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RESEARCH MEMORANDUM

AERODYNAMIC CHARACTERISTICS OF LOW-ASPECT-RATIO WINGS AT HIGH SUPERSONIC MACH NUMBERS

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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AERODYNAMIC CHARACTERISTICS OF LOW-ASPECT-RATIO WINGS AT HIGH SUPERSONIC MACH NUMBERS

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This paper presents some recently obtained data on the aerodynamic characteristics of low-aspect-ratio wings at supersonic Mach numbers of 4.04 and 6.9 and discusses some new methods of predicting the lift and drag of such wings. Data on lifting wings in the Mach number range above 2.5 are not plentiful and most of the available data may be found in references 1 to 8.

The plan forms, airfoil sections, and thickness ratios of the wings tested are given in figure 1. The wings shown in figures 1(a) and 1(b) all have double-wedge airfoil sections, with constant thickness ratios over the wing span. The wings of figure 1(c) are all of the same family, having hexagonal airfoil sections with constant thickness outboard to the 56-percent-semispan station and double-wedge sections with maximum thickness at the 69.2-percent-chord station from there to the wing tips. Exceptions to this are the two delta wings which have rounded leading edges and the clipped delta wings. The wings were selected to extend the Mach number range of data on wings previously tested and to investigate the effects of changes in the aspect ratio of delta wings, changes in wing plan form, and changes in airfoil section and thickness. The models tested at Mach number 6.9 in the Langley 11-inch hypersonic tunnel were sting mounted, and lift and drag data were obtained. The models tested at Mach number 4.04 in the Langley 9- by 9-inch Mach number 4 blowdown jet were tested as semispan models extending out into the stream from a boundary-layer bypass plate; lift, drag, pitching moment, and wing-root bending moment were measured. The test Reynolds numbers given in figure 1 are based on the wing root chords.

The aerodynamic characteristics of the double-wedge-section delta wings will be considered first. A summary of the lift-curve slopes at zero angle of attack for the double-wedge-section wings of this investigation is presented in figure 2, together with some data on delta wings of the same section from the Langley 9-inch supersonic tunnel at Mach numbers from 1.62 to 2.41. The ordinate in figure 2, the ratio of the delta-wing lift-curve slope to the linear-theory two-dimensional lift-curve slope, and abscissa, the ratio of the tangent of the semi-apex angle of the wing to the tangent of the free-stream Mach angle, are basic parameters obtained from the linear theory of delta wings (refs. 9 and 10). Tangent ratios less than 1 represent wings with subsonic leading edges, whereas at tangent ratios greater than 1 the wing leading edges are nominally supersonic, but may be actually subsonic because of shock detachment due to wing thickness. The shaded region in figure 2 includes points obtained in various other facilities throughout the country from tests of delta wings with thickness ratios equal to or less than 3 percent at Mach numbers from 1.2 to 2.4 (refs. 11 to 16). In the past, the analysis of delta-wing data for Mach numbers below 2.5, plotted to the variables of figure 2, has led to several conclusions: first, that delta wings having the same section and the same tangent ratio have lift ratios which are relatively independent of Mach number; and, second, that the linear theory gives a fairly accurate prediction of the lift of thin delta wings at low values of the tangent ratio, but overestimates the lift at tangent ratios from about 0.7 to 1.5. As wing thickness ratios increased, the lift-curve slopes were found to become increasingly less than the linear-theory values. The only theoretical methods which take leading-edge shock detachment into account, and thus might be expected to give better predictions of the lift of delta wings in the shock-detached region, are methods using conical characteristics solutions, such as that of Maslen (ref. 17). These nonlinear methods are very laborious and simpler methods are desirable. The data from the tests of double-wedge-section delta wings at Mach numbers 4.04 and 6.9 (fig. 2) indicate that these linear-theory parameters are not adequate for correlating higher Mach number data, since the high Mach number tests generally gave higher lift ratios than the low Mach number tests. In the region of attached leading-edge shocks, it was found that the lift-curve slopes were very close to the shock-expansion two-dimensional values for the wing airfoil sections. Accordingly, the data were plotted (fig. 3) as the ratio of the experimental lift-curve slope to the two-dimensional shock-expansion lift-curve slope for the streamwise airfoil section of the wing. In general, lift ratios close to 1 were obtained at high values of the tangent ratio, indicating that the two-dimensional shock-expansion theory gives good predictions of lift-curve slopes of delta wings when the leading-edge shock is attached.

At values of the tangent ratio close to those for shock detachment, the experimental lift ratios dropped abruptly below 1, as was noted at the lower Mach numbers by Love (ref. 18). Some simple method of predicting the variation of lift ratio in this region is desirable. Since the predictions of the linear theory are the same as those of the characteristics theory for wings of zero thickness, it was assumed that the lift of finite-thickness wings in the shock-detached region varies in a manner similar to linear-theory predictions for zero-thickness wings. The similarity constant was determined by the shock-detachment value of the tangent ratio for each wing. Using these constants, curves were drawn from the shock-detachment points to predict the wing lifts, as shown in figure 3. This modification to the linear theory predicts the experimental life curve slopes in the shock-detached regions with a maximum error of 5 percent for the five Mach numbers shown in figure 3. When extended to the prediction of lift-curve slopes of arrow- and diamond-plan-form wings tested at Mach number 4.04 by modifying the results of Puckett and Stewart's theory (ref. 19), given in chart form in reference 20, the method gave predictions within 7 percent of the experimental values for one arrow wing with a single-wedge section and one diamond-plan-form wing with a hexagonal section.

The previous figures have presented data on lift-curve slopes at zero angle of attack. Figure 4 presents typical lift curves for double-wedge-section delta wings at Mach numbers 4.04 and 6.9. At both Mach numbers the curves are essentially linear at low angles of attack. Nonlinearities are evident at angles of attack above approximately 6°, especially at Mach number 6.9. An estimate of the lift of the wing having a 30° semiapex angle at Mach number 6.9 at 10° angle of attack would be 20 percent low if based on the lift-curve slope at 0° angle of attack. The experimental data for the wings of figure 4 follow very closely the predictions of the shock-expansion two-dimensional theory for the streamwise airfoil sections of the wings at both Mach numbers, as long as the leading-edge shock is attached. When the angle of attack becomes so large that the leading-edge shock detaches, the experimental values begin to fall below the shock-expansion theory. This is especially noticeable at Mach number 6.9, where an abrupt change in the slope of the lift curves occurs at the angles of attack at which leading-edge shock detachment is predicted theoretically. At Mach number 4.04, the data for the 5-percent-thick wing, which has an attached leading-edge shock, agree very well with the shock-expansion theory, whereas the experimental lift coefficients for the much blunter 8-percent-thick wing, which has a detached shock at zero angle of attack, fall below the theoretical values. The shock-expansion theory gives predictions of the lifts of the double-wedge wings tested within about 2 percent of the experimental value at Mach number 4.04 and within 5 percent at Mach number 6.9, as long as the angle of attack is below that for leading-edge shock detachment.

The next section of this paper discusses methods of predicting and correlating the drag of low-aspect-ratio delta wings. The prediction of drag results involves, of course, three factors: predictions of friction drag, minimum pressure drag, and drag due to lift. In order to make a theoretical prediction of friction drag, predictions of the type of boundary layer and the location of boundary-layer transition must be made. Satisfactory theoretical methods of predicting boundary-layer transition on wings are not available at present, but the transition point, the nature of the boundary layer, and the value of the friction-drag coefficient can often be determined by experimental means in wind tunnels or in free flight. For example, an experimental value of the friction-drag coefficient at Mach number 4.04 was obtained by plotting the drag coefficients of wings having the same plan form and section against the square of the wing thickness ratio and making a straight-line extrapolation through the experimental points to the zero-thickness ordinate. A value of 0.0036 was obtained. Furthermore, the boundary-layer-transition lines on these same wings were determined by fluorescent-lacquer tests and, by using this information and by assuming no variation of C_{D_f} with wing thickness ratio, estimates of the friction-drag coefficients of the wings were made using Van Driest's value of laminar skin-friction-drag coefficient (ref. 21), corrected for differences in stream static temperature (ref. 22), and the Frankl and Voishel extended value of the turbulent skin-friction-drag coefficient (ref. 23). An estimated value of 0.0033 was obtained by this method, which compares favorably with the experimental value of 0.0036. The experimentally determined value of the skin-friction drag coefficient was used to obtain the minimum pressure-drag coefficients at Mach number 4.04 used in the following discussion. Theoretically determined friction-drag coefficients were used at Mach number 6.9.

The next component of wing drag which will be considered is the minimum pressure drag. The linear theory for delta wings as derived by Puckett (ref. 24) indicates that all delta wings with double-wedge airfoil sections having a given maximum-thickness location and the same value of the tangent ratio will have the same value of $\dfrac{C_{D_{P_{min}}}\sqrt{M^2 - 1}}{(t/c)^2}$, the ordinate of figure 5, for all thickness ratios and Mach numbers. Thus, the linear theory for each family of delta wings investigated appears as single curves in figures 5(a) and 5(b). The predictions of linear theory are rather poor for the wings shown in figure 5(a); however, all the experimental data for the wings with maximum thickness at 50 percent chord, wings $2\frac{1}{2}$, 5, and 8 percent thick, tested at Mach numbers from 1.62 to 6.9, fall very nearly on one curve, showing that these parameters successfully correlate experimental data for this family of wings. This result is found only for wings with sections that are symmetrical about the midchord point, since the higher order effects are small for such wings. For other wing sections with maximum thicknesses ahead of or behind the 50-percent-chord point, the higher order terms become important and the theory indicates Mach number effects in the shock-attached region which cannot be correlated by these parameters.

This point is illustrated by the results obtained from the wings with maximum thickness at 18 percent chord presented in figure 5(b). The predictions of the linear theory are poor for these blunt wings at low values of the tangent ratio due to the transonic nature of the flow over the wings; however, the lower Mach number data correlate well, since the second-order effects for this wing section are small at these Mach numbers. The data at the higher Mach numbers, the three experimental points obtained at Mach number 6.9 and the experimental value obtained at Mach number 4.04, indicate that the high Mach number data do not correlate with the lower Mach number data at tangent ratios close to and beyond the shock-attachment value. The trend of the data at each Mach number indicates that the pressure drags become constant at values close to those predicted by shock-expansion theory for the wing section at each test Mach number. This same trend was clearly evident in figure 5(a) for the symmetrical double-wedge wings. Thus, it can be seen that, with the aid of shock-expansion two-dimensional theory, satisfactory predictions of the pressure drags of double-wedge delta wings can probably be made throughout the supersonic Mach number range up to 6.9. (Some of the data of figure 5 were presented in figure 11 of reference 5. The discussion of the pressure-drag data in reference 5 and the second conclusion of that reference are correct with reference to the wings with maximum thickness at 50 percent chord but apply only for Mach numbers from 1.62 to 2.4 for the wings with maximum thickness at 18 percent chord.)

If the skin-friction drag and the minimum pressure drag of a wing have been determined, the variation of the drag due to lift must be known if any estimates of lift-drag ratios are to be made. For all the wings of this investigation it was found that the drag due to lift was equal to the normal force times the sine of the angle of attack. This has also been found to be the case for a large number of low-aspect-ratio wings tested at lower supersonic Mach numbers in the Ames 6- by 6-foot supersonic tunnel (ref. 16).

Some characteristics of the family of wings shown in figure 1(c), which have hexagonal sections and were tested at Mach number 4.04, are now considered and the experimental results will be compared with the predictions of the modified theory. The delta and the diamond-plan-form wings have constant-thickness sections out to 56 percent of the semispan and double-wedge sections from there to the wing tip. The tapered wing was made by cutting the tip from the delta wing at 56 percent of the semispan. Two of the wings were tested with both wedge leading edges and NACA 0003-63 leading-edge sections.

Wings with rounded leading edges are of interest at high Mach numbers, since rounded leading edges have better heat-conducting properties than sharp leading edges and thus will be more likely to keep their strength at the high temperatures which will be encountered at high supersonic Mach numbers. Figure 6 shows the effects on the lift and drag of two delta wings at Mach number 4.04 of replacing the wedge-leading-edge sections by NACA 0003-63 leading-edge sections. The shock was attached to the wedge leading edge of the wing having the 30° semiapex angle and was detached from the wedge leading edge of the wing having the 10° semiapex angle. The change from sharp to rounded leading edge resulted in a 50-percent increase in the minimum drag of the 30° wing, which is about a 90-percent increase in the pressure drag. This result has also been found at lower supersonic Mach numbers. The maximum lift-drag ratio was decreased 20 percent from 6 to 4.9 by rounding the leading edge of the wing.

The data for the lower-aspect-ratio wing, which has a subsonic leading edge, indicate that rounding the leading edge of this wing may have also caused an increase in drag. This is contrary to lower Mach number experience (for example, see ref. 25) and must be investigated further.

The methods discussed previously gave predictions of the lift of the sharp-leading-edge wings within 5 percent of the experimental values and predictions of pressure drag of the same wings within about 10 percent of the experimental values. The methods of predicting delta-wing lift and pressure drag which have been proposed here are, of course, not applicable to wings with rounded nose sections. Therefore, pressure distributions over the two wings having NACA 0003-63 nose sections were estimated by the Newtonian method as presented in reference 26, combined with a Prandtl-Meyer expansion over the lee surfaces of the wings and empirical values of base pressure. Drag coefficients were obtained by this method that were within 5 percent of the estimated experimental pressure drags. Using the modified method and the Newtonian method, the drag increments for these wings due to rounding the leading edges were predicted within 25 percent. It should be pointed out that the friction drag of the wings with the rounded leading edge is not known with the same accuracy as that of the sharp-leading-edge wings, so that the estimates of total drag may not be as accurate as the calculations indicate. The drag due to lift of these wings was found to be equal to the normal force times the sine of the angle of attack, as was the case for the double-wedge-section wings.

At Mach number 4.04 the locations of the wing-panel centers of pressure were determined experimentally. The chordwise location of the centers of pressure ranged from about 1.5 percent of the root chord downstream to 5 percent of the root chord upstream of the center of area of the wing panel. The spanwise location of the centers of pressure of the semispan models ranged from 2.5 to 5 percent of the semispan outboard of the center of area of the wing panel.

These methods of predicting wing lift and drag should give improved predictions of wing-body characteristics when used with wing-body-interaction methods such as the method of Nielsen and Kaattari (ref. 27). Figure 7 presents an example of some improvements in wing-body predictions obtained by the use of the more accurate values of wing lift obtained from the modified theory. The data are for four delta wing-body combinations for which the Mach lines, starting from the wing-body juncture, lie inside the wing leading edge, but which are actually operating with detached shocks due to wing thickness. Three of the configurations were tested at Mach number 1.93 (ref. 28) and one at a Mach number 4.06 (ref. 4). The ordinate of figure 7 is the experimental value of the increment in lift coefficient due to the addition of wings to a body. The abscissa is the theoretical value of the same quantity. The open points show the relatively poor predictions obtained by the use of the simple-linear-theory lift coefficients. The solid points show the improved predictions obtained by the use of the modified-theory wing lift coefficients. The good prediction by the linear theory at Mach number 4.06 is fortuitous, since it is the result of compensating effects, and such agreement should not be expected for other configurations at high Mach numbers.

To summarize, some simple methods of predicting lifts and pressure drags of thin delta wings at supersonic Mach numbers up to 6.9 have been presented. These methods are mainly modifications to the linear theory based on the physical realities of the flow, including shock detachment. Tests of a considerable number of low-aspect-ratio wings at Mach numbers from 1.6 to 6.9 have indicated that these methods accurately predict the wing lift and pressure drags. The effects on minimum drag of rounding the leading edges of two delta wings at Mach number 4.04 were predicted satisfactorily by the use of the Newtonian theory in combination with a Prandtl-Meyer expansion over the lee surfaces.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., September 1, 1953.

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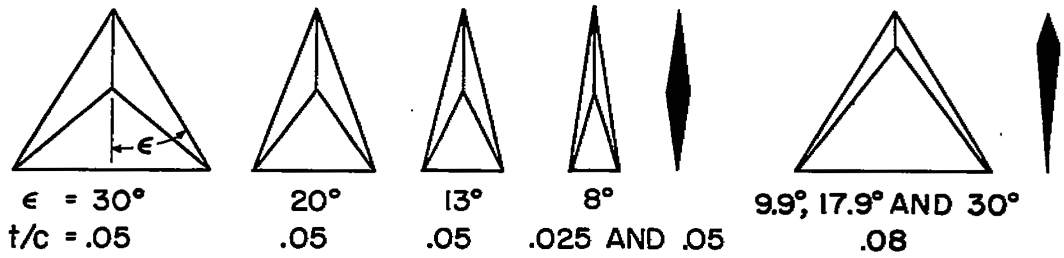
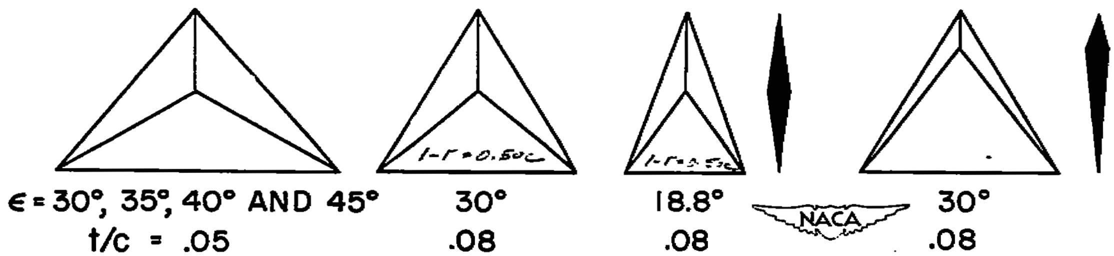
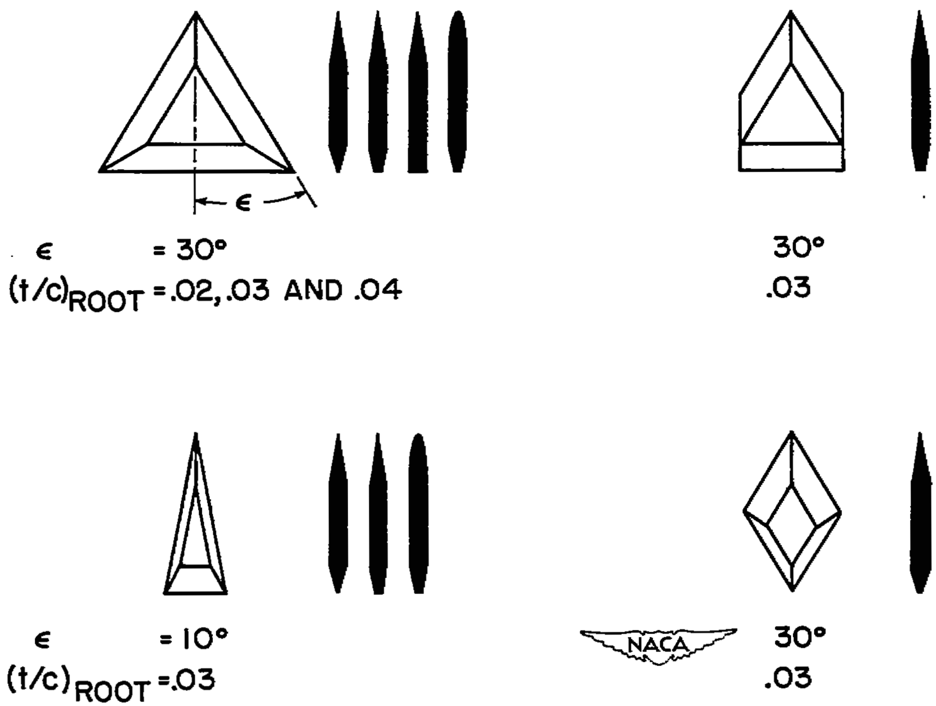


(a) $M = 6.9$; $R = 1.1 \times 10^6$ to 2.3×10^6.



(b) $M = 4.04$; $R = 9 \times 10^6$.



(c) $M = 4.04$; $R = 9 \times 10^6$.

Figure 1.- The geometry of the wings tested.

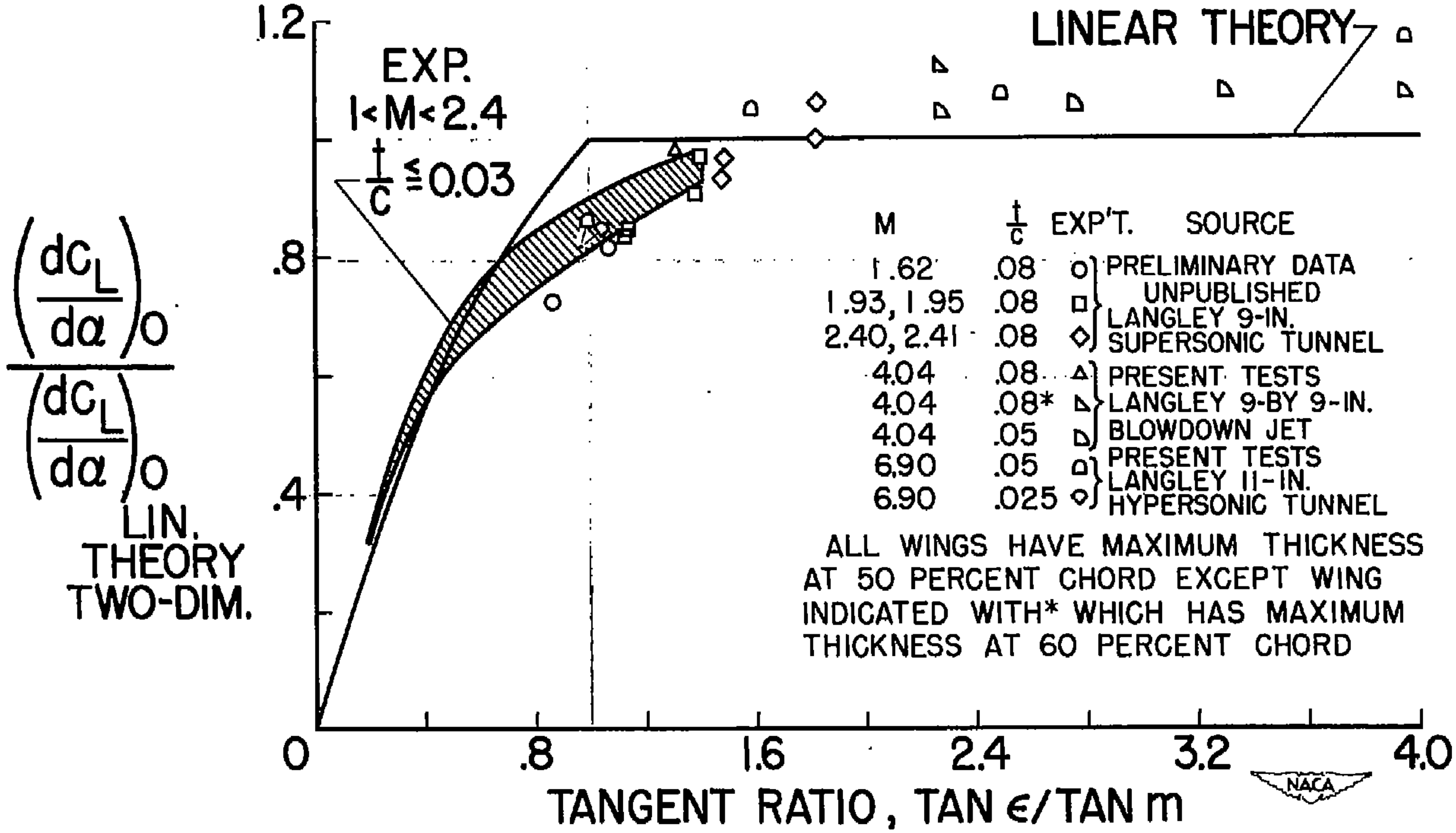


Figure 2.- Comparison of experimental lift-curve slopes at 0° angle of attack of double-wedge-section delta wings with the predictions of the linear theory.

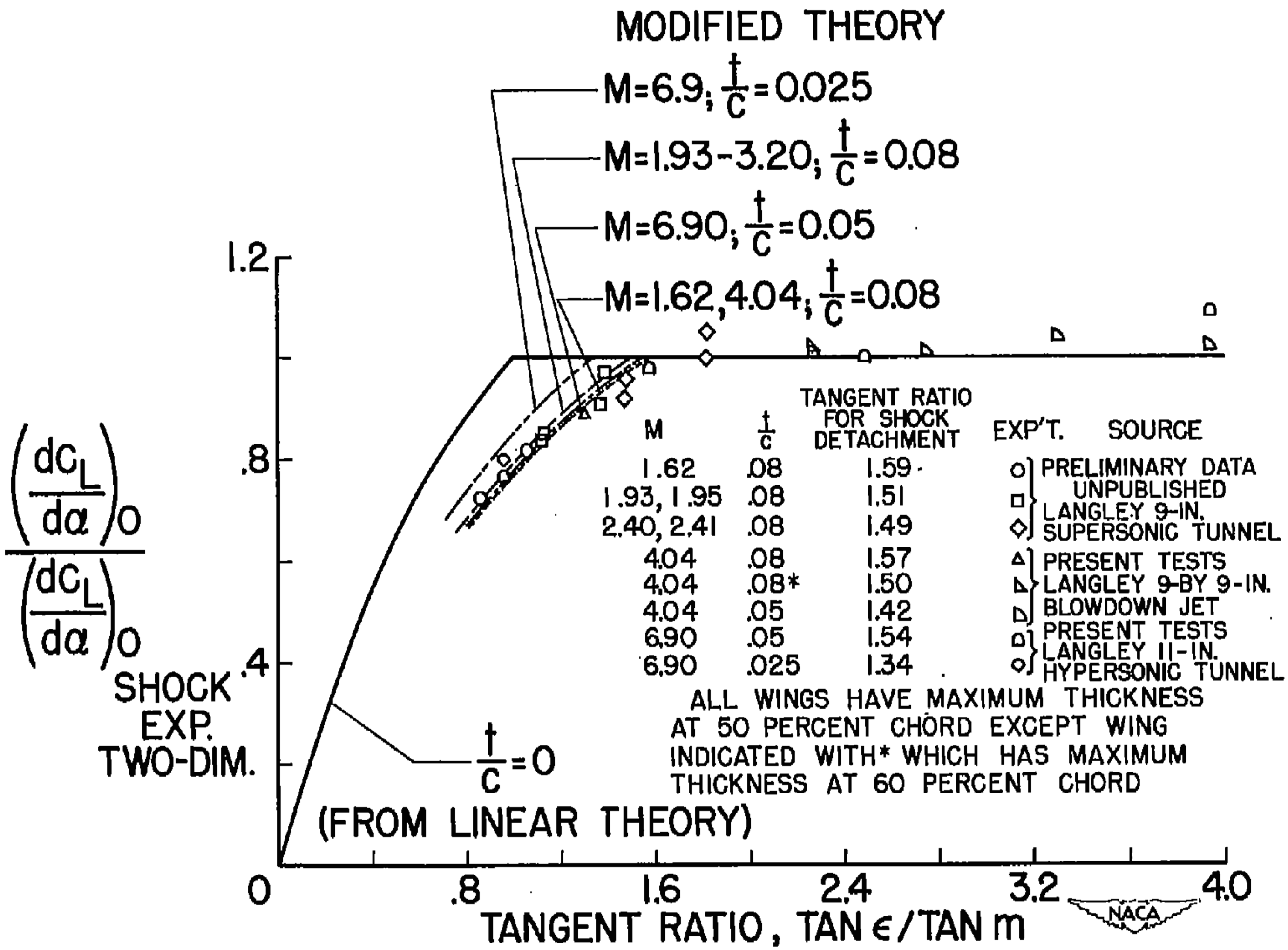
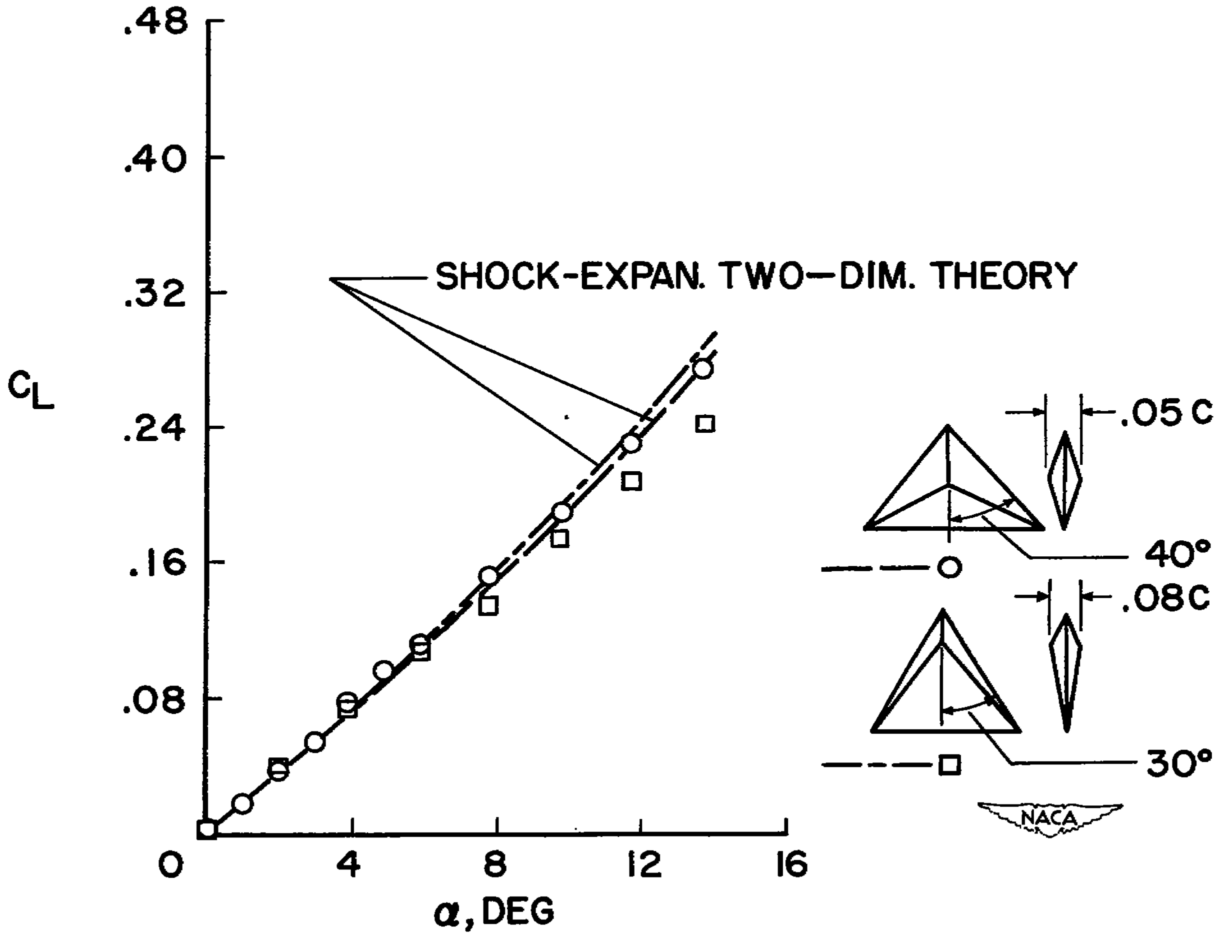
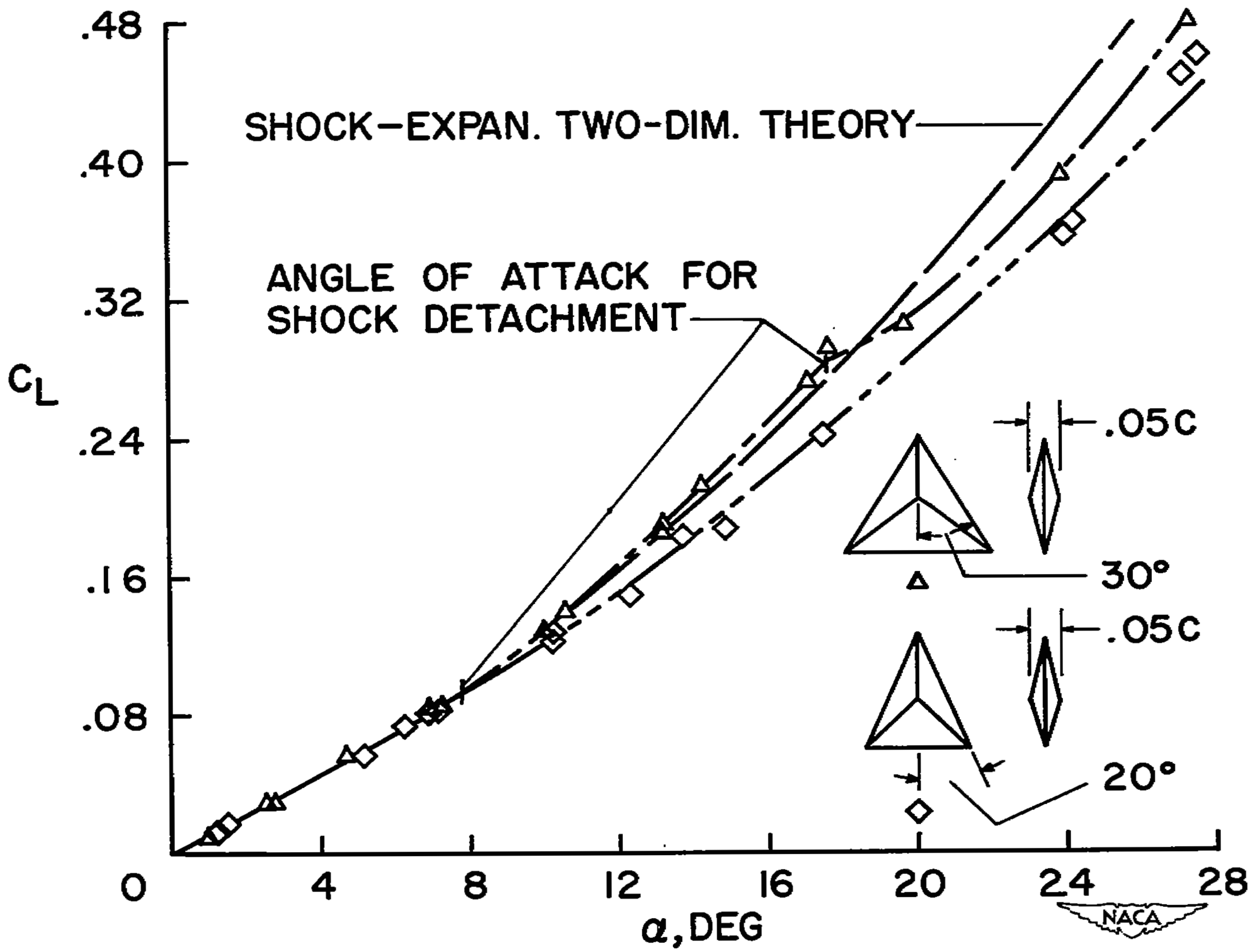


Figure 3.- Comparison of experimental lift-curve slopes at 0° angle of attack of double-wedge-section delta wings with the predictions of a modified linear theory.



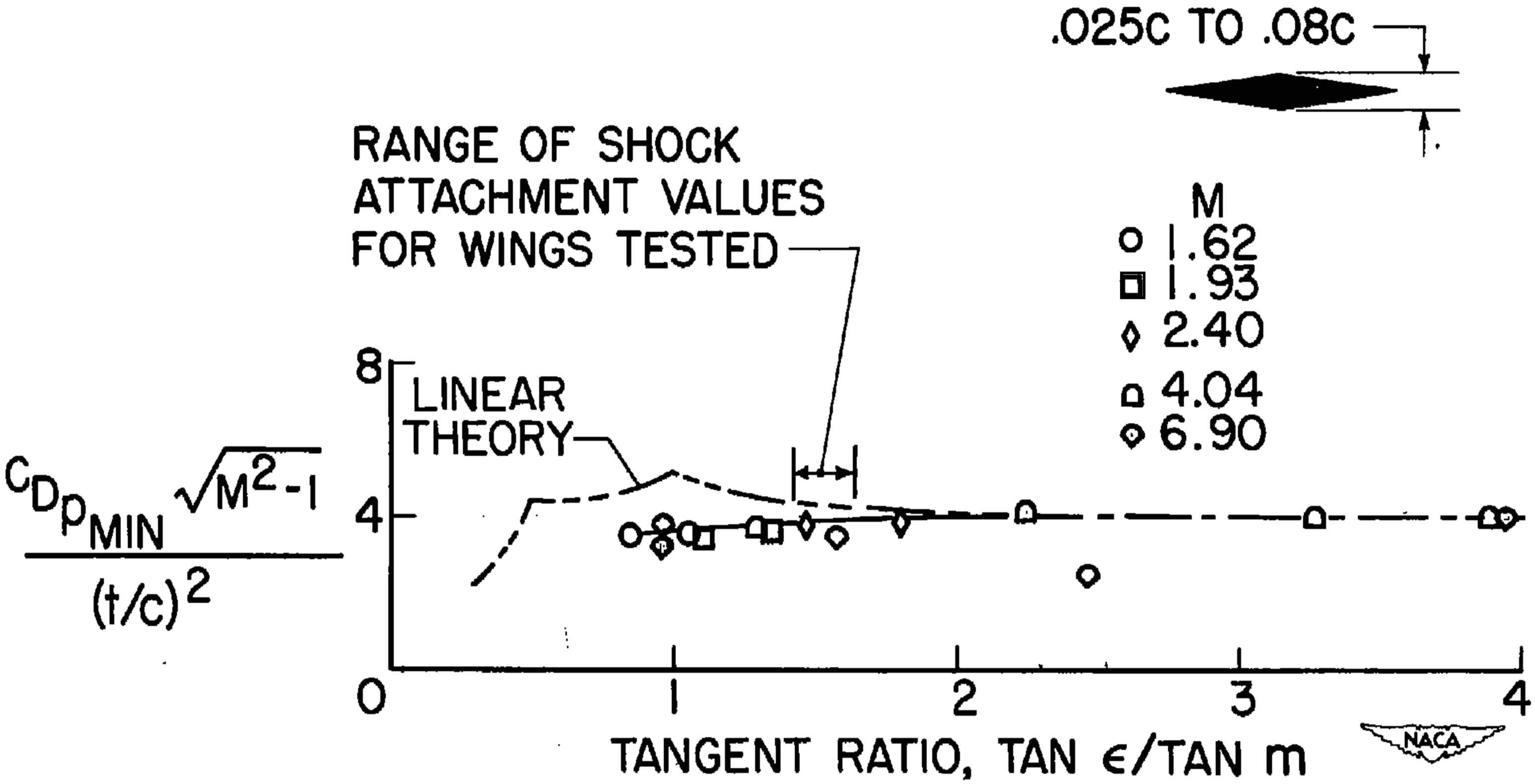
(a) $M = 4.04$; $R = 9 \times 10^6$.

Figure 4.- The variation of delta wing lift coefficient with angle of attack at high supersonic Mach numbers.



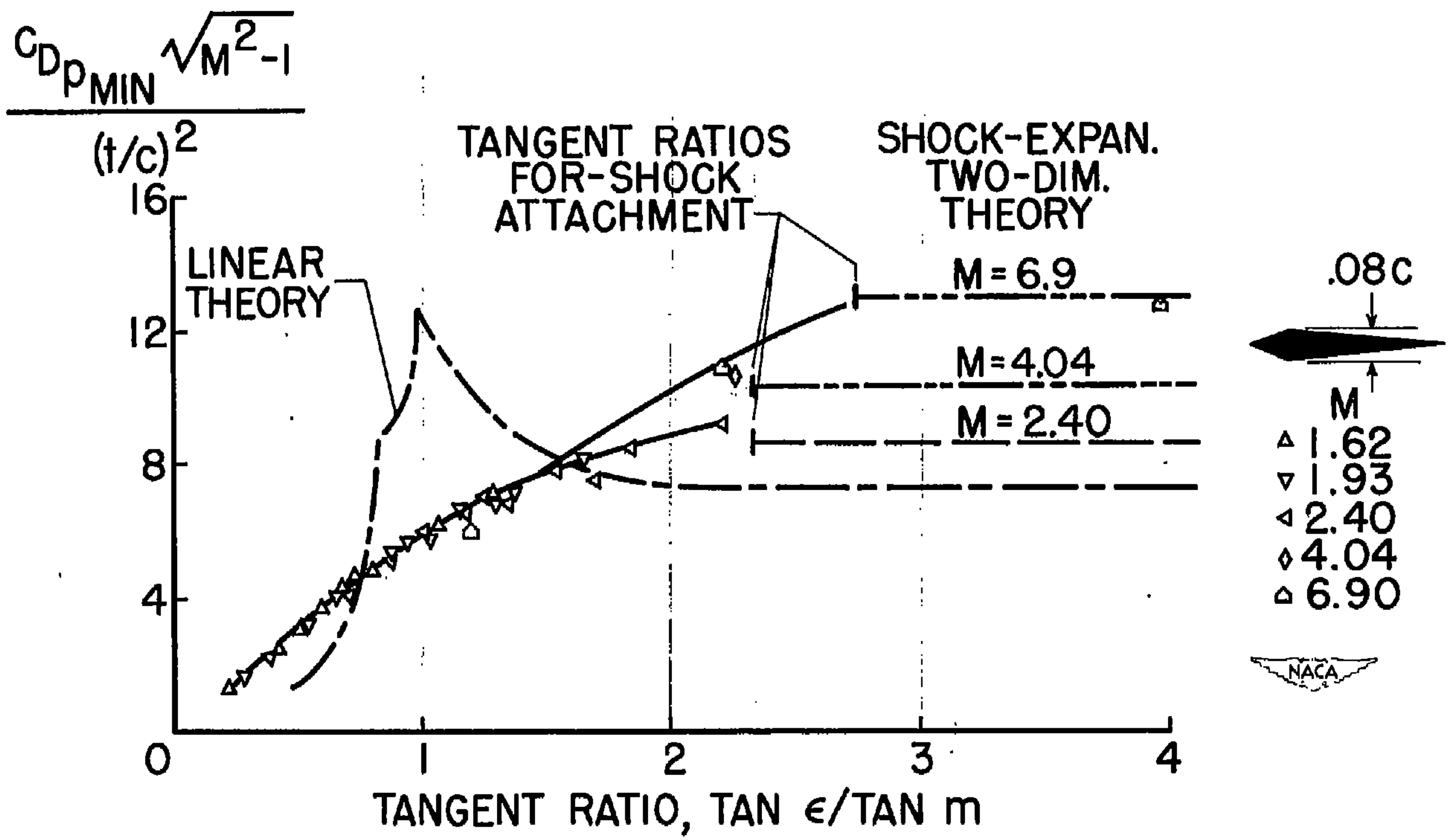
(b) M = 6.9; R = 1×10^6 to 2.0×10^6.

Figure 4.- Concluded.



(a) t/c_{max} at 0.5c.

Figure 5.- The correlation of minimum drag coefficients of double-wedge delta wings according to linear-theory parameters.



(b) t/c_{max} at 0.18c.

Figure 5.- Concluded.

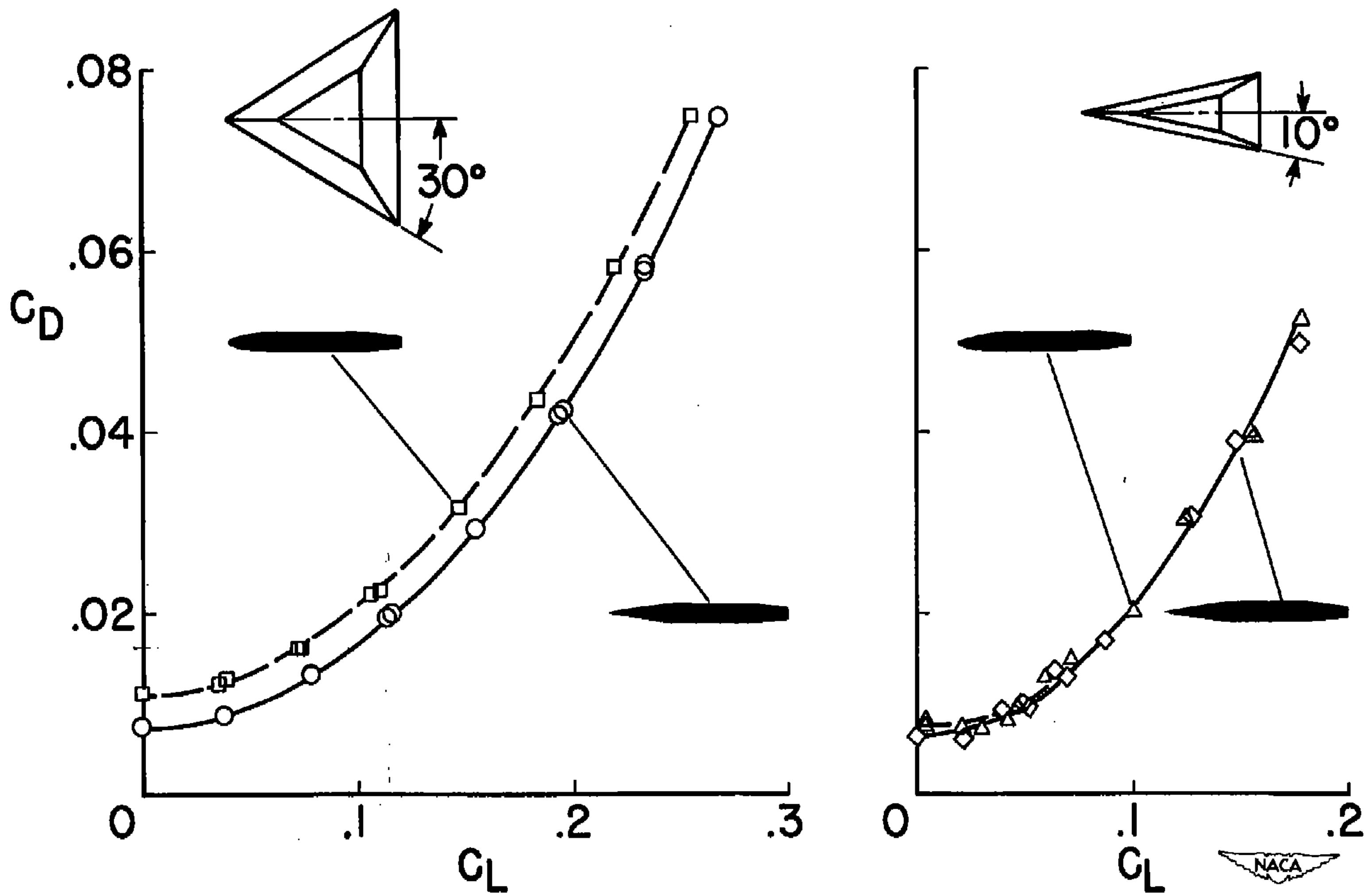


Figure 6.- The effect on the lift and drag coefficients of delta wings of changing the leading-edge shape from a wedge to an NACA 0003-63 profile. $M = 4.04$; $R = 9 \times 10^6$.

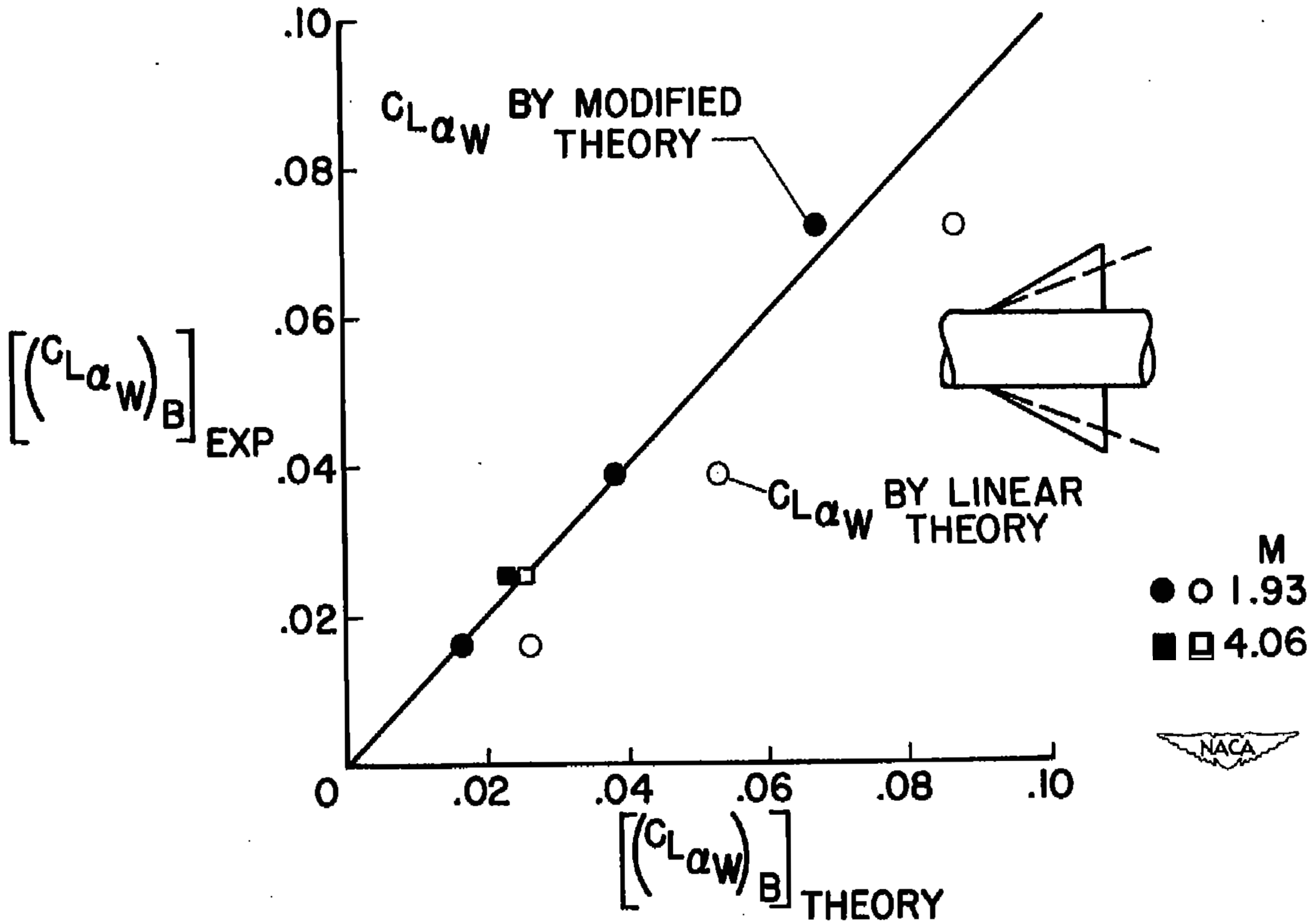


Figure 7.- The agreement obtained between theoretical and experimental values of the increment in lift coefficient due to the addition of wings to a body by using predictions of wing lift from linear theory and the modified theory in the method of Nielsen and Kaattari.