Engineering Notes

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Fast Viscous Correction Method for Full-Potential Transonic Wing Analysis

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Introduction

THE code TWING[1] was developed to solve the three-dimensional full-potential equation for transonic wing analysis through an iterative process that requires a minimum amount of computer time. Since the viscous effect was neglected, the numerical results deviated from the experimental data for moderately strong shock situations. A viscous correction method was developed to improve the accuracy of the inviscid solution and to maintain its computational efficiency.

The viscous ramp method[2] used an empirical relation to approximate the contour of a suddenly thickened boundary due to shock wave interaction. It resulted in a more realistic shock position with practically no increase in computational time. However, the ramp neglected the viscous effect before the shock. The code BLAYER[3] used a conventional integral boundary-layer method[4] to determine the displacement thickness and resulted in less computer time than the lag-entrainment method.[5] However, BLAYER, which modified the adverse pressure gradient behind the shock to avoid flow separation, became inadequate to locate shock positions.

Analysis

A generalized coordinate transformation was used to provide a three-dimensional computational domain as shown in Fig. 1. The streamwise direction consists of an O-grid, starting from the trailing edge over the upper and lower wing surfaces. The normal direction is from the wing surface to the outer boundary. The spanwise direction is from the inboard region extending beyond the wing tip. The three-dimensionality of the wing, due to sweep-back, tapering, and twist, was considered by the full-potential solution[1] that provided the boundary condition at the freestream for the boundary-layer solution.[3] Considering the shock/boundary-layer interaction along the wing surface in the streamwise direction, the method of superposition[6] was applied to the three-dimensional wings for viscous corrections at each spanwise cross section.

Using an empirical formula, the thickness of the viscous ramp per chord length, δ^*_R/C, is expressed as a function of the streamwise direction along the wing surface s, starting from the leading edge,

$$\frac{\delta^*_R}{C}=\beta_1\theta_{\max}\left[1-\exp\frac{(s_{sh}-s)}{C\beta_1}\right], \quad \text{for } s\geq s_{sh} \tag{1}$$

where β_1, an empirical constant, is equal to 0.1. The term $\theta_{\max}$ represents the maximum deflection angle for an attached shock at a given upstream Mach number, which is determined by the inviscid solution. The subscript sh denotes the location of the shock in the streamwise direction. The ramp thickness before the shock is zero. The boundary-layer displacement thickness δ^* is obtained using the integral boundary-layer code, BLAYER,[3] for both the upper and lower surfaces. The effective displacement thickness $\delta^*_{\rm eff}$ was obtained by superpositioning the ramp thickness δ^*_R and the boundary-layer thickness δ^*. The vertical component of the surface velocity W_v can then be obtained from the effective displacement thickness by the relation,

$$W_v=(\rho u\delta^*_{\rm eff})_s/\rho \tag{2}$$

where the subscript s denotes the first derivative in the streamwise direction, ρ the density, and u the velocity in the s direction. The code TWING[1] is then modified to consider W_v as the transpirational boundary condition. W_v is zero for the inviscid solution. W_v is evaluated by Eq. (2) during the iterative process to determine the velocity potential for TWING to include the viscous effect.[7]

Results and Discussions

Computations were made on an AMDAHL 470 computer. Several wings were studied from relatively weak to moderately strong shock situations. For weak shock situations, it was found that the inviscid solution gave reasonably accurate results for pressure coefficients on the wing surface. Viscous correction, which required approximately 10% additional computer time for the same number of iterations to produce practically the same results on pressure distributions, would not be necessary. For moderately strong shock situations, two different wings were investigated.

The experimental data of the ONERA M6 wing[8] with leading-edge sweep of 30 deg, taper ratio of 0.56, and aspect ratio of 3.8 was computed for Mach number 0.84 and an angle of attack of 3.06. The Reynolds number based on chord length was about 5×10^6. Figure 2 shows the pressure coefficient of the inviscid and viscous solutions in comparison with the experimental data at span stations of 20, 44, 65, and 90%. There was no abrupt change in boundary-layer displacement thickness on the lower surface. The viscous and inviscid solutions gave practically the same pressure distribution. A supersonic region occurred on the upper surface where the critical pressure coefficient was -0.31. In the inboard region, an oblique shock occurred near the leading edge. Even though the pressure coefficient downstream of the shock was rapidly increasing, it remained below the critical value. A normal shock occurred further downstream and created a sudden increase in boundary-layer displacement thickness. The pressure coefficient increased sharply

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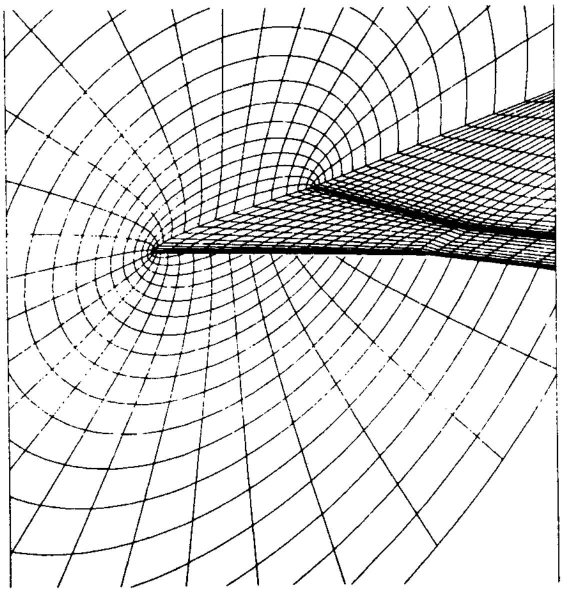
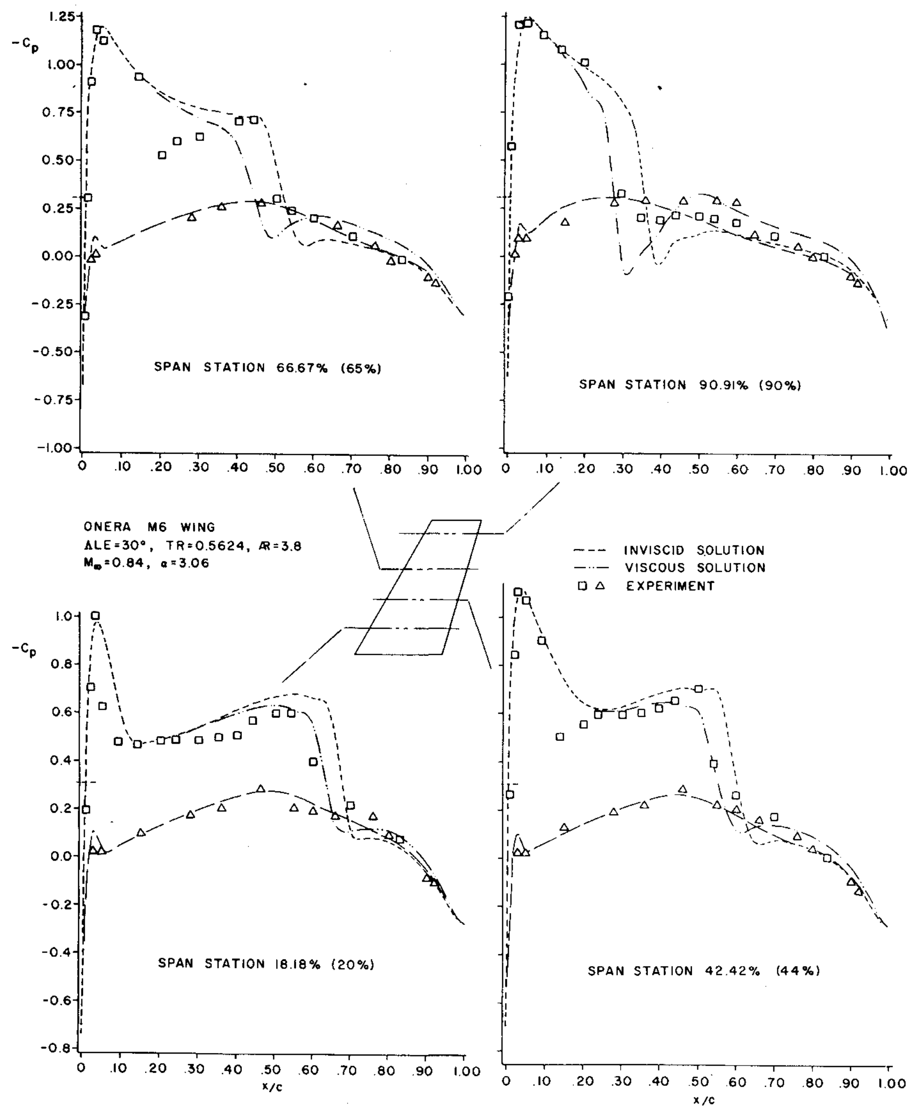


Fig. 1 Perspective view of wing grid.

from below to above the critical value. The viscous ramp was inserted at the location where the flow changed from supersonic to subsonic. The normal shock moved toward the leading edge by approximately 6% of the chord length. At the outboard region, the normal shock, which occurred at a higher Mach number, moved toward the leading edge by about 10% of the chord length. The viscous solution gave better agreement with the experimental data on shock locations. It took 58 iterations and 23.73 min of computer time, while the inviscid solution took 52 iterations and 20.93 min.

Wing A[9] was used for comparison with experimental data on twisted wings. It has a leading-edge sweep of 27 deg taper ratio of 0.4, aspect ratio of 8.0, and 4.8 deg twist between root and tip. Numerical solutions were obtained for Mach number 0.80 and an angle of attack of 1.60 at the root. The Reynolds number based on chord length was about 5×10^6. Figure 3 shows the results of the inviscid and viscous solutions in comparison with the experimental data at the same Mach number and a nominal angle of attack of 2.94 for span stations 15, 30, 50, 70, and 95%. Selection of the computed angle of attack was to give the best correlation with the experimental data at the leading-edge portion of the lower surface. Near the trailing edge, the viscous solution gave a slightly higher pressure coefficient than the inviscid solution. Since there is no abrupt change in δ^*_{eff}, the difference in pressure coefficient is too small to be distinguished at the scale of Fig. 3. Nevertheless, both inviscid and viscous solutions gave higher pressure coefficients than the experimental data near the trailing edge because of the

Fig. 2 Comparison with experimental data for ONERA M6 wing.



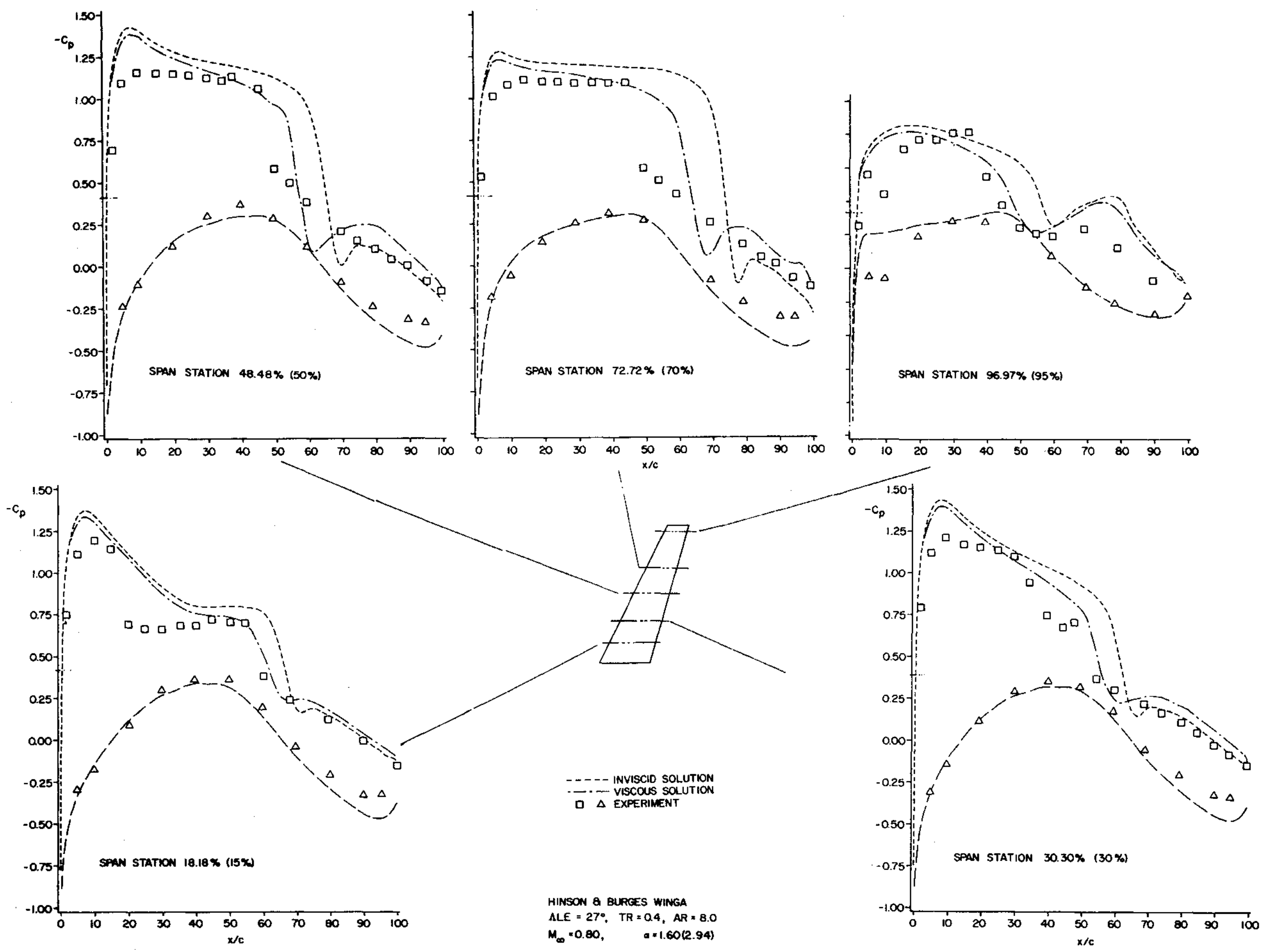


Fig. 3 Comparison with experimental data for wing A.

afterloading. The critical pressure coefficient on the upper surface was -0.42. The viscous ramp was inserted at the foot of the normal shock when the pressure coefficient jumped from below to above the critical value. The shock waves occurred at a lower Mach number in the inboard region than in the outboard region. The distance that the shock location moved toward the leading edge due to viscous correction again was proportional to the shock strength. In general, the viscous solution gave better agreement with the experimental data on shock locations than the inviscid solution. It took 93 iterations and 20.93 min for the viscous solution vs 99 iterations and 20.29 min for the inviscid solution. Consideration of the viscous effect reduced the number of supersonic points in the flowfield and thus reduced the number of iterations for converged solutions.

Conclusions

Full-potential analysis is the most economical method for evaluating transonic wingfields. At the preliminary design stage, TWING with viscous corrections can give a quick estimation of the wing loading with a reasonable accuracy. For weak shock situations, the inviscid solution of TWING gave good correlation with experimental data for a minimum amount of computer time. Viscous correction is not necessary.

For moderately strong shock situations, viscous correction is needed to improve the estimation in shock positions and pressure distributions. The increase in computer time does not have to be substantial, if a viscous ramp is used to approximate the suddenly thickened boundary layer due to shock wave interactions. The present study indicated that 13.5 and 3% increases in computer time were needed for the ONERA M6 wing and the wing A, respectively.

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