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RESEARCH MEMORANDUM

AERODYNAMIC STUDY OF A WING-FUSELAGE COMBINATION EMPLOYING A WING SWEPT BACK 63° - EFFECT OF SIDESLIP ON AERODYNAMIC CHARACTERISTICS AT A MACH NUMBER OF 1.4 WITH THE WING TWISTED AND CAMBERED

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CLASSIFICATION CHANGED TO UNCLASSIFIED
AUTHORITY: NACA RESEARCH ABSTRACT NO. 114
EFFECTIVE DATE: APRIL 8, 1957

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

September 29, 1950

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AERODYNAMIC STUDY OF A WING-FUSELAGE COMBINATION EMPLOYING A WING SWEPT BACK 63° - EFFECT OF SIDESLIP ON AERODYNAMIC CHARACTERISTICS AT A MACH NUMBER OF 1.4 WITH THE WING TWISTED AND CAMBERED

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SUMMARY

The longitudinal and lateral characteristics of a wing-fuselage combination employing a wing with the leading edge swept back 63° and cambered and twisted for a uniform load at a lift coefficient of 0.25 and at a Mach number of 1.5 are presented. The investigation was carried out through a range of sideslip angles from -5° to +5° at a Mach number of 1.4 at Reynolds numbers of 1.5, 2.7, and 3.7 million. The experimental results are compared with those predicted by an approximate theory.

The results showed that the longitudinal characteristics were essentially unaffected by Reynolds number or the sideslip angles investigated. The model exhibited an unstable variation of rolling moment with angle of sideslip up to a lift coefficient of 0.08 due to the twist and camber incorporated in the wing. The model was directionally unstable at all lift coefficients. Fuselage-alone tests showed that the fuselage was the primary factor in causing the instability.

INTRODUCTION

A comprehensive research program has been undertaken at the Ames Laboratory to investigate the characteristics of a wing-fuselage combination employing a wing with the leading edge swept back 63°, a configuration which was selected on the basis of a theoretical study to obtain improved lift-drag ratios at a Mach number of 1.5 (reference 1). The results of the investigation (references 2 to 4) have indicated that, at moderate supersonic Mach numbers, lift-drag ratios greater than 10 may be obtained with such a wing-fuselage combination and that reasonably efficient flight is therefore possible.

On the basis of these results, the experimental research has been extended to include an investigation of the stability characteristics of the wing-fuselage combination. The longitudinal-stability characteristics are given in references 2 through 5. Low-speed lateral- and directional-stability characteristics are given in references 6 and 7. The present investigation, conducted in the Ames 6- by 6-foot supersonic wind tunnel, is concerned with the lateral-stability characteristics and the effect of sideslip on the longitudinal characteristics at a Mach number of 1.4.

COEFFICIENTS AND SYMBOLS

All force coefficients were computed along the wind axes, and the moment coefficients were computed about the stability axes with the origin located at the quarter-chord point of the mean aerodynamic chord projected to the fuselage center line. The axes are described pictorially in figure 1. All angles and force and moment coefficients are shown in the positive sense.

The data are presented in the form of standard NACA coefficients as follows:

b wing span measured perpendicular to the plane of symmetry, feet

c chord parallel to plane of symmetry, feet

$\bar{c}$ mean aerodynamic chord $\left(\dfrac{\int_0^{b/2} c^2\,dy}{\int_0^{b/2} c\,dy}\right)$, feet

C_L lift coefficient $\left(\dfrac{\text{lift}}{qS}\right)$

C_D drag coefficient $\left(\dfrac{\text{drag}}{qS}\right)$

C_m pitching-moment coefficient $\left(\dfrac{\text{pitching moment}}{qS\bar{c}}\right)$

C_l rolling-moment coefficient $\left(\dfrac{\text{rolling moment}}{qSb}\right)$

C_n yawing-moment coefficient $\left(\dfrac{\text{yawing moment}}{qSb}\right)$

C_Y cross-wind-force coefficient $\left(\dfrac{\text{cross-wind force}}{qS}\right)$

C_{L_α} rate of change of lift coefficient with angle of attack, measured at zero lift, per degree

C_{l_β} rate of change of rolling-moment coefficient with angle of sideslip, measured at constant angle of attack, per degree

C_{n_β} rate of change of yawing-moment coefficient with angle of sideslip, measured at constant angle of attack, per degree

M Mach number

q dynamic pressure, pounds per square foot

R Reynolds number, based on the mean aerodynamic chord

S wing area, including that portion enclosed by the fuselage as determined by extending the leading and trailing edges to the plane of symmetry, square feet

α angle of attack, degrees

β angle of sideslip, degrees

EXPERIMENTAL CONSIDERATIONS

Apparatus

Wind tunnel and balance.- The investigation was carried out in the Ames 6- by 6-foot supersonic wind tunnel, which has been described completely in reference 8. The balance used was a four-component strain-gage type mounted in the fuselage of the model. Each force and moment was measured by an individual strain gage supported by ball bearings to reduce to a minimum any interaction between forces and moments. It has been found that the friction and interaction present are negligible.

Model and support.- The dimensions of the model are shown in figure 2, and curves showing the nonlinear variation of twist and camber over the span of the wing to obtain a uniform load are shown in figure 3. The streamwise airfoil sections of the 63° swept-back wing were NACA 64A005 sections, and the aspect ratio and taper ratio of the wing, including that portion enclosed by the fuselage, was 3.5 and 0.25, respectively. The dihedral angle, as measured between the leading edge and the horizontal plane, was zero. The model was constructed of steel, painted and sanded to obtain a smooth finish, and was mounted for testing on a sting-type support system as shown in figure 4.

Test Methods

Surveys of the stream in the test section of the tunnel (reference 8) have shown that at all Mach numbers the cross flow is negligible. However, significant variations of stream inclination and curvature occur in the vertical and axial directions at Mach numbers greater or less than 1.4.

With the exception of stream inclination, which is essentially constant over the test section, the stream variations are negligible at a Mach number of 1.4 and it has been found possible to obtain reliable lateral data with the wing in the horizontal plane. Therefore, the investigation was conducted at a Mach number of 1.4 with variable angle of attack at a constant angle of sideslip.

Aerodynamic forces and moments.– The quantities measured were normal, chord, and side forces, and pitching, rolling, and yawing moments. Use of the four-component balance necessitated running at each desired test condition twice in order that all forces and moments could be measured. The two sets of data for each condition were then combined and resolved to the wind and stability axes to obtain the final coefficients as presented.

Angle of attack.– The angle of attack, continuously variable during test, was computed from measurements of the relative vertical positions of two points on the fuselage. The measurements were taken during running at each test condition with a cathetometer. Direct reading of the measurements eliminated any correction due to the deflection of the sting support under aerodynamic load.

Angle of sideslip.– Sideslip angles, constant during each run, were obtained by using a sting bent to an angle of 5° and rotated to predetermined angles in the support body. Thus, when the bent portion of the sting was rotated 90° from the vertical plane of the tunnel, the model, which was maintained in a horizontal plane, was at an angle of sideslip of 5°; when the bent portion of the sting was in the vertical plane, the model was at zero sideslip angle. Intermediate angles were obtained by varying the degree of rotation.

Sideslip angles could not be measured directly during the test. The relative lateral positions of the nose and base of the fuselage were measured with the wind off and the angle was then computed. The change in sideslip due to deflection of the sting under aerodynamic load was found to be negligible.

Corrections to the Data

Angle of attack.– Because of the stream inclination, it was necessary to correct the angle of attack as computed from the cathetometer measurements. The correction was taken as one-half the difference in angles of zero lift for the inverted and upright positions of the model, which gave an integrated value of the effective angle of inclination of the stream. This correction (0.6°) was added to and subtracted from the measured angle of attack for the inverted and upright runs, respectively.

Pitching moment.— The pitching-moment curves were corrected in a manner analogous to that for the angle of attack, the correction being obtained from the difference in pitching-moment coefficient at zero lift for the inverted and upright runs.

Drag.— It was shown in reference 9 that, with an unseparated boundary layer over the fuselage and for the sting-to-base-diameter ratio used (0.93), the effect of support interference is confined to the base pressure alone. Liquid-film studies were attempted in this test, but no conclusion could be drawn regarding the nature of the boundary layer. It was found in reference 2, however, that, at a Reynolds number of 0.62 million, the boundary layer was turbulent over the rear of the fuselage and remained unseparated to the fuselage base. It is believed that at the relatively high Reynolds numbers of the present test a similar condition exists. Accordingly, the base pressure was measured for each test condition, and the drag force was corrected for the difference between the measured pressure and the static pressure of the free stream at the fuselage base.

The drag correction due to the longitudinal static-pressure gradient in the test section, shown in reference 8, was calculated and found to be negligible.

Precision of the Data

The precision of the data may be computed from the uncertainty of all quantities which enter into the final data. The uncertainty in each quantity was taken as one of the following:

1. The least reading of the instrument for quantities which were steady during reading.

2. One-half the magnitude of the fluctuation of quantities which were not steady during reading.

3. The magnitude of the variation from the arithmetic mean of quantities which varied during the period of the investigation.

The precision of the data was taken as being equal to the square root of the sum of the squares of all uncertainties entering into the final quantity. The test conditions for the tabulated quantities were: angle of attack, 4.4°; angle of sideslip, 5°; and Reynolds number, 1.5 million. The precision of the data for all other test conditions is within the limits given here.

The final uncertainty in each quantity is as follows:

Mach number	± 0.014
Reynolds number	$\pm 0.03 \times 10^6$
Angle of attack	$\pm 0.05^\circ$

Angle of sideslip	$\pm 0.01^\circ$
Lift coefficient	± 0.0028
Pitching-moment coefficient	± 0.001
Drag coefficient	± 0.0006
Rolling-moment coefficient	± 0.0008
Yawing-moment coefficient	± 0.0001
Cross-wind-force coefficient	± 0.0005

RESULTS AND DISCUSSION

Experimental Results

Longitudinal characteristics.- The lift, drag, and pitching-moment characteristics for the 63° swept-back wing model are shown in figure 5 for the three Reynolds numbers and positive range of sideslip angles. The results indicate that neither Reynolds number nor sideslip angle had any significant effect on the lift characteristics, a value for the lift-curve slope of 0.0515 being obtained throughout the test. The drag results also were unaffected to any important extent by Reynolds number in the range of the test. However, the minimum drag coefficient increased 0.0025 with increasing sideslip angle from 0° to 5°, with the greatest change taking place between 3° and 5°. The slope of the pitching-moment curve was unaffected by sideslip angle, but increased -0.013 with an increase of Reynolds number from 1.5 to 2.7 million, with no further change caused by an additional increase of Reynolds number to 3.7 million.

Lateral characteristics.- The lateral characteristics for the three Reynolds numbers and total range of sideslip angles are shown in figure 6. The cross-plotted data, discussed in detail later, show that the lift coefficient required to obtain a neutral variation of rolling moment with sideslip angle was approximately 0.08.

The extension of experimental values of rolling moment due to sideslip obtained from swept-back-wing models to full-scale aircraft may be questioned due to the large variance of structural rigidity. No experimental data are available for the determination of the effect of varying structural rigidity of a swept-back wing; it is possible, however, that an appreciable change in lateral stability may result and an investigation of such effects should be made.

Directional characteristics.- The variation of the directional-stability derivative C_{n_β} with lift coefficient for the three Reynolds numbers is shown in figure 7 for the wing-fuselage combination and the fuselage alone. As can be seen, the fuselage was the primary cause of the directional instability of the model. The increase in directional stability with lift coefficient for the wing-fuselage combination results from the fact that the variation of the yawing moment of a swept-back wing with sideslip is primarily a function of the drag due to lift of the wing. The directional-stability derivative of a swept-back wing alone tends to vary as the square of the lift coefficient, therefore, and is positive since the lift, and thus the drag due to lift, is greater on the leading wing panel.

Comparison of Theory and Experiment

The lateral characteristics of the swept wing under consideration at supersonic speed may be determined theoretically by a method similar to that used in reference 10, which is based on a lift cancellation process. The cancellation process involves the superposition of a number of constant load conical-flow sectors in the solution for an infinite triangular wing in sideslip (reference 11). The regions involved in the superposition of the conical flows are those encompassed by the Mach cone generated at the tip leading edge and the region ahead of the trailing edge encompassed by the Mach cone generated at the trailing-edge apex. The two regions, designated 1 and 2, are shown in figure 8 for a Mach number of 1.4 and zero sideslip angle. It is possible to obtain a rigorous solution for the flow in these regions by resorting to graphical integrations, but the work involved is prohibitive.

In an effort to simplify the problem for the present paper, certain approximations were made to eliminate the graphical integrations. The first assumption made was that over the region of the wing encompassed by the tip Mach cone no lift exists, a condition which can be seen in reference 10 to be a fair approximation. The second assumption made (fig. 8) was that the pressure over the region of the wing encompassed by the Mach cone from the trailing-edge apex varied linearly from the value that exists just ahead of the trailing-edge Mach cone to zero at the subsonic trailing edge.

The theoretical characteristics presented herein do not consider the effects of the elastic deformation of the model during test, the aerodynamic forces on the fuselage, nor the wing-fuselage interference.

Longitudinal characteristics.- The effects of sideslip angle on the slopes of the lift and pitching-moment curves as determined experimentally and as calculated by the approximate theory for the wing alone are shown in figure 9. As predicted by the theory, there was no appreciable effect of the sideslip on either characteristic. From the fair agreement with the experimental values it appears reasonable to use the theory as a quick method of obtaining the order of magnitude of the derivatives.

Effective-dihedral derivative C_{l_β}.- In figure 10, values of the parameter C_{l_β}, expressing the rate of change of rolling moment due to sideslip as computed by the approximate theory and determined experimentally for the three Reynolds numbers, are compared. The computed curve has a value of zero for the derivative C_{l_β} at a lift coefficient of zero as the effects of twist and camber were not included in the calculations. For lift coefficients below approximately 0.08, the variation of rolling moment with sideslip angle for the test wing was unstable.

It can be shown that the instability at the low lift coefficients is associated with the wing twist and camber. For a swept-back wing in sideslip, the total rolling moment may be considered as being primarily affected by the following:

Sweepback
Dihedral
Twist
Camber

It is shown in reference 12 that sweepback produces a positive dihedral effect, and that the rolling moment for a given angle of sideslip is directly proportional to the lift produced by the wing.

Positive geometric dihedral, of course, produces a variation of rolling moment with sideslip which is stable because of the differential in angle of attack of the two wing panels when the wing is sideslipping.

The manner in which twist and camber affect the rolling moment may also be visualized using the argument of reference 12. Consider first a swept-back wing in sideslip with no dihedral, symmetrical airfoil sections normal to the wing leading edge, but with negative twist (tip angle of attack less than root angle of attack), operating at zero lift. For the lift to be zero, the root sections must be at a positive angle of attack. Denoting the sweepback angle and sideslip angle as Λ and β, the positive and negative lifts on the leading panel of the swept-back wing in sideslip are increased by the factor $\dfrac{\cos^2(\Lambda-\beta)}{\cos^2\Lambda}$ due to the change in the magnitude of the velocity component normal to the leading edge, and those on the other panel are decreased correspondingly by the factor $\dfrac{\cos^2(\Lambda+\beta)}{\cos^2\Lambda}$. Since the outboard sections of the wing are more effective in producing a rolling moment due to the larger moment arm, the resulting rolling moment must be unstable.

A similar effect is caused by camber. Consider a swept-back wing with no dihedral, no twist, but with airfoil sections normal to the leading edge so cambered that their pitching moment at zero lift is negative, such as on the test model. In sideslip, the moment about the leading edge of the forward panel must increase by the factor $\dfrac{\cos^2(\Lambda-\beta)}{\cos^2\Lambda}$ as before, and that for the other must decrease by the factor $\dfrac{\cos^2(\Lambda+\beta)}{\cos^2\Lambda}$. As a consequence of the sweepback there is a component of this moment acting about the plane of symmetry to produce a rolling moment. The resulting rolling moment is the difference between these two components, and, as the moment due to the leading panel predominates, the resulting rolling moment will be unstable.

As can be seen from the preceding discussion, the unstable variation of rolling moment with sideslip angle at the low lift coefficients for the test wing may be attributed to the destabilizing effect of the twist and camber predominating over the stable effect of the sweepback. This effect is beneficial because, as discussed previously, sweepback causes a variation of the rolling moment with lift coefficient at angles of sideslip, a characteristic which tends to produce excessive values of effective dihedral at moderate lift coefficients. It can be seen that, in the range of lift coefficients for which there is no appreciable tip stall for an untwisted swept-back wing using symmetrical airfoil sections, twisting and cambering the wing will reduce the effective dihedral while simultaneously increasing the efficiency of flight as reported in reference 4. In the higher lift range it might be expected that, due to delayed flow separation at the tips, the value of C_{l_β} for the twisted and cambered wing would exceed that for the flat wing at the same lift coefficient.

It should be noted that the manner in which the absolute value of C_{l_β} is affected by the twist and camber is similar to the effect of a negative geometric dihedral angle. However, the rate of change of C_{l_β} with lift coefficient should be a function of the plan form only. That this is essentially true can be seen from figure 11, where the derivative dC_{l_β}/dC_L, shown as a function of Mach number as computed by the approximate theory, is compared with the experimental values obtained from figure 10. At the Mach number for which the flow components perpendicular to the trailing edges become supersonic, the theoretical results indicate a fairly rapid decrease in the effective dihedral. At a Mach number of 1.4, the fair agreement with the experimental values indicates that the magnitude of the derivative may be estimated using the approximate theory.

<u>Directional-stability derivative C_{n_β}</u>.- The variation of the directional-stability derivative C_{n_β} with lift coefficient as determined experimentally and computed for the wing alone are compared in figure 7. The calculated curve was determined from the difference in the drag due to lift on each panel of the wing in sideslip and includes the effect of leading-edge suction as determined from the expression derived in reference 13. The variation of the parameter with lift coefficient agrees fairly well with that determined experimentally. The computed value of C_{n_β} has the value zero at zero lift coefficient as the effect of the twist and camber and the fuselage were not considered.

CONCLUSIONS

An investigation has been made to determine the aerodynamic characteristics in sideslip of a wing-fuselage combination employing a wing with the leading edge swept back 63° and cambered and twisted for a uniform load at a lift coefficient of 0.25 and at a Mach number of 1.5. The investigation, carried out through a range of sideslip angles from -5° to +5° at a Mach number of 1.4 and Reynolds numbers of 1.5, 2.7, and 3.7 million showed the following:

1. The longitudinal characteristics were essentially unaffected by the sideslip angles investigated.

2. Reynolds number had no appreciable effect on the longitudinal characteristics for the range investigated.

3. As a consequence of the twist and of the camber incorporated in the wing, the variation of rolling moment with sideslip angle was unstable up to a lift coefficient of approximately 0.08.

4. The model was directionally unstable at all lift coefficients. Fuselage-alone tests showed that the fuselage was the primary factor in causing the instability.

5. The fair agreement between experiment and the approximate theory indicates that the approximate theory may be used as a quick method of obtaining the order of magnitude of the aerodynamic characteristics.

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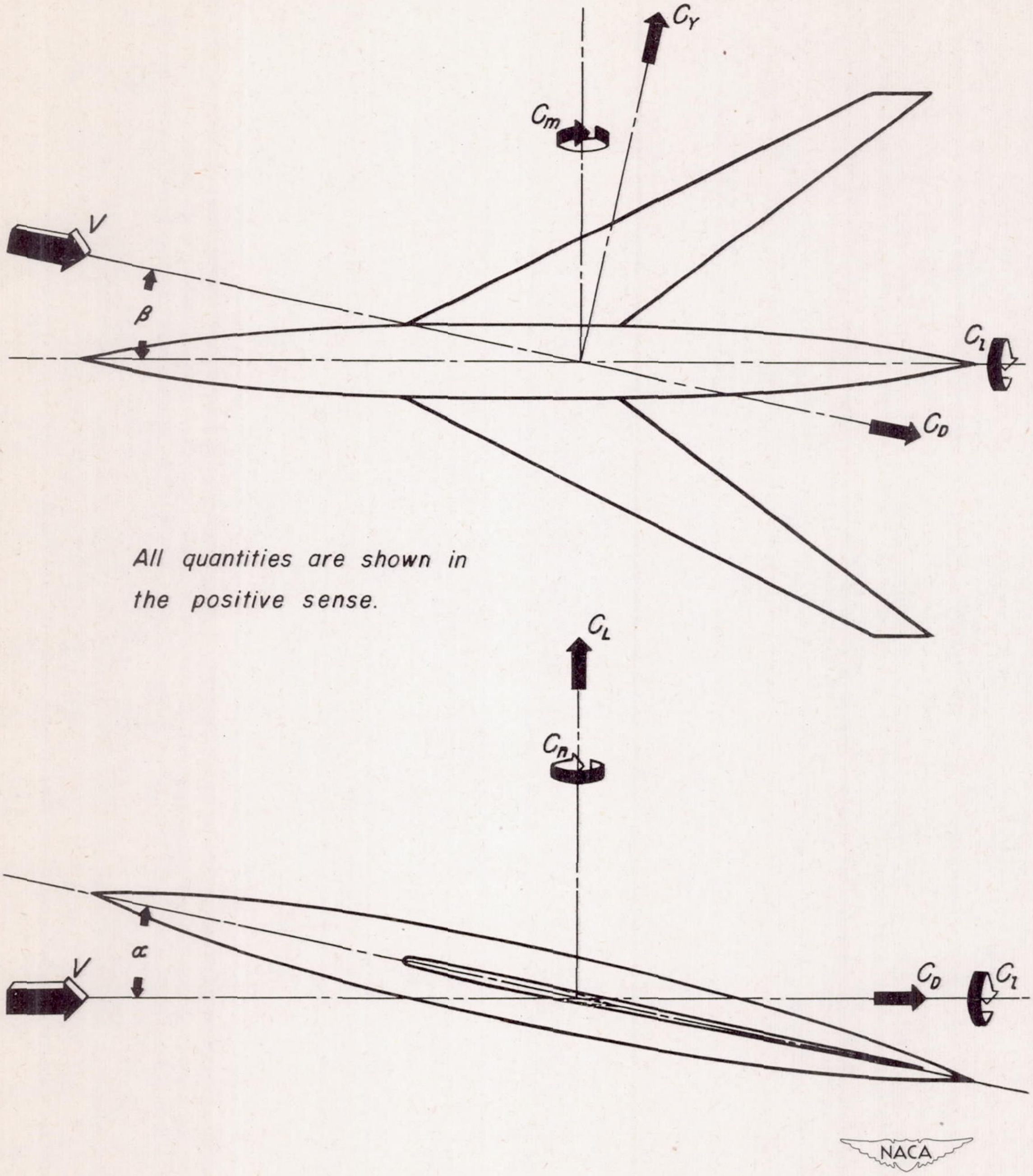

Figure 1.– Standard NACA sign convention.

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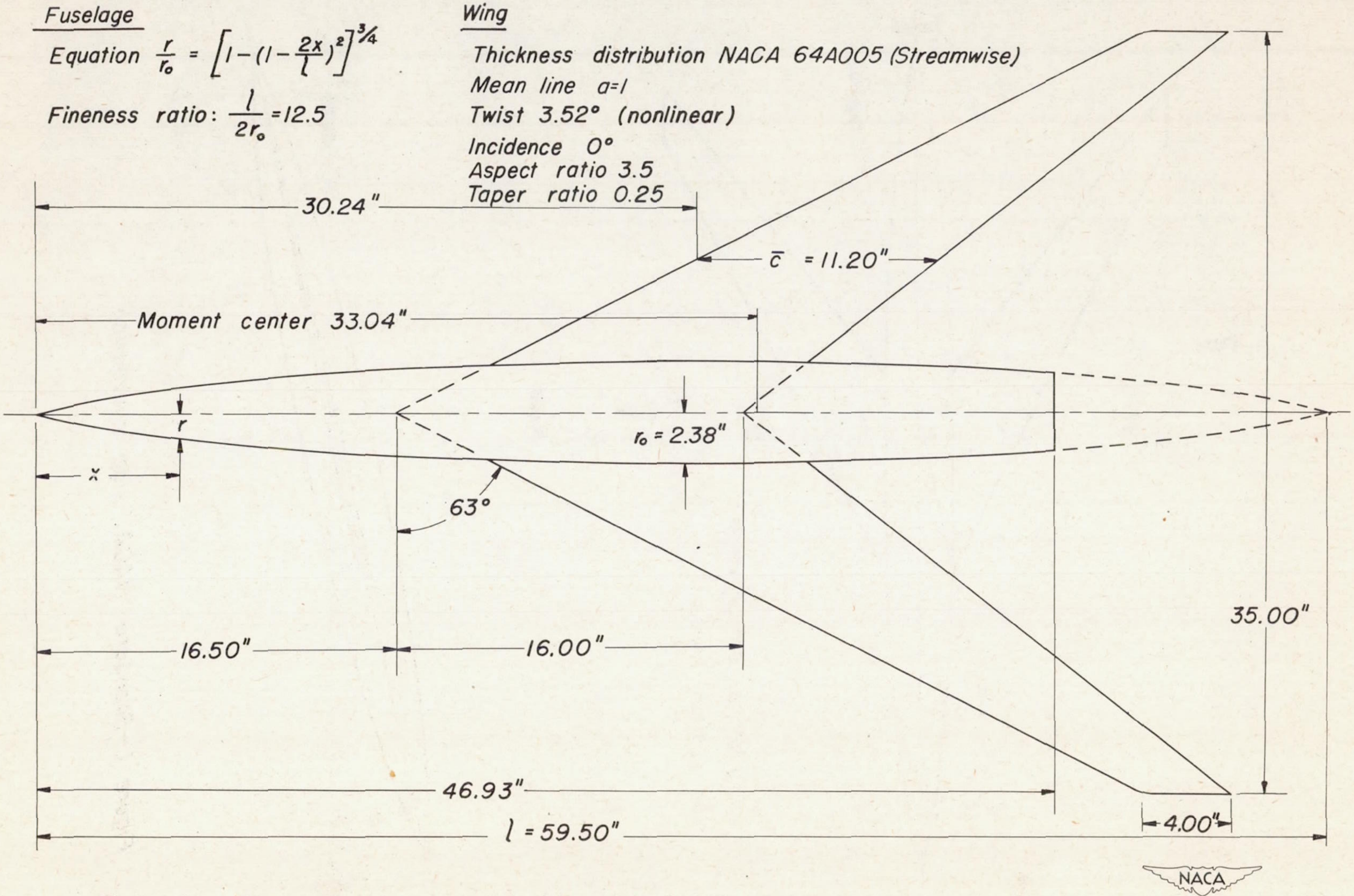

Figure 2.– Dimensions of the twisted and cambered 63° swept-back-wing model.

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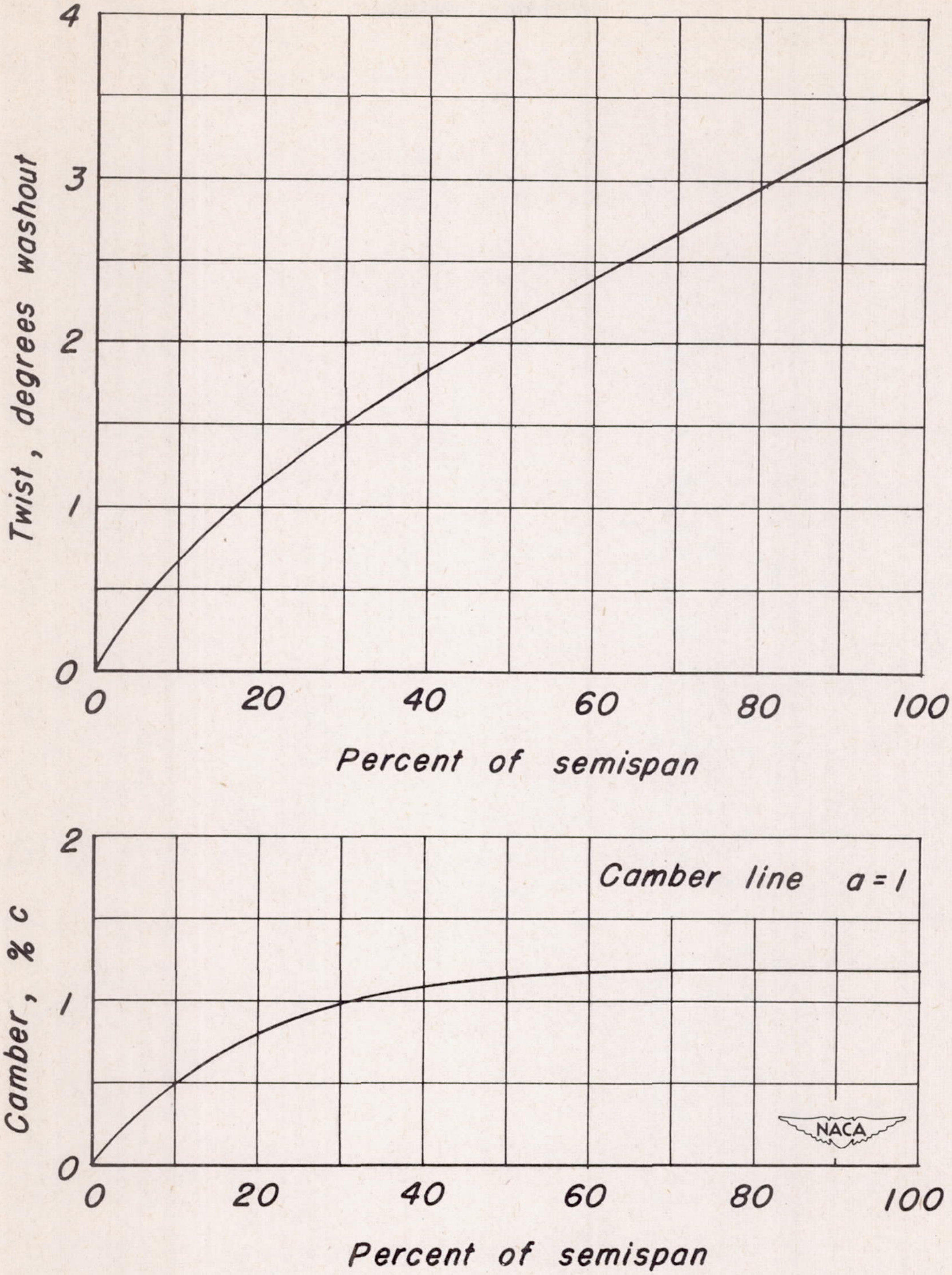
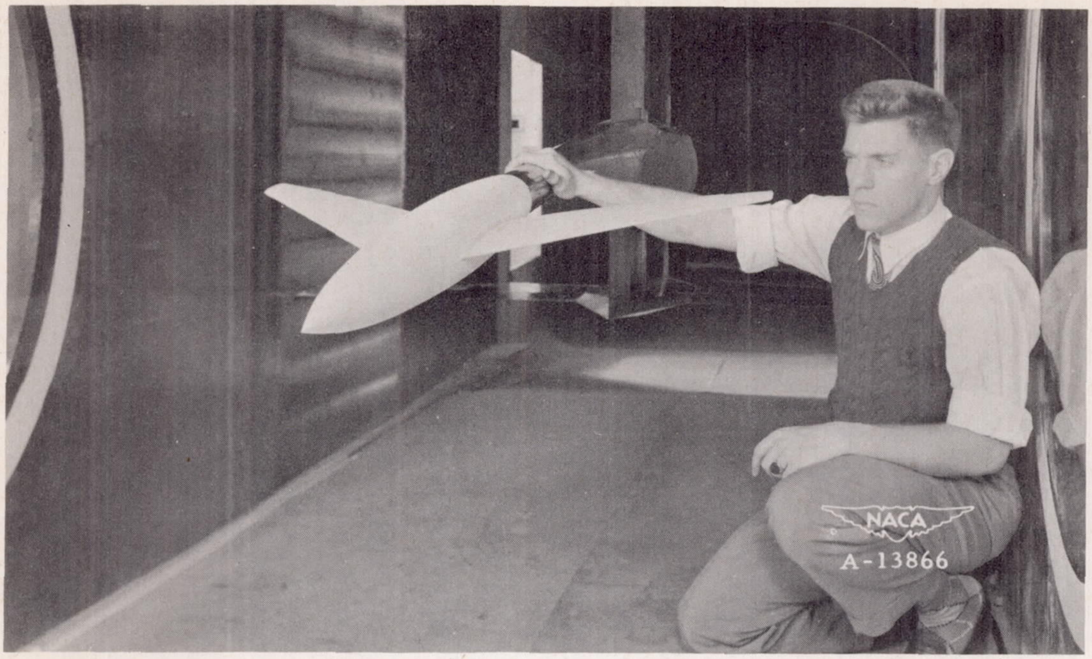
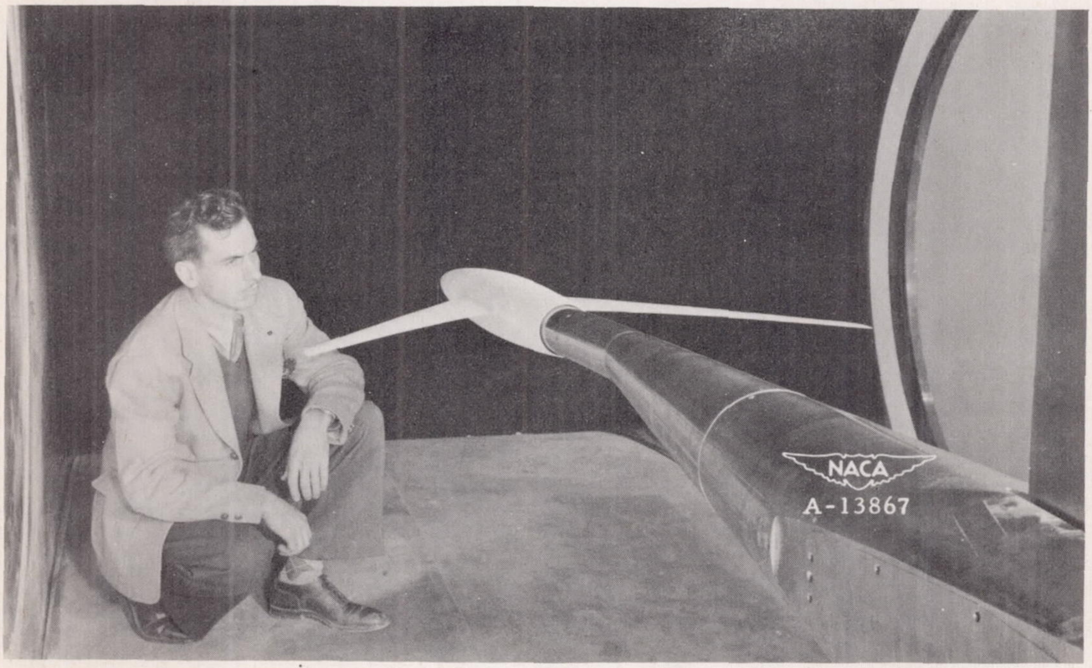


Figure 3.– The twist and camber distribution of sections parallel to the plane of symmetry of the 63° swept-back wing.

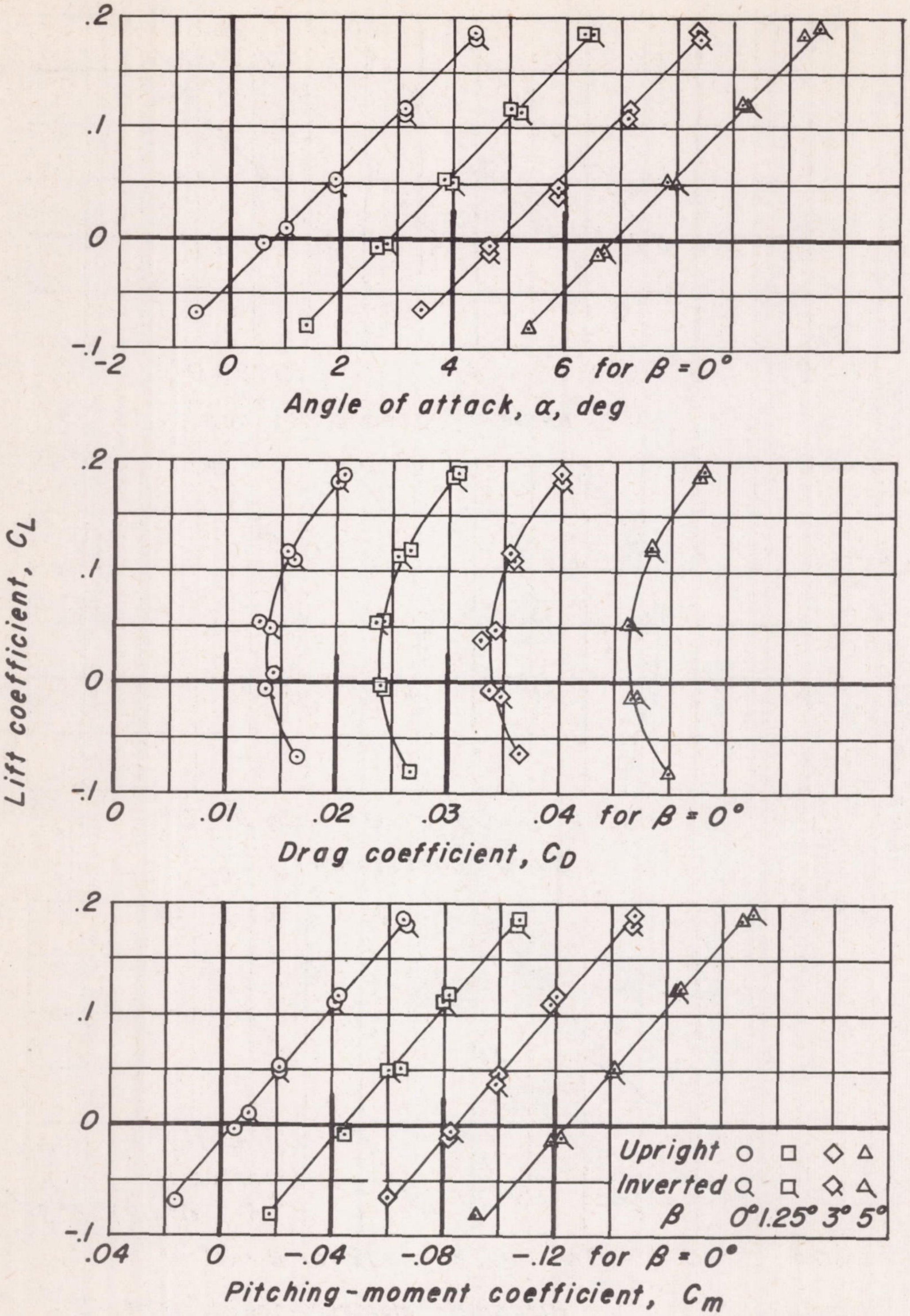


(a) Three-quarter front view.



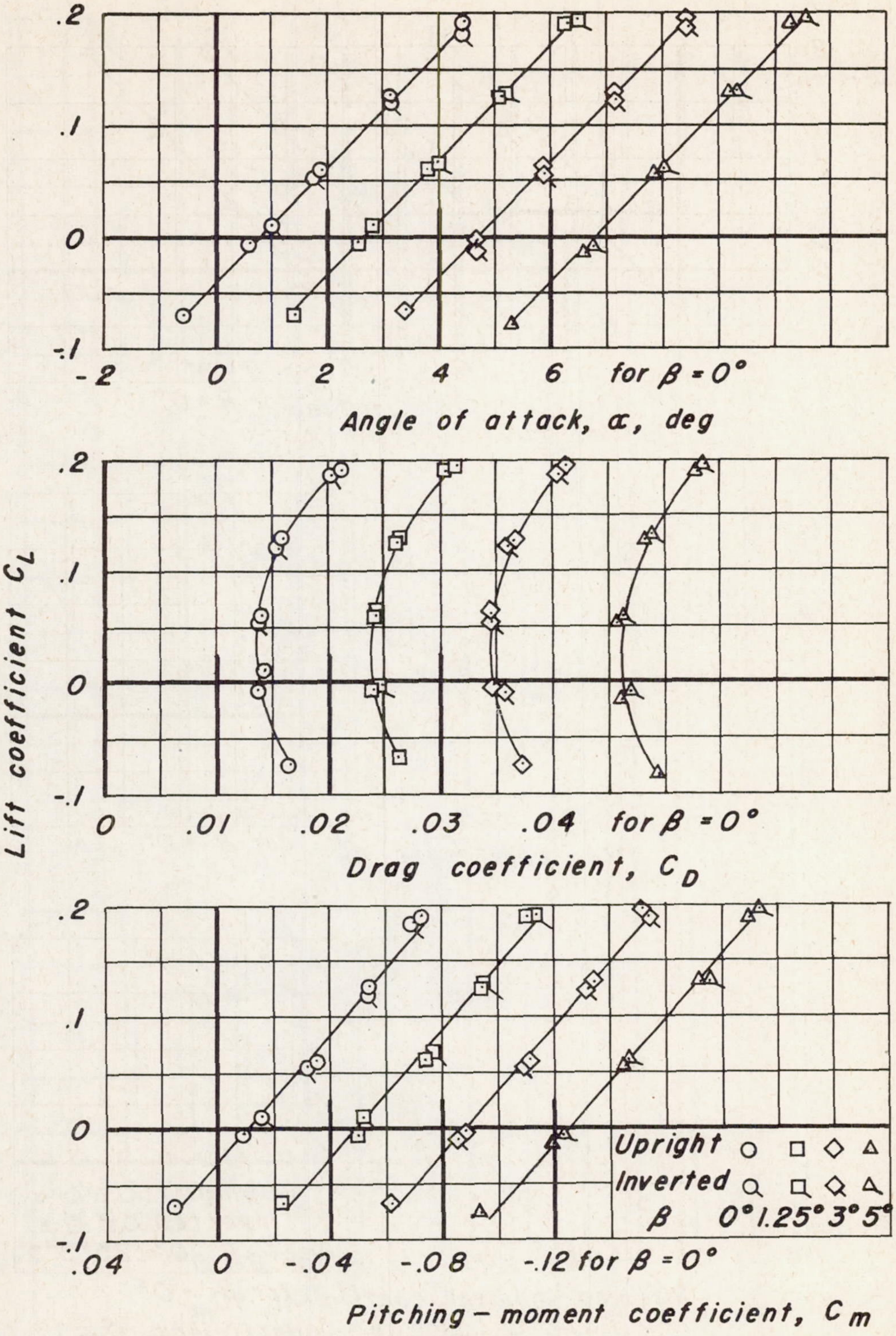
(b) Three-quarter rear view.

Figure 4.- Support system with model mounted for testing in the Ames 6- by 6-foot supersonic wind tunnel.



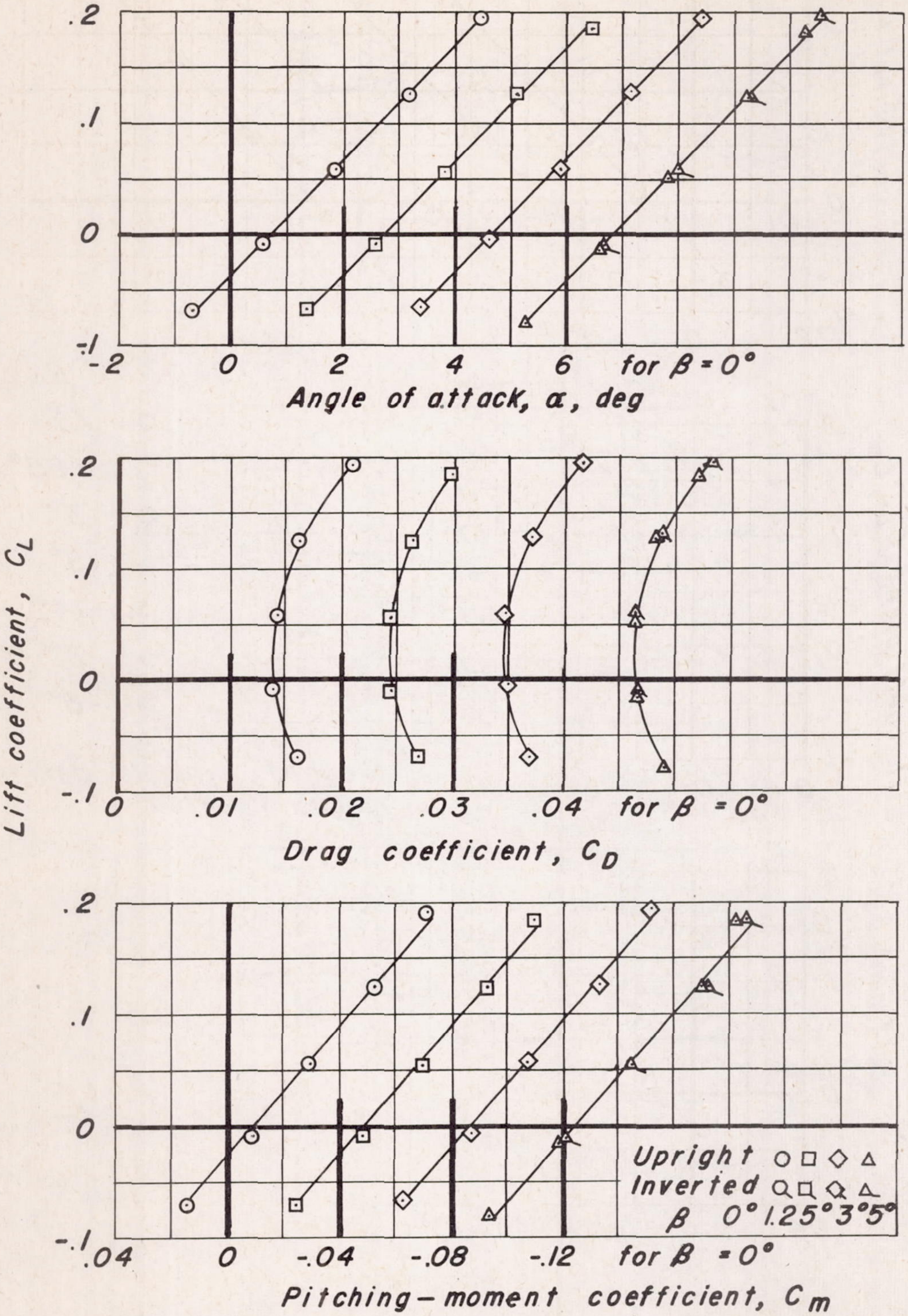
(a) $R = 1.5 \times 10^6$.

Figure 5.- Longitudinal characteristics of the 63° swept-back-wing model at various sideslip angles.



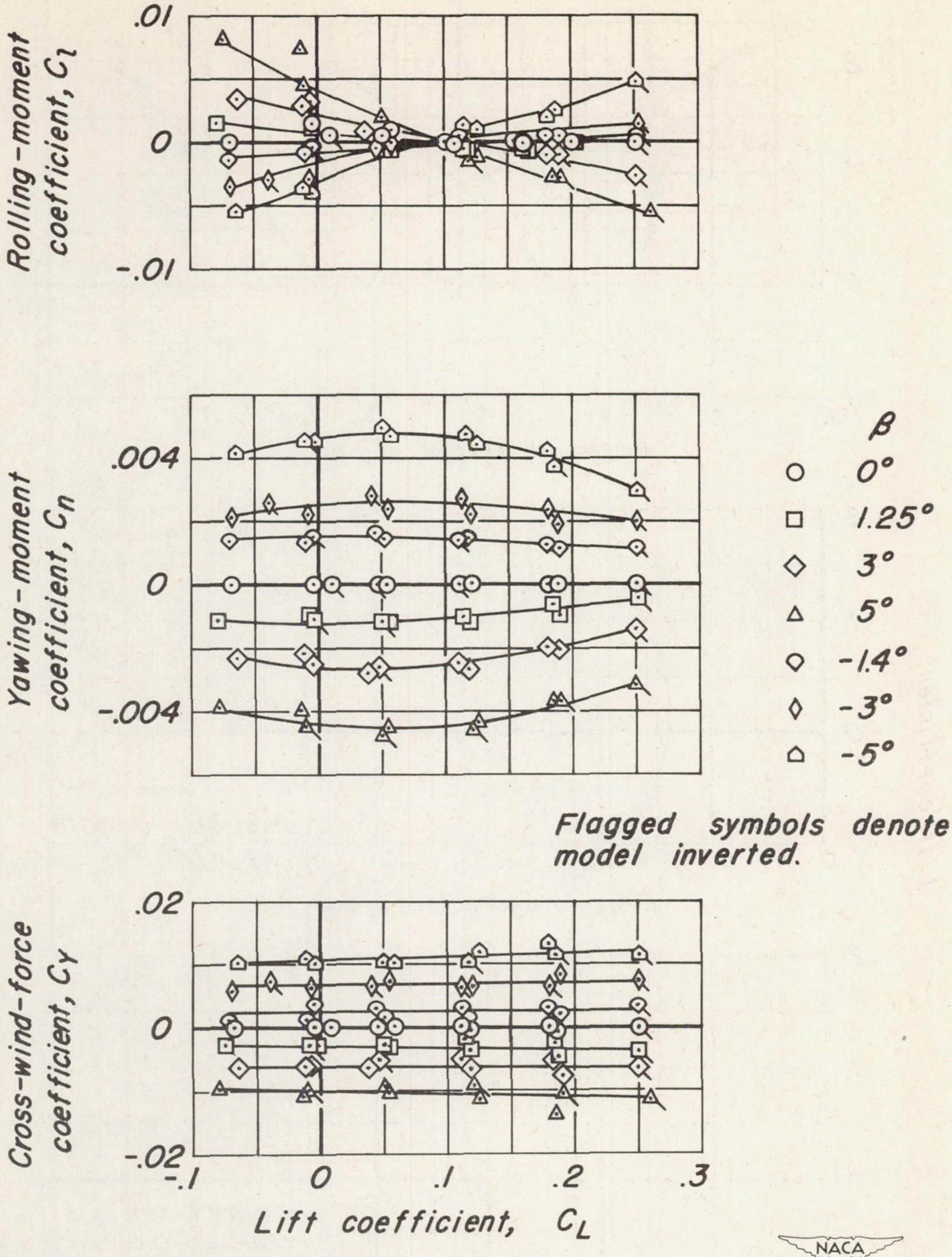
(b) $R = 2.7 \times 10^6$.

Figure 5.- Continued.



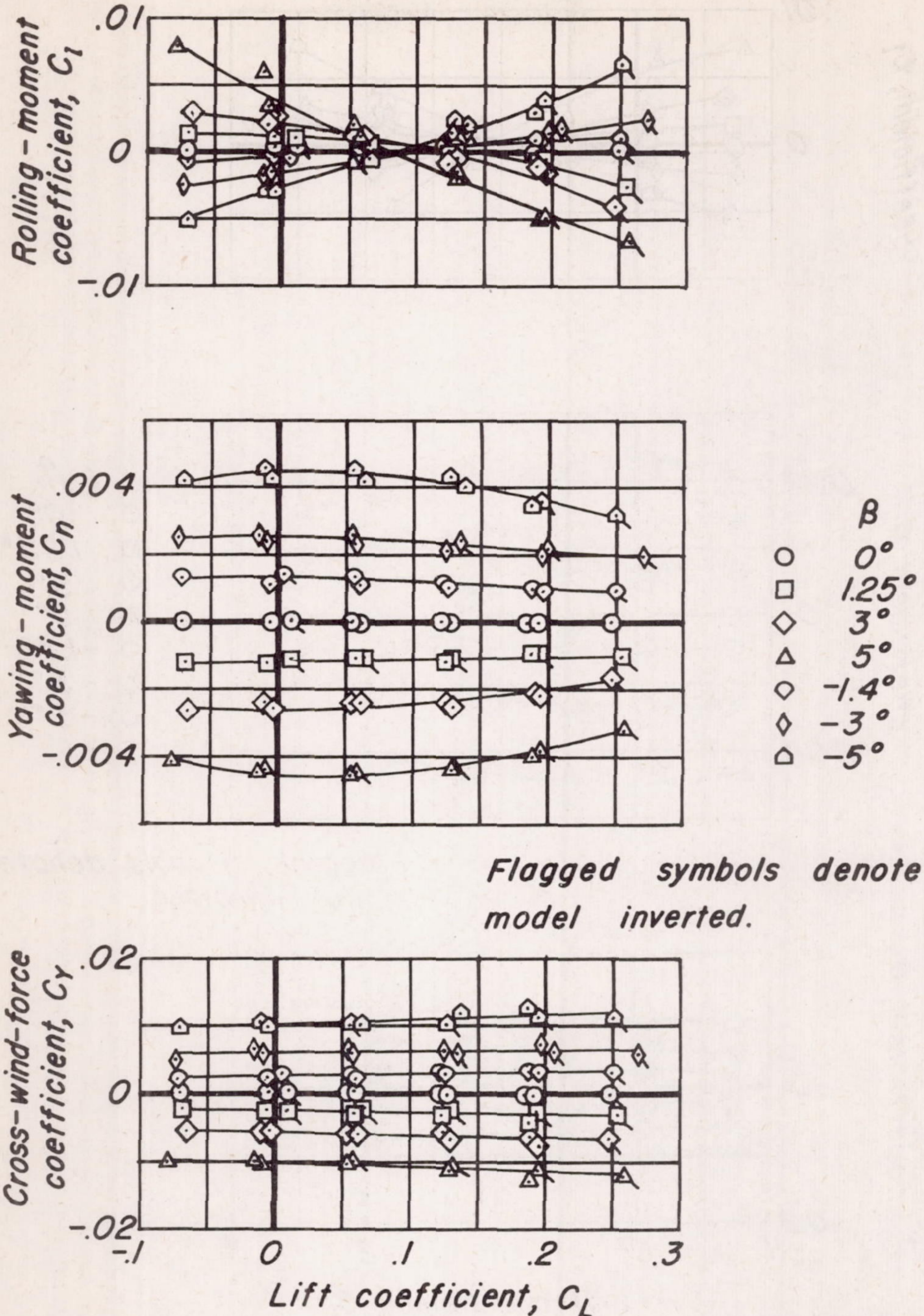
(c) $R = 3.7 \times 10^6$.

Figure 5.- Concluded.



(a) $R = 1.5 \times 10^6$.

Figure 6.– Lateral characteristics of the 63° swept-back-wing model at various sideslip angles.



(b) $R = 2.7 \times 10^6$.

Figure 6.- Continued.

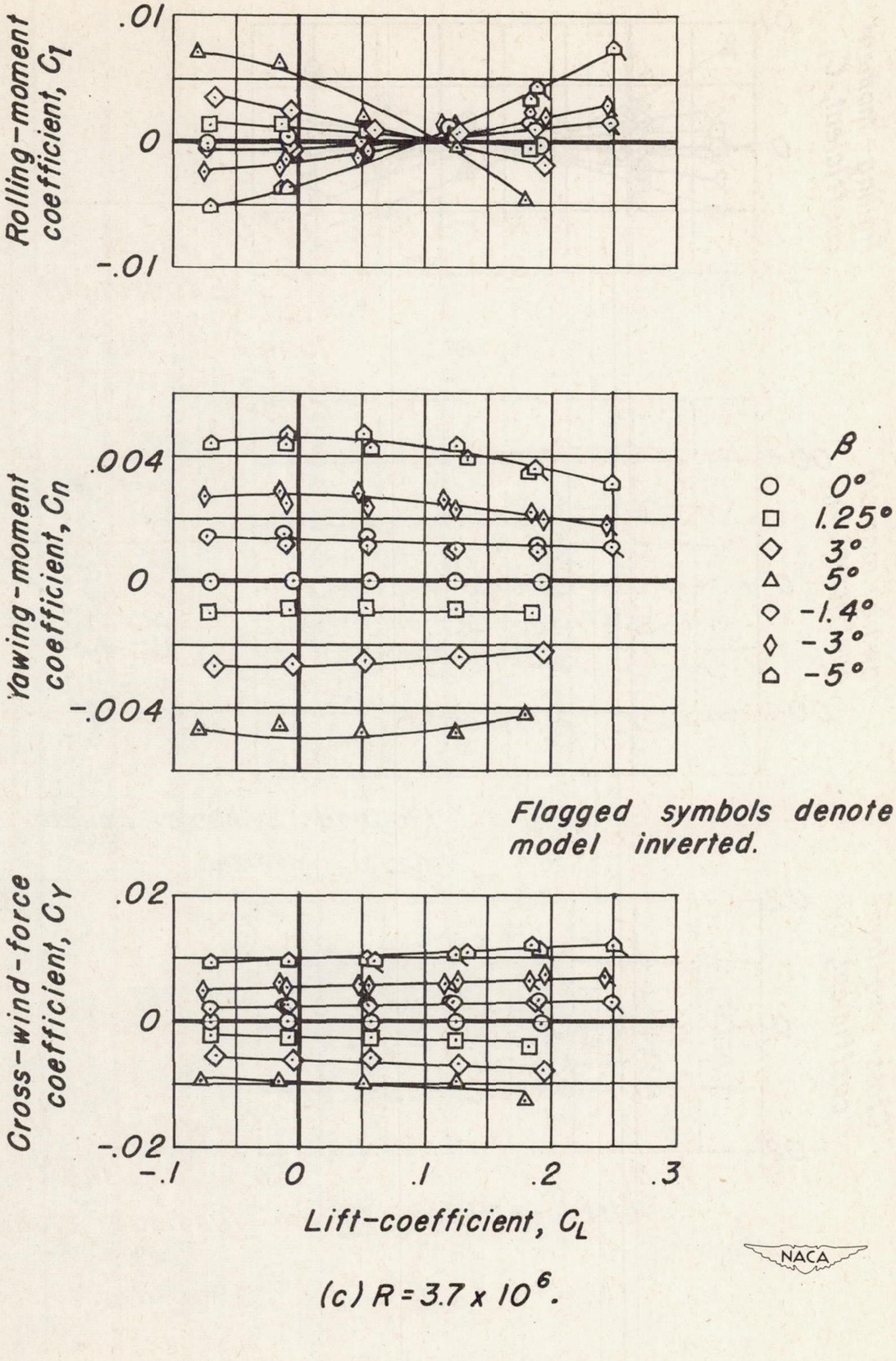


Figure 6.– Concluded.

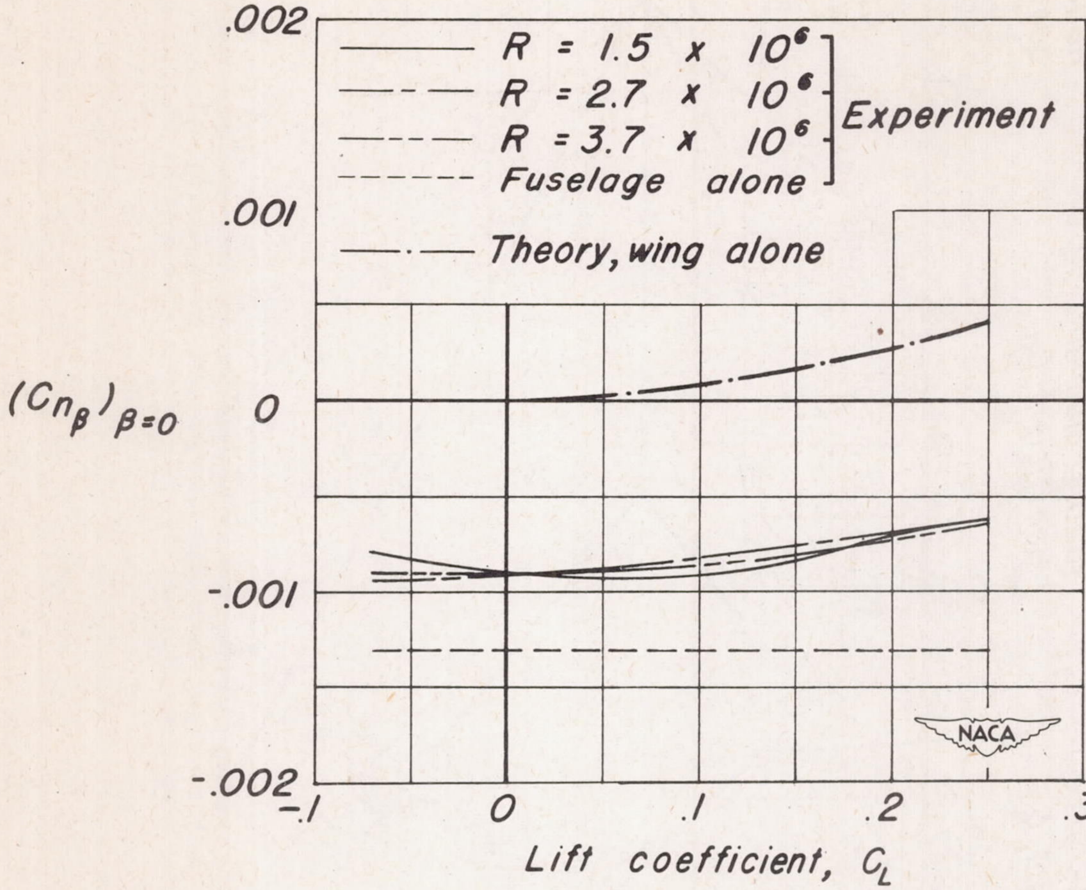


Figure 7.- Variation with lift coefficient of the directional-stability derivative C_{n_β} for the 63° swept-back-wing model and the fuselage alone.

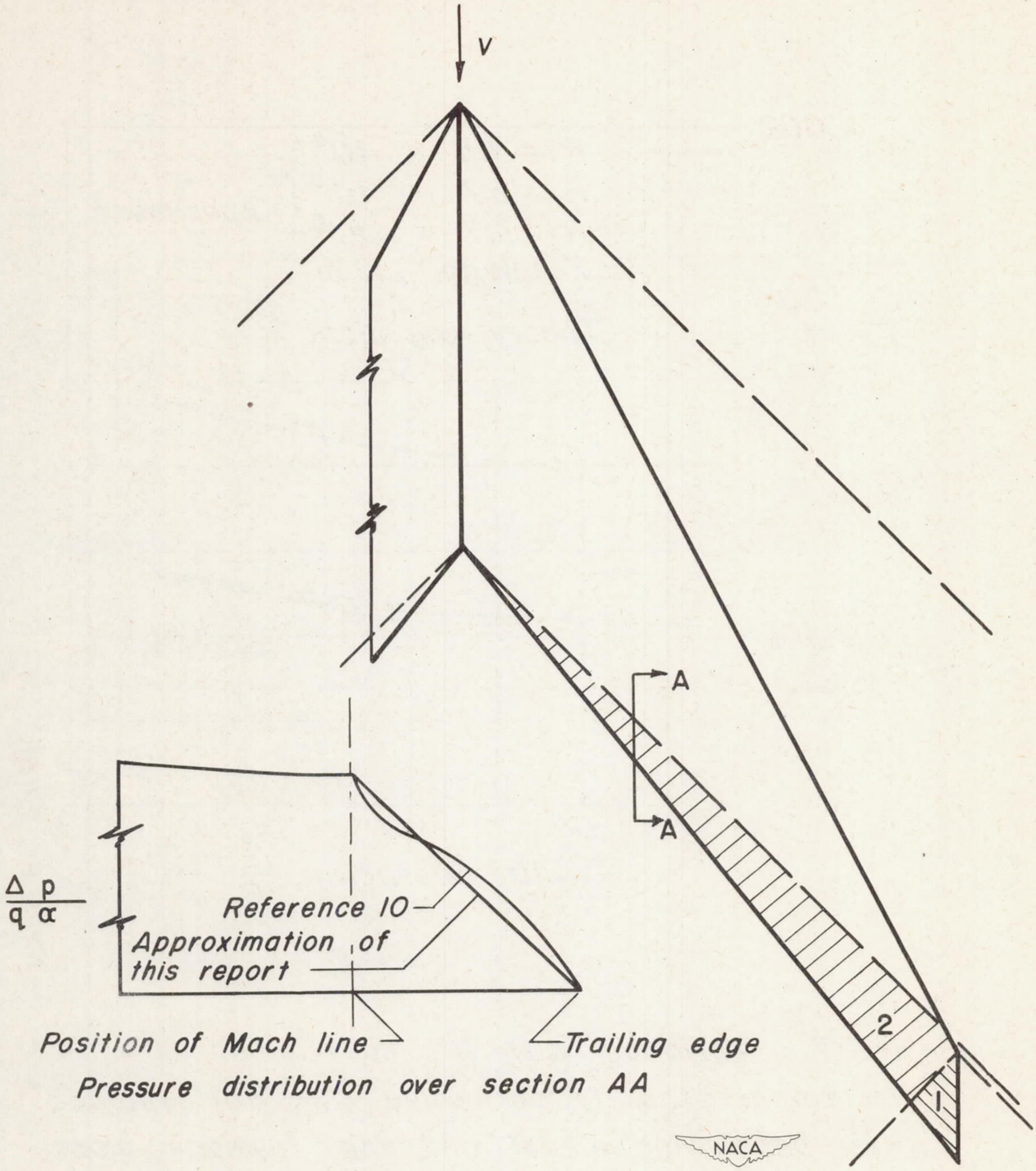


Figure 8.– Approximations made in the calculation of the theoretical aerodynamic characteristics.