Technical Notes

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Aerodynamics of High-Lift, Low-Aspect-Ratio Unswept Wings

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Introduction

ONE of the most important outcomes of Prandtl's lifting-line theory[1] was that two-dimensional airfoil shapes could be analyzed separately, before a more detailed examination of three-dimensional wing planforms. This led to the extensive development of airfoil sections[2] to meet a large range of operations, in terms of Reynolds and Mach number, and maximum lift or efficiency. For most applications, this simplified approach, which assumes that the wing aspect ratio (AR) is large ($AR \rangle 1$), is very informative, especially during the early stages of wing planform design.

In this Note, the difficulties arising from trying similar methodology for developing two-dimensional airfoil sections for highlift, unswept leading edge, and small AR wings, without considering the complete three-dimensional planform, will be highlighted. The primary reason for this difficulty is the large spanwise-vorticity gradients that will result in pressure distributions that differ considerably from the isolated two-dimensional airfoil data. Because of the strong downwash induced by these spanwise-vorticity gradients, wing camber can be increased far more than in the case of two-dimensional airfoils. The most important outcome is that such highly cambered wing sections cannot be tested in two dimensions, and the results of their two-dimensional analysis cannot be applied to the design of small-aspect-ratio wings.

Discussion

High-lift, multi-element wing segments are frequently used on the inner section of V/STOL aircraft[3,4] and on some high-speed ground vehicles where, due to a width limitation, the AR of the high-lift section is small. For this example, a highly cambered, four-element airfoil section (Fig. 1) with rectangular wing platform of $AR = 1.5$ was selected. It is assumed here that flow separations are to be avoided, and therefore the pressure field (for Reynolds number $> 0.5 \times 10^6$) can be adequately estimated by potential-flow (panel) methods. The computational method (VSAERO[5]) that has been used is based on quadrilateral, piecewise constant source and doublet surface distribution for modeling the incompressible flowfield.

Typical computational results for the two-dimensional pressure distribution on a four-element airfoil are shown in Fig. 1. Computed pressure distribution at the centerline of a rectangular wing ($AR = 1.5$), having the same airfoil section and the same attitude, is presented in Fig. 2. For these computations a vortex-wake model was specified along the elements trailing and side edges. An effort was made to use information from flow visualization for placing the wake lines parallel to the local streamlines without using an iterative wake relaxation routine.[5]

The most obvious difference between the two cases of Figs. 1 and 2 is the threefold reduction in the C_P range of the three-dimensional data when compared with the two-dimensional case. But, in addition, there is a change in the chordwise loading shape as well. The suction peak in the two-dimensional pressure distribution resembles typical airfoil data,[2,3] whereas in the three-dimensional case more of the load is concentrated at the aft part of the wing. Also, in this case pressure gradients are the strongest near the second flap (from the trailing edge) and, with increased angle of attack, flow separation can be initiated here. Since the pressure field computed by this method for larger aspect-ratio wings agrees

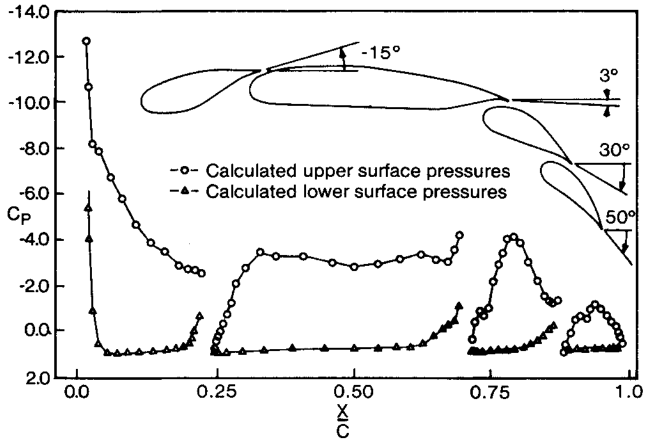

Fig. 1 Geometry of the four-element airfoil and computed two-dimensional pressure distribution ($c = 0.25$ m).

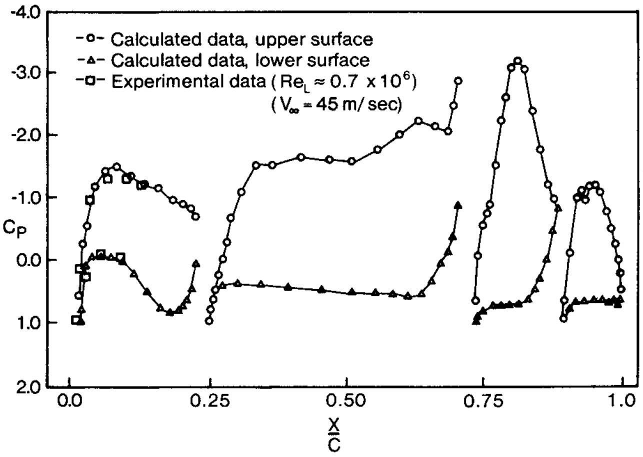

Fig. 2 Pressure distribution at the centerline of an $AR = 1.5$, rectangular wing, having the airfoil section shown in Fig. 1.

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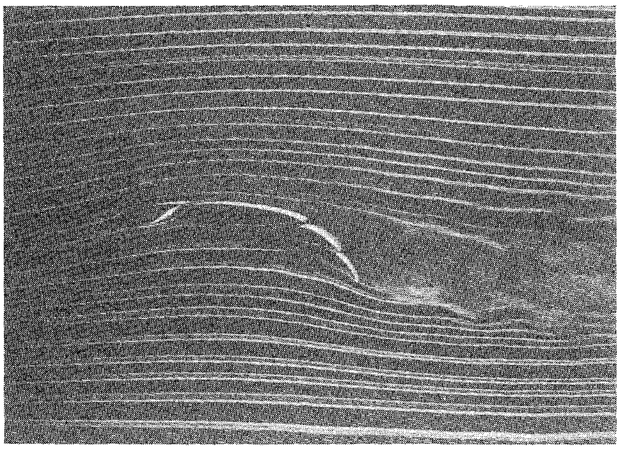

Fig. 3 Visualization of the two-dimensional flow over a four-element airfoil (geometry is same as in Fig.1, $Re_L \approx 0.3 \times 10^6$).

well within the attached flow regions with experimental results,[6] only a limited local validation of the three-dimensional computation was attempted. The large change in the shape of the pressures near the leading edge, along with an effort to simplify the test, influenced the decision to measure the pressures only in this area. The results of these experiments (at $Re_L \approx 0.7 \times 10^6$) with the $Æ = 1.5$ wing are shown in Fig. 2 by the rectangular symbols, and they compare well with the computations. The calculated section lift coefficient for the two-dimensional case was $C_L = 3.73$ and for the three-dimensional case $C_L = 1.78$. Because of this large difference, an attempt to follow lift-line practice and to show an "equivalent section angle of attack" (of $C_L = 1.78$) will lead to an extremely large negative incidence.

Experimental attempt to develop such high-lift airfoil shapes in two dimensions becomes difficult, too, because of the high-pressure gradients, shown in Fig. 1. Two-dimensional smoke-trace flow visualization (in Fig. 3) demonstrate this, and a large stalled area is visible behind the main wing. In this particular case, in spite of the large suction peak at the leading edge, the flow is attached there and separates later behind the main wing element. At this low Reynolds number, the boundary layer in the gap between the main wing and the second flap was thick and reduced the momentum transfer to the wing's suction side, resulting in the visible flow separation pattern. However, three-dimensional low visualizations with tufts indicated that for Reynold number of 0.5×10^6 to 2.0×10^6 (based on wing's chord), the $Æ = 1.5$ wing had a completely attached flowfield. Further experiments with these wings ($Æ = 1.5$) showed that with additional airfoil shape modifications, trailing-edge deflection of up to 90 deg were possible without flow separations (this was equivalent to about $C_L = 4.15$ for the two-dimensional).

Concluding Remarks

This short example shows that highly cambered airfoil shapes cannot be developed, neither experimentally nor numerically, with two dimensional tools. The strong dependency of airfoil shape on the Æ requires the definition of wing planform shape first, and only then and only with three-dimensional tools can the airfoil shape be formed (and optimized).

Consequently, experimental development of such wings is extremely difficult and requires an elaborative fabrication procedure. Therefore, even a simplified three-dimensional computational method, such as the one used here, can accelerate a similar multi-element-wing development program.

For high-lift wings with Æ and Reynolds number similar to the values mentioned here, trailing-edge flap detection of up to 90 deg is possible without causing flow separations.

Acknowledgments

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Diffuser Performance of Two-Stream Supersonic Wind Tunnels

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Introduction

A SIMPLE theoretical model is presented that determines the inviscid, steady-state diffuser performance of a tunnel with two plane, parallel supersonic streams that come into contact downstream of a splitter plate and form an infinitely thin interface. The model predictions can be useful to the design and operation of two-stream supersonic wind tunnels and of aircraft engines in which mixing is done at supersonic speeds.

A typical one-stream supersonic wind tunnel consists of a converging-diverging nozzle with a given sonic throat area, a test section in which uniform conditions are maintained, and a diffuser with a second throat for decelerating the flow and recovering the pressure. Choking criteria for such wind tunnels determine the minimum allowable second throat area for which the upstream conditions remain unchanged and are well established: assuming perfect-gas law and neglecting viscous effects, the minimum second throat area under steady-state conditions is the sonic throat area of the supersonic nozzle. In that case, pressure recovery is complete because the Mach number at the second throat is sonic and shock wave losses are eliminated.

For a stream with nonuniform distributions of Mach number and gas composition, choking criteria are more complex. The parameter that describes the pressure area relation for uniform or nonuniform flows is

$$\beta = \frac{\partial A}{\partial p} \quad (1)$$

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