Interactive Three-Dimensional Boundary-Layer Method for Transonic Flow over Swept Wings

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Abstract

A THREE-DIMENSIONAL, laminar and turbulent boundary-layer method has been coupled to the full-potential inviscid code, FL030, through a displacement surface interaction. Results of the present method are compared to those obtained with the integral viscous-inviscid interaction method, TAWFIVE, and with experimental data.

Contents

The viscous-inviscid interaction method developed by the authors in Ref. 1 solves the compressible boundary-layer equations in a nonorthogonal curvilinear coordinate system. The Reynolds shear-stress terms are modeled with the Cebeci-Smith eddy-viscosity formulation and the equations are transformed by the introduction of two stream functions so that the continuity equation is satisfied identically. The transformed governing equations are discretized with the second-order accurate predictor-corrector finite-difference technique of Matsuno,[2] which has the advantage that the crossflow derivatives are formed independent of the sign of the crossflow velocity component.

The viscous routine is coupled to the inviscid wing-body full-potential code,[3] FL030, through a displacement surface interaction. The three-dimensional displacement surface Δ^* is determined by applying Matsuno's predictor-corrector scheme to Moore's[4] partial-differential equation,

$$\nabla \cdot [\rho_e V_e \Delta^* \int_0^\infty (\rho_e V_e - \rho V) \mathrm{d}n] = 0$$

at each viscous marching step. If separation is encountered, the displacement surface is calculated from the separation line to the trailing edge by setting the velocity defects in the integral thicknesses to be constant and equal to their values at separation. Boundary-layer separation is assumed to occur when the streamwise skin-friction coefficient goes through zero.

The viscous-inviscid interaction is performed by running the inviscid code for the original wing planform and then running the viscous code to determine an initial displacement surface distribution. The displacement surface is added to the original wing coordinates in an under-relaxed manner, and this procedure is repeated for the new wing planform (wing plus displacement surface) until the viscous updates are less than a user-specified tolerance. This interaction procedure requires a new mesh to be generated for each interaction and is less efficient than the use of transpiration boundary conditions. However, the present method was obtained by replacing the integral boundary-layer solver in the existing TAWFIVE computer code[5] with a differential procedure, and in order to make the two codes compatible, the original interaction strategy was retained. Therefore, the only difference between the present method and the TAWFIVE code is the manner in which the boundary-layer equations are solved.

The present method is first applied to the ONERA M6 wing[6] for a freestream Mach number of $M_\infty = 0.84$, an angle of attack of $\alpha = 3.06$ deg, and a Reynolds number of $Re_\infty = 11.7 \times 10^6$. The results of the present method are compared to those obtained with the TAWFIVE code and with experimental values in Fig. 1. Five viscous updates and 500 inviscid iterations were performed by both methods. Viscous effects are moderate for this case and each calculation compares favorably with the measured values except at the tip where the rounded wingtip in the experiment was not modeled analytically.

The present method was also applied to a Douglas Aircraft Company wing-fuselage semispan model,[7] which was tested at the NASA Ames Research Center's 14-ft transonic wind tunnel. The Douglas wing planform has an aspect ratio of 0.68, a taper ratio of 0.3, and a quarter-chord sweep angle of 35 deg. The experiment was conducted at $M_\infty = 0.825$, $\alpha = 4$ deg, and $Re_\infty = 4.5 \times 10^6$ based on the mean aerodynamic chord. It was intended to compare the results of both calculation methods with the experiment; however, the TAWFIVE code computed unrealistically large values of the displacement surface on the

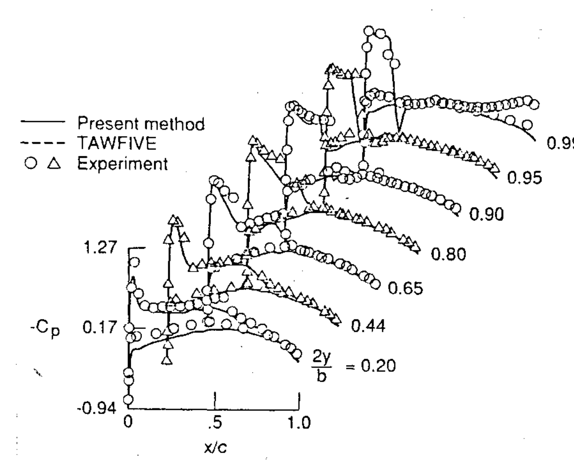

Fig. 1 Pressure coefficient distributions for the M6 wing.

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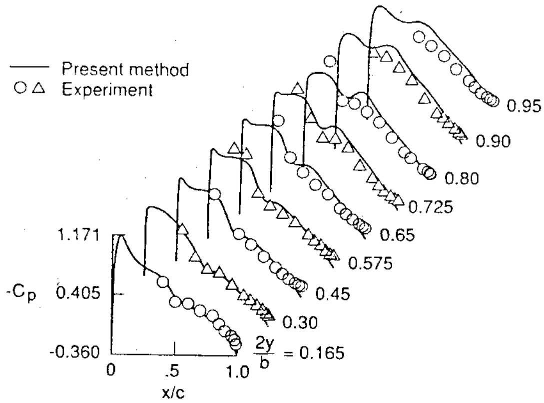

Fig. 2 Pressure coefficient distributions for the Douglas wing.

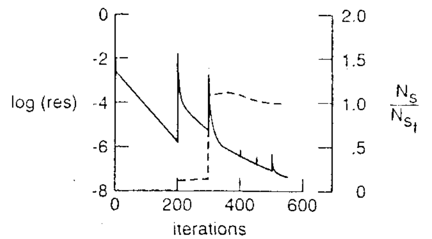

Fig. 3 Convergence history for the Douglas wing calculation.

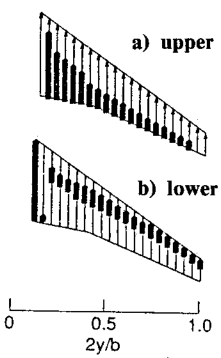

Fig. 4 Locations of spanwise profiles with velocity crossover for the Douglas wing.

lower surface of the wing and the solution became unstable. The present method had no difficulty computing a solution.

A comparison of the calculated and experimental upper surface pressure coefficient is given in Fig. 2. A total of five viscous updates and 550 inviscid iterations were performed for this case. The convergence history of the average residual is shown by the solid line in Fig. 3. In order to dampen out the low-frequency errors in an efficient manner, the first 200 inviscid iterations are performed on a coarse grid of $40 \times 8 \times 6$ grid points streamwise, spanwise, and normal to the surface, respectively. The inviscid grid points are then doubled for the next 100 iterations and then doubled again for each subsequent 50 iterations on a fine inviscid grid of $160 \times 32 \times 24$. After each 50 inviscid fine grid iterations, a boundary-layer calculation is performed on a viscous grid of $220 \times 21 \times 21$ and the wing coordinates are updated in an under-relaxed manner by the calculated displacement surface. This procedure is repeated until the solution has converged. The final inviscid potentials and wing coordinates are saved on an output file and the solution can be restarted if desired. The dashed line in Fig. 3 represents the total number of supersonic points normalized by its final value.

An advantage of the present method over integral boundary-layer methods is that the shape of the velocity profiles are determined as part of the solution and are not assummed a priori. Specifically, the majority of integral methods do not allow the spanwise velocity profile to have velocity crossover, where a profile has both positive and negative values across the boundary layer. The only restrictions placed on the spanwise profiles in the present method are the no-slip condition at the wing surface and that it asymptotically approach its inviscid value at the edge of the boundary layer. The locations where the spanwise velocity profiles have velocity crossover for the Douglas wing are shown in Fig. 4. The dark triangles in the figure are where the crossover profiles were calculated for both the upper and lower surfaces of the wing. The majority of the spanwise velocity profiles on the upper surface of the wing are of this type.

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