Computational/Experimental Pressure Distributions on a Transonic, Low-Aspect-Ratio Wing

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Abstract

AN experimental investigation was recently conducted to obtain detailed three-dimensional pressure distribution and boundary-layer data on a generic model of a modern advanced-technology wing and to compare these data with modern computer codes. The wing was designed for supercritical airfoil sections, moderate aft loading, mild shock waves, and a mild pressure recovery using a three-dimensional, full potential, transonic wing code (FL022) optimized at a Mach number of 0.85 and an angle of attack of about 5 deg. This Synoptic presents the results of the oil flow studies to determine the nature of the surface flow patterns.

Contents

A highly swept, low-aspect-ratio wing (designated wing C) was selected for this study. It has a large leading-edge sweep angle (45 deg) and a large mean chord to develop a thick, easily measured boundary layer. A large-scale (0.90 m) semispan model (Fig. 1) was tested in the NASA Ames 6×6 ft Transonic Wind Tunnel. Surface pressure measurements, oil flow studies, and three-dimensional boundary-layer surveys were obtained at the design angle of attack of 5 deg over a Mach number range of 0.25-0.96 and a Reynolds number range of $3.4\text{-}10 \times 10^6$, based on the mean aerodynamic chord. Measured pressures and comparisons with predictions are presented in Ref. 1.

Photographs of the oil flow tests are presented in Fig. 2 for $M = 0.82$, 0.85, and 0.90 at $Re = 6.8 \times 10^6$. At the design Mach number of 0.85 (Fig. 2a), where the flow was intended to be unseparated, local separation occurred on the outer third of the semispan. The measured pressure distributions[1] show that this was caused by a strong shock-wave/boundary-layer interaction. Arguments presented in Ref. 1 show that this flow separation at the design condition could be eliminated by certain adjustments to the design techniques.

Next, the Mach number was reduced to 0.82 (Fig. 2b) where the flow separation disappeared. Only the mild design shock wave is vaguely observed in the oil flow pattern, crossing the wing at about 15-25% chord. The most prominent feature is the lack of three-dimensionality in the surface flow pattern. At the trailing edge, the measured flow direction angle was 8 deg outboard at the midsection. The predicted inviscid flow direction angle at the trailing edge is about 5 deg inboard, so that the total change in flow angle through the boundary layer is only about 13 deg. In Ref. 1, it is argued that this lack of three-dimensionality in the boundary

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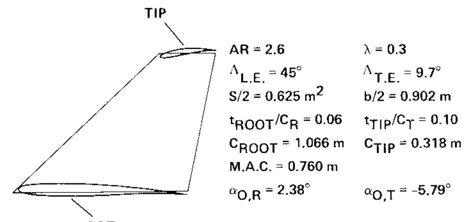
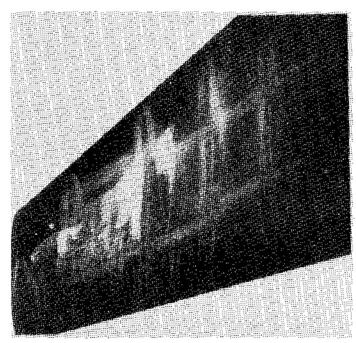
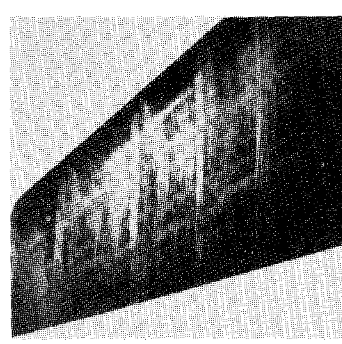


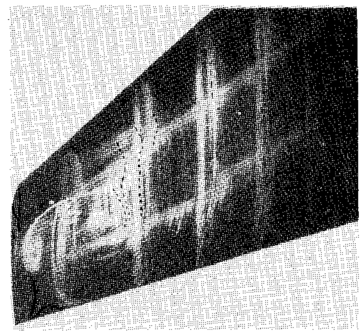
Fig. 1 Wing C geometry.



a) $M = 0.85$ (design test condition).



b) $M = 0.82$.



c) $M = 0.90$.

Fig. 2 Oil flow photographs at $a = 5$ deg, $Re = 6.8 \times 10^6$.

layer results from the wing design process, in which the wing was optimized for a mild shock wave and a mild pressure recovery.

When the Mach number was increased to 0.90 (Fig. 2c), the flow separation that existed at $M=0.85$ over the outer 30% of the wing increased in extent and moved slightly rearward.

Early in the design, it was expected that a low-aspect-ratio wing with large leading-edge sweep angle would have a significant three-dimensional boundary-layer flow, even at conditions for unseparated flow. Evidently, from the present results, this is not necessarily the case, except near the leading edge. These oil flow patterns indicate that the boundary-layer measurements will not be a substantial test of three-dimensional boundary-layer codes for unseparated flow. However, the oil flow patterns and pressure distributions with shock-wave/boundary-layer separation provide an interesting test case for Navier-Stokes computations of the wing flowfield.

Finally, Ref. 1 considers the question of what design criteria would result in a significant three-dimensional boundary-layer flow at conditions for unseparated flow. Results are cited from a test in the Ames 14 ft Transonic Wind Tunnel[2] for a transport-type wing, designed for a thick aft section and more available wing volume. This resulted in a strong pressure recovery and a boundary layer that was highly three-dimensional, but unseparated, in the last 15% chord and with flow angles measured to be as high as 30 deg outboard. These latter test results provide a good experimental test case and should be analyzed to determine the validity of three-dimensional boundary-layer methods.

References

[1]Keener, E. R., "Computational-Experimental Pressure Distributions on a Transonic, Low-Aspect-Ratio Wing," AIAA Paper 84-2092, Aug. 1984.

[2]Spaid, F. W., "Transonic Airfoil and Wing Flowfield Measurements," AIAA Paper 84-0100, Jan. 1984.