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Numerical Prediction of Transition of the F-16 Wing at Supersonic Speeds

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This work is part of the high speed research program currently underway at NASA. This project has the goal of gaining understanding of the technical requirements for supersonic-hypersonic flight. Specifically, this research is part of a continuing project to study the laminar flow over swept wings at high speeds and involves the numerical prediction of the flow about the F-16XL wing. The research uses the CNS/ARC3D codes and the resulting crossflow velocity components in order to estimate transition locations on the wing.

Before a full parametric study was to be conducted it was necessary to start with a baseline case that had a fair amount of laminar flow. This was necessary so that the effects of changing the various parameters could be distinguished. To do this and keep the baseline free stream Mach number equal to 1.5 at an altitude of 40 to 50 thousand feet, the Reynolds number was varied by changing the root chord length. The results showed that the extent of laminar flow was increased as the local Reynolds number was decreased. A root chord of 5 feet was selected for the baseline case and yielded laminar flow up to approximately 20% of the chord.

Effects of angle of attack on the extent of laminar flow was found to be minimal. This result can be attributed to the fact that a laminar flow airfoil was used in this study, which has a continuous favorable pressure gradient over approximately the first 20% of the chord for angles of attacks up to 10 degrees. It should also be noted that even after 20% chord the pressure gradient either slowly continued to increase, but never decreased before 90% chord, except for the higher swept cases when separation occurs. Angles of attack greater then 10 degrees were not considered since this study assumes natural laminar flow for normal supersonic cruise flight conditions.

In addition to investigating the effects of angle of attack, the effects of sweep were also studied. It was necessary to keep the wing's aspect ratio constant so that the comparison in sweep would not be misinterpreted by other changes in the wing's surface area or wing chord. It was also necessary to avoid sweeping the wing into the Mach cone, which would cause shock waves and distort the flow. Due to the above requirements it was necessary to shear the baseline clipped delta wing to obtain the different sweeps and maintain the same aspect ratio. The results obtained from the parametric sweep portion of this study to date show that an increase in sweep yields a decrease in the extent of laminar flow and therefore an earlier transition.

Once the above results have were obtained, it was necessary to validate that the result were grid independent. This required making sure that the results did not change with increased grid resolution. Results of this portion of the study are not yet obtained. This effort is on going.

Finally, investigation of the numerical methods being applied in this study have led to the following recommendations. It was found that the two dimensional boundary layer code uses a conical flow assumption that is not truly valid for swept wings and should be replaced with a three dimensional boundary layer code. Furthermore, it is recommended that future research directly use Navier-Stokes solutions in place of the Boundary Layer solutions.

The research carried out under this agreement was supplemental to work being performed under NASA Training Grant NGT-70228. The final technical reports for the work will be issued under that grant.