X 63 80183 Copy

NASA TM SX-805

CASE FILE COPY

TECHNICAL MEMORANDUM

SX-805

for

U.S. Army Transportation Research and Engineering Command

FULL-SCALE WIND-TUNNEL INVESTIGATION OF THE VZ-5 FOUR-PROPELLER DEFLECTED-SLIPSTREAM VTOL AIRPLANE

By Marvin P. Fink

Langley Research Center
Langley Station, Hampton, Va.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON FEB 20 1963

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

TECHNICAL MEMORANDUM SX-805

for

U.S. Army Transportation Research and Engineering Command

FULL-SCALE WIND-TUNNEL INVESTIGATION OF THE VZ-5 FOUR-PROPELLER DEFLECTED-SLIPSTREAM VTOL AIRPLANE

By Marvin P. Fink

ABSTRACT

The investigation was conducted to determine the static stability and control characteristics of the VZ-5 VTOL airplane over the speed range from hovering to forward flight. Force and moment data were taken over a range of angles of attack of 0° to 15° and a range of sideslip of $\pm 10^{\circ}$ for flap deflections from 0° to 77°. The longitudinal stability and trim characteristics were found to be quite unacceptable and it did not seem that they could be corrected with any reasonable modifications to the airplane.

L-3059

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

TECHNICAL MEMORANDUM SX-805

for

U.S. Army Transportation Research and
Engineering Command

FULL-SCALE WIND-TUNNEL INVESTIGATION OF THE VZ-5 FOUR-PROPELLER DEFLECTED-SLIPSTREAM VTOL AIRPLANE

By Marvin P. Fink

SUMMARY

An investigation of the aerodynamic and stability and control characteristics of the VZ-5 four-propeller deflected-slipstream VTOL airplane has been conducted in the Langley full-scale tunnel.

The investigation showed that the airplane was unstable over the speed range and could not be trimmed about the actual center of gravity at 0.64 chord for the low speed. In order to provide a reasonable degree of longitudinal stability for the basic airplane configuration and a bare capability of trim over the flap-deflection range, it would be necessary to ballast the airplane to move the center of gravity far ahead of its actual location. About 700 pounds of weight added to the cockpit area would be necessary to move the center of gravity the required amount. The airplane can develop a hovering lift of about 4,000 pounds which is approximately equal to its weight. The airplane had very high effective dihedral which, coupled with the certain directional instabilities, would be expected to produce highly undesirable flying qualities.

INTRODUCTION

As part of a general program to gain experience and knowledge in the field of V/STOL aircraft, a four-propeller deflected-slipstream research airplane, the VZ-5, was built under the auspices of the U.S. Army Transportation Research and Engineering Command (TRECOM).

A complete investigation of the longitudinal, lateral, and directional stability and control characteristics was made on a 1/5-scale model of the airplane and the results are reported in references 1 and 2.

The results of the small model tests indicated some unsatisfactory aerodynamic characteristics which, coupled with the fact that the flying characteristics of such an airplane were completely unknown, pointed out the desirability of

L-3059

testing the actual airplane in a wind tunnel before flight tests were attempted. Tests were, therefore, conducted in the Langley full-scale tunnel to determine the static stability and control characteristics of the VZ-5 airplane prior to actual flight tests of the airplane.

COEFFICIENTS AND SYMBOLS

The aerodynamic data are referred to the wind axes and to an arbitrary moment reference center located at the 40-percent-chord station and on the propeller thrust axis.

α	angle of attack, deg
b	wing span, ft
c	wing chord, ft
C_D	drag coefficient, Drag/qS
C_L	lift coefficient, Lift/qS
C_l	rolling-moment coefficient, Rolling moment/qSb
C_m	pitching-moment coefficient, Pitching moment/qSc
M_α	rate of change of pitching moment with angle of attack, ft-lb/deg
C_n	yawing-moment coefficient, Yawing moment/qSb
C_Y	side-force coefficient, Side force/qS
q	free-stream dynamic pressure, lb/sq ft
S	wing area, sq ft
δ_f	flap deflection, presented in terms of deflection of sliding flap with respect to wing-chord plane and deflection of the slotted flap with respect to sliding flap. For example, $\delta_f = 30/20$ denotes 30° deflection of sliding flap with respect to wing-chord plane and 20° deflection of slotted flap relative to sliding flap (See fig. 1.)
i_t	horizontal-tail incidence angle, positive when trailing edge is down, deg
β	angle of sideslip, positive when nose of airplane is to the left, deg
L	lift, lb

V velocity

C_{l_β} variation of rolling-moment coefficient with sideslip angle, C_l per deg

h/D ratio of height of airplane reference point above ground to propeller diameter

DESCRIPTION OF THE AIRPLANE

A drawing of the airplane with pertinent dimensions is presented as figure 1, and photographs and a sketch of the vehicle mounted in the full-scale tunnel test section are presented as figure 2. The flap system consisted of a 50-percent-chord sliding flap and a 30-percent-chord slotted flap as shown in figure 1. The geometric characteristics of the airplane are given in table I.

Longitudinal control during hovering was provided by a tail rotor mounted in an essentially horizontal plane behind the horizontal tail. It was driven by a hydraulic motor by using the main propulsion unit as a power source. The tail rotor was in operation at all times from VTOL operation through maximum forward-flight speed. The fan was driven at essentially constant speed and produced thrust by variation of the blade pitch angle as a result of longitudinal stick deflection. For aerodynamic reasons, the all-movable horizontal tail was programed to deflect in a prescribed manner with flap deflection. The programing of the horizontal tail, as prescribed by the manufacturer, is shown in figure 3 which gives the range of tail incidence corresponding to full stick as a function of flap deflection. The blade angle of the pitch-control rotor was not programed to the flap deflection, and at all flap settings full-stick throw gave the maximum deflection of $\pm 10^{\circ}$ blade angle from a mean setting of 0°.

Directional control in hovering was provided by a second tail rotor which was mounted vertically behind the vertical tail. This tail rotor was operated at constant speed and the blade angle ($\pm 10^{\circ}$) (and consequently thrust) was controlled by rudder pedal deflection. The rudder was also in operation at all times and was controlled by rudder pedal deflection.

Roll control in hovering was obtained by differential change of blade pitch of the main propellers which was a function of lateral stick deflection. For example, in order to raise the right wing, the blade angle of the propellers on the right side was increased and the blade angle of the ones on the left side was decreased from the normal blade angle required for VTOL operation. Thus, an increased lift was obtained on the right wing and a decreased lift was obtained on the left wing which resulted in a rolling moment. For roll control in conventional forward flight, the outboard portions of the slotted flaps were used as ailerons. These ailerson were inoperative in hovering flight but, as the flap deflection was reduced from that required to hover, the ailerons were phased into the lateral control system and the differential pitch device was phased out. The differential propeller pitch was completely phased out of the roll control system at flap deflections below 30/20 and the ailerons were then the sole roll-control device.

Vertical or height control in hovering was obtained by varying collective pitch on the four main propellers with the engine governed to hold a constant speed. A separate cockpit control was utilized to operate the propeller pitch collectively.

TESTS

Tests were made to cover the complete speed range of the airplane from hovering to cruise. For the hovering tests the speed of the gas turbine was advanced to maximum (a nominal 97 percent) and the propeller collective pitch was increased until a speed drop in the engine was noted. At this power setting and with the maximum flap deflection of $\delta_f = 50/27$, the angle of attack was varied until the drag scale in the balance system read zero. In this condition lift and pitching moment were recorded.

The forward-flight tests were directed primarily toward conditions simulating unaccelerated flight, that is, lift about equal to the airplane weight (4,000 pounds) and zero net drag. This simulation was accomplished by an iterative process wherein for any given airplane configuration the angle of attack was set at a representative value (usually 5°) and the tunnel speed was increased in conjunction with the collective pitch of the four propellers until the scale balances showed about 4,000 pounds lift and a net drag of zero. Once the condition of balance was obtained, the engine power and collective pitch were unaltered throughout the particular test run. The reason for trying to set a balance condition of full airplane weight, other than from considerations of scale effect, was to obtain the proper relationship between the tail fan output and the other aerodynamic forces. The tail fans always ran at constant speed and consequently full output was available; thus, unless the airplane were tested at a condition where the lift was equal to the gross weight, the other aerodynamic moments would not be proportionate to those produced by the fans. The forward-flight tests covered an angle-of-attack range from 0° to 15°, a range of yaw angles of $\pm 10^{\circ}$, and flap deflections of 0° to 50/27 over a range of tunnel speed from 0 to 65 knots. Structural limitations prescribed by the manufacturer established the top speed limit.

Because of the large size of the airplane relative to the tunnel size (see fig. 2(b)) and the doubtful nature of tunnel-wall corrections for large airplanes with such high values of slipstream velocity and wake deflection, no corrections for the influence of the tunnel walls have been applied to the data, although model blockage was accounted for in establishing tunnel speed. The results of the investigation with the 1/5-scale model (refs. 1 and 2) showed that there was an influence of ground effect on the model characteristics for ratios of height to propeller diameter h/D below 1.00. The relative h/D for the airplane was 0.67. The data for the lower lift coefficients are believed to be reasonably representative of the characteristics of the airplane in free air, but the data for the higher lift conditions can, at best, only be considered to represent the characteristics of the airplane in the presence of the ground.

RESULTS AND DISCUSSION

Hovering Flight

In the hovering tests it was found that the attitude of the airplane for zero drag was 30°. With the vehicle at an angle of attack of 30° and with $\delta_f = 50/27$, a range of blade angles (collective pitch) was tested. A plot of the lift (vertical-force component) against blade angle (fig. 4) shows that a maximum hovering lift of 3,998 pounds was obtained at a blade angle of about 13°. This value of lift is slightly more than the weight of 3,565 pounds without the pilot or the fuel.

The pitching moment in hovering is shown in figure 5 for a range of pitch-control settings. Tail incidence is used as an indication of pitch-control setting, since the blade angle of the pitch fans is linked to move simultaneously with the tail incidence. Tail incidence would not ordinarily be expected to have any effect on pitching moments in hovering flight, but in the case of the present tests which were run in ground effect, the tail incidence itself may have been significant since the tail might have been operating in some appreciable airflow as a result of slipstream deflection off the ground.

The location of the center of gravity of the airplane was determined before it was installed in the wind tunnel and was found to be at the 0.64-chord station. The hovering flight data of figure 5 show that large positive pitching moments occurred and that with this center-of-gravity location and the present tail arrangement the airplane was not quite trimmable with the available travel and tail-fan output. By computing the pitching moments about an assumed moment center of 0.40 chord (the center-of-gravity position used in refs. 1 and 2), the airplane would be trimmable with a tail incidence of about -37°. (See fig. 5.) However, moving the center of gravity to 0.40 chord would require adding about 700 pounds of weight to the cockpit area and would increase the gross weight above the airplane hovering lift capability.

As pointed out previously, lateral control is provided by differential propeller pitch changes for the hovering configuration. Several data points taken during the hovering tests indicate that a rolling moment of about ±4,000 foot-pounds was produced by the lateral control system with a full-stick displacement.

Forward Flight

Longitudinal characteristics.- The longitudinal aerodynamic characteristics of the model with propellers removed (power off) are presented in figure 6 for the range of angle of attack and flap deflection. The data cover only the positive angle-of-attack range because the airplane could not be tested at negative α without the tail fan protruding outside the tunnel airstream. (See fig. 2(b).)

Tuft tests for forward flight (power on) showed that with flap deflections as low as $\delta_f = 10/8$ the upper surface of the sliding flap became stalled, while the slotted flap remained unstalled for all test conditions. Two factors, in particular, are believed to contribute to the stalled condition. First at the leading edge of the sliding flap there was a forward opening gap with a ridge which acted very much like a spoiler; and second, and probably the more important, was the fact that the wing curvature ahead of the sliding flap caused an unfavorable pressure gradient in this critical area. It is believed that a major redesign of the wing to incorporate boundary-layer control or a double slotted flap would be necessary to alleviate this stall condition.

The power-on test data for flap deflections from $\delta_f = 0/0$ to $\delta_f = 50/27$ are presented in figure 7. As explained before, an angle of attack which might be representative of a flight condition was chosen and an initial balance condition (lift = weight and net drag = 0) was set up. A trim angle of attack of about 5° was chosen for flap deflections of 10/8, 20/14, 30/20, 40/20, and 50/27, and at $\delta_f = 50/27$ a trim condition was also set at $\alpha = 10^{\circ}$ to establish another condition approaching the hovering attitude. The actual lift and speed obtained at each of the nominal trim conditions is given in figure 7. In some cases this lift was as much as 12 percent different from the assumed 4,000 pounds weight. One major exception to this condition of trim at a lift of about 4,000 pounds occurred in the test at $\delta_f = 0/0$ in which it was attempted to establish lift and drag trim at $\alpha = 5^{\circ}$. (See fig. 7(a).) In this test it was found that the tunnel speed could not be raised above 65.2 knots without causing severe buckling in the vertical tail; thus the test was made at this speed and the lift at $\alpha = 5^{\circ}$ was only about 2,000 pounds. In order to see whether the basic airplane could attain a lift of approximately 4,000 pounds at a higher angle of attack, an approximate balance condition was established at $\alpha = 15^{\circ}$. At $\alpha = 15^{\circ}$, 3,817 pounds of lift were developed at approximately 58.3 knots.

The longitudinal stability characteristics from figure 7 are summarized in figure 8 which also shows the effect of varying tail incidence on the pitching moment. These values are referred to the actual center of gravity at 0.64 chord in figure 8(a) and to a moment center at 0.40 chord in figure 8(b).

An analysis of the data shows that the airplane with the existing tail arrangement could not be trimmed about the actual center of gravity for the low speed condition. The airplane was also statically unstable over most of the speed range. It should be pointed out here that the airplane in the wind tunnel was in ground effect. Reference 2 shows that, when the model went from in-ground effect to out-of-ground effect, the pitching moments showed an appreciable increase in the positive direction. Any positive pitching moment in addition to the large positive moment already exhibited by the airplane would make it untrimmable over even more of the low speed range.

In order that the airplane be barely trimmable over the speed range, and also have an appreciable degree of stability for the basic airplane configuration the center of gravity would have to be moved forward to approximately the 0.40-chord location as shown by the data of figure 8(b). As previously pointed out, such a shift in center of gravity might be accomplished by adding weight to the nose of the airplane. The large amount of weight required, however, may be beyond the structural limits of the airframe as well as prohibiting hovering because of the excess weight. There are, of course, alternate methods for changing the trim and stability characteristics such as relocating the wing on the fuselage or by the installation of a more powerful tail fan. In view of the fact that the present vertical tail was buckling under the previous load, the latter approach would seem impractical. It would seem, therefore, that there is no ready solution to improving the aerodynamic characteristics of the present configuration.

The data for the relocated center-of-gravity position (0.40 chord) (fig. 8(b)) show that the longitudinal stability varied progressively from stable for the flap-up condition to unstable for the fully deflected flap condition. This characteristic is perhaps more readily recognized in a plot of the variation in M_{α} with speed. (See fig. 9.) As the speed is reduced and the flaps are deflected to higher angles, as in the case of a transition to landing, the stability decreases - becoming zero at about 47 knots - and the airplane is unstable at speeds below this value.

The data of figure 9 also show a very abrupt change in trim with airspeed at a speed of about 35 knots. The data are for the case for an angle of attack of 5° and might not be so severe if the angle of attack as well as flap deflection were varied in this range. Actually, at $\alpha = 5^{\circ}$, the airplane has nearly the same airspeed with flap deflections of 40/24 and 50/27 although the pitching moments for these two flap settings are greatly different. This change from a fairly large diving moment at $\delta_f = 40/24$ to a large nose-up moment at $\delta_f = 50/27$ (see fig. 8(b)) may be attributed to the influence of the ground on the downwash at the tail. Trim is apparently very critical at low speeds with this airplane. The data of figure 5 (center of gravity at 0.40 chord) showed that about 50 percent of the available aft stick travel is required for trim in hovering whereas the data of figures 8 and 9 show that full forward stick is required for trim at $\delta_f = 50/27$ and a speed of about 33 knots.

Lateral and directional characteristics.- The static lateral and directional stability characteristics are presented in figure 10 for range of sideslip angles of $\pm 10^{\circ}$ at angles of attack of 5°, 10°, and 15° for flap deflections of 0/0, 20/14, and 30/20. Asymmetrical conditions may be noted throughout both the lateral and directional data. It is believed that this is due, at least in part, to the general flexibility of the airplane. During the wind-tunnel tests, twisting of the aft section of the fuselage as well as severe shaking of the entire tail assembly was noted. It is believed that under certain loading conditions aeroelastic effects resulted in apparent control moments even though the control position indicators showed no control deflection.

The data of figure 10 show that the airplane was generally directionally stable. However, close examination of the plots of C_n against β show that in some cases an instability exists at small sideslip angles. Considerable effort was expended in the small model tests of reference 1 to try to alleviate this condition but the condition was not completely eliminated and is shown here to be characteristic of the full-scale airplane as well as the model. The airplane configuration incorporated most of the changes which showed improvement on the model. The data of figure 10(a) for the $\delta_f = 0/0$ condition show that the variation of rolling moment with sideslip angle is fairly linear and that the value of $C_{l_{\beta}}$ is about -0.0048 (about 24° effective dihedral). These results, in general, concur with the findings of the small model lateral tests reported in reference 1. Such a high value of effective dihedral would be likely to cause undesirable handling characteristics, particularly in view of the poor directional stability characteristics and the obviously large nose-down inclination of the principal longitudinal axis of inertia. However, the flight reports of the VZ-3, a two-propeller deflected-slipstream airplane, state that, although the lateral and directional stability was weak, there was no particular problem in flying the airplane through the speed range. For the case with flaps deflected 20/14 and 30/20 (figs. 10(b) and 10(c)) the variations of C_l with β became very nonlinear and it would be difficult to obtain a slope value over the yaw range; however, mean values indicate that the dihedral effect has increased greatly over that for the $\delta_f = 0/0$ condition.

CONCLUSIONS

The full-scale wind-tunnel investigation of the longitudinal and lateral stability and control characteristics of the VZ-5 four-propeller deflected-slipstream VTOL airplane may be summarized as follows:

1. With the actual center of gravity of 0.64 chord, the airplane exhibited a longitudinal instability over almost the entire speed range and was untrimmable at low speed and in hovering. In order to provide a reasonable degree of longitudinal stability and a bare capability of trim over the speed range, it would be necessary to move the center of gravity forward almost 0.24 chord from its actual location. This change in the center of gravity would require the addition of a large amount (about 700 pounds) of weight to the cockpit area of the airplane.

2. At a propeller-height-to-diameter ratio of 0.6 the airplane can develop a hovering lift of about 4,000 pounds which is approximately equal to its weight.

3. The airplane had very high effective dihedral which coupled with directional instabilities would be likely to produce highly undesirable flying qualities.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., January 18, 1963.

REFERENCES

1. Kuhn, Richard E., and Grunwald, Kalman J.: Lateral Stability and Control Characteristics of a Four-Propeller Deflected-Slipstream VTOL Model Including Effects of Ground Proximity. NASA TN D-444, 1961.

2. Kuhn, Richard E., and Grunwald, Kalman J.: Longitudinal Aerodynamic Characteristics of a Four-Propeller Deflected-Slipstream VTOL Model Including the Effects of Ground Proximity. NASA TN D-248, 1960.

TABLE I.- GEOMETRIC DATA OF AIRPLANE

Wing:	
Area, sq ft	191.00
Span, ft	32.75
Aspect ratio	5.52
Taper ratio	1.00
Incidence, deg	5
Airfoil section	NACA 4415
Flap:	
Sliding flap, percent chord	50
Slotted flap, percent chord	40
Deflection range:	
Sliding, deg	50
Slotted, deg	27
Horizontal tail:	
Area, sq ft	61.12
Span, ft	15.50
Aspect ratio	3.93
Deflection range, deg	+25; -42
Airfoil section	NACA 0015
Vertical tail:	
Area, sq ft	29.97
Airfoil section	NACA 0012
Deflection range, deg	±10

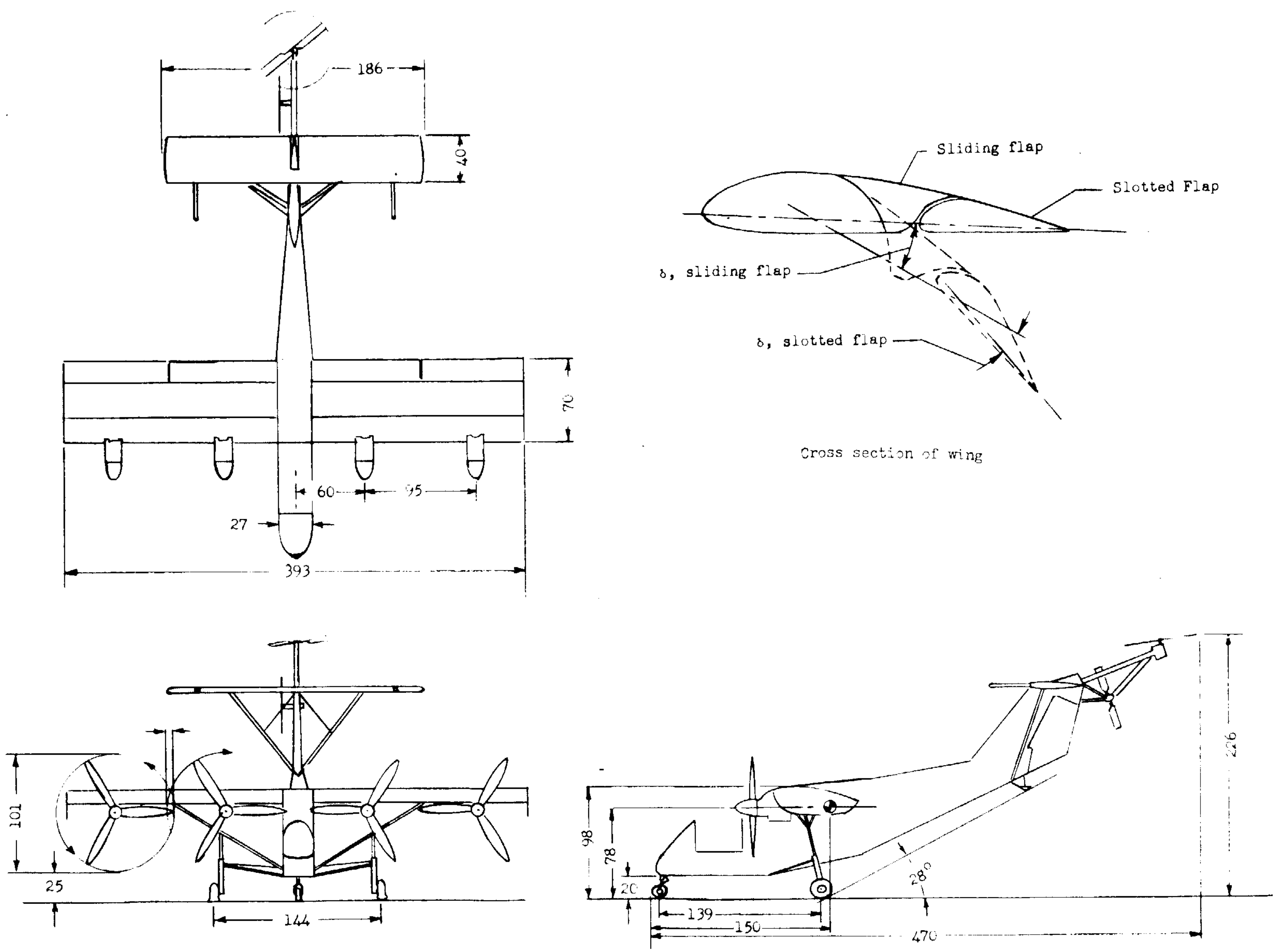
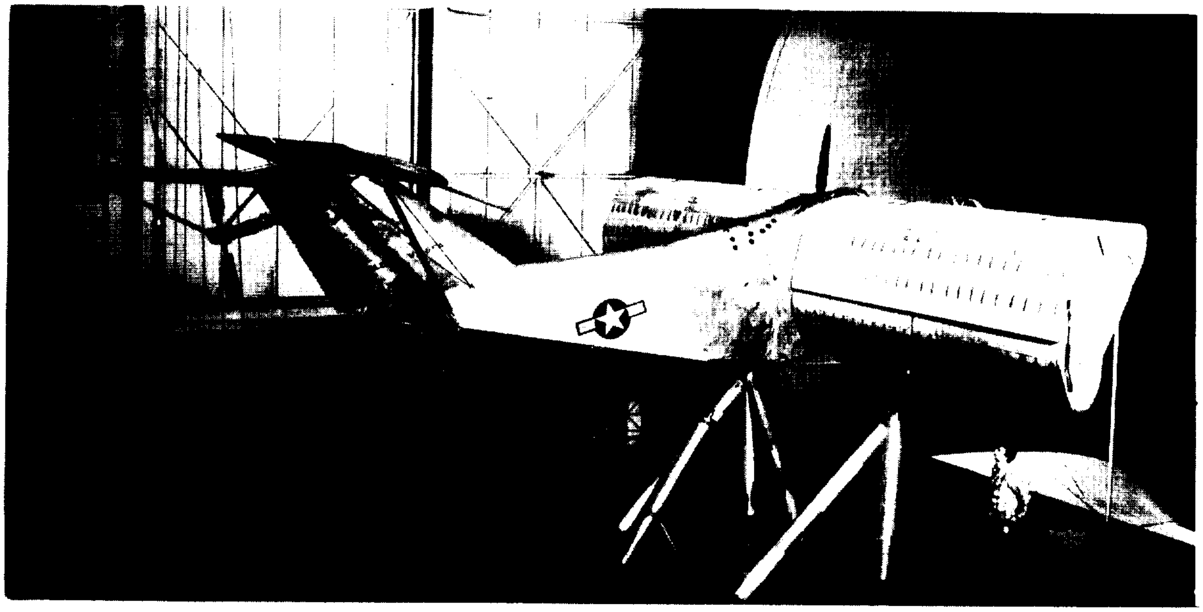
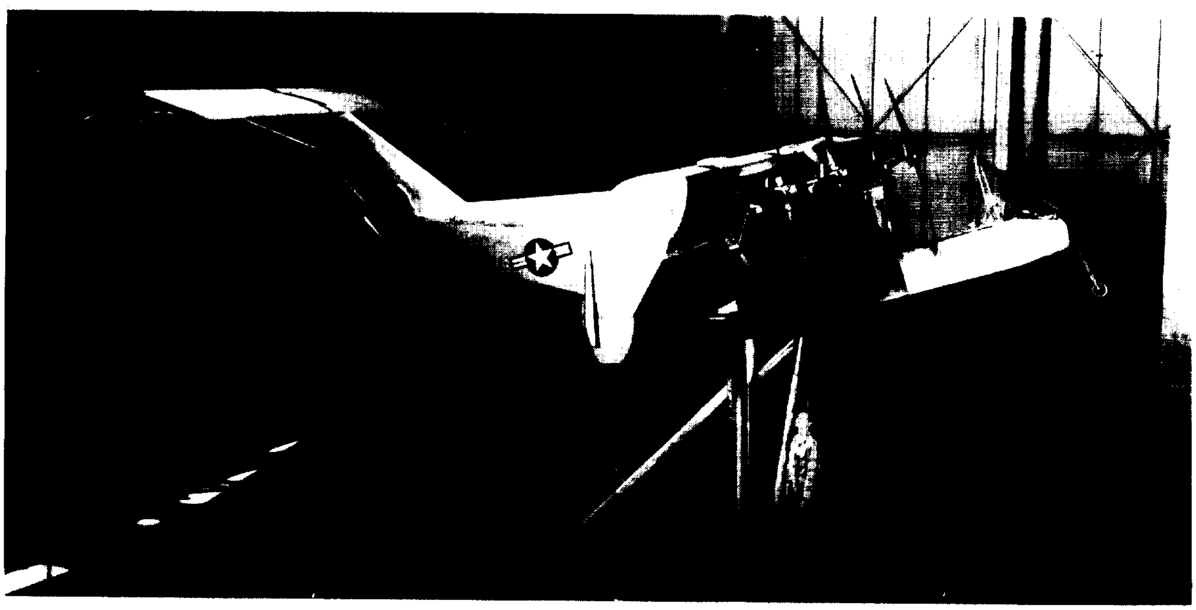


Figure 1.- General views and principal dimensions. All dimensions are in inches.

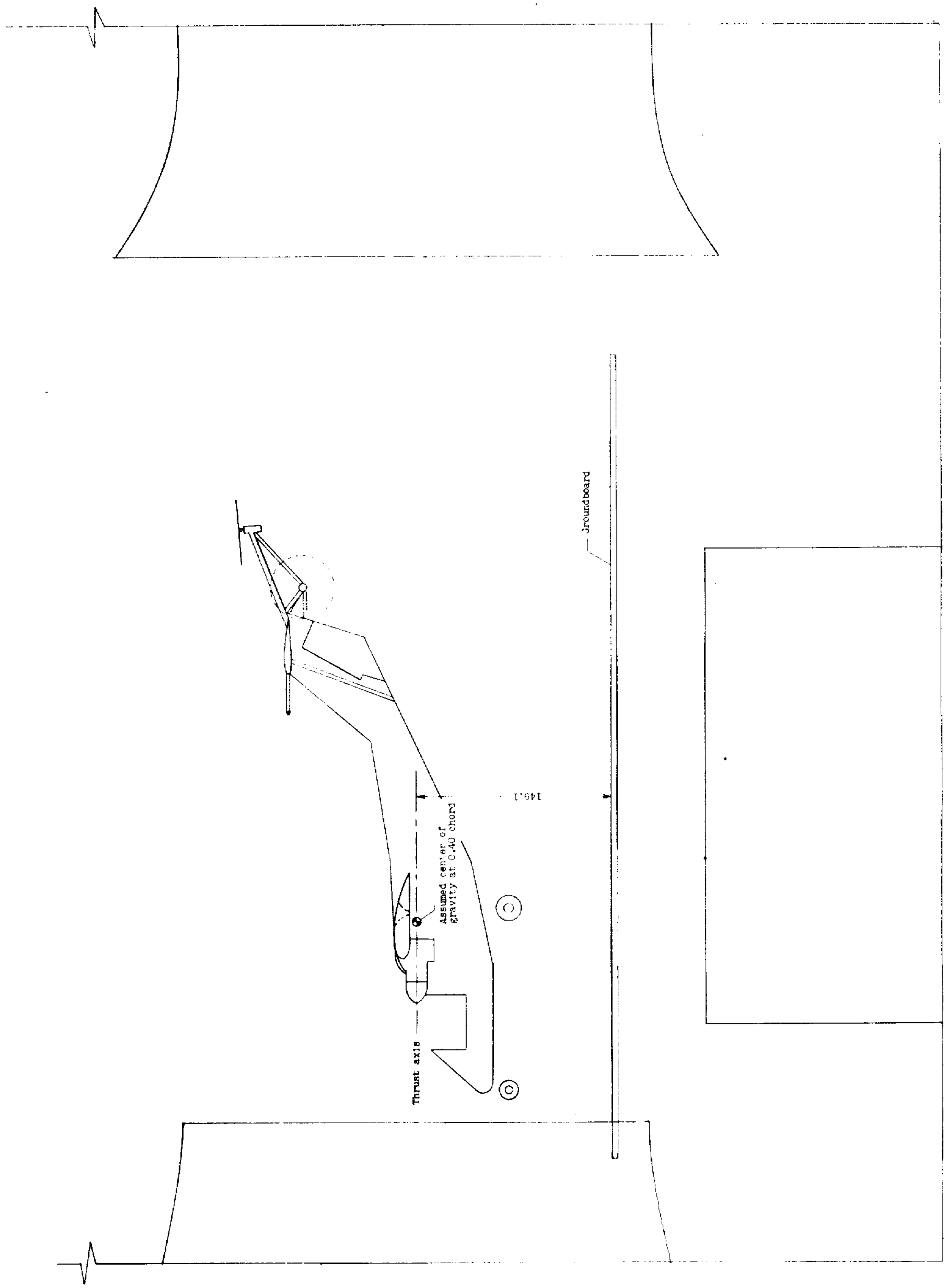


L-61-2822



(a) Photographs. L-61-2821

Figure 2.- Photographs and sketch of airplane in the tunnel.



(b) Sketch showing relative size of airplane and wind tunnel.

Figure 2.- Concluded.

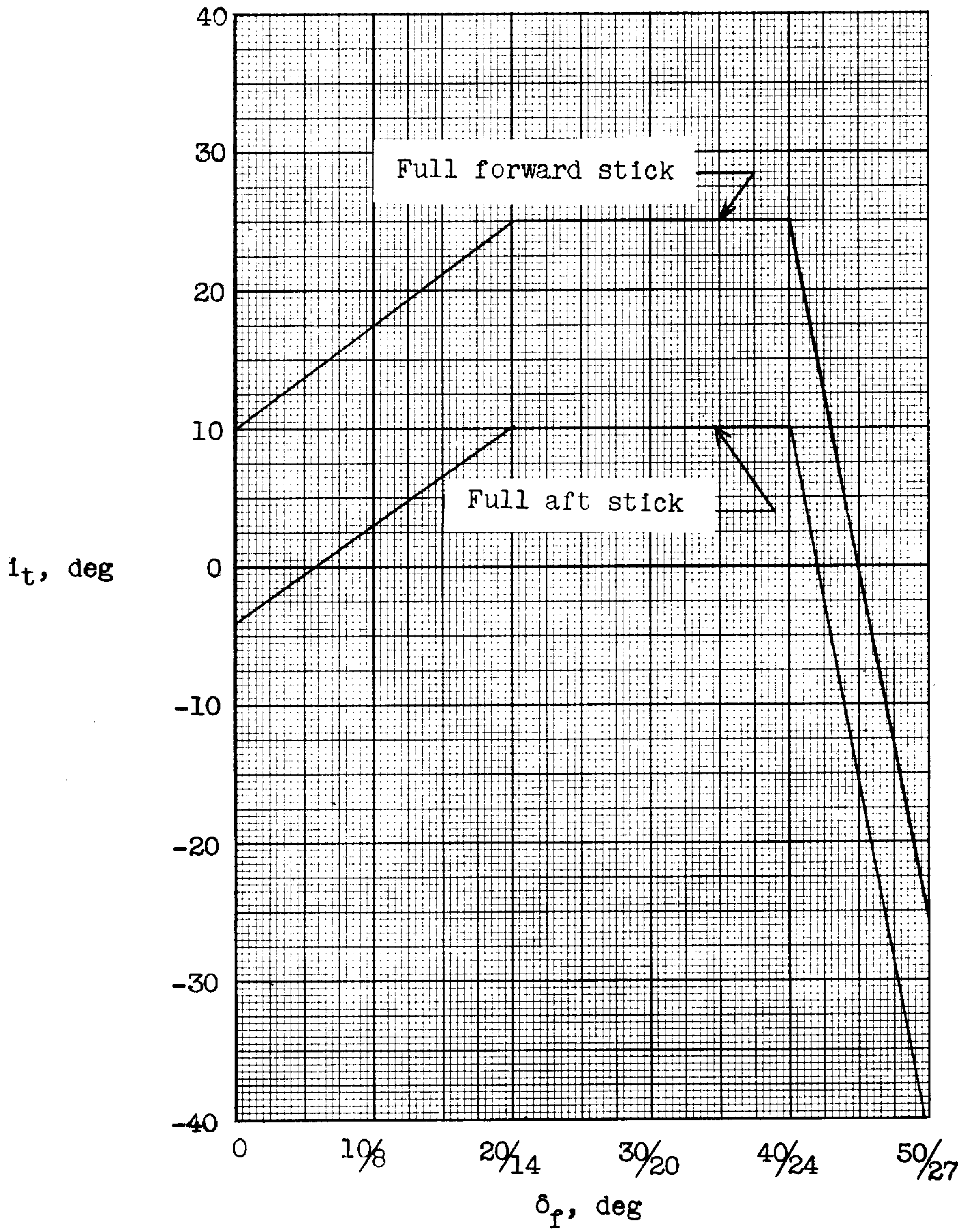


Figure 3.- Variation of tail incidence available for control with flap deflection.

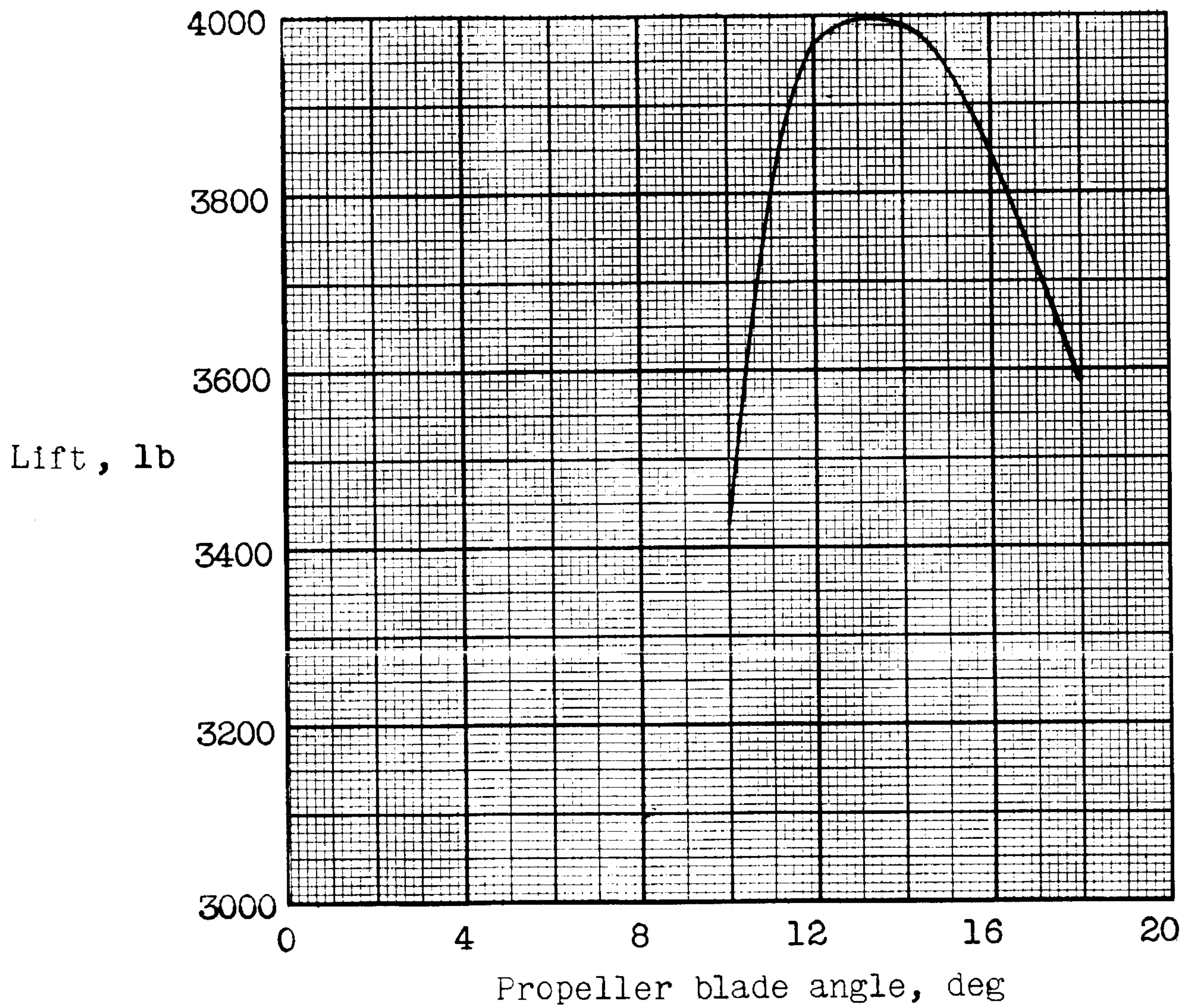


Figure 4.- Variation of lift force with propeller blade angle for collective pitch variation. $\alpha = 30^{\circ}$; $\delta_f = 50/27$.

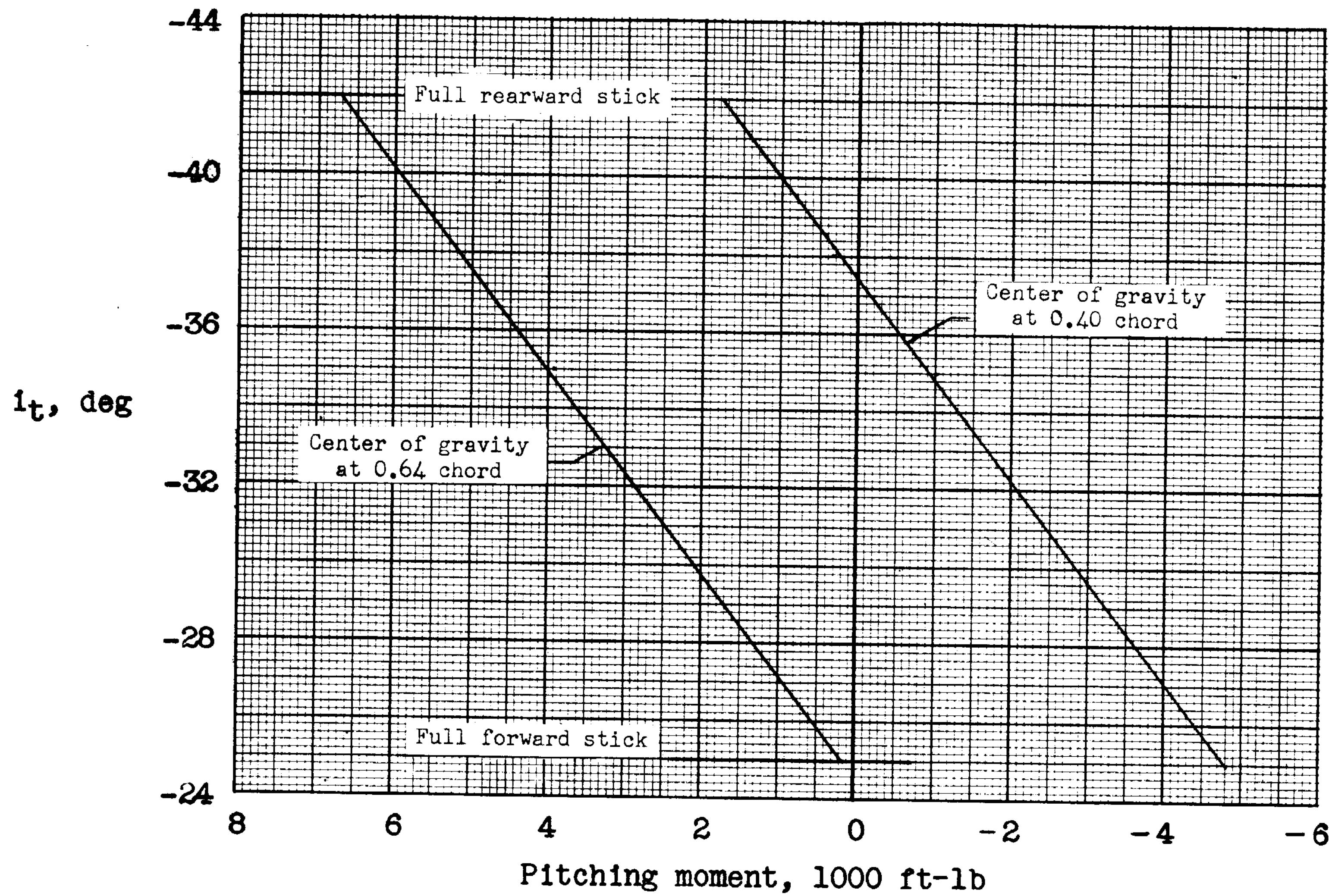


Figure 5.- Longitudinal control effectiveness in hovering.

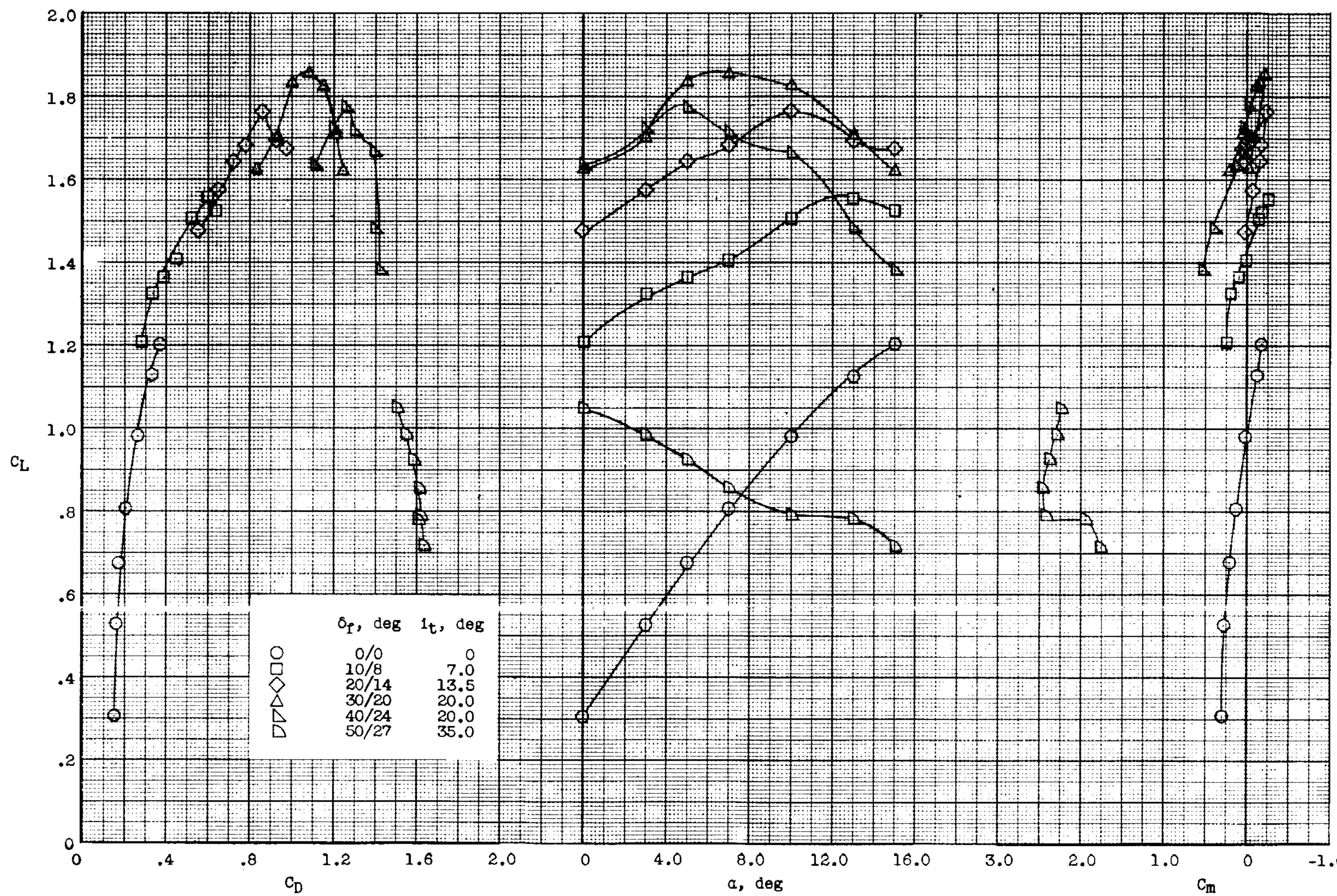
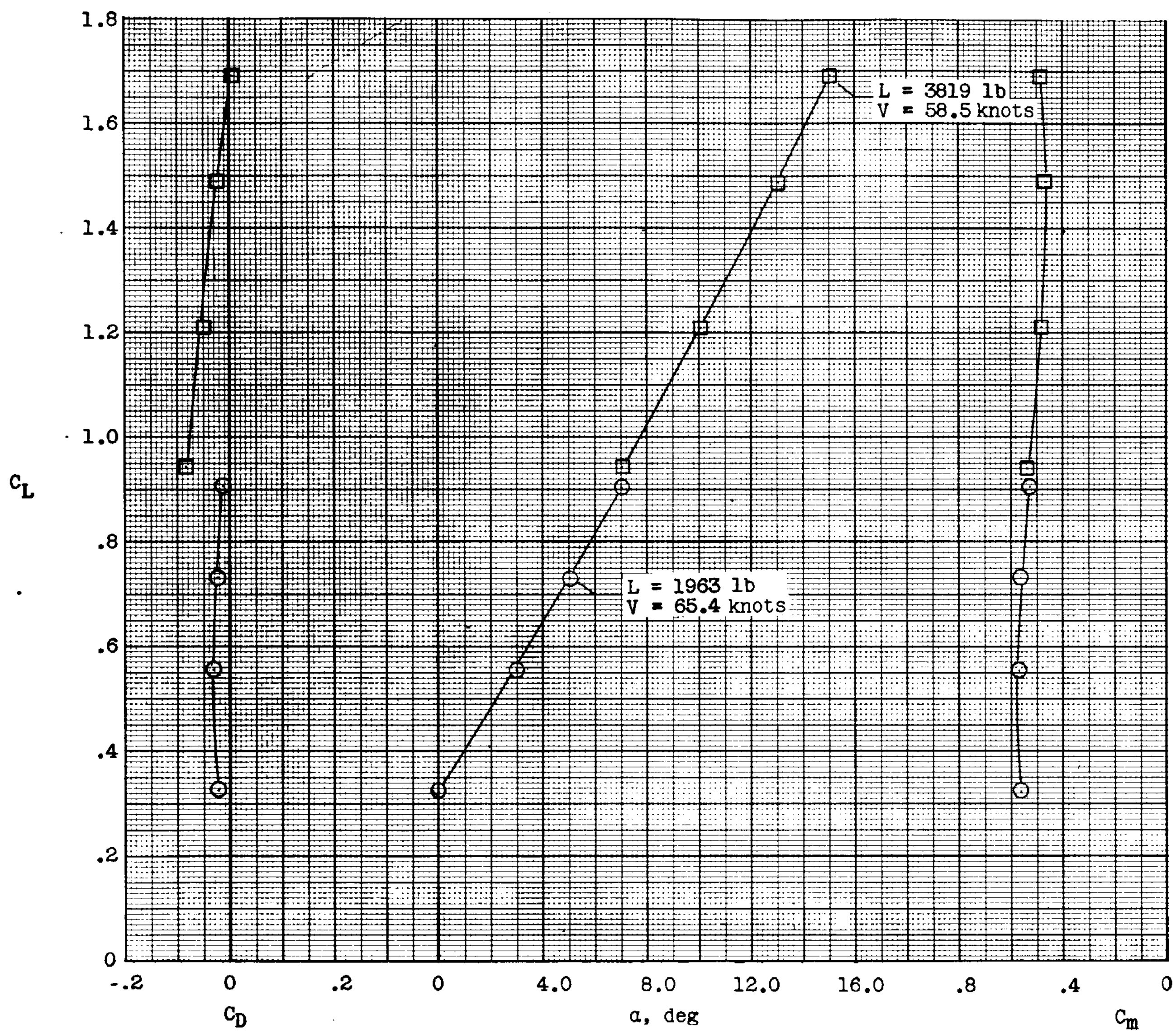
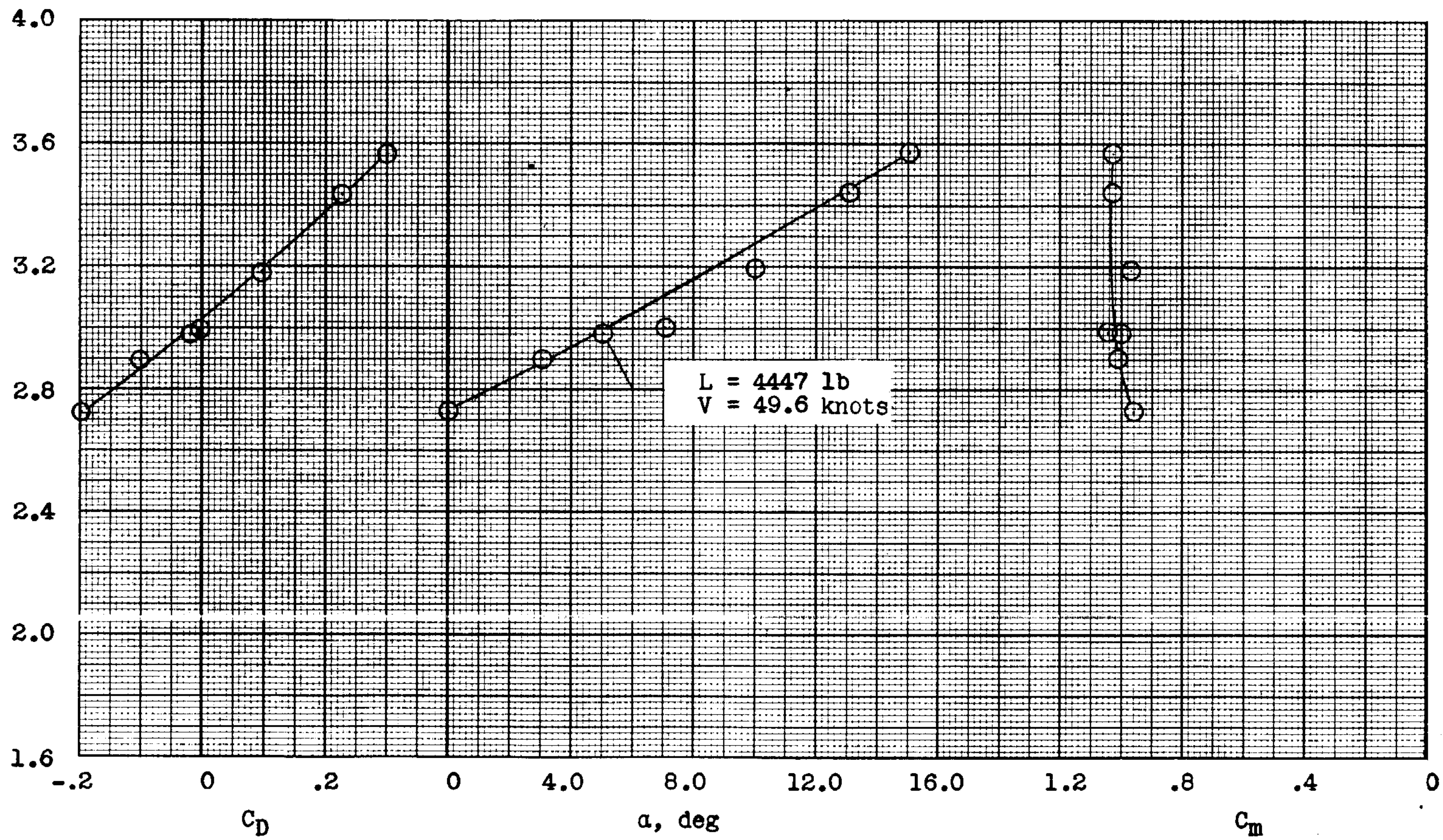


Figure 6.- Power of longitudinal aerodynamic characteristics of the airplane for the range of flap deflections with the propellers removed. Center of gravity at 0.40 chord.



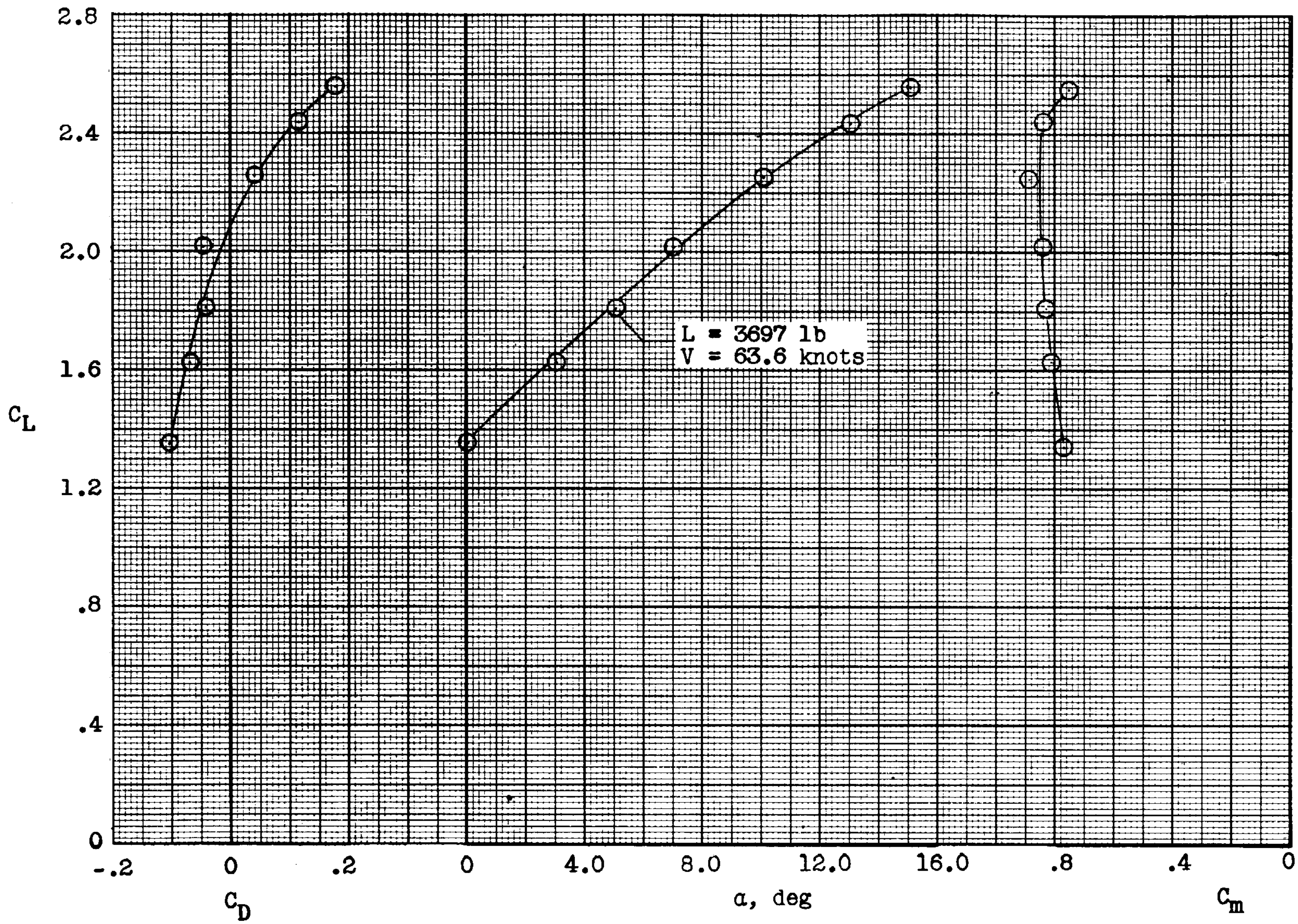
(a) $\delta_f = 0/0$; $i_t = 0^\circ$.

Figure 7.- Longitudinal aerodynamic characteristics of the airplane for the range of flap deflections with power to trim the various configurations at flying gross weight. Center of gravity at 0.64 chord.



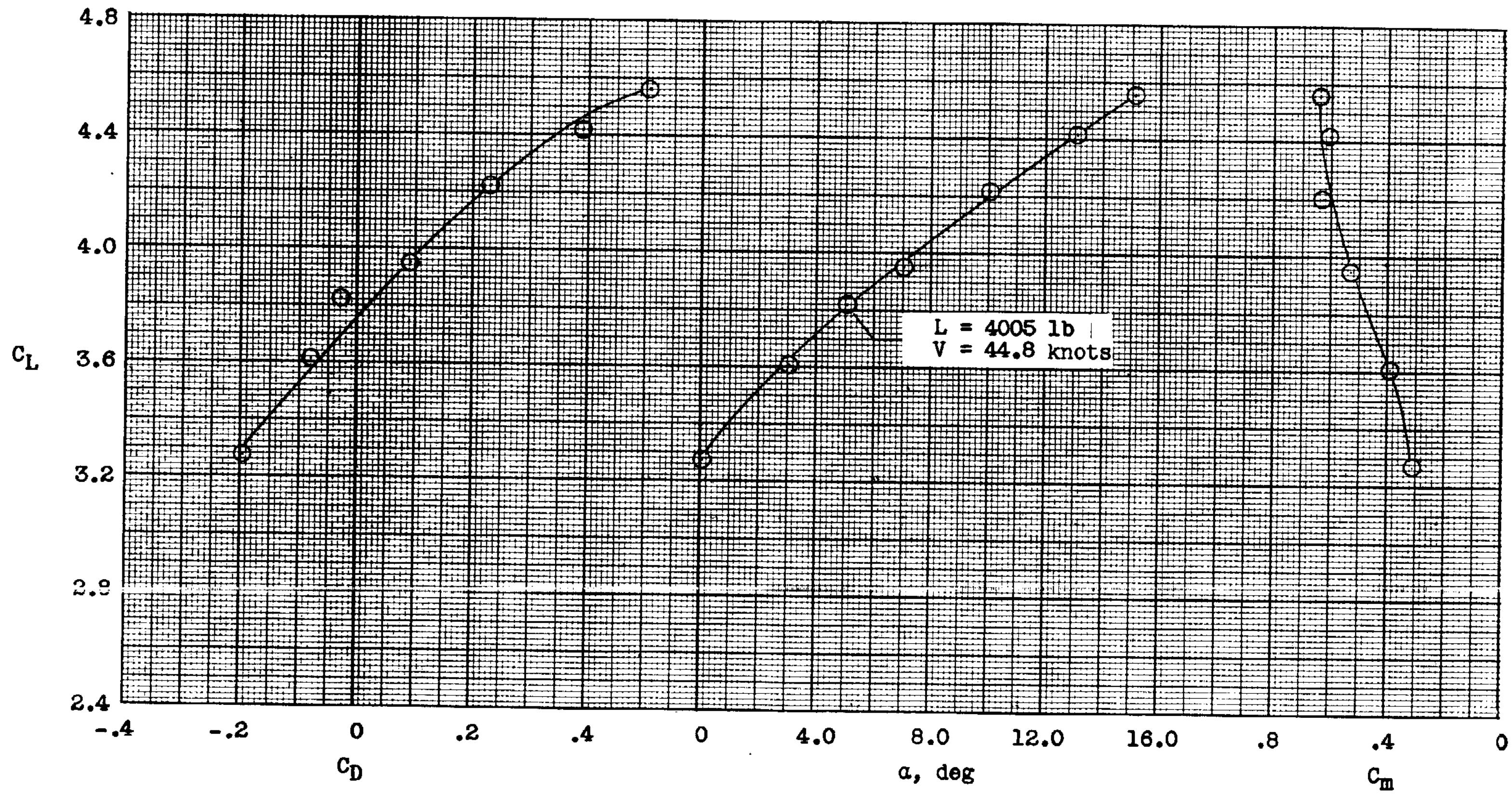
(b) $\delta_f = 10/8$; $i_t = 7.0^{\circ}$.

Figure 7.- Continued.



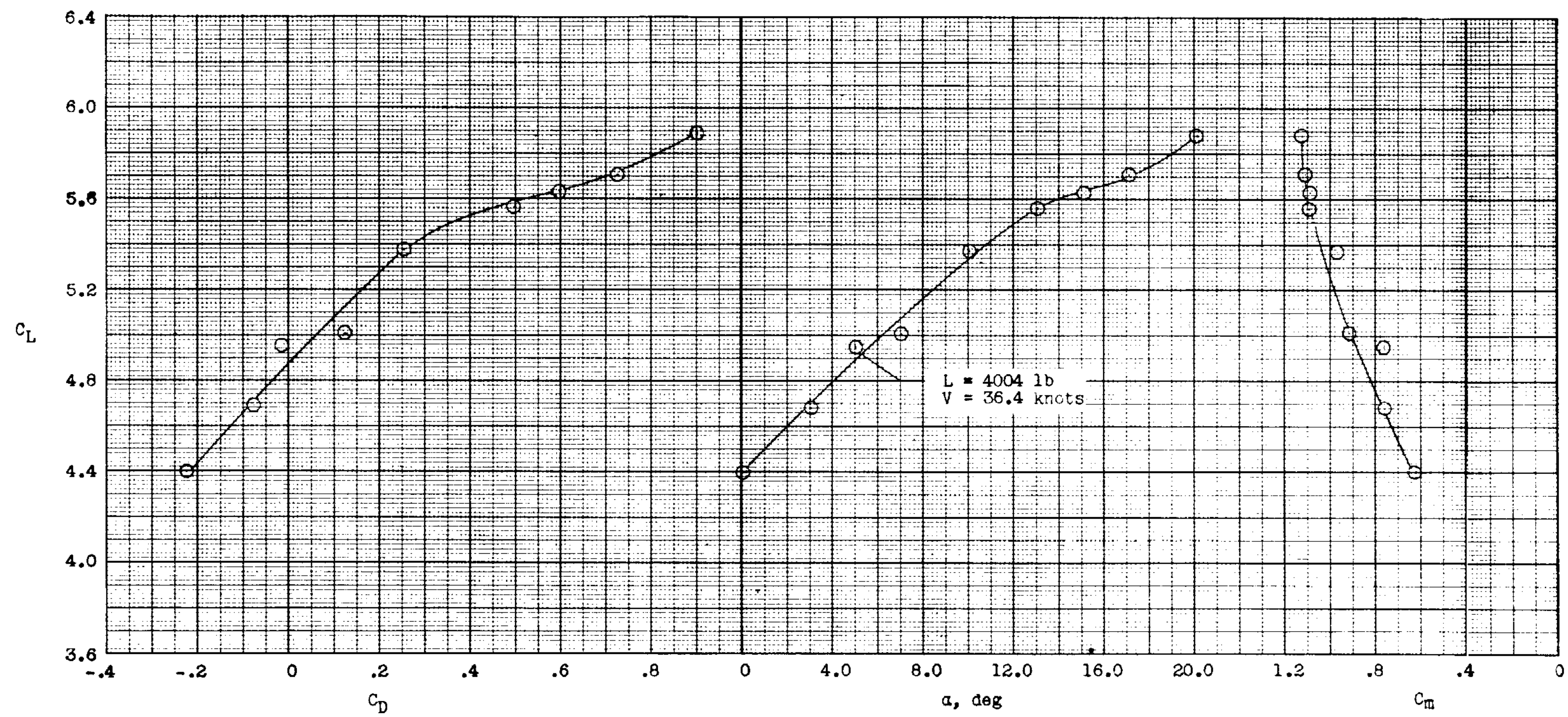
(c) $\delta_f = 20/14$; $i_t = 13.5^{\circ}$.

Figure 7.- Continued.



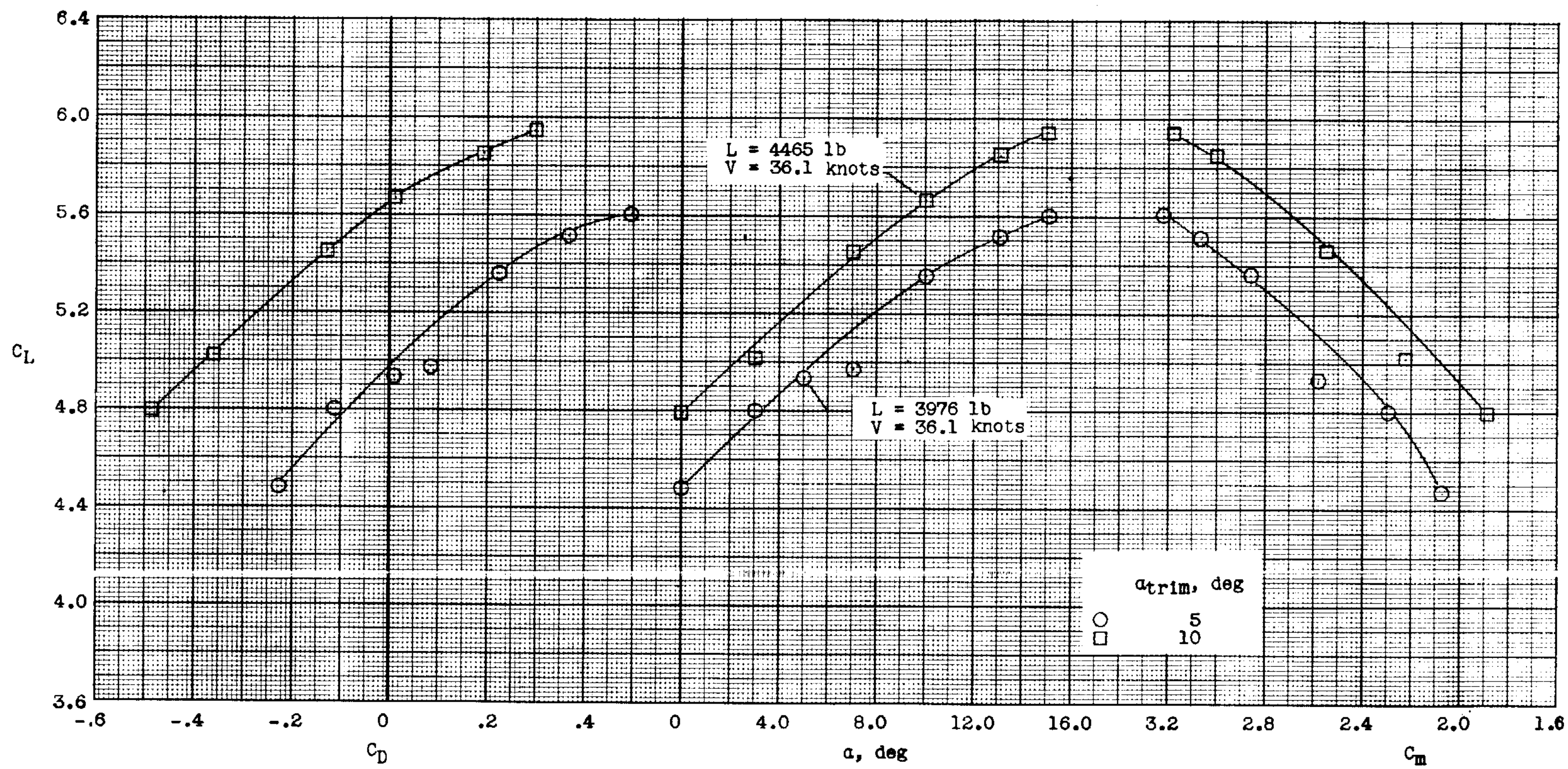
(d) $\delta_f = 30/20$; $i_t = 20.0^{\circ}$.

Figure 7.- Continued.



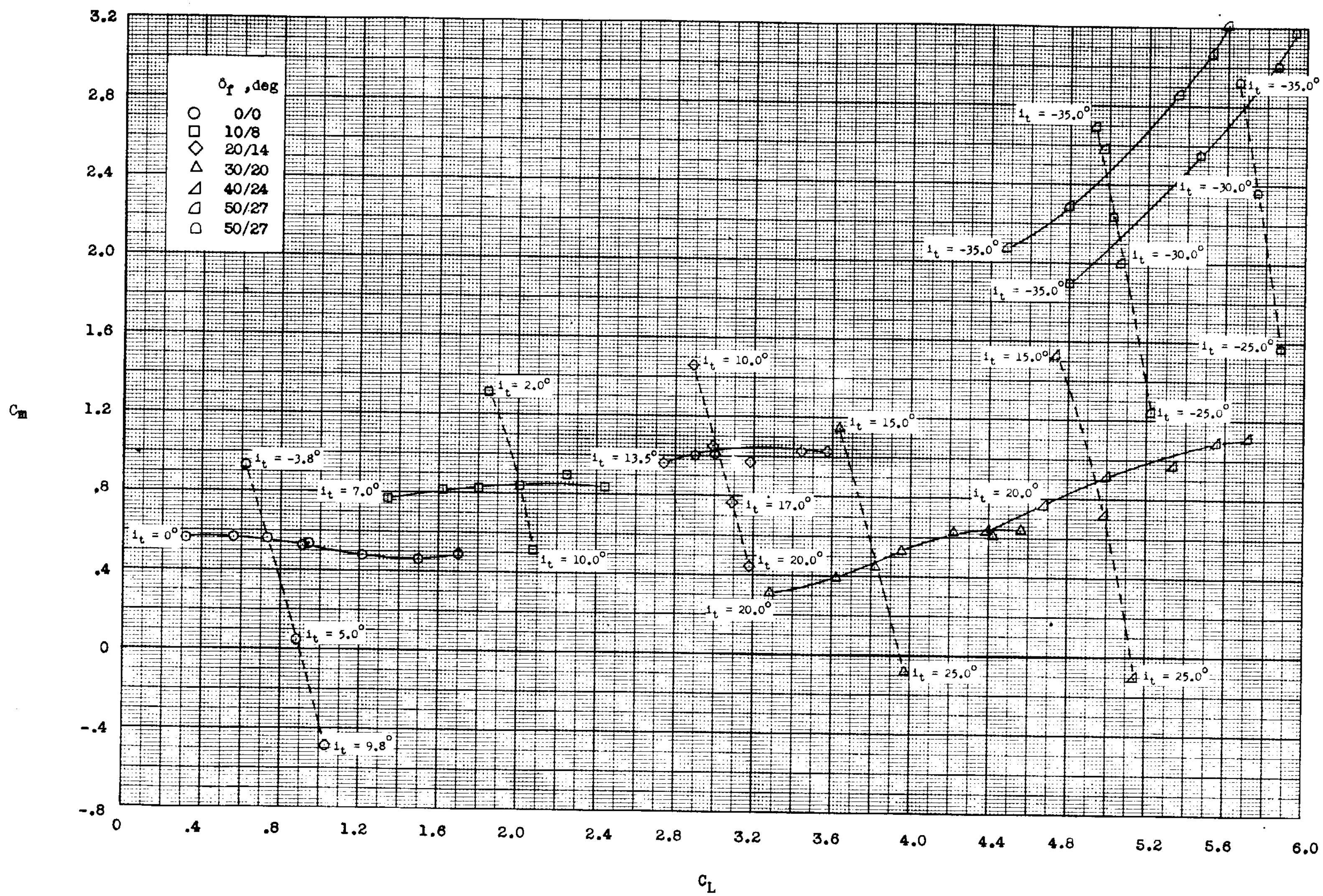
(e) $\delta_f = 40/20$; $i_t = 20.0^\circ$.

Figure 7.- Continued.



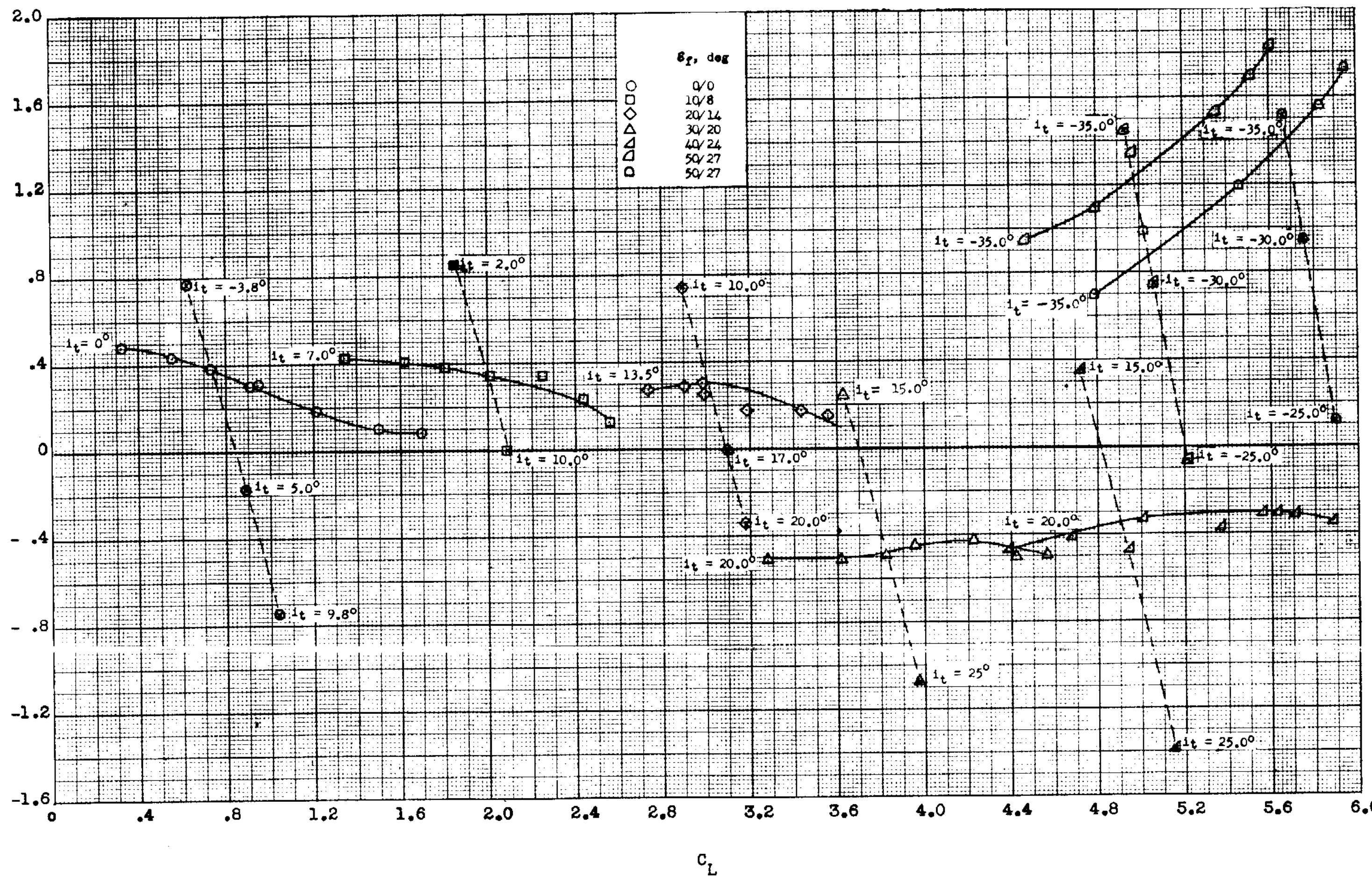
(f) $\delta_f = 50/27$; $i_t = -35.0^\circ$.

Figure 7.- Concluded.



(a) Center of gravity at 0.64 chord.

Figure 8.- Variation of pitching moment with lift coefficient and tail-incidence angle for various flap conditions.



(b) Center of gravity at 0.40 chord.

Figure 8.- Concluded.

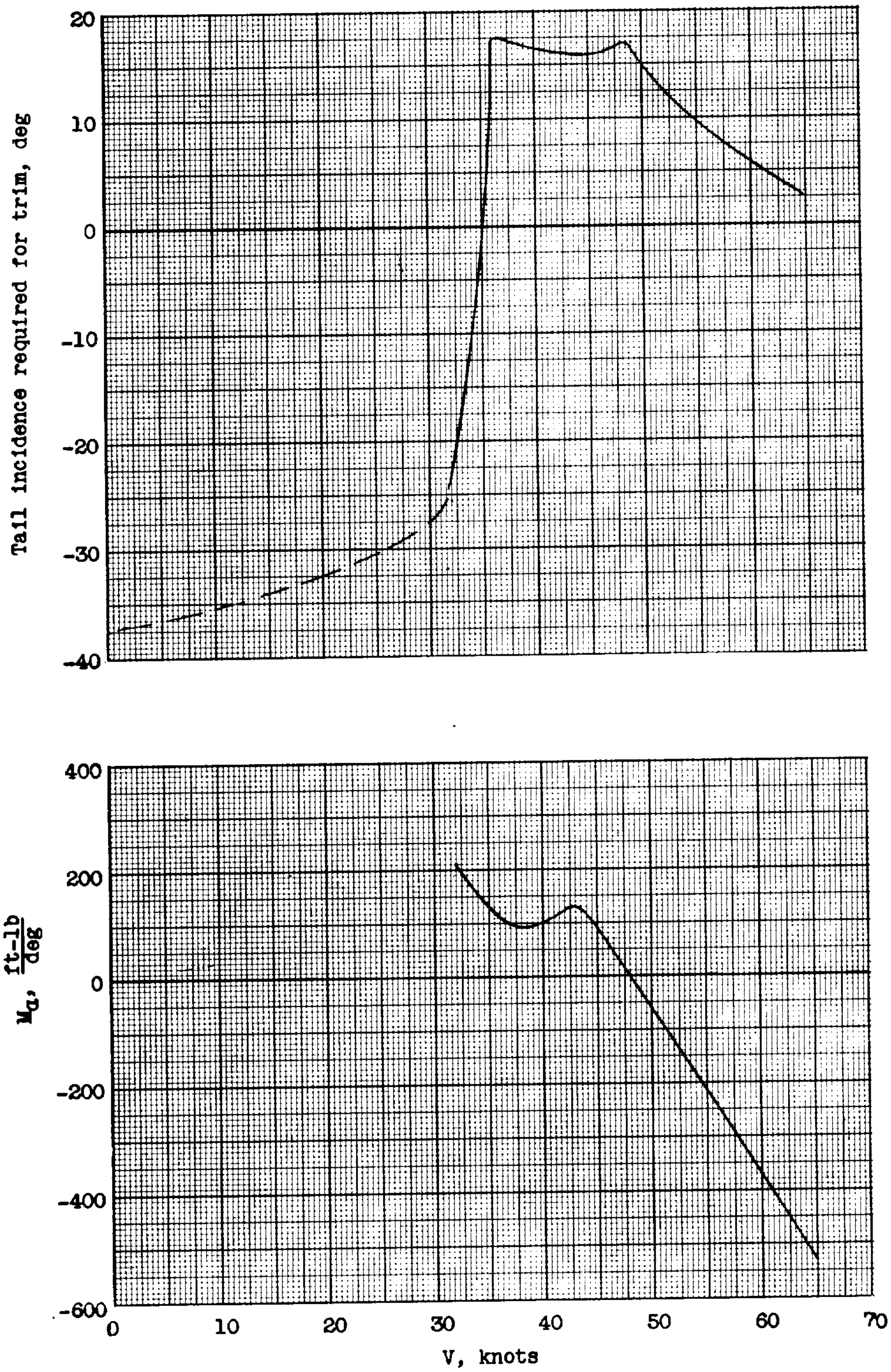
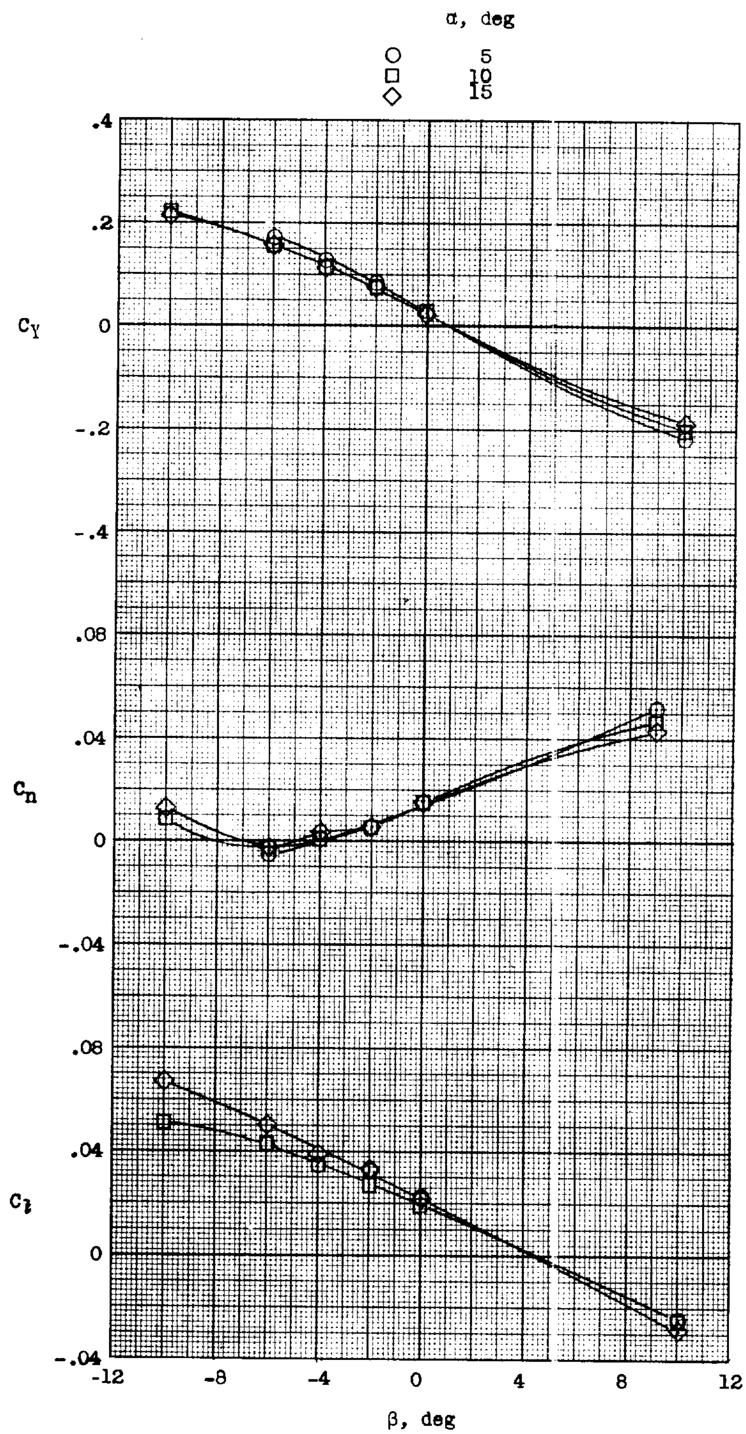
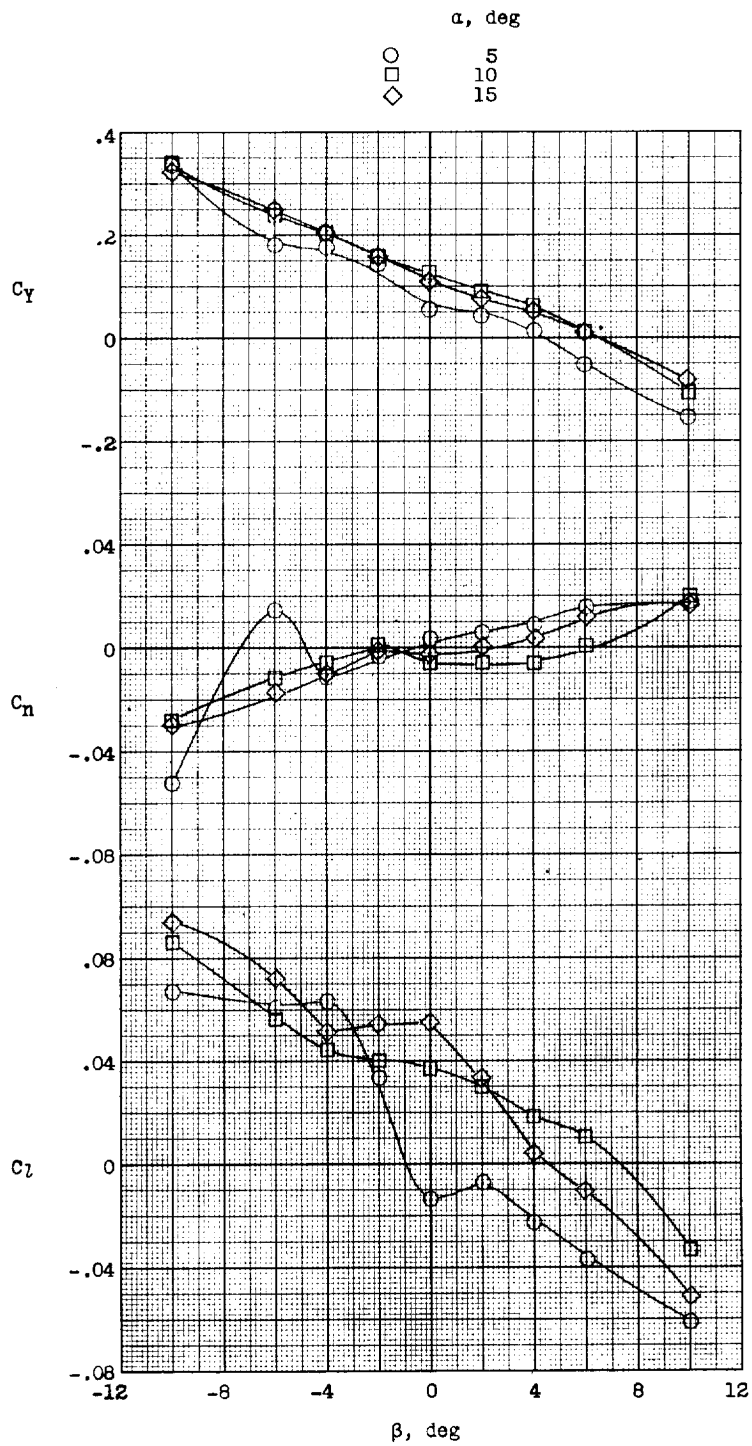


Figure 9.- Variation of tail incidence angle and longitudinal stability with forward velocity for center of gravity at 0.40 chord.



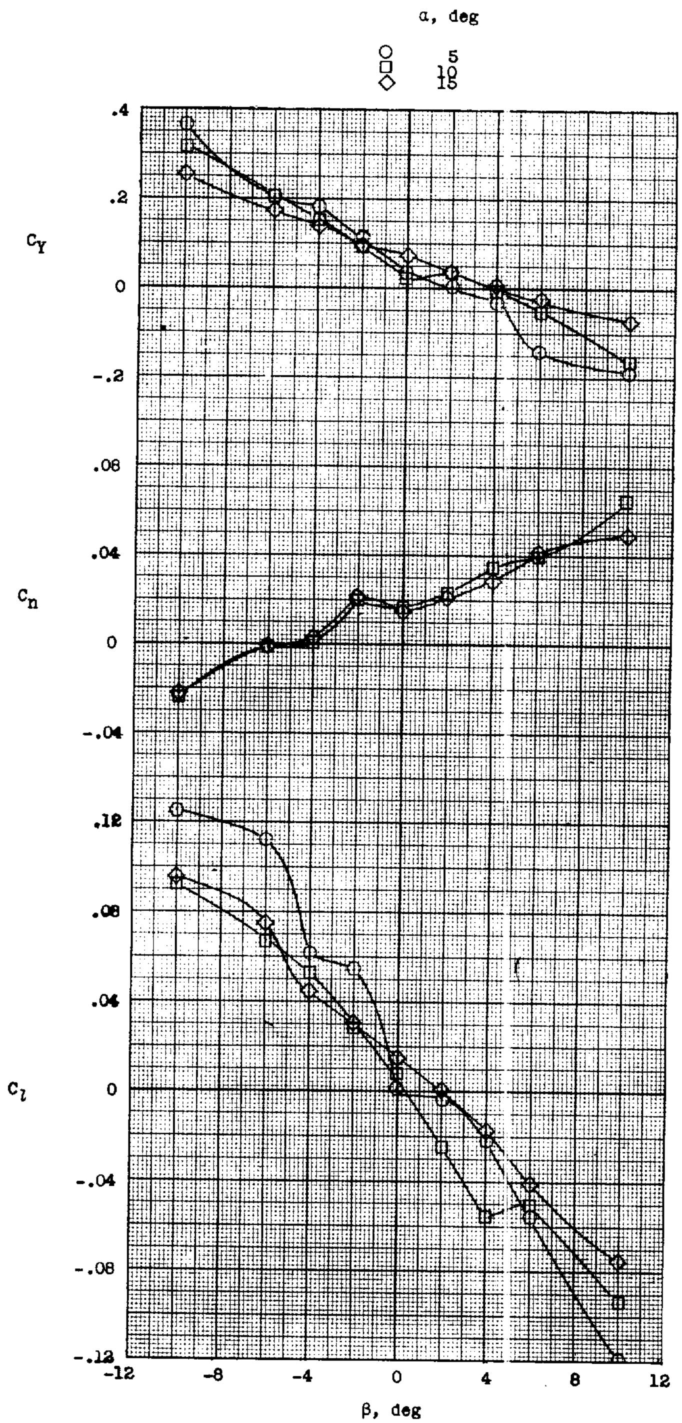
(a) $\delta_f = 0/0$; $i_t = 0^\circ$.

Figure 10.- Static lateral stability characteristics. $\delta_a = 0^\circ$; $\delta_r = 0^\circ$; center of gravity at 0.40 chord.



(b) $\delta_f = 20/14$; $i_t = 13.5^{\circ}$.

Figure 10.- Continued.



(c) $\delta_f = 30/20$; $i_t = 20.0^{\circ}$.

Figure 10.- Concluded.