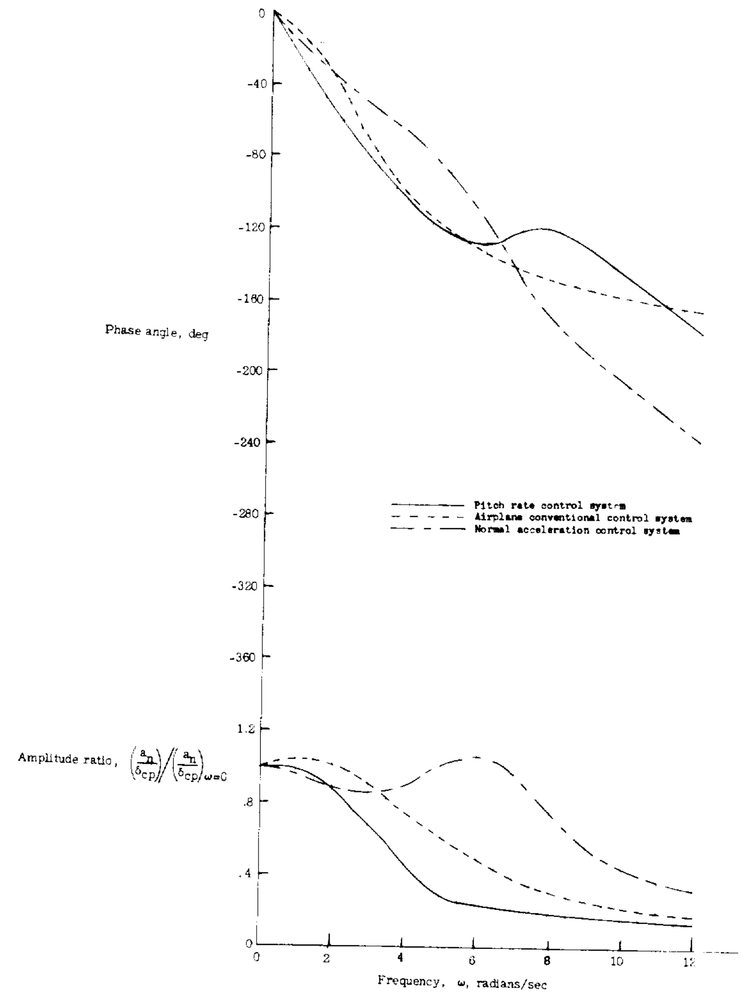
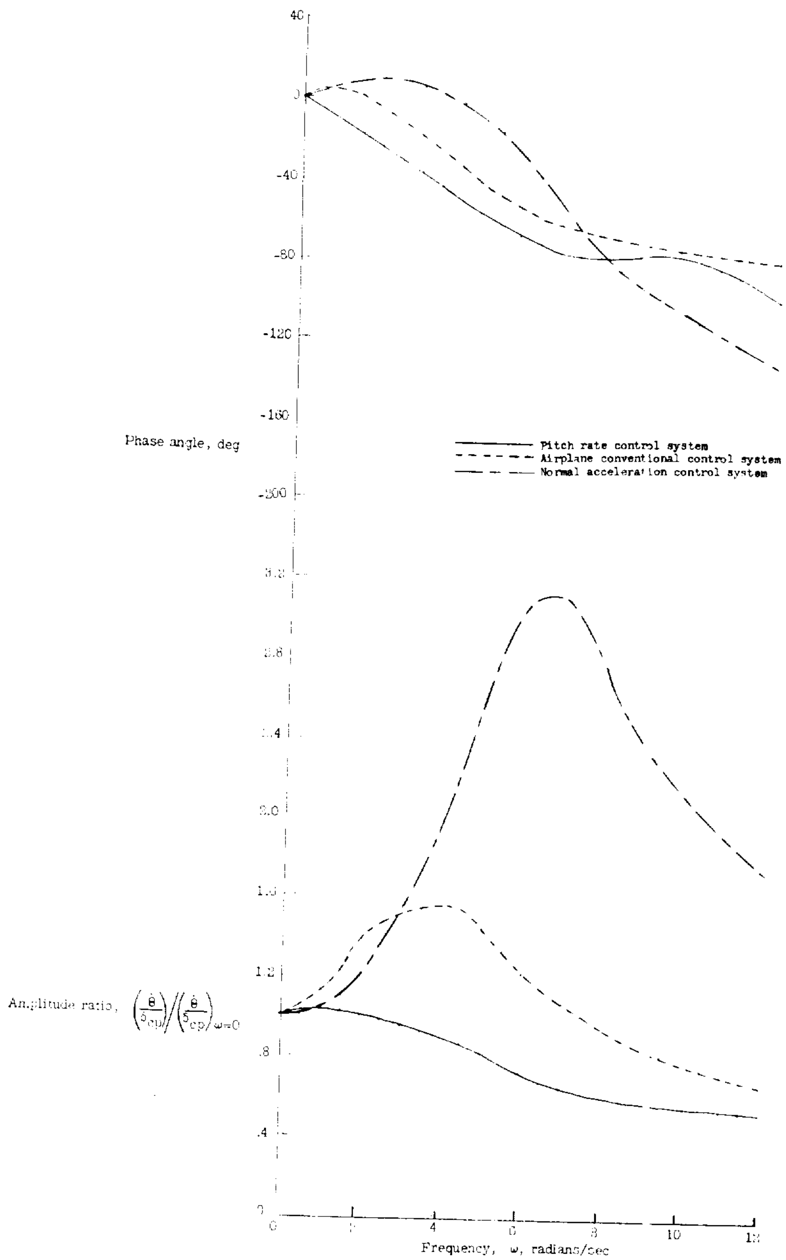


Figure 10.- Time history of a turn at decreasing Mach number with the side-located control stick held fixed. Clean condition: h_p = 30,000 feet; center of gravity approximately 27.5 percent mean aerodynamic chord; K_{a_n} = 1.35 volts/g; $K_{\dot{\theta}}$ = 6.5 volts/radian/sec; $K_{\delta_{cp}}$ = 0.162 volt/deg; K_{f_e} = 7 volts/radian.



$$(a)\ \frac{\left(\frac{a_n}{\delta_{cp}}\right)}{\left(\frac{a_n}{\delta_{cp}}\right)_{\omega=0}}.$$

Figure 11.- Frequency responses in pitch with the normal-acceleration control system, the rate control systems of reference 2, and the airplane conventional control system. Clean condition; V_i = 327 knots; M = 0.6; h_p = 10,000 feet; center of gravity approximately 27.5 percent mean aerodynamic chord. Normal-acceleration control system gains; K_{a_n} = 1.35 volts/g; $K_{\dot{\theta}}$ = 6.5 volts/radian/sec; $K_{\delta_{cp}}$ = 0.162 volt/deg; K_{f_e} = 7 volts/radian.



(b) $\dfrac{\left(\dfrac{\dot{\theta}}{\delta_{cp}}\right)}{\left(\dfrac{\dot{\theta}}{\delta_{cp}}\right)_{\omega=0}}$.

Figure 11.- Concluded.

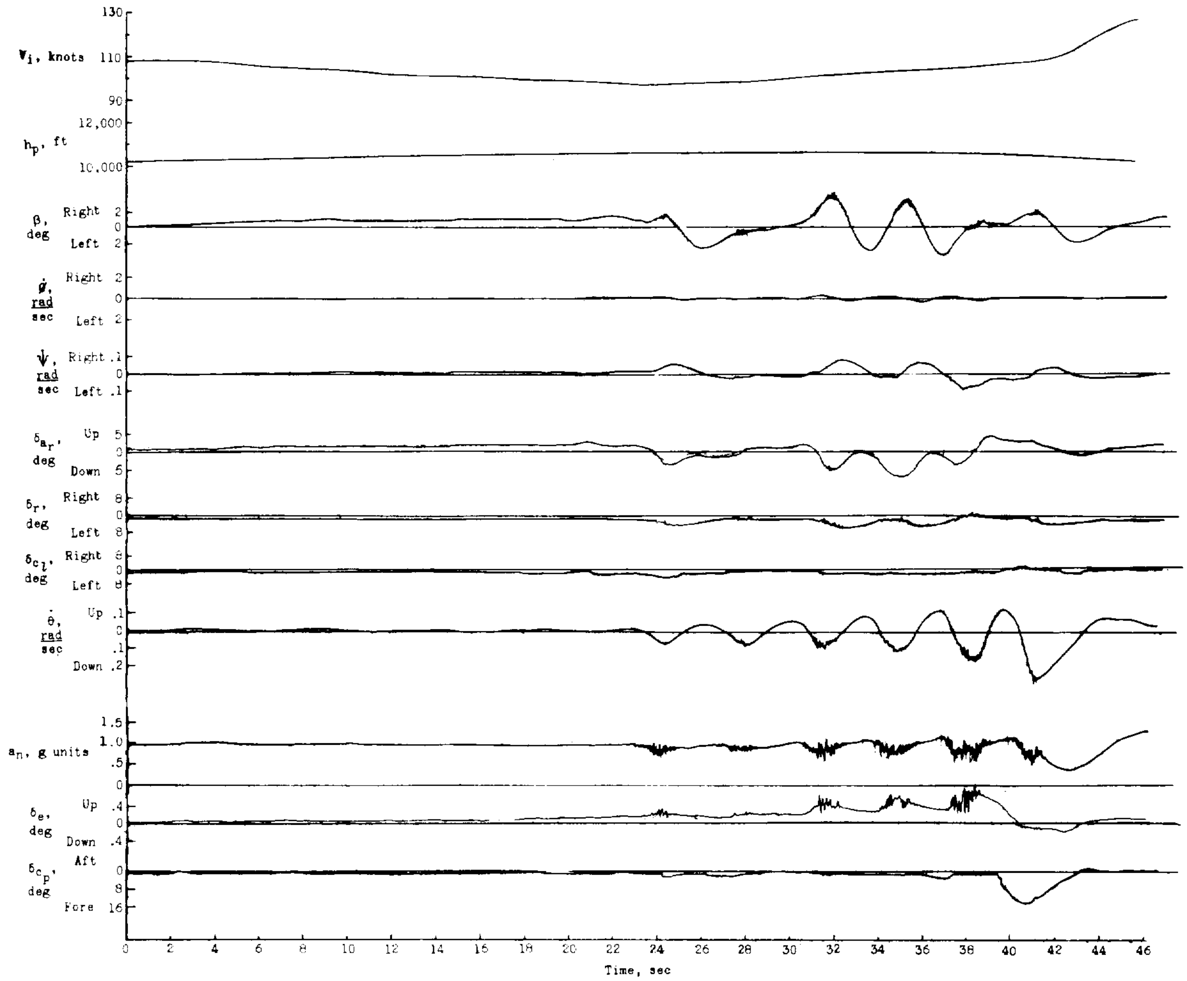


Figure 12.- Time history of a 1 g stall in the power approach condition. Center of gravity approximately 27.5 percent mean aerodynamic chord; K_{a_n} = 1.35 volts/g; $K_{\dot{\theta}}$ = 6.5 volts/radian/sec; $K_{\delta_{cp}}$ = 0.162 volt/deg; K_{f_e} = 7 volts/radian.

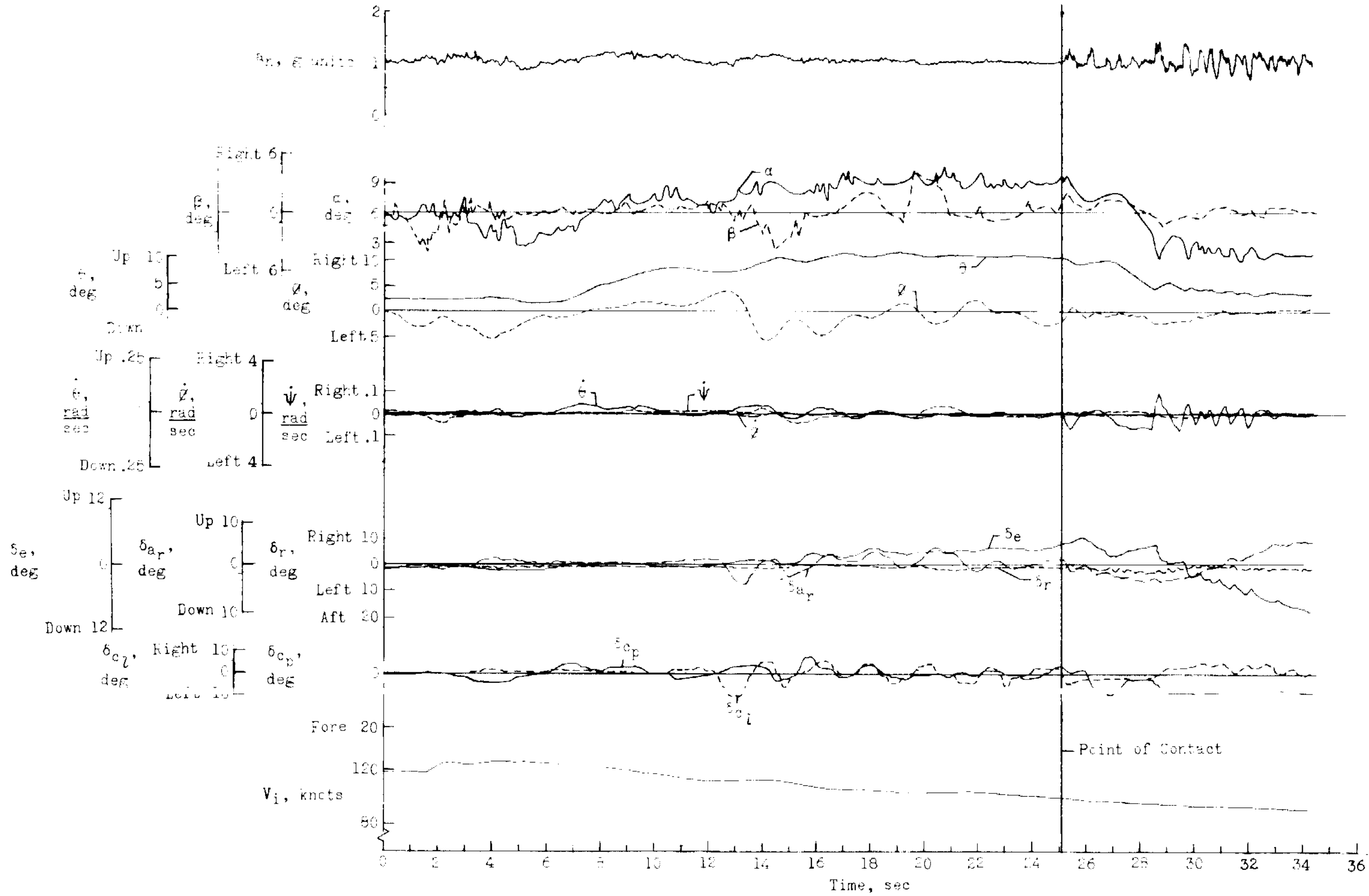


Figure 15.- Time history of a landing. Center of gravity approximately 27.5 percent mean aerodynamic chord; K_{a_n} = 1.35 volts/g; $K_{\dot{\theta}}$ = 6.5 volts/radian/sec; $K_{\delta_{cp}}$ = 0.162 volt/deg; K_{f_e} = 7 volts/radian.

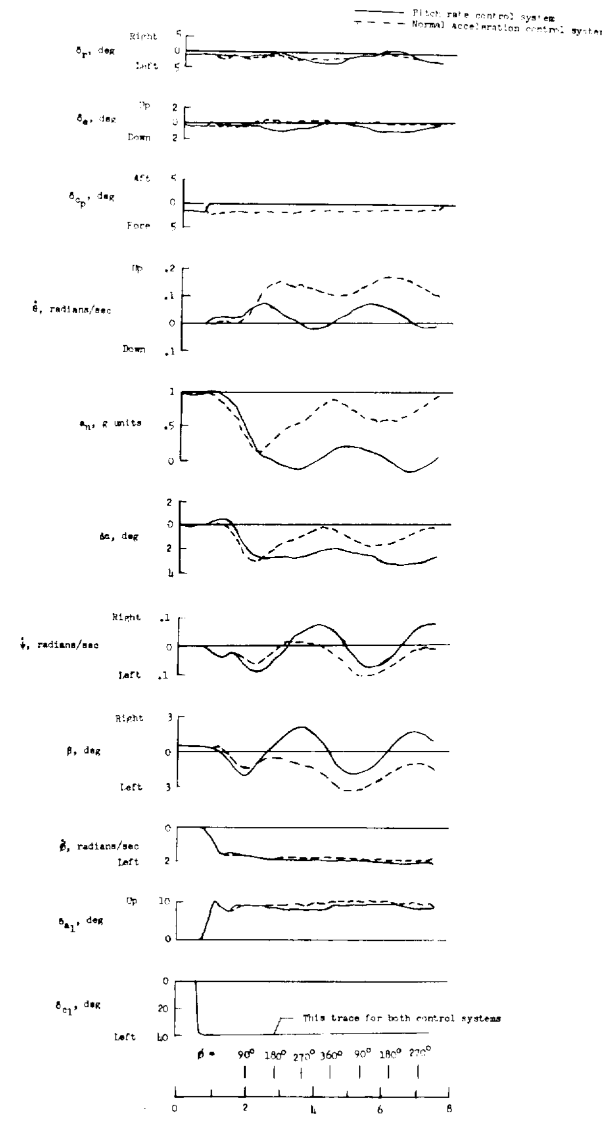


Figure 16.- Time histories of aileron rolls with the airplane normal-acceleration control system and a pitch-rate control system. Clean condition; V_i = 225 knots; M = 0.61; h_p = 30,000 feet; center of gravity approximately 27.5 percent of mean aerodynamic chord. Normal-acceleration control system gains; K_{a_n} = 1.35 volts/g; $K_{\dot{\theta}}$ = 6.5 volts/radian/sec; $K_{\delta_{cp}}$ = 0.162 volt/deg; K_{f_e} = 7 volts/radian. Pitch-rate control system gains: $K_{\dot{\theta}}$ = 11.6 volts/radian/sec; $K_{\delta_{cp}}$ = 0.162 volt/deg.