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RESEARCH MEMORANDUM

FORCE TESTS OF THREE THIN WINGS OF MODERATELY LOW ASPECT RATIO AT HIGH SUBSONIC MACH NUMBERS

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

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FORCE TESTS OF THREE THIN WINGS OF MODERATELY LOW ASPECT RATIO AT HIGH SUBSONIC MACH NUMBERS

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SUMMARY

An investigation was made in the Langley 24-inch high-speed tunnel to determine the effect of leading-edge shape and section profile on the aerodynamic characteristics of two thin wings of aspect ratio 4. Lift, drag, and pitching-moment data are presented for a range of angle of attack from -2° to 8° and a range of Mach number from 0.30 to approximately 0.90. Reynolds numbers are from 4.6×10^5 to 10.6×10^5. Test results are also included for a third wing having an aspect ratio of 3, a taper ratio of 0.4, and a 4.5-percent-thick modified-hexagonal section.

Lift and moment data showed no abrupt changes with Mach number for either of the two wings in the Mach number and lift-coefficient range where leading-edge-flow attachment would be expected to occur. The maximum lift-drag ratio of the wing having an NACA 66-006 section was about 20 percent greater than the maximum lift-drag ratio of the wing having a 6-percent-thick circular-arc section.

INTRODUCTION

Flow separation starting at the leading edge is found on the thin airfoils of current interest at moderate to high angles of attack in low-speed two-dimensional flow. As the Mach number increases, a high subsonic speed is reached at which attachment of the flow at the leading edge takes place more or less abruptly. Studies in two-dimensional flow (for instance, refs. 1 and 2) reveal that the principal force change accompanying this phenomenon is in the lift; in some cases an increase in lift is noted. The extent to which this attachment phenomenon is modified on finite wings is not well understood yet.

The purpose of the present tests is to determine in a preliminary way whether any important changes in force characteristics occur as a result of leading-edge-flow attachment at high subsonic speeds. An untapered wing of aspect ratio 4 was chosen for the tests. Because of the dependence of the leading-edge-flow phenomenon on leading-edge shape, two typical 6-percent-thick sections were selected, one (circular-arc) having a sharp leading edge and one (NACA 66-006) having a rounded leading edge. Test results are also included for a third wing which had a sharp-edge modified-hexagonal section of 4.5-percent thickness, an aspect ratio of 3, and a taper ratio of 0.4.

SYMBOLS

A	aspect ratio
C_L	lift coefficient
C_D	drag coefficient
$C_{m_{\bar{c}/4}}$	pitching-moment coefficient about quarter-chord point of mean geometric chord
C_{D_o}	minimum drag coefficient
ΔC_D	drag due to lift, $C_D - C_{D_o}$
L	lift
D	drag
M	Mach number
c	local chord
$\bar{c}$	mean geometric chord
α	angle of attack

APPARATUS AND TESTS

The tests of the present investigation were made in the Langley 24-inch high-speed tunnel, which is a nonreturn induction-type tunnel (ref. 3). The test section of the tunnel described in reference 3 has been modified by the installation of flats approximately 16 inches wide that reduce the width of the test section from 24 to 18 inches; thus, the shape of the test section is changed from circular to one approaching a rectangle. End plates were mounted flush with the flat sides of the test-section walls. The end plates had holes cut in them the same shape as the root section of the wing but with an 0.012-inch gap to permit forces to be transmitted without interference to the three-component recording balance. Since the forces encountered in these tests were small, a semispan model was mounted on each of the two side walls; thus, the magnitude of the forces was doubled and the accuracy of the data was increased.

The three wings investigated are shown in figure 1. Two of the wings had a trapezoidal plan form, a thickness-to-chord ratio of 6 percent, and an aspect ratio of 4. The parameter that was varied for these two wings was the airfoil section; one wing had an NACA 66-006 section and the other a 6-percent-thick circular-arc section. The trapezoidal-plan-form wings were mounted with the leading edge unswept. The modified-hexagonal-section wing had a thickness-to-chord ratio of 4.5 percent, a taper ratio of 0.4, and an aspect ratio of 3. This wing was mounted with its leading edge swept back 22.3°.

The two wings having an aspect ratio of 4 were constructed for a previous investigation and at that time the negatively raked tips were believed to be desirable. Subsequent investigations have made this tip shape obsolete and no significance is attached to it for this investigation.

Lift, drag, and pitching moment were measured on the three wings through a range of angle of attack from -2° to 8°, except for the modified-hexagonal-section wing which was not tested at -2°. The Mach number range extended from 0.30 to approximately 0.90, corresponding to Reynolds numbers from 4.6×10^5 to 10.6×10^5, based on the wing mean geometric chord.

During this investigation fluctuations of tunnel speed and forces acting on the model were encountered above a Mach number of about 0.70, at angles of attack of 6° and 8° for the three wings, and also at 4° for the 6-percent-thick circular-arc wing. Lift, drag, pitching moment, and velocity were continuously and simultaneously recorded for a period of about 10 seconds while the power input to the tunnel was held constant. Records obtained show that short periods of steady flow occurred during the fluctuations. The short-period steady-flow forces were used to extend the curves beyond a Mach number of about 0.70 for angles of attack of 4°, 6°, and 8°. All test points computed in this manner are indicated by flagged symbols and the curves faired through these points are shown dashed.

PRECISION

The initial angle of attack could be determined to within $\pm 0.1^\circ$ and subsequent changes in angle of attack were subject to an error of $\pm 0.01^\circ$. The data obtained were subject to errors that may be separated into two types. The first type of error arises from small inaccuracies in the calibration of the tunnel, calibration of the balance, and the inaccuracies in the balance at maximum sensitivity. Errors in coefficients due to this type of error are estimated to be of the following order:

C_L	± 0.010
C_D	± 0.002
$C_{m_{\bar{c}/4}}$	± 0.020

The second type of error arises from tunnel-wall interference and leakage at the model - tunnel-wall juncture. The data have been corrected for tunnel-wall interference by the method of reference 4. For this investigation the end-gap effects were believed to have been small in that repeat tests using 0.012- and 0.018-inch end gaps (25 percent and 38 percent of that used in ref. 5) showed negligible changes in the measured coefficients and, therefore, any corrections due to end gap have been neglected.

Calculations indicate that the 6-percent-thick wings (at an angle of attack of 0°) should choke the tunnel at a Mach number of about 0.95. This test Mach number of 0.95 was not obtained during the investigation, since the tunnel choked about 12 inches downstream of the test section as a result of water-vapor condensation. Static pressure at the tunnel wall along the test section and downstream of the model was observed for a given model configuration and indicated that this choking, which limited the maximum test Mach number, did not affect the pressures in the test section. The data presented are believed, therefore, to be relatively free of the usual tunnel effects attributed to the Mach number range within 0.03 of the choke Mach number.

RESULTS

The results of the investigation are presented in figures 2 to 8. Figure 2 shows the variation of lift coefficient with Mach number for constant angle of attack. Figure 3 presents the lift-curve slope of the three wings at zero lift and at a lift coefficient of 0.4. A comparison of the unsteady-flow boundaries of these wings with the two-dimensional normal-force break (ref. 1) is shown in figure 4. The variation of the quarter-chord pitching-moment coefficient with lift coefficient at various Mach numbers is shown by figure 5. Figure 6 presents the effect of Mach number on the minimum drag coefficient of the three wings. The effect of Mach number on drag due to lift at a lift coefficient of 0.4 is shown in figure 7. The variations of lift-drag ratio with Mach number for the three wings are shown in figure 8.

DISCUSSION

Flow Separation and Reattachment

Two-dimensional-flow data of reference 1 for thin airfoils with sharp or moderately sharp leading edges at a constant angle of attack showed that the reattachment of the flow at Mach numbers between 0.70 and 0.80 was accompanied by a moderate increase in lift coefficient. A small to imperceptible increase in the lift coefficient occurred at Mach numbers from 0.70 to 0.80 at angles of attack of about 6° and 8° in this three-dimensional-flow investigation. (See fig. 2.) A reduction in the magnitude of the effects of flow reattachment on a finite wing as compared with an infinite wing might be a consequence of a reduction in both the extensiveness of separation and in shock strength as a result of tip flow (ref. 6). The results of other high-speed tests of thin wings (for example, ref. 7) are in accord with the present results as regards the absence of any appreciable force changes associated with leading-edge-flow attachment.

Force Coefficients

Lift coefficient.- For a constant angle of attack, the wing with an NACA 66-006 profile produces an increase in lift coefficient with an increase in Mach number for all angles of attack tested up to a Mach number of about 0.87 (fig. 2(a)). The increase in lift coefficient with Mach number for the circular-arc-section wing and the modified-hexagonal-section wing is negligible for angles of attack from 0° to about 2° (figs. 2(b) and 2(c)); however, for angles of attack from approximately 2° to 8°, there is an increase in lift coefficient with an increase in Mach number. As a consequence, the lift-curve slope at zero lift (fig. 3) for the NACA 66-006 section wing shows an increase with Mach number, whereas the other two wings indicate minor increases in the lift-curve slope at zero lift throughout the Mach number range investigated. At a lift coefficient of 0.4, however, the lift-curve slopes of both 6-percent-thick wings indicate an increase in lift-curve slope with Mach number up to a Mach number of 0.85, with a more rapid increase occurring at a Mach number of about 0.81 for the circular-arc-section wing. The lift-curve slope of the modified-hexagonal-section wing at a lift coefficient of 0.4 remains relatively constant for a range of Mach number from 0.30 to 0.75. As the Mach number was increased above 0.75, the lift-curve slope increased moderately.

Unsteady flows were encountered during this investigation and caused violent shaking of the model and fluctuations of tunnel speed. Visual observation of the flow during a high-humidity run showed, by water-vapor condensation, that the width of the wake from the upper surface was fluctuating in the Mach number and lift-coefficient range where unsteady forces were encountered. The fluctuation in the width of the wake is probably a result of a chordwise oscillation in the separation point as discussed in reference 8. For low angles of attack, unsteady flows were not encountered within the Mach number range investigated, and this result is in accordance with the data of references 8 and 9.

The boundaries of the unsteady flows encountered on the three wings are compared with the two-dimensional normal-force breaks of reference 1 in figure 4. The unsteady-flow boundaries occurred at a Mach number approximately 0.05 less than the two-dimensional normal-force break and about 0.10 below the lift-break Mach number for these wings in three-dimensional flow. The two-dimensional normal-force-break Mach number (taken as the Mach number at which the inflection point occurs on the curve of normal force against Mach number), plus a Mach number increment of 0.06, has been suggested as one criterion for airplane buffeting (ref. 10). The significance of the unsteady-flow boundaries in the present tests is, of course, open to some question in view of the interaction between the unsteady wing flow and the tunnel flow which was evidenced by appreciable fluctuations in tunnel airspeed; thus, the significance of the boundaries shown in figure 4 as regards the buffeting of a similar wing in free air is uncertain.

Pitching-moment coefficient.- The variation of quarter-chord pitching-moment coefficient with lift coefficient for various Mach numbers is shown in figure 5. The moment-curve slope for the three wings is positive at lift coefficients below 0.1 throughout the Mach number range. The lift coefficient at which the moment-curve slope for the wing having the NACA 66-006 section becomes zero increases from a lift coefficient of 0.1 at a Mach number of 0.3 to a lift coefficient of approximately 0.5 at Mach numbers of the order of 0.75 and 0.80. At lift coefficients in excess of the value where the slope is zero, the slope becomes negative. This negative slope represents a rearward movement of the center of pressure which becomes more rapid at the higher Mach numbers. A similar behavior of the moment-curve slope is noted for the wing having the modified-hexagonal section (fig. 5(c)), with the rearward movement of the center of pressure occuring at a lift coefficient between 0.3 and 0.5 for a range of Mach number from 0.70 to 0.85. The rearward movement of the center of pressure for the modified-hexagonal-section wing is more pronounced than that observed for the wing having the NACA 66-006 section. This rearward movement of the center of pressure is similar to the results obtained on an aspect-ratio-4 wing having a modified-hexagonal section investigated at a Reynolds number of 1×10^6 and reported in reference 7. The more rapid movement of the center of pressure on the hexagonal-section wing as compared to that of the NACA 66-006 section wing might be attributed to the small pressure gradients which might be expected to exist along the central portion of the hexagonal section that would allow a more rapid chordwise shock movement or pressure change to occur. For the wing having a circular-arc section, the rearward movement of the center of pressure does not appear to be as pronounced as for the other two wings. The less rapid rearward movement of the center of pressure for this wing may be attributed to the larger trailing-edge angle.

Drag coefficient.- The minimum drag coefficient in figure 6 is approximately the same (about 0.0070) for the three wings through a range of Mach number from 0.40 to 0.70. The Mach number for drag rise at zero lift for each of the three wings is about 0.87.

The drag due to lift at a lift coefficient of 0.4 for the three wings investigated is shown in figure 7. These experimental drags are compared with theoretical values of drag due to lift with the resultant force acting normal to the relative wind $C_L^2/\pi A$ and with the resultant force acting normal to the chord line $C_L \tan \alpha$. As would be expected, the experimental value falls between the two theoretical values and thus gives an indication of the amount of leading-edge suction experienced by each of the wings at a lift coefficient of 0.4. The experimental drag due to lift for the two 6-percent-thick wings is in close agreement for a range of Mach number from 0.30 to 0.50. For a range of Mach number from 0.50 to 0.80, however, the wing having an NACA 66-006 section produced less drag due to lift. For Mach numbers above 0.80, the drag due to lift for the wing having the NACA 66-006 section increased to a value greater than that of the aspect-ratio-4 wing having a 6-percent-thick circular-arc profile.

The 4.5-percent-thick wing is inferior to both the 6-percent-thick wings in drag due to lift throughout the Mach number range investigated. The reduction in aspect ratio or thickness ratio could be partly responsible, since both tend to increase the drag due to lift, particularly at Mach numbers less than the drag-rise Mach number.

The maximum lift-drag ratio of the three wings occurs at a Mach number of about 0.70 and at lift coefficients from 0.2 to 0.3 (fig. 8).

The maximum lift-drag ratio of the wing having an NACA 66-006 section is about 20 percent greater than the lift-drag ratio of the wing having a 6-percent-thick circular-arc section and about 45 percent greater than the maximum lift-drag ratio of the 4.5-percent-thick modified-hexagonal-section wing with the lower aspect ratio.

CONCLUDING REMARKS

An investigation to determine the effect of leading-edge shape and section profile on the aerodynamic characteristics of two thin wings in a range of Mach number from 0.30 to about 0.90 indicated the following results:

1. Lift and moment data showed no abrupt changes with Mach number for either of the two wings in the Mach number and lift-coefficient ranges where leading-edge-flow attachment would be expected to occur.

2. The maximum lift-drag ratio of the wing having an NACA 66-006 section was about 20 percent greater than the maximum lift-drag ratio of the wing having a 6-percent-thick circular-arc section.

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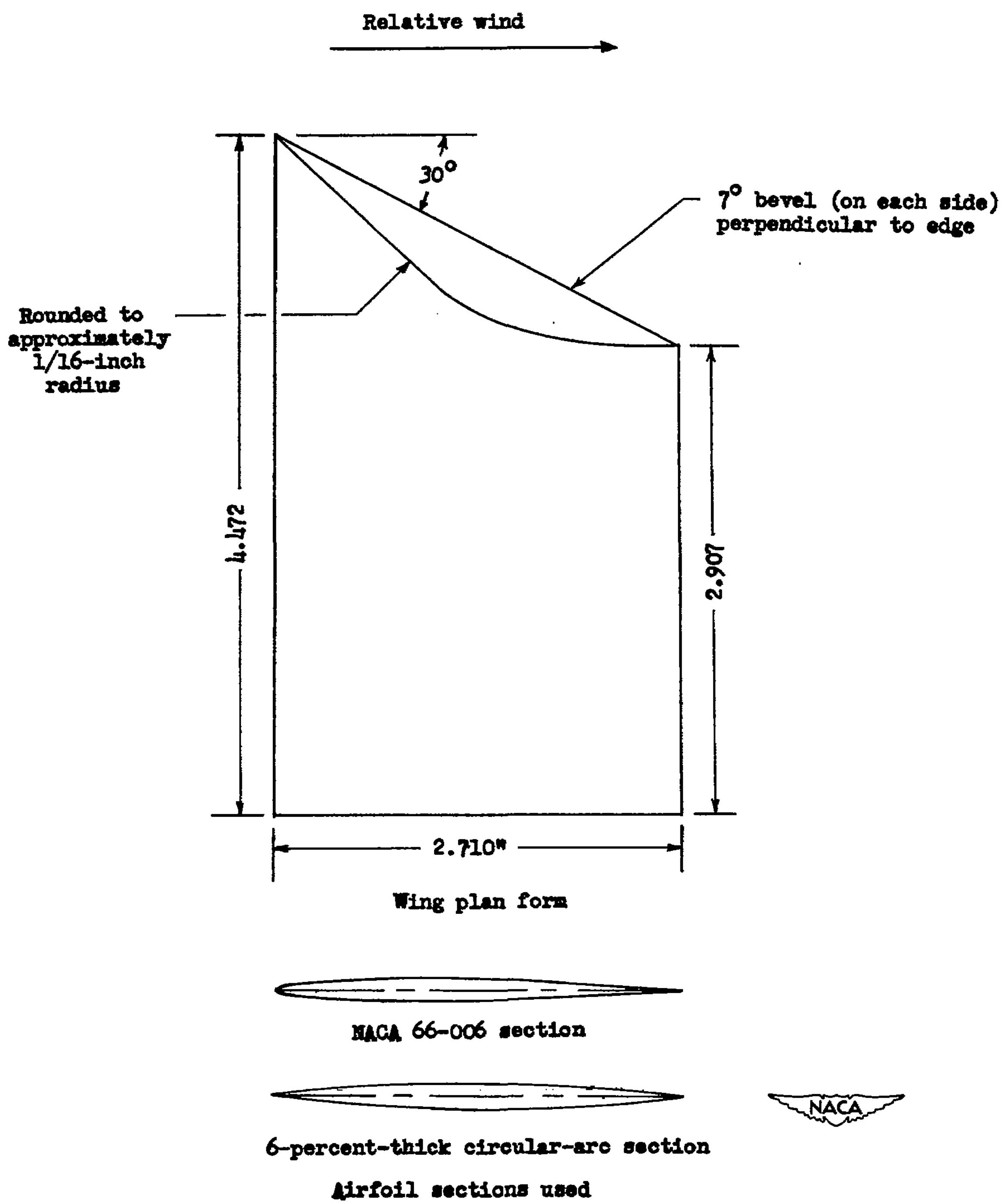
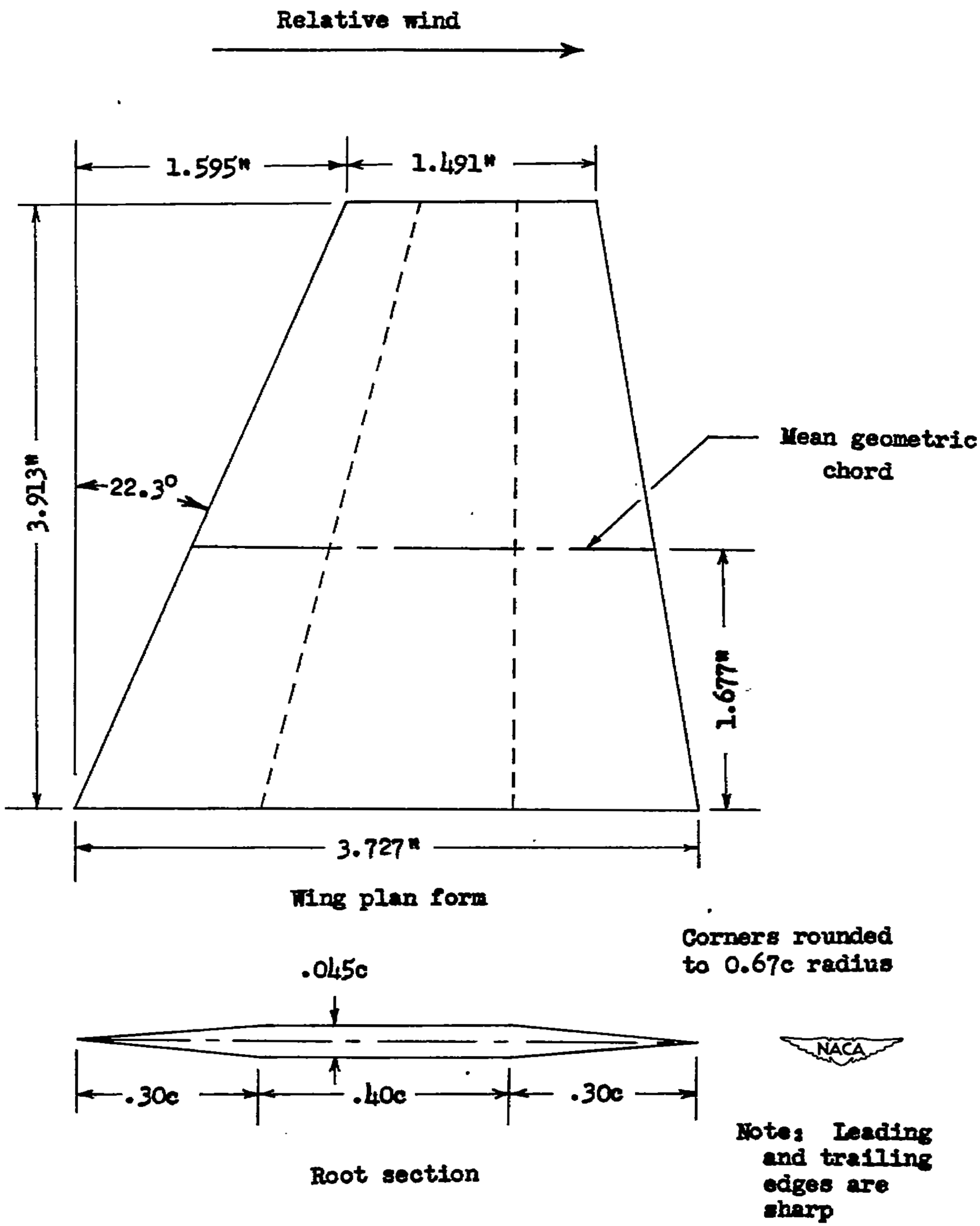


(a) Aspect-ratio-4 wings.

Figure 1.- Semispan models of low aspect ratio tested in the Langley 24-inch high-speed tunnel.



(b) Aspect-ratio-3 wing.

Figure 1.- Concluded.

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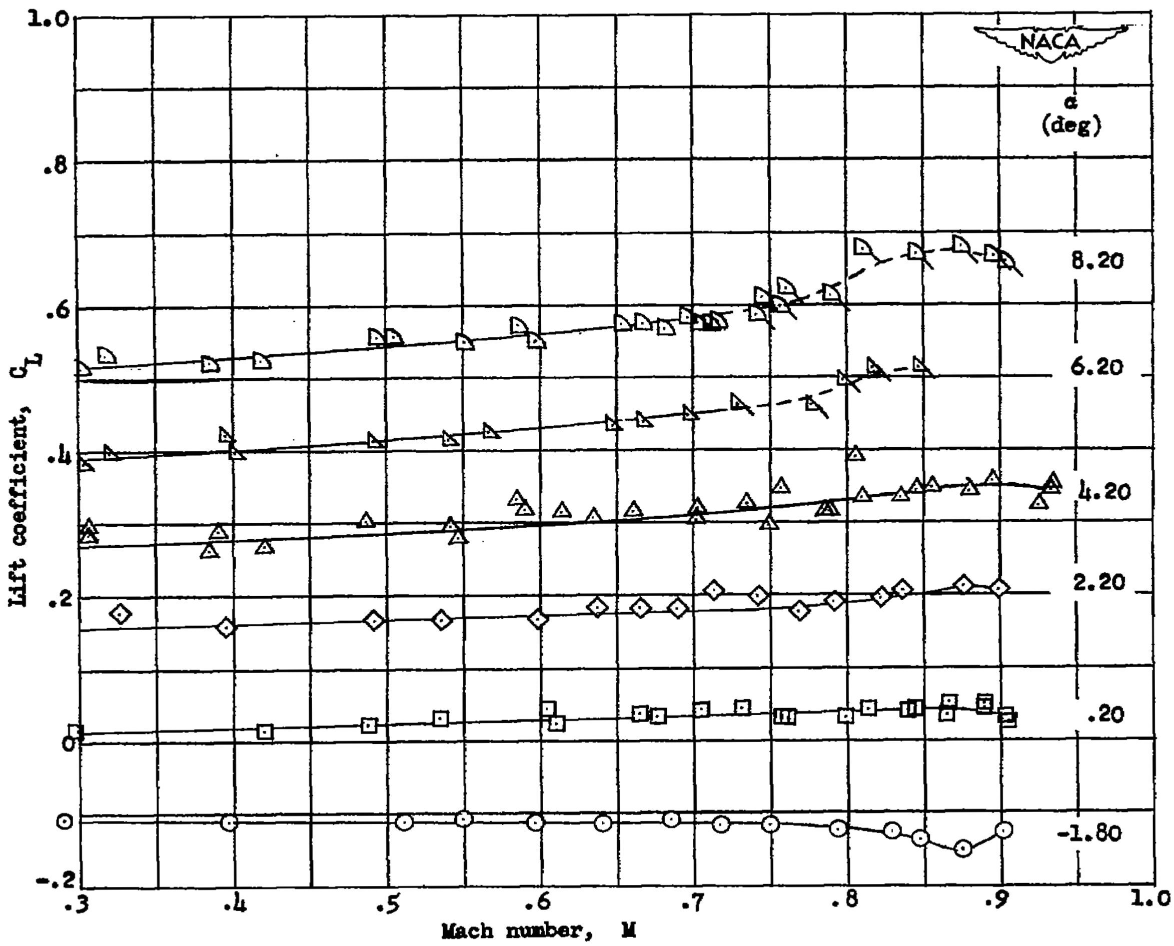
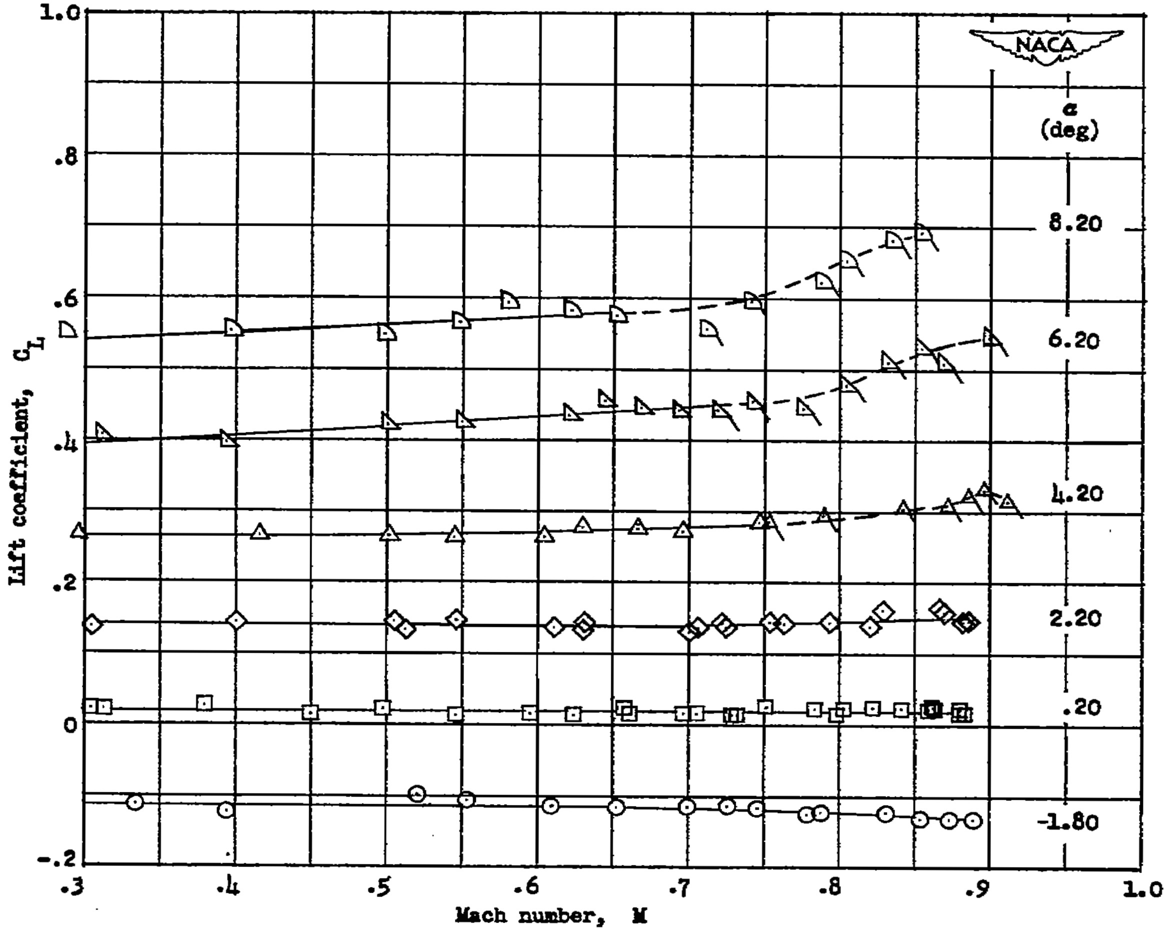


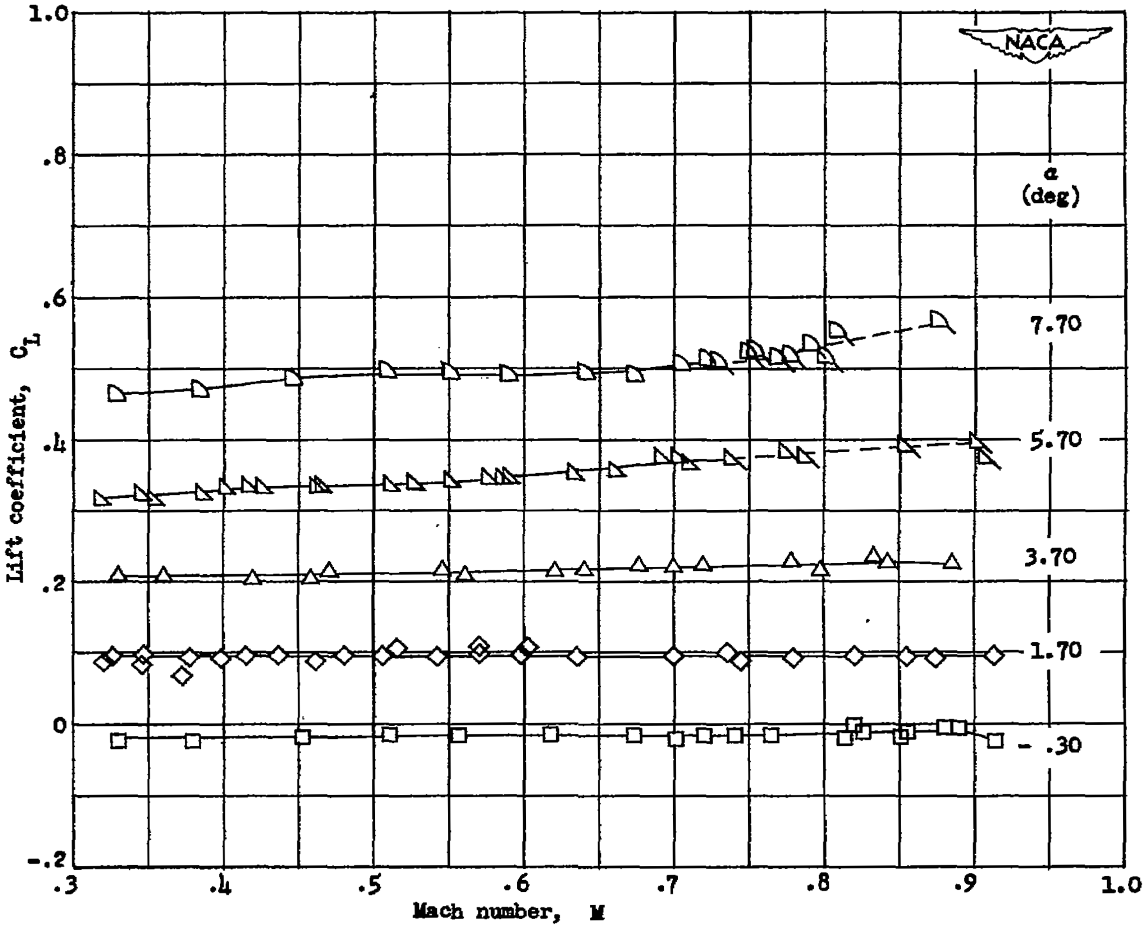
(a) NACA 66-006 section wing. A = 4.

Figure 2.- Variation of lift coefficient with Mach number.



(b) The 6-percent-thick circular-arc-section wing. A = 4.

Figure 2.- Continued.



(c) The 4.5-percent-thick modified-hexagonal-section wing. A = 3.

Figure 2.- Concluded.

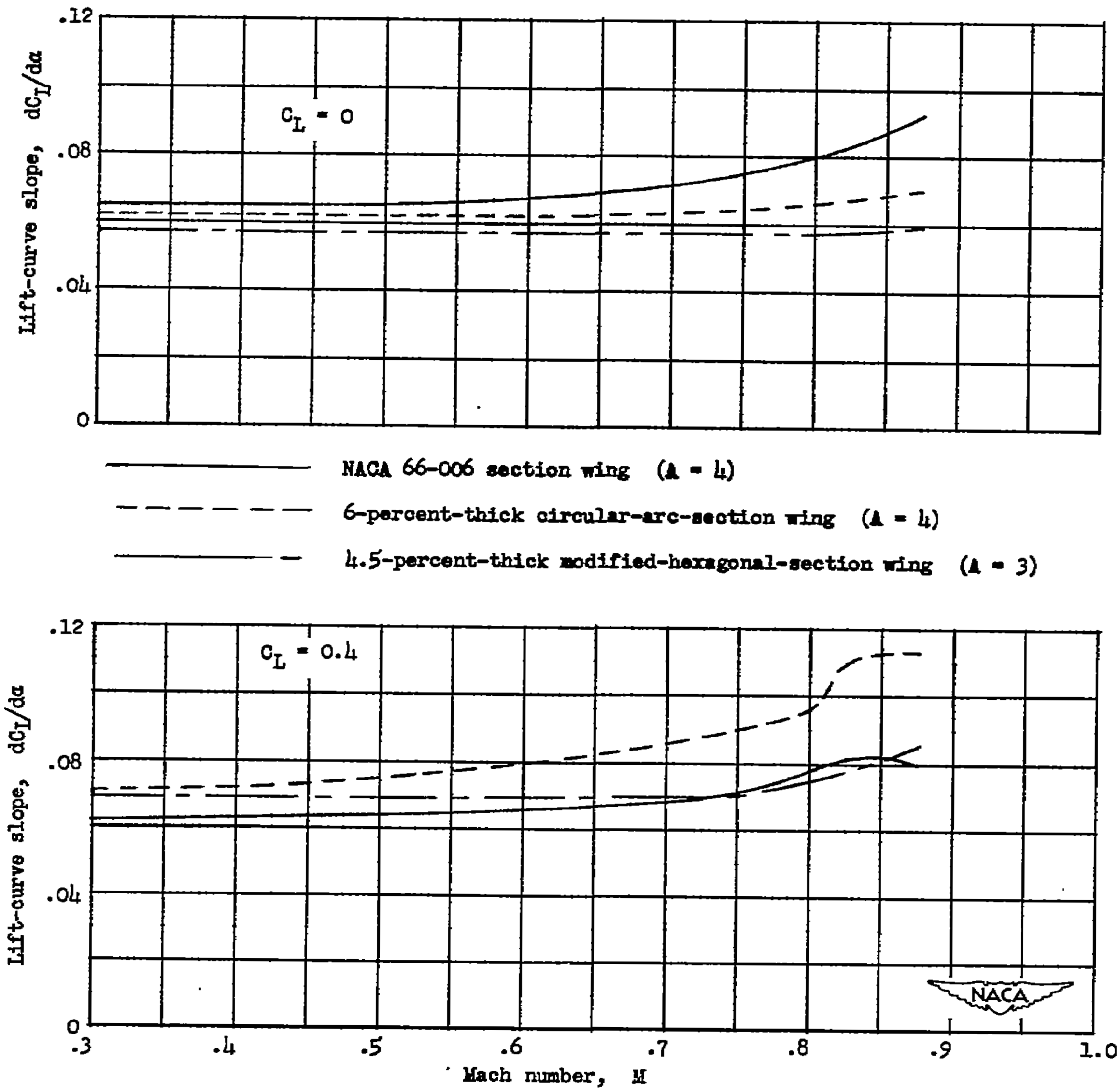


Figure 3.- Variation of lift-curve slope with Mach number.

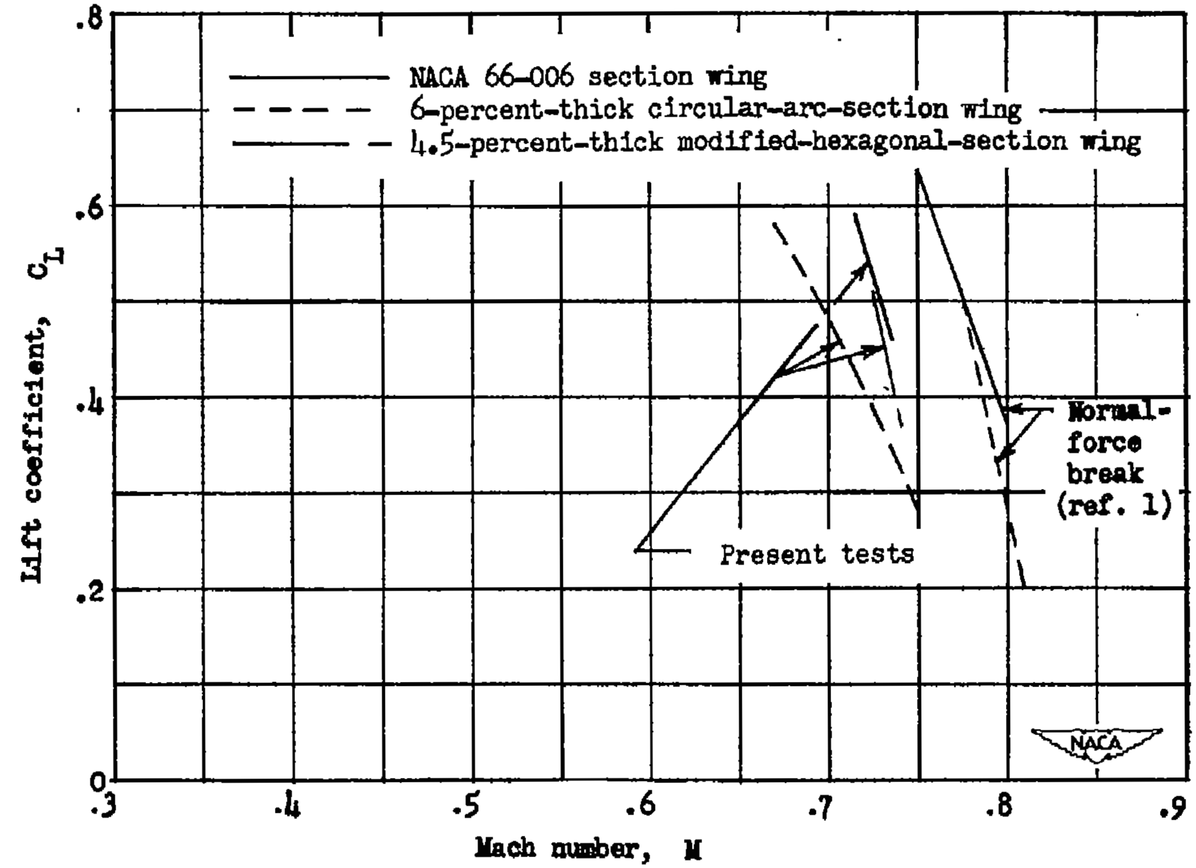
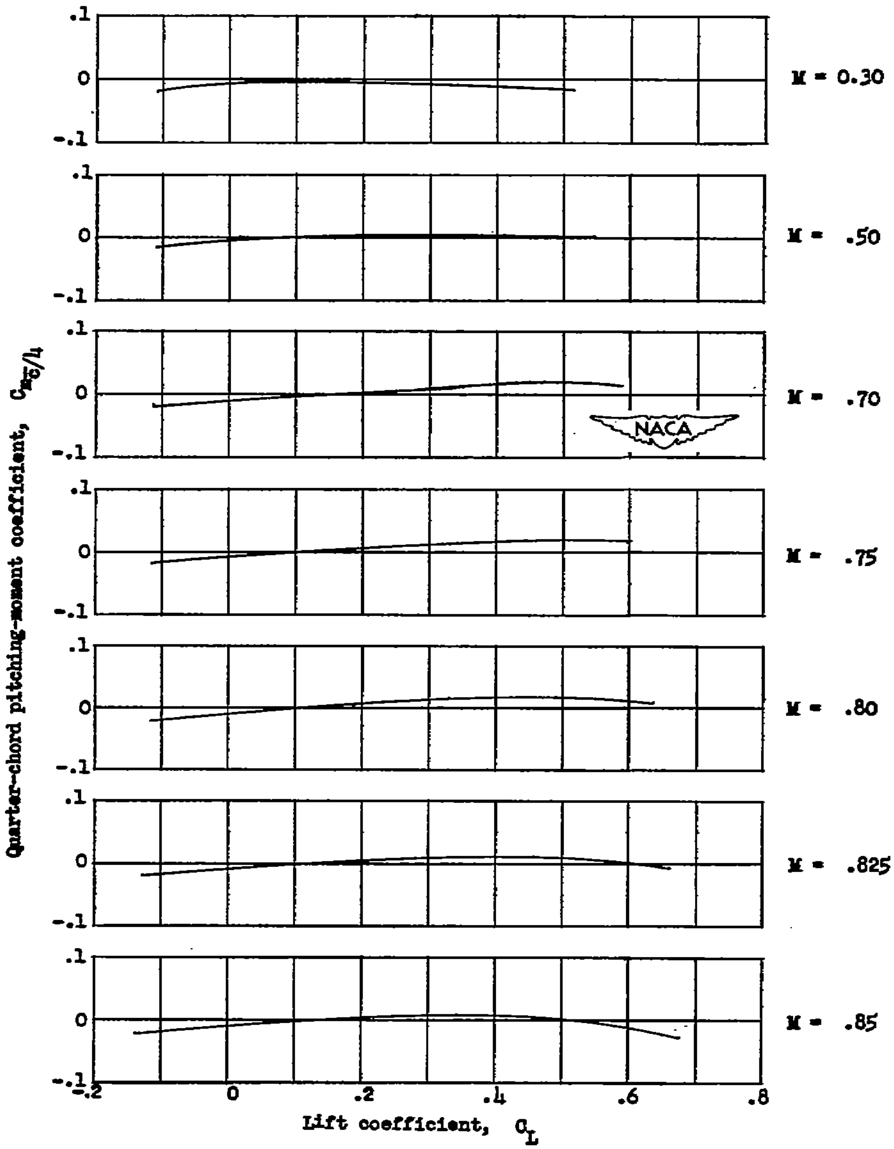
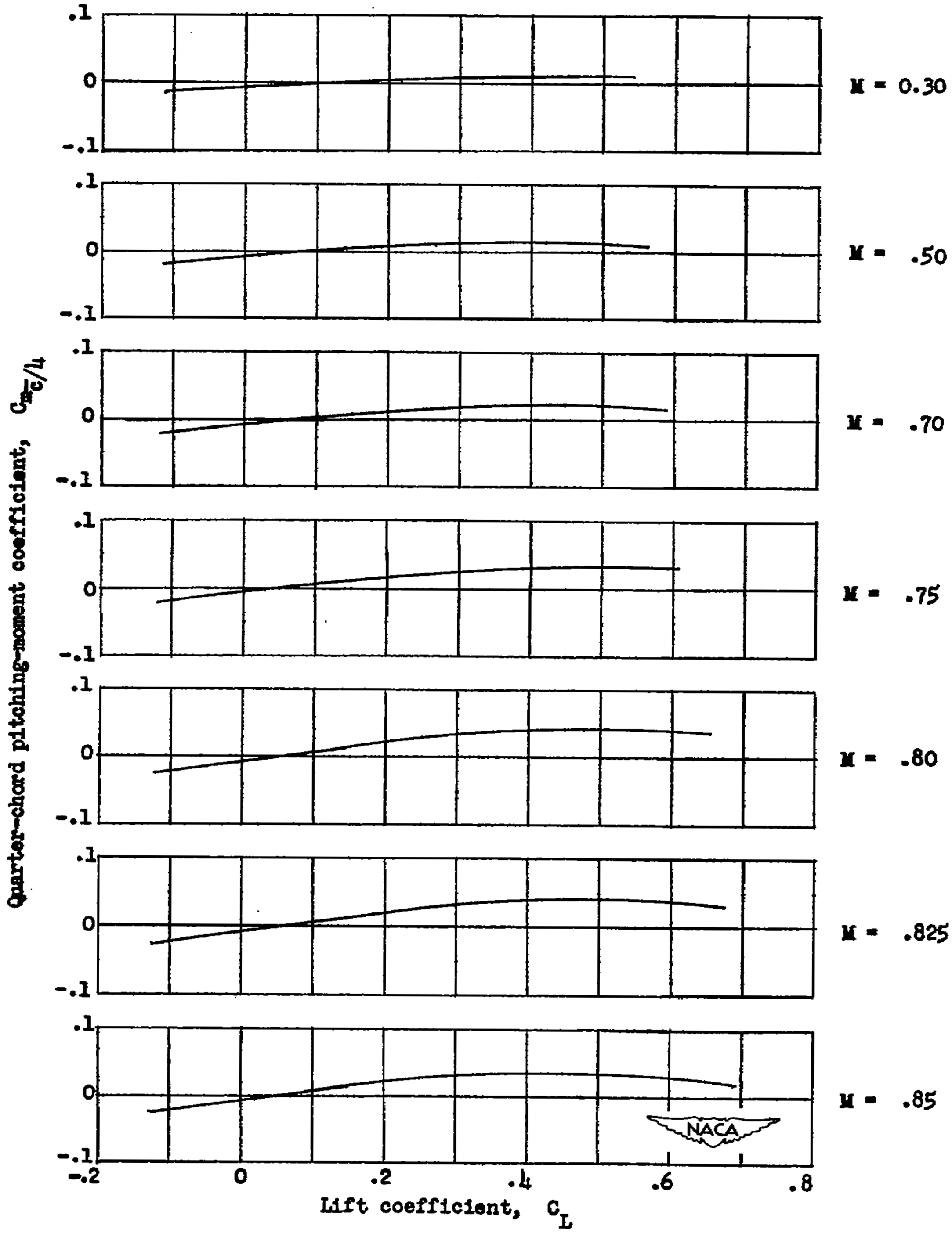


Figure 4.- Comparison of experimental unsteady-flow boundaries of three thin wings of low aspect ratio with two-dimensional normal-force break.



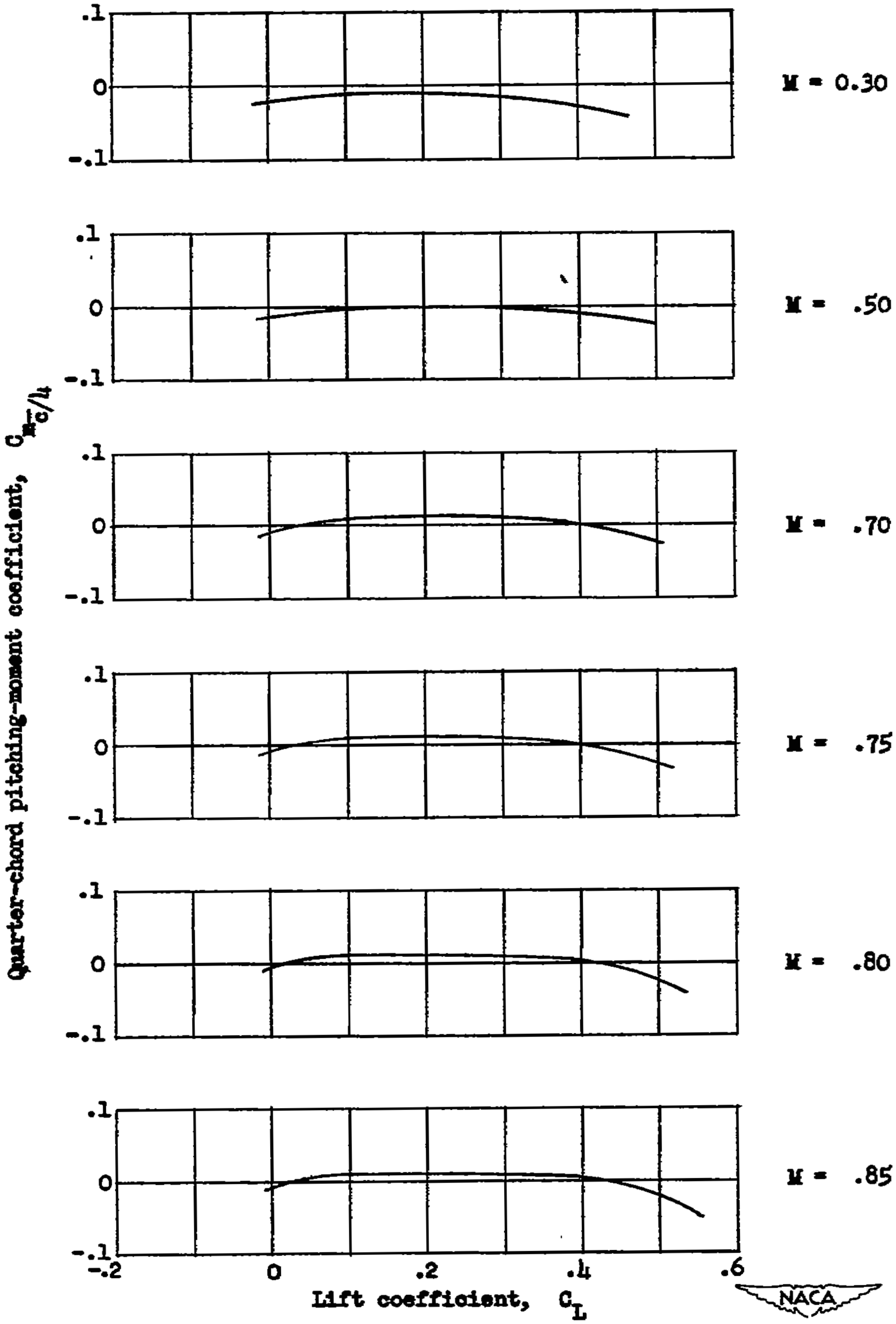
(a) NACA 66-006 section wing. A = 4.

Figure 5.- Variation of quarter-chord pitching-moment coefficient with lift coefficient at various Mach numbers.



(b) The 6-percent-thick circular-arc-section wing. A = 4.

Figure 5.- Continued.



(c) The 4.5-percent-thick modified-hexagonal-section wing. A = 3.

Figure 5.- Concluded.

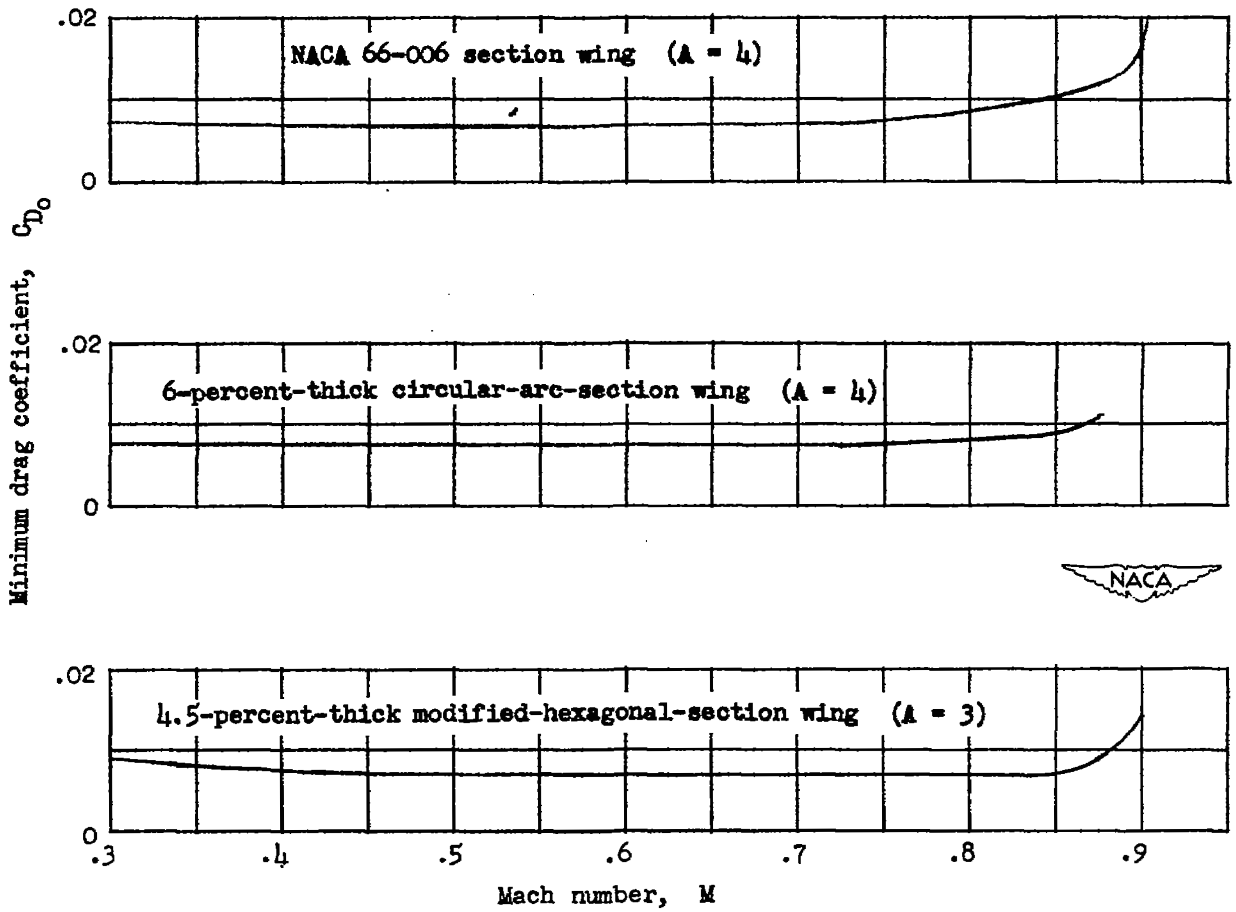


Figure 6.- Effect of Mach number on minimum drag coefficients of three thin wings of a moderately low aspect ratio.

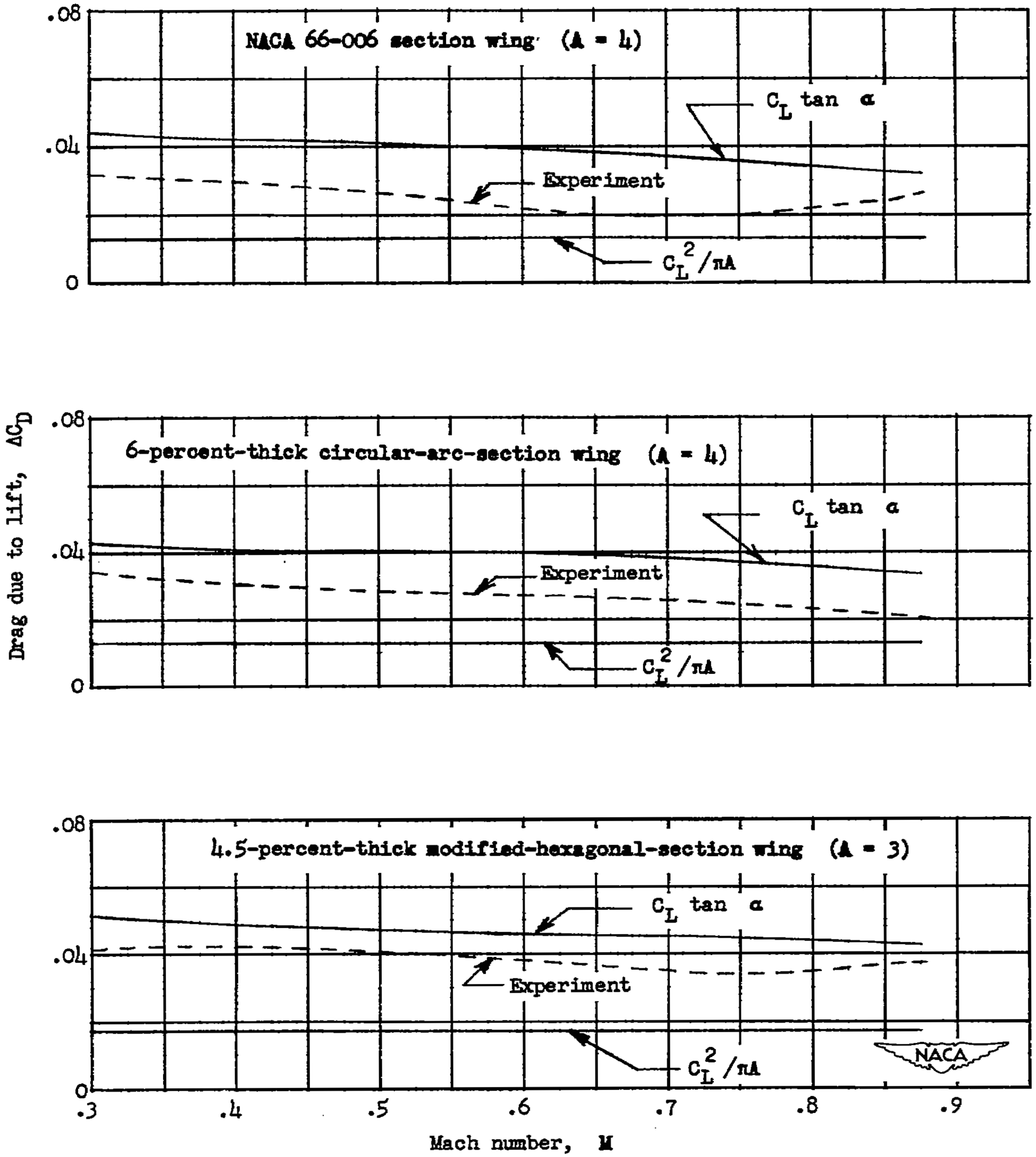


Figure 7.- Effect of Mach number on drag due to lift. $C_L = 0.4$.

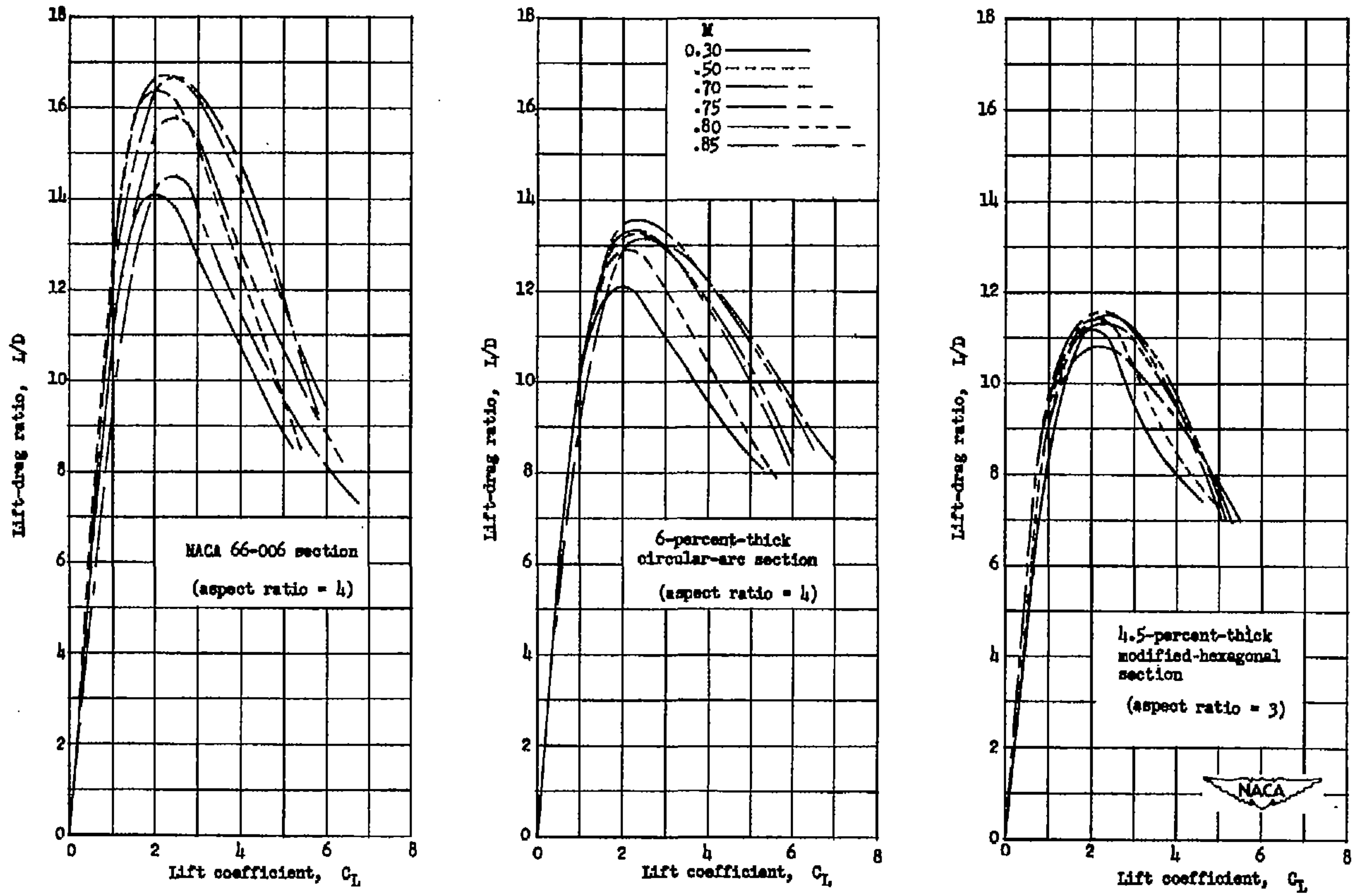


Figure 8.- Effect of Mach number on lift-drag ratio of three thin wings of moderately low aspect ratio.

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