As Dowell discussed, the size of the governing matrix equations is substantially reduced by eliminating the usual displacement coordinates in favor of the joint forces or modal force vector, between substructures. This is the key difference between the Receptance method and the modal force method.[1,2] The latter directly uses the physical coordinates at joints to generate the system equation. The emphasis is on the determinant of the modal force matrix $(\bar{H})$ for natural frequencies and on the modal force vector $(\bar{f})$ for mode shapes of the synthesized structure.

The Modal Force method has been extended and applied for the system modification,[2,3] rotor dynamics,[4] and system transient response analysis.[5]

We share the same feelings as Dr. Dowell and hope that the method will be more widely used for system dynamic analysis.

References

[1]Yee, E. K. L., and Tsuei, Y. G., "Direct Component Modal Synthesis Technique for System Dynamic Analysis," *AIAA Journal*, Vol. 27, August 1989, pp. 1083-1088.

[2]Tsuei, Y. G., Yee, E. K. L., and Lin, A. L. Y., "Physical Interpretation and Application of a Component Modal Synthesis Technique (Modal Force Method)," *Proceedings of the Eighth International Modal Analysis Conference Proceedings*, Orlando, Florida, Feb., 1990.

[3]Tsuei, Y. G., and Yee, E. K. L., "A Method for Modifying Dynamic Properties of Undamped Mechanical Systems," *American Society of Mechanical Engineering Dynamic Systems, Mearsurment, and Control*, Vol. 111, Sept. 1989, pp. 403-408.

[4]Tsuei, Y. G., Yee, E. K. L., and Wang, S. S., "A Component Modal Synthesis Technique for the Lateral Vibration Analysis of Aircraft Engine Systems," *Proceedings of the Seventh International Modal Analysis Conference*, Las Vegas, Nevada, Feb., 1989, pp. 377-383.

[5]Tsuei, Y. G., and Yee, E. K. L., "Component Synthesis Method for Linear and Nonlinear Transient Response," *AIAA Journal*, (to appear).

Comment on "Optical Boundary-Layer Transition Detection in a Transonic Wind Tunnel"

P.M.H.W. Vijgen*
High Technology Corporation, Hampton, Virginia 23666
C. P. Van Dam†
University of California, Davis, California 95616
and
C. J. Obara‡
Lockheed Engineering and Sciences Co. Hampton, Virginia 23666

DURING recent years, methods to nonintrusively measure details of the laminar-to-turbulent transition process have been developed mostly for high-Mach-number flows where, for example, hot-wire probes are not practical. A differential laser interferometer is applied in the experiment discussed in Ref. 1 to detect relative variations in density fluctuations in a laminar, transitional, and turbulent boundary layer over a wind-tunnel model at freestream Mach numbers ranging from $M = 0.7$ to $M = 0.8$. The purpose of this Comment is to clarify some details of the transition process in the flowfield of Ref. 1 and to point out a few inconsistencies in the experimental setup and the presented analysis of the transition mode in Ref. 1.

Figure 1 presents the predicted subcritical pressure distribution along the upper surface of an NACA 66-006 airfoil, which has a slightly negative angle of attack. The two-dimensional method of Ref. 2 that models wind-tunnel walls is used to analyze the potential flowfield for the test configuration as described in Ref. 1. The compressible laminar boundary-layer development and the linear stability characteristics of the boundary layer along the airfoil are analyzed for the appropriate conditions (chord-Reynolds number R_c = 1.96 million and $M = 0.70$) using the methods of Refs. 3 and 4, respectively. The onset of laminar separation is predicted to occur at 11.6 cm ($x/c \cong 0.77$, where the airfoil chord is $c = 15$ cm) from the leading edge. The well-defined transition pattern in the spanwise direction indicated by the sublimating chemical technique in Fig. 5 of Ref. 1 (the segment unaffected by the turbulent wedges) is indicative of transition due to shear-layer instability at laminar separation[5]. Transition occurs at x/c = 0.80 in the experiment. Also, the increased spreading angle of the turbulent wedge after $x/c \cong 0.66$ in Fig. 5 of Ref. 1 corresponds with the predicted onset of streamwise pressure increase at $x/c = 0.65$ in Fig. 1 of this Comment. Thus, the predicted flowfield along the upper surface of the airfoil appears to match the experimentally measured flowfield reasonably well.

The predicted logarithmic amplification growth ("n-factor") for several Tollmien-Schlichting (T-S) disturbance frequencies is presented in Fig. 1 for the given test conditions. The most amplified unstable frequencies in the attached boundary-layer ahead of the separation point are in the range of 15-40 kHz. The calculated wavelength for the 25-kHz disturbance is about 7δ (δ is the boundary-layer thickness, δ = $0.004c$ at $x/c = 0.65$). The combination of the favorable pressure gradient over the front portion of the airfoil, the relatively low chord-Reynolds number and the damping effect of flow compressibility[6] prevents the growth of these disturbances until the onset of pressure recovery at 9.8 cm (x/c = 0.65) from the leading edge. The n-factor for the T-S frequencies around 25 kHz grows rapidly in the adverse-pressure-gradient region but does not exceed 4. (If the Mach number is increased to $M = 0.8$, even lower amplitudes are obtained for the unstable T-S frequency range.) Natural transition due to catastrophic growth of unstable T-S waves has been correlated for wind-tunnel experiments to coincide with amplification factors of 7–11.[7]

Unfortunately, the range of most amplified unstable frequencies lies below the chosen bandwidth of the inter-

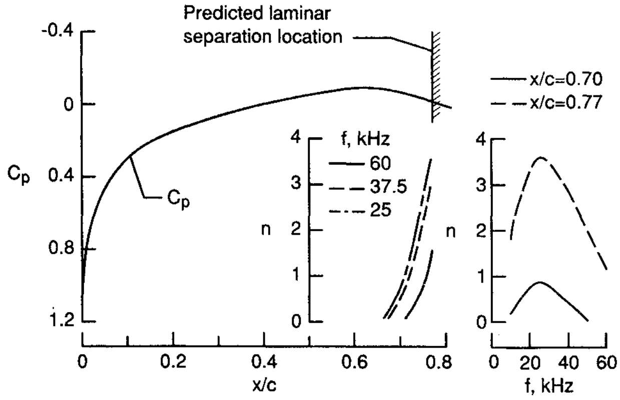

Fig. 1 Predicted surface-pressure distribution (C_p-Pressure coefficient) over NACA 66-006 airfoil in tunnel geometry of Ref. 1 and predicted amplification ratios for several unstable T-S disturbance spectrum (*f*-frequency); $M = 0.70$, $R_c = 1.96$ million.

Received Dec. 4, 1989; revision received Feb.20, 1990; accepted for publication March 20, 1990.

*Research Engineer. Member AIAA.

†Associate Professor, Department of Mechanical, Aeronautical and Materials Engineering. Member AIAA.

‡Research Engineer.

ferometer system used in the experiment of Ref. 1. The 40 kHz - 1 MHz bandpass filter of the data acquisition system makes it very difficult to detect the predicted T-S waves ahead of separation. The effect of the bandpass filter is clearly visible in the frequency spectra of Figs. 12 and 13 in Ref. 1. In all cases the recorded power density reaches a maximum at a frequency of 40–50 kHz; below 40 kHz the power-density level drops quickly. In these two figures, dramatic changes over the complete frequency range are indicated between the power-spectra levels of the laminar and the transitional boundary layer. Taking into account the previously discussed calculated frequency range, it appears that there is a strong dependence of the power spectrum on the measurement location in the tunnel. This dependence may have overpowered the effect of the boundary-layer state on the power spectrum. It is, therefore, impossible to use the frequency data of Figs. 12 and 13 in Ref. 1 as an indication of the transition mode. However, based on the computational results given here and the flow visualization in Ref. 1, it is most probable that transition occurred due to laminar separation and was not caused by attached-flow T-S instability as suggested in Ref. 1.

In conclusion, proper tuning of the laser inferometer system to the T-S instability frequencies at given test conditions and the simultaneous measurement of disturbance spectra in the laminar boundary layer using, for example, hot-film techniques will allow a more in-depth verification of the proposed nonintrusive test technique. Further analysis of the wind-tunnel results may also reveal important information on the transition mechanism in the separation bubble at compressible conditions.

References

[1]Azzazy, M., Modarress, D., and Hall, R. M., "Optical Boundary-Layer Transition Detection in a Transonic Wind Tunnel," *AIAA Journal,* Vol. 27, No. 4, April 1989, pp. 405–410.

[2]Stevens, W. A., Goradia, S. H., and Braden, J. A., "Mathematical Model for Two-Dimensional Multi-Component Airfoils in Viscous Flow," NASA CR-1843, 1971.

[3]Kaups, K., and Cebeci, T., "Compressible Laminar Boundary Layers with Suction on Swept and Tapered Wings," *Journal of Aircraft,* Vol. 14, No. 7, July 1977, pp. 661–667.

[4]Malik, M. R., "Finite-Difference Solution of the Compressible Stability Eigenvalue Problem," NASA CR-3584, 1982.

[5]Obara, C. J., "Sublimating Chemical Technique for Boundary-Layer Flow Visualization in Flight Testing," *Journal of Aircraft,* Vol. 25, No. 6, June 1988, pp. 493–498.

[6]Vijgen, P. M. H. W., Dodbele, S. S., Holmes, B. J., and van Dam, C. P., "Effects of Compressibility on Design of Subsonic Natural Laminar Flow Fuselages," *Journal of Aircraft,* Vol. 25, No. 9, Sept. 1988, pp. 776–782.

[7]Jaffe, N. A., Okamura, T. T., and Smith, A. M. O., "Determination of Spatial Amplification Factors and Their Application to Predicting Transition," *AIAA Journal,* Vol. 8, No. 2, Feb. 1970, pp. 301–308.