Transonic Wind-Tunnel Wall Interference Prediction Code

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Abstract

A SMALL-DISTURBANCE transonic wall interference prediction code capable of modeling solid, open, perforated, and slotted walls as well as slotted and solid walls with viscous effects has been developed. This code was developed by modifying the outer boundary conditions of an existing aerodynamic wing-body-pod-pylon-winglet (WBPPW) analysis code. Comparisons are presented at transonic flow conditions between computational results and experimental data for a wing alone in a solid-wall wind tunnel.

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Methods to deal with or account for the influence of tunnel walls or data obtained in wind tunnels are diverse. Facilities are designed to minimize wall influences on the flowfield near the models. Model sizes are chosen and test conditions often are restricted to minimize wall interference effects. Analytical techniques have also been developed to predict or assess wall interference. Typically, these are categorized based on the type of measurements required to make the prediction or assessment.[1] These measurements may include static pressures and flow angularity at the wall, wall characteristics (porosity, slot geometry, divergence angles, etc.), and/or static pressures or simulation of the model. The various methods have specific advantages and disadvantages relative to their accuracy, ease of use, and resource requirements.

A method has been developed that simulates the transonic flow over complex configurations in a wind tunnel.[2] A variety of wall characteristics may be modeled through the implementation of appropriate homogeneous wall boundary conditions applied at the outer boundary in an extended small-disturbance flow solver WBPPW.[3] The code has incorporated boundary conditions that simulate solid, porous, slotted, and open jet walls, as well as viscous effects on solid and slotted walls. The importance of this approach is that prediction of wall interference effects can be determined a priori or when static pressures are sparse or unavailable.

To demonstrate this method, comparisons of computational results and wind-tunnel data were made for an untapered NACA 0012 semispan wing swept 20 deg with an aspect ratio of 3. Lockman and Seegmiller[4] performed this experimental investigation in the High Reynolds Number Channel I Tunnel at the Ames Research Center on the NACA 0012 wing specifically for obtaining data that could be used to assess computational methods. The Channel I Tunnel is 10 × 15 in. with parallel solid sidewalls and 0.15 deg of divergence in the solid top and bottom walls to account for channel boundary-layer growth. The semispan wing was mounted on

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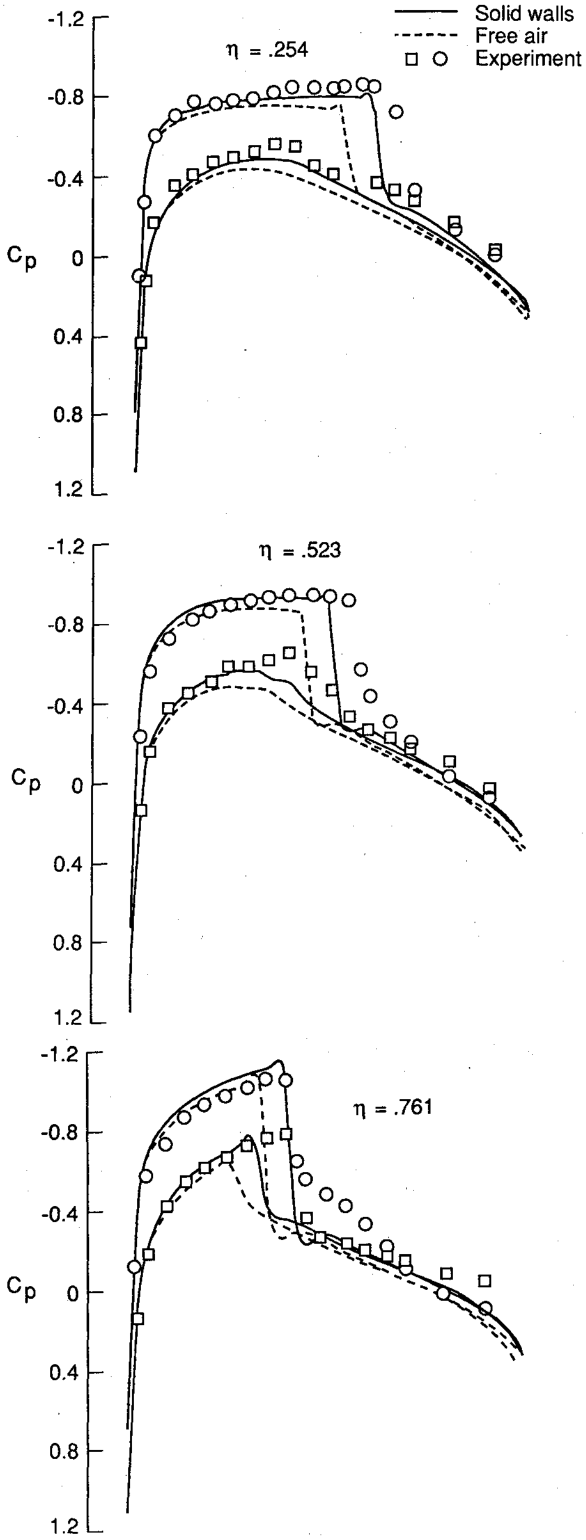

Fig. 1 Comparison of computations and experiment on a swept NACA 0012 wing at $M = 0.826$ at $\alpha = 2.0$ deg; η is the fractional span position %.

the centerline of the tunnel sidewall. In this study, the particular data used were obtained at $M = 0.826$, $\alpha = 2.0$ deg, and a Reynolds number of 8.07×10^6 based on wing chord, where M and α represent freestream Mach number and angle of attack, respectively. No transition strips were used on the wing at this particular condition since the boundary layer was known to be fully turbulent.

Comparisons of the data and computational results with free-air and solid-wall outer boundary conditions are presented in Fig. 1 for three span stations. The model-in tunnel and free-air computations were made at 2.5-deg angle of attack to match the lift levels in the calculations to the experiment. Significant differences between the computations modeling in the wind-tunnel walls and free-air conditions can be seen for all three span stations. The pressure-level changes on the upper and lower surfaces and in the shock location resulting from simulating free-air and solid wind-tunnel wall boundary conditions are indicative of the strong interference effects of the tunnel walls.

At the most inboard station, there is good agreement between the experimental and computational results, with the exception of the upper surface shock location, which is predicted computationally approximately 5% chord forward of the data. This same discrepancy in the upper surface shock location can also be seen at the midchord station, where the computations also indicate a stronger shock than the data. At the most outboard station, there is good agreement on the shock location, but again the computations predict a much stronger shock. As the shock develops out of the span on the lower surface, the discrepancies between computations and experiment also increase.

Wing oil-flow patterns obtained during the experimental investigation show flow separation on the outboard portion of the wing at the preceding conditions. The boundary-layer computations did not detect separation until approximately $0.93c$ (c is the local chord, in.), which is much farther aft than indicated by the oil-flow patterns. This inability of the boundary-layer algorithm to predict accurately flow separation is not surprising since the details of shock-wave/boundary-layer interaction are not computed. This attribute can also account for the overprediction of the shock strength.

The discrepancies seen in the shock location are most likely due to the nonconservative differencing used in the computational method. Lock[5] has shown that, for cases where there is a relatively strong shock, there can be obvious differences in the shock location predicted by conservative and nonconservative differencing techniques. Lock specifically demonstrates this by comparing the shock location for an NACA 0012 airfoil at $M = 0.85$ and $\alpha = 0.0$ deg using a nonconservative potential flow method and the Euler method of Ref. 6. For this case, the nonconservative method predicts the shock approximately 7% chord ahead of the shock location computed by the Euler method.

This partial demonstration of the wall interference prediction method illustrates drawbacks using a nonconservative small-disturbance formulation and the particular boundary-layer method employed by the WBPPW code. Despite the complexity of the flow environment of this case, reasonable correlation is seen between computational and experimental results that could not have been obtained without modeling the wind-tunnel walls as shown in Fig. 1.

In summary, the results obtained from this prediction method thus far are promising. However, many more comparisons between experiment and computations need to be made to assess fully the wall boundary conditions modeled by the method. Additional comparisons are also needed to define the limitations of the wall interference prediction method.

References

[1]Kraft, E. M., "An Overview of Approaches and Issues for Wall Interference Assessment/Correction," NASA CP-2319, Jan. 1983.

[2]Phillips, P. S., and Waggoner, E. G., "A Transonic Wind Tunnel Wall Interference Prediction Code," AIAA Paper 88-2538, June 1988.

[3]Boppe, C. W., "Aerodynamic Analysis for Aircraft with Nacelles, Pylons, and Winglets at Transonic Speeds," NASA CR-4066, April 1987.

[4]Lockman, W. K., and Seegmiller, H. L., "An Experimental Investigation of the Subcritical and Supercritical Flow About a Swept Semispan Wing," NASA TM 84367, June 1983.

[5]Lock, R. C., "A Review of Methods for Predicting Viscous Effects on Airfoils and Wings at Transonic Speeds," AGARD CP 291, 1980.

[6]Sells, C. C. L., "Numerical Solutions of the Euler Equations for Transonic Flow Past a Lifting Aerofoil," Royal Aircraft Establishment, Bedford, England, TR-80065, 1980.

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