Engineering Notes

ENGINEERING NOTES are short manuscripts describing new developments or important results of a preliminary nature. These Notes cannot exceed 6 manuscript pages and 3 figures; a page of text may be substituted for a figure and vice versa. After informal review by the editors, they may be published within a few months of the date of receipt. Style requirements are the same as for regular contributions (see inside back cover).

C80-132

Illustration of Airfoil Shape Effect on Forward-Swept Wing Divergence

20018
80001

Samuel R. Bland*
NASA Langley Research Center, Hampton, Va.

Introduction

FORWARD-swept wings offer several aerodynamic and performance advantages over conventional wing configurations. Because forward-swept wings are much more susceptible to divergence than rearward-swept wings, a renewed interest in this static aeroelastic instability has arisen. Recent tests in the NASA Langley Transonic Dynamics Tunnel showed significant differences in the divergence behavior of forward-swept wings with the same planform and stiffness, but with different airfoil sections. In an attempt to understand these differences, some divergence calculations were made which took airfoil shape into account. In this Note, divergence boundaries for untapered wings with conventional and supercritical airfoil sections are compared for two sweep angles, zero and −15 deg. Simple strip theory was used with the two-dimensional aerodynamic characteristics computed from a finite difference formulation of the full potential equation.

Analysis

The static aeroelastic analysis employed one bending and one torsion mode for a uniform rectangular cantilevered beam with the elastic axis at midchord. The analysis[1] was based upon the differential equations formulation in the structural coordinate system. The aerodynamic coefficients were assumed to be constant across the span and were computed from two-dimensional theory using the Mach number component normal to the leading edge.

The geometry is sketched in Fig. 1. The effective angle of attack α in the structural coordinate system is

$$\alpha = \Theta - W' \tan\Lambda$$

where Θ is the torsional displacement and W is the bending displacement. The prime denotes d/dy. The differential equations are

$$EIW'''' - cc_{l_\alpha}\alpha\, q\cos^2\Lambda = 0$$

$$GJ\Theta'' + c^2 c_{m_\alpha}\alpha\, q\cos^2\Lambda = 0$$

where $q\cos^2\Lambda$ is the dynamic pressure component normal to the leading edge and the pitching moment coefficient axis is at midchord. Expressing the displacements in terms of mode shapes (w and θ) and generalized coordinates (ζ_w and ζ_θ)

$$W = w\zeta_w, \qquad \Theta = \theta\zeta_\theta$$

leads to the homogeneous system

$$\begin{bmatrix} EIw'''' + cc_{l_\alpha} qw' \sin\Lambda\cos\Lambda & -cc_{l_\alpha} q\theta\cos^2\Lambda \\ -c^2 c_{m_\alpha} qw' \sin\Lambda\cos\Lambda & GJ\theta'' + c^2 c_{m_\alpha} q\theta\cos^2\Lambda \end{bmatrix} \times \begin{Bmatrix} \zeta_w \\ \zeta_\theta \end{Bmatrix} = 0$$

The divergence speed is obtained by setting the coefficient determinant to zero, which yields a linear equation for the divergence dynamic pressure q_D. In terms of the parameters:

$A = cl$ (structural planform area)

$P = (l/c)\cos^2\Lambda$ (half aerodynamic aspect ratio)

$S = GJ/EI$ (stiffness ratio)

one obtains

$$-1 = (q_D A^2/GJ)(l^{-2}\theta/\theta'')c_{m_\alpha}\cos^2\Lambda + (q_D A^2/EI)(l^{-3}w'/w'''')Pc_{l_\alpha}\tan\Lambda$$

Within the framework of simple strip theory, EI, GJ, c_{l_α}, and c_{m_α} are independent of y. In order for q_D also to be a constant, it is necessary that the dimensionless ratios $l^{-2}\theta/\theta''$ and $l^{-3}w'/w''''$ be constant. This condition is met by requiring the mode shapes θ and w to satisfy the differential equations

$$l^2\theta'' + \mu^2\theta = 0$$

and

$$l^3 w'''' - \nu^3 w' = 0$$

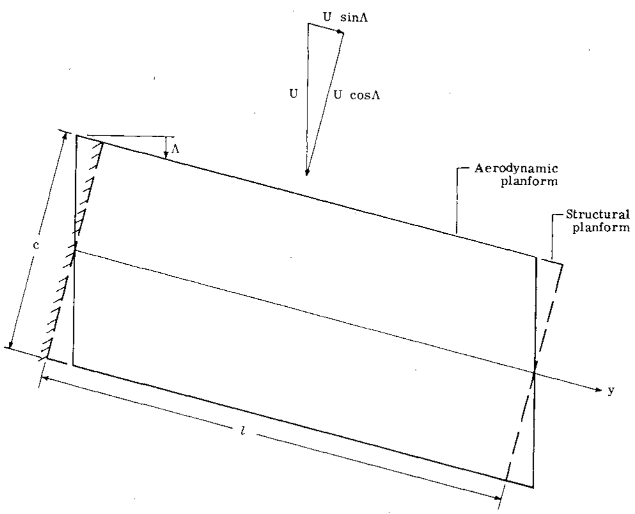

Fig. 1 Wing geometry.

Received Dec. 12, 1979. This paper is declared a work of the U.S. Government and therefore is in the public domain.

Index categories: Aeroelasticity and Hydroelasticity; Transonic Flow.

*Aero-Space Technologist, Aeroelasticity Branch, Structures and Dynamics Division. Associate Fellow AIAA.

in which μ and ν are the respective eigenvalues of the torsion and bending equations. These equations could be used to determine the exact divergence shape of the wing, but only the eigenvalues are required here. In terms of these eigenvalues, the divergence equation becomes

$$1 = q_D (A^2/GJ)(c_{m_\alpha} \mu^{-2} \cos^2 \Lambda - PS c_{l_\alpha} \nu^{-3} \tan \Lambda)$$

The eigenvalues are found by assuming an exponential solution of each equation, applying the cantilever boundary conditions and setting the determinant of each of the resulting homogeneous systems to zero. One finds

$$\cos \mu = 0$$

with first eigenvalue $\mu = \pi/2$ and

$$\exp(-3\nu/2) = -2 \cos(\nu\sqrt{3}/2)$$

with first eigenvalue $\nu = 1.84981280$. In terms of the dimensionless divergence parameter

$$Q = 4A^2 q_D / (\pi^2 GJ)$$

the divergence equation is

$$Q = 1/(c_{m_\alpha} \cos^2 \Lambda - \lambda PS c_{l_\alpha} \tan \Lambda)$$

where $\lambda = \pi^2/(4\nu^3) = 0.38981309$.

Within the limitations of the simple strip theory this divergence equation is exact, only two modes are needed.

All of the aerodynamic parameters were obtained from inviscid, two-dimensional theory. The linear theory values are

$$c_{l_\alpha} = 2\pi/\beta \quad \text{and} \quad c_{m_\alpha} = \pi/(2\beta)$$

where $\beta = \sqrt{1-M^2}$ and M is the freestream Mach number. The lift and moment coefficients from the nonlinear theory[2] were computed for an angle of attack of 0.5 deg. For these symmetric airfoils, a linear relation was assumed between these values and zero in computing the required coefficients c_{l_α} and c_{m_α}. The aerodynamic method[2] gives a solution of the nonlinear, full potential equation which treats thickness and shock wave effects accurately.

Results and Discussion

Calculations were made for wings with constant aerodynamic aspect ratio $P=2$ and stiffness ratio $S=48/35$ (suitable for a thin aluminum beam). The aerodynamic parameters, c_{l_α}, c_{m_α}, and aerodynamic center

$$ac = ½ - c_{m_\alpha}/c_{l_\alpha}$$

are shown in Fig. 2. The full potential equation values for an NACA 64A010 airfoil and for a symmetric supercritical airfoil (both 10% thick) are compared with linear theory. There are two principal differences in the behavior of the two airfoils: first, the supercritical airfoil achieves a higher lift coefficient before the force break which occurs with strong shock wave formation; and second, the rearward shift of the aerodynamic center begins at about 0.025 Mach number higher for the supercritical airfoil. The linear theory shows a continuing increase in c_{l_α}, with the aerodynamic center remaining at the quarter-chord point.

The divergence parameters Q are given in Fig. 3 for sweep angles of zero and -15 deg. At $\Lambda = 0$ the divergence parameter is inversely proportional to the midchord pitching moment coefficient. For the two airfoils, the rapid increase in Q that occurs at transonic speeds is due to the shift of aerodynamic center rearward toward the elastic axis. This shift, which begins at Mach 0.8 for the supercritical airfoil and at Mach 0.775 for the conventional airfoil, marks the bottom of the divergence bucket, even though the force break occurs at a higher Mach number in each case. Since there is no aerodynamic center shift in the linear theory, Q continues to decrease in accordance with the Prandtl-Glauert rule.

The divergence pressures of the 15 deg forward-swept wings are about half those of the unswept wings. The unfavorable effect of wing bending causes this trend. The divergence bucket occurs at a somewhat higher Mach number. This Mach number shift results principally from the aerodynamic coefficients being calculated for the Mach number component normal to the leading edge, but then plotted against the higher freestream value.

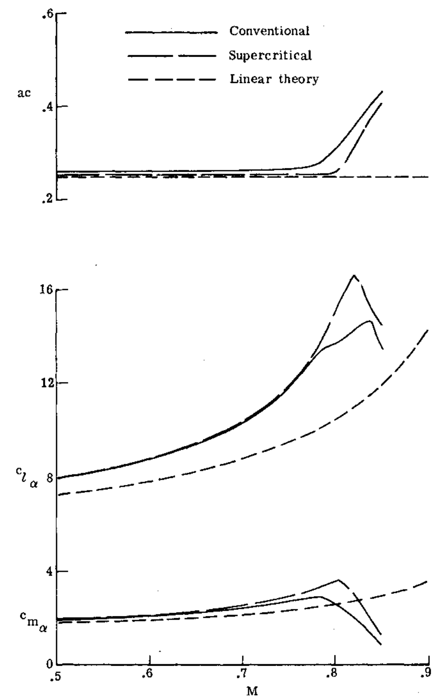

Fig. 2 Aerodynamic parameters.

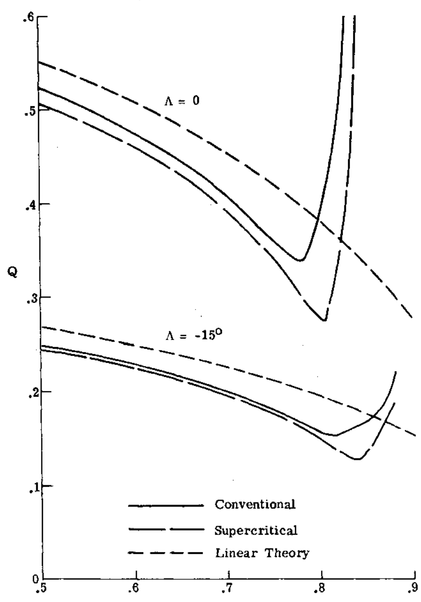

Fig. 3 Effect of airfoil shape and wing sweep on divergence.

Conclusion

Calculations based upon simple strip theory show that a nonlinear aerodynamic method predicts a minimum divergence speed in the transonic range which is associated with the rearward shift of aerodynamic center. In addition, airfoil shape has a significant influence upon divergence as shown by a 17% difference in minimum divergence dynamic pressure between a supercritical wing and a conventional wing. Although the relative sensitivity of various airfoil shapes to divergence may be assessed by the strip theory approach used here, sophisticated, three-dimensional transonic aerodynamic methods should be employed to predict wing divergence characteristics accurately.

References

[1]Bisplinghoff, R.L., Ashley, H., and Halfman, R.L., *Aeroelasticity*, Addison-Wesley Publishing Co., Reading, Mass., 1955, pp. 479-481.

[2]Bauer, F., Garabedian, P., Korn, D., and Jameson, A., "Supercritical Wing Sections II," *Lecture Notes in Economics and Mathematical Systems*, Vol. 108, Springer-Verlag, New York, 1975.

C.80-133

Analysis of High Velocity Impact on Hybrid Composite Fan Blades

80003
80006

C. C. Chamis* and J. H. Sinclair†
NASA Lewis Research Center, Cleveland, Ohio

Introduction

APPLICATION of advanced fiber composites to turbine engine fan blades offers several potential advantages. The most significant of these are: 1) elimination of the midspan shroud which results in improved engine efficiency, 2) lighter engine and aircraft weights which result in reduced fuel consumption, and 3) higher possible tip speeds which can reduce the number of fan or compressor stages and result in fewer parts and lower initial and operating costs.

One major problem with the application of advanced fiber composites to fan blades has been their low resistance to impact in general, and to bird ingestion and other foreign object damage (FOD) in particular. A major difficulty in predicting impact resistance of composite fan blades has been the limitations of available analysis methods for coping with the complex problem of predicting the local and global dynamic response when subjected to high velocity impact. This paper describes recent developments in the analysis of high velocity impact of composite blades using a computerized capability which is under development at the NASA Lewis Research Center.

Briefly, at the present stage of development, the method consists of coupling a composite mechanics computer code with the direct-time integration feature of NASTRAN. All aspects of composite mechanics from micromechanics to laminate analysis and combined stress failure are handled via the composite mechanics computer code. The structural dynamics aspects of the high velocity impact are handled via NASTRAN. The discussion of the application of the computerized method in this paper is limited to the linear dynamic response of an interply hybrid composite fan blade subjected to a high velocity impact of a 2 lb bird.

Presented as Paper 79-0783 at the AIAA 20th Structures, Structural Dynamics, & Materials Conference, St. Louis, Mo., April 4-6, 1979.

Index categories: Structural Composite Materials; Structural Dynamics.

*Aerospace and Composite Structures Engineer. Member AIAA.
†Aerospace Materials Engineer.

Blade Geometry, Hybrid Composite, and Finite Element Model

A photograph of the blade investigated is shown in Fig. 1. The blade was designed and made by the General Electric Co.[1] The nominal dimensions of the blade are 21 in. long, and 12 in. wide at the tip. The thickness varied along the blade centerline from about 0.90 in. at the root to 0.30 in. at the tip. The nominal leading and trailing edge thicknesses were 0.13 in. The angle of twist was approximately 33 deg. The nominal tip radius from the center of the shaft was about 35 in.

The interply hybrid composite blade consisted of boron/epoxy outer plies at ±45 deg (measured from the radial line), S-glass/epoxy plies at 0 and ±45 deg near the root, S-glass cloth filler plies at the root, and Kevlar 49/epoxy plies at 0, ±45, and 90 deg interspersed through the thickness of the blade. The blade also had metallic leading edge protection. The total number of different materials in the hybrid composite blade was five. The nominal ply thickness was 0.010 in. The fiber volume ratio was about 0.60. The several 0 and ±45 deg S-glass/epoxy plies near the root were added to increase bending strength of the blade for improved impact resistance.

The blade was modeled using 127 nodes (grid points) and 210 anisotropic-material, triangular plate elements (CTR1A2 in the NASTRAN[2] element library). Each element had different material properties (Fig. 2).

Analysis, Results, and Discussion

The high velocity impact analysis of the hybrid composite blade was performed using the computerized capability depicted schematically in Fig. 3. This computerized capability

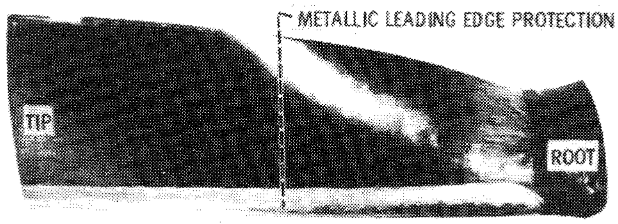

Fig. 1 Hybrid composite fan blade nominal dimensions (21 in. long, 12 in. wide at tip).

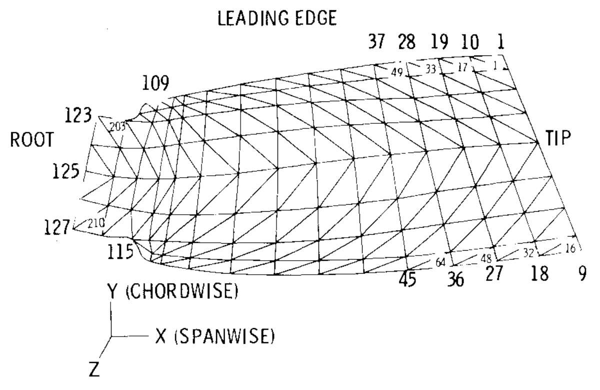

Fig. 2 Hybrid composite fan blade finite element representation schematic (127 nodes, 210 elements, 210 material properties).