NACA TN 1955

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE 1955

PRELIMINARY INVESTIGATION AT SUPERSONIC SPEEDS OF TRIANGULAR AND SWEPTBACK WINGS

By Macon C. Ellis, Jr., and Lowell E. Hasel

Langley Aeronautical Laboratory
Langley Air Force Base, Va.

NACA

Washington
October 1949

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE 1955

PRELIMINARY INVESTIGATION AT SUPERSONIC SPEEDS OF TRIANGULAR AND SWEPTBACK WINGS

By Macon C. Ellis, Jr., and Lowell E. Hasel

SUMMARY

A series of thin, triangular plan-form wings have been investigated in the Langley model supersonic tunnel. The series consisted of eight triangular wings of vertex angles such that a range of leading-edge positions both inside and outside the Mach cone at the two test Mach numbers of 1.43 and 1.71 was obtained. Three sweptback wings having angles of sweep of 45°, 55°, and 63° were also tested at a Mach number of 1.43. These sweptback wings had circular-arc sections with rounded leading edges and thicknesses of 13.3 percent of the chord measured normal to the leading edge. For each angle of sweep, wings having two values of aspect ratio were tested.

Lift results for the triangular wings indicated that Jones' theory for the lift of slender pointed wings is applicable for thin wings in the range of test Mach numbers up to values of $\frac{\tan \epsilon}{\tan m} \approx 0.3$, where ϵ is the wing vertex half-angle and m is the Mach angle. The center of pressure of the triangular wings was coincident with the center of area for all the wings tested at both Mach numbers. The lowest minimum drag coefficients were obtained for the wings with smallest vertex angles relative to the Mach angle. Also in this smallest vertex-angle region, the highest values of maximum lift-drag ratio of about 7 for both Mach numbers were obtained. The tests indicated that wings having triangular plan forms should be operated well within the Mach cone for maximum efficiency.

Results of the sweptback-wing tests compared with triangular wing results for a Mach number of 1.43 show the same trends of lift and drag as the sweep angle is changed. For corresponding sweep angles, the swept-wing lift-curve slopes were lower than those for triangular wings, probably because of the increased thickness. The tests indicate that for a Mach number of 1.4, the angle of sweep must be increased to about 60° to obtain low drag coefficients of the same magnitude as those due to skin friction.

INTRODUCTION

Recent theories of low-aspect-ratio triangular wings and swept wings by Jones (references 1 and 2) have indicated the advantages to be gained by the use of pointed plan-form wings for high-speed flight. Numerous tests conducted both in this country and in Germany have shown that the drag rise with Mach number just below sonic velocity usually associated with wings having their leading edges normal to the flight direction may be delayed to higher speeds by the use of sweepback. Some of these tests have been conducted at high subsonic and up to moderate supersonic speeds; however, the largest amount of experimental work appears to be in the low subsonic-speed region and is mostly concerned with development of means for making the stability and control characteristics of swept wings satisfactory. In reference 1, Jones has indicated by use of theories assuming small disturbances that the lift distribution at small angles of attack of a slender airfoil having a pointed or triangular plan form is relatively unaffected by the compressibility of the air below or above the speed of sound. The required condition for small changes in aerodynamic characteristics with Mach number at supersonic speeds is that the triangular wing have its vertex angle so small that the entire surface lies near the center of the Mach cone. With this condition satisfied, the changes in lift-curve slope with Mach number are expected to be small and the position of the center of pressure at the center of area is not expected to change. The direction of the resultant force was shown to lie halfway between the normal to the surface and the normal to the air stream. This result suggests that higher values of lift-drag ratio might be expected for these wings than for wings having essentially two-dimensional characteristics; that is, wings with the resultant force normal to the surface. An isolated test of a slender triangular airfoil at a Mach number of 1.75 (reference 1) verified the theoretical values of lift and center of pressure; however, the value of maximum lift-drag ratio was not obtained. Thus the present tests of a series of thin triangular wings at supersonic speeds were made to explore the possibilities of high values of maximum lift-drag ratio, to find the limits of Jones' slender wing theory, and to provide preliminary design information for such wings beyond this limit.

A series of eight triangular wings of various vertex angles were tested at the Langley model supersonic tunnel which was the forerunner of the present Langley 9-inch supersonic tunnel. The tests were brief and preliminary in nature, because at the time they were started the date for starting modification of this tunnel to the present closed-return tunnel was imminent. The vertex angles of the wings were of such values that a range of leading-edge positions both inside and outside of the Mach cone were covered for the two test Mach numbers of 1.43 and 1.71.

After the triangular wing tests were made, time permitted only very brief tests of six sweptback wings at one Mach number of 1.43. Results of these sweepback tests are included herein mainly for their qualitative indications. All of the tests were made during July and August of 1945.

Many significant contributions, especially theoretical, relating to the aerodynamics of triangular and swept wings have been made since the present paper was originally issued for limited distribution in 1946. Inasmuch as the experimental data obtained at that time, however, still appear to be of general interest, the original version is being reissued by the NACA in the present form to provide for wider distribution.

SYMBOLS

M	Mach number
V	stream velocity
ρ	stream density
q	dynamic pressure $\left(\frac{1}{2}\rho V^2\right)$
R	Reynolds number referred to c
α	angle of attack
ϵ	triangular wing vertex half-angle
m	Mach angle
Λ	sweepback angle
b	maximum span of wing
c	maximum chord
S	wing area
A	aspect ratio $\left(b^2/S\right)$
l	distance to center of area from vertex
C_L	lift coefficient $\left(\frac{\text{Lift}}{qS}\right)$

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

$C_{m_{ca}}$ pitching-moment coefficient for triangular wings $\left(\frac{\text{Moment about center of area}}{qSc}\right)$

L/D lift-drag ratio

Subscripts:

max maximum

min minimum

DESCRIPTION OF SUPERSONIC TUNNEL AND TEST MODELS

The Langley model supersonic tunnel in which the tests were made was the direct-action type; that is, atmospheric air was continuously inducted, compressed, and after passing through the nozzle and diffuser, exhausted back to the atmosphere. Thus the air in this tunnel was subject to condensation in the supersonic nozzle during periods of high outside air humidity. The supersonic nozzles and test sections for the tunnel were formed by interchangeable nozzle blocks inserted between fixed side walls $7\frac{1}{2}$ inches apart. The test sections were approximately square. A three-component balance and shielded-sting-support system provided means for measuring lift, moment, and drag forces on models.

In order to expedite the tests in the limited time available, the triangular wings were made simply from flat sheets of $\frac{1}{32}$-inch-thick steel. The leading edges were beveled slightly and rounded off, and the trailing edges were beveled to a sharp edge as shown in figure 1 which also gives dimensions of the wings. For the tests, the wings were mounted on a sting support which passed through a sharp-edged conical shield to the three-component balance. The size of the wings were limited by the forces the balance was capable of measuring; the reflected shock from the wing vertex was always well back on the shield.

Details of the sweptback-wing models are shown in figure 2. Circular-arc sections were selected mainly for ease of construction and duplication. The leading edges were rounded because it was considered that the wings would operate always behind the Mach angle.

TEST RESULTS AND DISCUSSION

Description of Tests

When air of sufficiently low temperature and high humidity flows through a supersonic nozzle the water vapor becomes supercooled and finally condenses at a shock front somewhere along the nozzle. This condensation results in an increase in stagnation temperature and a decrease in total pressure of the air. For given initial stagnation conditions of the air before expansion through the nozzle, the effect of varying humidity is to vary the stream conditions in the test section. Most of the tests were made during periods of low humidity; however, stream conditions did vary to some extent. The two test Mach numbers of 1.43 and 1.71 are actually averages for the series of wings; the maximum variation in Mach number for the results presented was about plus or minus 0.02 and the maximum variation in stream pressure in the region of the model for any one test was about 4 percent. Variations within these values did not seriously affect the scatter of data, although they made it necessary to obtain, in some cases, a large number of test points in order to find differences in characteristics among the wings. Fewer test points were obtained for the triangular wings at the lower Mach number because the more consistent test conditions gave less scatter for the same number of points.

Tares for the triangular and swept wings were obtained by measuring the lift and drag forces on the support cones alone. The drag tare was composed of the small cone drag and a relatively large pressure force acting on the spindle area. The pressure force was due to atmospheric pressure acting on one end of the spindle and stream pressure acting on the other end. The drag tares were of about the same magnitude as the drag forces and, therefore, the variations in the pressure force leave the absolute values of drag more in doubt than the lift results. Tares for the swept wings were obtained similarly; however, the lift tare for the relatively longer supports was larger than for the small cones for the triangular wings.

Test Results for Triangular Wings

Lift results for the eight triangular wings at $M = 1.43$ are shown in figure 3. The lift appears to vary linearly with angle of attack up to about 5°, the limit of the tests, for all wings. Variations in angle of zero lift for the wings are due to varying stream inclination and to inadvertently different asymmetries in the wings. The lift results for $M = 1.71$ shown in figure 4 are similar except that for wings 5, 6, 7, and 8, the angle range is about 7° and the lift variation is still linear. These four wings have their leading edges inside the Mach cone for $M = 1.71$. The lift-curve-slope values from figures 3 and 4 are collected and shown in figure 5 as the ratio of measured lift-curve slope to the theoretical two-dimensional lift-curve slope against the parameter $\frac{\tan \epsilon}{\tan m}$. The theoretical two-dimensional lift-curve slope values are taken from Ackeret's theory where $\frac{dC_L}{d\alpha} = \left(\frac{1}{57.3}\right)\frac{4}{\sqrt{M^2 - 1}}$.

Theoretical considerations indicated that $\frac{\tan \epsilon}{\tan m}$ is a fundamental parameter as pointed out to the authors by C. E. Brown of the Langley Laboratory. (This parameter later appeared in references 3 and 4.) The quantity $\frac{\tan \epsilon}{\tan m} \approx \frac{\epsilon}{m}$ for the range of test Mach numbers. When $\frac{\tan \epsilon}{\tan m} = 1.0$, it is identical to ϵ/m. Thus, values of $\frac{\tan \epsilon}{\tan m} < 1$ correspond to cases where the leading edge is behind the Mach angle and values of $\frac{\tan \epsilon}{\tan m} > 1$ correspond to cases where the leading edge is ahead of the Mach angle. As $\frac{\tan \epsilon}{\tan m}$ approaches zero, the test results for both Mach numbers show a single curve for the slope ratio that asymptotes Jones' theory. The limit of applicability of Jones' theory for slender triangular wings in the range of test Mach numbers thus appears as a value of $\frac{\tan \epsilon}{\tan m} \approx 0.3$. In reference 5, Jones has developed a theory for calculating the pressure drag of thin oblique airfoils at supersonic speeds. It was pointed out by C. E. Brown of the Langley Laboratory that the equations in Jones' report could be used to calculate the lift of a thin triangular wing for cases where the wing leading edge is outside the Mach cone. Calculations for wings outside the Mach cone at the test Mach numbers showed the lift-curve slopes to be the same as the two-dimensional theoretical values for a straight wing. (This result later appeared in reference 6.) A suitable theory for the lift of triangular flat plates that bridges the gap between Jones' slender wing theory and the theory for wings outside the Mach cone might be expected, therefore, to result in a curve that follows the lower part of the experimental slope ratio curve but continues smoothly to 1.0 or the two-dimensional value at $\frac{\tan \epsilon}{\tan m} = 1.0$. (Linearized theories for the general case of lift of triangular wings appeared at about the same time in references 7 and 8.) The variations in slope ratio, shown by the tests as the wing leading edge approaches and moves ahead of the Mach cone, are believed to be primarily due to the flow in the region of the rounded leading edge. Wing 1 was tested with its point aft, that is, with its leading edge normal to the stream; and values of lift-curve slope approximately equal to the values obtained by Ackeret's theory were measured.

The pitching-moment coefficients in figures 6 and 7 show that the center of pressure is coincident with the center of area for all the triangular wings tested at both Mach numbers. At low values of $\frac{\tan \epsilon}{\tan m}$, this result is as predicted by the theory and is verified by a test of a single wing in reference 1. The fact that the center of pressure is coincident with the center of area may also be reasoned simply for all values of $\frac{\tan \epsilon}{\tan m}$ from considerations of the conical flow. Any supersonic flow in which the pressure and velocity are constant along lines radiating from a point is a conical flow field. Supersonic flow about a point-foremost triangular flat plate is such a flow. Conical supersonic flows are discussed in detail by Busemann in reference 9.

Minimum drag-coefficient values for the wings at zero lift are collected from figures 3 and 4 and shown in figure 8 plotted against the parameter $\frac{\tan \epsilon}{\tan m}$ as were the lift-curve slopes. The tests show increasing minimum drag coefficient as the wing leading edge moves away from the center of the Mach cone. Estimates were made from the calculations in reference 4 to indicate the theoretical trends of the minimum drag coefficient as the leading-edge angle and Mach number are varied. The calculations in reference 4 were for the pressure drag of a series of thin, sharp-edge, double-wedge-section triangular wings of various thickness ratios and points of location of maximum thickness. The assumption of geometrical similarity between the wings of reference 4 and those of the present tests is rather crude; nevertheless, calculations were made assuming the present wings to have an equivalent thickness ratio equal to the maximum value for the average chord. It was further assumed that the maximum thickness was located at the midchord point and was constant from root to tip. Results of these calculations showed the same trend with $\frac{\tan \epsilon}{\tan m}$ as did the test results; that is, values of the minimum drag coefficient smoothly increased as the Mach angle approached and passed over the leading edge.

The test points at the lowest value of $\frac{\tan \epsilon}{\tan m}$ on each drag curve in figure 8 are for the same wing at the two test Mach numbers. For this wing (wing 8), the calculations of drag give about the same value of $C_D = 0.002$ due to pressure forces for both values of the Mach number. From the low value of pressure drag indicated by the calculations for wing 8, most of the drag shown by the tests for low values of $\frac{\tan \epsilon}{\tan m}$ would be expected to result from skin friction. Inasmuch as no appreciable difference should be expected in skin-friction drag for the two Mach numbers, the displacement of the drag curves at the lowest value of $\frac{\tan \epsilon}{\tan m}$ is probably incorrect. A constant error in drag-tare measurements for the tests at either Mach number very likely exists and this error is different for the two Mach numbers. Thus, an approximation of the true drag curves appears possible by displacing the upper test curve downward and the lower test curve upward by equal amounts so that they both asymptote the same line at $\frac{\tan \epsilon}{\tan m} = 0$. This asymptotic value of minimum drag coefficient minus an allowance for pressure drag of $\Delta C_D = 0.002$ is of the right order of magnitude for skin friction. For corresponding wings at the two Mach numbers, the displaced curves show no difference in drag values within the scatter of the test points about a smooth curve. The drag results appear to show the correct trend with $\frac{\tan \epsilon}{\tan m}$ but are not of sufficient accuracy to show the trends for a given wing with Mach number. The results of these tests indicate that the pressure drag may be reduced to a small value by operating triangular wings well within the Mach cone.

Although the accuracy of the absolute values of drag is somewhat doubtful, the indicated rise with angle of attack is believed to be reliable because of the systematic nature of the tests for each wing and because a smooth curve can be drawn through the points with small scatter. A check of the drag rise with angle of attack shows the resultant incremental force on all the wings for both Mach numbers to be normal to the surface. This result may be obtained by first assuming the resultant incremental force to be normal to the surface, then calculating ΔC_D above C_D for 0° angle of attack as $\Delta C_D = C_L \tan \alpha$. These calculated values fall on each drag curve within the probable test accuracy.

The measured values of L/D are shown in figures 3 and 4. The maximum lift-drag ratio results (fig. 8) show an increase in $(L/D)_{max}$ as the wings become more slender for each Mach number. The trend of the curves obtained at a maximum lift-drag ratio of about 7 indicated the possibility of still higher values for more slender wings. For a comparison with two-dimensional values of $(L/D)_{max}$, wing 1 was tested reversed, that is, with its sharp, straight trailing edge forward and normal to the stream. Approximate values of $(L/D)_{max}$ obtained were 4.0 for $M = 1.43$ and 3.8 for $M = 1.71$. The curves of L/D are seen to be approaching these values as the wing leading edges approach the normal to the stream.

Test Results for Sweptback Wings

The lift results shown in figure 9 for the six sweptback wings indicate no significant change in slope with aspect ratio except for the 45° sweep angle where the slope for the lower aspect ratio appears higher. For the 45° sweep angle at the test Mach number of 1.43, the Mach cone lies approximately along the wing leading edge, and the different flow arising from the strong initial shock may lead to different characteristics than for the higher angles of sweep for which cases the initial disturbance must be smaller.

The most significant result of the drag coefficients shown for the wings in figure 9 is the high drag for the wing with 45° sweep. For the Mach number of 1.43, drag coefficients as low as subsonic values are not obtained until the sweep angle is increased to approximately 60°. Practical use of this high degree of sweep appears difficult in relation to present knowledge and capability of handling the low-speed stability and control problems. The upward trend of the curves of L/D shown in figure 9 for the highest sweep angle suggests a high value of $(L/D)_{max}$ and invites solution to these stability and control problems.

The moment results of figure 9 show the center of pressure to be moving forward as the sweep angle decreases. At the highest sweep angle, the center of pressure appears about on the center of area. This result might be expected because most of the wing is in an approximately conical field except in the regions near the tips and along the trailing edge.

A comparison between the lift and drag test results for the sweptback and triangular wings at a Mach number of 1.43 is given in figure 10. The lift-curve slopes for the sweptback wings are appreciably lower than those for the triangular wings for corresponding sweep angles. A part of this difference may be due to thicker sections and some increases in lift may be affected by use of thinner sections. Further tests are necessary to explore this possibility. The drag comparison shows about the same minimum drag coefficient for the triangular and swept wings at the higher angles of sweep, however, for the lower sweep angles, the swept wing values are higher. The higher drags are probably due to the increased thickness ratio for the swept wings. The drag test results are not sufficiently accurate to show effects of aspect ratio. Variations in drag with aspect ratio and sweep angle can be calculated by the theory presented in reference 5.

The schlieren photographs of the lower aspect-ratio swept wings shown in figure 11 were made at a higher stream Mach number than that for the force tests but serve to show some significant points in regard to the flow over the wings. The photographs were made at a stream Mach number of 1.55. The leading edge of the 45° wing (fig. 11(d)) is in a position slightly ahead of the Mach angle. The disturbance ahead of the wing is seen to be strong as indicated by an appreciable curvature of the shock. This strong shock leads to the idea that high pressures are acting along the wing leading edge and resulting in high drag. This relatively high drag has been shown by the force tests. A comparison of figure 11(d) with figure 11(a) and 11(c) for higher angles of sweep indicates that the intensity of the initial disturbance from the point of the wing decreases. This decrease of intensity is in line with the decreasing drags shown by the force tests. The side view of the 63° swept wing in figure 11(b) shows the initial disturbance still small but shows a fairly strong shock originating at the vertex of the trailing edge and, therefore, indicates an accelerating region over the rear part of wing near the trailing-edge vertex, that results in relatively high velocities. As regards the tip sections, reasoning based on Jones' theory in reference 5 suggests that the tips should probably be made parallel to the stream for lower tip drag.

CONCLUSIONS

A supersonic wind-tunnel investigation of a series of thin, triangular plan-form wings at Mach numbers of 1.43 and 1.71, and an investigation of three sweptback wings of 13.3-percent-thickness ratio at a Mach number of 1.43 have indicated the following conclusions:

1. The lift of thin, triangular plan-form wings may be calculated by Jones' slender wing theory up to values of $\frac{\tan \epsilon}{\tan m} \approx 0.3$, where ϵ is the wing vertex half-angle and m is the Mach angle. For values of $\frac{\tan \epsilon}{\tan m}$ above 1.0, the lift is essentially the same as that obtained theoretically for a two-dimensional wing.

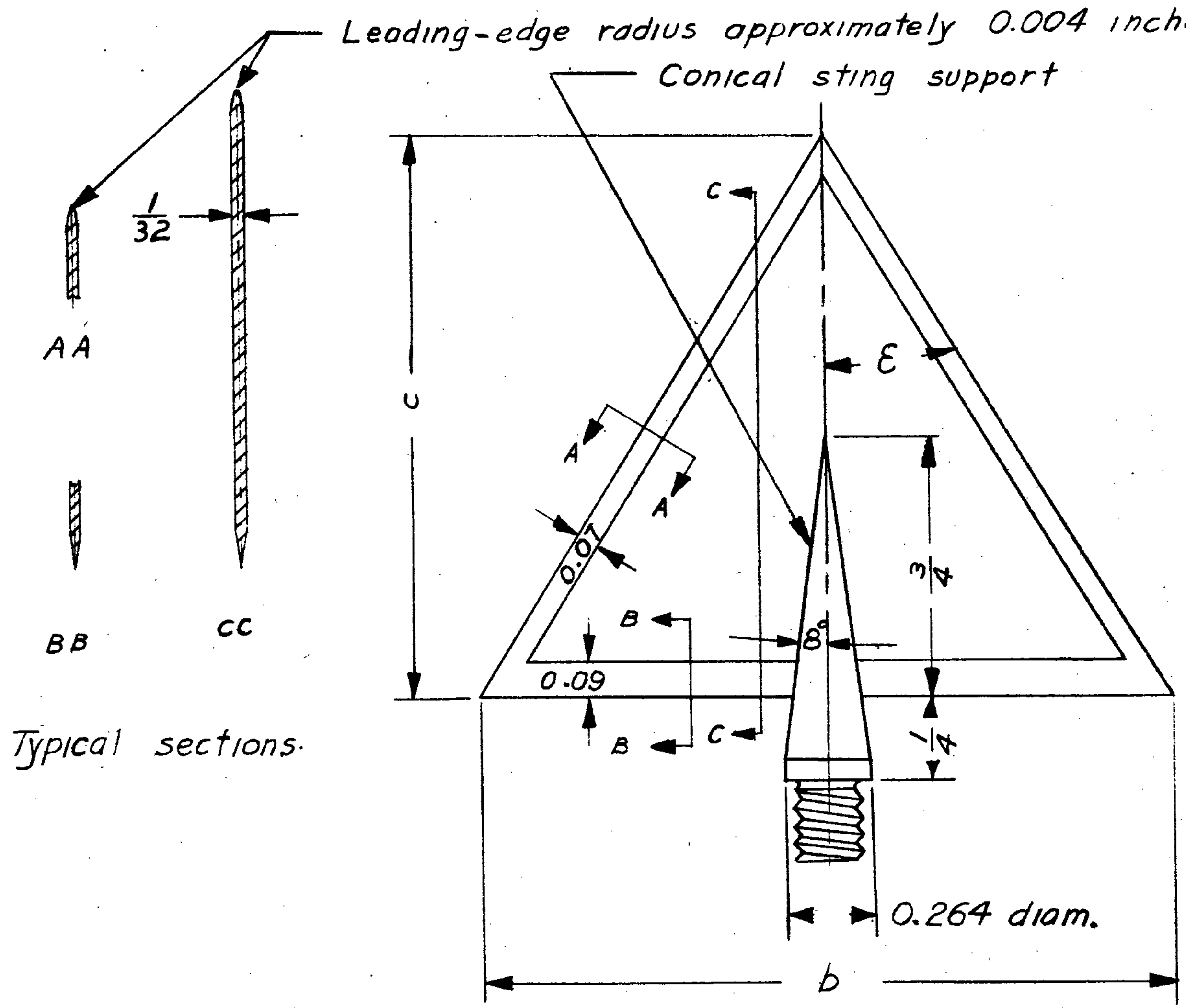
2. The center of pressure of thin, triangular plan-form wings is coincident with the center of area.

3. For low drag coefficients approaching those due to skin friction alone and for the highest values of maximum lift-drag ratio, both triangular and sweptback wings should be operated with their leading edges well behind the Mach cone.

Langley Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Air Force Base, Va., July 6, 1949.

REFERENCES

1. Jones, Robert T.: Properties of Low-Aspect-Ratio Pointed Wings at Speeds below and above the Speed of Sound. NACA Rep. 835, 1946.
2. Jones, Robert T.: Wing Plan Forms for High-Speed Flight. NACA Rep. 863, 1947.
3. Busemann, Adolf: Infinitesimal Conical Supersonic Flow. NACA TM 1100, 1947.
4. Puckett, Allen E.: Supersonic Wave Drag of Thin Airfoils. Jour. Aero. Sci., vol. 13, no. 9, Sept. 1946, pp. 475-484.
5. Jones, Robert T.: Thin Oblique Airfoils at Supersonic Speed. NACA Rep. 851, 1946.
6. Puckett, A. E., and Stewart, H. J.: Aerodynamic Performance of Delta Wings at Supersonic Speeds. Jour. Aero. Sci., vol. 14, no. 10, Oct. 1947, pp. 567-578.
7. Brown, Clinton E.: Theoretical Lift and Drag of Thin Triangular Wings at Supersonic Speeds. NACA Rep. 839, 1946.
8. Stewart, H. J.: The Lift of a Delta Wing at Supersonic Speeds. Quarterly Appl. Math., vol. IV, no. 3, Oct. 1946, pp. 246-254.
9. Busemann, A.: Drücke auf kegelförmige Spitzen bei Bewegung mit Überschallgeschwindigkeit. Z.f.a.M.M., Bd. 9, Heft 6, Dec. 1929, pp. 496-498.



Wing	b (in)	c (in)	ε (deg)
1	2.36	0.87	53.6
2	2.29	1.13	45.4
3	2.19	1.29	40.3
4	2.14	1.40	37.4
5	2.09	1.53	34.3
6	2.01	1.64	31.5
7	1.87	1.90	26.2
8	1.63	2.32	19.4

Figure 1.- Triangular wings and support dimensions.

Sweepback, Λ (deg)	A	l	c
45	5.13 3.32	1.97 1.41	1.33
55	3.44 2.10	2.36 1.52	1.64
63	2.27 1.36	2.74 1.73	2.07

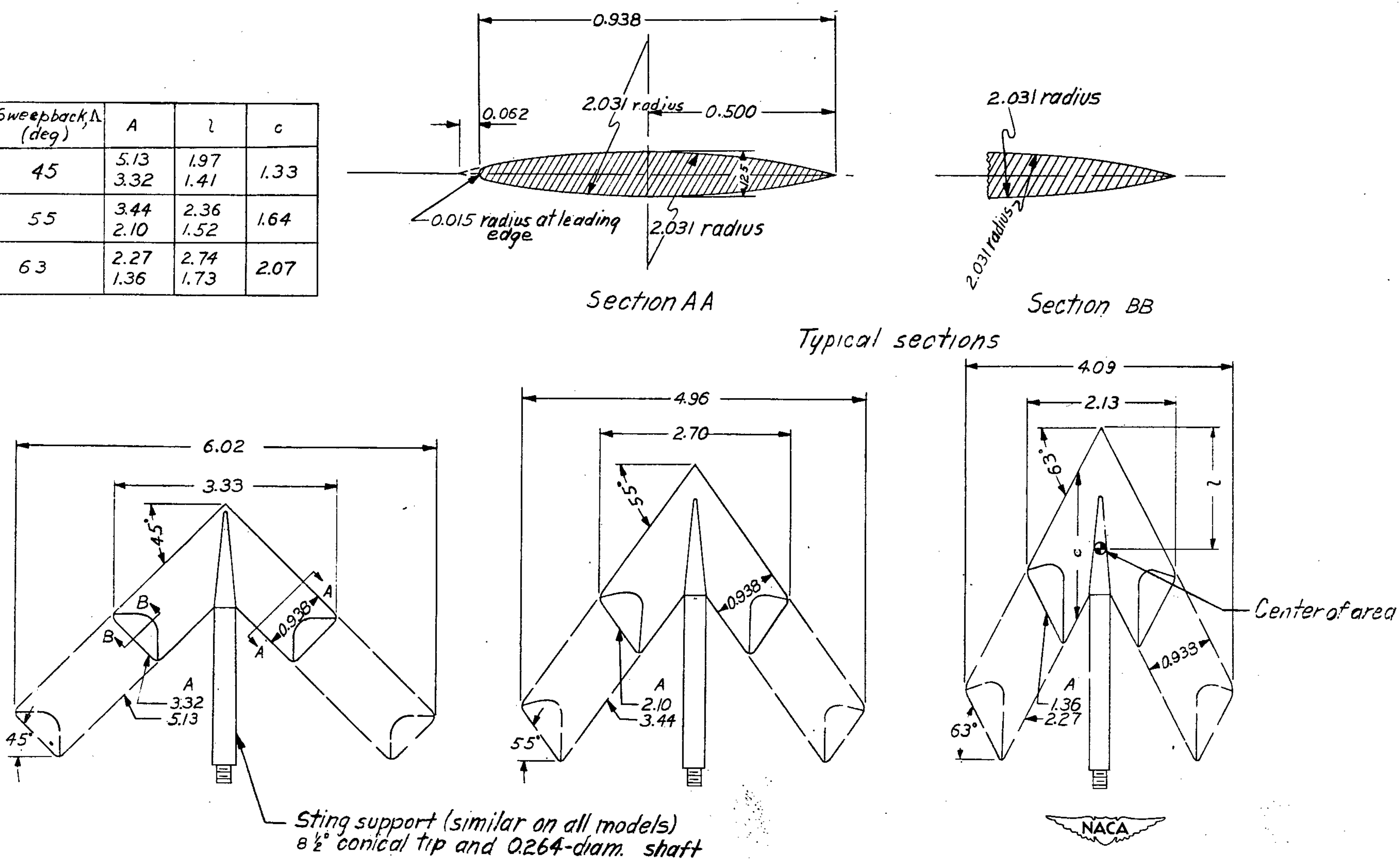
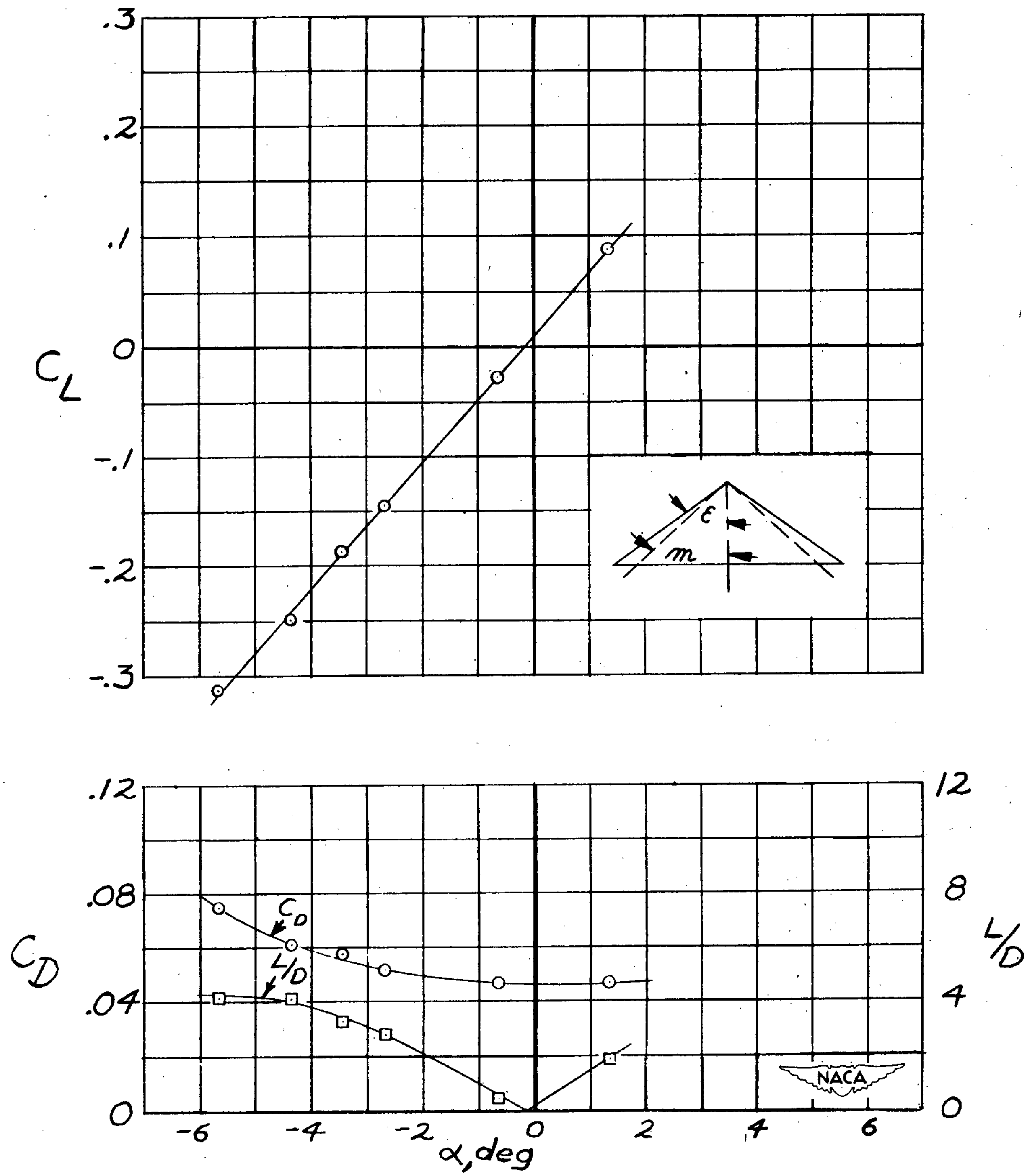
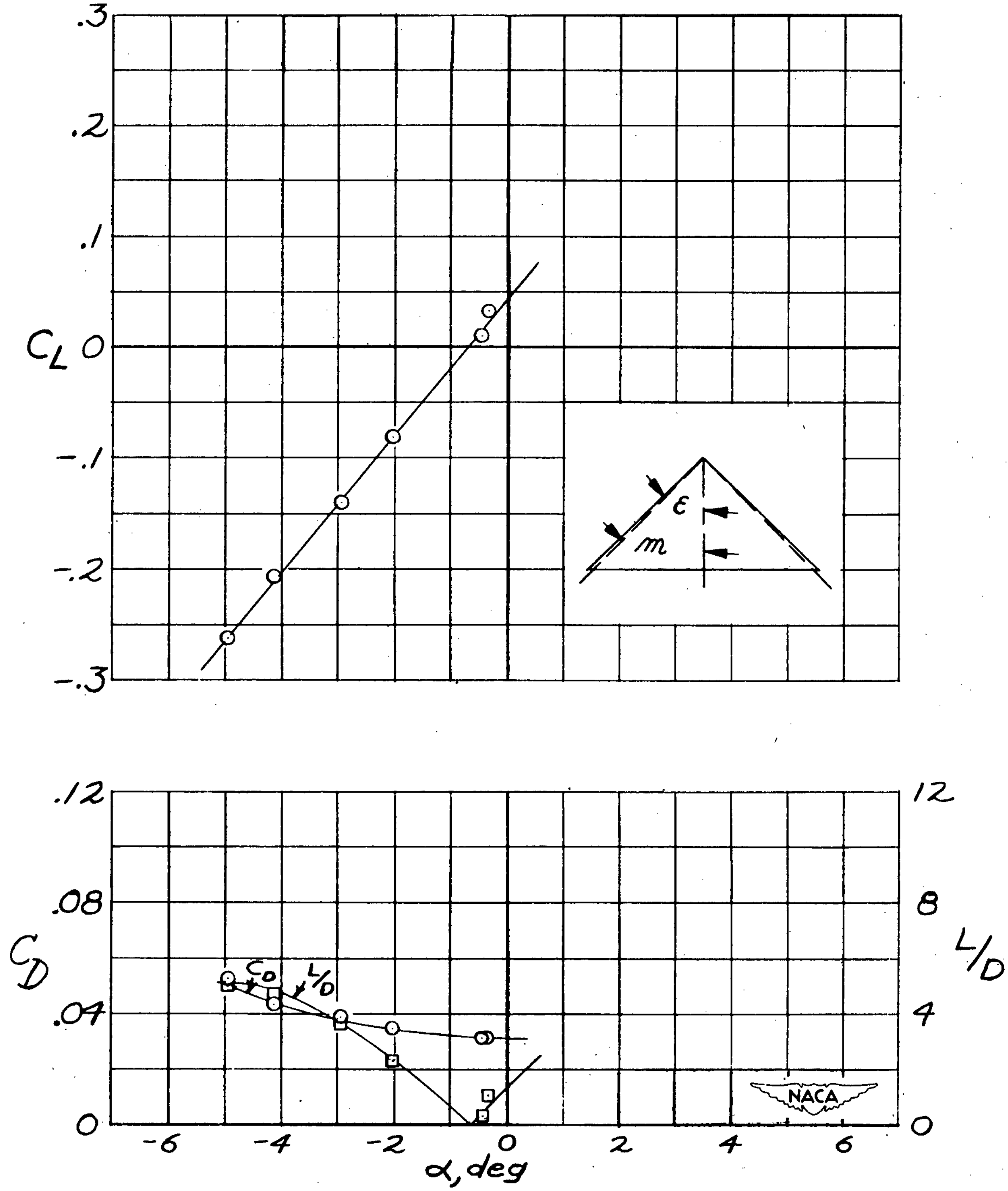


Figure 2.- Sweptback wings and support dimensions. All dimensions are in inches.



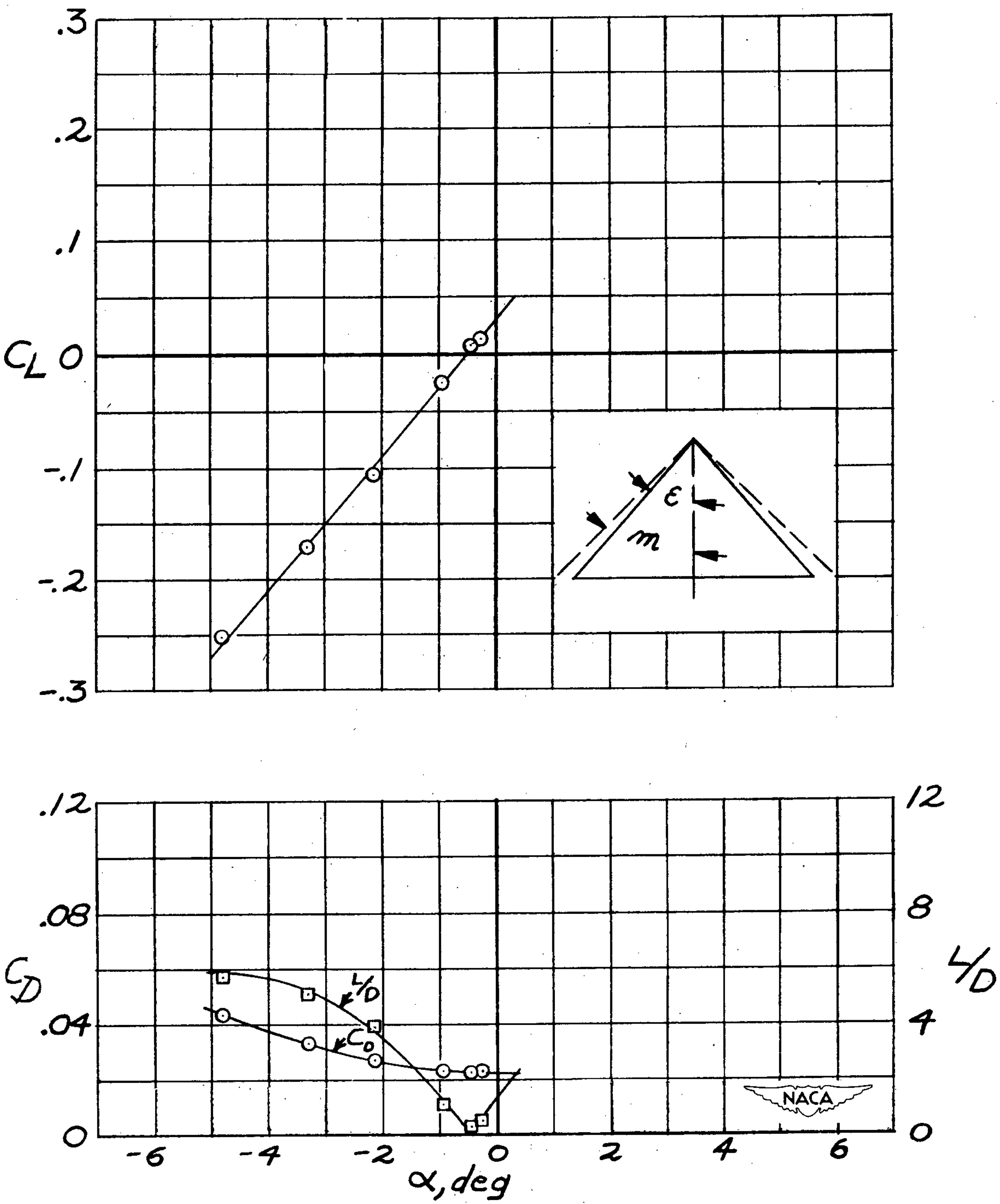
(a) Wing 1; R = 360,000.

Figure 3.- Triangular wing lift and drag test results for M = 1.43.



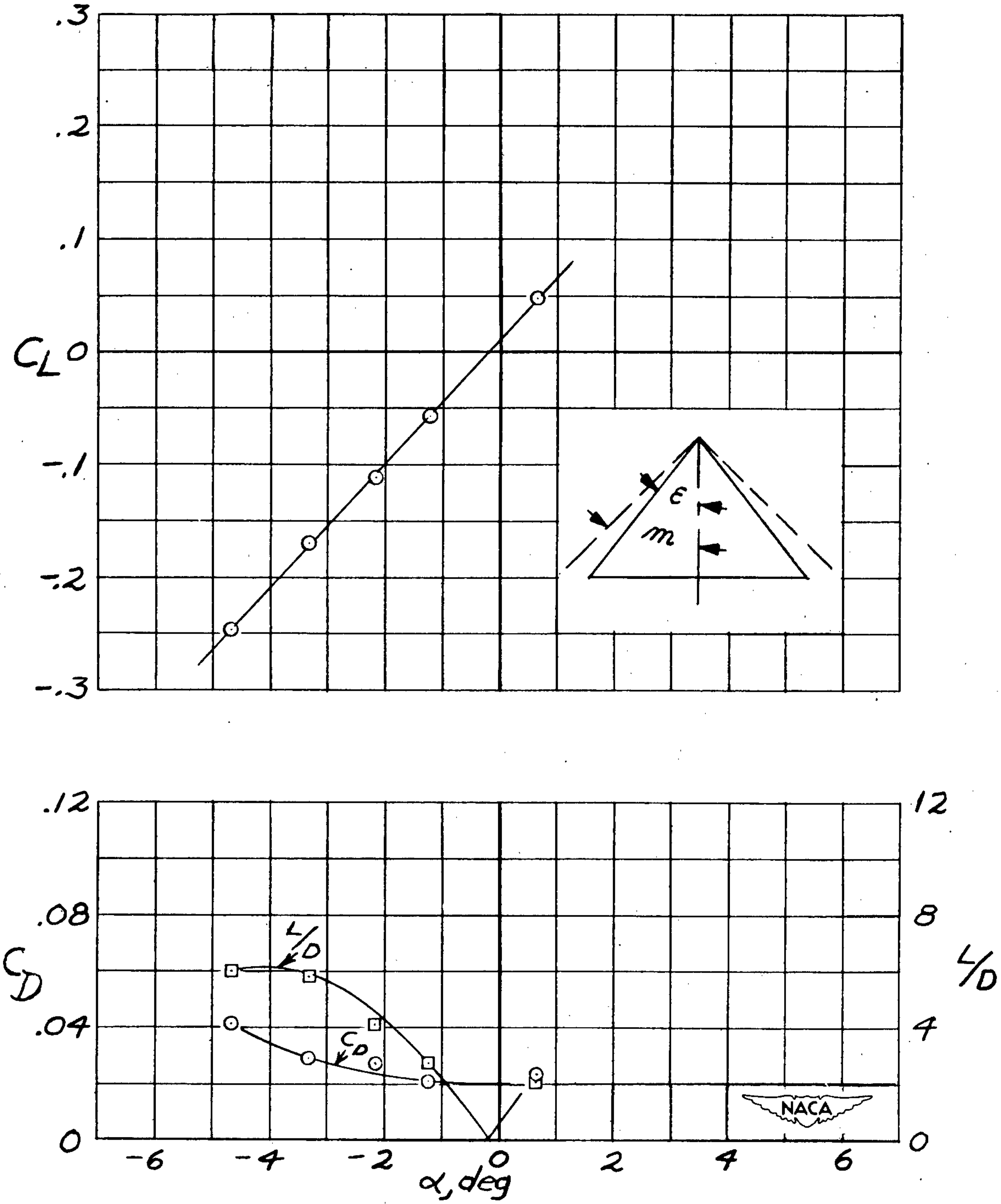
(b) Wing 2; R = 480,000.

Figure 3.- Continued.



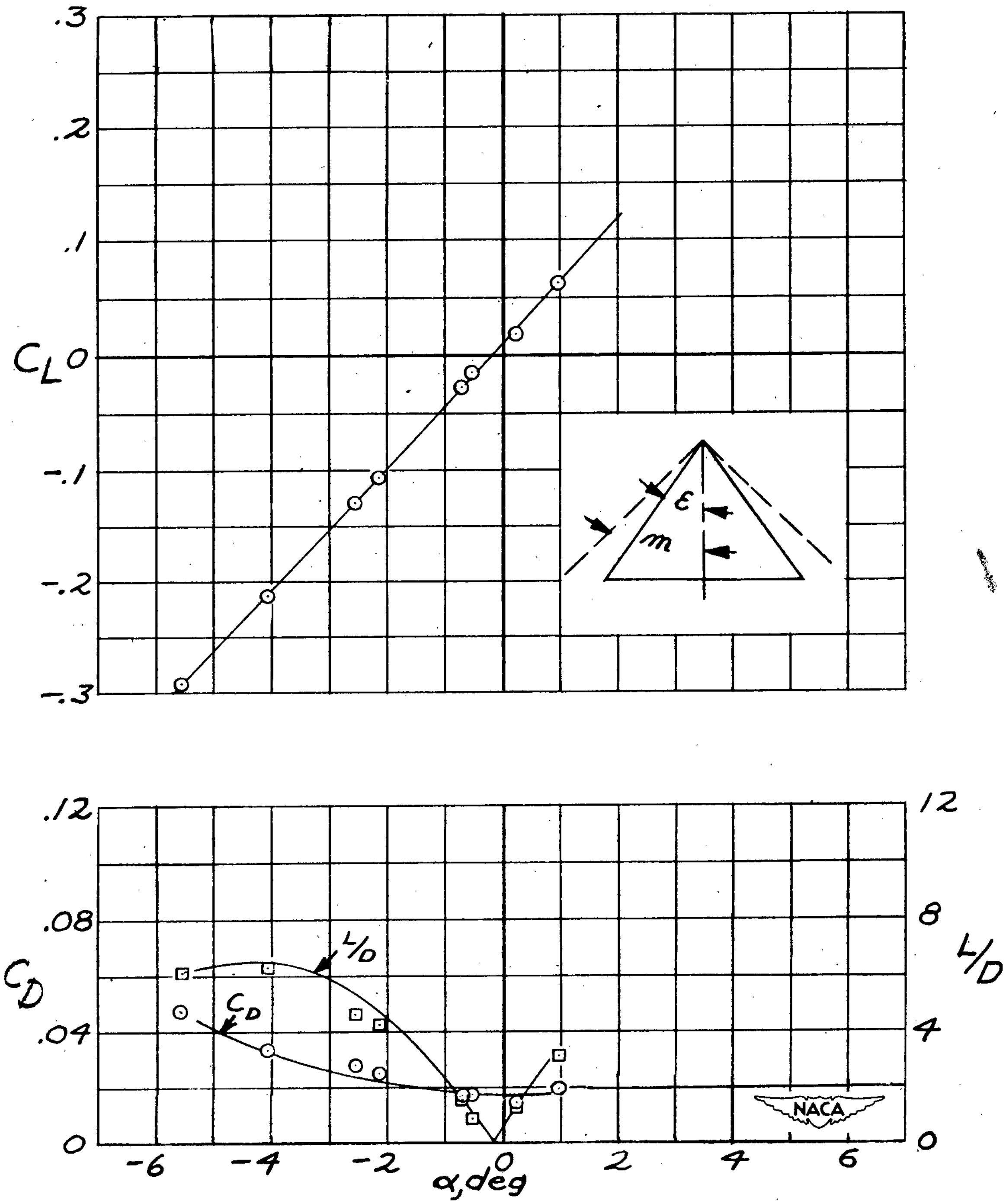
(c) Wing 3; R = 540,000.

Figure 3.- Continued.



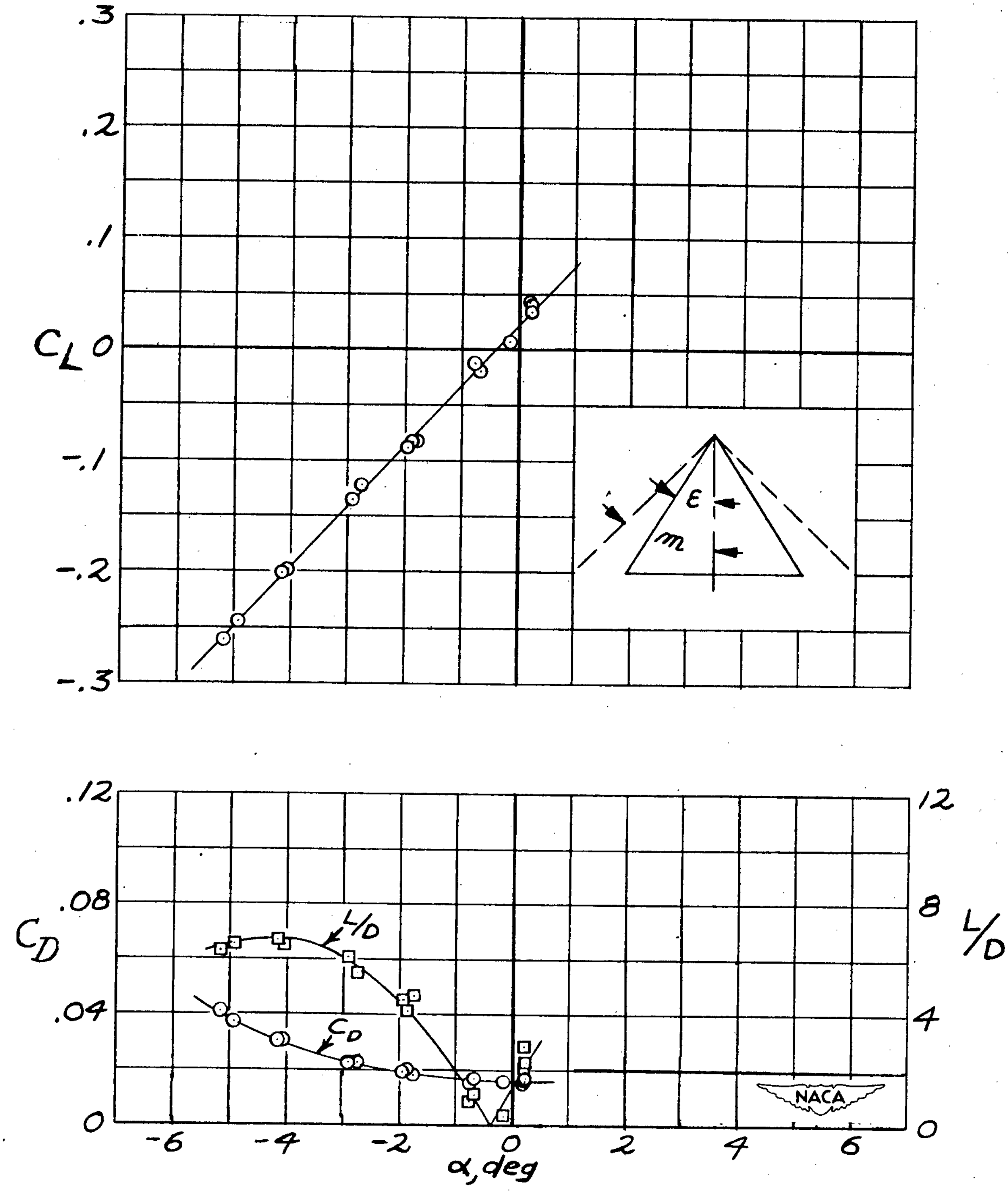
(d) Wing 4; R = 600,000.

Figure 3.- Continued.



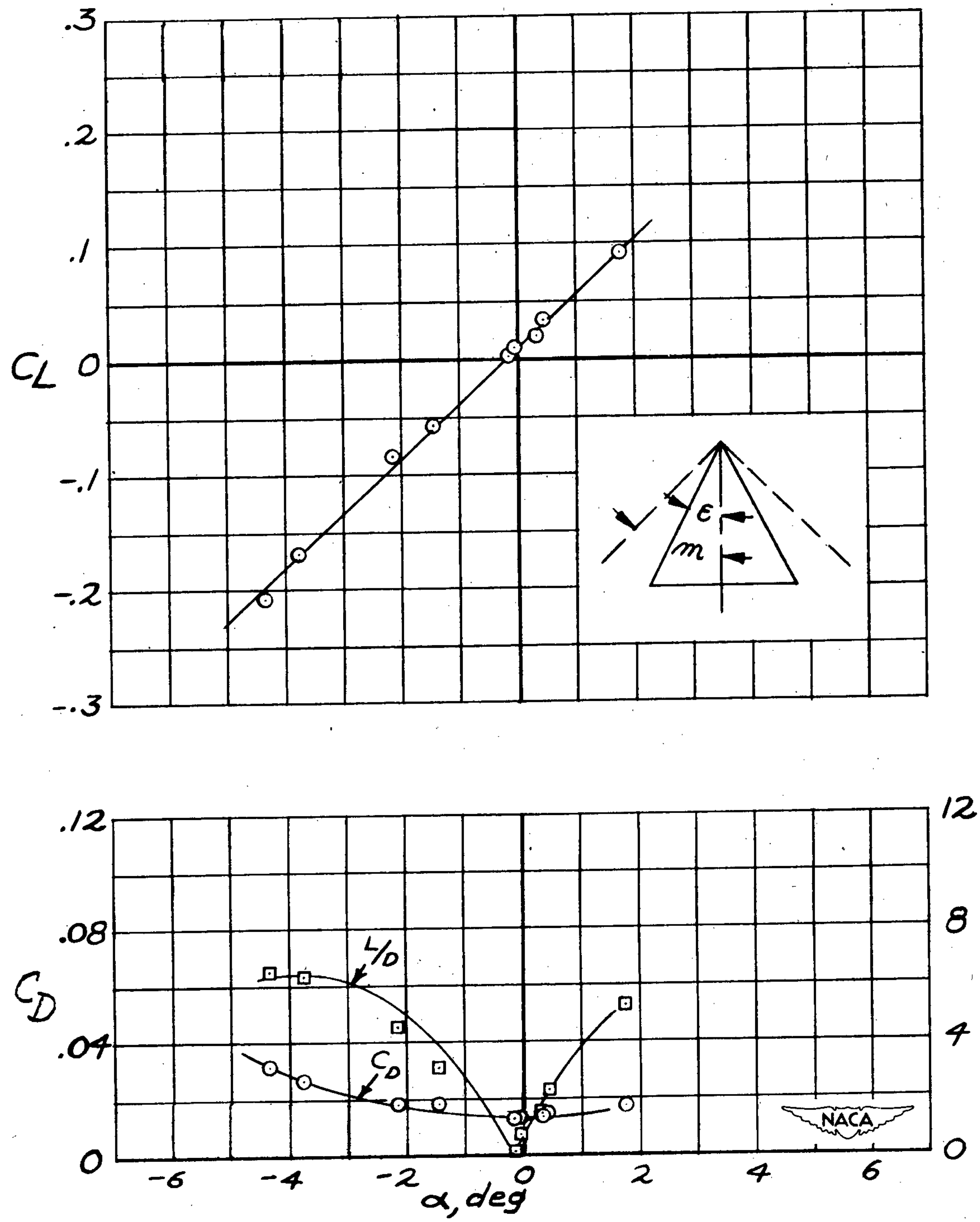
(e) Wing 5; R = 640,000.

Figure 3.- Continued.



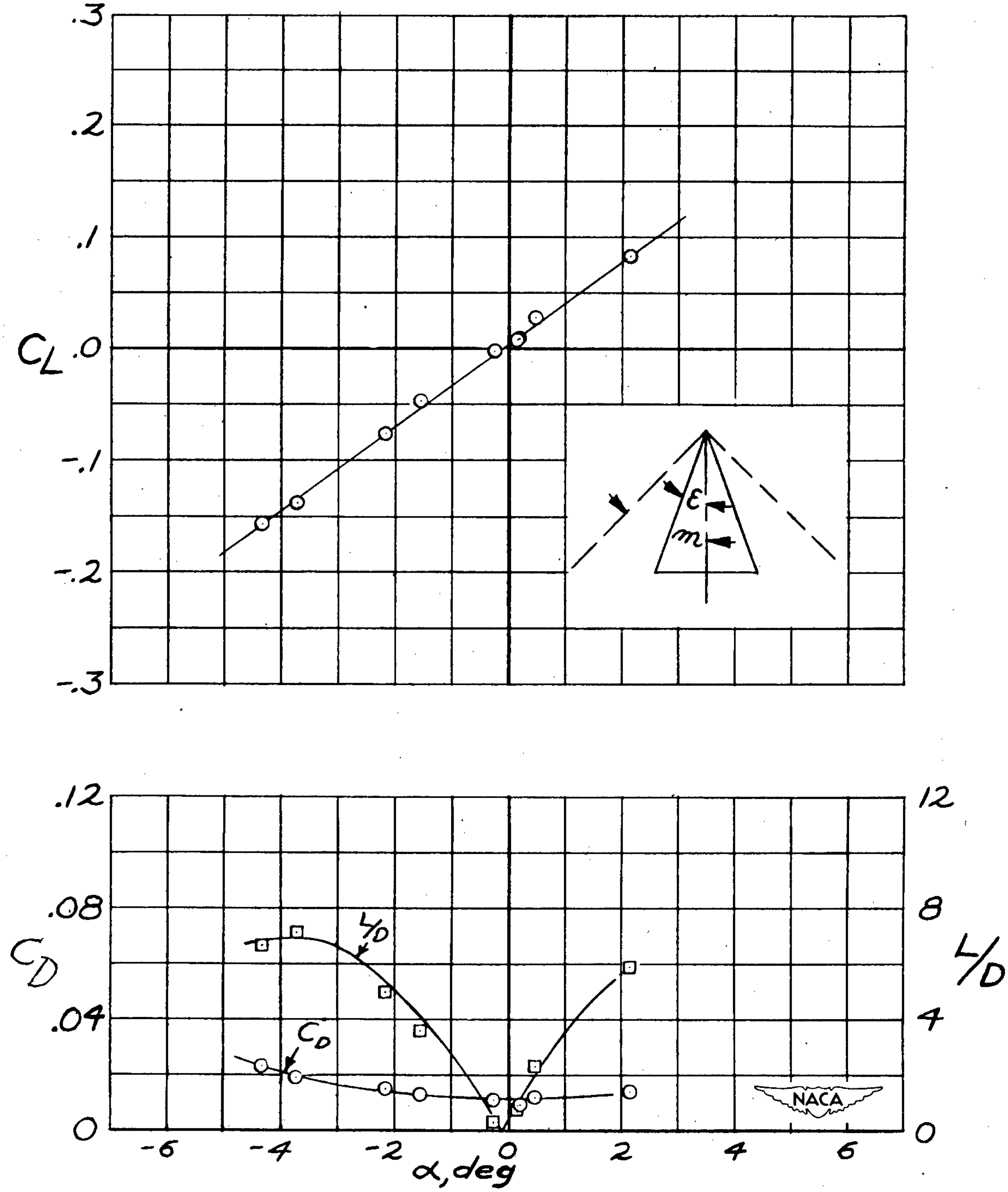
(f) Wing 6; R = 700,000.

Figure 3.- Continued.



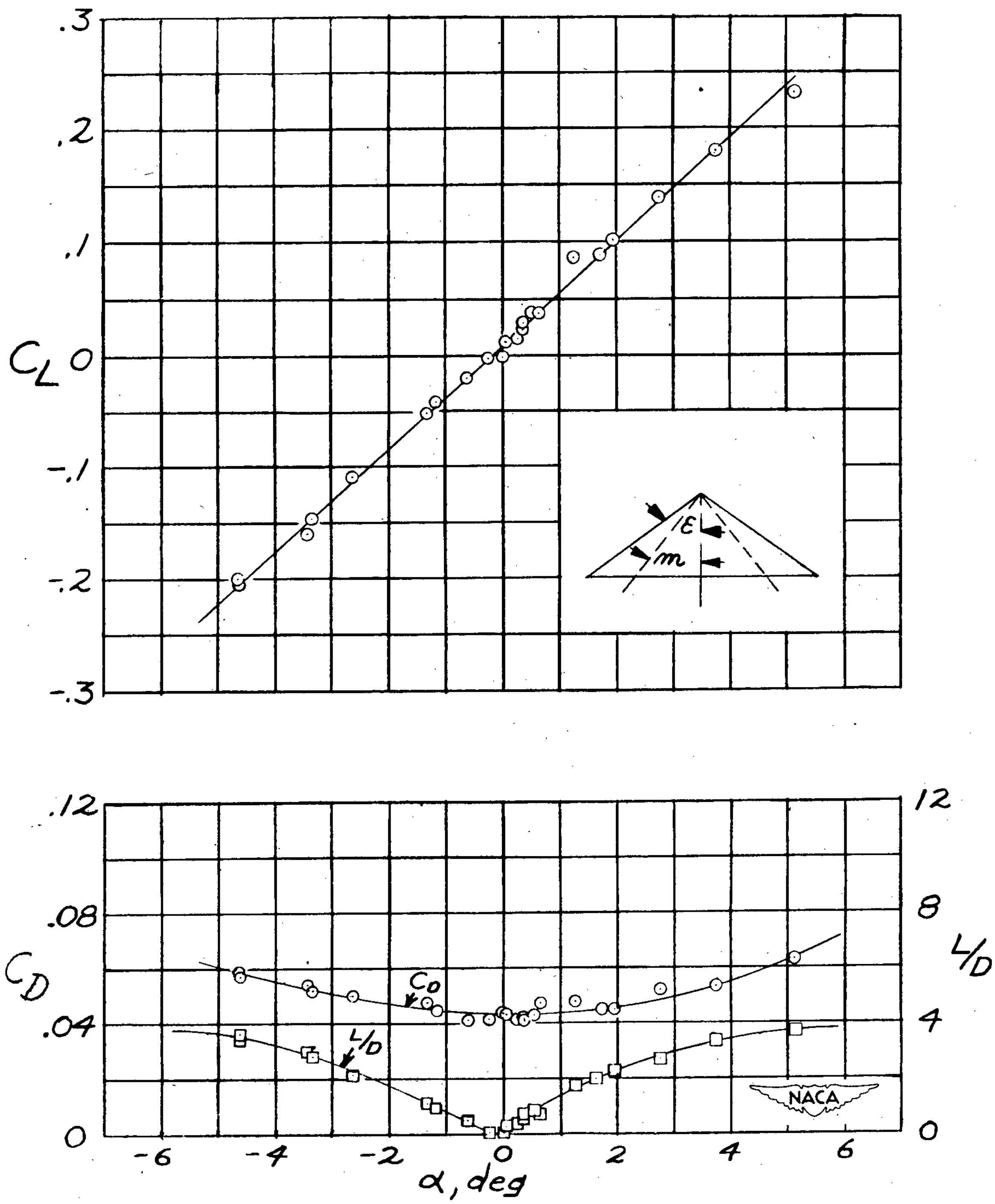
(g) Wing 7; R = 800,000.

Figure 3.- Continued.



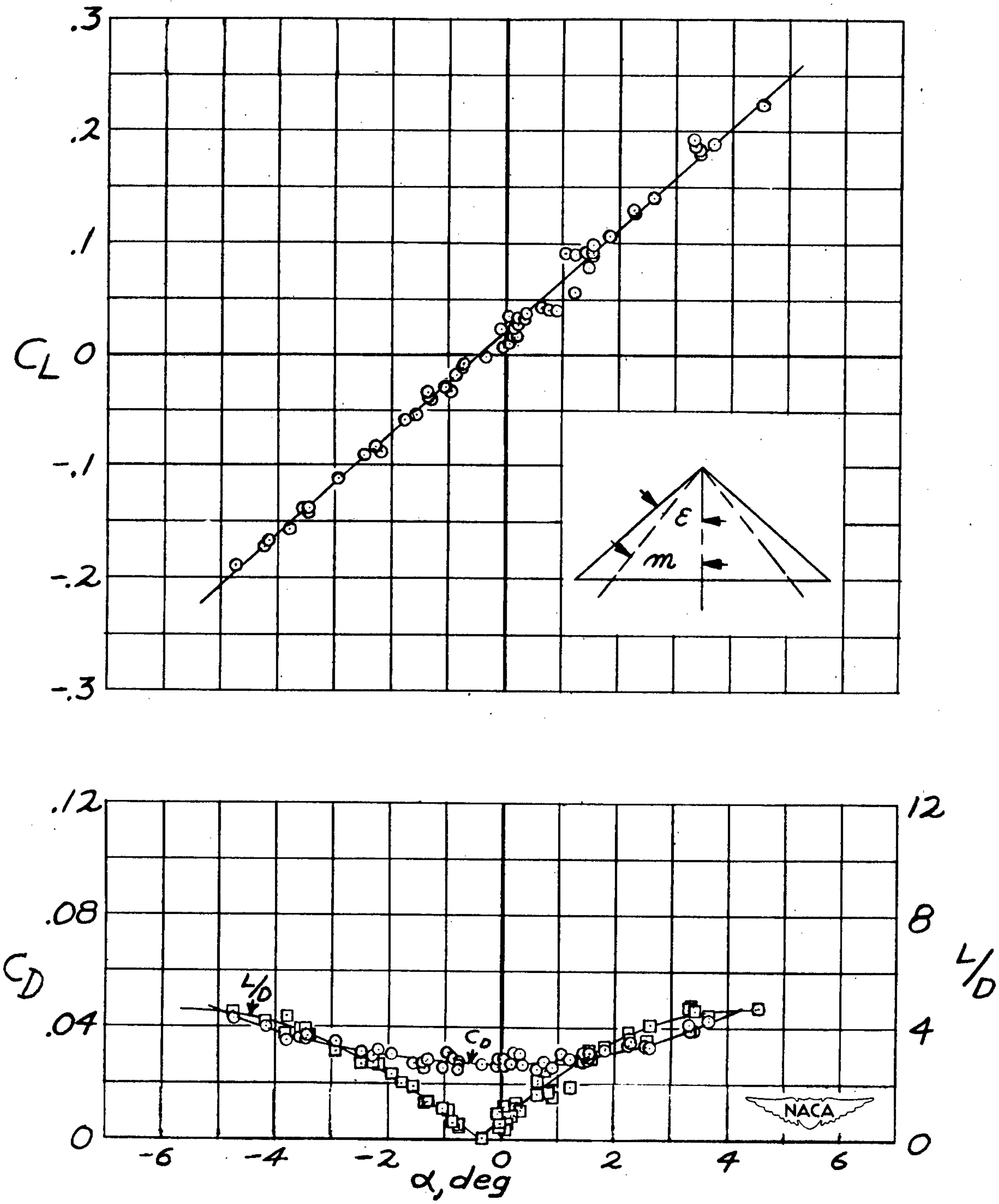
(h) Wing 8; R = 980,000.

Figure 3.- Concluded.



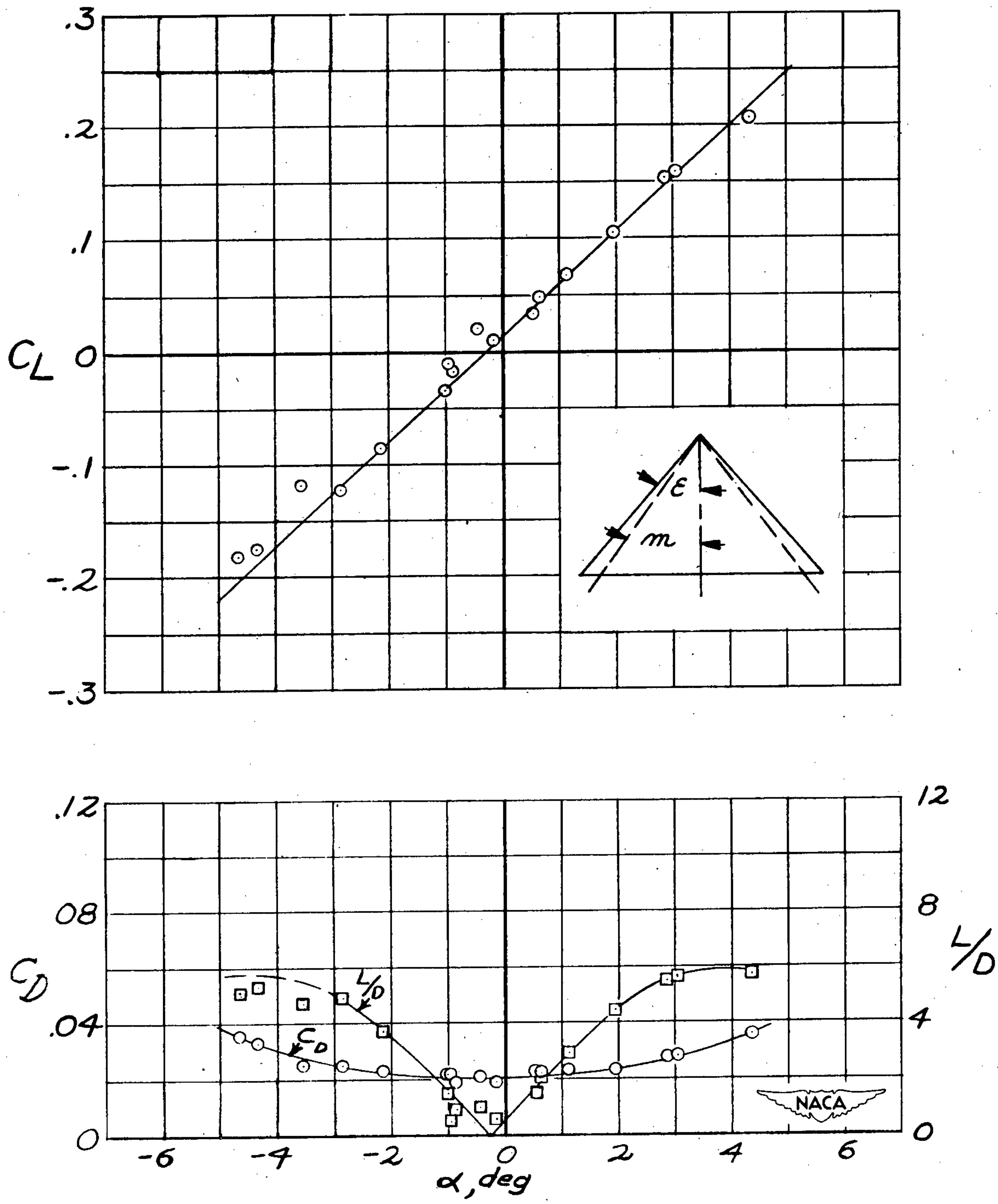
(a) Wing 1; R = 340,000.

Figure 4.- Triangular wing lift and drag test results for M = 1.71.



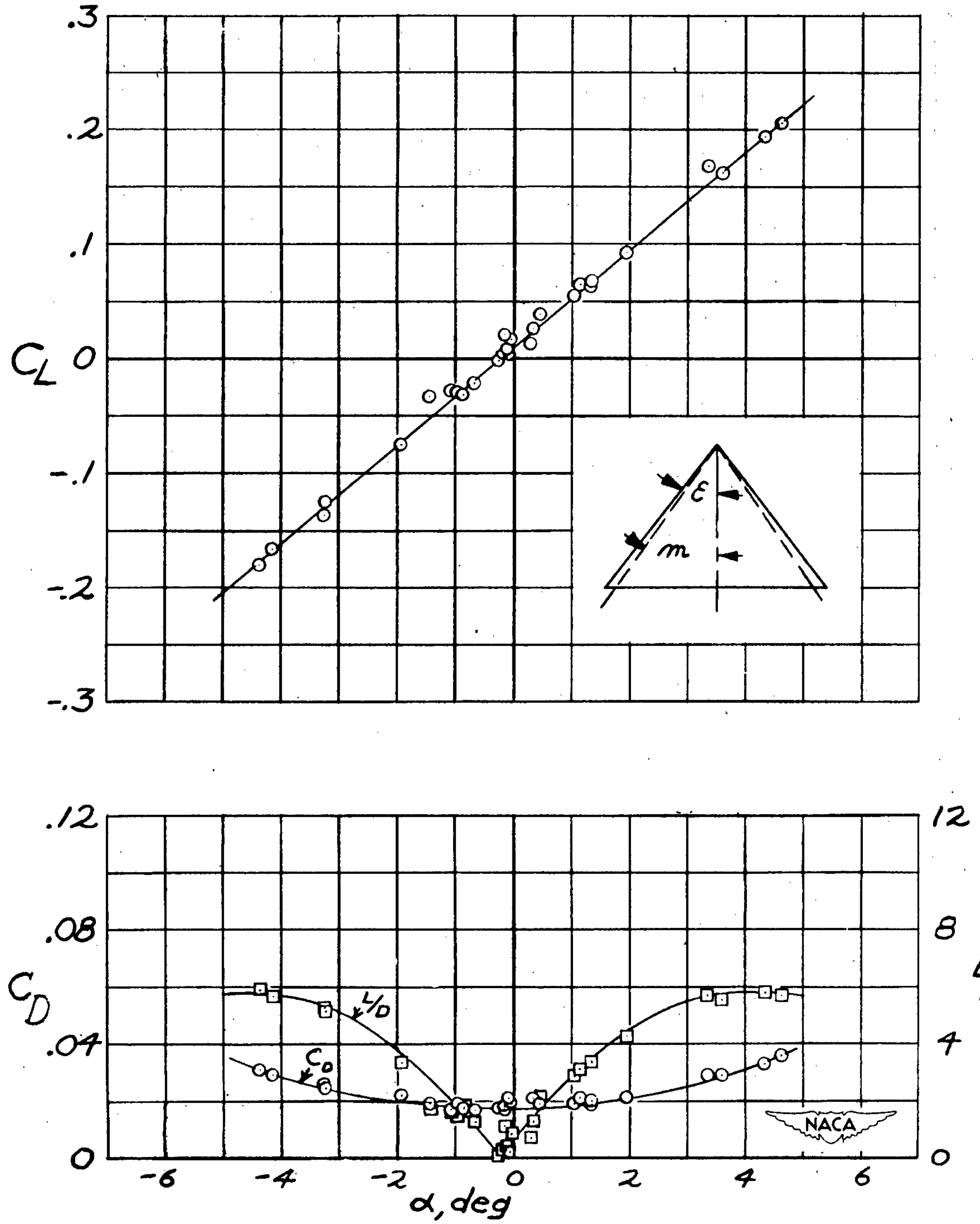
(b) Wing 2; R = 440,000.

Figure 4.- Continued.



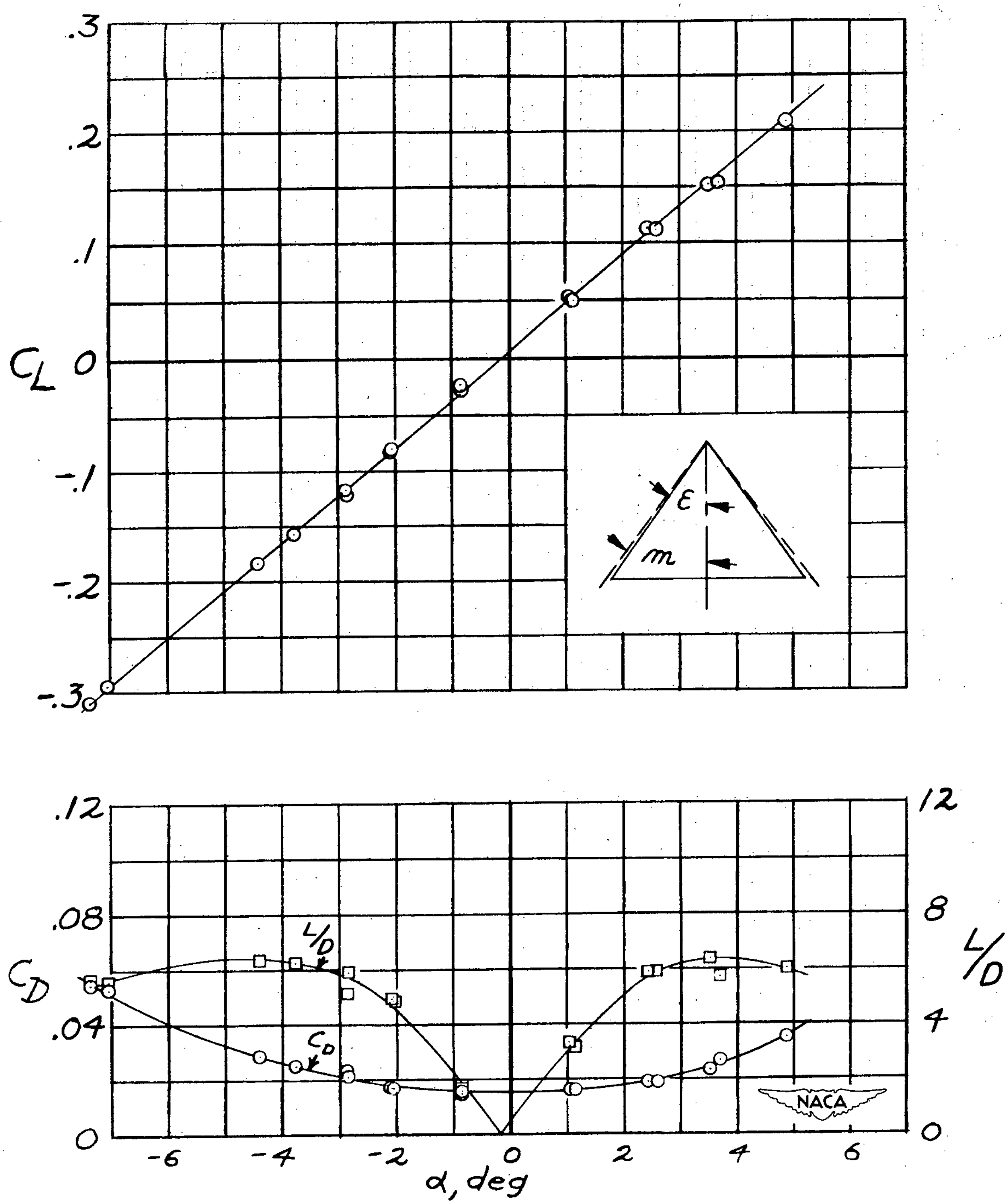
(c) Wing 3; R = 520,000.

Figure 4.- Continued.



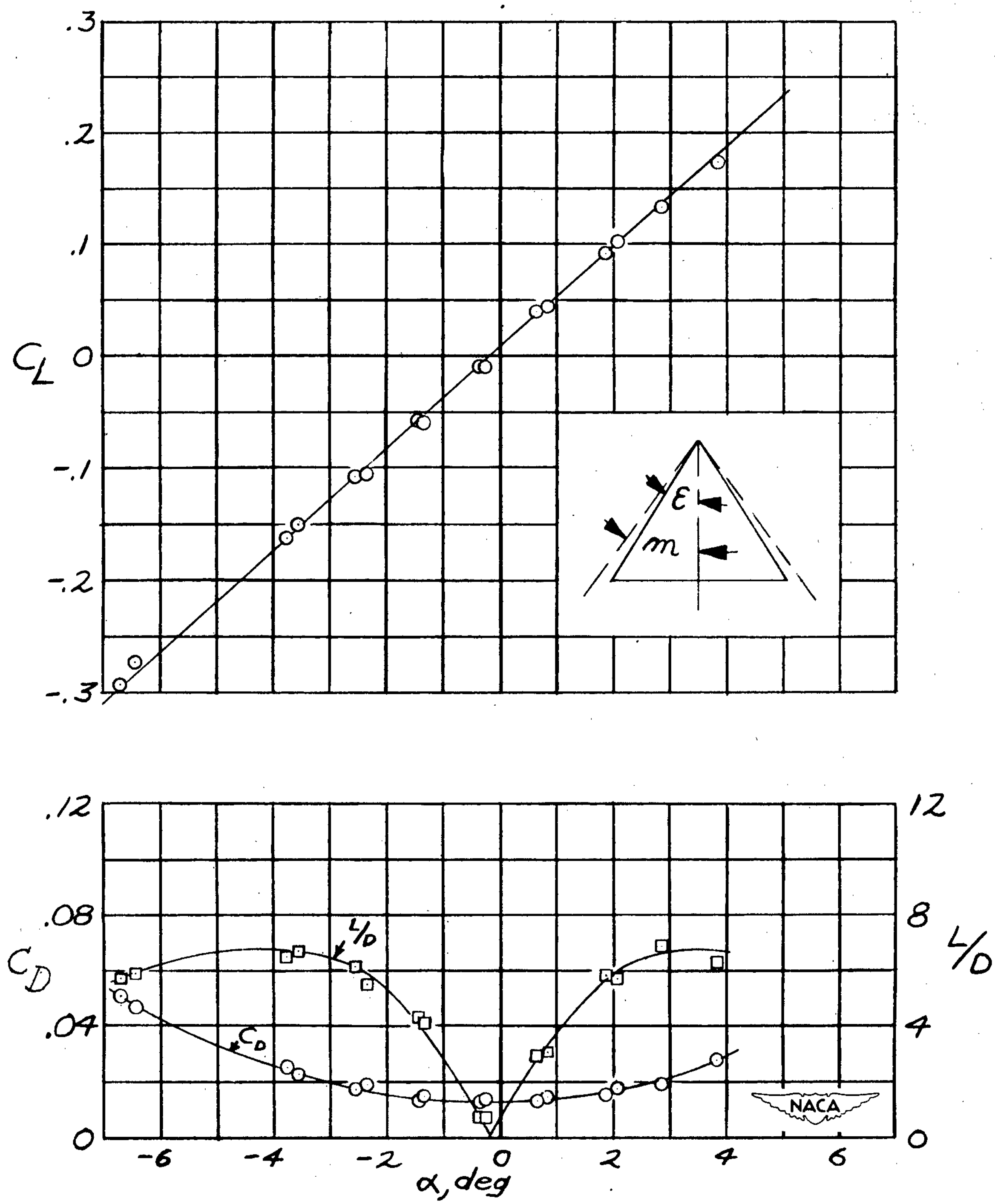
(d) Wing 4; R = 560,000.

Figure 4.- Continued.



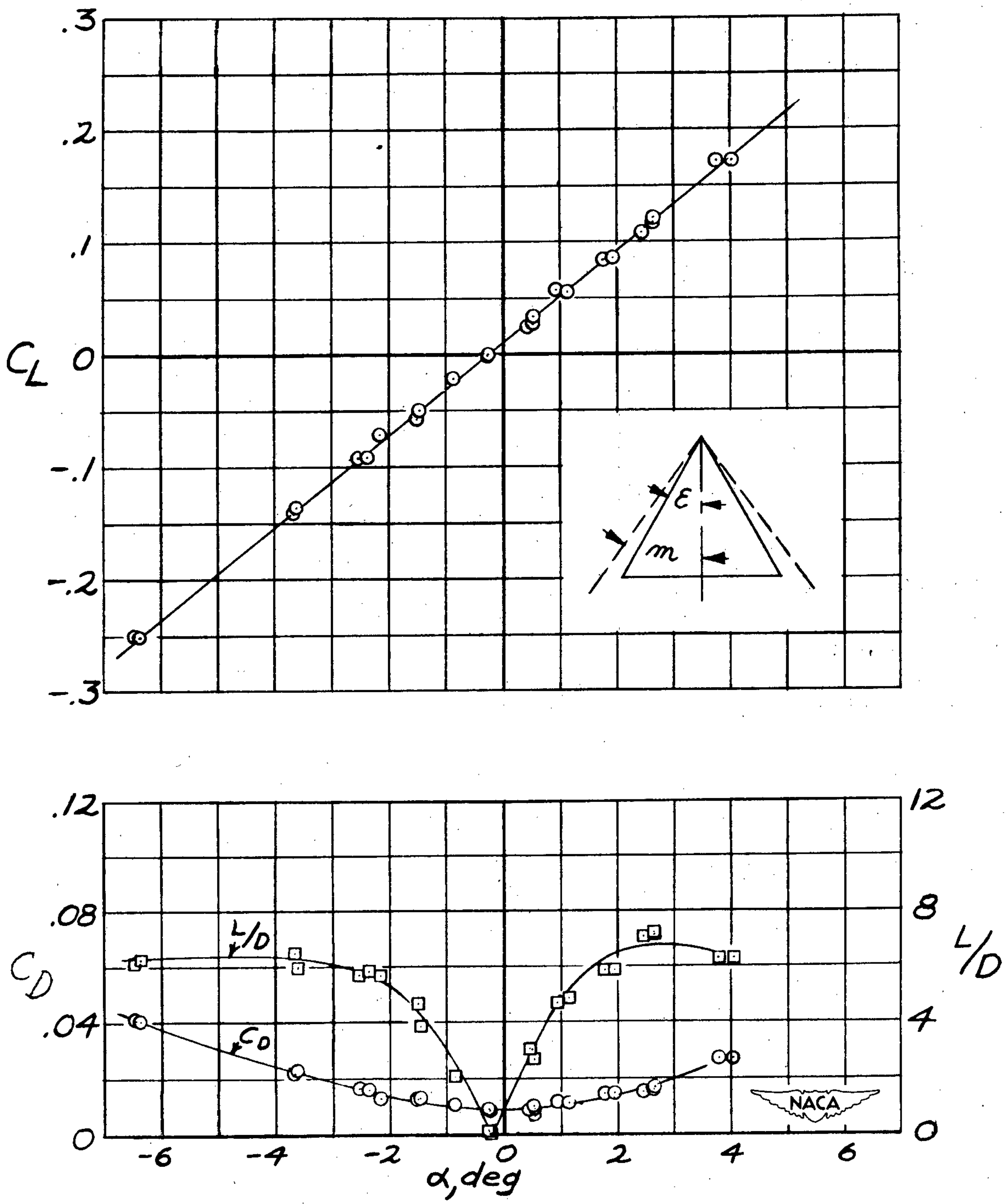
(e) Wing 5; R = 620,000.

Figure 4.- Continued.



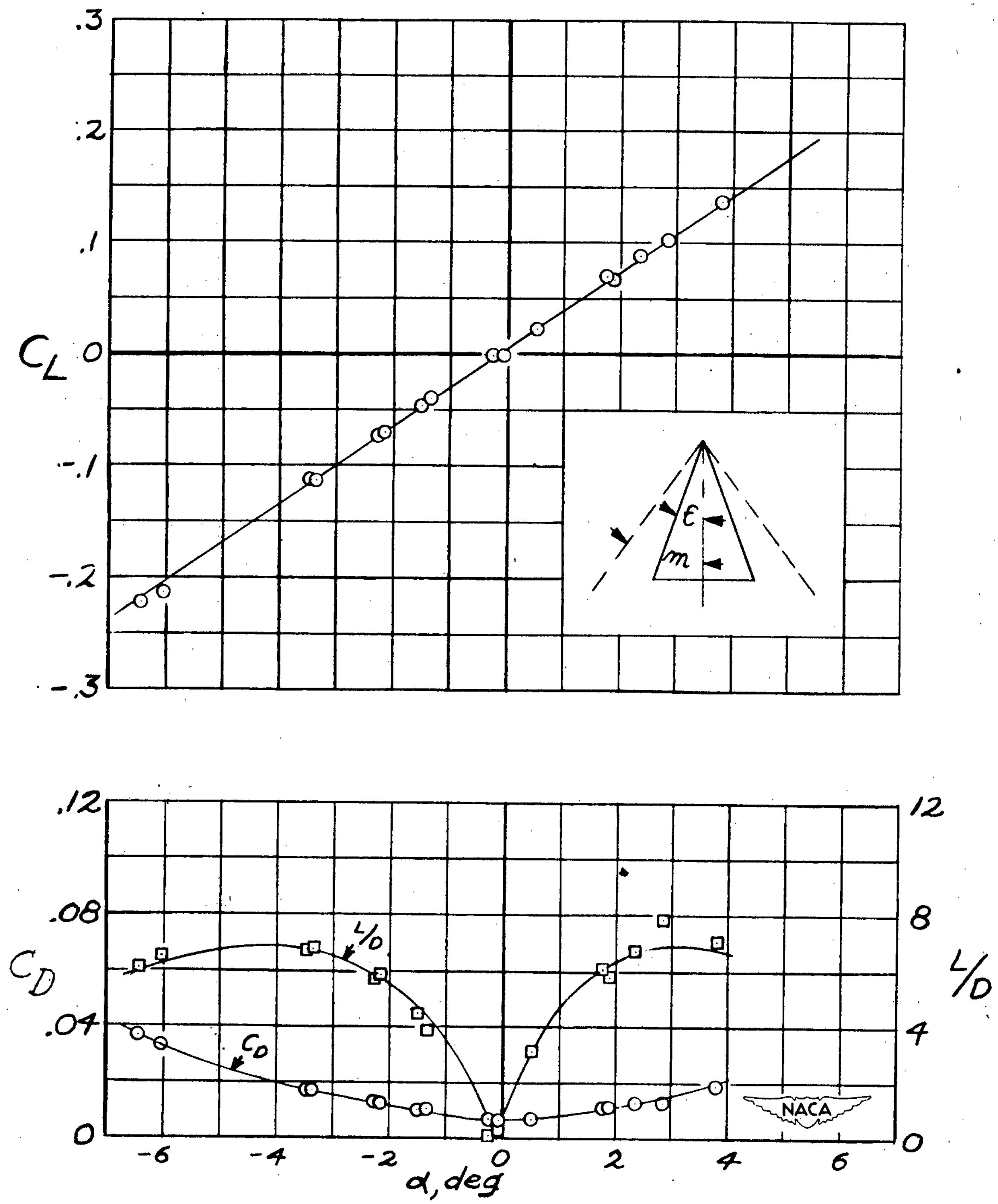
(f) Wing 6; R = 660,000.

Figure 4.- Continued.



(g) Wing 7; R = 760,000.

Figure 4.- Continued.



(h) Wing 8; R = 940,000.

Figure 4.- Concluded.

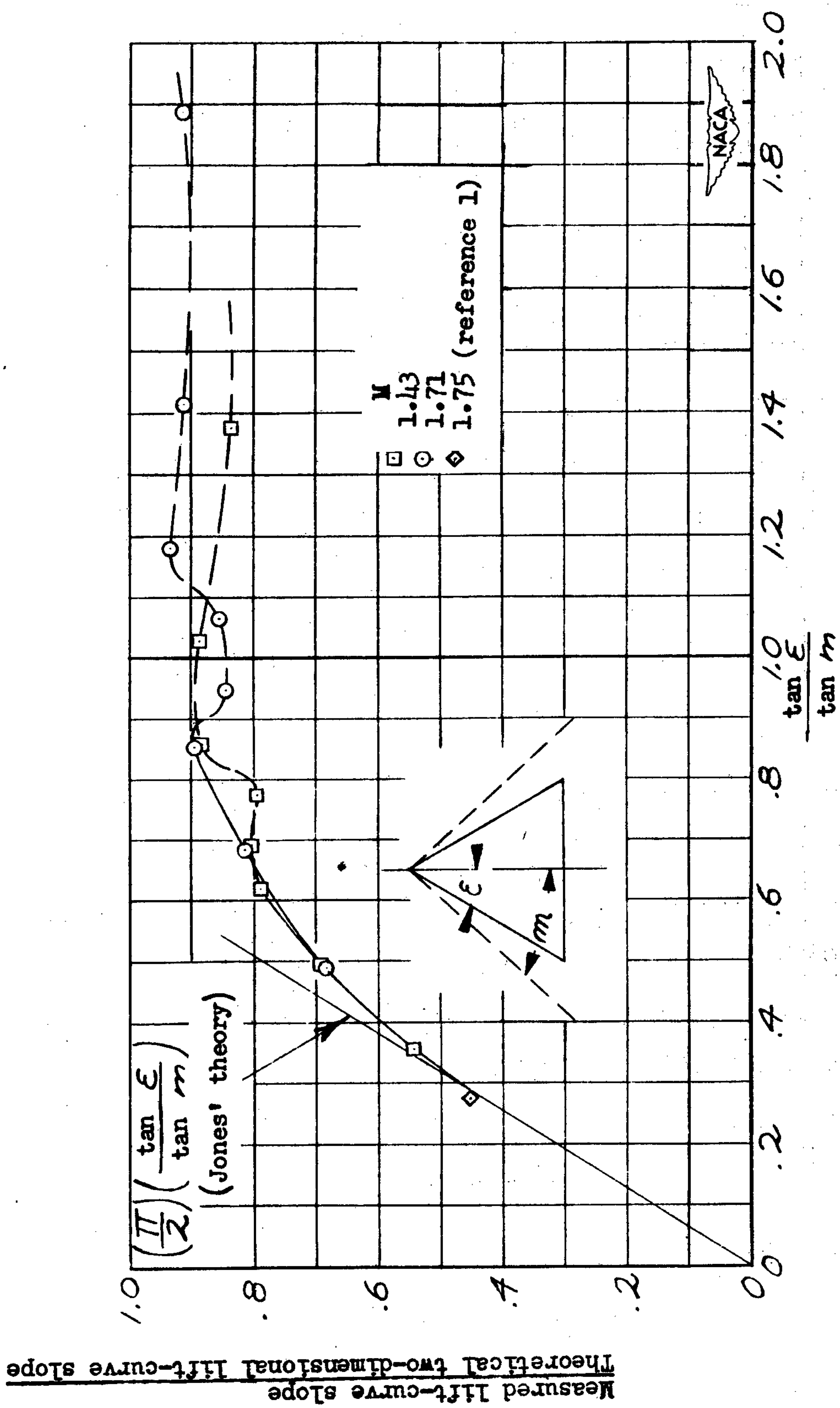


Figure 5.- Triangular-wing lift-curve-slope results from figures 3 and 4.

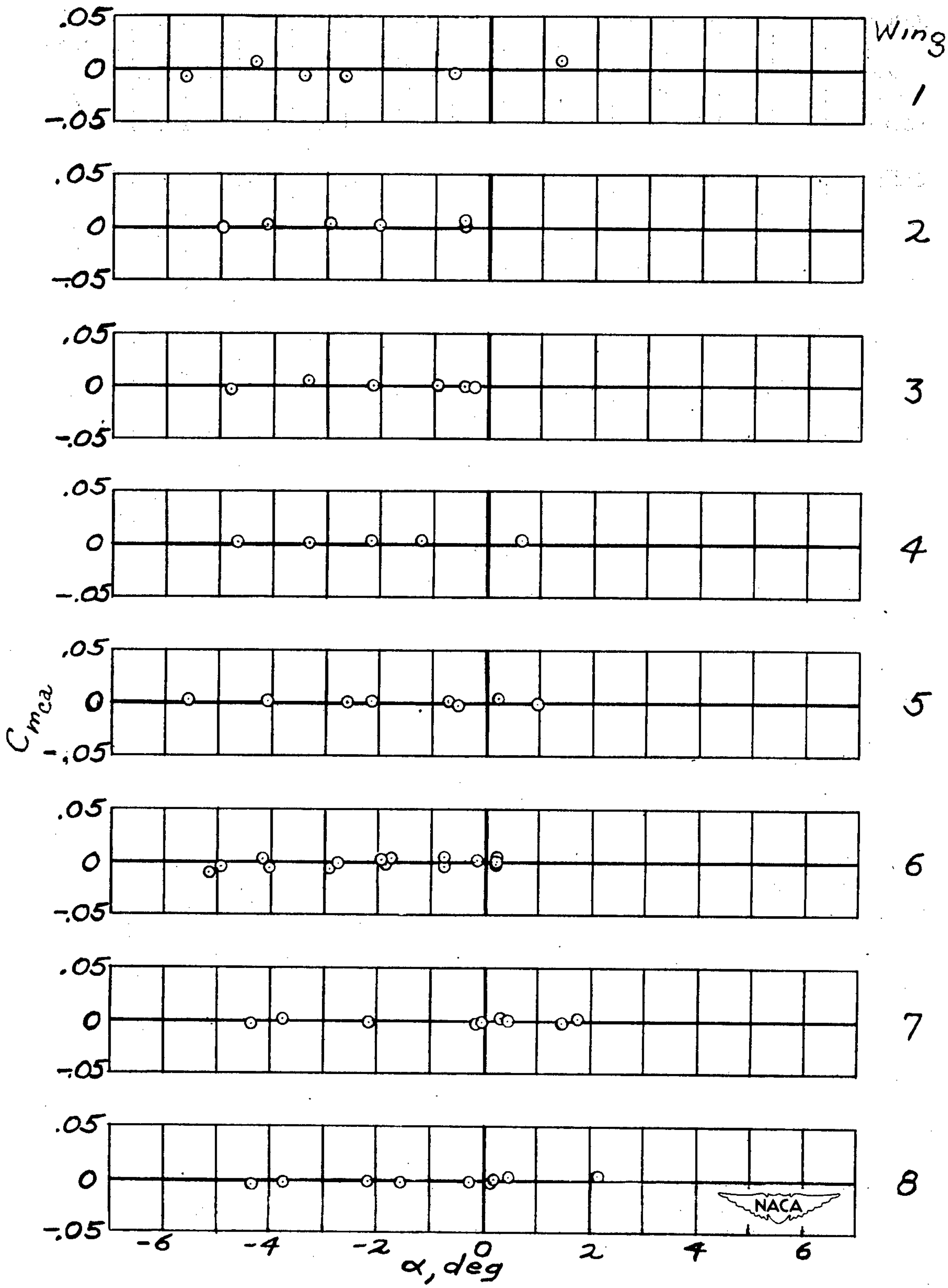


Figure 6.- Triangular wing moment test results for M = 1.43.

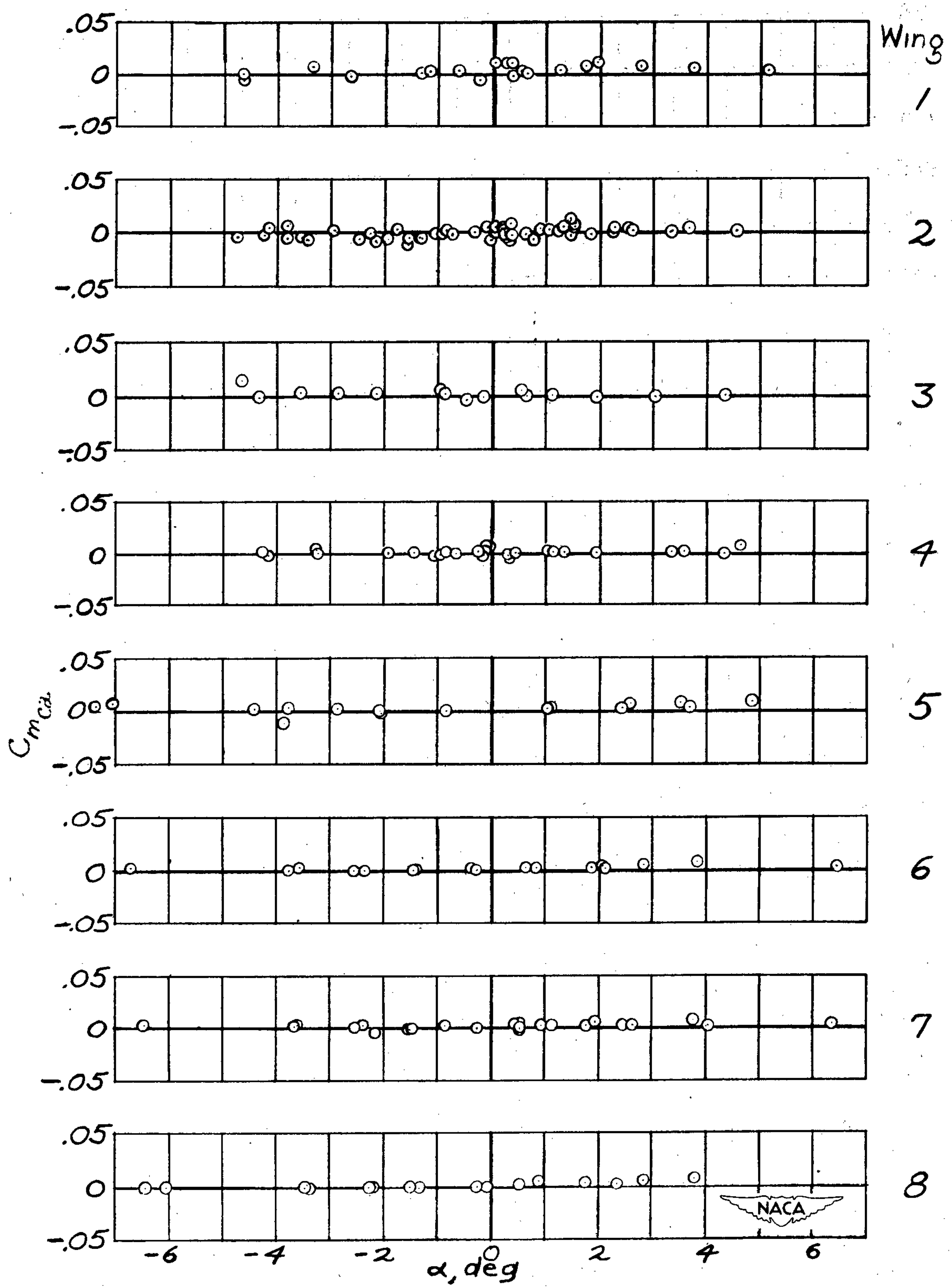


Figure 7.- Triangular wing moment test results for M = 1.71.

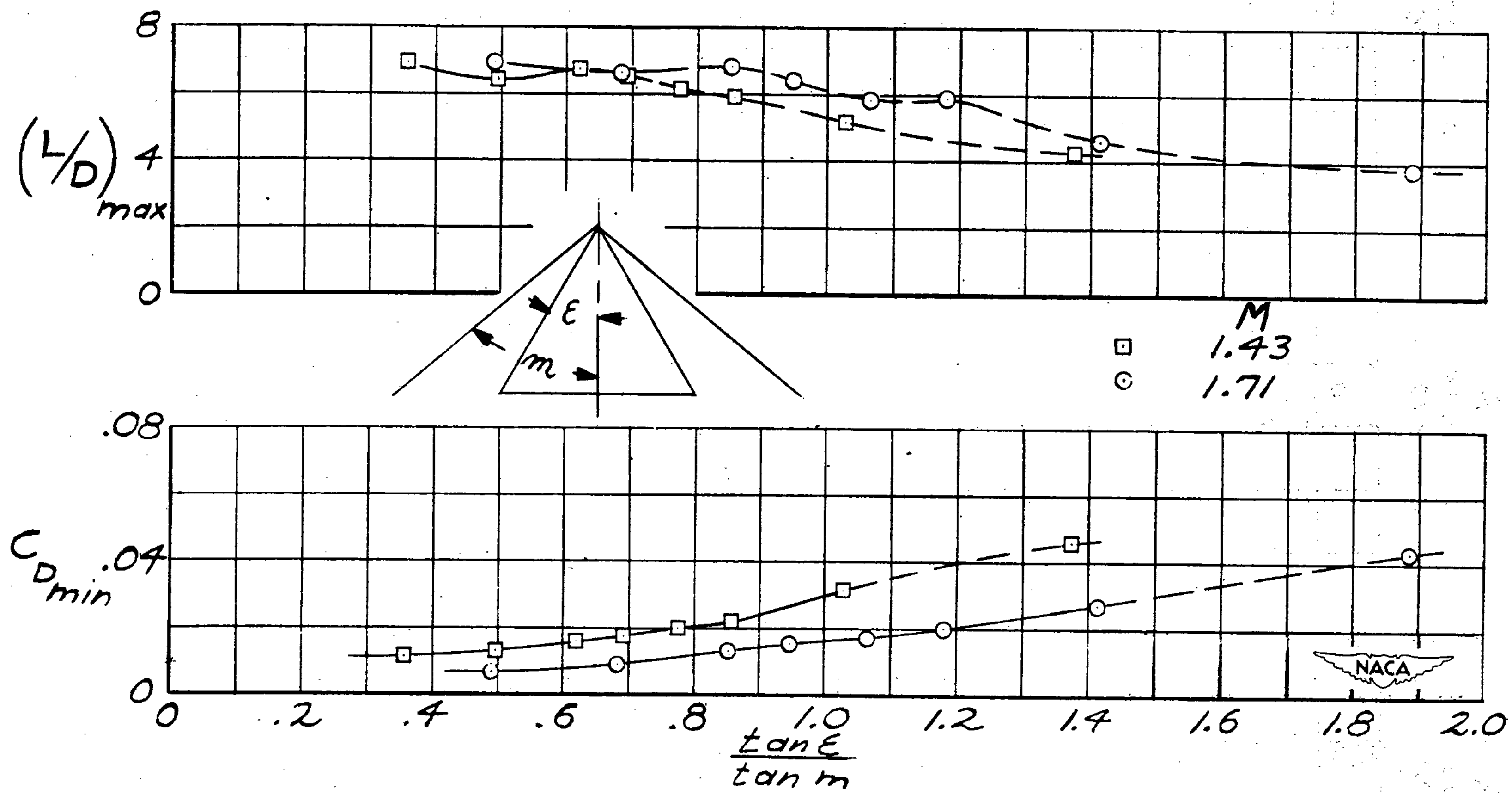
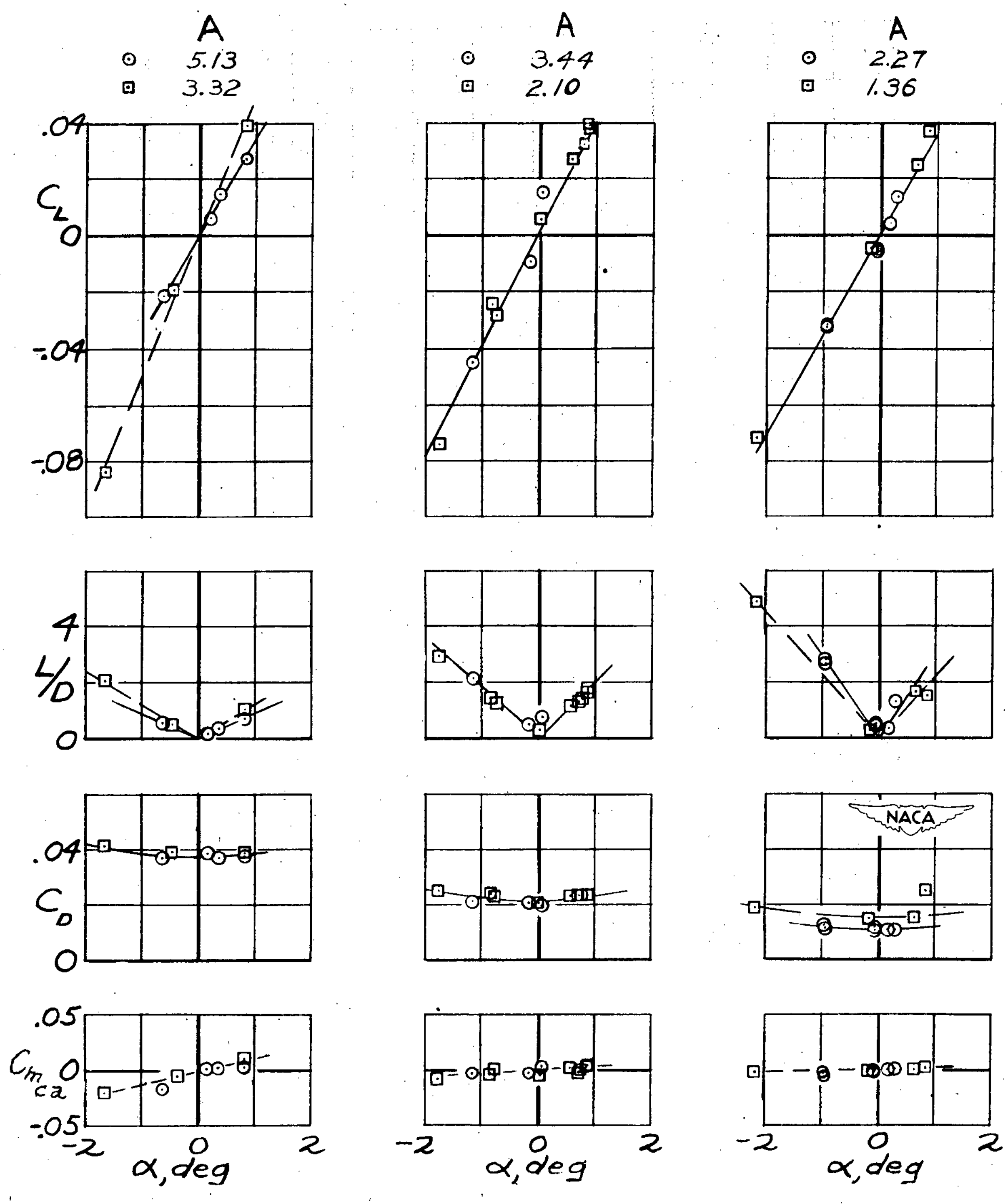


Figure 8.- Triangular wing minimum drag coefficient and maximum lift-drag ratio test results from figures 3(h) and 4(h).



(a) Λ = 45°.
(b) Λ = 55°.
(c) Λ = 63°.

Figure 9.- Sweptback-wing test results for M = 1.43.

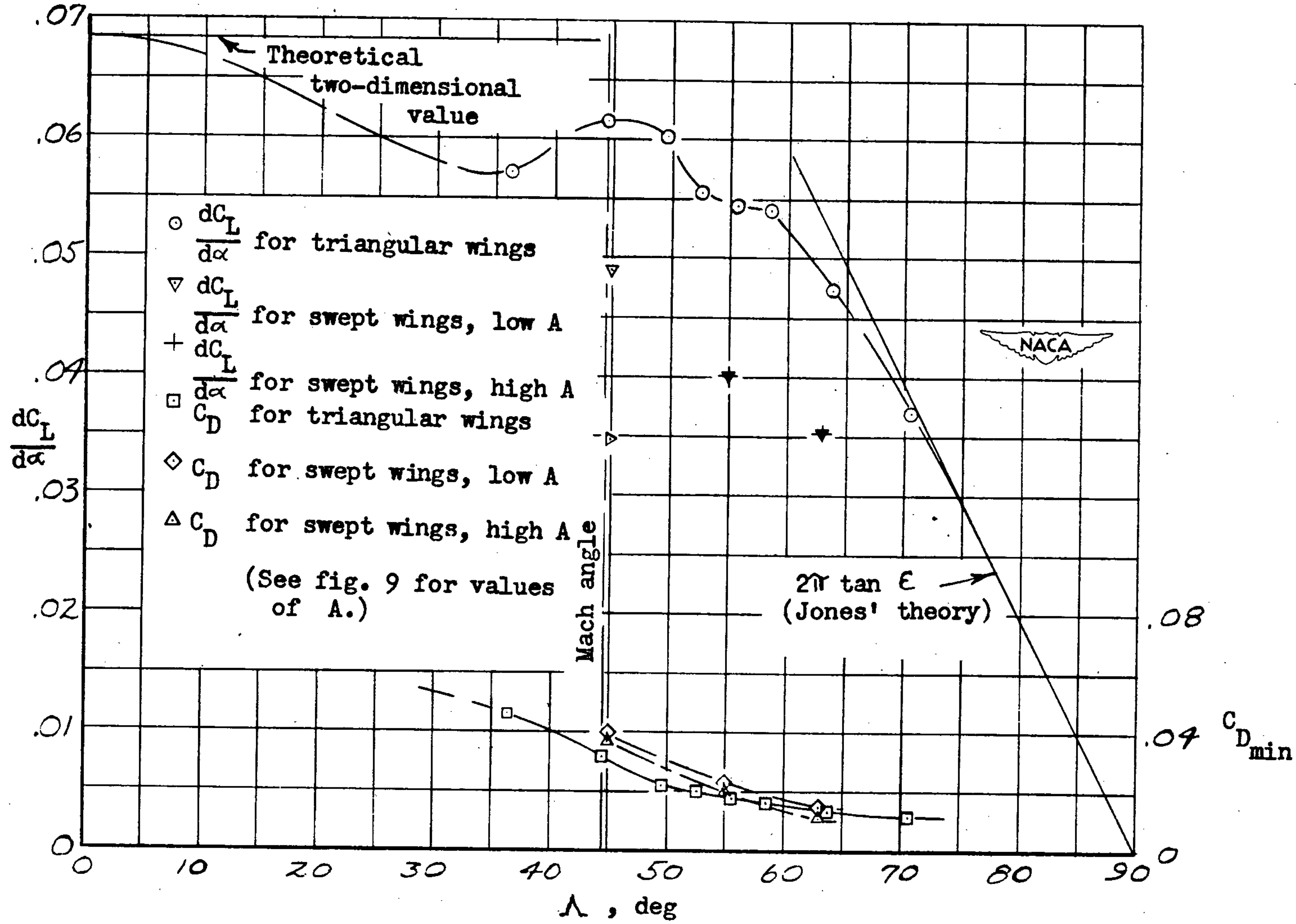
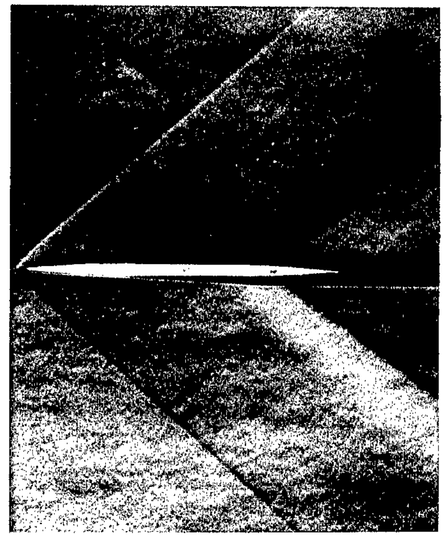
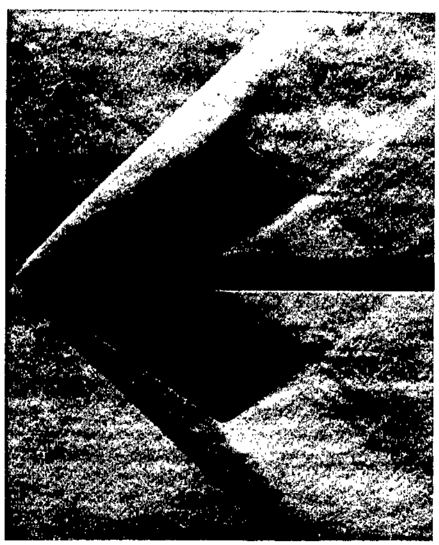


Figure 10.- Comparison of lift and drag results for triangular and sweptback wings of M = 1.43.

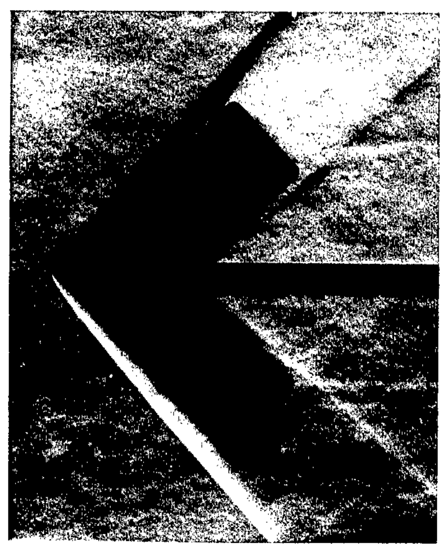
(a) $\Lambda = 63^{\circ}$.



(b) $\Lambda = 63^{\circ}$; side view.



(c) $\Lambda = 55^{\circ}$.



(d) $\Lambda = 45^{\circ}$.

Figure 11.- Schlieren photographs of low aspect ratio sweptback wings at $M = 1.55$.

NACA

L-60581