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SUMMARY

Measurements of the velocity profiles in the turbulent boundary layer on the upper surface of a wing of semielliptical plan form at an angle of yaw of 25° and angles of attack of 12° and 14° are reported along with pressure distributions. These show considerable spanwise flow near the surface. The chordwise (velocity components inclined 25° to the main wind direction) and spanwise boundary layers were calculated. The chordwise velocity profiles are in good agreement with those obtained at the National Bureau of Standards on an unyawed airfoil shape.

INTRODUCTION

The existence of a spanwise component of velocity near the upper surface of a yawed wing is easily demonstrable by means of tufts. This component persists over a wide range of angle of attack, reaching high values near the stall, and increases in magnitude with distance from the root up to a point near the tip. The spanwise flow is initiated by a spanwise pressure gradient arising from the displacement downstream of the pressure profiles. As a result, slower moving air in the boundary layer is transported outward; this causes an increase in boundary-layer thickness near the tip. The wing will therefore stall first near the tip unless a strong washout is introduced.

The study reported here, initiated in 1944, had for its object an experimental study of the details of the flow in the boundary layer at large angles of attack and sweepback. When the work was begun, a sweepback angle of 25° was considered large and most of the measurements were made with this condition.

Recent analyses (references 1, 2, and 3) have shown that for the laminar boundary layer on an infinite yawed cylinder the "chordwise" boundary-layer velocity profile, that is, the profile of the velocity component normal to the leading edge of the cylinder, is unaffected by yaw. Further, if the solution for the chordwise profile is substituted in the equation of motion for the spanwise flow, the resulting differential equation may be solved for the spanwise velocity profile. While this analysis holds strictly only for the laminar boundary layer on an infinite wing, the results reported here for a turbulent boundary layer on a yawed finite wing indicate that in this case as well the profile of the chordwise component is relatively unaffected by an angle of yaw of 25°.

This investigation was conducted under the sponsorship and with the financial assistance of the National Advisory Committee for Aeronautics.

APPARATUS AND METHOD

The investigation was carried out in the 8-foot open-throat wind tunnel at the University of Michigan. The airspeed was 50 miles per hour and the corresponding Reynolds number was 700,000 (based on a root chord of 18 inches).

The wing model used for the investigation was of semielliptical plan form and Clark Y section.[1] The semiaxes of the ellipse were 9 and $48\frac{1}{4}$ inches. The wing model was constructed of laminated wood. An end plate 40 inches in diameter was secured to the root section. A support was attached to the end plate and secured to the tunnel floor so that the wing was in an upright position, as shown in figure 1.

A hot-wire instrument was adapted for measurements of velocity in magnitude and direction. A hot-wire probe in the shape of a V pointed into the wind was used and was calibrated as follows: The potential drop across each arm of the V was measured, by means of a potentiometer circuit, at a series of angles (in the plane of the V) and airspeeds; plotting the sum of the two potential drops against their difference permits the drawing of lines of constant speed and direction such as are shown in figure 2. The calibrations were carried out in the undisturbed stream, after which the probe was mounted on a micrometer screw which projected through the wing, and traverses of speed and direction through the boundary layer were taken. At some positions near the trailing edge it was necessary to traverse the boundary layer in two steps because the total change in wind direction exceeded the range of the hot-wire instrument for a given orientation. The instrument was

[1]The Clark Y section was chosen because it is relatively easy to construct and because it seems probable that the effects observed would be the same, within the experimental error, for any conventional airfoil profiles of the same thickness. Low-drag sections, in which a considerable area of laminar flow exists, would undoubtedly show different effects at low angles of attack.

oriented at a large angle to the potential-flow direction for measurements close to the surface, then reoriented at a smaller angle for the traverse of the remainder of the boundary layer. Calibrations were made before and after each traverse and the run was discarded if the calibrations deviated from each other by more than 3° in direction or more than 5 feet per second in airspeed. These values indicate the approximate limits of the accuracy of the results.

The measurements reported here were made at the positions shown in figure 3 for a sweepback angle of 25° at angles of attack of 12° and 14°. Observations were made at other angles of attack and for angles of sweepback of 0° to 30° but these were not extensive enough to permit conclusions to be drawn and are therefore not included in this report.

Preliminary observations included tuft observations and an investigation of boundary-layer transition at 25° sweepback by the method of reference 4. A mixture of camphor and ether was sprayed on the upper wing surface and after it had dried the wind was turned on. Because of the greater rate of evaporation of the camphor in a turbulent boundary layer, the line of demarcation between camphor layer and clean surface is taken to be the line at which transition takes place. Observations were visual since the contrast was not great enough for photographic methods.

After the hot-wire measurements were completed, copper tubes were embedded in the wing flush with the upper surface and extending in the spanwise direction. Holes were drilled in the tubes at 11 spanwise stations so that the pressure profile could be measured at any station by closing the other 10 holes with airplane-model glue. The pressures were measured on a multiple manometer. Pressure distributions at angles of attack of 12°, 14°, and 15° at an angle of yaw of 25° were measured and pressure gradients were calculated.

RESULTS AND DISCUSSION

The camphor-evaporation tests described in the preceding section indicated that, for the results given here (25° sweepback, 12° and 14° angles of attack), transition on the upper surface occurred well within the first 10 percent of the local chord from the leading edge. Hence the main scale effect on these results is, in all likelihood, that associated with the turbulent boundary layers.

The tuft photographs of figure 1 show the over-all character of the flow at 25° sweepback at angles of attack from 12° to 16.5°. Even at an angle of attack of 12° there is considerable spanwise flow near the trailing edge. This spanwise component becomes more and more pronounced as the angle of attack increases and some unsteadiness of the flow, as evidenced in the photographs by appreciable angular deviation between adjacent tufts, is noticeable at an angle of attack of 13°. This tendency continues and, at a 14° angle of attack, at least the tip portion appears to be stalled. The stalled area increases with angle of attack until at 16.5° it extends over the rear half of at least the outer third of the wing.

Figures 4 and 5 show spanwise and chordwise velocity profiles along the lines designated 1 and 2, respectively, in figure 3 for an angle of attack of 14°. As the trailing edge is approached, the chordwise profiles show qualitatively the change characteristic of a turbulent boundary layer in an adverse pressure gradient. The spanwise profiles indicate that the spanwise boundary layer is considerably thinner than the chordwise. On the other hand, the calculations of Sears and Wild (references 1 and 3, respectively) show that, for laminar flow over an infinite yawed cylinder, the two boundary layers are of approximately the same thickness. For a finite wing such as is used in the measurements reported here, there is a spanwise component near the wing due to the trailing vortex system. On the upper surface this component will be toward the root and will therefore oppose the outward spanwise flow due to sweepback. It will furthermore be variable with distance from the surface. Hence any comparison between spanwise velocity profiles for finite and infinite wings must take into account the trailing vortex system.

The change in flow direction through the boundary layer is quite large. The maximum change measured, for position 2-d of figure 5, was about 100°.

The chordwise profile of figure 5, position 2-d, shows a region of reverse flow near the surface. In a two-dimensional flow the point where the reverse flow begins, in general, marks the beginning of a turbulent wake in which the fluctuations are so high as to preclude measurements of mean-velocity profiles. In this case, however, while fluctuations were greater than nearer the leading edge, they were not especially troublesome. On the other hand, at the two outermost positions of figure 3 the fluctuations were too high to permit observations. These positions were therefore presumed to be in the turbulent wake; the pressure gradients of figure 6 bear out this presumption. Accordingly, the criterion for the point where the turbulent wake begins on a yawed wing does not appear to be clearly defined. The most reasonable criterion seems to be $\left(\frac{\partial|V|}{\partial y}\right)_{y=0} = 0$ where $|V|$ is the absolute velocity and y is the coordinate normal to the surface.

According to the work of Von Doenhoff and Tetervin (reference 5) on the characteristics and separation of the two-dimensional turbulent boundary layer, the velocity profiles plotted against y/θ constitute a one-parameter family, depending only on $H = \delta^*/\theta$, where δ^* and θ are, respectively, the displacement and momentum thickness of the boundary layer. For laminar flow over a yawed airfoil, Wild (reference 3) calculated δ^*_x, δ^*_z, θ_{xx}, and θ_{zz}, where x and z are, respectively, the coordinates in the chordwise and spanwise directions. They are defined as follows:

$$\delta^*_x = \int_0^\delta \left(1 - \frac{u}{u_1}\right) dy$$

$$\delta^*_z = \int_0^\delta \left(1 - \frac{w}{w_1}\right) dy$$

$$\theta_{xx} = \int_0^\delta \left(1 - \frac{u}{u_1}\right) \frac{u}{u_1}\, dy$$

$$\theta_{zz} = \int_0^\delta \left(1 - \frac{w}{w_1}\right) \frac{w}{w_1}\, dy$$

where u_1 and w_1 are, respectively, the chordwise and spanwise velocities at the outer edge of the boundary layer, and δ is the boundary-layer thickness.

Two sets of curves are shown in figure 7. The long-dashed curves were reported by Dryden (reference 6) on the basis of measurements on the two-dimensional flow over an airfoil shape at a high Reynolds number. The short-dashed curves were taken from the compilation by Von Doenhoff and Tetervin of a large number of measurements on airfoils (reference 5). It appears that experimental points for the yawed wing reported here agree better with Dryden's curves, though there is a slight systematic deviation from the curve at $y/\theta_{xx} = 1/4$. His curves extend only to $H = 2.8$; the results in figure 7 indicate that the extrapolation to $H_x = 3.2$ is well justified. The results indicate that for angles of yaw up to 25° the turbulent chordwise profiles are in good agreement with two-dimensional results. It is quite possible that appreciable systematic differences will occur at higher angles of yaw.

The two-dimensional results reported by Dryden and by Von Doenhoff and Tetervin indicated that separation occurred when H reached a value of 2.6 to 2.8. Figure 7 shows, however, that, for the yawed wing, separation has not yet occurred for $H_x = 3.18$. For the separated chordwise profile of figure 5, $H_x = 6.7$.

The values of δ^*_z, θ_{zz}, and H_z were calculated for a number of spanwise profiles but no systematic behavior could be detected. For instance, in figures 4 and 5 the values of these quantities are given along with δ^*_x, θ_{xx}, and H_x. While H_x shows a steady increase as the trailing edge is approached, no such consistent trend is shown by H_z. As was pointed out earlier, one would expect the trailing vortex sheet to have an influence on the spanwise velocities, so that if the experiment were carried out for an airfoil which spanned a tunnel it is possible that the results would show a systematic dependence on H_z.

Von Doenhoff and Tetervin (reference 5) developed a method, based on the Von Kármán integral relation and on empirical results for calculating the growth of the turbulent boundary layer on a two-dimensional wing. They calculated H and θ for several cases and found good agreement with experiment.

Although the method was worked out for the two-dimensional case, an attempt was made to apply it to the chordwise flow for the yawed finite wing. The faired measured values of H_x and θ_{xx} for the first station along each of four chordwise lines were taken as a starting point and the method was used to predict the values of H_x and θ_{xx} as functions of x. Near the root the agreement with experiment was fair but for other positions the deviation was marked. In all cases the predicted H_x increased much more rapidly with x than did the measured values, especially near the trailing edge.

For these calculations the pressure gradients shown in figure 6 were used. These vectors represent the pressure gradient in magnitude and direction for three angles of attack. The change in the gradients near the tip for angles of attack of 14° and 15° shows the change caused by tip stalling.

CONCLUSIONS

On the basis of the results of measurements of the velocity profiles in the turbulent boundary layer of a yawed wing and analyses of the results, the following conclusions are drawn:

1. The chordwise flow in the turbulent boundary layer of a finite wing at an angle of yaw of 25° may be expressed:

$$\frac{u}{u_1} = f\left(\frac{H_{x,y}}{\theta_{xx}}\right)$$

The functional relationship is very nearly the same as that reported in NACA Technical Note 1168 for the unyawed wing. The extrapolation of the curves given in NACA Technical Note 1168 from $H_x = 2.8$ to $H_x = 3.2$ in the present paper seems well justified.

2. The maximum measured value of H_x for an unseparated chordwise profile was 3.18. For the unyawed wing the measurements of NACA Report 772 and NACA Technical Note 1168 showed that separation occurred for values of H of 2.6 to 2.8.

3. The separation point for the chordwise profile does not necessarily mark the beginning of the turbulent wake for a yawed cylinder. The criterion for the beginning of the turbulent wake is probably $\left(\frac{\partial|V|}{\partial y}\right)_{y=0} = 0$, where $|V|$ is the absolute velocity.

4. The boundary layer for the chordwise flow is considerably thicker than that for the spanwise flow. The spanwise flow due to the trailing vortex system of a finite wing probably causes the distribution of this component to be highly dependent on the lift of the wing.

University of Michigan
Ann Arbor, Mich., April 22, 1949

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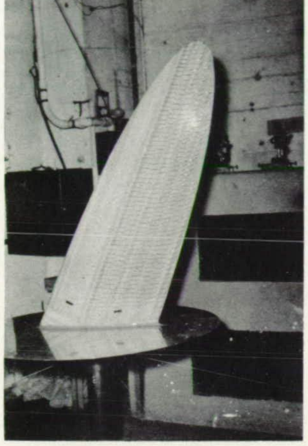

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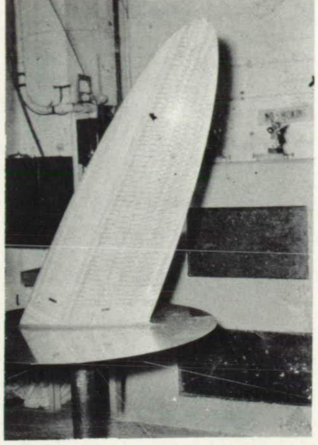

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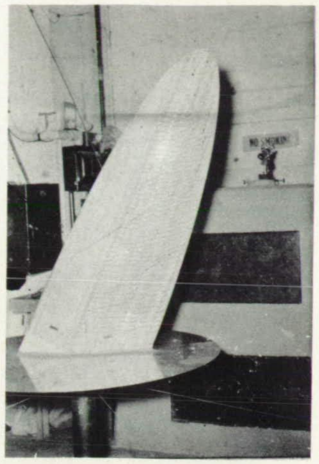

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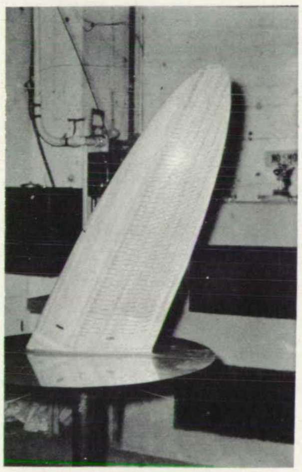

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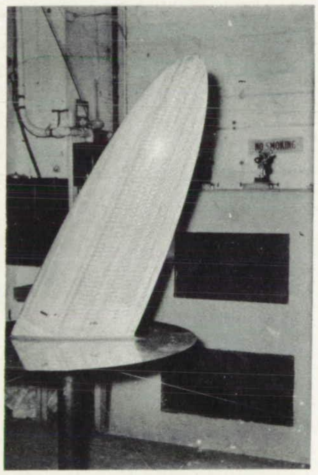

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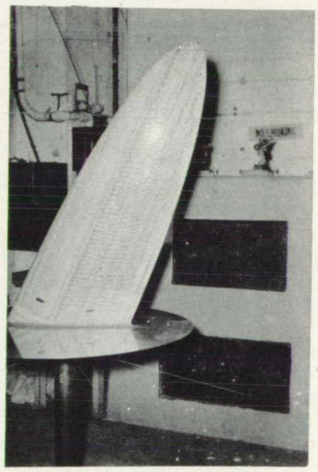

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Figure 1.- Photographs of wing model mounted in tunnel for tuft studies. Sweepback angle, 25°; various angles of attack α.

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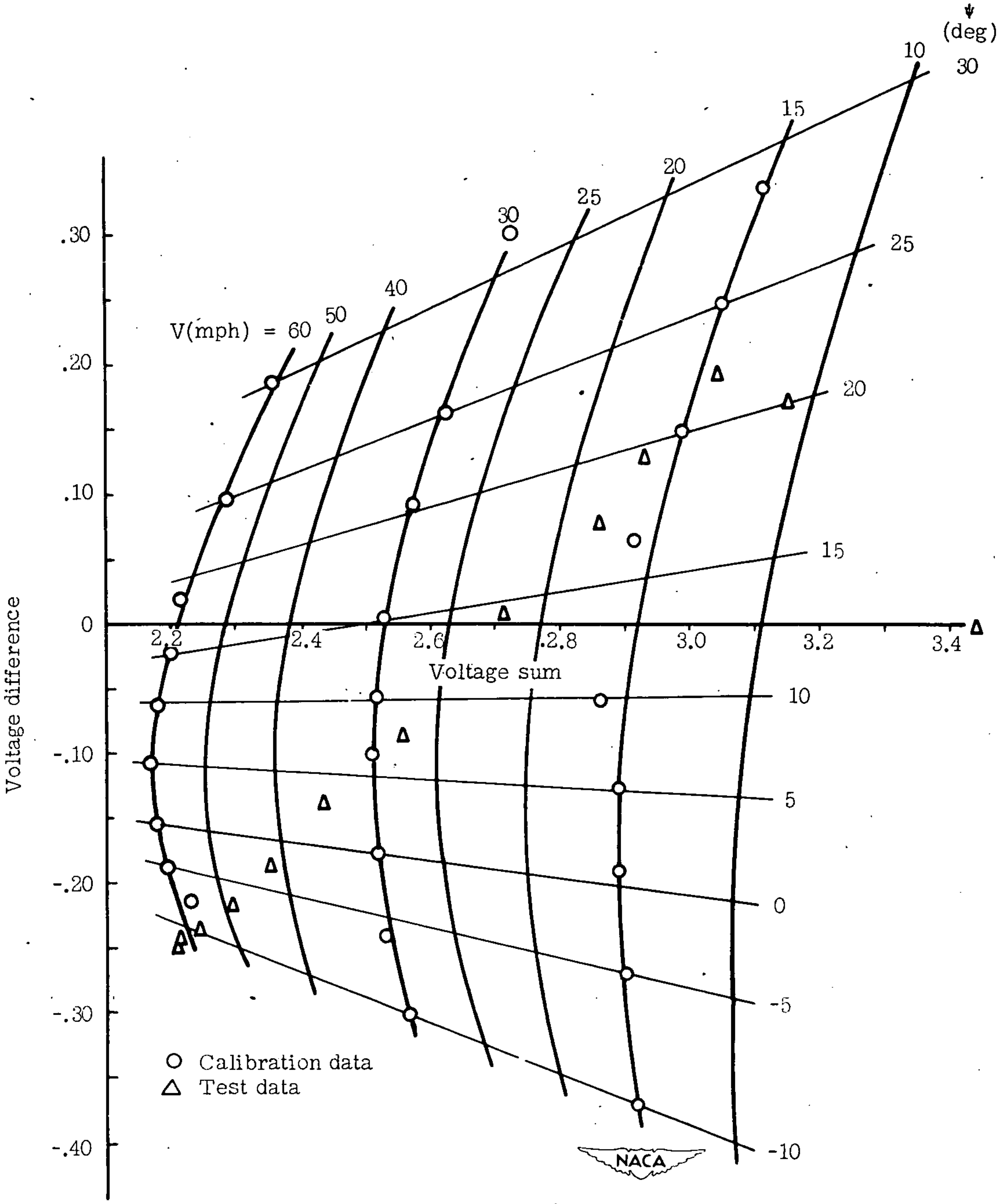

Figure 2.- Calibration curves of hot-wire unit. The abscissa is the sum of the voltage drops across the two arms of the V unit, the ordinate is the difference. Calibrations were made at various angles of yaw ψ at air-speeds of 15, 30, and 60 miles per hour. The curves for intermediate velocities are interpolated. Triangles represent the separate points of a traverse of the boundary layer. Positive values of ψ indicate outward flow; ψ measured from free-stream direction.

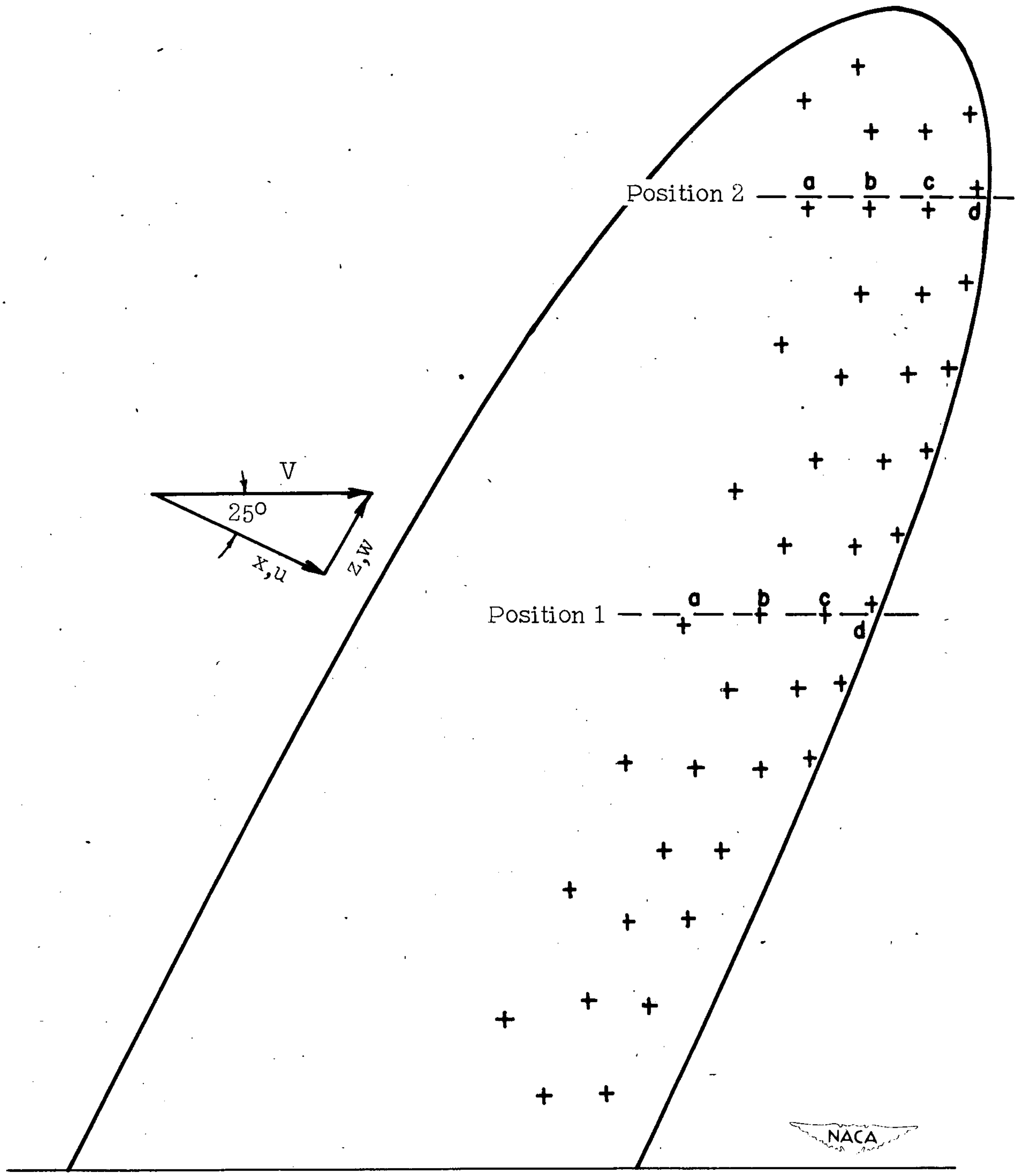

Figure 3.- Diagram of wing showing positions at which measurements were made. The system for the chordwise and spanwise directions is shown. The points at which measurements of velocity profiles were made are indicated by crosses. Sweepback angle, 25°; angle of attack, 12° and 14°.

Position	δ^*_x	θ_{xx}	H_x	δ^*_z	θ_{zz}	H_z
1-a	0.096	0.058	1.66	0.060	0.042	1.43
1-b	.172	.085	2.02	.067	.052	1.28
1-c	.246	.106	2.32	.057	.042	1.35
1-d	.390	.140	2.77	.142	.096	1.48

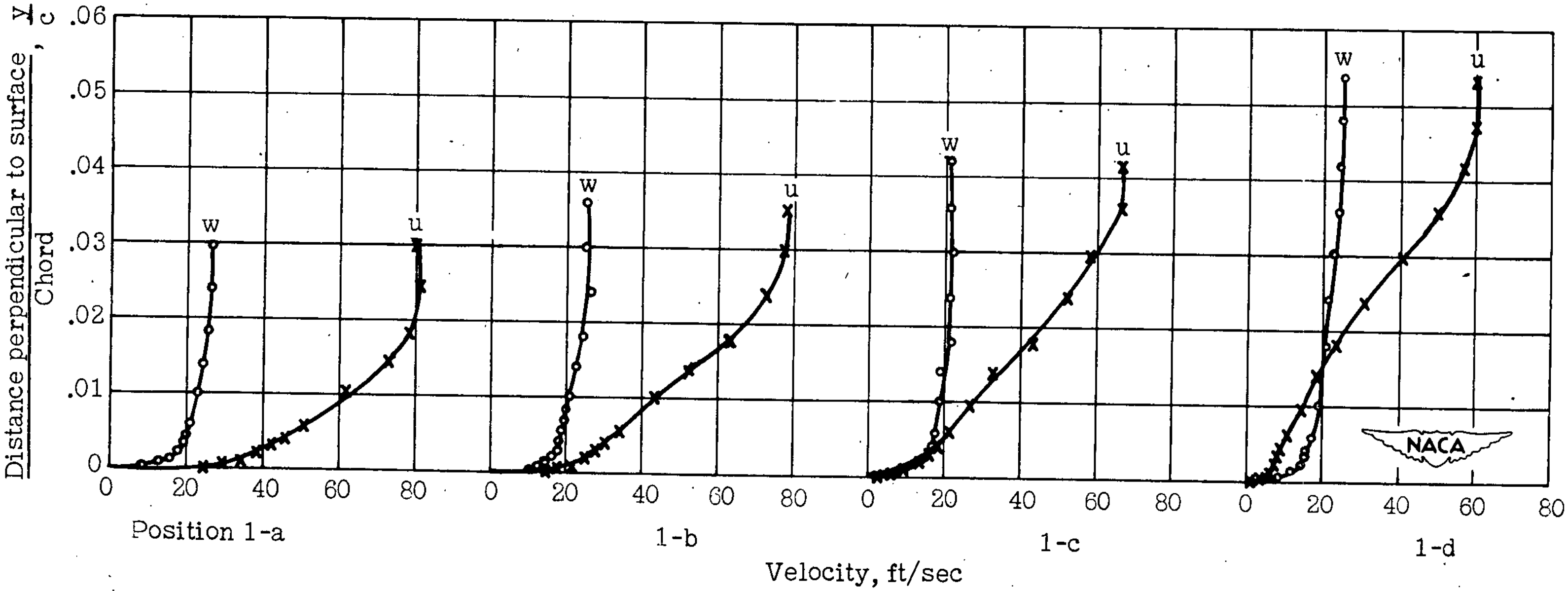

Figure 4.- Chordwise and spanwise velocity profiles along line 1 (fig. 3) for 25° sweepback and 14° angle of attack.

Position	δ^*_x	θ_{xx}	H_x	δ^*_z	θ_{zz}	H_z
2-a	0.071	0.039	1.83	0.036	0.027	1.33
2-b	.166	.073	2.29	.0248	.0141	1.76
2-c	.282	.099	2.85	.086	.065	1.32
2-d	.517	.077	6.70	.0085	.0063	1.35

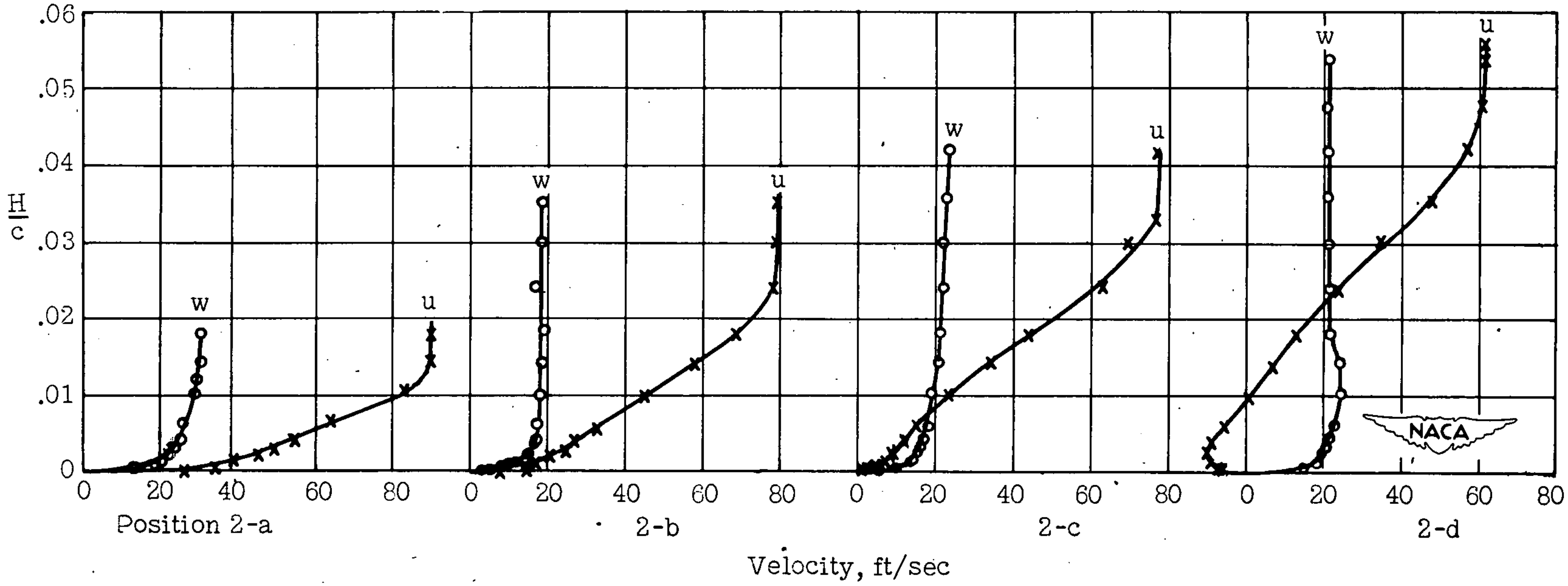

Figure 5.- Chordwise and spanwise velocity profiles along line 2 (fig. 3) for 25° sweepback and 14° angle of attack.

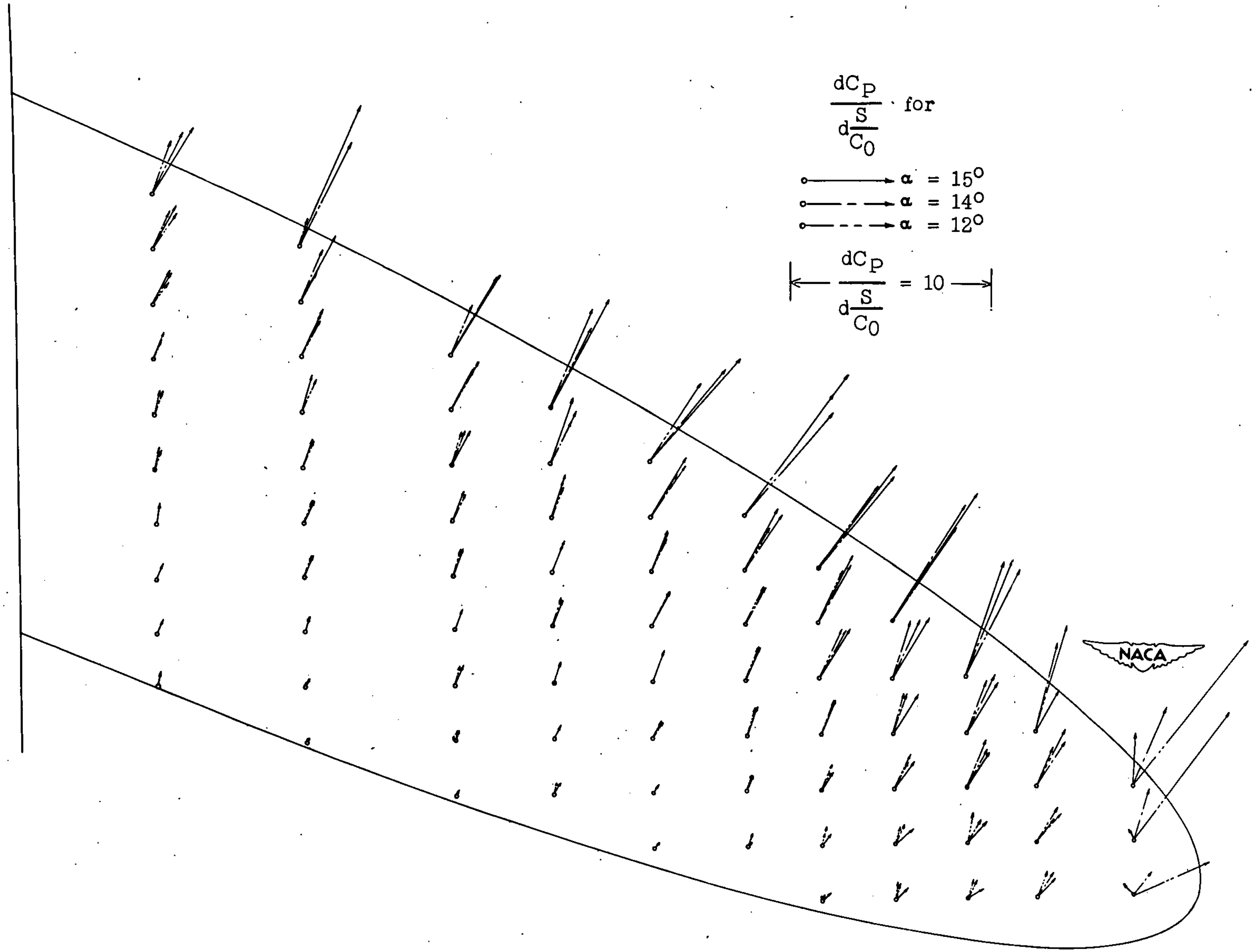

Figure 6.- Vector representation of pressure gradients on upper surface of wing. Values are derived from the measured pressure distribution.

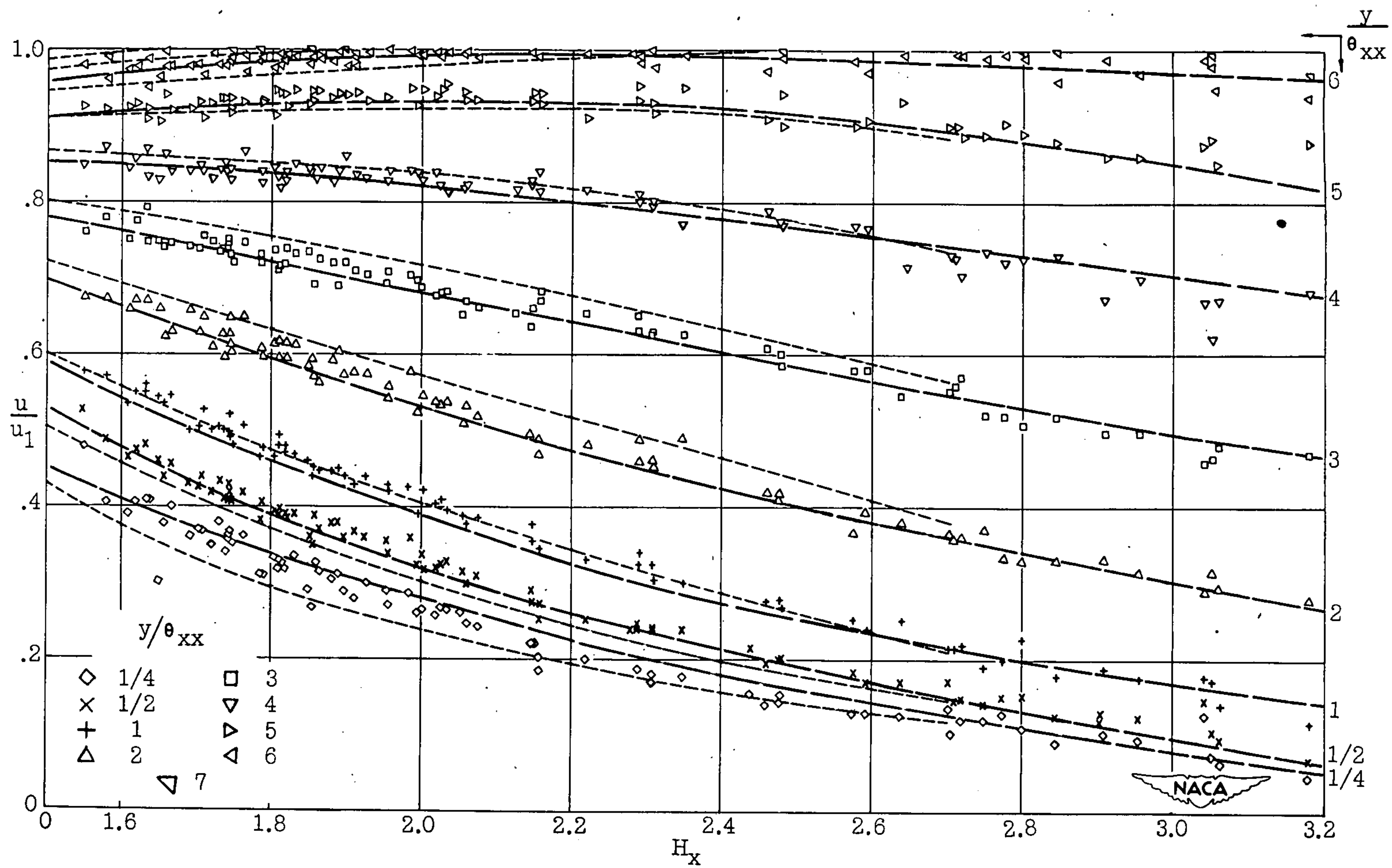

Figure 7.- Representation of the chordwise velocity profiles for angles of attack of 12° and 14°. The profiles which showed flow reversal are not shown. The long-dashed curves are those given in reference 6; the short-dashed are given in reference 5. Beyond $H_x = 2.8$ the curves are extrapolated from those in reference 6.