A Fast Viscous Correction Method for Unsteady Transonic Flow About Airfoils

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Abstract

THIS study modified the inviscid, small-disturbance, unsteady, transonic code, LTRAN2, to consider the viscous effect. The boundary condition of the vertical surface velocity needs to include an additional term which is derived from the effective displacement thickness of the boundary layer. The effective thickness consists of a viscous wedge to simulate the suddenly thickened boundary layer behind a shock and a displacement thickness obtained from a conventional integral boundary-layer method. Unsteady aerodynamic performances were calculated for several airfoils. Comparisons were made with experimental data of two supercritical airfoils. The viscous solutions gave better agreement with experimental data with practically no increase in computational time.

Contents

Numerical simulation with wind tunnel verification has played an important role in transonic aircraft development. Currently there are two different approaches. The viscous approach is physically more realistic, but numerically it takes too much computational time for practical applications.[1] The inviscid approach, which neglects the viscous effect, gives reasonable results with only 1% of the computational time of the viscous approach if the shock is relatively weak.[2,3] Until faster computers are available, numerical method for inviscid solution will remain the primary tool for industrial applications.

For moderately strong shock situations, the inviscid approach encounters difficulties in determining the actual shock locations. To correct this problem, several viscous correction methods have been deveoped at the expense of computational time. Improvements made by Melnik et al.[4] required at least three times the computational time for the inviscid solution. With an improved algorithm for inviscid solutions,[2,3] it became necessary that the viscous correction method not only improve the accuracy but also maintain the computational efficiency. This study considers the boundary layer in two segments. The suddenly thickened boundary layer behind a shock was simulated by inserting a viscous wedge at the foot of the shock with the following empirical relation[5] to approximate the wedge thickness w,

$$w/c=0 \qquad s<s_{sh} \tag{1}$$

$$=\beta_l\theta_{\max}\{1-\exp[(s_{\rm sh}-s)/c\beta_l]\} \qquad s\geq s_{sh}$$

where s is the distance in the main-flow direction along the surface, $s_{\rm sh}$ the shock location, c the chord length of the airfoil, $\theta_{\max}$ the maximum wedge angle for an attached shock at a given upstream Mach number, and $\beta_l=0.1$ an empirical constant. The displacement thickness δ^* before the shock was evaluated by using a conventional integral boundary-layer method[6] to solve for the momentum thickness and the shape factor H. Since the shock-induced separation had been included in w, δ^* was calculated only for the portion of the boundary layer without separation by limiting $H\leq 1.8$, a preassigned value for $\delta^*=$const as $H>1.8$. During low-frequency maneuvering, δ^* before the shock undergoes no significant changes, while the shock position varies with the instantaneous angle of attack. Consequently, δ^* was calculated only for the steady solution and assumed to be independent of time, while w was evaluated at every time step. Numerical stability was achieved by superimposing w on δ^* as an effective thickness $\delta^*_{\rm eff}$. The normal component of the surface velocity v_n can then be written as

$$v_n=\beta_2\partial(M\cdot\delta^*_{\rm eff})/\partial s \tag{2}$$

where M is the Mach number at the edge of the boundary layer and $\beta_2=2.0$ a second empirical constant. Detailed implementation to accelerate the computational procedure while retaining the numerical accuracy was discussed by Lee.[7] Several airfoils were investigated. Comparisons with experimental data were made for pressure coefficients of two supercritical airfoils which encountered moderately strong shock situations during low-frequency maneuvering.

Figure 1 shows the pressure coefficients for both steady and unsteady states in comparison with the experimental data of Cook et al.[8] for the RAE 2822 airfoil oscillating 1.0 deg about the midchord with a reduced frequency of 0.2. The viscous solution gives better agreement with the experimental data than the inviscid solution. With reference to a stationary frame, the vertical and horizontal components of the pressure coefficient, $(C_p)_v$ and $(C_p)_H$, were obtained by integrating with respect to the instantaneous angle of attack for a complete cycle. The magnitude, $\sqrt{(C_p)_v^2+(C_p)_H^2}$, and the phase angle, $\tan^{-1}[(C_p)_v/(C_p)_H]$, are shown in Fig. 2. The shock wave occurs only on the upper surface where significant differences appear between the inviscid and viscous solutions that are identical at the lower surface. The viscous solution took 100.5% of the computational time of the inviscid solution. Figure 3 shows the pressure coefficients for both steady and unsteady states in comparison with the experimental data of Davis and Malcolm[9] for the NLR 7301 airfoil pitching 2.01 deg about 40% of the chord with a reduced frequency of 0.4. The magnitude and phase angle of the integrated pressure coefficent are shown in Fig. 4. In general, the viscous solution gave a slightly better agreement with the experimental data than the inviscid solution at the upper surface, while the lower surface remains identical. However, some detailed variations, which were observed in the experimental data but were not revealed from the inviscid solution, became clear in the viscous solution. The viscous solution took 88% of the computational time of the inviscid solution. The computational time was reduced because the numbers of supersonic point were reduced in the flowfields due to the viscous effect. In conclusion, a viscous correction

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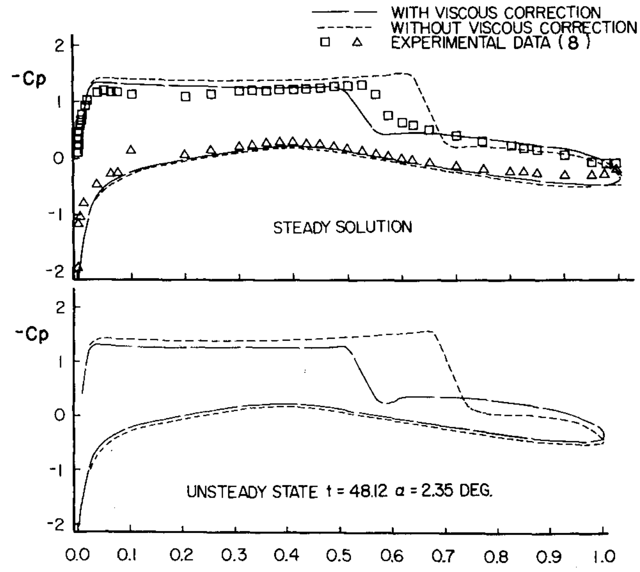

Fig. 1 Pressure coefficient for RAE 2822 airfoil at steady-state $M_\infty = 0.730$, $\alpha_0 = 3.19$ deg and unsteady-state oscillating 1.0 deg about $x/c = 0.5$ with $K = 0.2$.

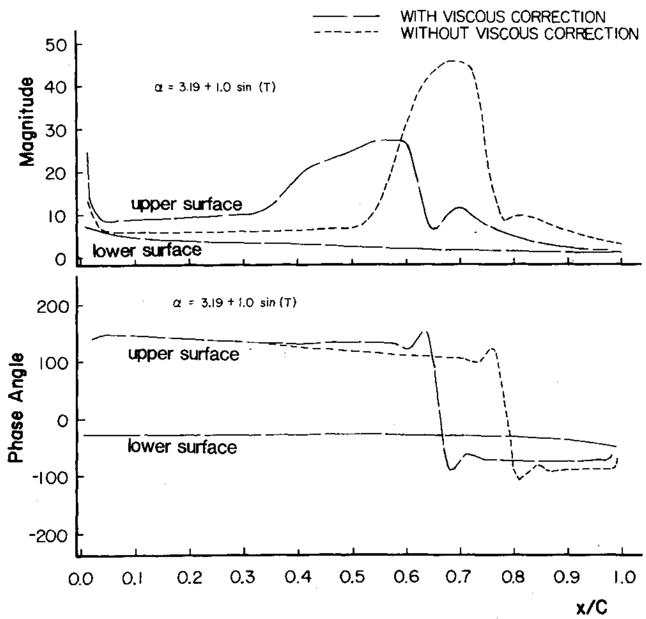

Fig. 2 Integrated pressure coefficient for RAE 2822 airfoil with initial state $M_\infty = 0.730$, $\alpha_0 = 3.19$ deg, oscillating 1.0 deg about $x/c = 0.5$ with $K = 0.2$.

method for unsteady transonic flow was developed to improve the inviscid solution without substantially increasing the computational time.

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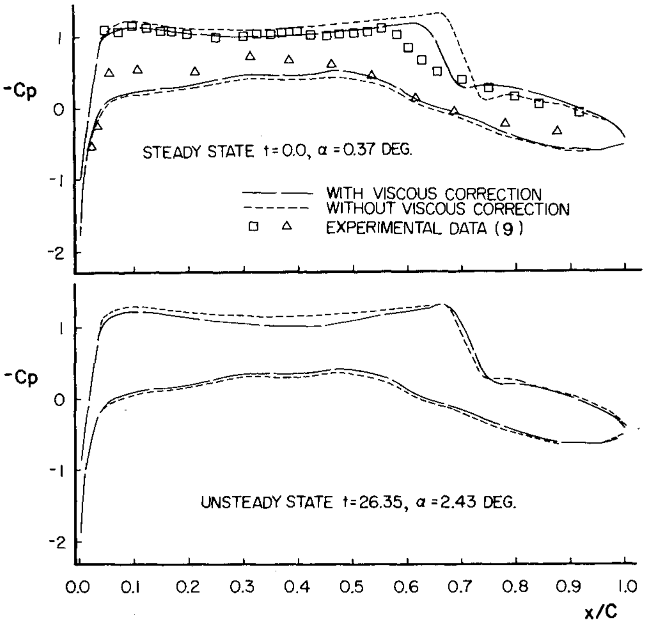

Fig. 3 Pressure coefficient for NLR 7301 airfoil at steady state $M_\infty = 0.752$, $\alpha_0 = 0.37$ deg and unsteady-state pitching 2.01 deg about $x/c = 0.399$ with $K = 0.4$.

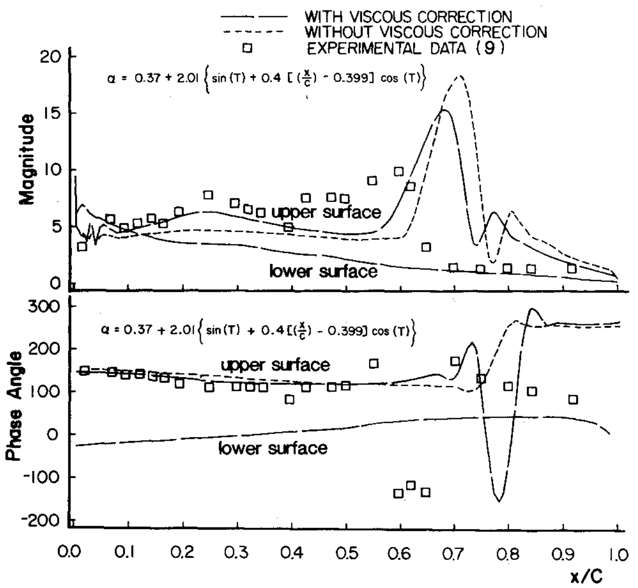

Fig. 4 Integrated pressure coefficient for NLR 7301 airfoil with initial state $M_\infty = 0.752$, $\alpha_0 = 0.37$ deg, pitching 2.01 deg about $x/c = 0.399$ with $K = 0.4$.

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