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AIRFOIL SECTION CHARACTERISTICS AT HIGH SUBSONIC SPEEDS

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The design of airplane lifting surfaces and the prediction of their aerodynamic characteristics have always depended, to a great extent, on airfoil section data. The drag data are of particular significance from the standpoint of airplane performance, whereas the lift and pitching-moment data are of appreciable importance from the point of view of airplane stability and control. It was first believed, nevertheless, that airfoil section data would be of limited value for swept wings. Recent theoretical work of V. V. Struminsky and also of R. T. Jones reported in references 1 and 2, however, have indicated that airfoil section data should have a wider scope of application in the design of high-aspect-ratio swept wings. In such use, the aerodynamic characteristics of the airfoil section normal to the swept leading edge are used together with the Reynolds and Mach numbers corresponding to the normal component of the free-stream velocity. According to the present knowledge, this is the most logical procedure for the selection of airfoil sections of a high-aspect-ratio wing either straight or swept. Although the qualitative use of airfoil section data is made in the design of swept wings, it should not be inferred that such data may be used quantitatively. In view of the aforementioned application it was thought that a review of important high-speed properties of several NACA 6-series airfoils would be of interest. A large part of the data which will be presented has already been reported in references 3 and 4.

It is generally known that the reduction in the lift and the abrupt increase in the drag of an airfoil section (in other words, the divergence of forces) occur at speeds somewhat greater than the airfoil critical speed. Since the force-divergence Mach numbers for a given airfoil are of particular interest in the design of wings for high-speed aircraft, it has been suggested that the critical Mach number might be used as a conservative indication of these Mach numbers. In figure 1 is shown a calculated curve of critical Mach number for the NACA 64-210 airfoil section. This curve was determined from the Kármán-Tsien relationship between the critical Mach numbers and the peak incompressible pressure coefficients of the airfoil. The extremities of the curve are determined by the peak pressures at the airfoil leading edge, whereas the center part is determined by the peak pressures which are, for this airfoil, at 0.4 chord. For comparison, curves of lift- and drag-divergence Mach numbers determined from experimental results are also shown.

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It can be seen in figure 1 that the critical Mach number curve is entirely inadequate in indicating the range of lift coefficient over which the lift- and drag-divergence Mach numbers are the highest. Only in the center part does the curve of critical Mach number approximate the force-divergence Mach number curves. This center part of the critical Mach number curve is quite close to the curve of drag-divergence Mach number, but the curve of lift-divergence Mach number is about 0.025 Mach number higher. For every lift coefficient shown in this figure, the Mach numbers of lift divergence are greater than the Mach numbers of drag divergence. A comparison of experimental and calculated critical Mach number curves has shown very good agreement between the center part of the curves; the extremities of the experimental curve, however, came between those of the curves of calculated critical Mach number and the curves of force-divergence Mach numbers. It is apparent, then, that the Kármán-Tsien relation overestimates the increases in peak pressures at the airfoil leading edge. It is quite significant to note that there are greater differences between the force-divergence Mach numbers and the critical Mach numbers when the critical Mach numbers have been determined from peak pressures at the airfoil leading edge than when they are determined from peak pressures located somewhat behind the leading edge. This is explained by the fact that at a given increment in Mach number above the critical the extent of the region of supersonic flow at the leading edge is much less than the corresponding region of flow at a position on the airfoil somewhat behind the leading edge. These remarks are also applicable to other airfoils besides the NACA 6-series type.

The disposition of these curves of calculated critical Mach numbers and experimental lift- and drag-divergence Mach numbers for other thin NACA 6-series airfoils can be expected to be similar to that shown in figure 1. At low lift coefficients the cambered airfoils have lift-divergence Mach numbers that are 0.02 to 0.06 greater than the drag-divergence Mach numbers. In general, the differences between these force-divergence Mach numbers appear to be the greatest for the NACA 64- and 65-series airfoils. The ranges of lift coefficients over which the force-divergence Mach numbers are the highest also appear to be somewhat greater for these series of airfoils.

Associated with the divergence of airfoil lift at supercritical Mach numbers is the variation of angle of attack required to maintain a given lift coefficient and the reduction in airfoil lift-curve slope. Shown in figure 2 for several NACA 64-series airfoils is the effect of airfoil thickness ratio and camber on the section angle of attack required to maintain a lift coefficient of 0.1. Variations in this angle of attack are significant in that they lead

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to corresponding changes in airplane trim. This figure shows that for the cambered airfoils the Mach number of the abrupt change in angle of attack increases with a decrease in airfoil thickness ratio. It appears, also, that the large variation in angle of attack is not alleviated by a reduction in thickness ratio but is delayed to higher Mach numbers. The advantage of the symmetrical airfoil is clearly evident when the curves for the NACA 64-210 and 64-010 airfoils are compared. Up to the highest Mach numbers shown, the data indicate that the symmetrical airfoil can maintain a constant lift coefficient of 0.1 with only very small changes in angle of attack.

In figure 3 are shown the effects of thickness ratio and extent of favorable pressure gradient on the section lift-curve slopes per degree of several NACA 6-series airfoils. From the standpoint of the static longitudinal stability of an airplane, the variations of the lift-curve slope of the wing are of appreciable importance. When the lift-curve slope of the wing increases, the downwash at the tail plane increases correspondingly, leading to a decrease in airplane stability; and for a decrease in the lift-curve slope the converse is true. The thickness effect for several NACA 64-series airfoils is indicated in the upper group of curves in figure 3. It can be seen that the Mach numbers at which the lift-curve slopes begin to decrease and the values of the lift-curve slopes at these Mach numbers increase as the thickness ratio decreases. It is of interest to note that the maximum value of the lift-curve slope for the airfoil with a 6-percent thickness is nearly three times as large as the value at low speeds. The effect of a change in the extent of favorable pressure gradient from 0.4 chord to 0.6 chord for 10-percent thick NACA 6-series airfoils is indicated in the lower group of curves in this figure. The data show that the maximum values of lift-curve slope decrease as the region of favorable pressure gradient becomes more extensive or as the position of maximum thickness moves rearward. (The positions of maximum thickness for the NACA 64-, 65-, and 66-series airfoils are at approximately 38-, 41-, and 45-percent chord, respectively.) The Mach numbers at which the lift-curve slopes begin to decrease seem to be the least for the airfoil having the greatest extent of favorable pressure gradient. The preceding remarks can be expected to apply only to airfoils with small trailing-edge angles such as those of the NACA 6-series airfoils.

Data of reference 4 show that the lift-curve slopes for the NACA 63-210 airfoil are practically identical with those for the NACA 64-210 airfoil. Unpublished data indicate that camber has very little effect on the lift-curve slopes of the thin NACA 6-series airfoils.

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Presented in figure 4 are the section drag characteristics of several NACA 64-series airfoils as affected by camber and thickness ratio. The lower group of curves indicate the effect of thickness at a section lift coefficient of 0.2. These data show that the Mach number of drag divergence increases as the thickness ratio decreases. Above this Mach number the increases in drag coefficient appear to be independent of thickness ratio. The upper two curves of this figure show the variation of section drag coefficient at a lift coefficient of 0.2 for the NACA 64-210 and 64-010 airfoils. Even though a comparison at this lift coefficient is disadvantageous for the symmetrical airfoil, the data show that it has a slightly higher drag-divergence Mach number. At Mach numbers just above those for drag divergence the data show that the NACA 64-010 airfoil has the least drags; hence, the lift-drag ratios are the highest for this airfoil at these Mach numbers.

Pitching-moment data for the NACA 6-series airfoils show, in general, no large changes until a Mach number in the vicinity of the lift- and drag-divergence Mach numbers has been reached. Corresponding data for other types of airfoils having trailing-edge angles considerably larger than those for the NACA 6-series airfoils, however, have shown abrupt changes in the pitching moments at high lift coefficients which have indicated rearward shifts of the center-of-pressure position. Even at low lift coefficients, the latter type of airfoil has shown forward movements of the position of center of pressure. For these airfoils with large trailing-edge angles, there is appreciable variation of flow separation near the trailing edge with small changes in airfoil angle of attack. The action is effectively the same as if there were a flap at the trailing edge deflected in opposition to the airfoil angle of attack.

The data which have been presented thus far have been obtained at Mach numbers as high as 0.9. It is noteworthy that in the Langley transonic tunnel, which is now in operation, airfoil pressure-distribution measurements may be made at a Mach number of approximately unity. Shown in figure 5 is a schematic diagram of this tunnel. The tunnel working section is actually a 3-inch annulus between two concentric circular cylinders. The airfoil models are fixed to the rim of a rotor having a diameter equivalent to that of the inner cylinder. The model rotates within the annulus at speeds which correspond to Mach numbers as high as 1.4. Since the ratio of tunnel height to model thickness for this tunnel is almost infinite, the choking effects of the usual subsonic tunnel are eliminated. In order to prevent the model from operating in its own wake and to control the model angle of attack, a low axial velocity is induced through the annulus. In order to reduce the effects of the boundary

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layer on the tunnel walls in the vicinity of the model, air from the boundary layer is removed at three annular slots upstream of the test section.

Shown in figure 6 are preliminary pressure-distribution data obtained in the Langley transonic tunnel at a Mach number of approximately unity for the NACA 66-006 airfoil at zero angle of attack. The data are presented as pressure ratios: the ratio of the local pressure p on the surface of the airfoil to the stagnation pressure p_s. For comparison, the Prandtl-Meyer expansion was computed for the supersonic region of the airfoil. A comparison of these curves shows that the pressure ratios given by the Prandtl-Meyer expansion are somewhat lower than the corresponding pressure ratios shown by the test data. There is a very good agreement, however, in the shapes of the curves and the chordwise position of the peak pressures. This agreement is remarkable in view of the fact that the Prandtl-Meyer expansion is based on the assumptions that no boundary layer exists on the airfoil and that the sonic flow field extends from the airfoil surface to infinity. The magnitude of the experimental peak pressure corresponds to approximately 80 percent of the calculated Prandtl-Meyer increment, whereas at about the 25-percent-chord position the experimental pressure corresponds to approximately 40 percent. An analysis of the local supersonic region of NACA airfoil sections tested up to Mach numbers of approximately 0.9 has been made by Nitzberg and Sluder in reference 5. It was shown that values of the Prandtl-Meyer increments from 40 to 60 percent, depending upon the conditions at the beginning of the sonic region, occur on the forward parts of the airfoils. A comparison of the calculated value of pressure drag, from the experimental data presented in figure 6, with corresponding data obtained from a freely falling body shows good agreement.

Data from the Langley 24-inch high-speed tunnel for NACA 16-series airfoil sections (reference 6) indicate that the camber for best lift-drag ratio L/D decreases rapidly as the Mach number increases beyond the point of force divergence. Figure 7 presents typical results for the NACA 16-X09 airfoil family at $M = 0.775$. For this particular speed best L/D at $c_l = 0.5$, for example, was obtained with a section cambered for a design lift coefficient c_{l_i} of only 0.2. The results indicated that at somewhat higher Mach numbers best L/D would probably occur with zero camber. Reduction in camber also reduced the angle-of-attack variations required to maintain a given lift coefficient throughout the transonic-speed range.

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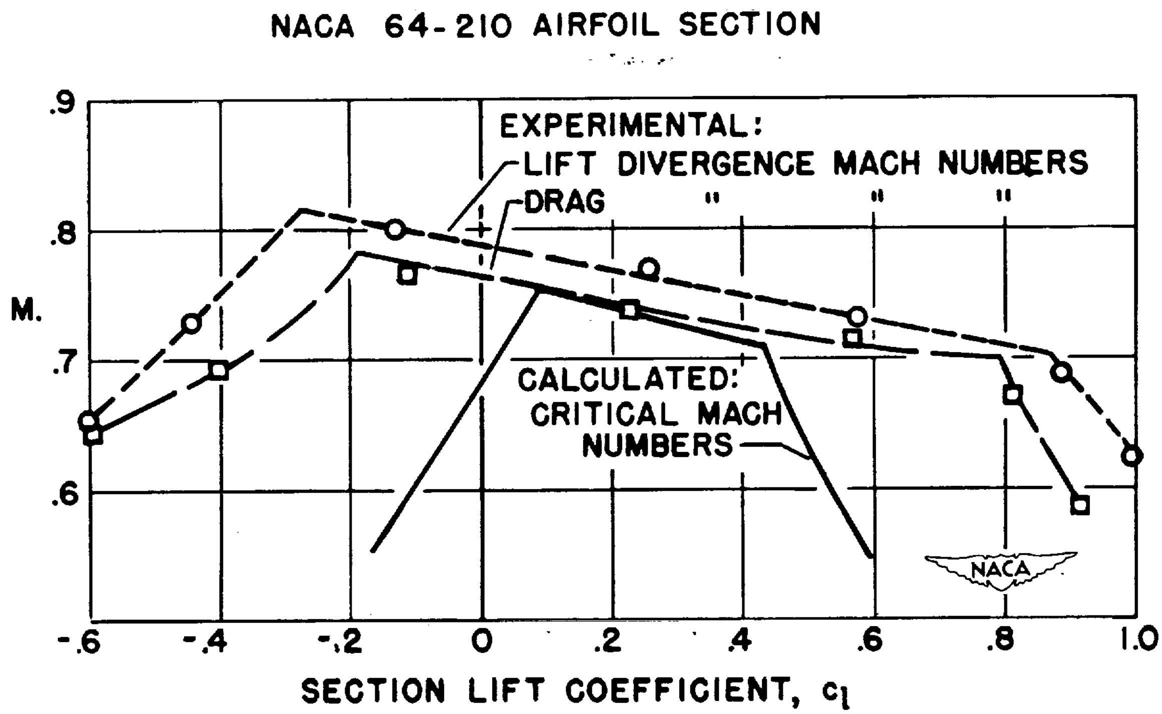

Figure 1.- Variation of the force-divergence and critical Mach numbers with section lift coefficient for the NACA 64-210 airfoil.

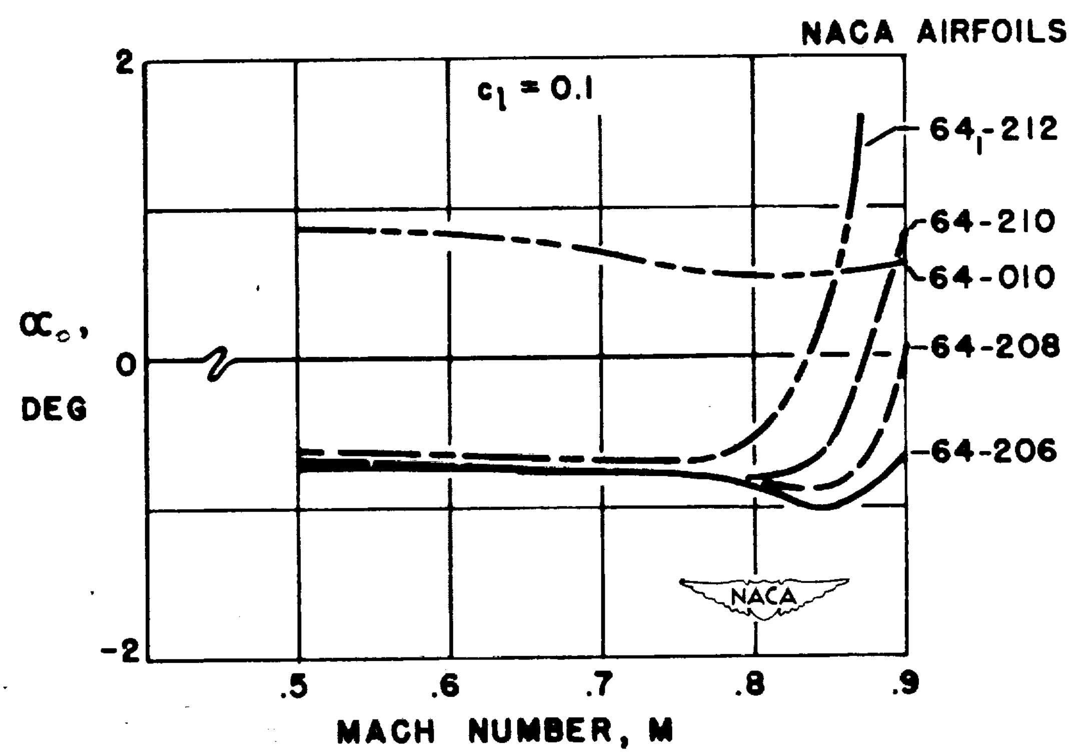

Figure 2.- Variation with Mach number of the angle of attack required to maintain a section lift coefficient of 0.1 for several NACA 64-series airfoils.

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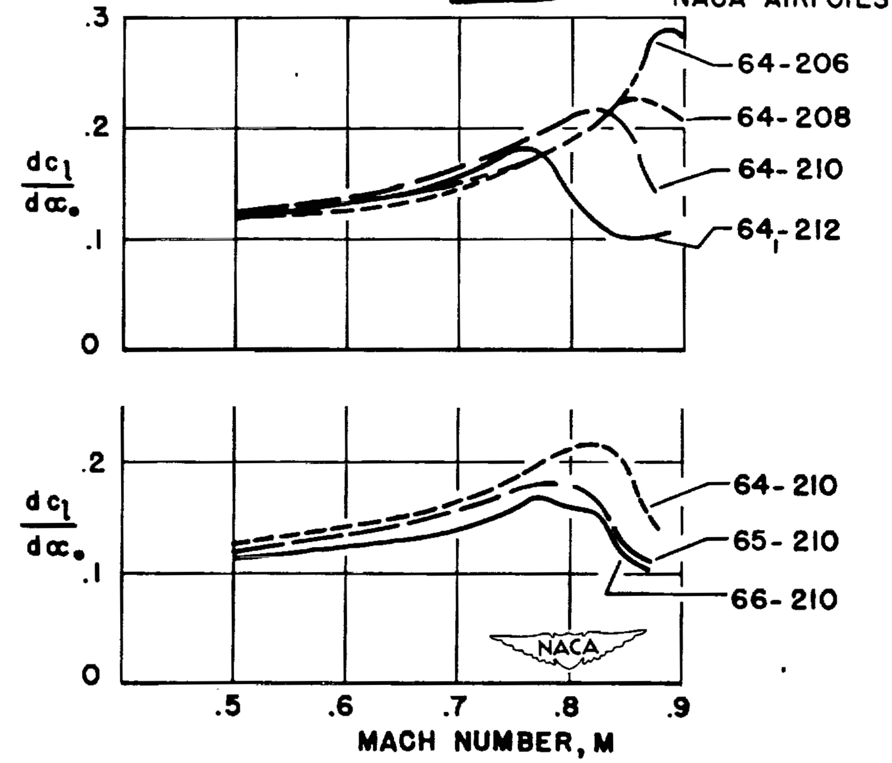

Figure 3.- Variation with Mach number of section lift-curve slope per degree for several NACA 6-series airfoils.

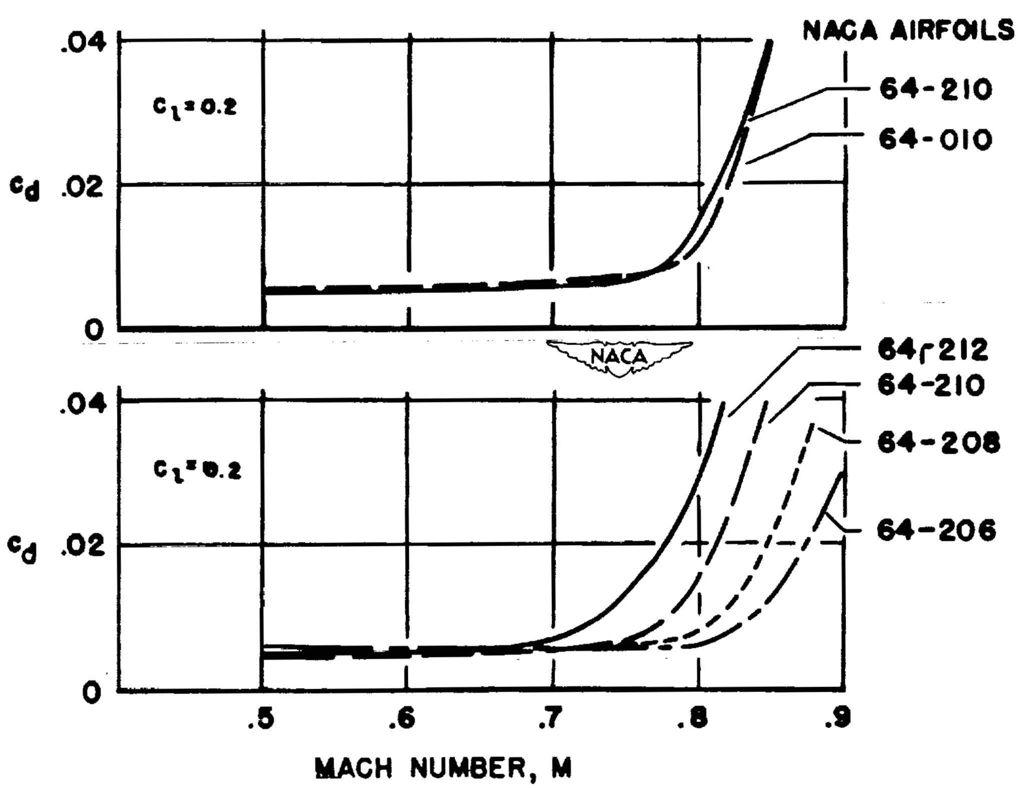

Figure 4.- Variation with Mach number of section drag coefficient at a lift coefficient of 0.2 for several NACA 64-series airfoils.

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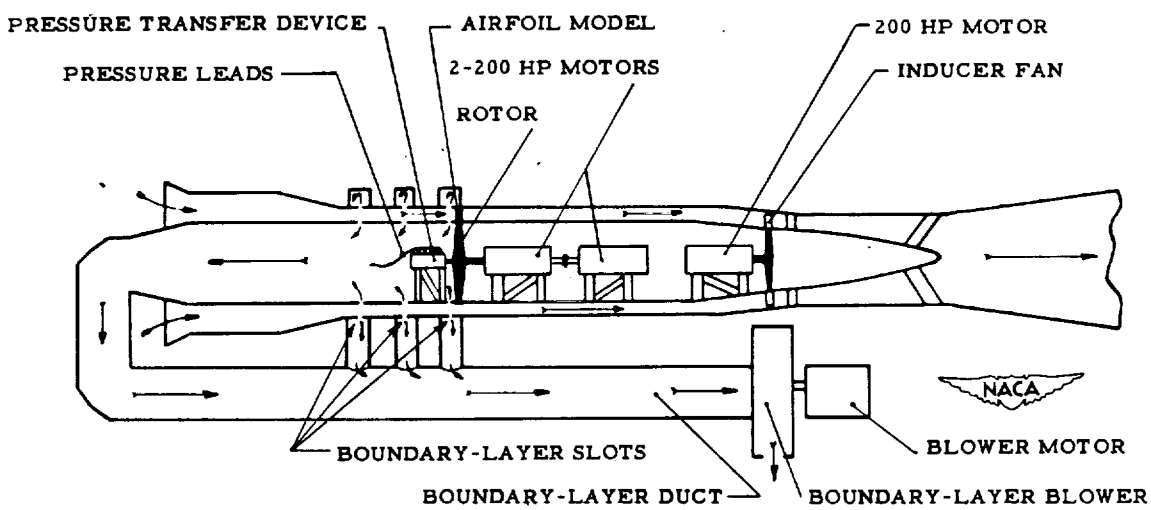

Figure 5.- Schematic diagram of the Langley transonic tunnel.

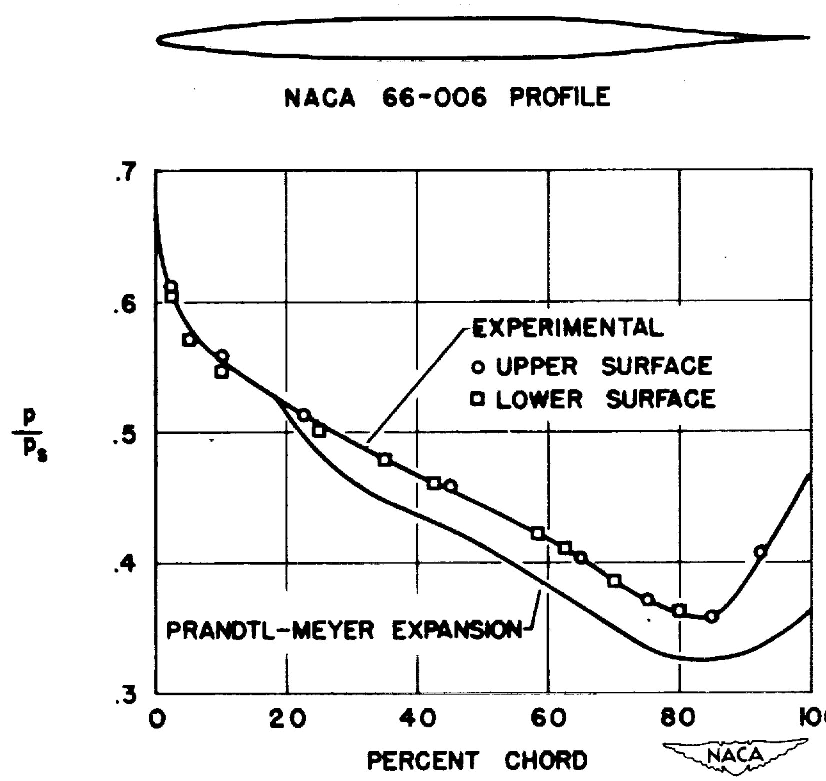

Figure 6.- Preliminary pressure-distribution data obtained in the Langley transonic tunnel at a Mach number of approximately unity for the NACA 66-006 airfoil at zero angle of attack.

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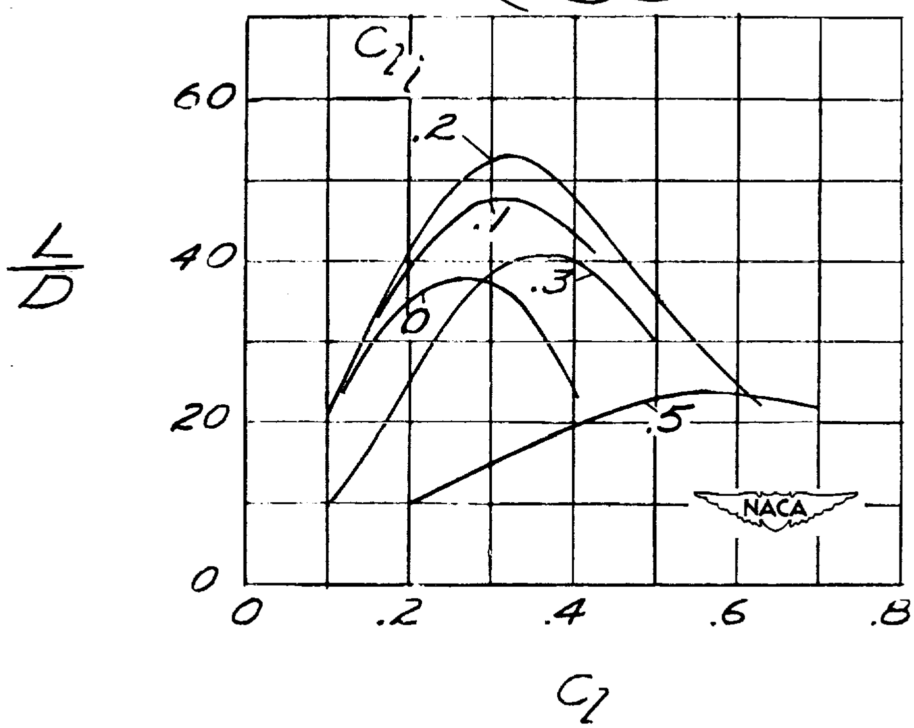

Figure 7.- Lift-drag ratios for NACA 16-X09 airfoils at M = 0.775.

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