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STABILITY CHARACTERISTICS OF A TANDEM-ROTOR TRANSPORT HELICOPTER AS DETERMINED BY FLIGHT TEST

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SUMMARY

Selected, unaugmented stability characteristics of a modern tandem-rotor transport helicopter were determined by a flight investigation. The angle-of-attack instability was the predominant factor which resulted in unacceptable maneuver stability characteristics. Also present were speed and directional instabilities.

Based on pilots' comments, the current V/STOL specifications concerning handling qualities appeared applicable to a helicopter of this size and configuration. Theoretical calculations of the pitch and roll damping showed good agreement with flight measurements.

INTRODUCTION

The advantage of having information available on the unaugmented stability characteristics of a specific aircraft configuration is well recognized. This information has several applications; it provides designers with an indication of the inherent stability characteristics of a particular configuration and, when the physical characteristics of the aircraft are available, it supplies a source of data for comparisons with theory. Where comparisons with theory accompany the data, the designer can obtain an insight into the degree of correlation expected from applying a particular theory to similar configurations. Knowledge of the basic stability characteristics of a given configuration also provides preliminary information as to the degree of artificial stabilization required to provide satisfactory handling characteristics.

The helicopter configuration used in the present study has been independently evaluated in two previous investigations (refs. 1 and 2); however, the emphasis in both references was placed on evaluation of the stability augmentation system (SAS) installed in the aircraft. As a result, only limited SAS-off stability data were presented (ref. 2). In an attempt to fill the existing void, this report presents additional SAS-off stability characteristics and a relatively complete listing of the pertinent physical characteristics of the configuration.

The helicopter which was evaluated in references 1 and 2 was obtained by the Langley Research Center for use as a variable-stability aircraft. Prior to the application of the computer model simulation technique (described in ref. 3) to the variable-stability helicopter, a brief investigation of the basic unaugmented stability characteristics of the helicopter was made in what were considered to be possibly critical areas. The results of this investigation have been documented and are presented herein. Although it is not within the scope of this report to present a comprehensive theoretical treatment of the stability characteristics of the configuration investigated, the pitch and roll damping obtained from flight data have been compared with theory. Pilots' comments on the handling characteristics have also been included where appropriate in order to determine whether existing handling-qualities criteria are applicable to a helicopter of this size and configuration. It should be emphasized that the pilots' comments refer to the basic unaugmented characteristics since the dual stability augmentation system installed in the helicopter was not engaged during the present flight investigation.

TEST HELICOPTER AND INSTRUMENTATION

Helicopter

The test helicopter (fig. 1) is a modern twin-turbine tandem-rotor configuration. A three-view drawing is shown in figure 2. Physical dimensions and characteristics are given in table I.

Control moments are generated in the following manner: Pitching moments are created by longitudinal displacement of the center stick to produce differential collective pitch on the front and rear rotors; rolling moments, by lateral displacement of the center stick to introduce lateral cyclic pitch to both rotors; and yawing moments, by pedal inputs to produce differential lateral cyclic pitch of the front and rear rotors.

The aircraft is equipped with a dual stability augmentation system (SAS) which is employed to improve the basic handling characteristics. Since the SAS was not engaged during the present investigation, it is not described.

Instrumentation

The helicopter was instrumented to record angular velocities, angular accelerations, and linear accelerations about the principal inertia axes. Control positions, airspeed, altitude, and rotor rotational speed were also recorded. The sideslip-angle and angle-of-attack sensors were boom-mounted (fig. 3) and provided reliable information above approximately 25 knots; below this speed the sensors were affected by the rotor downwash. Standard NASA recorders equipped with synchronized timers were employed.

RESULTS AND DISCUSSION

Sensitivity and Damping

The sensitivity (initial angular acceleration per inch of control) and angular-velocity damping-to-inertia ratios about the pitch, roll, and yaw axes were determined for the hovering-flight condition. Several step inputs were made independently for each axis and the pilots' control inputs and resultant angular velocities were simultaneously recorded. Typical response time histories are shown in figures 4, 5, and 6 for the pitch, roll, and yaw axes, respectively. The pitch and roll time histories in figures 4 and 5 indicate that the step input is preceded by a slight control input in the opposite direction. The initial control displacement is an intentional input referred to as a "false start" and is used to obtain a step input of longer duration and to minimize linear velocity effects. This practice yields more precise results.

Standard methods were used in arriving at the sensitivity and the damping-to-inertia ratios, and the results given in table II represent the average values obtained from several step inputs in the pitch, roll, and yaw axes. For convenient reference, table II includes sensitivity and damping requirements obtained from references 4 and 5. Table II also includes the yaw control power requirements. (Control power is defined as the maximum angular acceleration which can be produced from a trimmed flight condition.) The damping requirements given by references 4 and 5 are a function of helicopter inertia only and are therefore readily obtainable. On the other hand, the sensitivity requirements are expressed in terms of an angular-displacement requirement in a given time interval following a 1-inch input from trim (control power is given in similar terms). For this report the angular-displacement requirement was converted to an angular-acceleration requirement by the following equation which assumes a first-order system (that is, a system containing only a mass and a damper):

$$\frac{M_\delta}{I} = \theta_1 \left[\frac{\left(\frac{M_{\dot\theta}}{I}\right)^2}{-\frac{M_{\dot\theta}}{I} t_1 + e^{\frac{M_{\dot\theta}}{I} t_1} - 1} \right]$$

where

$\dfrac{M_\delta}{I}$ sensitivity, (moment per unit control to moment of inertia), required to produce the given displacement

$\dfrac{M_{\dot\theta}}{I}$ angular-velocity damping-to-inertia ratio (negative values indicate stable damping)

t_1 given time (specified in refs. 4 and 5)

θ_1 given angular displacement after time t_1 (specified in refs. 4 and 5)

It should be emphasized that the value specified in references 4 and 5 for the angular-velocity damping-to-inertia ratio (M_θ/I) is used rather than the measured value.

When inputs were made in the longitudinal direction, the pilot stated that the helicopter was "touchy," especially in forward flight. Although many factors enter into this characteristic, the main contribution in the present case is believed to be the high longitudinal control sensitivity, which is indicated by the measured value presented in table II. The pitch-sensitivity requirements given in references 4 and 5 are minimum values, and no mention is made of a maximum allowable sensitivity. Since the helicopter appears to be approaching some sort of maximum it seems desirable that future criteria should consider a limitation on the maximum allowable pitch sensitivity. The fact that the helicopter became more sensitive in forward flight is attributed primarily to the angle-of-attack instability, which is discussed in a subsequent section.

The pilot commented that the lateral response was somewhat high but satisfactory. Reference 4 states that the lateral control effectiveness (used in this context to mean the angular roll rate) shall be considered excessive if the maximum rate of roll per inch of stick displacement is greater than 20 degrees per second. By assuming that the aircraft response is described by a first-order system, the steady-state angular-rate capability about the roll axis, obtained by dividing the lateral control sensitivity by the damping-to-inertia ratio, is approximately 30 degrees per second. Reduction of the lateral control sensitivity or an increase in damping would be necessary to reduce the steady-state rate capability to less than 20 degrees per second. It is interesting to note that if the roll damping-to-inertia value met the minimum visual-flight requirement of reference 4, the roll-rate capability for the existing roll sensitivity would be slightly less than 20 degrees per second. This condition implies that the sensitivity in itself is satisfactory and the high response is due primarily to the low damping. The indication that the sensitivity is satisfactory is also substantiated by the results of a recent investigation (ref. 6). In reference 6, for the minimum damping-to-inertia value required by AGARD in reference 5 (which is more than twice the damping-to-inertia ratio of the basic unaugmented helicopter), approximately the same sensitivity as that of the present helicopter was investigated and found to be satisfactory.

When inputs were made to the yaw axis, the pilot stated that even though the helicopter was relatively more powerful and had better directional control than earlier tandem-rotor helicopters, he would prefer at least twice the existing control power. From the comparison between the measured yaw control power and the yaw control power requirements of references 4 and 5 (see table II), it can be seen that the requirements for yaw control power are from approximately $1\frac{1}{2}$ to $2\frac{1}{2}$ times the measured value depending upon which requirement is used. The investigation reported in reference 6 indicates that the minimum yaw control power requirement for maneuvering under visual flight conditions should be somewhere between the requirements of MIL-H-8501A (ref. 4) and AGARD (ref. 5). The pilots' comments obtained during the present investigation tend to support this view.

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Comparison of Calculated Pitch and Roll Damping

With Measured Values

Theoretical pitch and roll damping-to-inertia ratios were calculated for the helicopter having a gross weight of 15 500 pounds and a center of gravity located approximately 15 inches forward of the center line between the rotors. In the past, pitch-damping theory for tandem-rotor helicopters that neglected the change in induced velocity due to the vertical velocity of the rotor disks was reported to provide a good estimate of the pitch damping for an overlapped tandem helicopter configuration. This theory, when applied to the present configuration, resulted in a calculated pitch damping which was three times the actual value. Consequently, the pitch damping was calculated by the theory presented in reference 7 in which the induced-velocity effect was included. Although intended primarily for tail-rotor studies, reference 7 is also applicable to studies of the pitching response of a tandem-rotor helicopter. The computed pitch damping-to-inertia ratio of $-0.60 \frac{rad/sec^2}{rad/sec}$ was found to be in good agreement with the measured value of $-0.50 \frac{rad/sec^2}{rad/sec}$. The damping-to-inertia ratio about the roll axis was computed by the method of reference 8. However, since the method derived in reference 8 was for rotors having flapping hinges on the rotor shaft, it was necessary to add the damping contribution due to the offset flapping hinges of the present configuration. The computed roll damping-to-inertia ratio was found to be $-0.82 \frac{rad/sec^2}{rad/sec}$, which agreed closely with the measured value of $-0.76 \frac{rad/sec^2}{rad/sec}$.

Speed Stability

The speed stability of the helicopter was investigated for a wide range of airspeeds at six different power settings. The speed range from 30 to 105 knots was covered by all six power conditions. Data were obtained for airspeeds as low as 25 knots for three power conditions and as high as 145 knots for one power condition. These data are presented in figure 7, which shows the variation of longitudinal stick position with airspeed. The figure indicates that the aircraft is unstable with speed for all conditions investigated. One pilot stated that the instability with speed was annoying since constant retrimming was necessary to hold a given airspeed; nevertheless, the pilot did not consider the instability dangerous because a sufficient control margin for maneuvering existed at all trim speeds.

Angle-of-Attack Stability

Measured.- An attempt was made to measure the angle-of-attack stability of the aircraft by the procedure described in reference 9. The value of the

stability derivative determined by this method is approximately $0.8 \frac{rad/sec^2}{rad}$, and indicates that the aircraft is unstable. It should be noted, however, that in arriving at this value a severe limitation in the application of the procedure of reference 9 was encountered. One requirement of this technique is that rotor speed and forward speed be varied in direct proportion. However, the speed-governing system installed on the engines of this helicopter limits the range of obtainable rotor speeds which, in turn, limits the change in forward speed allowed for the measurement. Consequently, the angle-of-attack stability could not be determined with any degree of precision.

Pull-and-hold maneuver.- The pull-and-hold maneuver was employed to determine the maneuver stability characteristics of the helicopter, and to provide a qualitative measure of the angle-of-attack instability. Several pull-and-hold maneuvers were performed at trim airspeeds ranging from approximately 40 to 85 knots. During the maneuvers, time histories of the longitudinal stick position, pitching angular velocity, and normal acceleration were recorded and are presented in figure 8 for trim airspeeds from 50 to 80 knots. The time history of normal acceleration is divergent during the maneuver for all trim airspeeds. With a given input, the time history of pitching angular velocity does not appear to be reaching any maximum rate for any of the conditions investigated.

The pilots commented that these maneuver stability characteristics were definitely unsatisfactory. One pilot stated that these characteristics would be dangerous at higher trim airspeeds (about 120 knots and up) especially if the pilot was flying on instruments, since the aircraft could "get away" from the pilot before he could initiate corrective action. Current requirements for acceptable maneuver stability characteristics (refs. 4 and 5) state, in part, that the time history of normal acceleration shall become concave downward within 2 seconds following the start of the maneuver and remain concave downward until the attainment of maximum acceleration. Based on the time histories presented in figure 8, the characteristics shown do not satisfy the current requirements. Since the pilots considered these maneuver stability characteristics unacceptable, the current requirements appear to be adequate in this area.

Longitudinal Trim Change With Power

The longitudinal trim change with power was measured at a forward speed of 20 knots. The results of this measurement are given in figure 9 where longitudinal stick position is plotted against vertical velocity; the vertical velocity is used as a measure of power.

The pilot commented that the magnitude of the trim change (approximately 2.3 inches) was acceptable under the conditions investigated. The direction of the trim change, however, would definitely be undesirable for accomplishing certain tasks since the helicopter pitched up with a decrease in power and pitched down with an increase in power in the low-rate-of-descent region

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(<5 ft/sec). This feature was found to be somewhat annoying during a low-speed low-angle instrument landing system approach. For example, when the pilot was above the glide slope while holding a desired airspeed and made a correction by decreasing power, the aircraft pitched up due to the trim change effect. This pitch-up caused the helicopter to flare and thus resulted initially in an even greater deviation from the glide slope and a loss of airspeed. The flare was eventually followed by the desired increase in rate of descent necessary to acquire the glide slope. Present requirements only specify a maximum allowable trim change independent of direction. Based on the pilot comments noted herein, it appears that future criteria should possibly give consideration to the direction as well as the magnitude of a trim change.

Dihedral Effect

The dihedral effect at various flight conditions is presented in figure 10. The variation of lateral control displacement with sideslip angle is approximately linear and indicates that the helicopter is stable at the various flight conditions of this investigation. At any trim airspeed the dihedral effect is essentially independent of power condition; however, there is a definite increase in dihedral effect with an increase in airspeed.

Directional Stability

The results of the directional-stability measurements are given in figure 11 where the variation of pedal position with sideslip angle is presented for several flight conditions. Because of structural limitations on the aircraft the maximum sideslip angles investigated were ±26°. The results indicate that the helicopter is unstable for left sideslip angles and for small right sideslip angles for all the flight conditions. The helicopter may be either stable or unstable for large right sideslip angles, depending upon the power condition and trim airspeed considered. In general, there is a tendency for the helicopter to become more unstable directionally as the speed is increased.

The pilot commented that for visual flight operations the directional stability was unsatisfactory but not necessarily dangerous for the range of sideslip angles covered. However, under instrument conditions, the pilot considered that the directional stability characteristics would be unacceptable. This comment is substantiated by the investigation of reference 10 in that the directional stability and directional angular velocity damping are in the unacceptable region of figure 3 in reference 10. The pilot noted that since the helicopter constantly diverged from a desired heading, it would be very tiresome to fly in the navigation mode even under visual flight conditions.

CONCLUDING REMARKS

An investigation of the unaugmented stability characteristics of a modern tandem-rotor helicopter has been conducted. The predominant characteristic of the configuration is the angle-of-attack instability which results in

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unacceptable maneuver stability characteristics. Also present are speed and directional instabilities, but these instabilities are of a much lesser degree.

Within the flight conditions covered by this investigation, current V/STOL specifications concerning handling qualities appear applicable to a helicopter of this size and configuration. It is shown that a good estimate of the pitch and roll damping is provided by existing theory.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., March 16, 1965.

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5. Anon.: Recommendations for V/STOL Handling Qualities. AGARD Rept. 408, Oct. 1962.

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TABLE I.- PHYSICAL CHARACTERISTICS OF TEST HELICOPTER

Overall dimensions (blade turning):
  Length, ft 81.66
  Width, ft 48.33
  Height, ft 18.00

Rotor characteristics:
  Distance between rotors, ft 33.33
  Number of blades per rotor 3
  Blade airfoil section . . . . NACA 0012
  Blade chord, ft 1.5
  Blade twist, deg 8.05
  Radius, ft 24.17
  Flap hinge offset, ft 0.383
  Solidity 0.0593
  Swept disk area, ft^2 3670
  Projected disk area, ft^2 3310
  Normal rotational speed, rpm 268
  Normal tip speed, ft/sec 678
  Percent overlap, $100\left(1 - \dfrac{\text{Distance between rotors}}{\text{Rotor diameter}}\right)$ 31
  Rotor shaft forward tilt (with respect to fuselage reference line):
    Front rotor, deg 9.5
    Rear rotor, deg 7.0
  Rotor height (above fuselage reference line):
    Front rotor, ft . . . . ≈3.3
    Rear rotor, ft . . . . ≈9.3
  Normal disk loading, lb/ft^2 . . . . 4.24
  Blade weight, lb . . . . ≈130

Inclination of principal inertia axis relative to fuselage reference line, deg . . . . ≈6.5 (nose down)

Power-plant rating (two turbine engines), hp 1050

Normal power loading, lb/hp 9.15

Maximum take-off weight:
  Normal, lb 15 500
  Overload, lb 16 600

Operating gross weight, lb 13 000

Fuel capacity, gal 310

Most forward center-of-gravity location (ahead of midpoint between rotors), ft 2.33

Most aft center-of-gravity location (behind midpoint between rotors), ft 0.83

Moments of inertia:
  Pitch, slug-ft^2 . . . . 75 000
  Roll, slug-ft^2 9200
  Yaw, slug-ft^2 . . . . 71 000

Control travel:
  Longitudinal stick, in. . . . . ±5.5
  Lateral stick, in. . . . . ±3.6
  Pedal, in. . . . . ±2.3
  Collective pitch lever, in. 12.8

Blade travel:
  Collective pitch (at 0.75 radius), deg . . . . 1 to 17
  Longitudinal differential collective pitch, deg . . . . ±3
  Longitudinal cyclic trim:
    Front rotor (fixed), deg . . . . -2
    Rear rotor (fixed for test, normally variable), deg 0.5
  Lateral cyclic pitch:
    Front rotor, deg . . . . ±6.15
    Rear rotor, deg . . . . ±4.60
  Directional differential lateral cyclic:
    Front rotor, deg . . . . ±7.13
    Rear rotor, deg . . . . ±7.13

TABLE II.- HOVERING CONTROL CHARACTERISTICS

(MEASURED AND SPECIFIED BY CRITERIA)

Control characteristic	Measured	AGARD Rept. 408 (ref. 5)	MIL-H-8501A (ref. 4)	
			Instrument	Visual
Pitch axis				
Sensitivity, $\frac{rad/sec^2}{in.}$ . . .	0.31	0.12	0.12	0.07
$\frac{Damping}{Inertia}$, $\frac{rad/sec^2}{rad/sec}$	-0.50	-0.52	-0.52	-0.28
Roll axis				
Sensitivity, $\frac{rad/sec^2}{in.}$ . . .	0.40	0.22	0.23	0.18
$\frac{Damping}{Inertia}$, $\frac{rad/sec^2}{rad/sec}$	-0.76	-1.63	-1.63	-1.17
Yaw axis				
Sensitivity, $\frac{rad/sec^2}{in.}$ . . .	0.11	0.11	0.20	0.20
Control power, rad/sec^2 . .	0.25	0.33	0.61	0.61
$\frac{Damping}{Inertia}$, $\frac{rad/sec^2}{rad/sec}$	≈0	-0.95	-0.95	-0.95
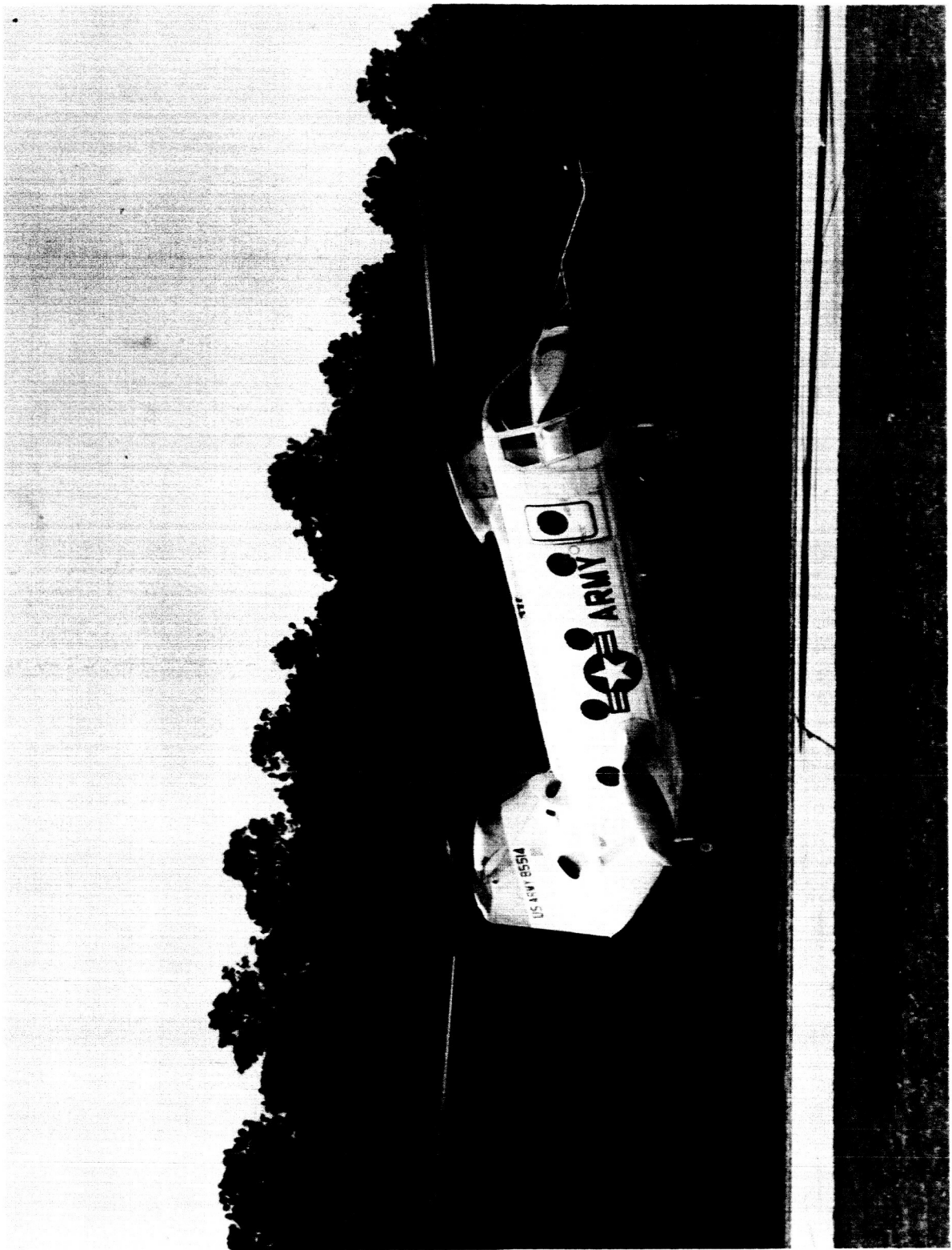
Figure 1.- Test helicopter.

L-63-8406
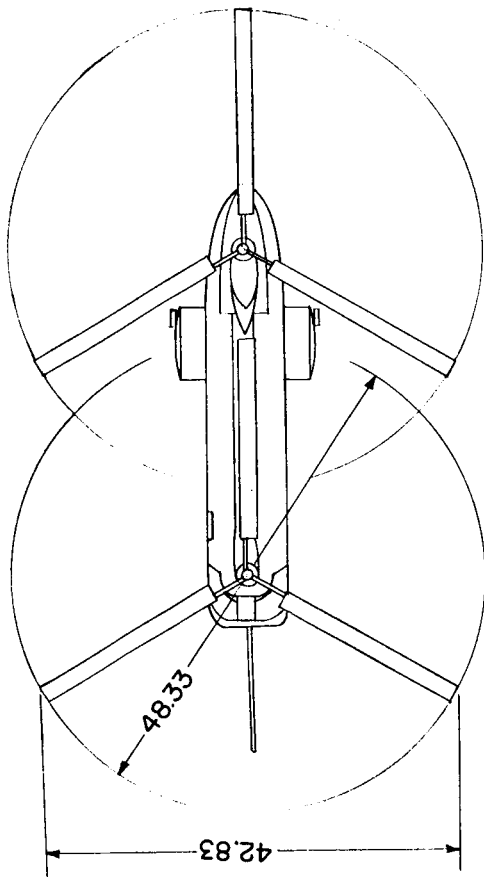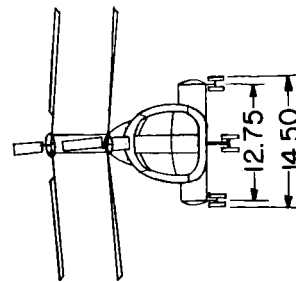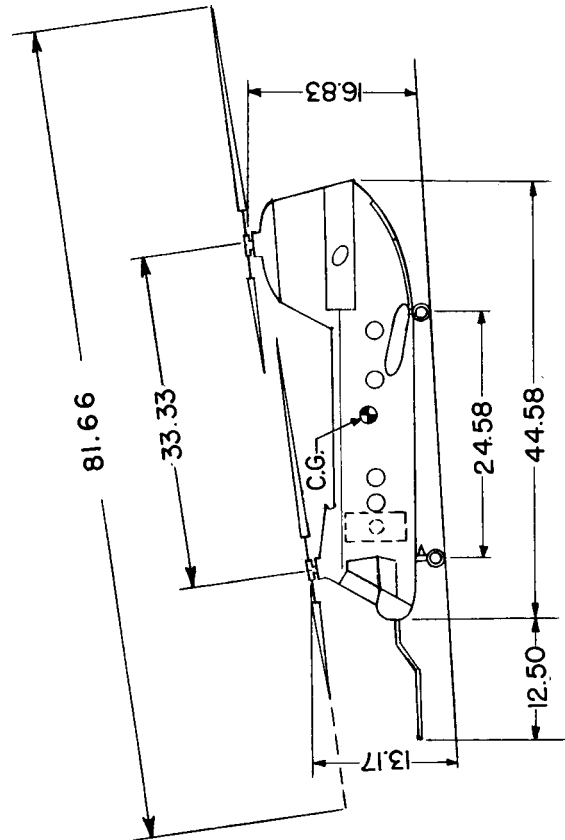
Figure 2.- Three-view sketch of test helicopter. All dimensions are in feet.
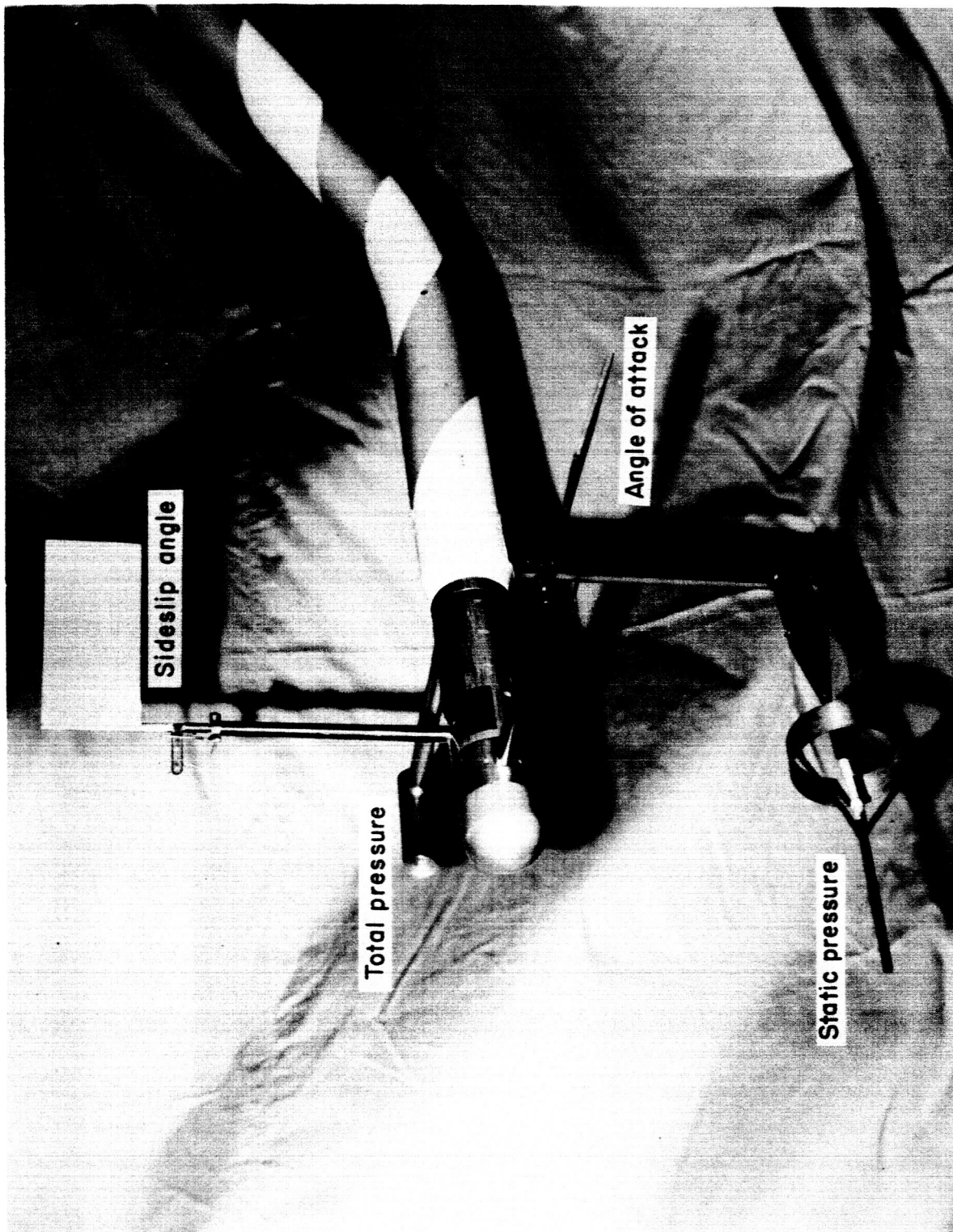
Figure 3.- Boom-mounted sensors.

L-63-8281.1

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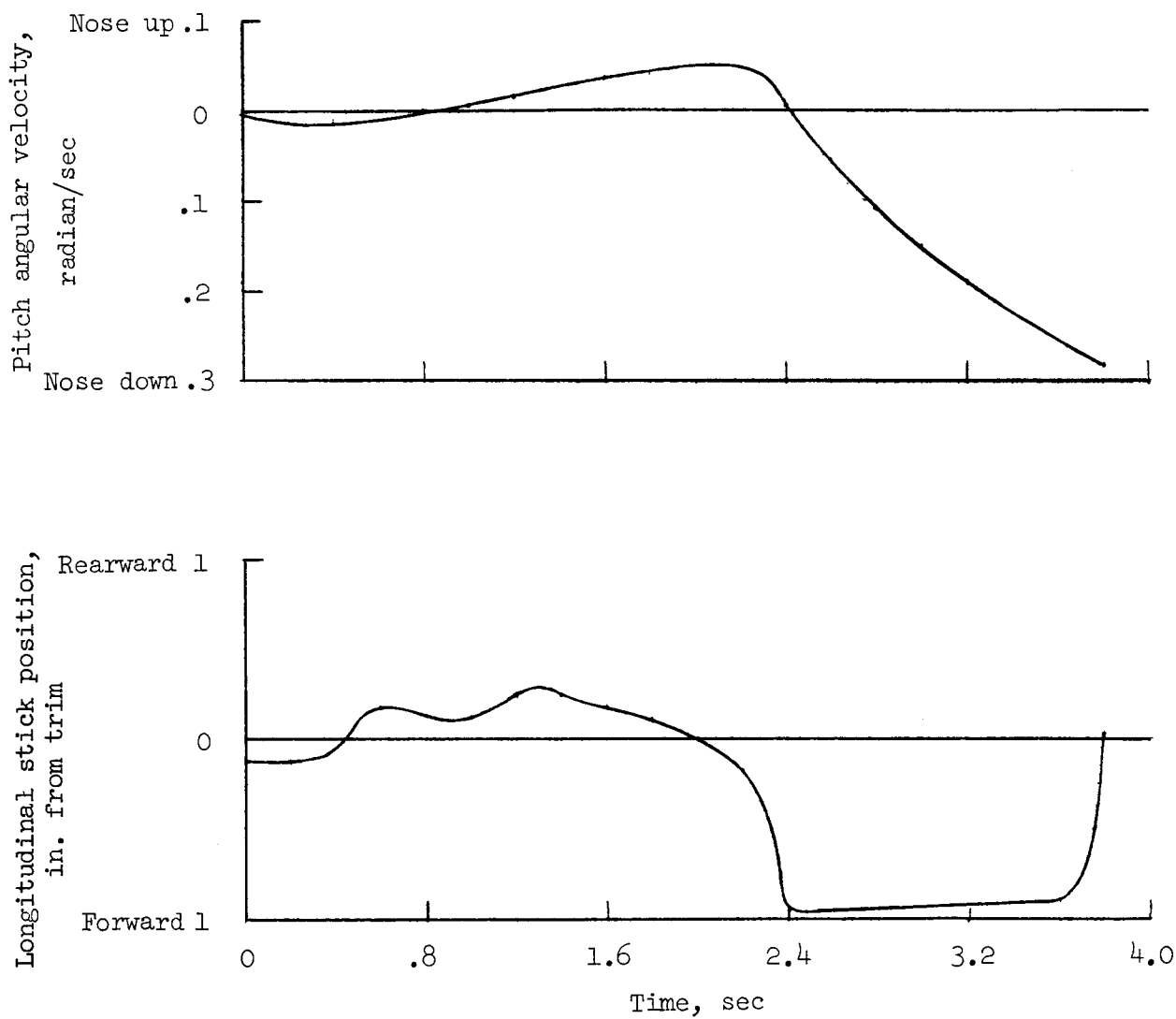
Figure 4.- Typical time history of a longitudinal step input (hovering).

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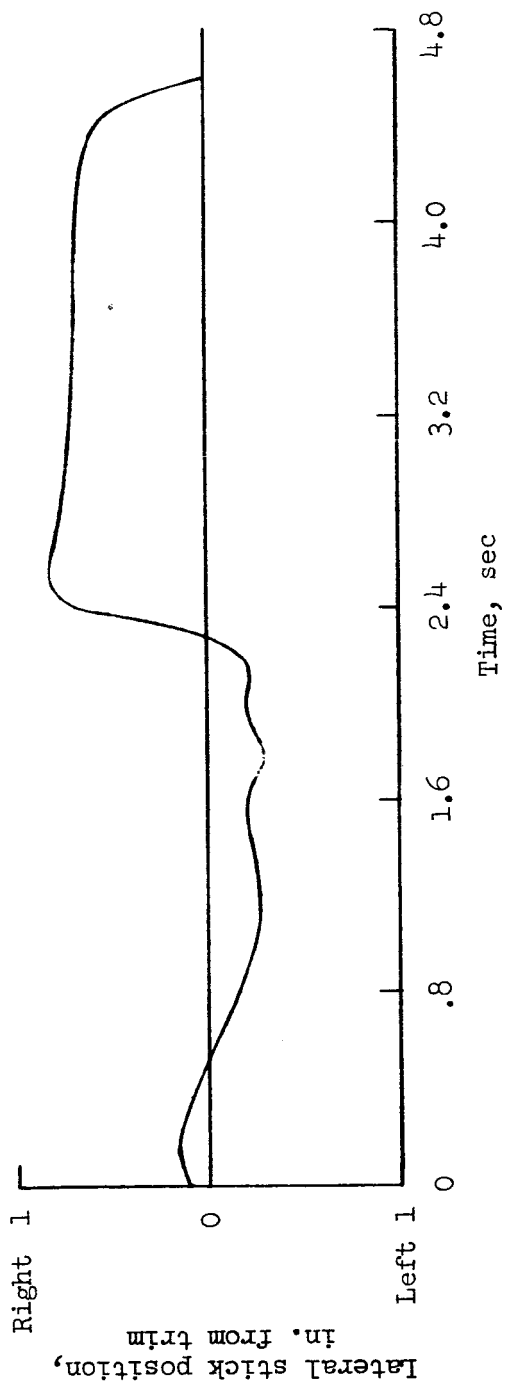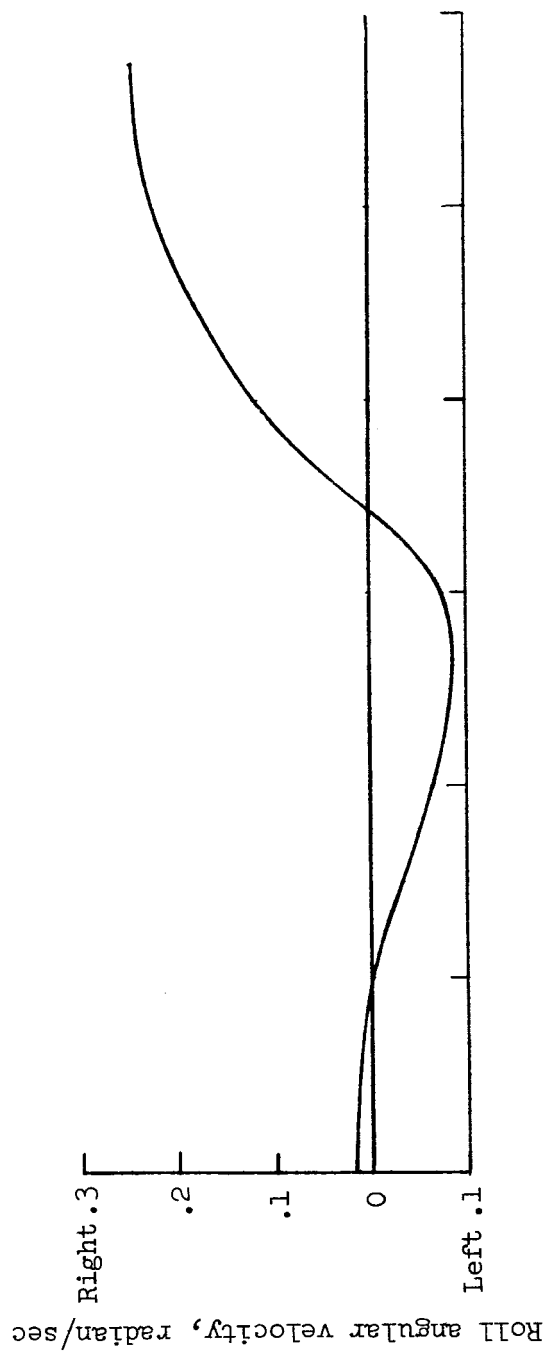
Figure 5.- Typical time history of a lateral step input (hovering).

15
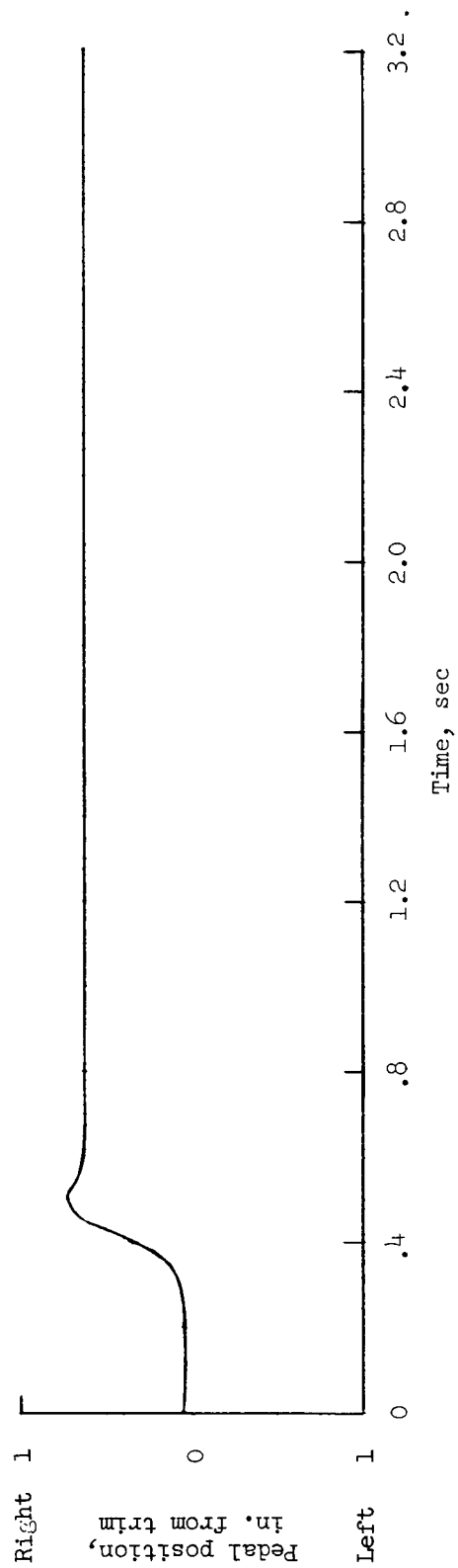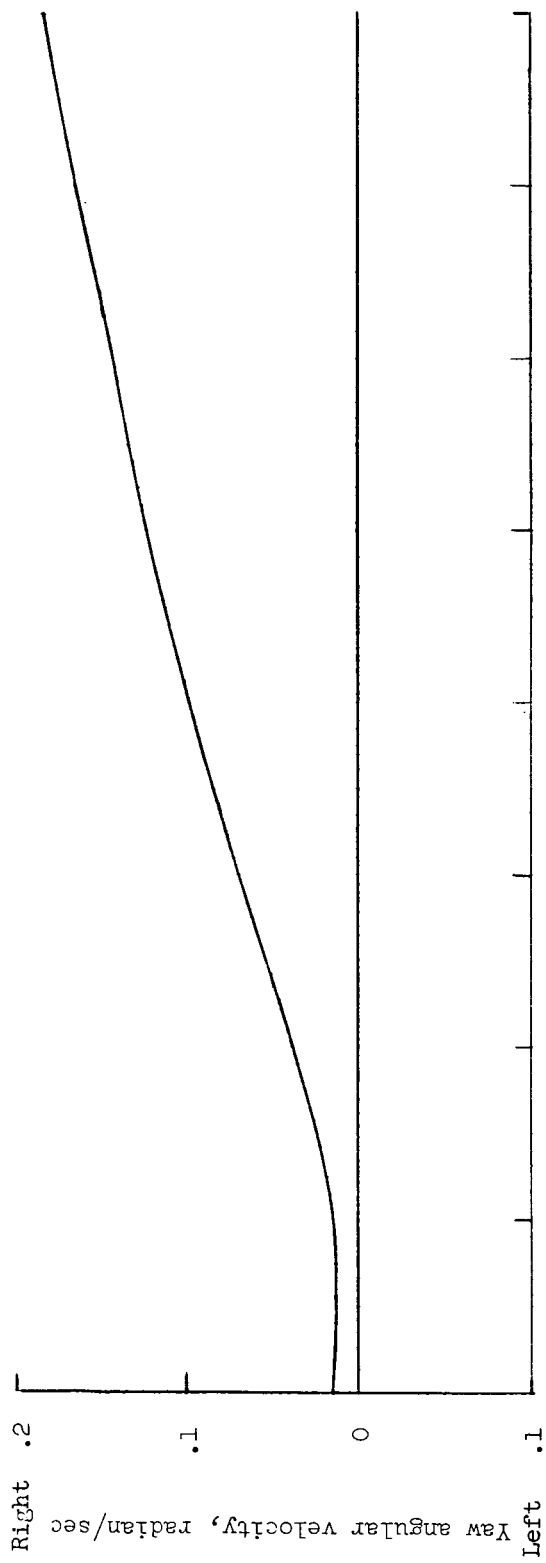
Figure 6.- Typical time history of a yaw step input (hovering).

16
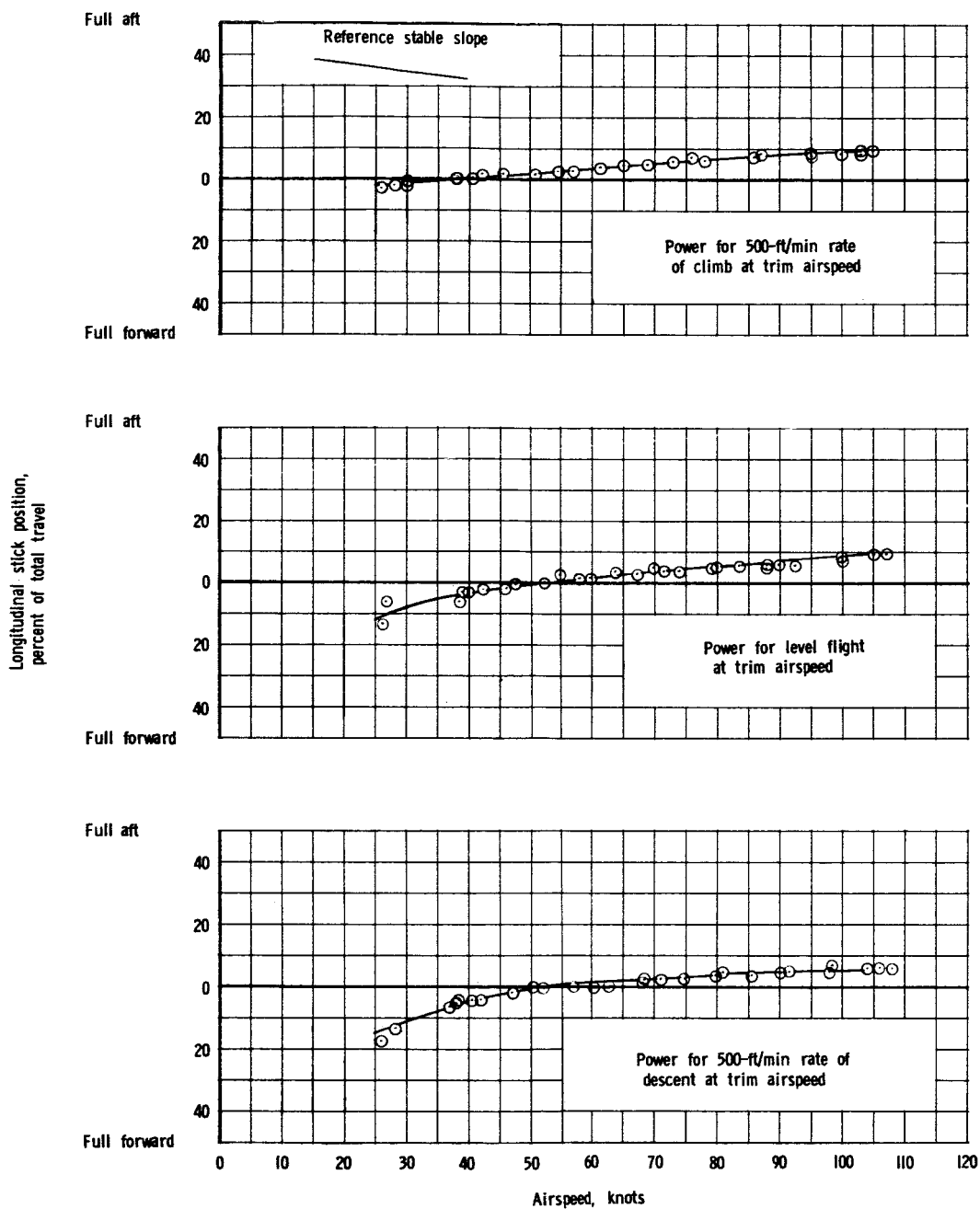
(a) Trim airspeed, 45 knots.

Figure 7.- Variation of longitudinal stick position with airspeed.
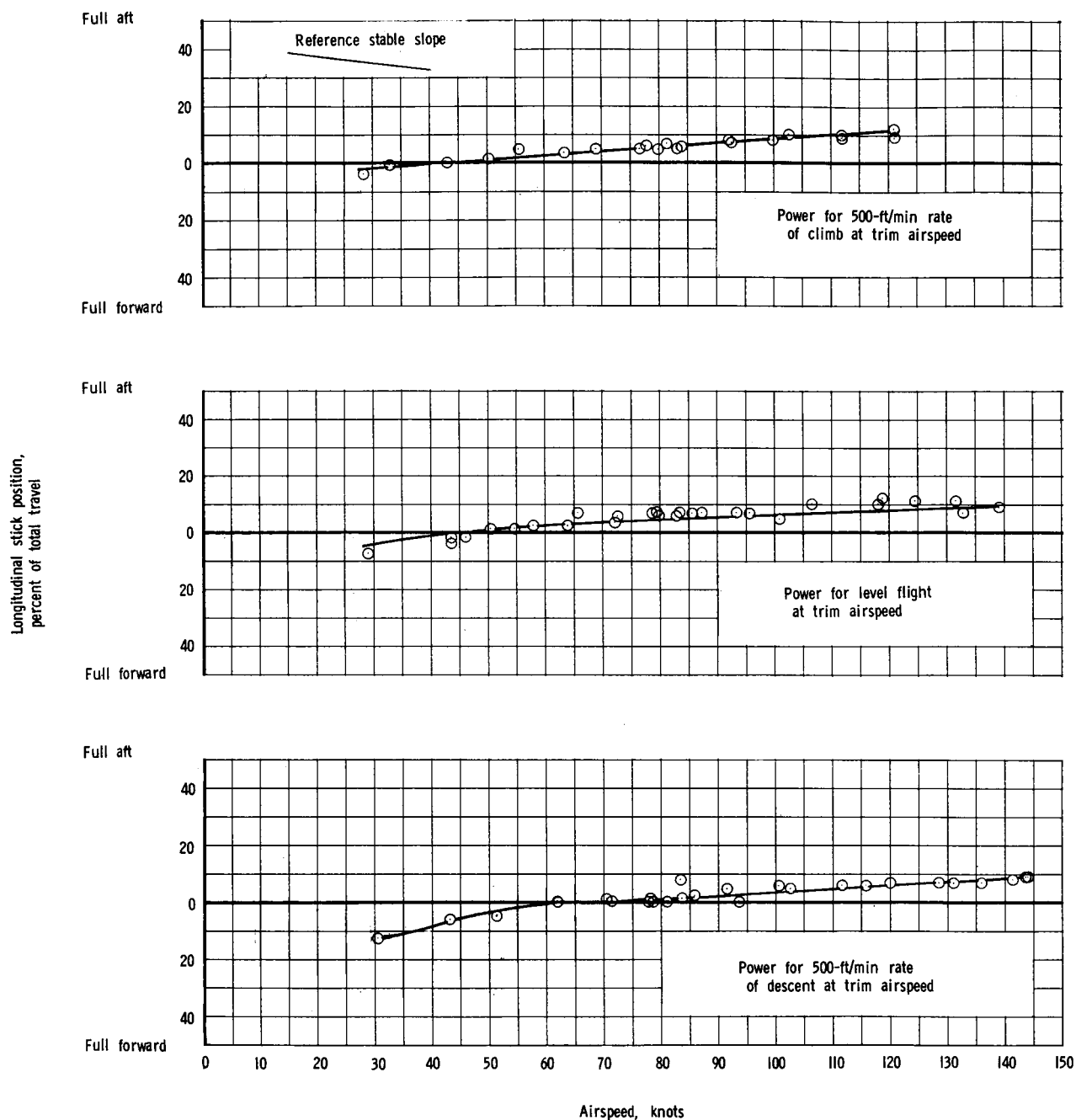
(b) Trim airspeed, 80 knots.

Figure 7.- Concluded.
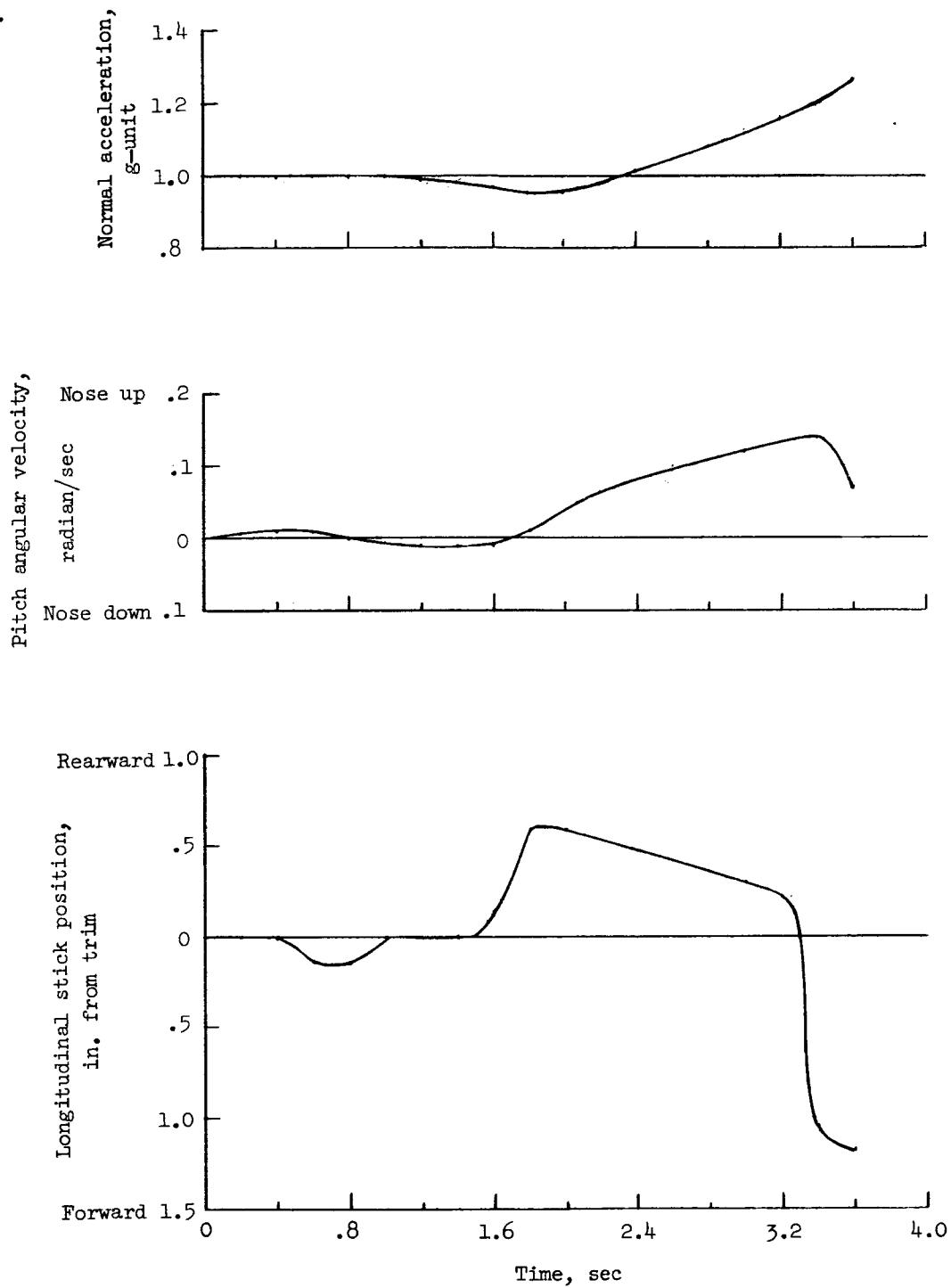
(a) Trim airspeed, 50 knots.

Figure 8.- Pull-and-hold maneuver.
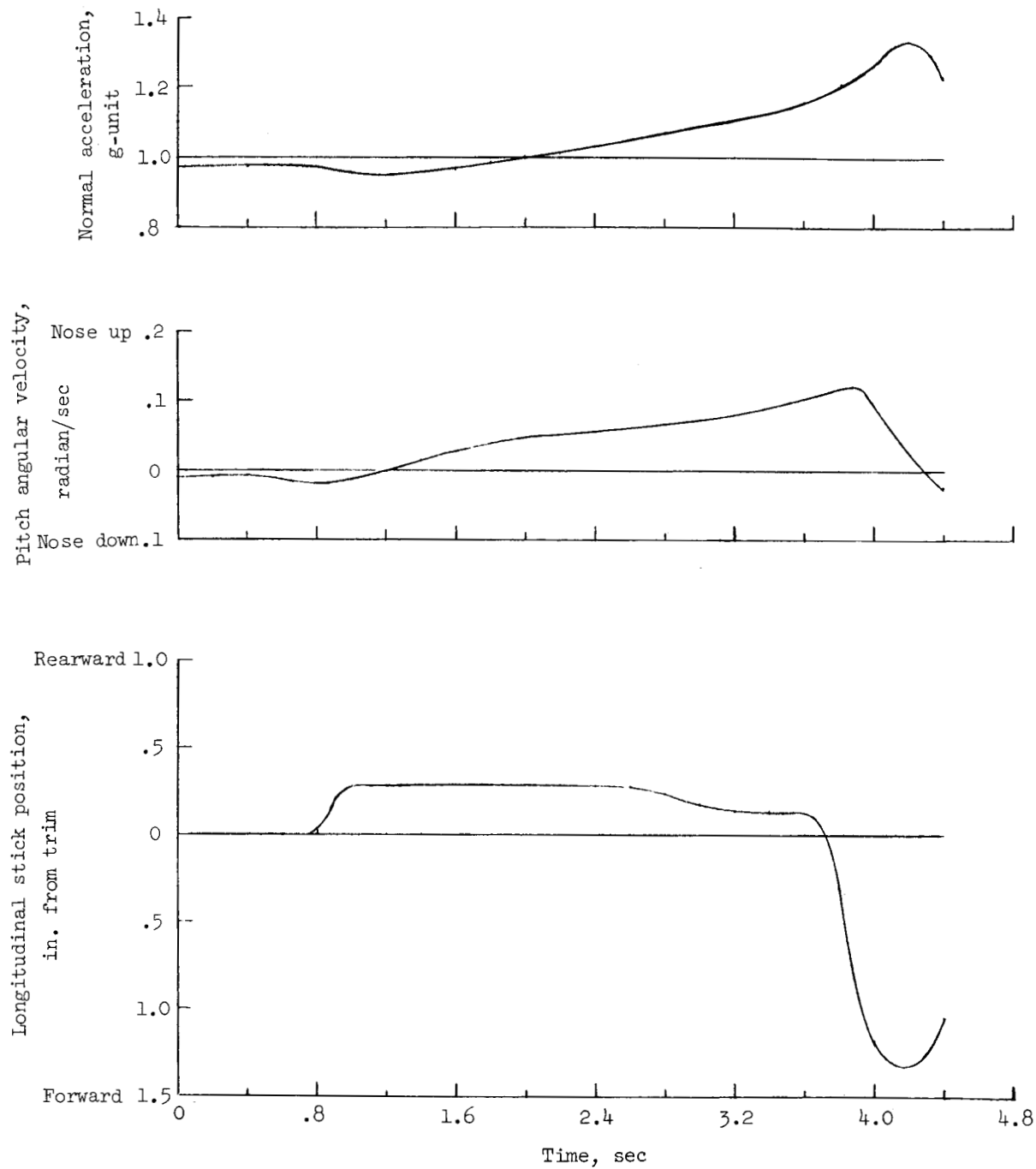
(b) Trim airspeed, 60 knots.

Figure 8.- Continued.
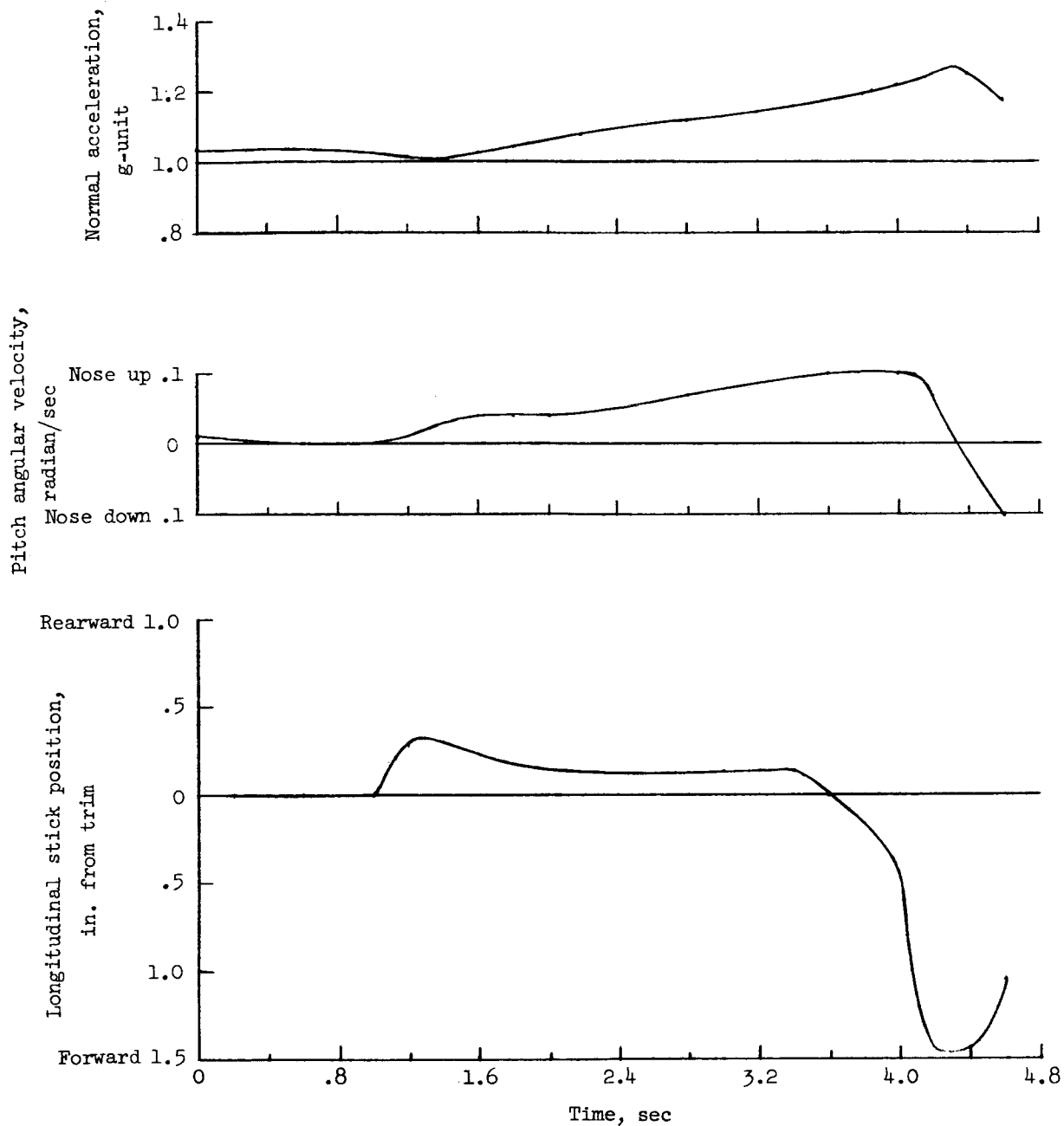
(c) Trim airspeed, 70 knots.

Figure 8.- Continued.
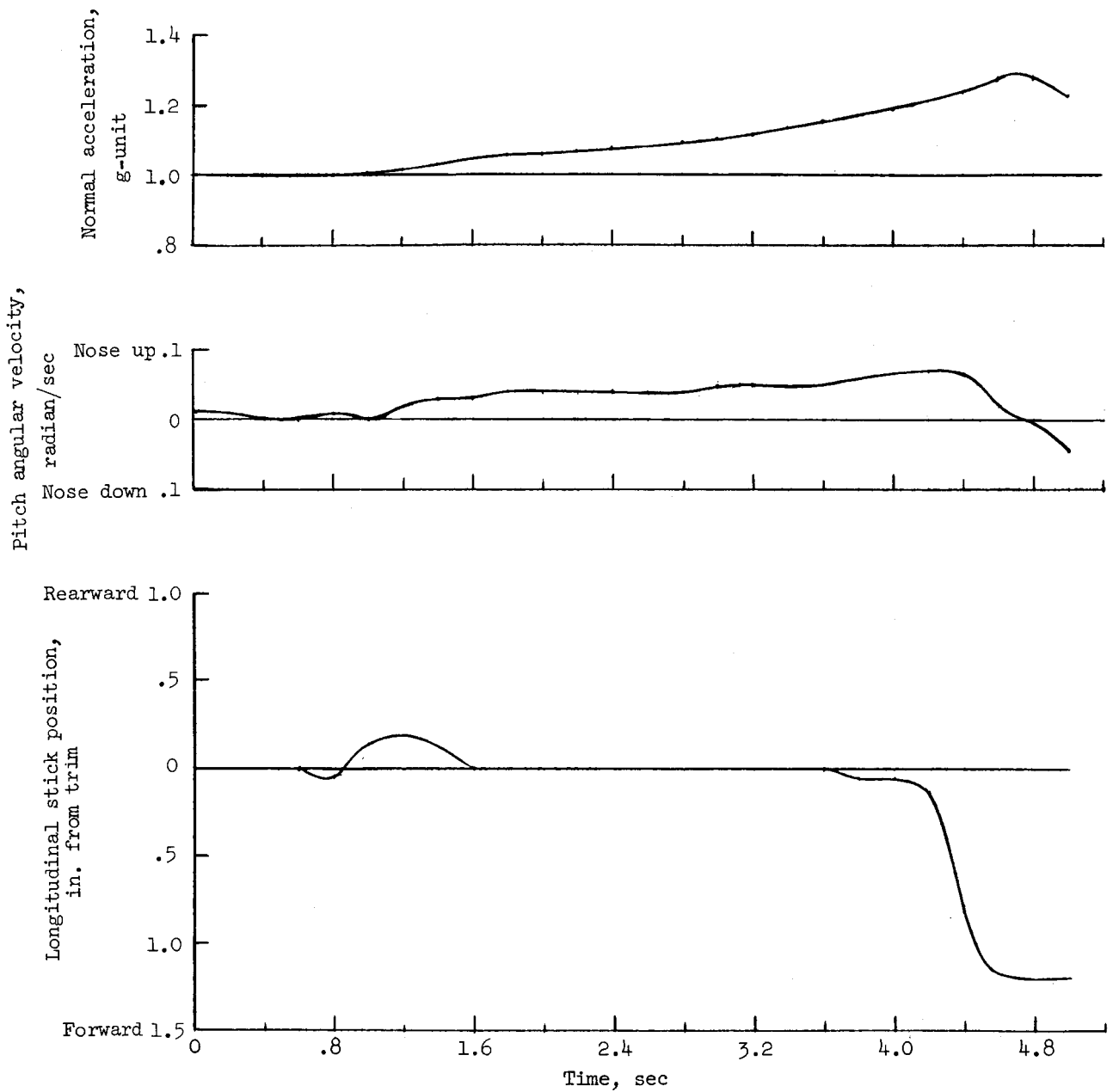
(d) Trim airspeed, 80 knots.

Figure 8.- Concluded.
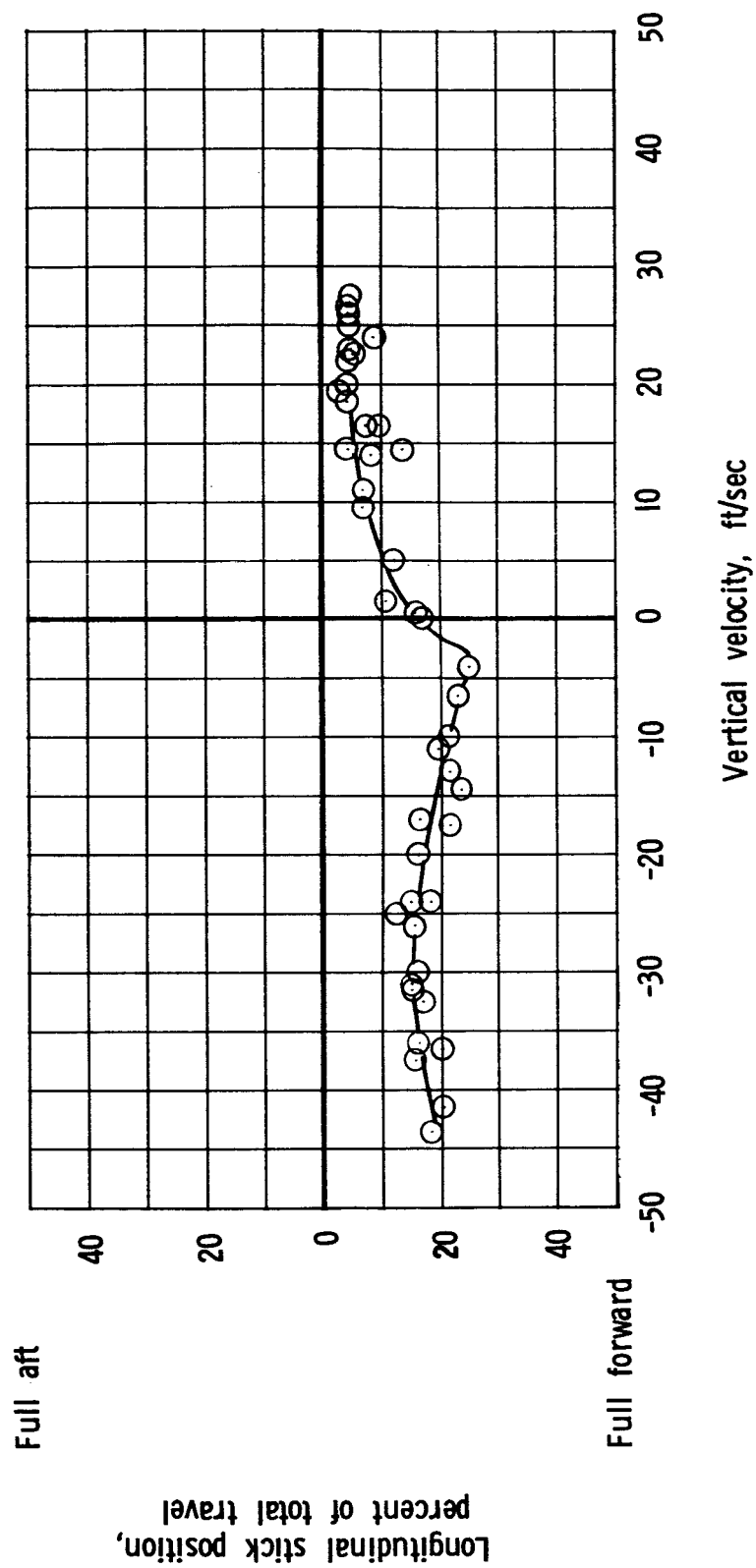
Figure 9.- Variation of longitudinal stick position with power. (Vertical velocity is used as a measure of power.) Airspeed, 20 knots.

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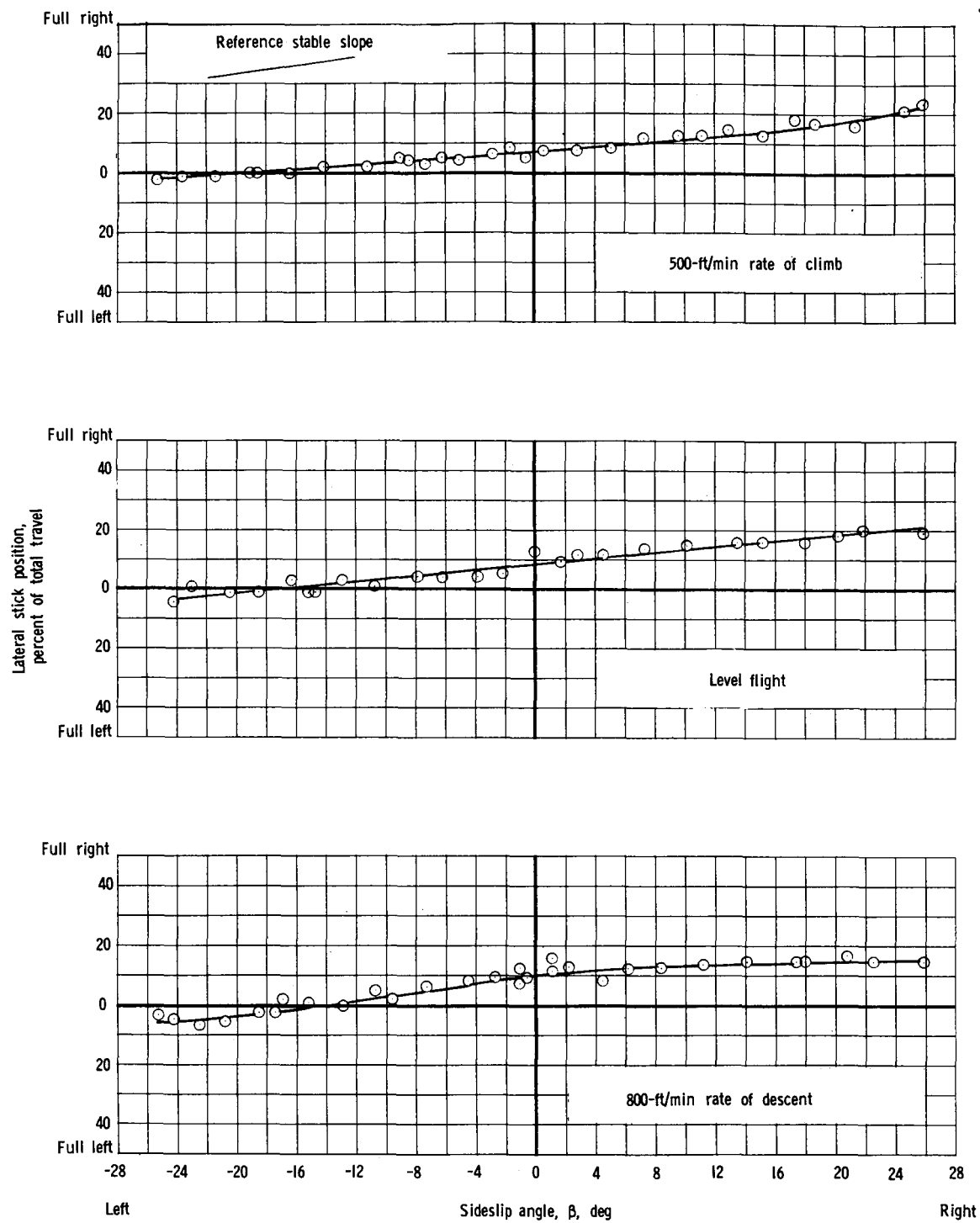
(a) Airspeed, 45 knots.

Figure 10.- Variation of lateral stick position with sideslip angle.
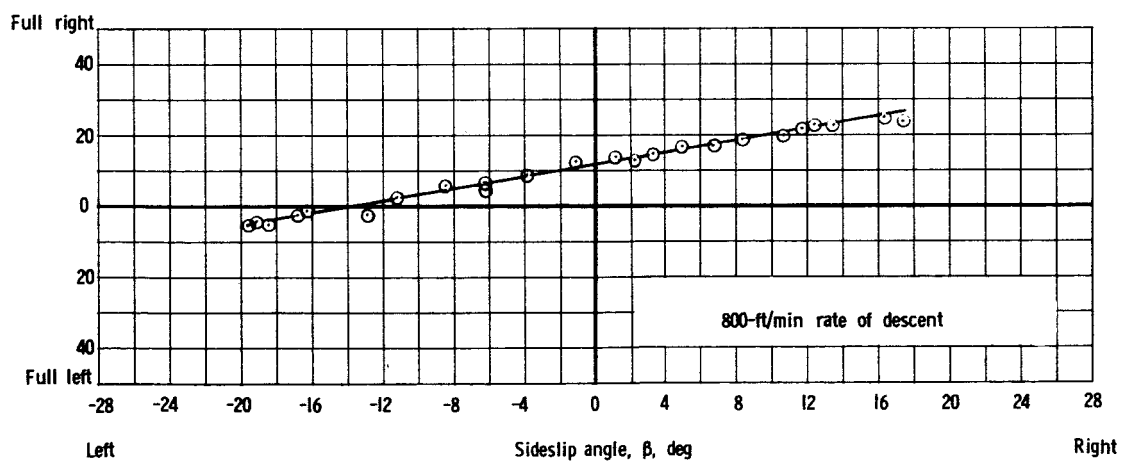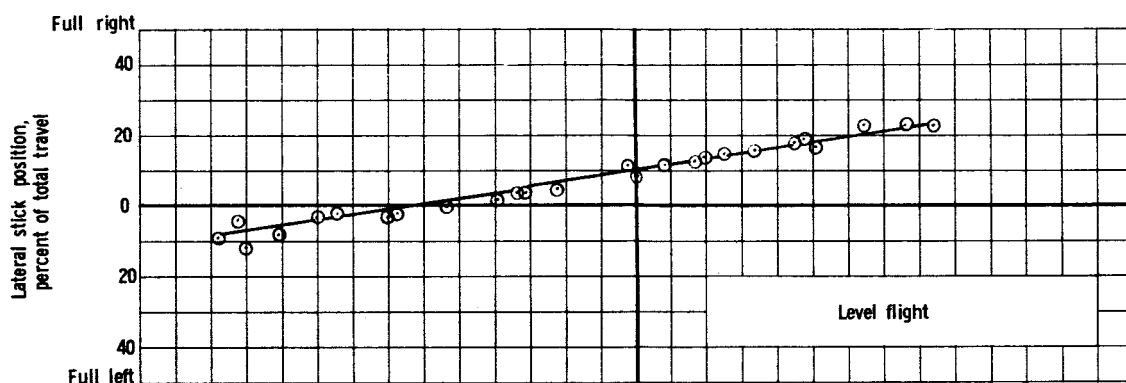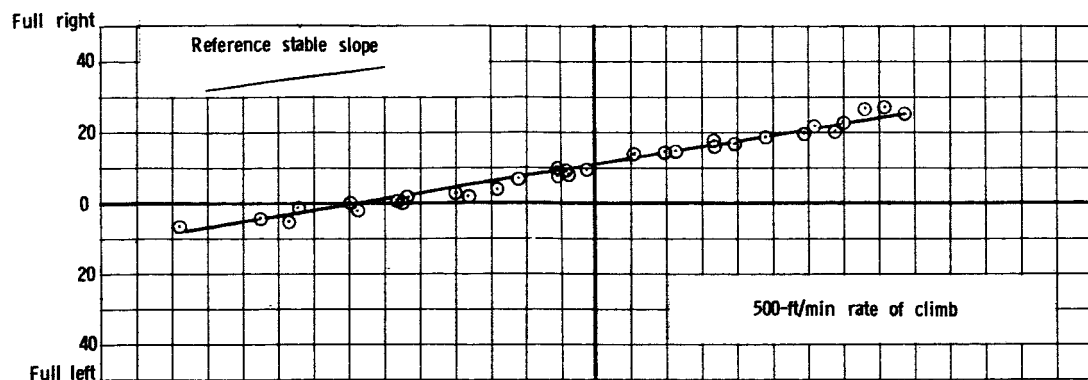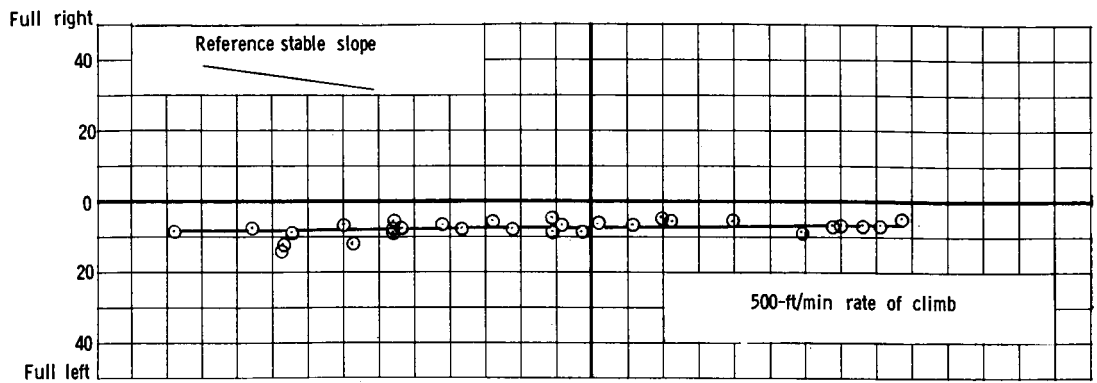
(b) Airspeed, 60 knots.

Figure 10.- Continued.
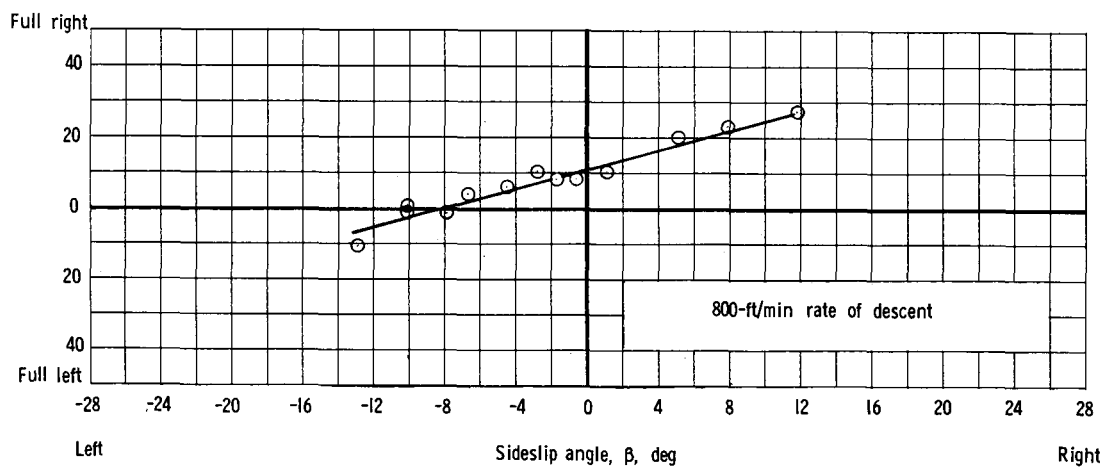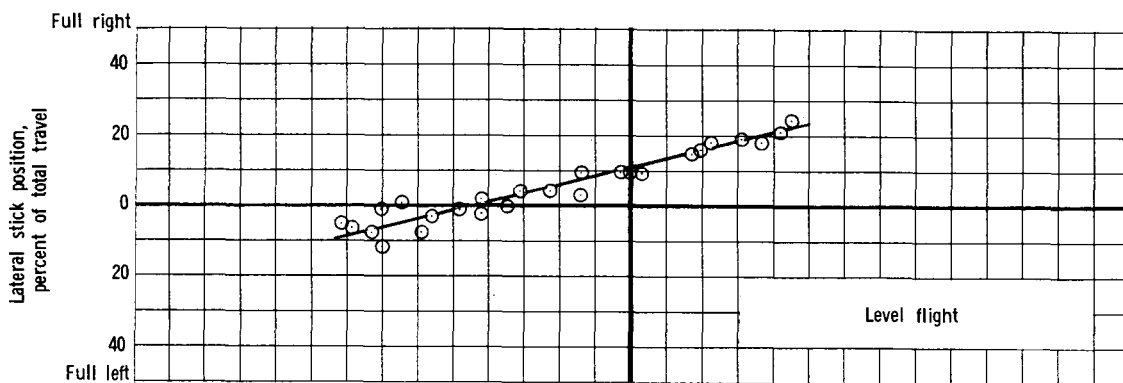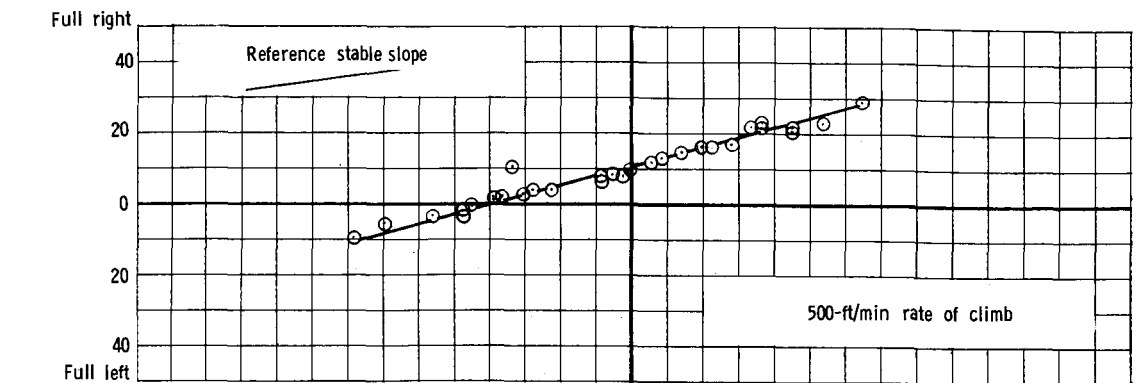
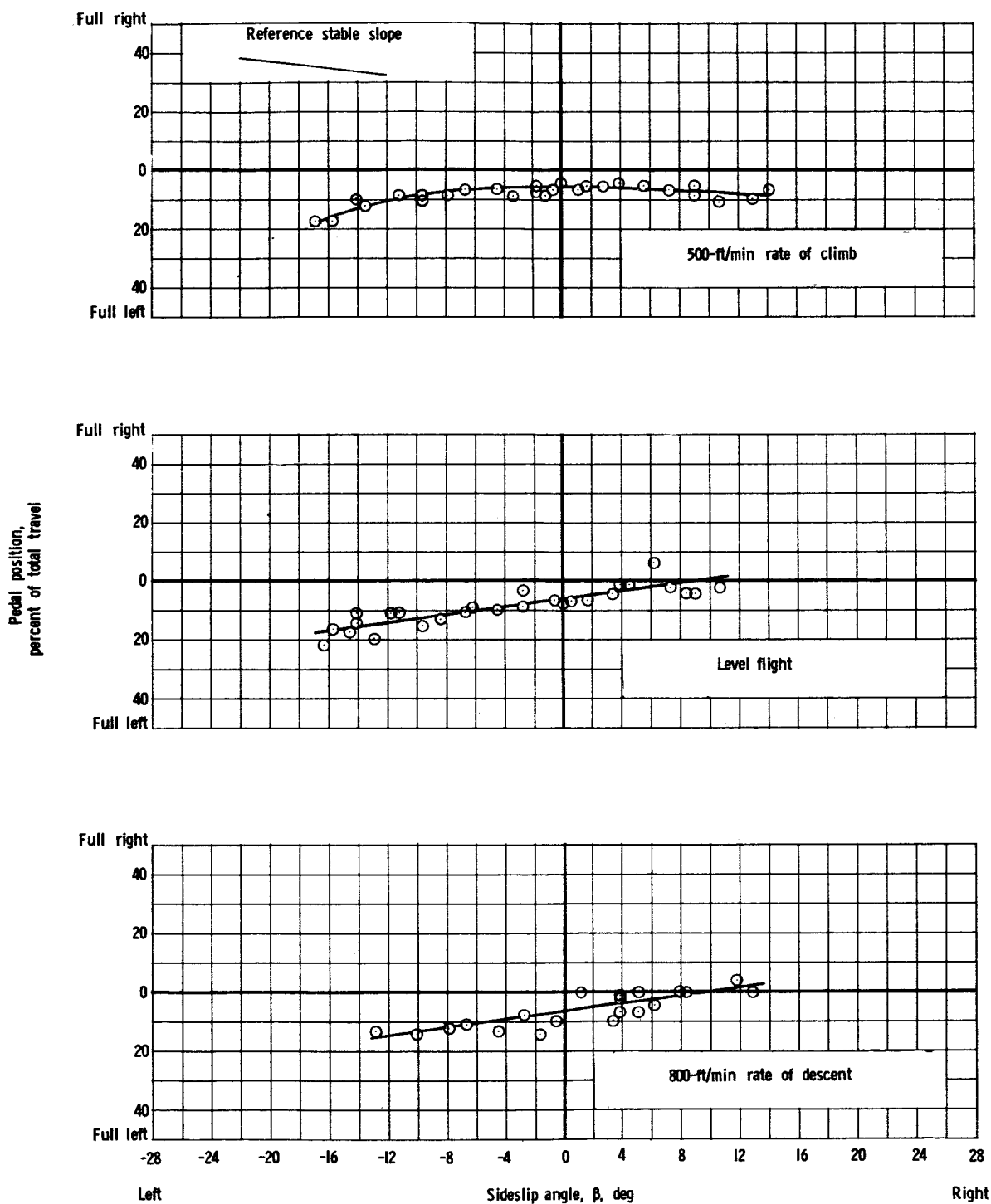(c) Airspeed, 80 knots.

Figure 10.- Concluded.
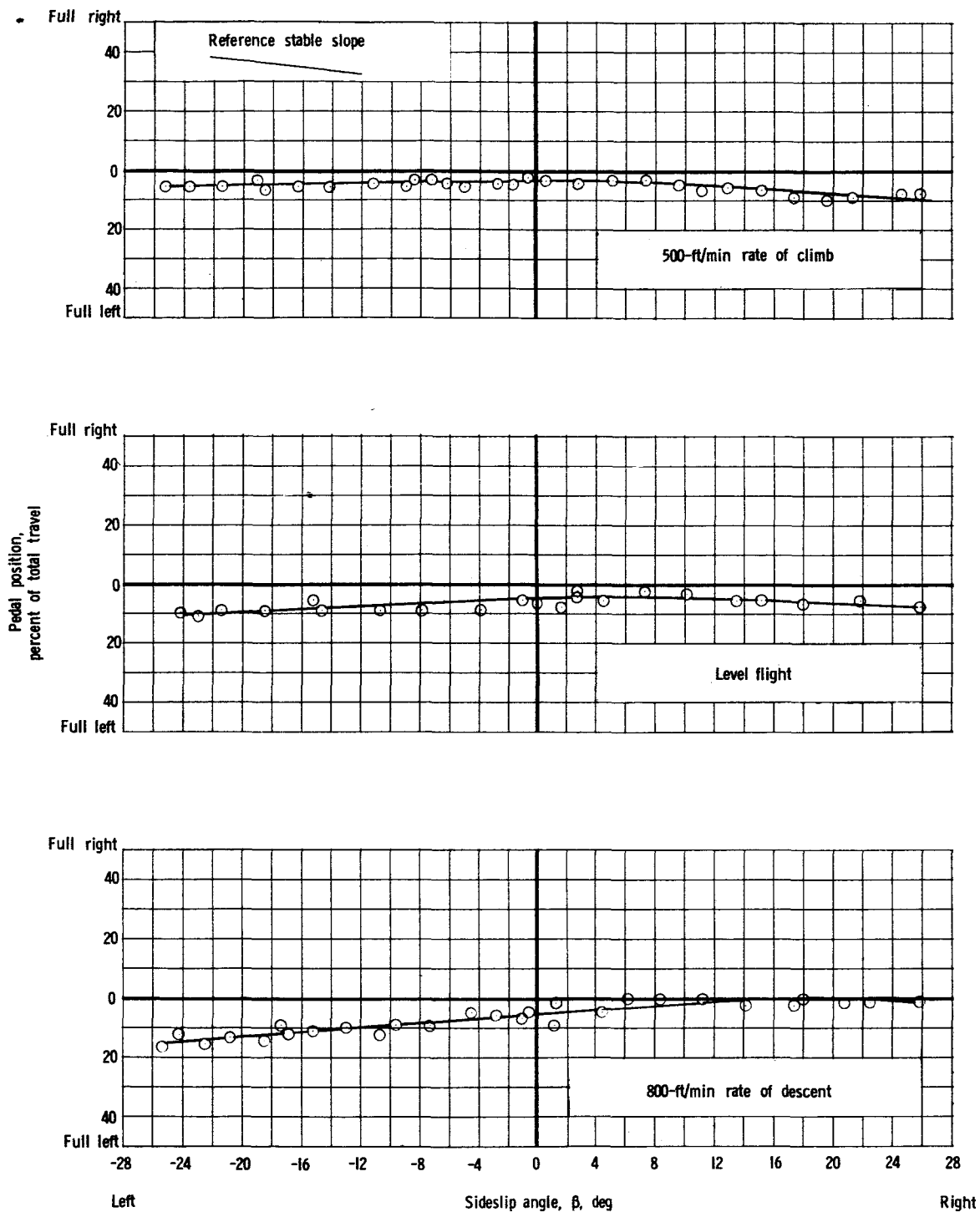
(a) Airspeed, 45 knots.

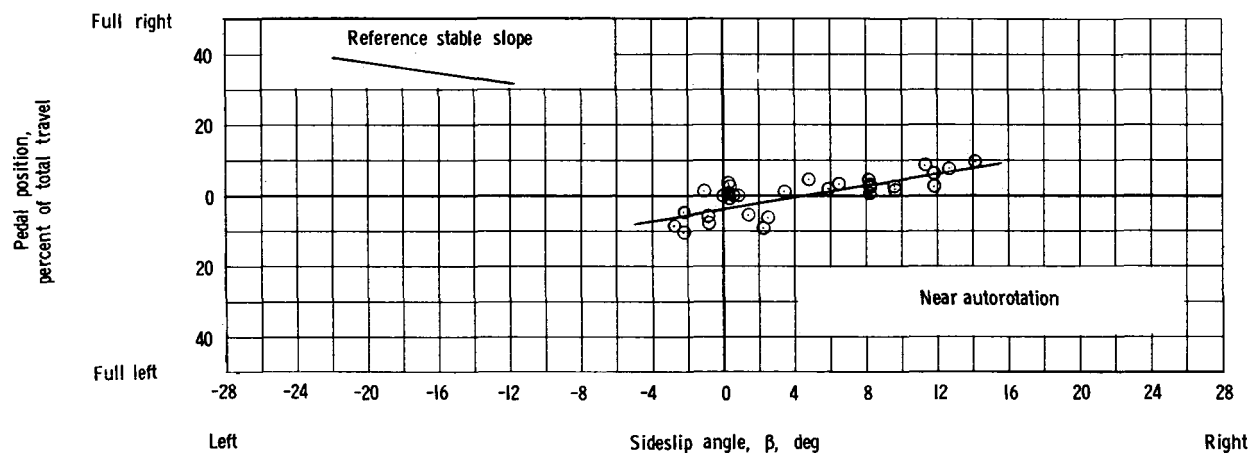Figure 11.- Variation of pedal position with sideslip angle.
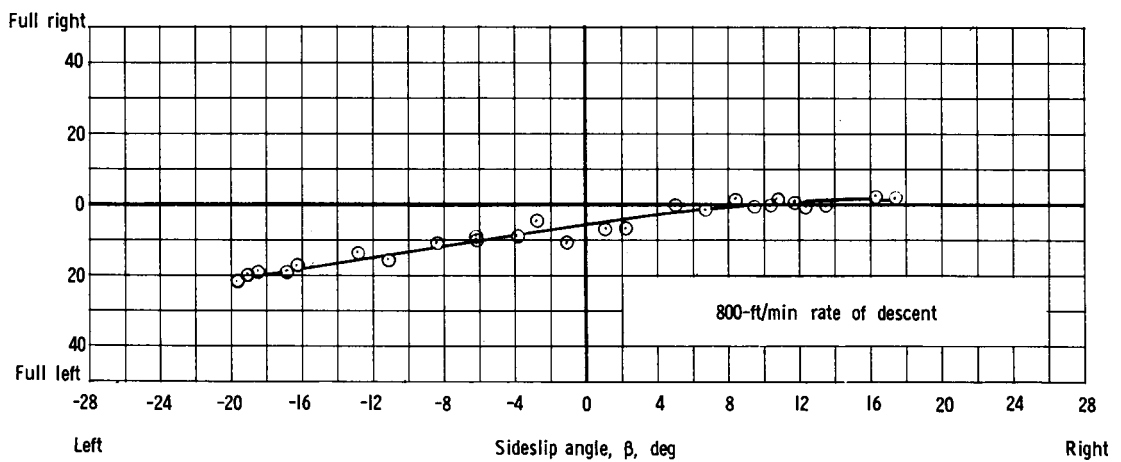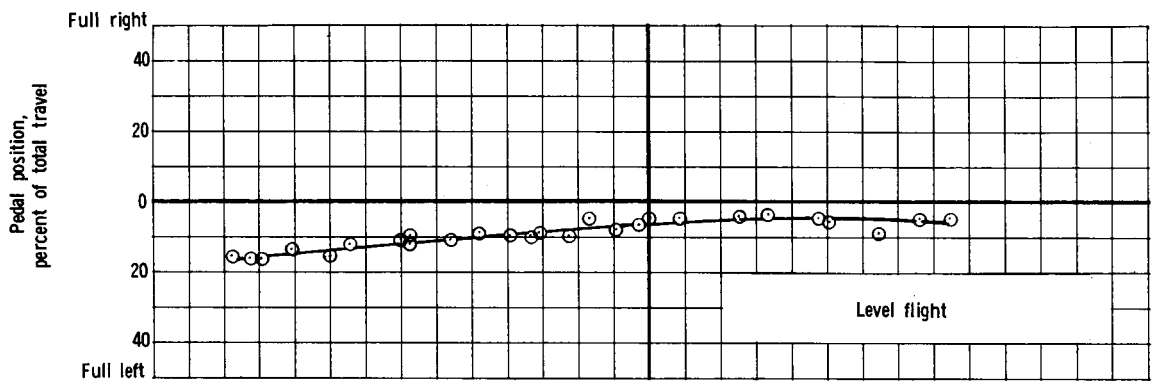
(b) Airspeed, 60 knots.

Figure 11.- Continued.

(c) Airspeed, 80 knots.

Figure 11.- Continued.

(d) Airspeed, 60 knots.

Figure 11.- Concluded.