Engineering Notes

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Incipient Stall Detecton through Unsteady-Pressure Monitoring on Aircraft Wings

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MODERN high-performance aircraft must be able to operate over a broad range of flight regimes, from low-speed landing and takeoff to high-speed level flight and maneuvers. Such aircraft, which typically have swept-back wings of low-aspect ratio, often need to operate at high-lift coefficients, i.e., near stall. As stall is associated with loss of lift—and often with the occurrence of unstable nose up pitching moments—the pilot must be warned of incipient stall.

Some current aircraft are equipped with angle-of-attack indicators that can warn the pilot when he is about to exceed a predetermined angle of attack. For commercial aircraft, these indicators can be set conservatively; for high-performance aircraft, however, the problem is more complex. For high-performance aircraft, there is no unique angle of attack at which stall occurs because maneuvers change the flow patterns over the wings, and thus change the stall conditions. In addition, the presence of the ground affects the flow patterns on the wings of all types of aircraft during takeoff and landing. A simple means for the direct detection of incipient stall is needed to reduce the angle-of-attack safety margin requirements and to increase the operational capabilities of high-performance aircraft.

This Note describes some results of a feasibility study of a system designed for the detection of incipient stall. Such a system would utilize the detection of increased levels of fluctuating pressure associated with flow disturbances o then wing. This approach arises from the expectation that at or near stall there occur, at certain locations, amplitude changes in the fluctuating pressure which are orders of magnitude larger than corresponding changes in the static pressure on the wing surface. Since unsteady pressures can be measured precisely by flush-mounted microphones, a potential stall-warning system could use appropriately processed information to indicate critical flight conditions with a high degree of accuracy.

An experimental program was undertaken to study the various typical flow patterns on a variety of model wings to: a) determine the relation between surface flow patterns and lift characteristics; b) determine strategic locations where easily identified characteristic flow-patterns occur at critical lift conditions; and c) investigate in detail the fluctuating-pressure characteristics at these locations.

Measurements on two-dimensional airfoil configurations were carried out in a low-turbulence, low-noise, wind-tunnel facility. These measurements served to establish the general relations between flow patterns and associated fluctuating pressures.

A NACA 0006 airfoil section of 10-in. chord with circular end plates 14 in. apart—typical thin uncambered airfoil—was exposed to uniform airflow. Figure 1 shows flow patterns typical of this airfoil type at various angles of attack. A mixture of kerosene, titanium-dioxide, and motor oil was applied to the upper wing surface for surface flow visualization. The flow near the centerline was almost purely two-dimensional. Figure 2 shows the position of the reattachment line vs angle of attack; the flow is fully attached up to approximately 4°, the flow is fully separated at 13.5°.

Flush-mounted $\frac{1}{4}$-in.-diam condenser microphones were located at a 20% and a 70% chord location on the centerline. Fluctuating-pressure spectra measured at these locations at one flow speed—(36 fps) for angles of attack α from 0–20°—are shown in Figs. 3 and 4. At $\alpha = 0°$, the microphones are under a laminar attached boundary layer and sense only the environmental background noise. After transition ($\alpha > 1.5°$), the spectra exhibit a haystack shape with a peak frequency that may correspond to a frequency of an oscillatory motion of the reattachment line. This latter frequency decreases as the separation bubble becomes larger. As the reattachment line approaches the microphone location, the peak levels increase and reach their highest value when the reattachment

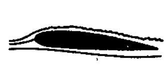
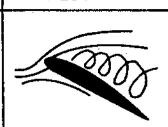
ANGLE OF ATTACK	0° → 1.5°	1.5° → 4°	4° → 13.5°	> 13.5°
FLOW REGIME	LAMINAR ATTACHED	INITIAL BUBBLE FORMATION; TRANSITIONAL FLOW	GROWTH OF SEPARATION BUBBLE	FULLY SEPARATED FLOW
FLOW PATTERN				

Fig. 1 Two-dimensional flow patterns on thin airfoil.

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Index categories: Aircraft Performance; Airplane and Component Aerodynamics; and Aircraft and Component Wind-Tunnel Testing.

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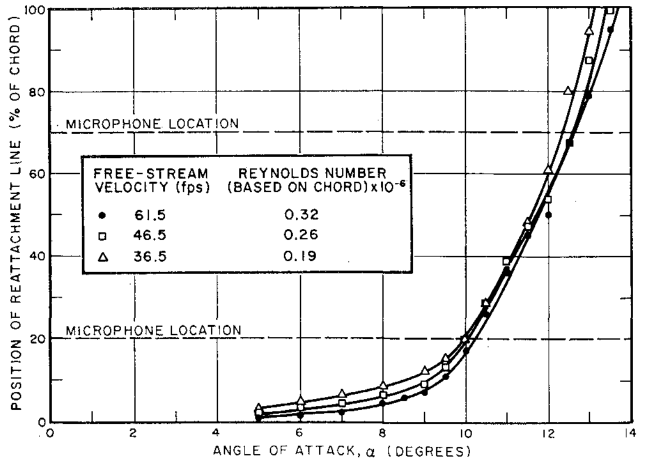

Fig. 2 Location of reattachment line as function of angle of attack for NACA 0006 airfoil.

line oscillates directly above the microphone at $\alpha = 10.5°$ (Fig. 3). As the reattachment line continues to move downstream, the peak frequency decreases only slightly; the levels decrease rapidly but the spectrum shape changes little. The microphone is now far from the reattachment line and well within the separation bubble where no phenomenological changes occur.

The same phenomenological sequence is evident in Fig. 4. Here, the reattachment line passes the microphone location at $\alpha = 12.5°$, and the peak levels are highest for this angle. The decrease in peak frequency for large angles of attack can be deduced from the general shape of the spectra, although the actual peaks were below the measuring range.

One may generalize the results pertaining to the spectra under the reattachment location by normalizing the observed rms pressure p in a band with respect to the freestream dynamic pressure q, and a dimensionless frequency, $S = f.\, l/U_\infty$, where l denotes the distance of the leading edge to the microphone, when located at the reattachment line, and U_∞ represents the freestream velocity. The normalized spectrum has its peak at $S \approx 1$ (Fig. 5).

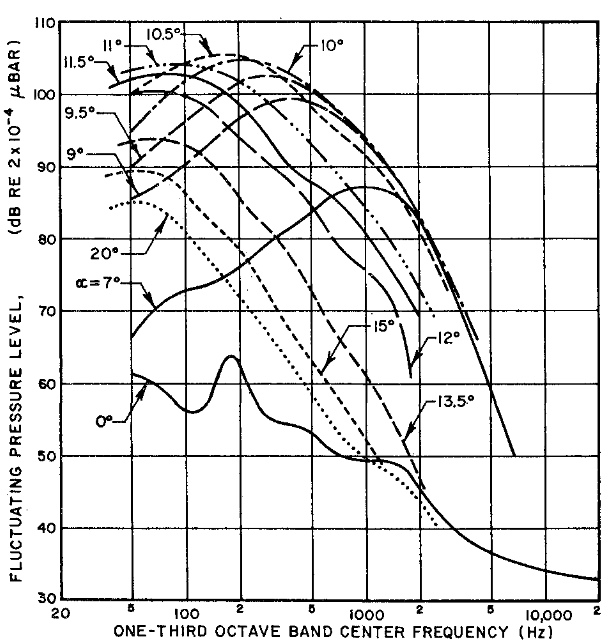

Fig. 3 Fluctuating-pressure spectra at 20% chord location on upper surface of NACA 0006 airfoil, at 36 fps airspeed.

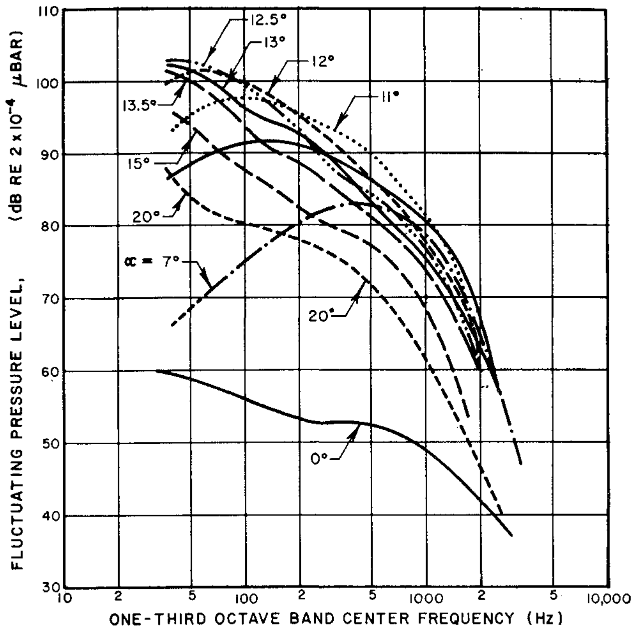

Fig. 4 Fluctuating-pressure spectra at 70% chord location on upper surface of NACA 0006 airfoil, at 36 fps airspeed.

Experiments with a NACA 4415 profile (a typical thick cambered airfoil) showed less distinctive features. Although the growth of the trailing-edge separation region with angle-of-attack increase was substantiated, identification of a distinct separation line was difficult. Corresponding microphone signals showed qualitatively a decrease in peak frequency with increasing angle of attack, probably a result of the progressive thickening of the boundary layer with which an increase in the characteristic size of the turbulent eddies[1] is associated.

Thus, for the wings tested, changes in angle of attack are accompanied by changes in flow characteristics and by changes in fluctuating-pressure-level (FPL) spectra at certain locations. In particular, the rapid change in location of the reattachment line with angle of attack and the dramatic FPL-increase, that is sensed as the line passes over the microphone indicates that pressure spectra should serve as a useful basis for a stall-warning system.

As evident from Fig. 3, even small angle-of-attack changes lead to pronounced changes in the FPL spectra measured at appropriately selected locations on the upper surfaces of unswept wings. One finds, for example, that a change in the

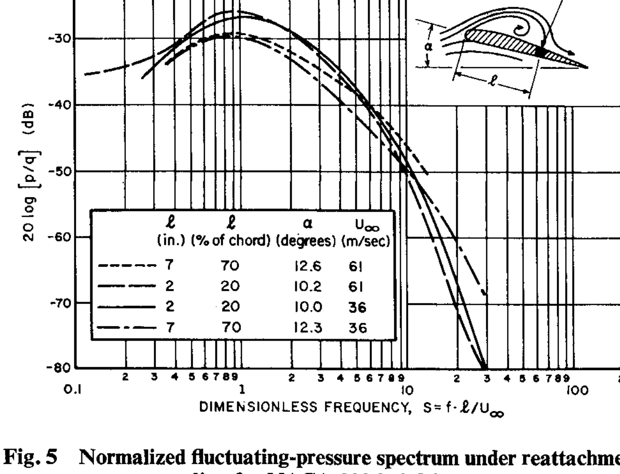

Fig. 5 Normalized fluctuating-pressure spectrum under reattachment line for NACA 0006 airfoil.

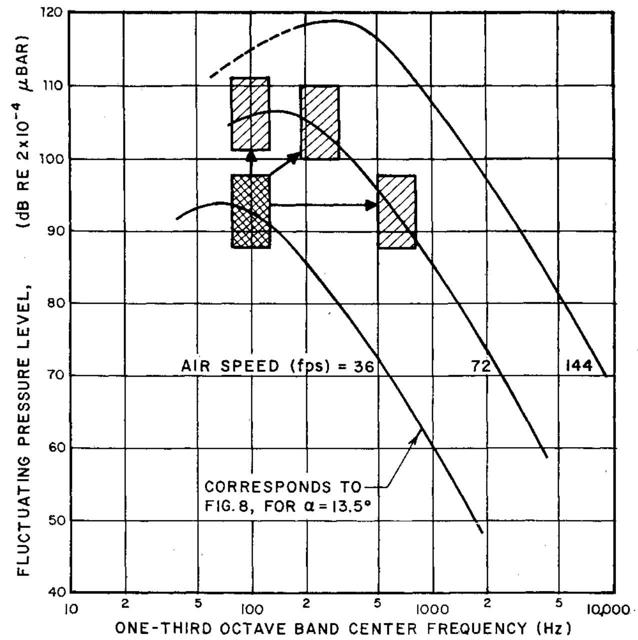

Fig. 6 Shift of typical fluctuating-pressure spectrum with air-speed; possible adjustments of frequency/level detection windows.

angle of attack from 7°–10° increases the pressure level in the 100-Hz frequency band by almost 30 db and that a change from 10°–12° decreases the level in the 1000-Hz band by about 20 db.

It appears that a stall-warning system need not necessarily use the FPL spectrum in the entire audio-frequency range. One may generally obtain positive identification of a spectrum at a given air speed by observing the fluctuating-pressure signal in only two one-third octave bands.

In the design of a warning system for stall at a wide range of speeds, the effects of flowspeed are also an important consideration. In scaling the FPL spectra, one takes the flow speed into account by referring the fluctuating pressure to the freestream dynamic pressure $q = \rho U_\infty^2/2$ and referring the frequency to U_∞/l (where l represents a characteristic length dimension). Figure 6 indicates how a spectrum may be expected to change with airspeed and also shows a number of possibilities as to how one might shift a level/frequency detection window of a stall-warning system to account for these speed-associated spectrum shifts.

Thus, detection of incipient stall by means of microphones flush-mounted at properly selected locations on the upper surfaces of aircraft wings appears feasible. The greatest and most easily detectable changes in the fluctuating-pressure spectrum sensed by a microphone occur as the flow reattachment line passes over the microphone; therefore, microphones located where reattachment occurs at a critical angle of attack (at a given speed) are particularly well-suited for detecting when the aircraft reaches that critical angle.

In designing a stall-detection system, one must also ensure that acoustic noise, such as results primarily from the propulsion system, does not mask the fluctuating-pressure signal used as the basis for stall detection. In many aircraft, the fluctuating-pressure spectrum at reattachment greatly exceeds that due to noise and thus the problem of masking is resolved; however, where this condition does not exist, additional signal processing (e.g., narrowband filtering, correlation, or averaging of signals from two sensors) may be used to extract the desired signal.

References

[1] Heller, H. H., Bliss, D. B., Ungar, E. E., and Widnall, S. E., "Feasibility of Aircraft Stall Detection by Means of Pressure Fluctuation Measurements," AFFDL-TR-70-147, Bolt, Beranack and Newman, Cambridge, Mass., Nov. 1970.

Turbulent Coaxial Jet Mixing in a Constant Area Duct

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Nomenclature

P_T = total pressure
r = radial coordinate
r_e = radius of external duct
r_j = radius of inner jet
T_T = total temperature
u = axial velocity component
χ = axial coordinate
ρ = density
ε = eddy viscosity
μ = turbulent viscosity $\mu = \rho\varepsilon$
$\bar{\mu}$ = nondimensionalized turbulent viscosity $\bar{\mu} = \mu/\rho_j u_j r_j$
$(\ )_e$ = initial external stream conditions
$(\ )_j$ = initial inner jet stream conditions
$(\ )_L$ = properties on the centerline
$\overline{(\ )}$ = nondimensionalized by respective jet values

Introduction

THE mixing process which takes place when two moving streams come into contact is a flow problem which has application to many areas of fluid mechanics. For example, such mixing occurs in combustion chambers, jet pumps, and various propulsion systems. This present study is concerned with the turbulent mixing between two compressible, axisymmetric, coaxial jets confined in a constant area duct.

The flowfield in the duct consists of two distinct mixing regions: 1) a potential core region in which the flow properties are constant along the centerline, and mixing occurs only in the radial direction; and 2) a main mixing region in which flow properties change along the centerline and in the radial direction. This flowfield is analogous to that of a freejet mixing system in which the outer stream is infinite.

The freejet mixing process is fairly well understood. Many experiments have been verified by solutions of the turbulent boundary-layer equations utilizing the turbulent viscosity concept. Most of the experimental data for free mixing and the various models used in representing turbulent transport are summarized and discussed by Schetz[1] and Harsha.[2]

On the other hand the process of ducted coaxial jet mixing is not very well understood. In this study experimental centerline and radial distributions of total pressure are compared with theoretical predictions for compressible air/air mixing. Good agreement exists between theory and experiment provided the proper value of the turbulent viscosity is used in the calculations.

Theory

The analytical approach taken in this study is an explicit finite difference solution[3] of the turbulent boundary-layer equations. The theory directly yields axial and radial distributions of velocity and static temperature throughout the flowfield. The static pressure is assumed to be a function only of the axial coordinate, and is obtained by an iteration procedure which is based on the total mass flow rate through the duct. With the velocity, temperature, and static pressure known, the total pressure is obtained by assuming locally isentropic flow. The boundary layer, which builds up on the external duct wall, is approximated by an incompressible flat plate model.

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