Engineering Notes

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Transonic Aeroelasticity of Fighter Wings with Active Control Surfaces

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Introduction

AIRCRAFT structural weight can be reduced if aerodynamic means can be developed to counter the forces and moments that drive flutter and dynamic instability. One such means is flutter suppression by actively using control surface oscillations. Use of active control is becoming increasingly important for future aircraft, which will tend to be more complex for high maneuverability. So far, analytical aeroelastic studies with active control surfaces have been restricted to the linear subsonic and supersonic regimes.[1]

Aeroelastic characteristics of wings are especially sensitive in the transonic regime because of flow nonlinearities and the presence of moving shock waves. The influence of the control surface on both aerodynamic and aeroelastic performance of wings is more pronounced in the transonic regime. This pronounced influence of the control surface on wing aerodynamics can be constructively used by active controls.[2]

The lack of efficient computational tools previously restricted the theoretical studies of active controls to the linear subsonic and supersonic regimes. Recent developments in computational fluid dynamics make available efficient methods for computing the unsteady transonic aerodynamic loads. To date, the most advanced codes for aeroelastic applications are those based on the potential flow theory. One such code, ATRAN3S,[3,4] is based on a time-accurate, finite-difference scheme. This code has the capability of conducting static and dynamic aeroelastic analysis by simultaneously integrating aerodynamic and structural equations of motion. In this Note, the new capability of ATRAN3S to simulate active control surfaces of fighter wings in the transonic regime is demonstrated.

Coupling of Aerodynamics, Structures, and Active Controls

In this work it is assumed that a control law is known from detailed control theory analysis for a given configuration. The analysis required for the derivation of such a control law is beyond the scope of this work. Using the following procedure, the validity of a given control law in the transonic regime with shock waves can be verified. Necessary corrections can then be applied to the control law so that effectiveness is maintained throughout the transonic regime. To demonstrate this capability, a simplified control law in the time domain is assumed to be

$$\delta(t) = \sum G_j L_j(t) e^{i\psi_j} \tag{1}$$

where $\delta(t)$ is the control surface deflection at time t; G_j the gain factor for the jth term; $L_j(t)$ a selected quantity obtained from the response analysis such as wing deflection, angle of attack or generalized displacements, velocities, and accelerations at t; and ψ_j the phase angle for the jth term.

By representing the active control law in the above form, the coupled phenomenon of structures, aerodynamics, and active controls can be studied in the realistic time domain. This active control equation is coupled with the aeroelastic equations of motion that are derived assuming that the deformed shape of the wing can be represented by a set of discrete displacements at selected nodes. From the modal analysis, the displacement vector $\{d\}$ can be expressed as $\{d\} = [V]\{q\}$ where $[V]$ is the modal displacement matrix interpolated to the finite-difference grid points and $\{q\}$ is the generalized displacement vector.

The final matrix form of the aeroelastic equations of motion is

$$[M]\{\ddot{q}\} + [G]\{\dot{q}\} + [K]\{q\} = \{P\} \tag{2}$$

where $[M]$, $[G]$, and $[K]$ are modal mass, damping, and stiffness matrices, respectively. The aerodynamic force vector $\{P\}$ is defined as $(\frac{1}{2})\rho U_\infty^2 [V]^T[A]\{\Delta C_p\}$ and $[A]$ is the diagonal area matrix of the aerodynamic control points that are the same as the grid points used for the finite-difference modeling of the flow.

The aerodynamic force vector $\{P\}$ in Eq. (2) is obtained using the modified, three-dimensional, small-disturbance, unsteady transonic equation of motion given by

$$\begin{aligned} A\phi_u + B\phi_{xt} &= \\ &[E\phi_x + F\phi^2{}_x + G\phi^2{}_y]_x \\ &+ [\theta_y + H\theta_x\theta_y]_y + [\theta_z]_z \end{aligned} \tag{3}$$

where

$$A = M_\infty^2; \qquad B = 2M_\infty^2; \qquad E = (1 - M_\infty^2);$$
$$F = -(\tfrac{1}{2})(\gamma+1)M_\infty^2; \qquad G = -(\tfrac{1}{2})(\gamma-3)M_\infty^2;$$
$$H = -(\gamma-1)M_\infty^2$$

and ϕ is the disturbance velocity potential. The freestream Mach number is M_∞ and γ is the ratio of specific heats.

This equation is solved using an efficient time-accurate, finite-different method based on the alternating direction implicit scheme. Using this procedure, moving shock waves can be accurately captured. Further details of the simulation of active controls are given in Ref. 5.

Results

The main objective of this Note is to demonstrate the capability and importance of accounting for shock waves in active control analysis for fighter wings. In order to study the basic

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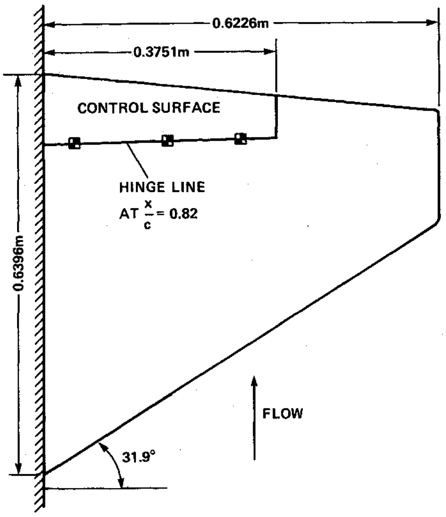


Fig. 1 F-5 wing planform with trailing-edge flap.

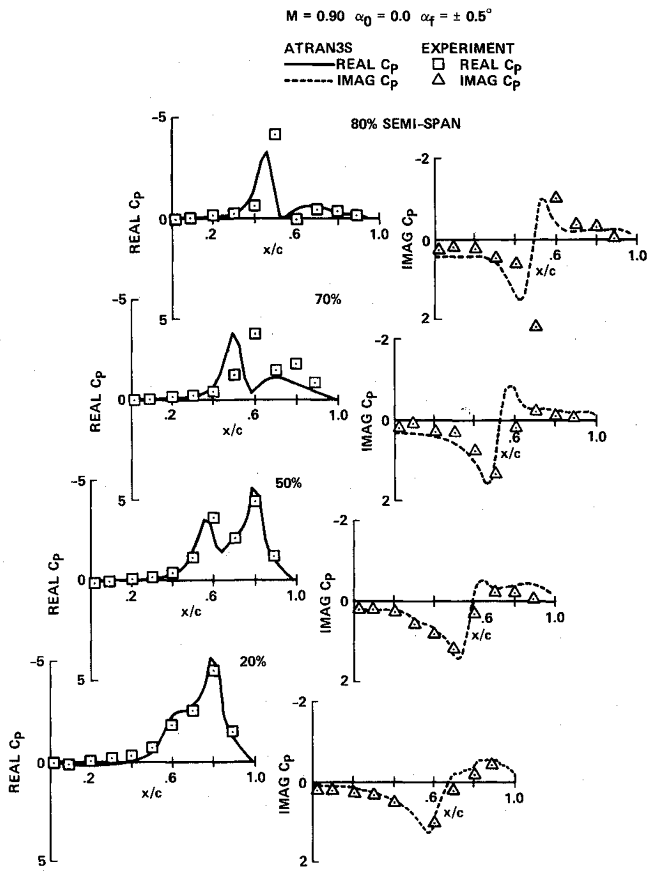


Fig. 2 Unsteady upper surface pressure comparisons at $M = 0.900$ between theory and experiment for the F-5 wing with oscillating control surface.

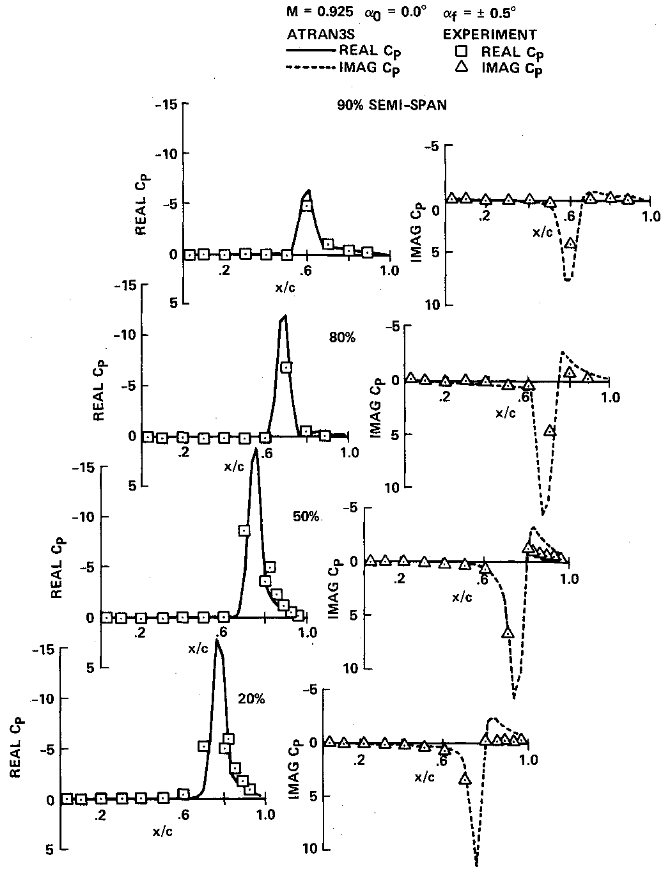
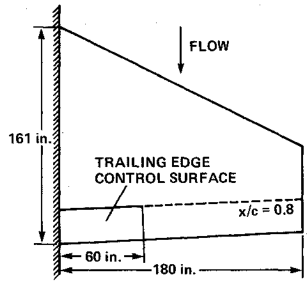


Fig. 3 Unsteady upper surface pressure comparisons at $M = 0.925$ between theory and experiment for the F-5 wing with oscillating control surface.



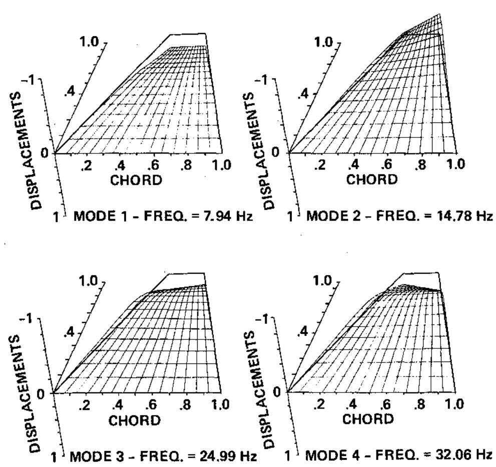


Fig. 4 Mode shapes of a typical fighter wing.

unsteady aerodynamics of flows with shock waves, unsteady transonic computations were made for the F-5 wing with an oscillating control surface and the results are compared with the experiment.[6] The wing planform is given in Fig. 1. Figures 2 and 3 show the unsteady results with the flap oscillating at a frequency of 20 Hz and an amplitude of 0.5 deg at $M = 0.900$ and 0.925, respectively. It is noted that for $M = 0.900$, the shock wave location is forward of the control surface hinge line located at 80% chord. The unsteady pressure distribution on the wing is significantly influenced by the control surface oscillations. For $M = 0.925$, the effect of the control surface oscillations on the wing pressures forward of the shock location is considerably reduced. At this higher Mach number, the shock location has moved aft toward the hinge line. Such rapid changes in unsteady pressures with moving shock waves

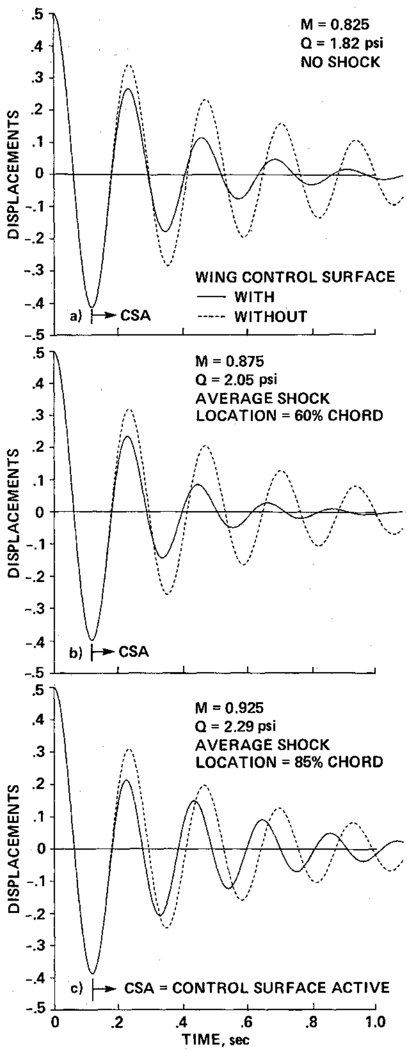


Fig. 5 Aeroelastic responses of the first mode for a typical fighter wing with and without, control surface at $M = 0.825$, 0,875, and 0.925.

will have direct influence on the effectiveness of the active control surface. This phenomenon is demonstrated for a typical fighter wing.

The modes and frequencies of a typical fighter wing are given in Fig. 4. Based on the modal data, aeroelastic response analyses were conducted at the subsonic Mach number of 0.825, and the transonic Mach numbers of 0.850, 0.875, 0.900, and 0.925 for wings with and without control surface. For these cases, a control law that is effective in the lower transonic regime was used. The active control law selected corresponds to $G_1 = -0.0002$, $L_1 = \dot{q}_1$, and $\psi_1 = 0.0$ of Eq. (1). The first modal generalized velocity $\dot{q}_1$ is obtained from solving Eq. (2) by numerical integration.[5] The results from these analyses showed that the control surface with the given control law was effective in the subsonic regime and slightly more effective in the transonic regime when the shock was forward of the control surface hinge line. The first modal responses of the fighter wing cases for the subsonic Mach number of 0.825 and the transonic Mach numbers of 0.875 and 0.925 are shown in Fig. 5. Higher modes were observed to play less significant roles in the overall response and were relatively unaffected by the active controls. By observing the damping rate of the first modal responses, it is noted that the control law remains effective in both the subsonic and transonic regimen up until the shock location has moved aft of the control surface hinge line. At $M = 0.925$, when the shock has moved aft of the control surface hinge line, the control law becomes significantly less effective. This trend was also observed in earlier rectangular wing analyses[5] and is a direct effect of shock wave location. This result further demonstrates the importance of accounting for the presence of a shock wave and its location in designing an effective control law.

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Effects of Contamination on Riblet Performance

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Introduction

THE parametric studies by Walsh and Lindemann and Wilkinson and Lazos,[2] among others, of various riblet geometries established riblets as an effective means for turbulent viscous drag reduction. These favorable experimental results have created an interest in the use of riblets for various applications (e.g., Refs. 3-6). Since many applications would expose the riblet surface to environments that could result in riblet surface contamination, a brief study of the effect of surface contaminants on riblet drag reduction has been conducted.

For riblet flight applications, contamination modes of particular interest include 1) atmospheric particulate matter that could collect on or between the ribs, and 2) contamination from liquids, such as oil and other miscellaneous condensates, that would concentrate in the riblet valleys but, in general, not extend above the riblet peaks. In the present study, thin-element plastic film riblets (provided by the 3M Corporation) were used that had been shown previously to behave similarly to v-groove riblets. Two different methods of contamination were used in an effort to simulate the two contamination modes of interest.

Test Facility and Model Construction

The experiments were conducted in the closed-return, 7×11-in. (178 high $\times 279$ wide $\times 914$-mm long) low-speed wind tunnel at the NASA Langley Research Center over a velocity

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