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UNSTEADY TRANSONIC FLOW PAST AN AIRFOIL IN A CHANNEL

Sanjay Mittal
Department of Aerospace Engineering, IIT Kanpur, Kanpur 208 016, India.

ABSTRACT Results are presented for computation of unsteady viscous transonic flows past a stationary NACA 0012 airfoil in a channel at various angles of attack. Stabilized finite-element method is employed to solve the compressible Navier-Stokes equations in the conservation variables formulation. Interesting flow patterns are observed as a result of the interference between the lateral boundaries and shocks in the flow. Solutions at various angles of attack are qualitatively different from each other. Hysteresis in the flow is observed at an angle of attack 8°. The free-stream Mach number is 0.85 while the Reynolds number based on the chord-length of the airfoil is 10,000.

1. Introduction

Viscous transonic flows are associated with complex interactions between the boundary/shear layers and shock/expansion waves. Computation of viscous transonic flows past aerospace vehicles and their components will play an increasingly important role in their designs in the future. Though there have been numerous efforts to compute steady transonic flows [1,2,3], fewer studies have been conducted for unsteady, transonic, viscous flows [1,4]. In this article, results are presented for the computation of Mach 0.85 Re 10,000 flow past an airfoil in a channel at various angles of attack.

2. Formulation

Stabilized finite-element method [3] is employed to solve the compressible Navier-Stokes equations in the conservation variables formulation. The SUPG (streamline-upwind/Petrov-Galerkin) stabilization technique is employed to stabilize the computations against spurious numerical oscillations due to advection dominated flows. In addition to the SUPG stabilizations, a shock-capturing term is added to the formulation to provide stability of the computations in the presence of discontinuities and large gradients in the flow. The equation systems, resulting from the finite element discretization, are solved iteratively by using the preconditioned GMRES technique. Time integration of the governing equations is carried out for large values of the non-dimensional time to understand the unsteady dynamics and long-term behavior of the flows.

3. Results & Discussion

A NACA 0012 airfoil is placed in a Mach 0.85 and Re 10,000 flow. The airfoil surface is assumed to be adiabatic and the no-slip condition is specified for the velocity on its wall. At the upstream boundary density and both components of velocity are assigned free-stream values while the component of heat flux vector normal to the boundary is assumed to be zero. At the downstream boundary the pressure is prescribed while the viscous stress vector is assigned zero value. On the upper and lower boundaries and the components of velocity and heat flux vector normal to and the component of viscous stress vector along these boundaries are prescribed zero values. The airfoil is located at the origin of the coordinate axes. The location of the upstream and downstream boundaries, from the mid-chord point of the airfoil, correspond to 4.25 and 7.75 chord-lengths, respectively. The upper and lower boundaries are, each, located at 4.25 chord-lengths away from the mid-chord point of the airfoil. The mesh employed consists of 18,772 quadrilateral elements and 19,014 nodes. The computations are initiated with free-stream conditions in the entire domain. A time step of 0.01 is used to ensure time-accuracy of the results.

Figures 1 and 2 show the density fields for the computed solutions at various angles of attack. The figures for $\alpha=7^\circ$ and less correspond to the peak value of the lift coefficient for the temporally periodic solution while those for $\alpha=8^\circ$ and 10° are for the steady-state solution. At low angles of attack ($\alpha=0^\circ$, 2°) the lateral boundaries do not interact with the flow significantly.. The flow patterns at $\alpha=0^\circ$ compare quite well with the computational results reported in [1] and with the experimental results reported in Shapiro [5] (page 414). At $\alpha=5^\circ$ and beyond the effect of the side-walls becomes important and leads to interesting solutions. At $\alpha=7^\circ$ the downstream part of the lambda shock becomes weaker compared to the flow at lower angle of attack. This causes a decrease in the adverse pressure gradient as a result of which the unsteadiness in the wake decreases. At $\alpha=8^\circ$ the flow reaches a steady-state. Compared to the solution at $\alpha=7^\circ$, the expansion fan at the front half of the airfoil is much stronger for this case, while the downstream branch of the lambda shock, close to the airfoil surface, is a much more gradual compression. Therefore, the shear layer emanating from the upper surface of the airfoil remains stable and results in a steady wake. At $\alpha=10^\circ$ the shock structure as seen for $\alpha=8^\circ$ is unstable and degenerates to a reflection shock system as shown in Figure 2. Using this solution as an initial condition, the angle of attack of the airfoil is changed from $\alpha=10^\circ$ to $\alpha=8^\circ$ in 200 time steps. Figure 2 shows the steady solution obtained in this case. On comparing it with the solution in Figure 1 it can be seen that the close field solutions are quite similar in the two cases while the shock structures are quite different. As α is reduced further, the solution goes back to the temporally periodic state. Figure 3 shows the mean values of lift and drag coefficients and the Strouhal number at various angles of attack.

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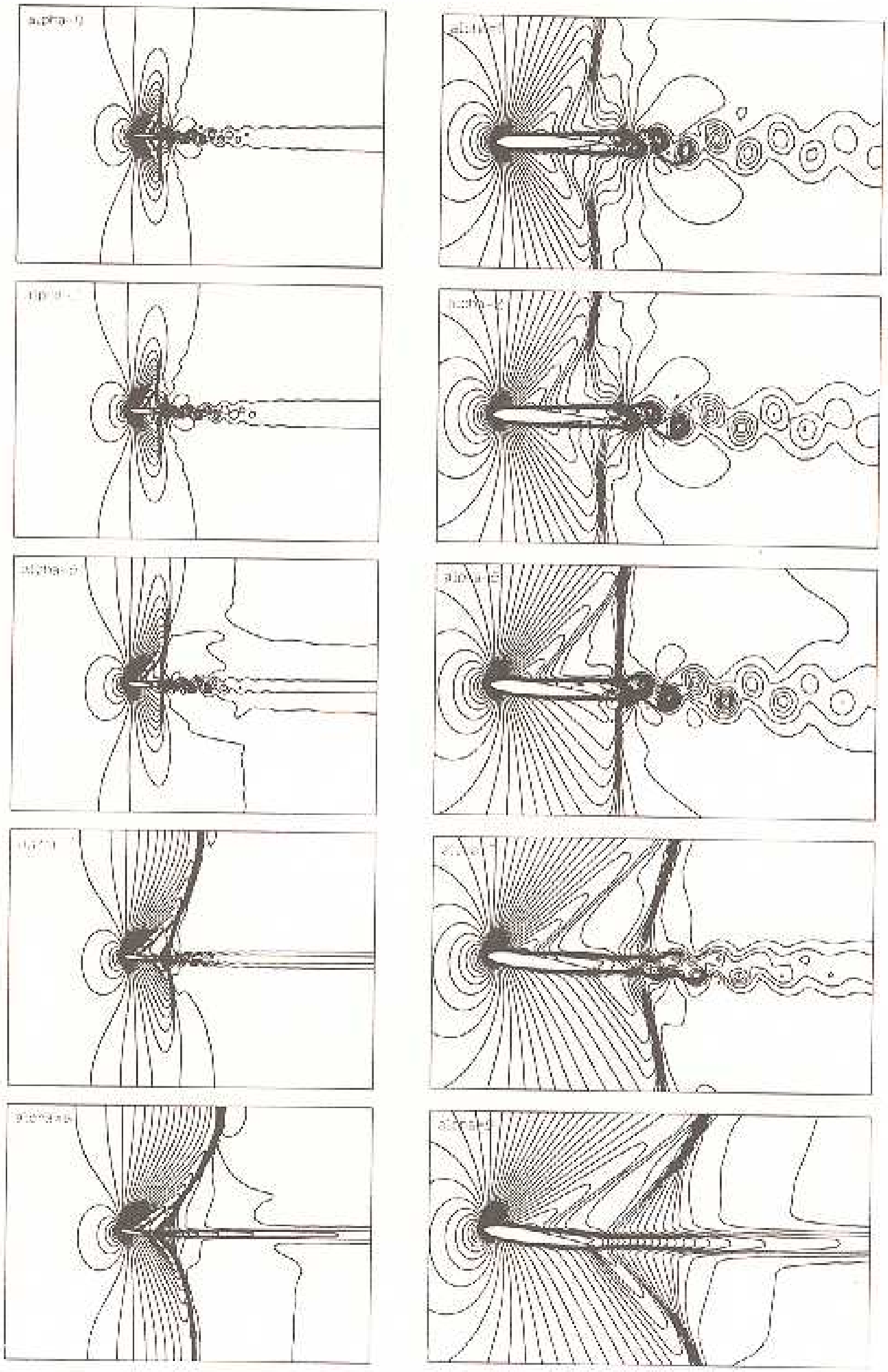

Fig. 1. $Re = 10,000$, $Mach = 0.85$ flow past a NACA 0012 airfoil: density field and its close-up at $\alpha = 0^o, 2^o, 5^o, 7^o, 8^o$ (from top to bottom).

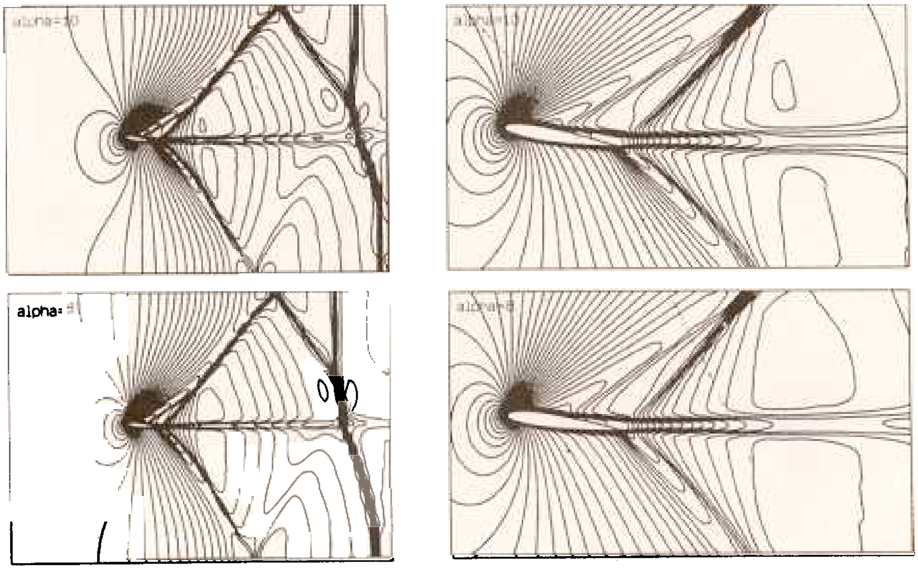

Fig. 2. $Re = 10,000$, $Mach = 0.85$ flow past a NACA 0012 airfoil: density field and its close-up at $\alpha = 10^o, 8^o$ (from top to bottom). The computation for $\alpha = 8^o$ uses the solution at $\alpha = 10^o$ as the initial condition.

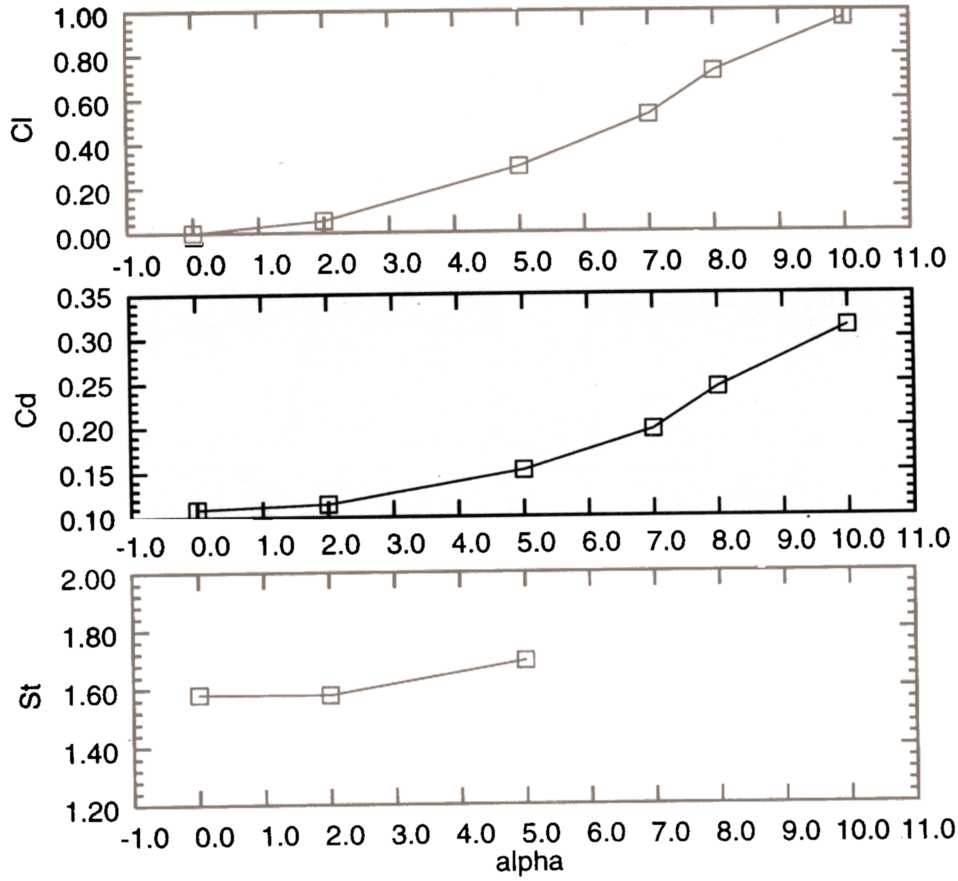

Fig. 3. $Re = 10,000$, $Mach = 0.85$ flow past a NACA 0012 airfoil: mean values of lift and drag coefficients and the Strouhal number at various angles of attack.