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RESEARCH MEMORANDUM

LIFT AND MOMENT CHARACTERISTICS AT SUBSONIC MACH NUMBERS OF FOUR 10-PERCENT-THICK AIRFOIL SECTIONS OF VARYING TRAILING-EDGE THICKNESS

By James L. Summers and William A. Page

Ames Aeronautical Laboratory
Moffett Field, Calif.

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RESEARCH MEMORANDUM

LIFT AND MOMENT CHARACTERISTICS AT SUBSONIC MACH NUMBERS OF FOUR 10-PERCENT-THICK AIRFOIL SECTIONS OF VARYING TRAILING-EDGE THICKNESS

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SUMMARY

The results of a wind-tunnel investigation from 0.3 to approximately 0.9 Mach number of the lift and moment characteristics of four 10-percent-thick circular-arc airfoil sections are presented. The thickness at the trailing edge was varied from 0 to 100 percent of the maximum thickness. The Reynolds number of the investigation varied with Mach number within the limits of 1×10^6 to 2×10^6.

Increases in the trailing-edge thickness resulted in increases in maximum lift coefficient and lift-curve slope at all Mach numbers, and also in increases in lift-divergence Mach number at all lift coefficients. As the trailing-edge thickness was increased, proportionately more lift was carried over the rear portion of the airfoil sections with an accompanying increase in the slopes of the pitching-moment curves. These improvements were ascribed to a reduced region of decelerated flow over the aft portions of the airfoil sections as the trailing edge was thickened, with a consequent reduction in the area of separation and in the effects of compressibility.

Strongly developed Kármán vortex streets were observed in the wakes of the sections with appreciable trailing-edge thickness. Associated with the vortex-street development were rapid lift fluctuations and large drag coefficients at low lift coefficients. The lift fluctuations are considered to be of possible importance with respect to airplane tail buffeting and trailing-edge control-surface flutter. The attachment of a thin splitter plate to the airfoil trailing edge was found to be an effective means for preventing the development of the vortex street and its adverse effects.

INTRODUCTION

It has been indicated by Chapman in reference 1 that airfoil sections having blunt trailing edges possess certain advantages over sharp trailing-edge sections at supersonic Mach numbers. The analysis shows that, for given structural strength, higher lift-curve slopes and lower drags can be expected for the blunt trailing-edge sections than for the sharp trailing-edge sections. In references 2 and 3, it was shown that airfoil sections with thick trailing edges have favorable characteristics from the standpoint of trailing-edge control-surface effectiveness at transonic Mach numbers. Favorable lift characteristics in this respect were also observed at high subsonic Mach numbers in reference 4 for airfoil sections with maximum thickness at or near the trailing edge.

The present investigation was undertaken to provide information on the aerodynamic characteristics of blunt trailing-edge sections at Mach numbers up to 0.9. The trailing-edge thickness was varied from 0 to 100 percent of the maximum thickness. All airfoil sections were of circular-arc profile and 10 percent thick.

NOTATION

a_o section lift-curve slope at zero section lift coefficient, per degree

c airfoil chord

c_l section lift coefficient

$c_{l_{max}}$ maximum section lift coefficient

$c_{m_{c/4}}$ section pitching-moment coefficient about the quarter-chord point

M free-stream Mach number

M_l lift-divergence Mach number

α_o section angle of attack, degrees

$\frac{h}{t}$ ratio of trailing-edge thickness to maximum thickness

APPARATUS AND TESTS

The investigation was conducted in the Ames 1- by 3-1/2-foot high-speed wind tunnel which is a two-dimensional-flow, low-turbulence, closed-throat wind tunnel.

The airfoil models, constructed of aluminum alloy, were of 6-inch chord and entirely spanned the short dimension of the wind-tunnel test section. Contoured sponge-rubber gaskets compressed between the model ends and the tunnel walls were used to prevent end leakage. The four airfoil sections, illustrated in figure 1, were of circular-arc profile, 10 percent thick, with trailing-edge thicknesses of 0, 25, 50, and 100 percent of the maximum thickness.

Measurements of lift and quarter-chord pitching moment were made at angles of attack varying from -1° to 10°, this latter value being sufficient to encompass the lift stall at most test Mach numbers. The Mach numbers ranged from 0.3 to approximately 0.9 with corresponding Reynolds numbers varying from 1×10^{6} to 2×10^{6}.

Airfoil lift and pitching moments were evaluated, using a method similar to that described in reference 5, by integration of the pressure reactions of the airfoil forces on the tunnel floor and ceiling. All the data of the present report have been corrected for wind-tunnel-wall interference by the method of reference 6.

RESULTS AND DISCUSSION

Section lift and quarter-chord pitching-moment coefficients of the airfoils (identified in terms of the ratio of trailing-edge thickness to maximum thickness) are presented as functions of Mach number at constant angles of attack in figures 2 and 3, respectively. The dashed lines in the figures at the higher Mach numbers indicate the region of possible influence of wind-tunnel choking effects on the results.

The variation of lift coefficient with angle of attack is presented in figure 4 for the various profiles. The airfoil section with zero trailing-edge thickness (fig. 4(a)) exhibits low values of lift-curve slope in the vicinity of 0° angle of attack. For example, at 0.3 Mach number the lift-curve slope is approximately one-half the value at 4° angle of attack.[1] Schlieren observations indicated the flow to be

[1]This phenomenon has been observed to occur on at least one other section having a large trailing-edge angle; the NACA 0035 section exhibits this characteristic at a Reynolds number of 3.2×10^{6}. (See reference 7.) The trailing-edge angles of the NACA 0035 section and of the section of the present investigation are 43.2° and 22.8°, respectively. The effects of separation would be expected to be less severe for the latter airfoil section because of the smaller trailing-edge angle. However, because of differences in both profile and Reynolds number the opposite effect is noted.

separated on both upper and lower surfaces at 0° angle of attack. It is believed that, as the angle of attack increased, the point of separation moved rearward on the lower surface, placing the center of the separated wake above the trailing edge of the airfoil. This resulted in reductions of the effective angle of attack and the section lift-curve slope from the values that would pertain if the wake were unseparated. At higher angles of attack, approximately 2° and greater, the separation point on the lower surface remained fixed at the trailing edge and, as a result, the slopes of the lift curves are indicated to be greater than at zero angle of attack.

The effects of trailing-edge thickness on the respective variations with Mach number of maximum lift coefficient and lift-curve slope are shown in figures 5 and 6. Figure 7 illustrates the lift-divergence Mach number as a function of lift coefficient. (Lift-divergence Mach number, determined from the curves of figure 2, is defined as that Mach number at which the first point of inflection occurs.) It is noted that, for increasing trailing-edge thickness, the values of maximum lift coefficient and lift-curve slope are increased at all Mach numbers and the lift-divergence Mach number is increased at all lift coefficients. The improvements in the lift characteristics are ascribed to the decreasing magnitude and extent of adverse pressure gradient over the aft portion of the airfoil sections with increasing trailing-edge thickness. As a result, the detrimental viscous and compressibility effects over the trailing-edge region are reduced. For this reason, increasing effectiveness of trailing-edge control surfaces, as is observed from the results of references 2 and 3 at high subsonic Mach numbers, would also be expected to accompany increases in the trailing-edge thickness. Also for this reason, it is believed that, although the magnitude of the improvements in characteristics observed to accompany increases in trailing-edge thickness may be exaggerated because of the poor characteristics of the basic circular-arc section, decided improvements will be realized from thickening the trailing edges of more conventional sections.

It is further noted from figure 2(d) that, for the airfoil section having maximum thickness at the trailing edge, lift divergence does not occur within the Mach number limits of this investigation. Experimental data (see reference 1) indicate the lift-curve slope of this section at 1.5 Mach number to be about 30 percent of the value shown in figure 6 for 0.85 Mach number. Since the lift-curve slope must decrease from the subsonic value to the supersonic value, it should be expected that divergence must occur for this section somewhere between a Mach number of 0.85 and a Mach number of 1.5.

Quarter-chord pitching-moment coefficient plotted against lift coefficient for several Mach numbers is shown in figure 8 for the various airfoil profiles. For the airfoil section having zero trailing-edge thickness, positive slopes of the pitching-moment-coefficient curves are observed at small values of lift coefficient (fig. 8(a)). The

magnitudes of the slopes of the curves are such as to indicate that, for Mach numbers of 0.5 and greater, the center of lifting pressures was ahead of the leading edge at small lift coefficients. This probably results from the negative additional lift produced over the rear portion of the section by the movement of the lower-surface separation point toward the trailing edge with increase in angle of attack. More negative slopes of the moment curves at all values of lift coefficient are observed from figure 8 to accompany thickening of the trailing edge. This result should be expected since proportionately greater lift is carried over the rear portions of the airfoil sections as the maximum thickness position moves rearward. (See reference 5.)

The improvements in lift characteristics were not achieved without some accompanying undesirable characteristics of the thick trailing-edge sections. Schlieren observations indicated that a strongly developed Kármán vortex street and an accompanying system of pressure waves, both indicative of large energy losses, were present in the flow fields about the airfoil models. Typical schlieren photographs illustrating the flow fields about the several profiles are presented in figure 9. From this figure it is noted that there is nothing unusual (with the exception of the aforesaid separation) about the flow over the section with the sharp trailing edge, but that a regular Kármán vortex street appears in the wake of the sections having finite thickness at the trailing edge. This phenomenon is most clearly evident in part (b) of this figure. Also noted in the figure is a system of pressure waves which originate at the points of discharge of the individual vortices from the trailing edge and are propagated upstream throughout the flow field. A close examination of the photograph reveals that the waves are emitted alternately from the upper and lower surfaces at the trailing edge with a frequency corresponding to that of the vortex discharge. It would appear, then, that the lift is fluctuating periodically, a factor of important significance with respect to airplane tail buffeting and trailing-edge control-surface flutter. Measurements made with a stroboscopic schlieren device indicated that the frequencies of the vortex streets varied directly with Mach number. At 0.65 Mach number, the measured frequencies were roughly 9500, 6500, and 3500 cycles per second for the airfoil sections having, respectively, values of 0.25, 0.50, and 1.00 for the ratio of trailing-edge thickness to maximum thickness. It was also observed that, qualitatively, the drag coefficients of the thick trailing-edge sections were high and increased with trailing-edge thickness as a result of the energy losses associated with the Kármán vortex street. For reasons to be discussed later, the drag coefficients of the thick trailing-edge sections, which were determined from wake-survey measurements, are considered quantitatively unreliable and, as a consequence, are not presented.

One means has been devised for preventing the development of the vortex street, namely, the attachment of a thin splitter plate to the trailing edge to prevent the interaction of the vortices generated at the upper and lower surfaces of the airfoil section. The effectiveness of this device is illustrated by the schlieren photographs of figure 10.

Part (a) of this figure is an illustration of the flow about the section having a trailing-edge thickness equal to one-half the maximum thickness before the attachment of the splitter plate. The unsteady flow characteristics in the wake and about the model are clearly evident. Part (b) illustrates the flow over the airfoil section under identical conditions with a splitter plate attached to the trailing edge. It is immediately apparent that the wake width has been greatly decreased and the pressure waves totally eliminated. A photograph of the empty test section with no flow is included in the figure (part (c)) to permit separation of the optical defects of the tunnel windows from the physical characteristics of the flow. The destruction of the strong vortex street ordinarily formed at the thick trailing edge would indicate also that the periodic fluctuations of the lift forces are no longer present. The drag coefficients of the airfoil with the splitter plate were determined to be approximately twice those for conventional, sharp, trailing-edge sections. For this airfoil section, the length of the splitter plate employed was approximately three times the trailing-edge thickness.

The results of limited measurements of the characteristics of the airfoil section with the splitter plate (determined only for the section having a trailing-edge thickness of one-half the maximum thickness) indicated that the addition of the splitter plate appeared to decrease the lift-divergence Mach numbers somewhat under those of the section without the plate. A comparison of the lift curves of the airfoil section with and without the splitter plate with those of the section having zero trailing-edge thickness is illustrated in figure 11 for Mach numbers of 0.65 and 0.85. For the section with the splitter plate, the lift coefficients were based on the total area, including that of the plate. Also included in figure 11 are the lift curves of a conventional, sharp, trailing-edge section, namely, the NACA 64-010 section. (These data were obtained from reference 8.) All the curves have been adjusted to pass through zero lift coefficient at zero angle of attack so that the comparison is not obscured by small errors in the angle-of-attack setting or by model asymmetry. It is observed from figure 11 that the section with the splitter plate displays somewhat inferior lift characteristics to to those of the same section without the splitter plate. The NACA 64-010 section, at 0.65 Mach number, having lift characteristics superior to the other sections, is perceived to have at 0.85 Mach number a lift curve not significantly different from that of the section with the splitter plate. The circular-arc section having zero trailing-edge thickness displays lift characteristics quite inferior to the other sections, particularly at 0.85 Mach number.

As was stated previously, the measured drags are considered unreliable. By virtue of the vorticity in the wakes of the blunt trailing-edge airfoils, the direction of the local flow in the plane of the wake total-pressure measurements was periodically variant with time. As a result, it was suspected that true total pressures were not being measured by the wake-survey rake and that the drag coefficients so

determined for the blunt trailing-edge airfoils were considerably in error. To assess the probable magnitude of this error, the drags of two circular cylinders of different diameters were determined from simultaneous measurements of the total-pressure defect in the wake, the pressure distribution about the cylinders, and the reactions of the drag forces on a strain-gage balance. The drag coefficients determined from the wake-survey measurements always were considerably greater than the corresponding values determined from the force and pressure-distribution measurements, the difference being roughly proportional to the cylinder diameters. It may be inferred from the cylinder investigation that the error in the wake-survey measurements of the drag of thick trailing-edge airfoils increases with trailing-edge thickness and, furthermore, that this error may be as great as 50 percent for the airfoil section having maximum thickness at the trailing edge.

CONCLUSIONS

The results of a wind-tunnel investigation from 0.3 to 0.9 Mach number of the lift and moment characteristics of four 10-percent-thick circular-arc airfoil sections having trailing-edge thicknesses ranging from 0 to 100 percent of the maximum thickness lead to the following conclusions:

1. Increases in the trailing-edge thickness result in increases in maximum lift coefficient and lift-curve slope at all Mach numbers, and also in increases in lift-divergence Mach number at all lift coefficients.

2. Increases in the trailing-edge thickness result in more negative slopes of the pitching-moment-coefficient curves.

3. Kármán vortex streets were present in the wakes of the sections with appreciable thickness at the trailing edge. As a consequence of the vortex-street development, the airfoil lift is subject to rapid fluctuations of possibly important significance with respect to airplane tail buffeting and trailing-edge control-surface flutter. Also, the drag coefficients of the thick trailing-edge sections were indicated to be high because of the energy losses associated with the vortex street. The vortex-street development and its adverse effects can be prevented by the attachment of a thin splitter plate to the airfoil trailing edge.

Ames Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Moffett Field, Calif.

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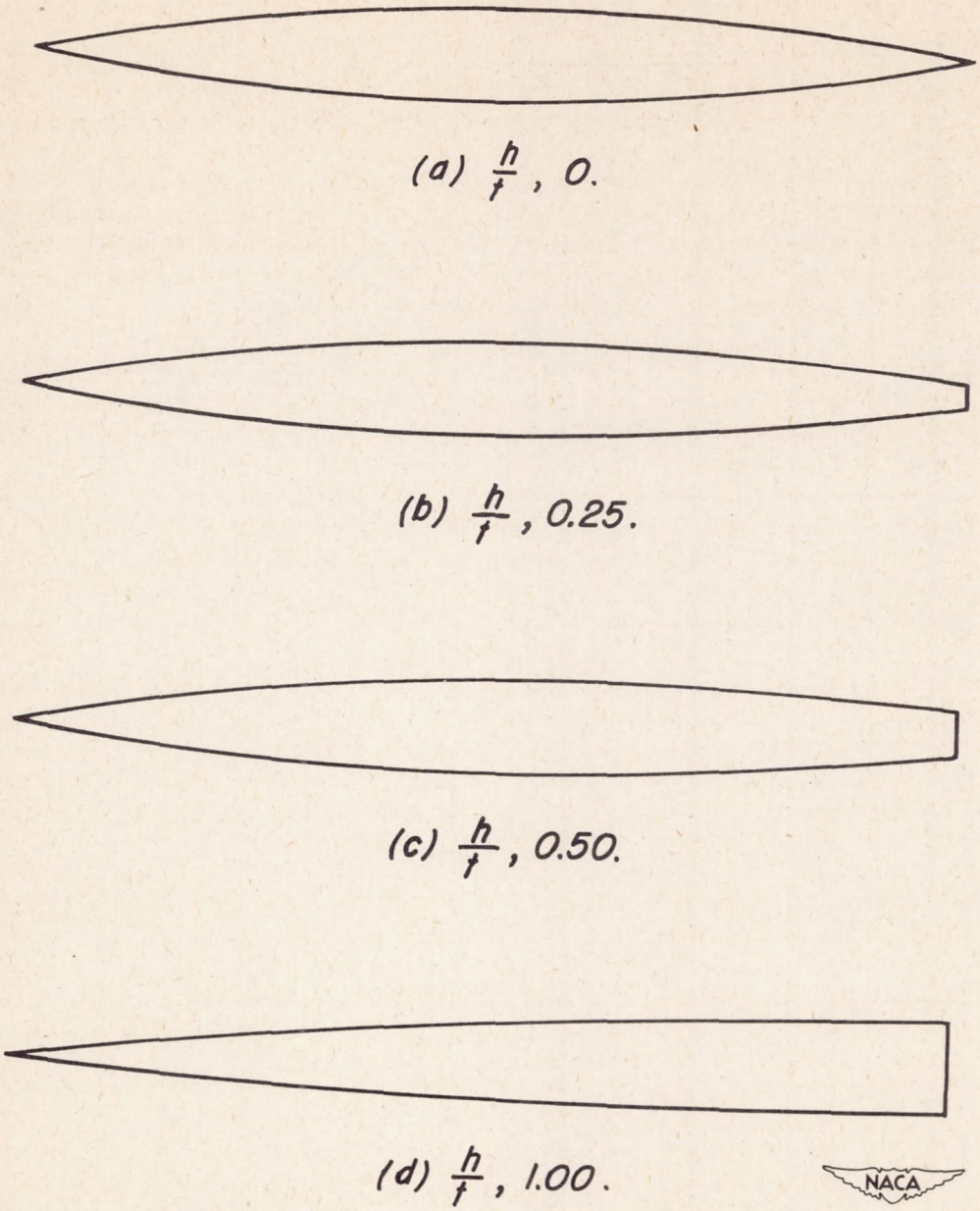
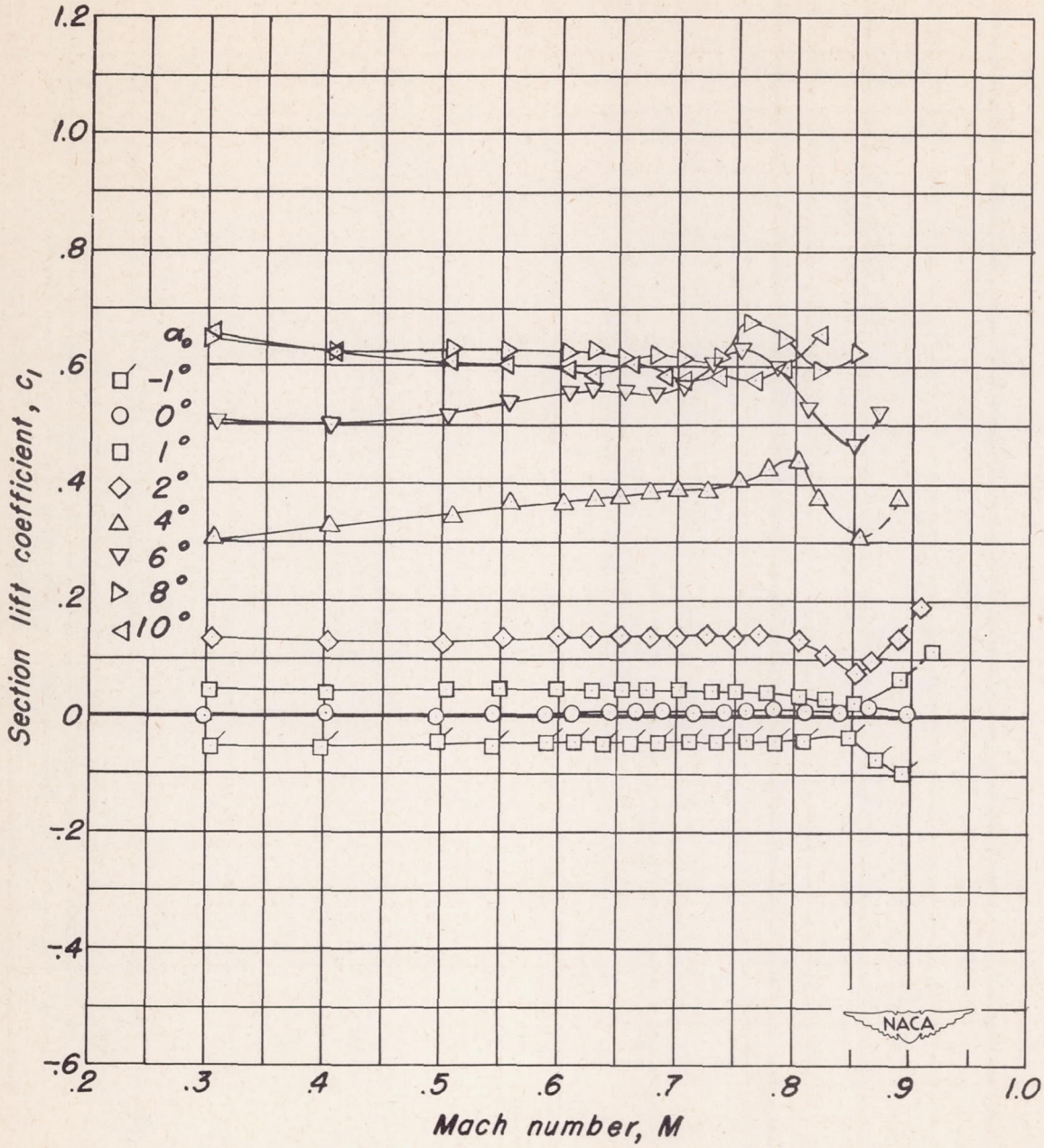
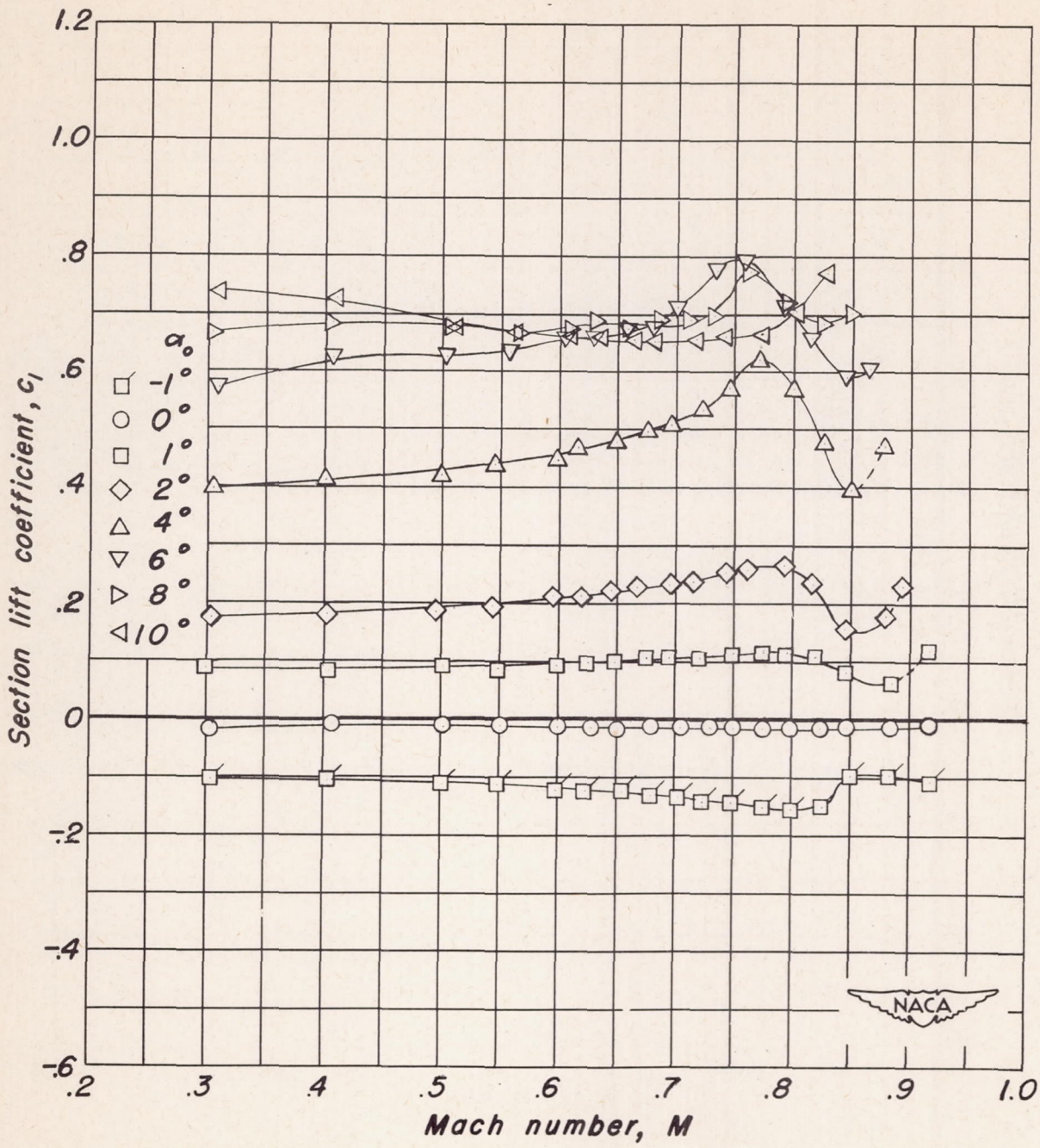


Figure 1.- Circular-arc airfoil profiles.



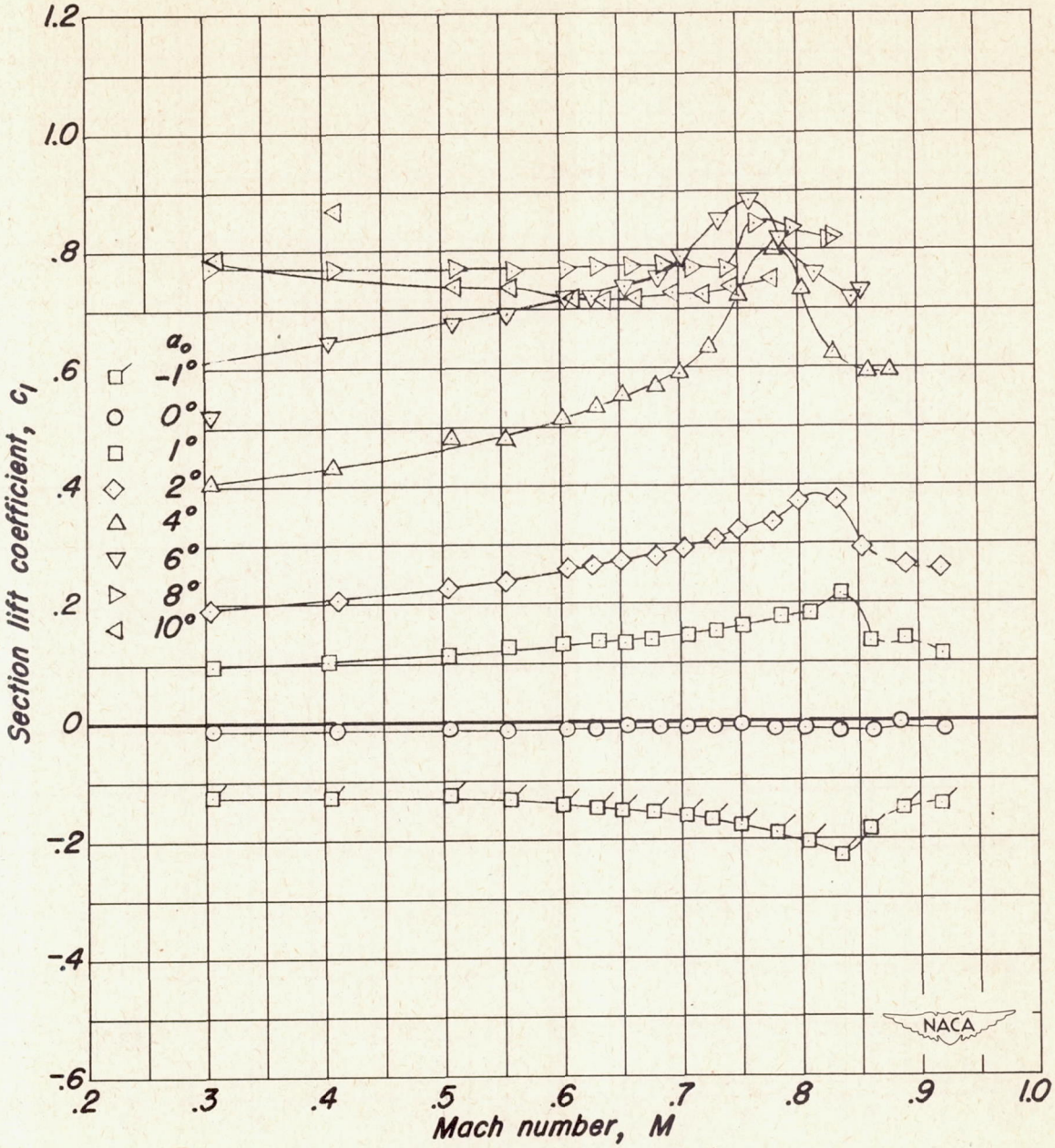
(a) $\frac{h}{t}$, 0.

Figure 2.–The variation of lift coefficient with Mach number at various angles of attack.



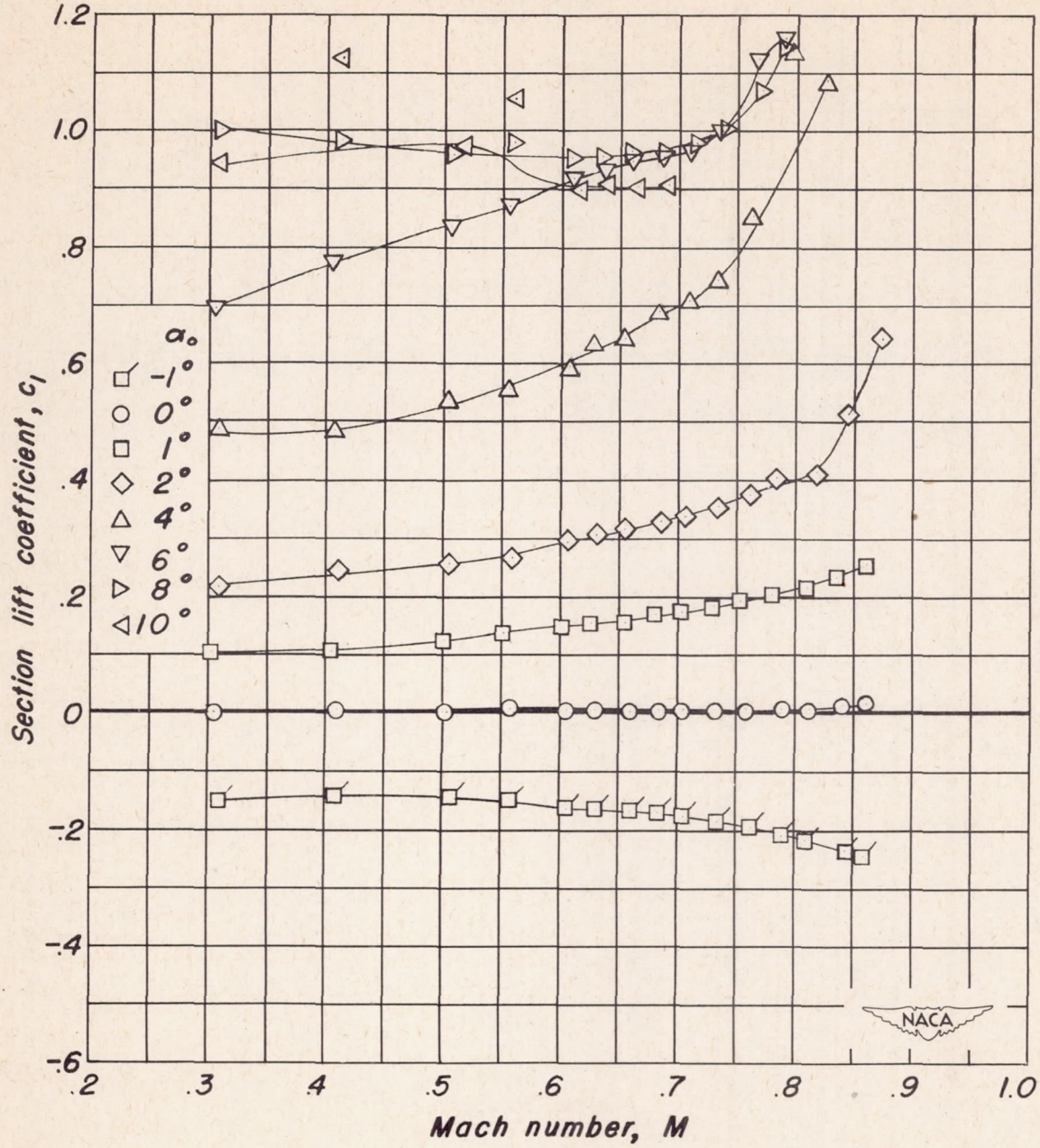
(b) $\frac{h}{t}$, 0.25.

Figure 2.- Continued.



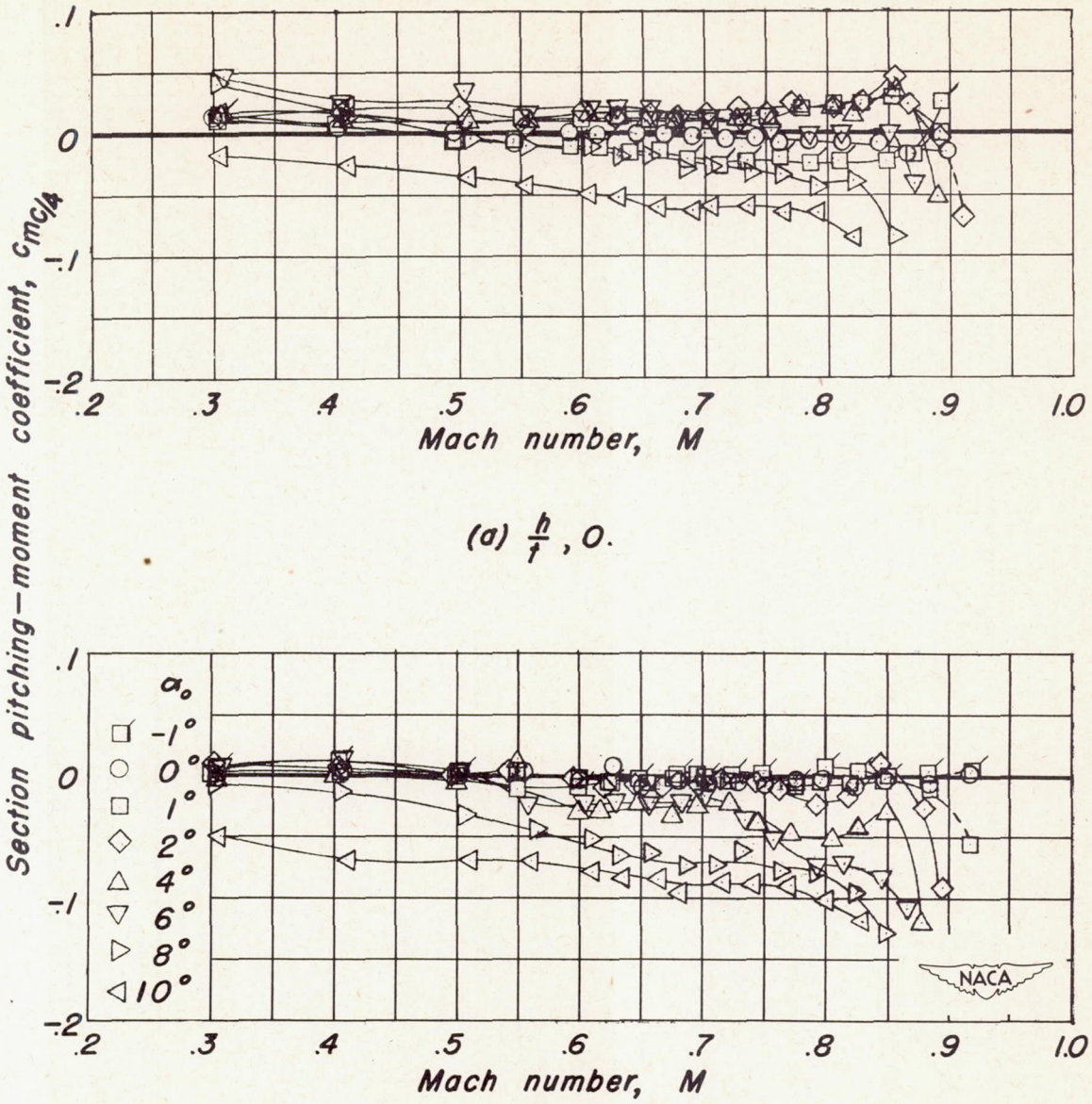
(c) $\frac{h}{t}$, 0.50.

Figure 2.-Continued.



(d) $\frac{h}{t}$, 1.00.

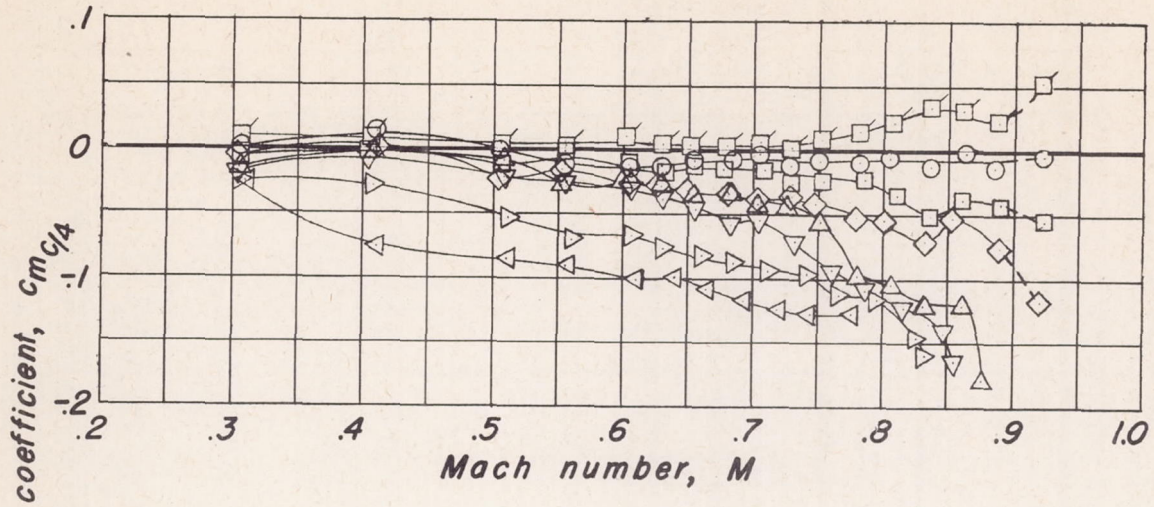
Figure 2.–Concluded.



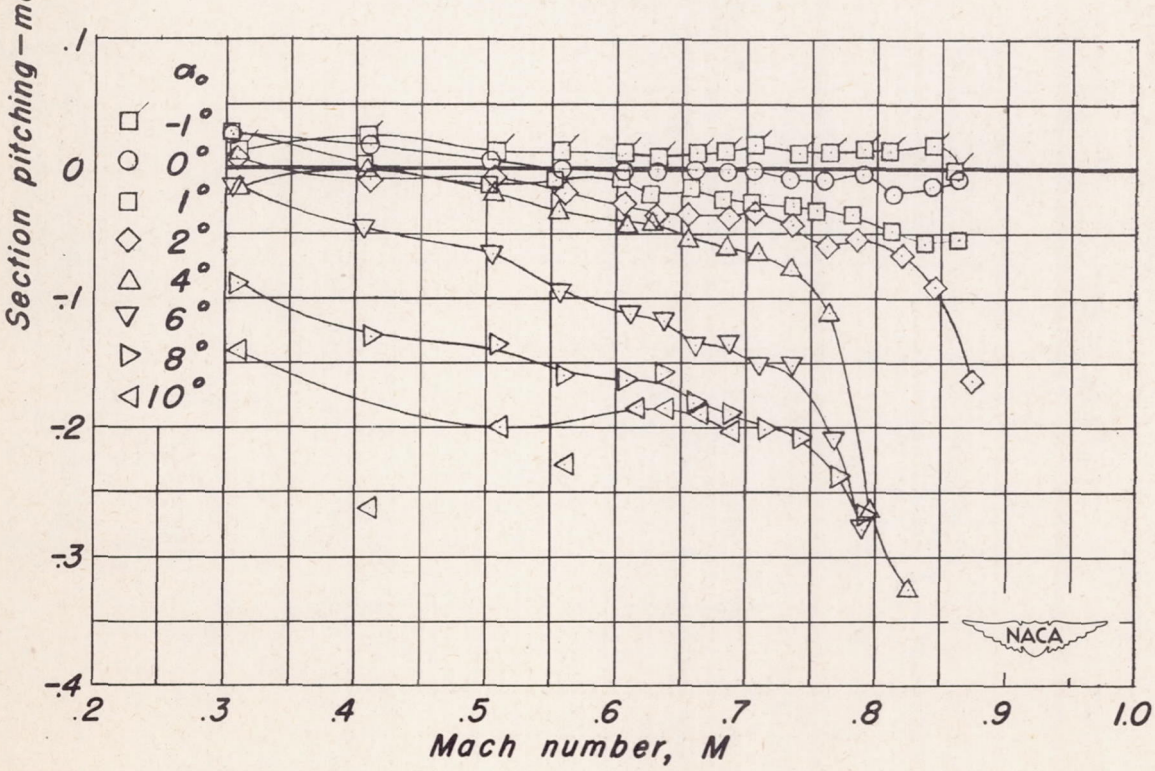
(a) $\frac{h}{t}$, 0.

(b) $\frac{h}{t}$, 0.25.

Figure 3.–The variation of quarter-chord pitching-moment coefficient with Mach number at various angles of attack.



(c) $\frac{h}{t}$, 0.50.



(d) $\frac{h}{t}$, 1.00.

Figure 3.-Concluded.

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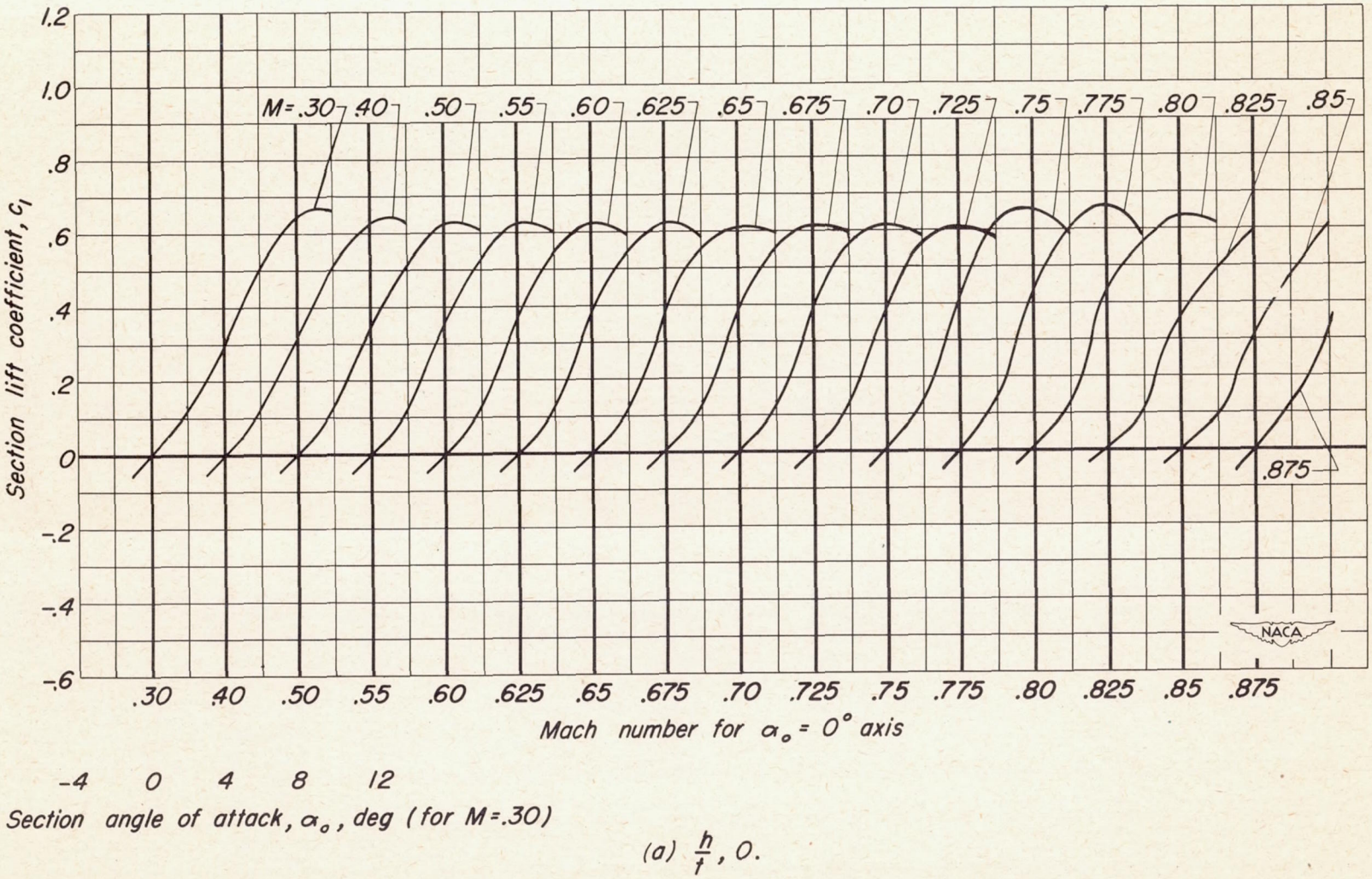

-4 0 4 8 12

Section angle of attack, α_o, deg (for M=.30)

(a) $\frac{h}{t}$, 0.

Figure 4.—Variation of lift coefficient with angle of attack at various Mach numbers.

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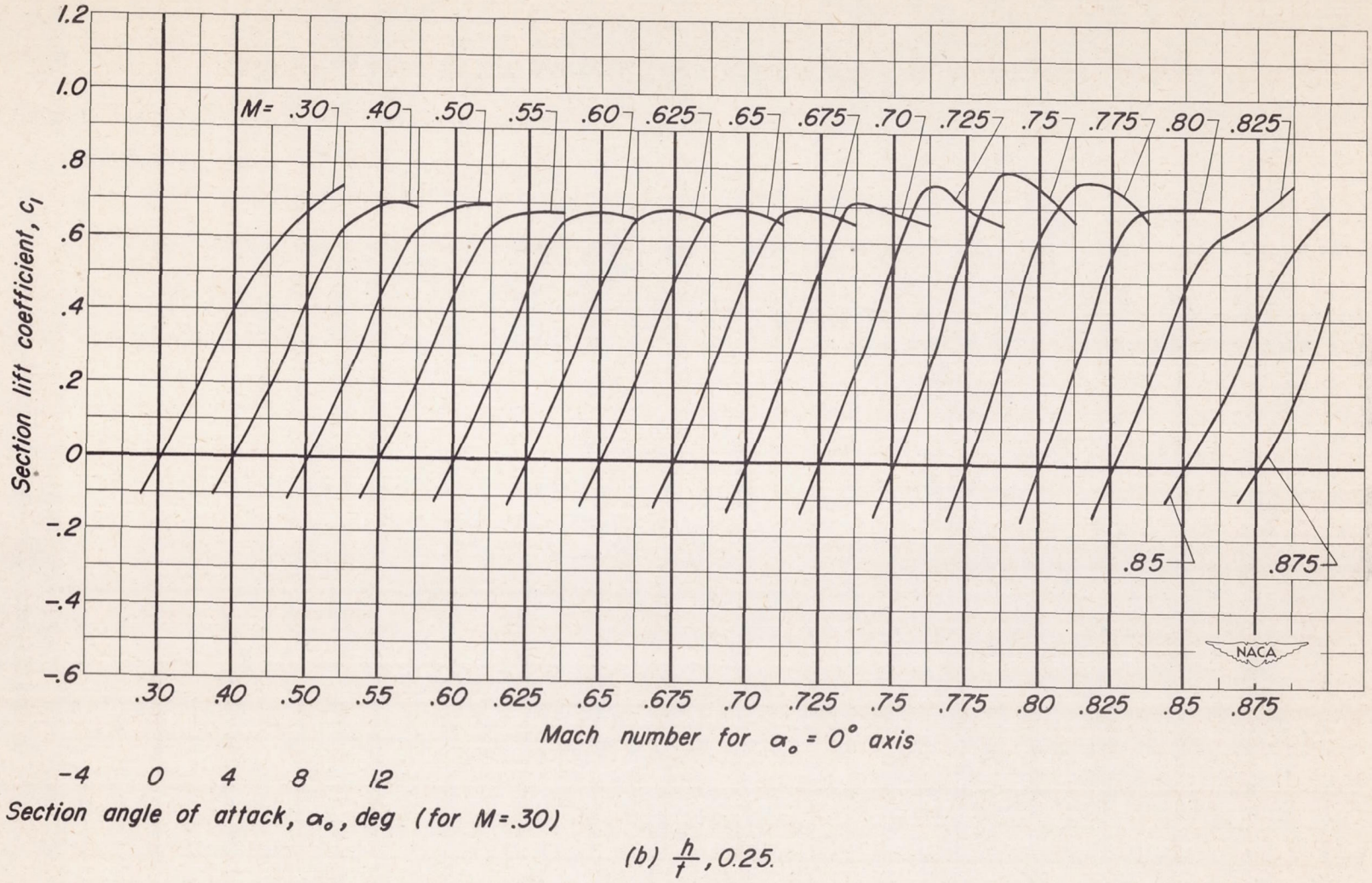

(b) $\frac{h}{t}$, 0.25.

Figure 4.–Continued.

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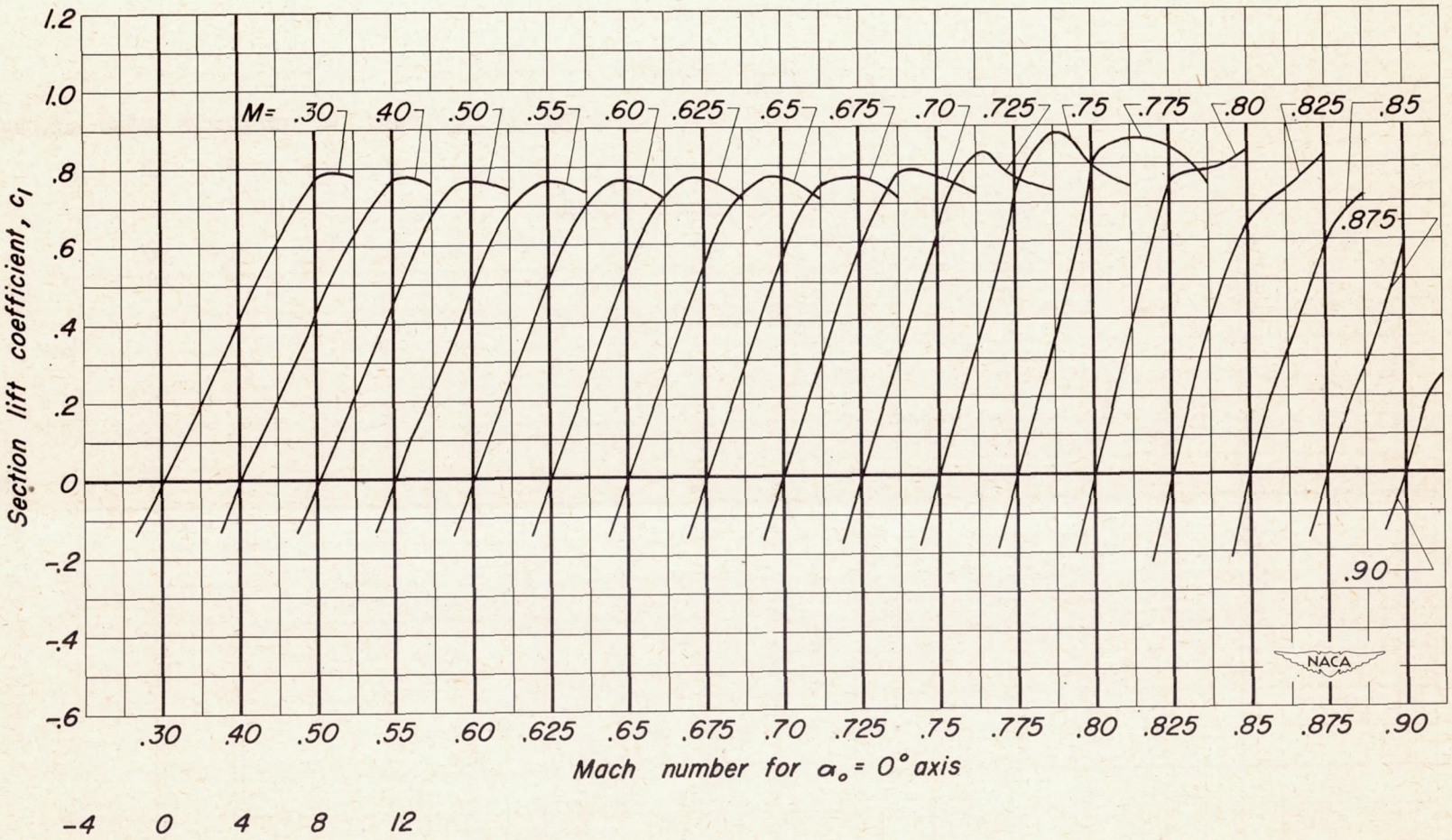

(c) $\frac{h}{t}$, 0.50.

Figure 4.—Continued.

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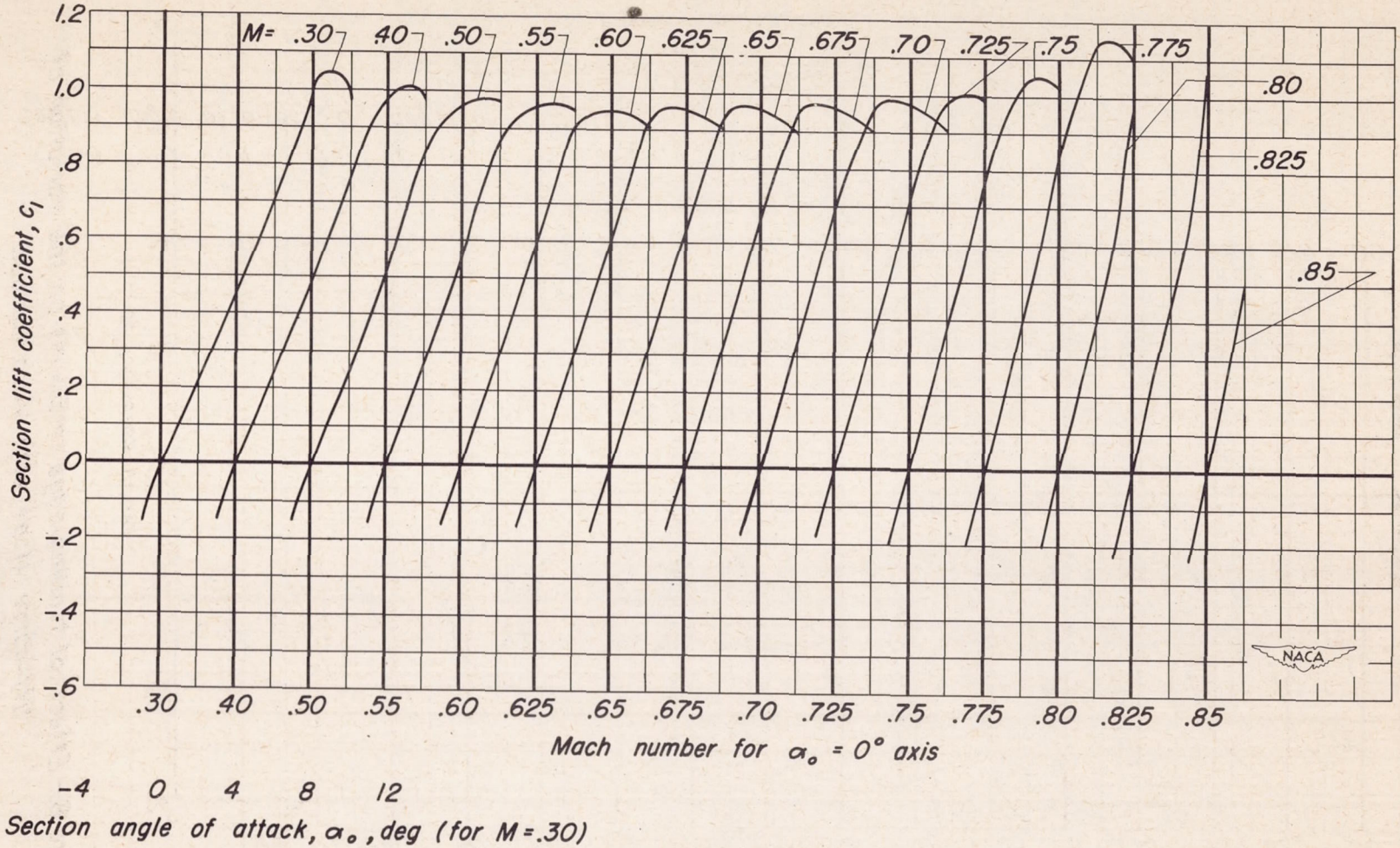

(d) $\frac{h}{t}$, 1.00.

Figure 4.—Concluded.

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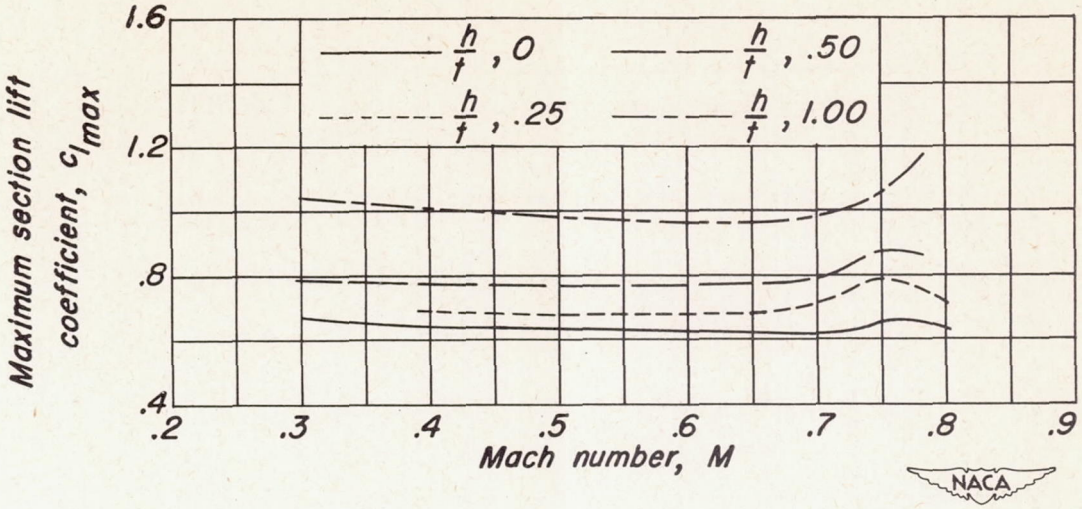

Figure 5.– Effect of trailing-edge thickness on the variation of maximum lift coefficient with Mach number.

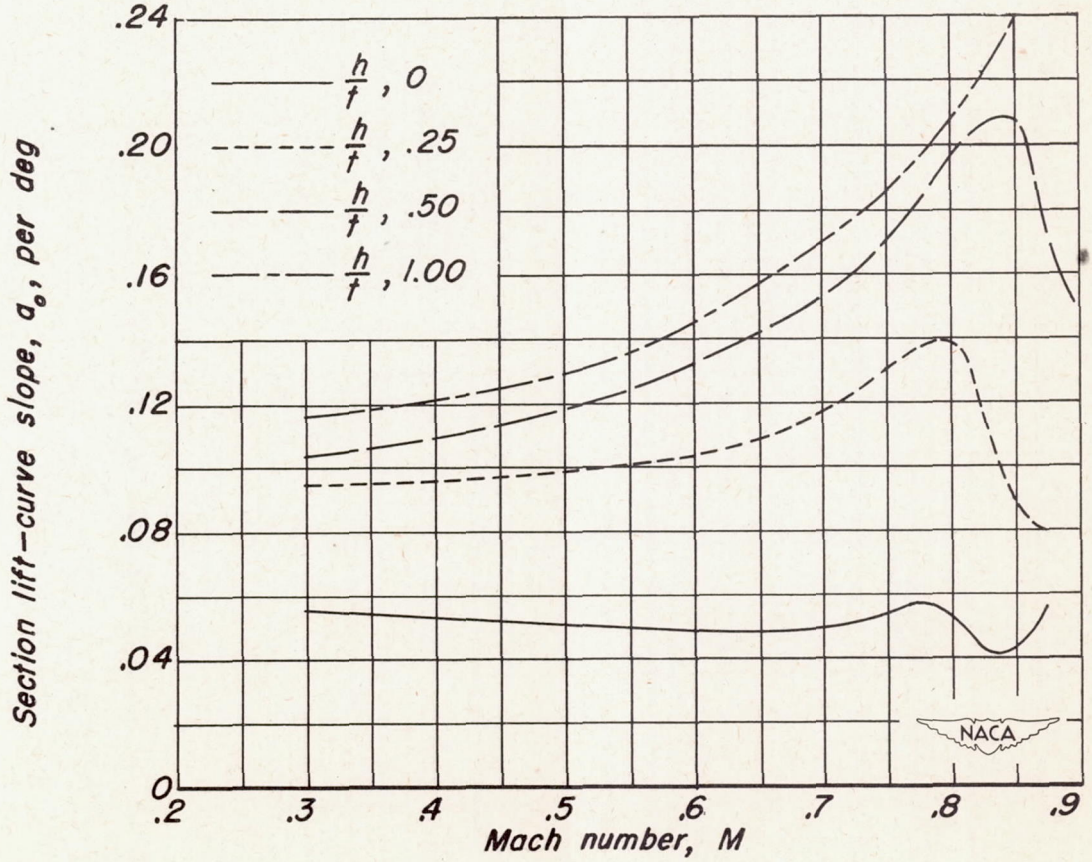

Figure 6.– Effect of trailing-edge thickness on the variation of lift-curve slope with Mach number.

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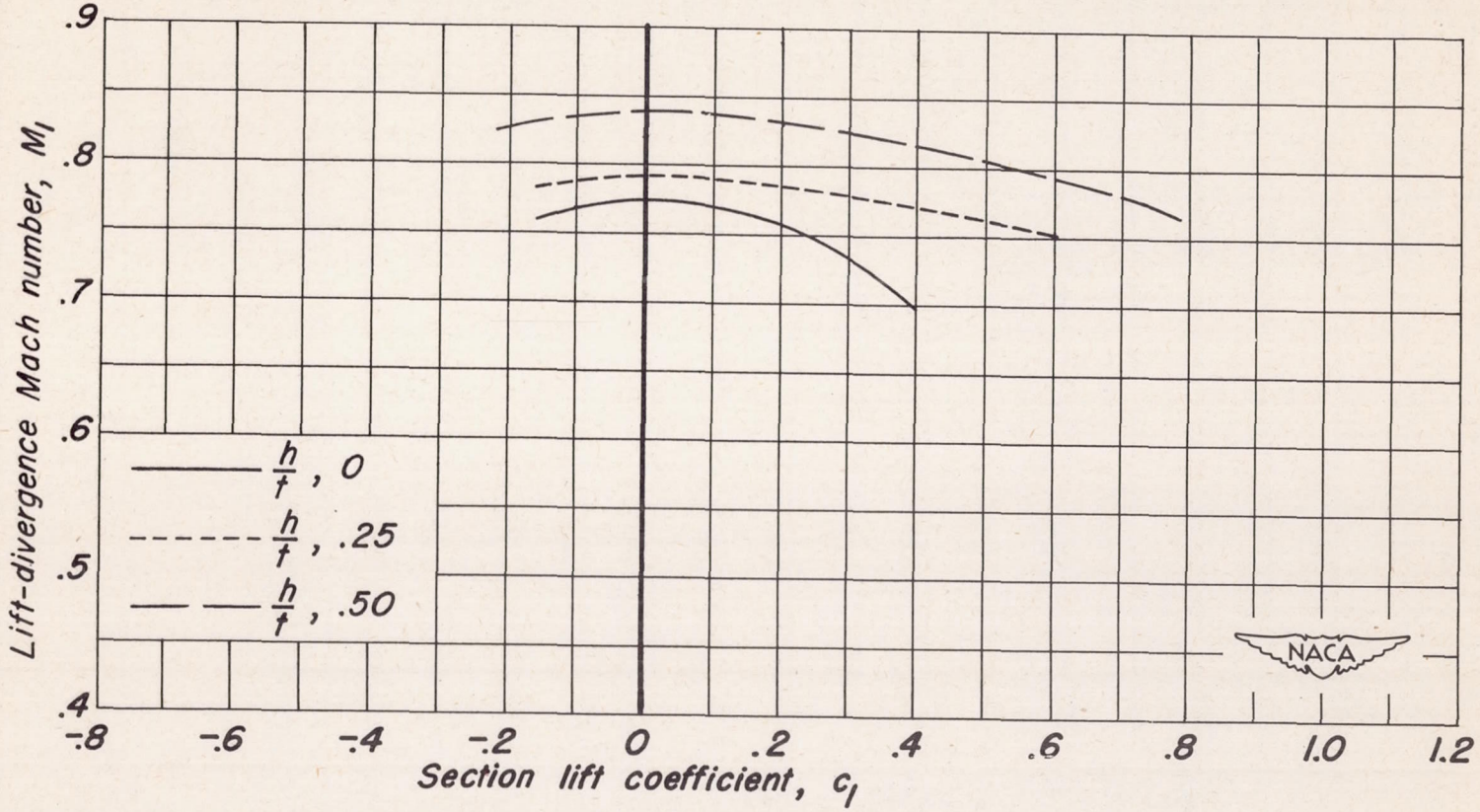

Figure 7.—Effect of trailing-edge thickness on the variation of lift-divergence Mach number with lift coefficient.

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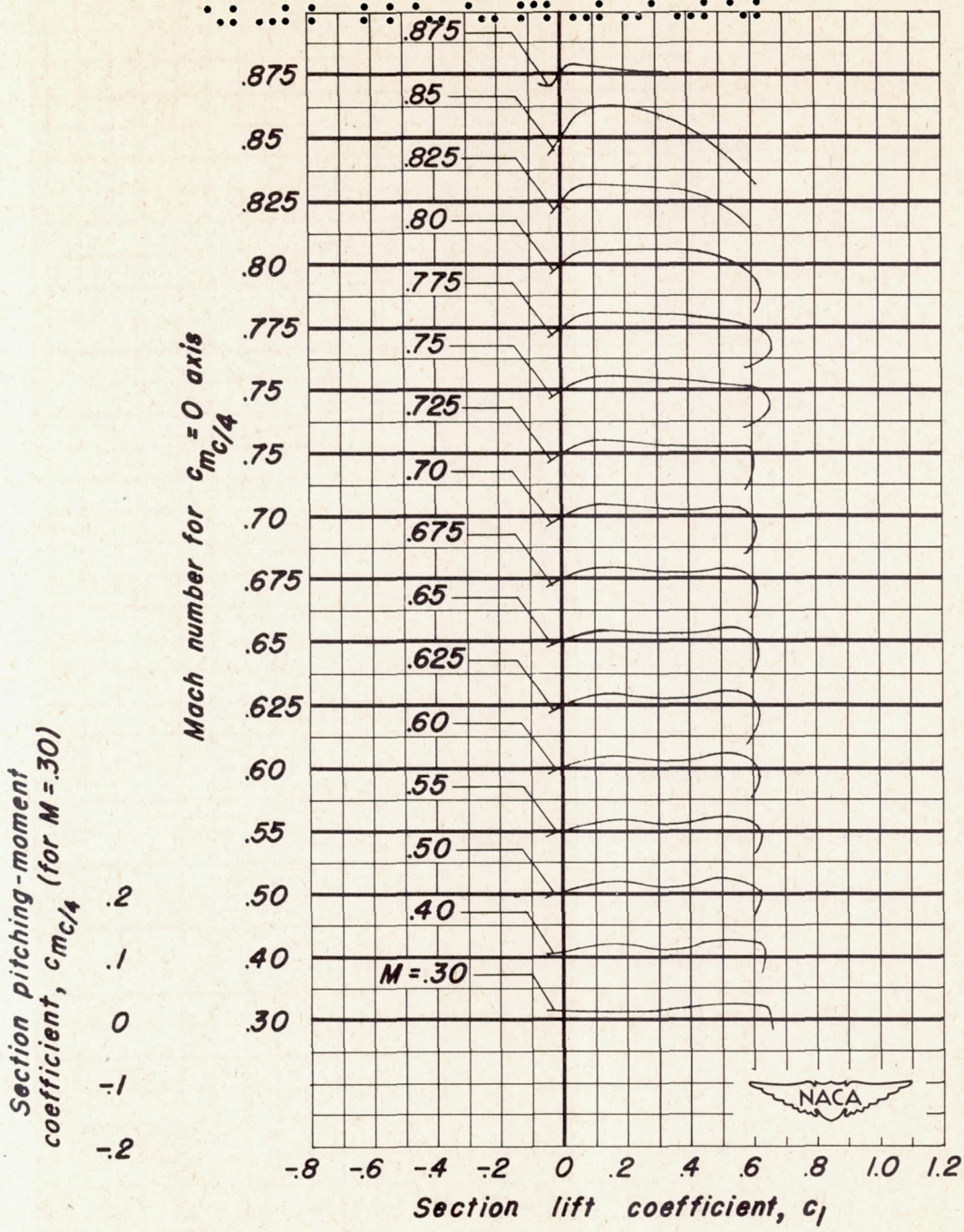

(a) $\frac{h}{t}$, 0.

Figure 8.–Variation of quarter-chord pitching-moment coefficient with lift coefficient at various Mach numbers.

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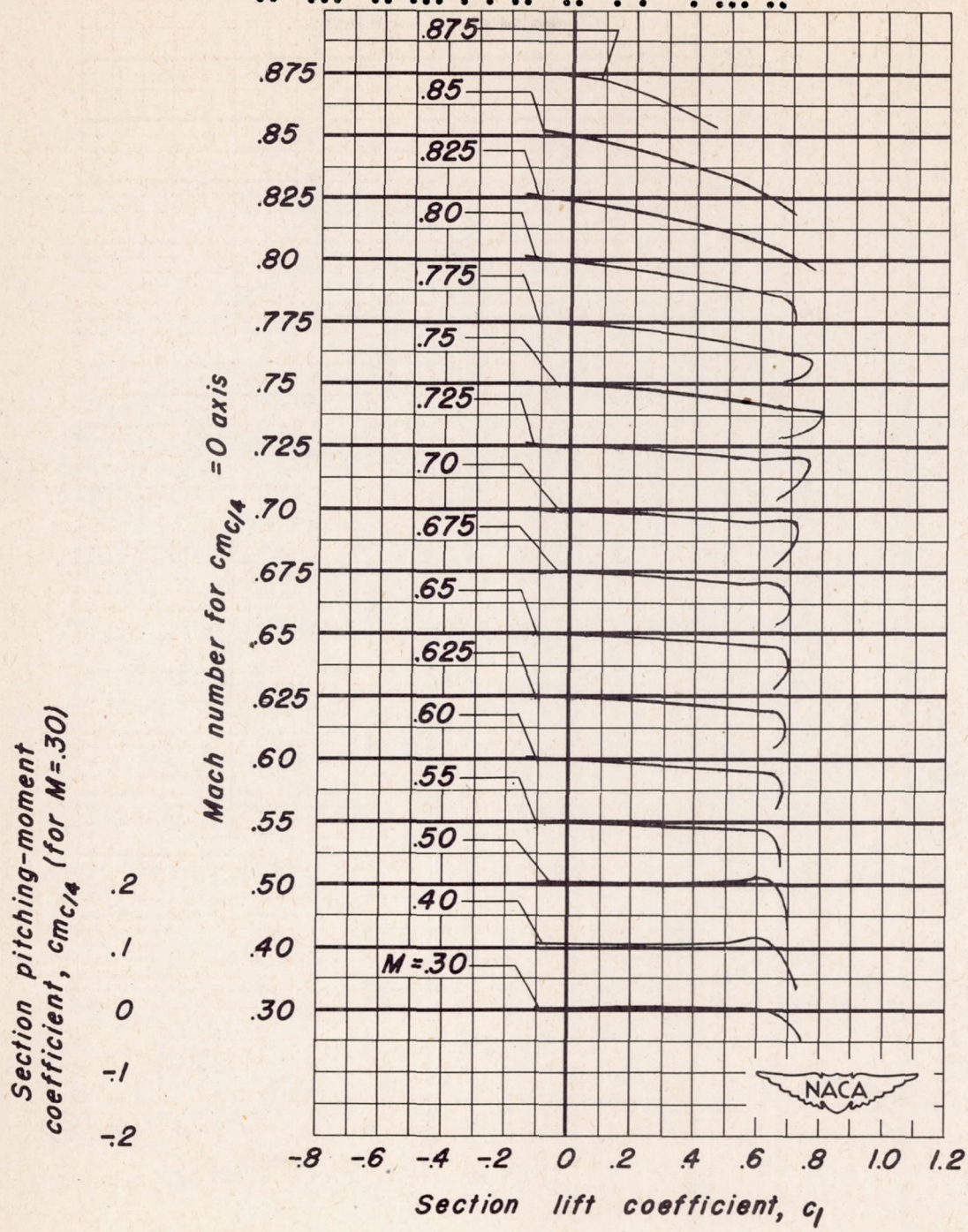
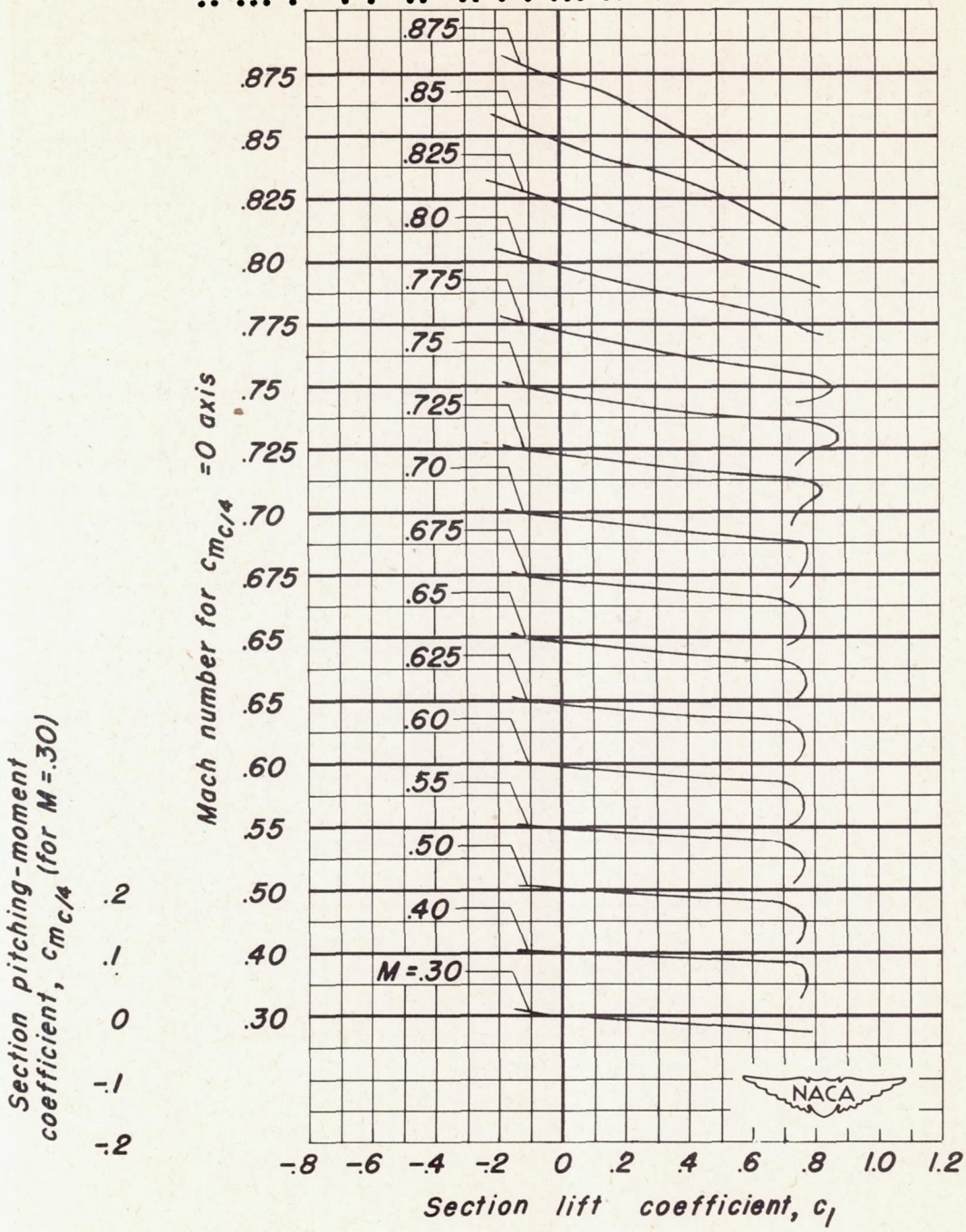


(b) $\frac{h}{t}$, 0.25.

Figure 8.–Continued.



(c) $\frac{h}{t}$, 0.50.

Figure 8.–Continued.

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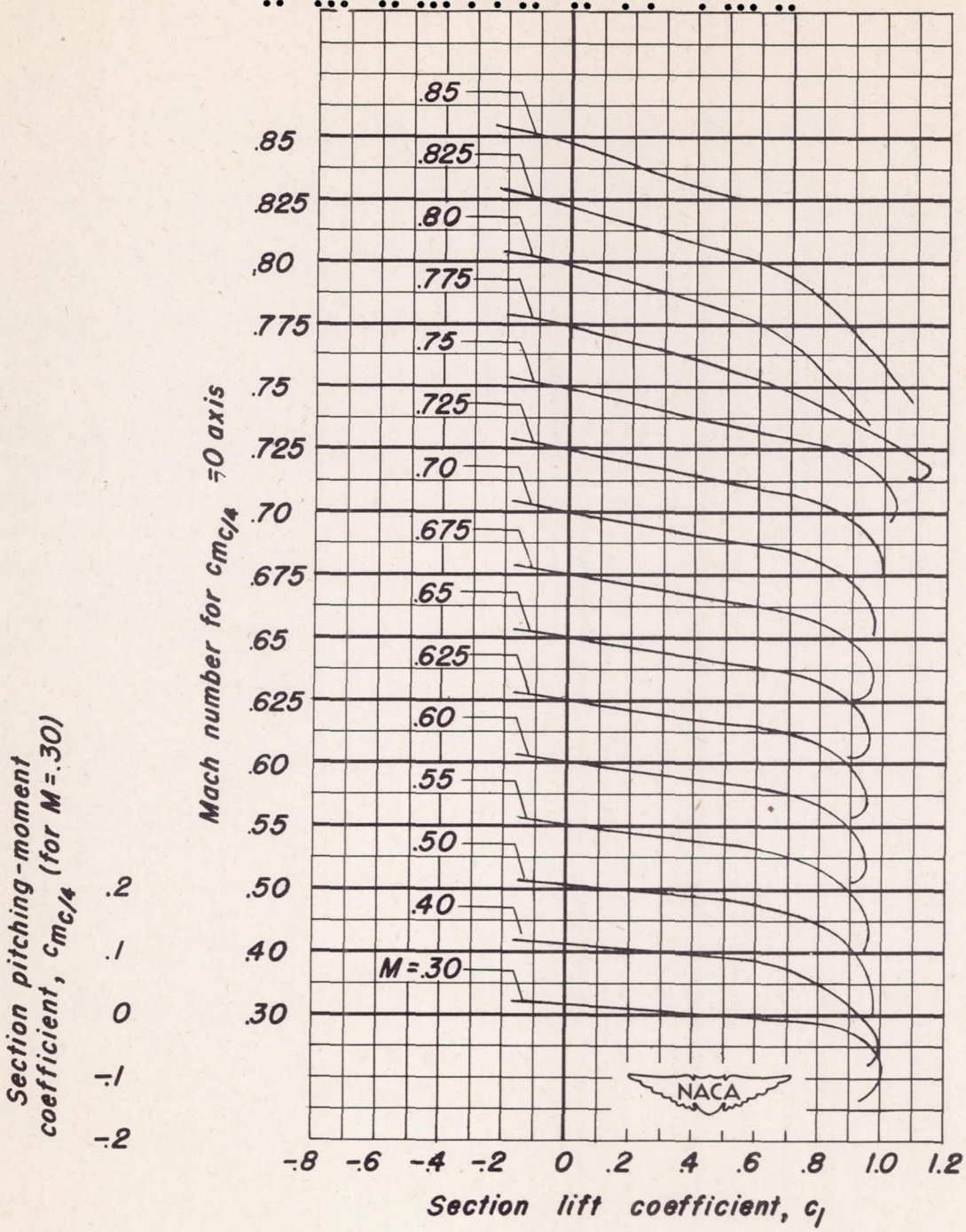

(d) $\frac{h}{t}$, 1.00.

Figure 8.–Concluded.

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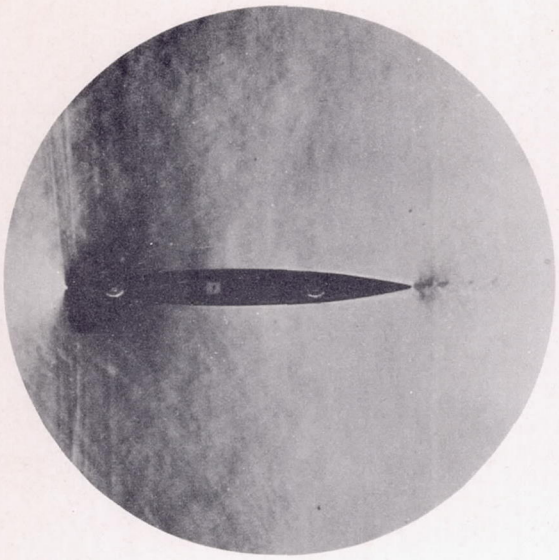
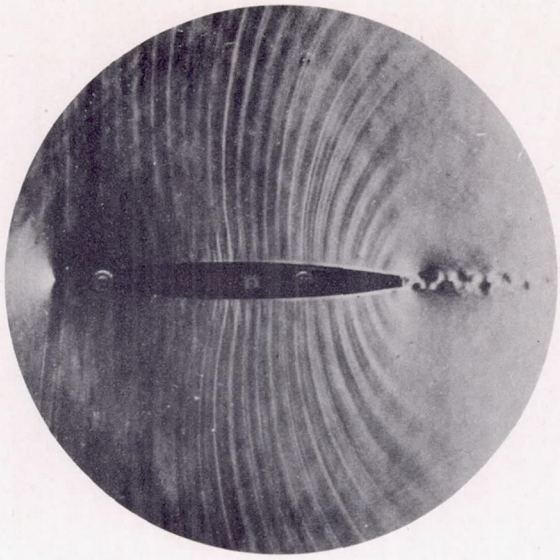
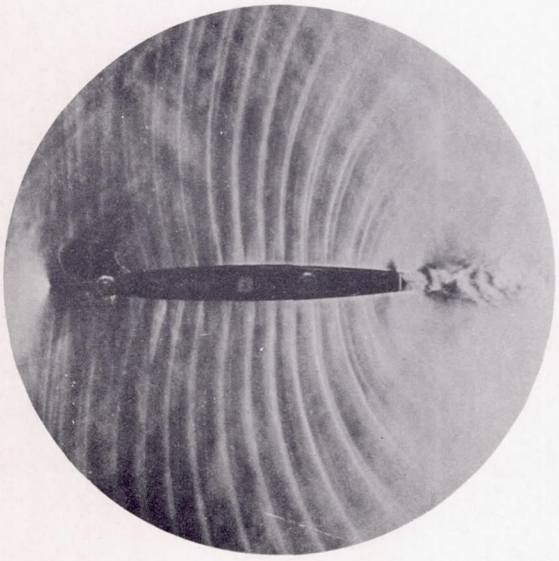


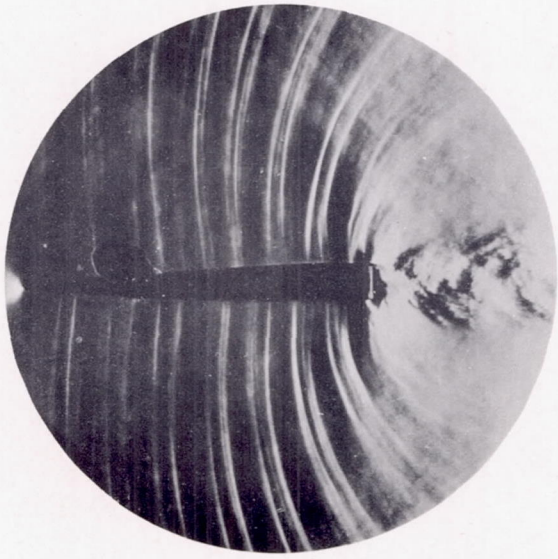
(a) $\frac{h}{t}$, 0.



(b) $\frac{h}{t}$, 0.25.



(c) $\frac{h}{t}$, 0.50.



(d) $\frac{h}{t}$, 1.00.

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Figure 9.– Schlieren photographs of the flow fields of the various airfoil sections at 0.7 Mach number and zero angle of attack.

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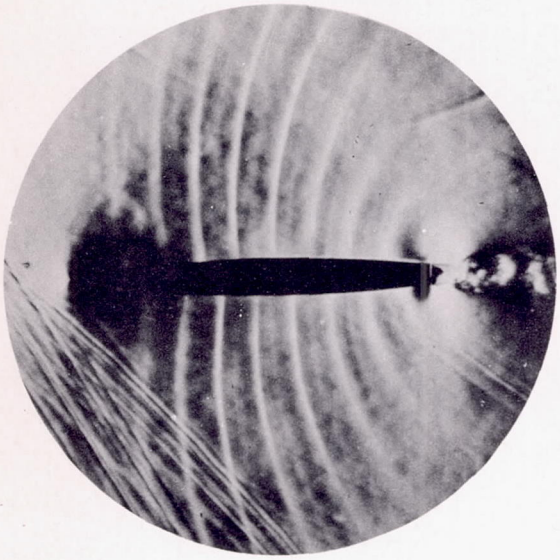
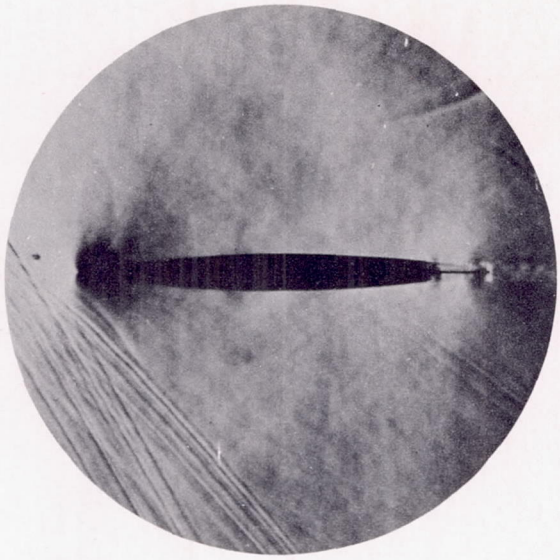
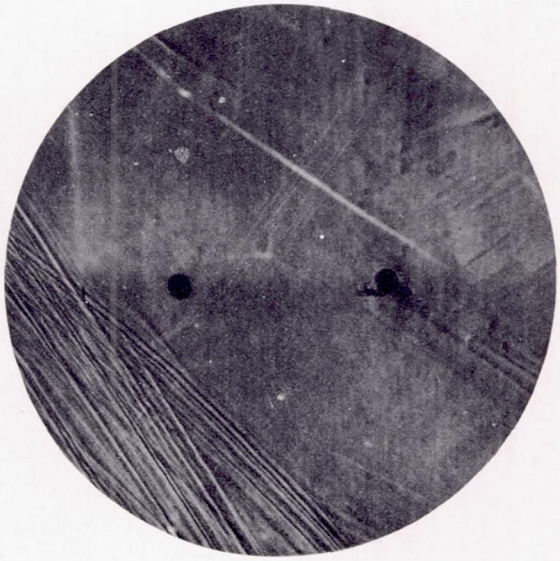


(a) Without splitter plate.



(b) With splitter plate.



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(c) Model removed; no flow.

Figure 10.- Schlieren photographs of the effect of the splitter plate on the flow field of the airfoil section having a trailing-edge thickness equal to one-half the maximum thickness at 0.65 Mach number and zero angle of attack.

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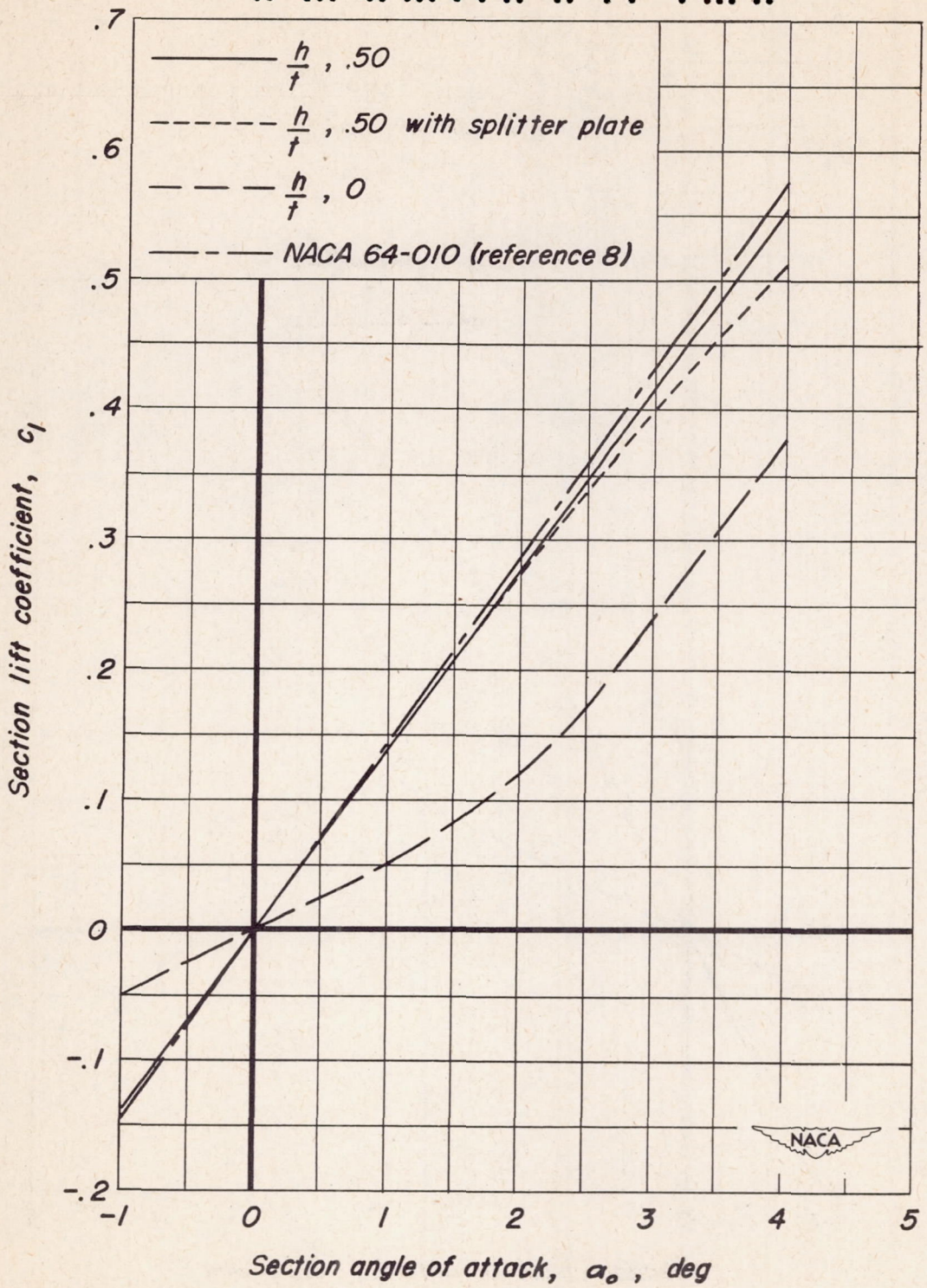
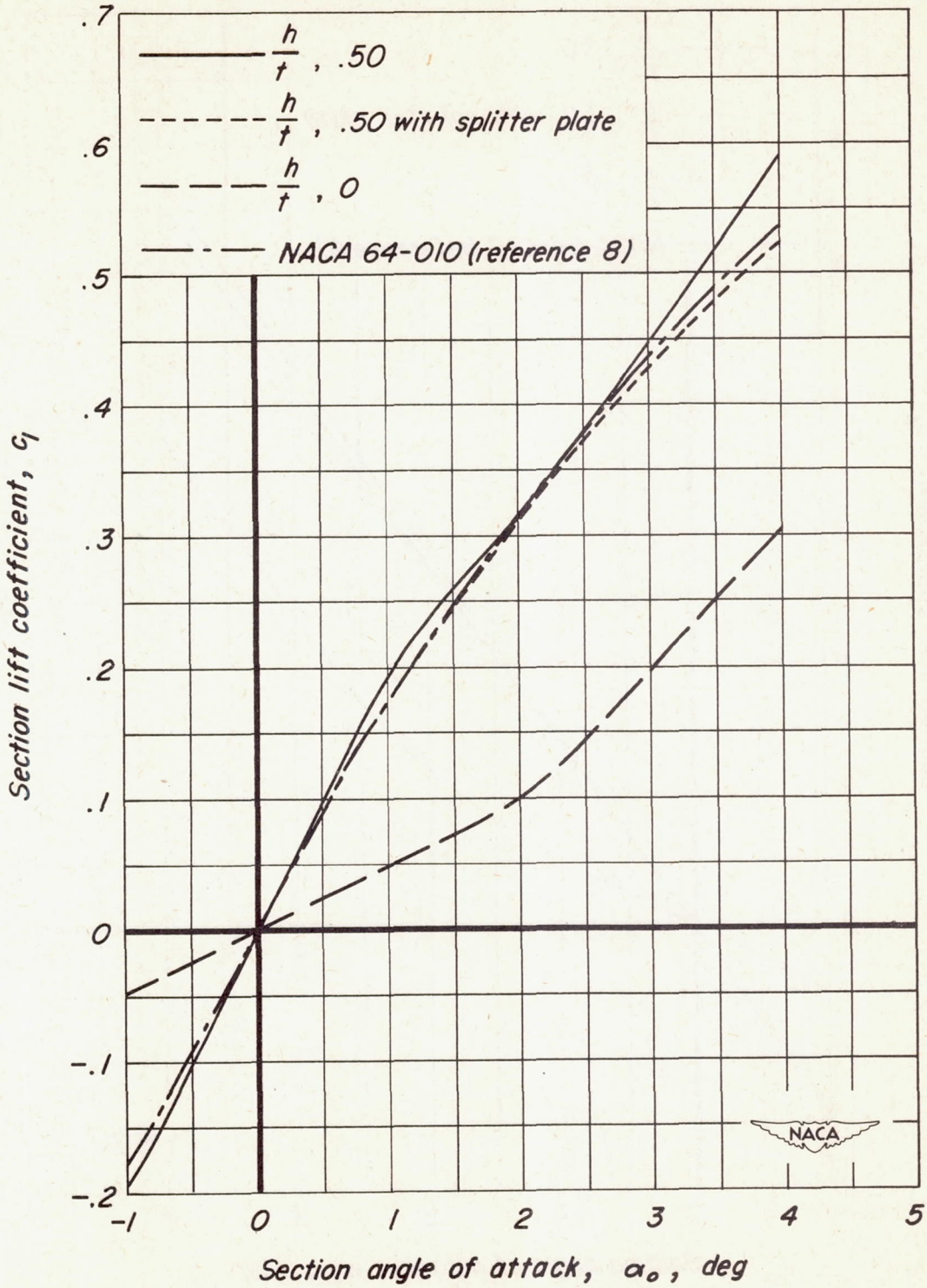


(a) M, 0.65.

Figure 11.- Comparison of the lift curves of several airfoil sections.



(b) M, 0.85.

Figure 11.- Concluded.

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