HIGHLIGHTS FROM THE HISTORY OF AIRFOIL DEVELOPMENT

By R. T. Jones (EAA 114934)
Senior Staff Scientist
Ames Research Center, NASA
Moffett Field, CA 94035

OUR UNDERSTANDING OF aerodynamic phenomena is complicated by the fact that the equations of fluid motion give several possible answers to a given problem (Fig. 1). Probably for this reason prominent scientists of the nineteenth century (e.g., Rayleigh, Helmholtz, Kirchoff) gave incorrect solutions for the lift and drag of airfoils. As a model for the lifting airfoil they assumed a thin flat plate at an angle of attack. To avoid excessively high velocities at the sharp edges they further assumed that the flow separated, leaving both the leading and trailing edges in a tangential direction (see Fig. 1b). This model gave a region of "dead" air above the wing and, because the dead air gave no lift, all the lift came from the excess of positive pressure on the underside of the airfoil. Moreover, the resultant force was always at right angles to the chord plane of the airfoil and the lift was always accompanied by a sizable drag force. We now associate this kind of flow with the stalled condition of the airfoil; it is not surprising that the prospects of powered flight seemed rather dim in the light of this theory.

It was not until experiments such as those of Langley, Lilienthal, and the Wright brothers showed more favorable values of lift and drag that a reasonably correct basis for the understanding of airfoil behavior was devised. In 1902, W. M. Kutta calculated the lift of a thin, cambered plate at zero angle of attack and obtained a substantial lift force without drag in a frictionless fluid. In 1906, a theory of airfoils having rounded leading edges and varying angles of attack was developed by N. E. Joukowski. In the Joukowski theory the flow was assumed to cling to the airfoil around the leading edge but to separate in a tangential direction from the trailing edge. The formulas of Kutta and Joukowski showed a substantial lift force at right angles to the flight direction and no pressure drag. Figure 2 shows a Joukowski airfoil and its pressure distribution as derived by a mathematical transformation from a circle. The formulas for the airfoil shape and its pressure distribution can be found in textbooks on aerodynamics and are sufficiently simple to be programmed on a pocket computer (e. g., HP 25 or 67).

The prediction that an airfoil should be able to develop lift without pressure drag in two-dimensional flow is surprisingly close to reality. Figure 3 shows a smooth NACA 64-421 airfoil section at an angle of attack of 5^0 compared with a circular wire having the same drag, in pounds, at the same airspeed. The drag coefficient of the wire is about 1.0, and that of the airfoil is 0.006; hence, the diameter of the wire is only 0.006 the chord of the airfoil. At an angle of attack of 5^0 the airfoil develops a lift coefficient of 0.9 so that, for example, a lift force of 150 lbs. can be developed for a drag force of only 1 lb. These are experimental results at a Reynolds number of 6 million (see ref. 1). Inventors who seek to devise a more efficient means for supporting an airplane must take these results into account.

The airfoil shown in Figure 3 has a thickness-to-chord ratio of 21% and yet its drag hardly exceeds the skin friction. Early airplanes, such as the Wright biplane and the Bleriot monoplane, had extremely thin wings; as a consequence, they required lots of bracing. Figure 4 shows the Eiffel 36 which was used on the

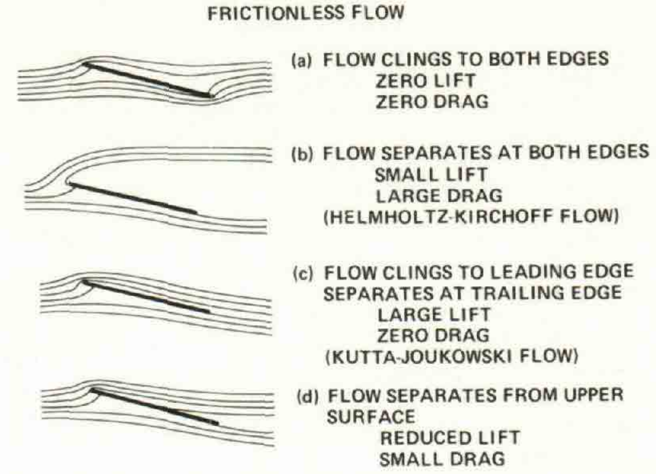


FIGURE 1 — Variety of flow patterns given by hydrodynamic theory.

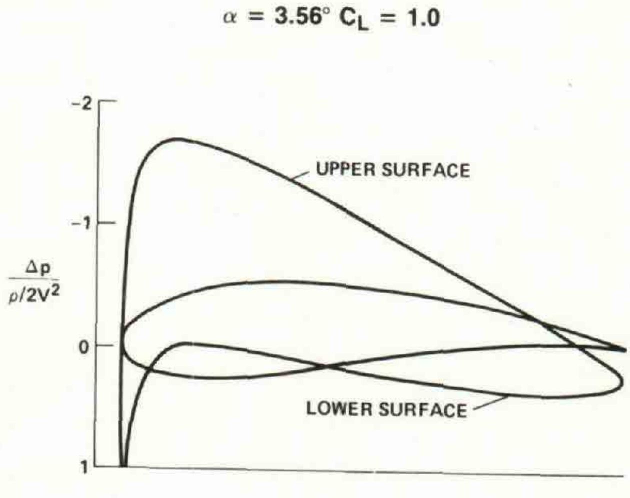


FIGURE 2 — Joukowski airfoil and theoretically derived pressure distribution.

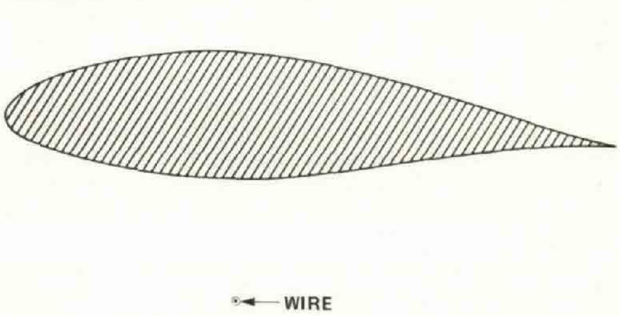


FIGURE 3 — NACA 64-421 airfoil compared with a circular wire having the same drag.

Jenny. Why did the early builders consider it necessary to use such thin sections? I believe the answer can be found in the small scale of the experimental apparatus available at that time. If we examine the behavior of airfoil shapes at very low Reynolds numbers, comparable to those available in the Wright brothers' small wind tunnel, we find that modern thick shapes do not work very well and the best shape resembles a thin, cambered plate.

Figure 5 shows a comparison of three shapes tested at Reynolds numbers of 40,000 and 120,000 (see ref. 2). A Reynolds number of 40,000 corresponds (in sea level air) to a wing chord of 3 in. at 17 mph. At this low scale, the thin flat plate is superior to the N-60 airfoil and the thin, cambered plate is better than either. At a Reynolds number of 120,000, the N-60 airfoil, which is 12% thick and similar to the Clark Y, begins to show its superiority.

The relatively poor performance shown by conventional thick airfoils at these low Reynolds numbers results from the fact that the flow tends to remain laminar for long distances. Such a laminar boundary layer, while it has low skin friction, shows very little ability to flow against a region of increasing pressure. In frictionless flow the velocity increases on approaching the region of reduced pressure on the upper surface of the airfoil and just enough kinetic energy is acquired to negotiate the increase of pressure toward the trailing edge. In the boundary layer, however, the velocity is kept low by friction and if the boundary layer remains laminar it will separate almost immediately on encountering a region of increasing pressure. The result is that the flow separates in going around the nose of the airfoil, thus leaving a region of dead air on the upper surface, similar to that assumed in the earlier theories (Fig. 1b). At a Reynolds number of about 20,000 the laminar separation bubble may extend over the whole width of the airfoil. As the speed is increased, however, the bubble shrinks in size and is followed by a turbulent flow which clings to the airfoil. At Reynolds numbers of about 1 million, the laminar bubble becomes too small to detect easily.

During World War I, aeronautical laboratories sprung up in different parts of the world and hundreds of shapes were tested. Figure 6 shows U.S.A. airfoil No. 1, based on a design by Col. V. E. Clark, and tested at 30 mph. in the Massachusetts Institute of Technology wind tunnel. Later, Clark introduced the X, the Y, and the Z airfoils, all three quite similar although the Y seemed marginally superior. Among the U.S.A. airfoils, to find practical application, the U.S.A. 35-B still provides reliable support for hundreds of Taylor and Piper Cubs.

Beginning in 1921, NACA began collecting airfoil data from laboratories around the world and presenting them in a uniform notation. These reports were entitled "Aerodynamic Characteristics of Airfoils" and ultimately included test results on more than 800 different shapes (see NACA TR's 93, 124, 182, 244, 286, and 315). Although the conditions of the tests varied considerably from one laboratory to another, this collection provided valuable assistance to aircraft manufacturers.

During this period, the design of airfoils was largely intuitive and based on the experience gained in testing numerous variations. While the experimenters were thus occupied, attempts were being made to extend the theory of Kutta and Joukowski to cover a greater variety of shapes. Thus, R. von Mises, a well-known aerodynamicist, was able to extend Joukowski's theory to cover an infinite series of profiles, all obtained by transformation of a circle. Also, H. Glauert, and T. von Karman obtained generalizations of the Joukowski airfoils. These

FIGURE 4 — Eiffel 36 airfoil as used on the JN-4 airplane.

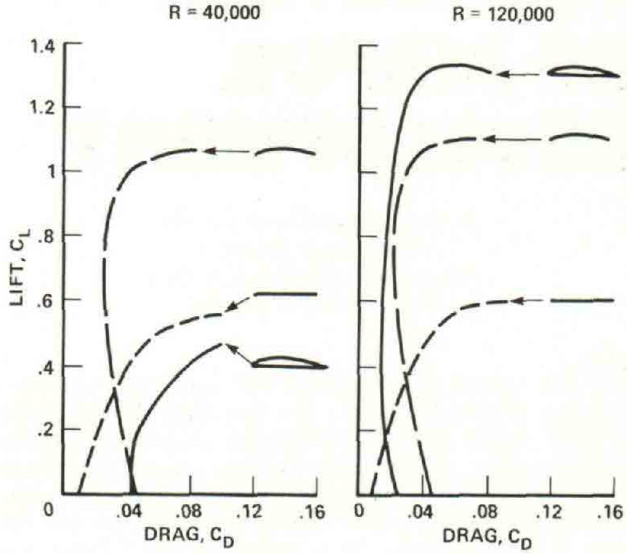


FIGURE 5 — Airfoil shapes at low and high Reynolds numbers.

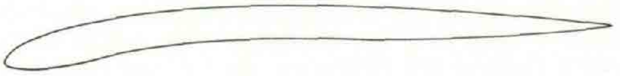


FIGURE 6 — U.S.A.-1 airfoil.

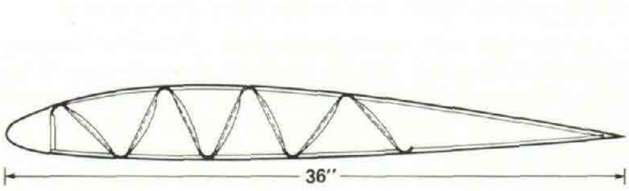
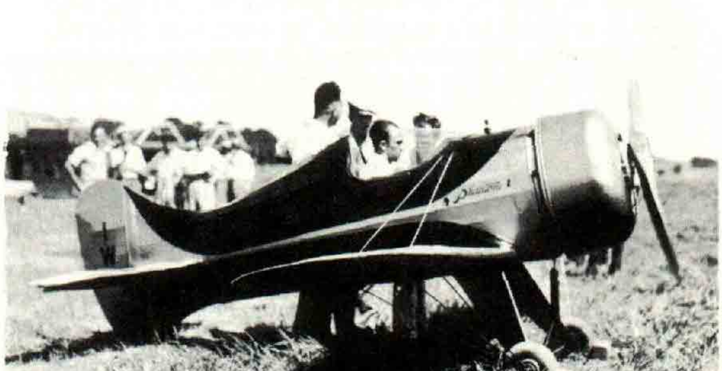


FIGURE 7 — NACA M-12 wing section on Pobjoy Phantom.



The Phantom I, powered by a Pobjoy geared radial engine. It used a M-12 airfoil.

(The Lightplane by John Underwood)

theories gave reasonable predictions of the lift, moment, and pressure distribution but, being based on frictionless flow, they could not predict either the small but extremely important drag or the maximum lift. It can be added that experiment did not predict these quantities very well either, since they varied rather widely with Reynolds number and with the quality of flow in the wind tunnel.

A new direction in experimental technique appeared when Max Munk of NACA Langley proposed the variable density wind tunnel (see NACA TN 60, 1921). One of the more surprising predictions of the kinetic theory of gases is that the viscosity of a gas is not increased by compressing it to a higher density. Since the Reynolds number (RN) involves the ratio of density to viscosity,* high Reynolds numbers could be obtained in a small wind tunnel by compressing the air to a smaller volume and a higher density. The construction of the variable density tunnel at Langley began soon after, and it became possible to test small models at a Reynolds number of 3 million, corresponding approximately to a 3-ft. chord at 100 mph.

Soon after the variable density tunnel, Munk developed his "thin airfoil theory." Although less accurate, this theory was a great simplification of earlier theories and permitted the relations between shape, pressure, and lift to be seen more clearly than before. With thin airfoil theory it was relatively easy to design airfoils that had a fixed or even a stable center of pressure. A systematic series of these, known as the M sections, were tested in the variable density tunnel at Langley in the 1920s. Of these, the M-6 and the M-12 found application; for example, on the Waco Taperwing. I used the M-12 on a small racing airplane (Pobjoy Phantom, Fig. 7) which flew in the 1930 National Air Races. Another important result of the thin airfoil theory, brought out by Glauert, is the magnifying effect of a plain trailing-edge flap when used as a control surface. Thus, on the basis of area alone, we might expect a 20% chord flap to have effectiveness of 20%. According to the theory, however, its effectiveness is more than 50%. For small deflection angles this is confirmed by experiment, provided there is no leakage through the gap.

An important aspect of airfoil behavior which was missed by the early Helmholtz-Kirchoff theories is the forward direction of the chordwise force, referred to sometimes as "leading-edge suction." Some early airplanes, such as the JN-4 and the Standard, were equipped with strong cables, called "drag wires," bracing the wings to the nose of the fuselage. It was observed in flight that these wires became slack during rapid pullouts showing that the wings tended to pull forward when the angle of attack was increased. This did not mean, of course, that the wing had negative drag but simply that the resultant of the lift and drag forces was inclined forward relative to the chord plane of the airfoil.

Figure 8 shows a plot made by R. M. Pinkerton of measured pressure forces around an NACA 4412 airfoil. Here the individual pressure forces are plotted in their true directions, perpendicular to the airfoil surface. At an angle of attack of 16° there is a high velocity flow around the nose and consequent large negative pressures which act in a forward direction. In spite of this forward suction, the resultant of all the pressure and friction forces lies behind a line perpendicular to the wind direction but ahead of a line perpendicular to the chord.

The forward-chord force, or leading-edge thrust, is extremely important in maintaining low drag at high lift coefficients. Thus, in the case of a wing of aspect ratio 10 at a lift coefficient of 1.0, reducing the camber or sharpening the leading edge to eliminate the forward-chord force could increase the drag by a factor of 4.

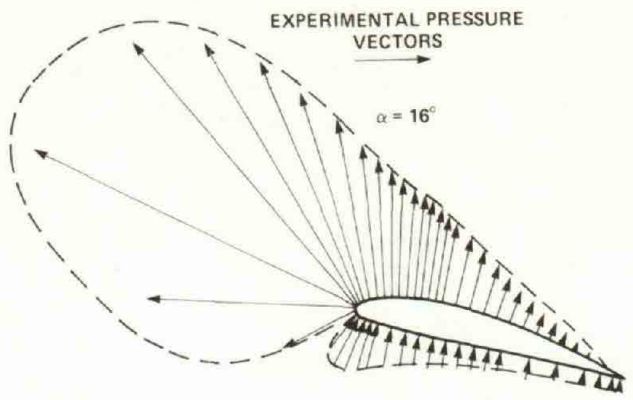


FIGURE 8 — Pressure-vector diagrams for the NACA 4412 airfoil at 16° angle of attack.

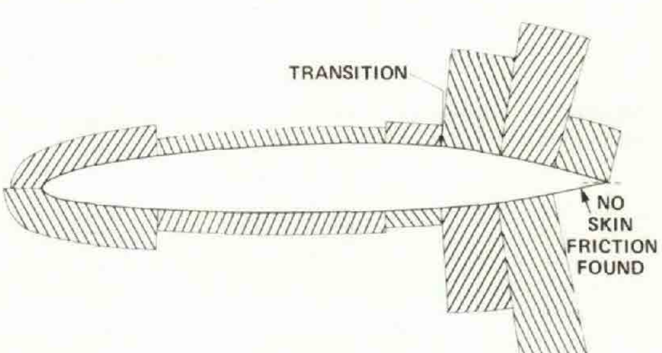


FIGURE 9 — Distribution of skin friction over surface of laminar flow airfoil (NACA 27-212).

The problem of determining the pressure distribution over a thick airfoil of arbitrary shape was finally solved in a rather definitive way by Theodore Theodorsen (see NACA TR 411, 1931). By combining Theodorsen's method with a correction for the boundary layer due to G. I. Taylor, the pressure distribution over any reasonable airfoil shape can now be calculated with a high degree of accuracy in the lower range of Mach numbers.

Up to this point, airfoil theories had been based on the assumption of frictionless flow. At Reynolds numbers of 1 million or so the normal pressures may be expected to dominate over the friction and hence this seems a reasonable assumption. However, such a theory cannot predict the small but very important drag force, nor the maximum lift. The next step in the progress of airfoil development came when the relation between the pressure distribution and the friction of the boundary layer was studied more carefully.

Early experiments of Reynolds (1883) had shown that laminar motion in a pipe tends to become unstable and turbulent when the parameter $\rho VC/\mu$, now known by his name, exceeded a certain value. Subsequently, both laminar and turbulent boundary layers were studied extensively by L. Prandtl (who originated the boundary-layer concept) and his students at Goettingen in Germany. It was known, for example, that at a Reynolds number of 6 million the skin friction of a turbulent boundary layer is more than 3 times that of a laminar layer. Furthermore, the laminar layer tended to be more stable

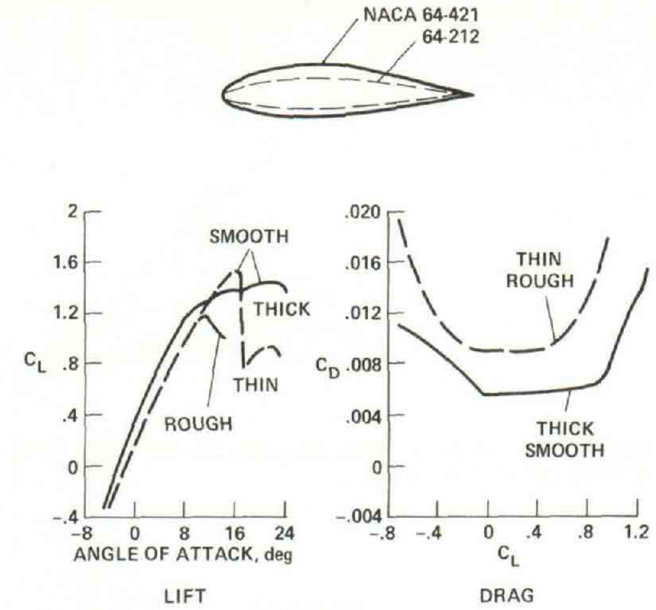
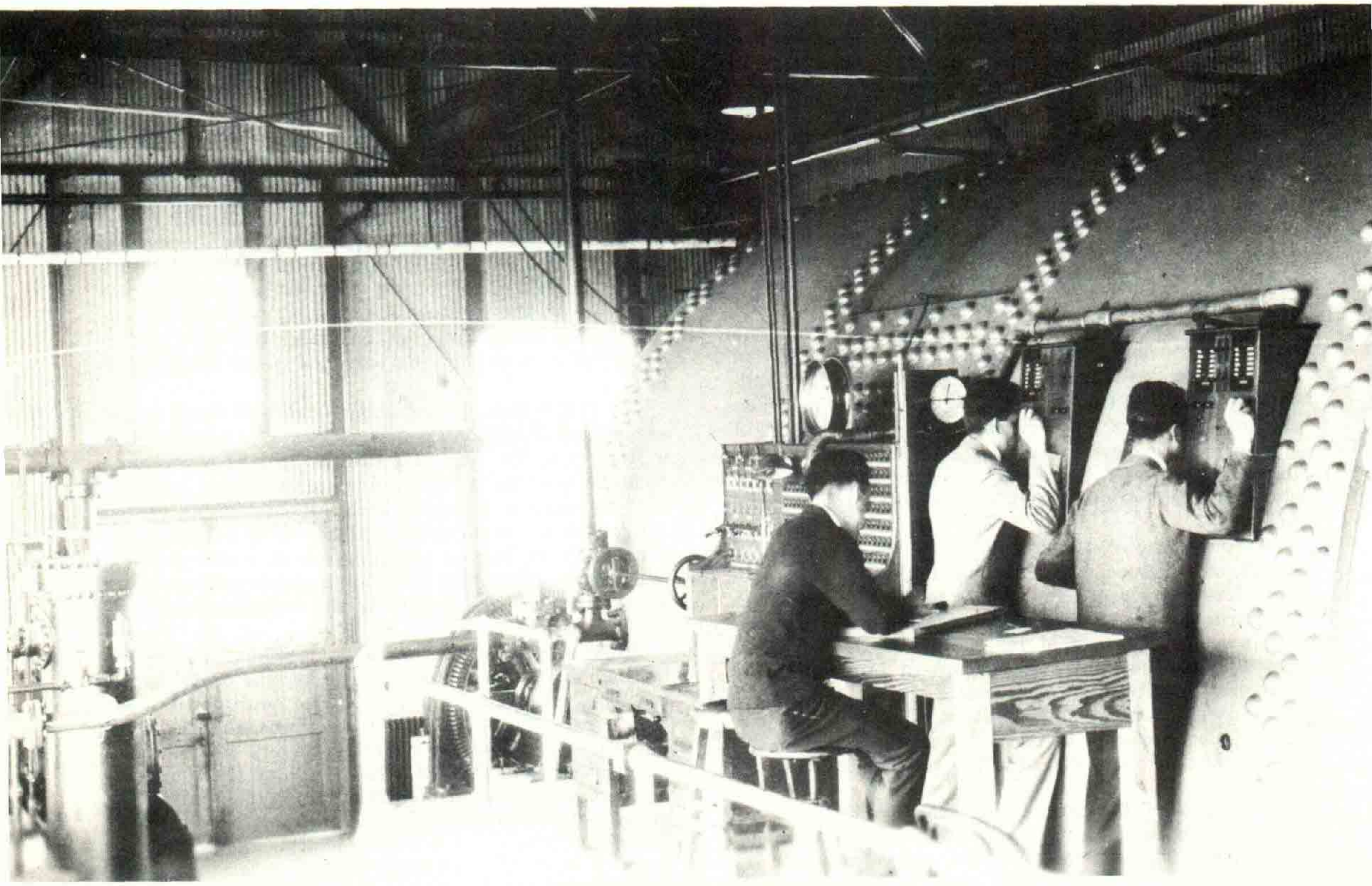


FIGURE 10 — Effect of surface finish on airfoil drag and lift (data from ref. 1).

when flowing in a region of falling pressure, that is, in a "favorable" pressure gradient. B. M. Jones in England had observed that the drag of airfoils measured in flight in smooth air varied considerably with the extent of laminar flow.

In the early 1930s, Eastman Jacobs at Langley Field decided to put this knowledge to work by designing airfoil shapes that would definitely promote stable laminar flow for considerable distances along the wing chord. Of course, one has to be careful here since too much laminar flow will lead to separation and poor L/D ratios, as shown by the early small model tests. We need to develop reduced pressures over the upper surface of the airfoil to provide lift; however, the flow must come back to a positive pressure at the trailing edge or else a large pressure drag will develop. Typically, a boundary layer with strong turbulence will permit 60 to 70% pressure recovery, while a laminar layer separates after only 10% recovery. Although the turbulent layer has much higher skin friction, we need the turbulence at the rear of the airfoil to recover pressure. The laminar flow airfoil thus has gradually falling pressures over a part of the upper surface followed by a region of increasing pressure where it is expected that the flow will become turbulent.

Figure 9 shows the distribution of skin friction over a laminar flow airfoil (NACA 27-212) as measured by I. H. Abbot and Harry Greenberg at Langley in 1939. The height of the blocks gives the total increment of drag between two points on the surface and hence gives only a rough picture of the distribution. In this rather extreme case the flow remained laminar back to about 70% chord. The skin friction at the nose is high because the laminar layer is thin there. The large increase of skin friction following transition to turbulent flow is evident at the rear of the airfoil. It appears that the flow separated from the lower surface at about 90% chord since no skin friction was found there. Although some friction is saved here the increase in pressure drag would undoubtedly more than compensate.



NACA Variable Density Wind Tunnel.

Theodore Theodorsen

N. E. Joukowski

Eastman Jacobs

Max Munk

Early low drag airfoils were designed to maintain laminar flow back as far as 70% of the chord, and drag values less than half those of conventional shapes were measured. However, such extensive laminar runs are difficult to maintain in practice, especially at high Reynolds numbers where the boundary layer becomes sensitive to small surface imperfections. The maintenance of laminar flow over 50% of the chord at a Reynolds number of 6 million requires a surface finish and smoothness about comparable to that of a new automobile. Anyone who has examined airplane surfaces knows that such finishes are not the usual practice in the aircraft industry.

It is generally believed that thinner sections have smaller drag; they do, but differences in surface smoothness may outweigh the effect of thickness. Figure 10 shows lift and drag curves for a thin airfoil having sufficient roughness to produce turbulent flow over most of its surface compared with those for a thick airfoil having a smooth surface. Further details concerning the degree of smoothness required to obtain these low drag values can be found in the book by Abbott and Von Doenhoff (ref. 1).

The danger inherent in the behavior of airfoils at high angles of attack was recognized very early. The loss of control and entry into a spin were traced to the reversal of the lift-curve slope beyond the stall. In the late 1920s a concerted effort was made to find shapes that would have gentler stalling characteristics — preferably a lift curve that did not drop after the stall. At one point it was believed that the objective had been achieved. An airfoil tested at Langley in 1928 showed a remarkably flat-topped lift curve. Unfortunately, on checking the setup it was found that the phenomenon was not aerodynamic, but mechanical — the wind-tunnel balance had reached its stop.

No satisfactory theory for the behavior of airfoils at or beyond the stall has yet been developed but some empirical rules have emerged from extensive testing. First of all, the phenomenon seems to be distinctly three dimensional and also unsteady with considerable buffeting so that the usual plot of section curves against angle of attack gives only a limited picture of the actual behavior in flight.

Thick, cambered profiles show a progressive separation beginning at the trailing edge and moving forward as the angle of attack is increased. Thus, the thicker laminar-flow profiles, such as the 64-418 and the 64-421, show rather gentle stalling characteristics combined with a wide range of low drag in the smooth condition. Thinner profiles of this series have smaller nose radii and a rather abrupt breakaway of flow from the leading edge. Figure 10 shows the difference in behavior of the 64-212 and the 64-421 at a Reynolds number of 3 million. In some instances it has been found desirable to modify the thinner laminar-flow profiles as suggested in reference 3.

Recently, a new series of airfoils with exceptionally high values of the maximum lift coefficient has been introduced by R. W. Whitcomb who is well known for his high-speed "supercritical" wing and for the transonic "coke bottle" shape for the fuselage. Both the 13% thick and the 17% thick airfoils of this series can achieve a maximum lift coefficient of 2 without the necessity of a movable flap; hence they should find application to airplanes in the general aviation category.

In this brief account I have called attention to the remarkable efficiency of modern airfoil sections in what are now thought of as conventional shapes. Unfortunately, such high efficiencies appear to be limited to speeds considerably below the speed of sound. At Mach numbers of 0.7 or more, thinner profiles are needed to avoid shock waves. And at still higher speeds it becomes necessary to sweep the wings to recapture at least in part the efficiency of the straight airfoil of high aspect ratio.

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* RN = ($\rho VC/\mu$) where ρ is the air density, V the flight velocity, C the wing chord, and μ the viscosity. At V = 100 mph and C = 1 ft., the RN in sea level air is about 1 million — an easily remembered rule.