Computation of Viscous Transonic Flow about a Lifting Airfoil

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Abstract

A NAVIER-STOKES computation method has been extended to consider turbulent transonic flow about lifting airfoils in free air. A brief description of the computational methods is given and results are compared with experimental data. The most striking feature of the calculations was that, in some cases, unsteady periodic motion was obtained along the aft portion of the airfoil and in its wake, even though the airfoil and boundary conditions were stationary. Recent experimental results for the same airfoil indicated oscillations downstream of the shock at a frequency near that which was calculated, but with significantly smaller amplitude.

Contents

For an airfoil at transonic speeds, there is an interaction between the laminar or turbulent boundary layer and the inviscid outer flow. Separation may occur, whether induced by shock or the trailing edge. Furthermore, experimental results[1,2] indicate that under some conditions the flow is periodic rather than steady.

To investigate these phenomena numerically, a first-order, time-dependent Navier-Stokes code[3] was modified by incorporating a turbulence model and a far-field solution along the external boundaries of the computational mesh. The turbulence model is based on mixing length theory with the modifications employed by Deiwert.[4] A far-field solution, obtained from transonic small disturbance theory,[5] was used on the external boundaries to simulate flight in free air. This was necessary because, at transonic speeds, an inordinate amount of coordinate stretching in the vertical direction would be required to yield an accurate solution if freestream conditions were applied on the lateral boundaries. Along the downstream boundary, the method of characteristics is used to allow convection and propagation of disturbances through that boundary with a minimum of spurious reflections into the computational region. In developing this boundary condition, it was verified in test calculations that acoustic waves were propagated without reflection and a plane traveling shock produced only a slight reflection. In the present calculations, vortices have been successfully passed through the boundary.

Computations were performed for the NACA 64A010 airfoil at 2 deg incidence, a Mach number of 0.8, and a Reynolds number based on chord of 4 million. These conditions were selected to allow comparison with steady experimental results[6] and recent unpublished unsteady measurements by D. A. Johnson (private communication). The calculations were performed on a 130×68 nonorthogonal curvilinear mesh extending 3.34 chords above and below the airfoil, 3 chords upstream of the leading edge, and 2 chords downstream of the trailing edge. Mesh spacing was designed to provide about 6 points in the boundary layer at the 40% chord station, which is insufficient to accurately define the turbulent boundary layer in the expansion region. However, in order to limit computer time to less than 2 h, the mesh was not made finer.

For this calculation, the initial field employed was the inviscid small-disturbance solution by TSFOIL,[7] with the shock situated at 85% chord and a lift coefficient C_L of 0.871. The calculation was advanced to a characteristic time, $\tau = tU_\infty/c$, of 9.05 and required 1.88 h computing time on the CDC 7600.

The general flowfield structure for this case was periodic. From $\tau = 0\text{-}3.75$ the shock moved toward the leading edge with vortices shedding in its wake. The shock then moved downstream to approximately the 45% chord station at $\tau = 7.0$. From then to $\tau = 9.05$, there was a small oscillation of the shock about the 45% chord station with an attendant periodic shedding of vorticity. Figure 1 shows a velocity vector plot of the flowfield at $\tau = 8.62$ and illustrates shedding and the sinusoidal pattern of the wake.

Averaging the instantaneous lift and drag coefficients over a period of motion ($\Delta\tau \sim 1.4$) results in mean values of 0.137 and 0.038, respectively, which are markedly different from the measured values[6] of 0.415 and 0.047. This discrepancy is due to the coarseness of the mesh near the airfoil leading edge. The thin boundary layer on the upper surface is contained within the first layer of zones adjacent to the airfoil almost to the 30% chord station.

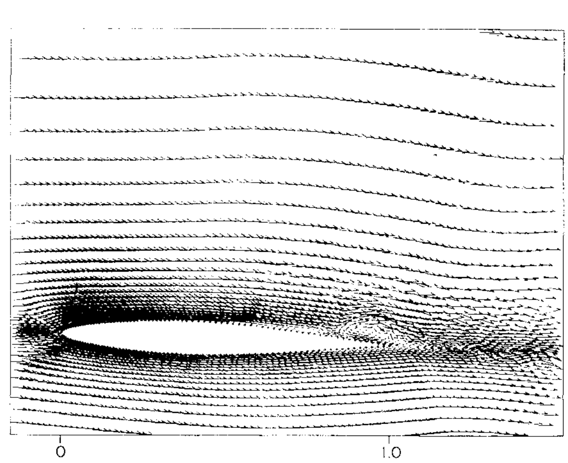

Fig. 1 Velocity field at a characteristic time of 8.62.

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Index categories: Nonsteady Aerodynamics; Transonic Flow; Computational Methods.

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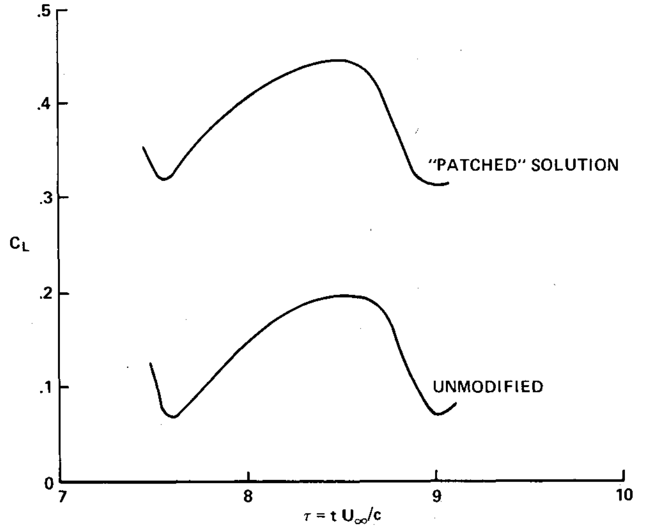

Fig. 2 Variation of lift coefficient over one period.

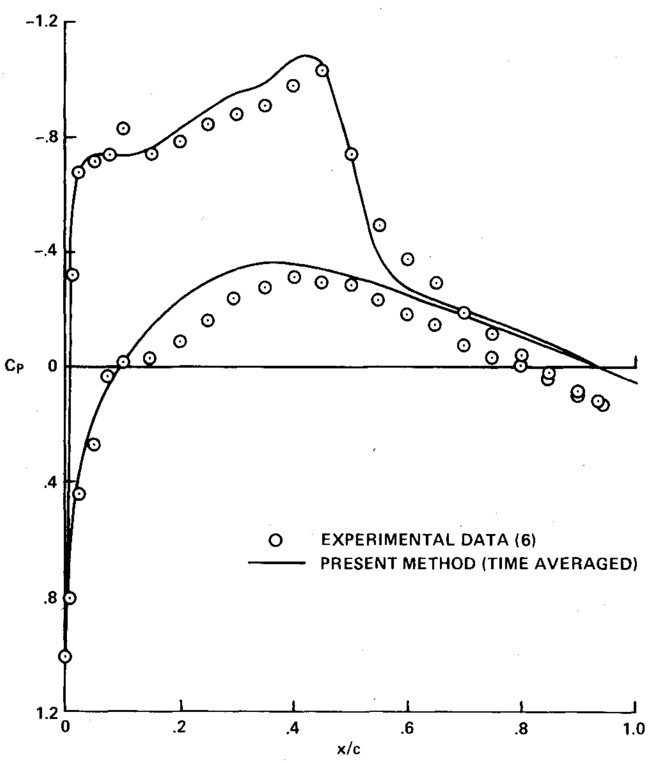

Fig. 3 Time-averaged surface pressure distribution.

Also, the coarseness of the mesh inhibited re-expansion around the leading edge after passage of the shock back downstream toward its equilibrium position. As a result, the high suction pressures near the nose were not obtained numerically and the supersonic region is much smaller than that observed experimentally.

Examination of the time history of the pressure distribution in the leeward expansion region indicates an initial adjustment to body thickness and viscous effects followed by little change until the shock moved into the region. Since the shock adopted an equilibrium position near that observed experimentally and because of the constancy of the expansion region prior to passage of the shock, it was decided to reinstate the former values in the expansion region by "patching" in the solution obtained at $\tau = 1.6$. The resulting instantaneous lift coefficient over a period of motion is shown in Fig. 2, along with the original values.

The instantaneous lift has a period of 1.4 characteristic time units which converts to a reduced frequency $k = 2\pi fc/U_\infty$ of 4.49. Measurements of Johnson on the same airfoil, but at a Reynolds number of 2 million, indicated shock oscillations with a reduced frequency of 5.21. This differs by 16% from the calculated value of 4.49 at a Reynolds number of 4 million. Lending credence to these results are recent experiments and calculations[8] in which periodic motion was obtained on a biconvex section.

Averaging the calculated lift and drag coefficients from the patched solution over a period of oscillation results in mean values of 0.384 and 0.034, respectively. These compare with the experimental results[6] of 0.415 and 0.047. The lower calculated drag is a result of the coarseness of the mesh, particularly near the leading edge. It is believed that, for the thin boundary layers near the leading edge, boundary-layer theory must be integrated with Navier-Stokes computations everywhere else to produce correct drag results.

A comparison of the calculated time-averaged pressures on the NACA 64A010 airfoil with corresponding steady results obtained by interpolating the measurements of Ref. 6 to 2 deg incidence is presented in Fig. 3. Both the calculations and the experimental results place the shock at about the 45% chord station. Pressure recovery in the separated region was not as great as obtained experimentally, possibly because of incorrect prediction of the strength of the vortices being shed, which may, in turn, be related to mesh resolution and turbulence modeling. Spark Schlieren photographs taken by Johnson do not clearly indicate vortices being shed periodically, which may well mean that the amplitude of the periodic disturbances is less than predicted. It should also be mentioned, as discussed in Ref. 9, that the pressure distribution obtained in a slotted wind tunnel may differ from that obtained in free air.

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