Technical Notes

Brief discussions of previous investigations in the aerospace sciences and technical comments on papers published in the AIAA Journal *are presented in this special department. Entries must be restricted to a maximum of 1000 words, or the equivalent of one* Journal *page including formulas and figures. A Discussion will be published as quickly as possible after receipt of the manuscript. Neither the AIAA nor its editors are responsible for the opinions expressed by the correspondents. Authors will be invited to reply promptly.*

Dynamic Stall Inception Correlation for Airfoils Undergoing Constant Pitch Rate Motions

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Introduction

IN recent years the study of unsteady aerodynamics in the stall and poststall regimes has become important. On the one hand, it is important to be able to predict the actual "stall margin" for devices such as axial flow turbines and compressors that may be experiencing unsteady flow situations due to inlet distortions, etc. On the other hand, it is important to understand how the occurrence of dynamic stall affects helicopter and vertical axis wind turbine performance and how it might be exploited in cases such as the supermaneuverability concept of poststall turns of fighter aircraft.[1] While there are a number of issues that can be raised with regard to unsteady stall characterization, an important one is stall inception, which is the topic of the present paper.

Most of the experimental data obtained for unsteady separated flow over airfoils have been obtained from oscillating airfoils undergoing relatively small sinusoidal pitch oscillations (± 1-± 10 deg) about a relatively low mean angle of attack (0-15 deg) as typified by the experiments reported by McCroskey and Philippe,[2] McAlister and Carr,[3] Martin et al.,[4] and Robinson and Luttges.[5] The types of data obtained in these studies include flow visualization data, hot-film and hot-wire data, and surface pressure measurements. Such airfoil motions are, of course, applicable to many of the fluid devices studied in the past. On the other hand, there are few experimental data available for situations in which an airfoil undergoes very large angle-of-attack excursions that take it into deep dynamic stall. Applications such as the recently conceived "supermaneuverability" of fighter aircraft[1] require a more thorough understanding of dynamically stalled airfoils at angles of attack that may exceed 45 deg. In addition, the motion of the airfoil for this application will perhaps be more closely related to a constant pitch rate ($\dot{\alpha}$) motion as opposed to a sinusoidal motion.

A limited amount of experimental data have been obtained for airfoils undergoing constant pitching rate motions up to moderate angles of attack of at least 30 deg. These works include the study of Harper and Flanigan,[6] who obtained force balance data on a small aircraft model pitching up to 30 deg, the work of Ham and Garelick,[7] who obtained surface pressure measurements on an airfoil pitching up to 30 deg, and the work of Francis et al.,[8] who obtained surface pressure measurements on an airfoil pitching up to 60 deg. None of these works contain any flow visualization data. Deekens and Kuebler[9] obtained flow visualization data from a NACA 0015 airfoil and observed the dynamic leading-edge separation phenomenon for several low Reynolds numbers (less than 3×10^4) and nondimensional pitch rates up to 0.26. Daley[10] obtained leading-edge dynamic stall data for Reynolds numbers up to 3×10^5 and nondimensional pitch rates up to 0.06. Walker et al.[11] obtained flow visualization data along with some hot-wire data on a NACA 0015 airfoil undergoing constant pitch rate motions. These data were obtained for Reynolds numbers on the order of 4.5×10^4 and nondimensional pitch rates up to 0.30. Although correlations for the onset of leading-edge stall were not obtained in these works, it has been noted by Jumper[12] that the data[9,10] tend to collapse when the dynamic stall angle minus the static stall angle is plotted vs the nondimensional pitch rate.

Present Experiment

Data were taken in the Texas Tech tow tank, which is approximately 5 m wide and 10 m long. Water depth was maintained at 1.3 m. A NACA 0015 airfoil with a 35.6-cm chord was mounted vertically from a carriage, which imparted translational motion as well as constant rates of pitching about the quarter-chord. The airfoil was operated at a Reynolds number of 1×10^5 with nondimensional pitch rates ranging from 0.1 to 1.0.

Flow visualization data were obtained by producing hydrogen bubbles on a "ladder" wire stretched across the flowfield 45 cm below the free surface. The wire and a 35-mm camera were mounted so as to translate with the airfoil. The flow appeared to be very two-dimensional, judging from visual data obtained by observing hydrogen bubble streaks in the vertical plane. This is not surprising since the "effective" airfoil aspect ratio tends to be large due to the close proximity of the airfoil end to the bottom wall and also to the penetration of the free surface by the airfoil, which doubles the aspect ratio based on the submerged portion. In addition, the "effective" aspect ratio (based on equivalent downwash at midspan) is more than an order of magnitude higher due to the relatively small clearance between the airoil end and the bottom wall (approximately 7% chord). Typical flow visualization pictures taken during this series of tests are shown in Fig. 1. In the upper picture it can be seen that the flow over the nose of the aifoil has just become detached. The picture taken later shows a large vortex formation in the leading-edge region.

In order to account for variations in the pitch axis location between various sets of data, angle-of-attack data were corrected to reflect the effective angle of attack at the nose due to the freestream and pitching motion. The relationship between the geometric angle of attack α and the nose angle of attack α_N is given by

$$\tan\alpha_N = \tan\alpha - \frac{2\xi}{\cos\alpha}\left(\frac{\dot{\alpha}c}{2U_\infty}\right) \tag{1}$$

where ξ is the pivot point location (fraction of chord from nose), c the airfoil chord, and U_∞ the freestream speed. The

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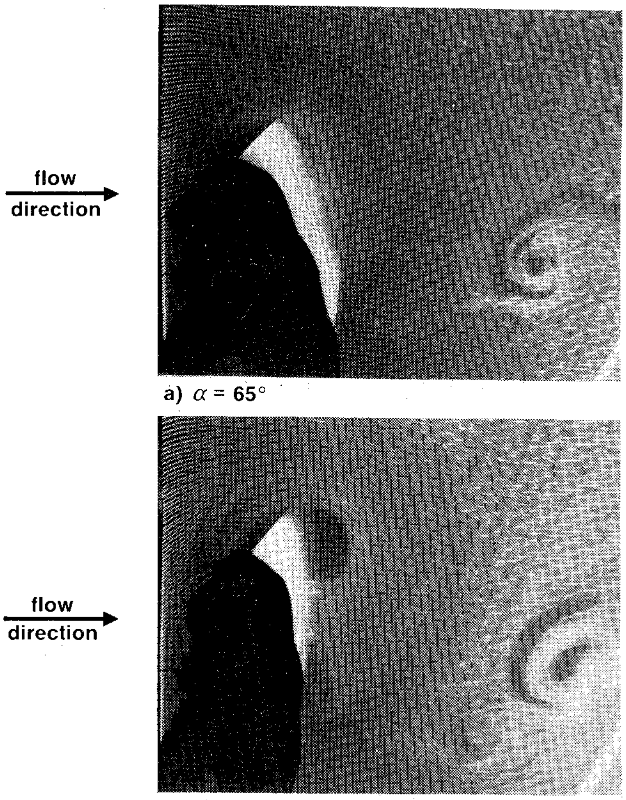

Fig. 1 Typical hydrogen bubble data showing leading-edge stall (Re = 100,000, $\dot{\alpha}c/2U_\infty$ = 0.5, pivot at 25% chord).

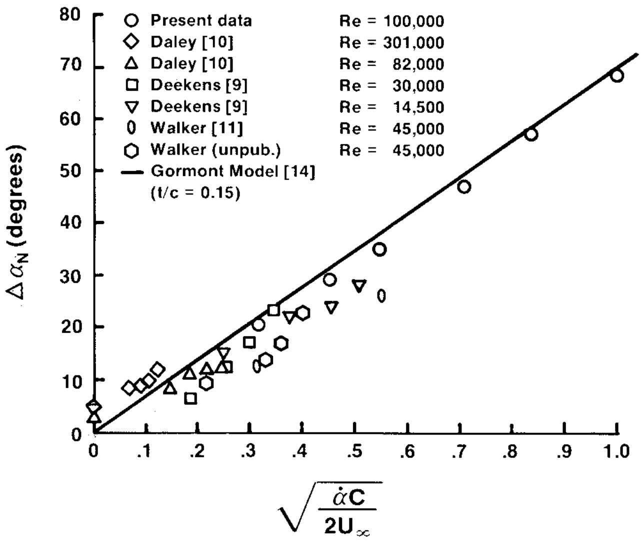

Fig. 2 Stall delay correlation.

effective angle of attack at the nose is as much as 8 deg smaller than the geometric angle in the present set of data.

Results

For the dynamic situation, separation usually occurs suddenly at the leading edge at angles of attack that may greatly exceed the static stall angle. However, for some low Reynolds number cases, boundary-layer separation may occur near the trailing edge as well as at the leading edge, thus forming two separation regions.[11] Data that will be presented subsequently are based upon the occurrence of leading-edge separation.

It is customary to present stall inception data in terms of a stall delay angle defined by

$$\Delta\alpha_{N\,\text{stall}} = \alpha_{N\,\text{dyn. stall}} - \alpha_{\text{static stall}} \tag{2}$$

For some of the data, which will be used subsequently, the static stall angle of attack was not measured. Therefore, a correlation for the static stall angle for a NACA 0015 airfoil was obtained from a curve fit of data due to Jacobs and Sherman.[13] This correlation, which is valid for Jacobs and Sherman data in the range $3\times10^4 < Re < 3\times10^6$, is given in degrees by

$$\alpha_{\text{static stall}} = 1.79\ell n(Re/202) \tag{3}$$

Leading-edge stall results obtained from visual data from several sources,[9-11] including the present work, are shown in Fig. 2. As can be seen from this figure, the stall delay angle at the nose is proportional to the square root of the nondimensional pitching rate. The moderate scatter in the data may be in part due to Reynolds number effects, although no clear trends are discernible. It should also be noted that all of the data except the present data were taken using relatively short-aspect-ratio airfoils; three-dimensional effects may thus play some role in the scatter.

It is interesting to note that a correlation by Gormont,[14] which is based upon oscillating airfoil data, tends to fit the experimental data reasonably well. This correlation is based upon the data of Liiva et al.[15] and Gray et al.[16] for four airfoils (V23010-1.58, NACA 0012 MOD, V13006-0.7, and NACA 0006), which were operated at Mach numbers between 0.2 and 0.6. The Gormont model for low Mach number "moment stall" is given by

$$\Delta\alpha_{N\,\text{stall}} = K_1\gamma(\,|\dot{\alpha}c/2U_\infty|\,)^{1/2}\,\text{sgn}(\dot{\alpha})$$

$$\gamma = 1.0 - 2.5(.06 - t/c)$$

$$K_1 = 3/4 + 1/4\,\text{sgn}(\dot{\alpha}) \tag{4}$$

It should be noted that the Gormont model also includes a correlation for "lift stall," which occurs subsequent to moment stall. The occurrence of moment stall is, however, the appropriate indicator since it represents the *first* major disturbance of the potential flow by the boundary layers. It should also be noted that all of the cases presented in Fig. 2 represent the onset of stall, and thus it remains to be seen whether the Gormont correlation is also valid for the cessation of stall for the constant pitch rate situation.

Acknowledgments

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Simulation of Inviscid Vortex-Stretched Turbulent Shear-Layer Flow

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Introduction

REALISTIC numerical simulations of vortex flows around aircraft require three-dimensional discrete models of significant size.[1] The recent construction of a 16-million-word memory for the CYBER 205 has allowed tests of 1-million-grid-point models on a practical basis within reasonable elapsed times. Prior to this it was hoped that this type of computation could be performed with disk I/O, but experience showed that heavy time penalties were encountered due to the demands of the flow-solving algorithm. Therefore, good virtual-memory-management techniques within a sufficient working set of *real* memory seem to be the only effective way to carry out such large-scale simulations. The other crucial requisite is a large bandwidth communciation channel between the memory and the arithmetic unit.

What do we expect to see in the inviscid simulation as the discrete model grows larger, with more and more grid points added in the discretization? For one thing, the accuracy will improve but, more importantly for the physical understanding, the allowable degrees of freedom in the solution increase so that we may begin to study fundamental flow instabilities such as the interacting of stretched vortices as described by the continuum equations. The advent of supercomputers with very large real memory just now offers the possibility to explore such phenomena numerically.

The type of vortex flowfield we want to simulate develops when a delta-shaped wing meets an oncoming stream of air at a high angle of attack. The flow separates from the wing in a shear layer along the entire sharp leading edge and, under the influence of its own vorticity, coils up to form a vortex over the upper surface of the wing. The high velocities in the vortex create a low-pressure region under it, which gives the wing a nonlinear lift. The flow also separates in a shear layer from the trailing edge of the wing, which forms into a wake vortex. The primary and wake vortices then interact with each other behind the wing. The appropriate model to describe the fluid mechanics of this vortex flow is the compressible Navier-Stokes equations. But if the Reynolds number of the flow is very large—say, over 10 million—the shear stresses and dissipation terms take effect only in very thin layers of the flow on the surface of the wing and across the shear layers that separate from the leading and trailing edges. For this type of flow, it is these free shear layers, and not the boundary layer, that contribute the greatest amount of vorticity. The intent of this paper is to study the dynamics and the stability of these free shear layers by means of numerical simulation. Because the layers are thin and not influenced by the no-slip boundary conditions, the shear stresses and dissipation terms do not have to be accurately represented. They can instead be modeled by simpler nonphysical expressions. In this way the equations we solve are the Euler equations with an artificial viscosity model. The flow, therefore, is inviscid except across thin discontinuities like shock waves and vortex sheets that are

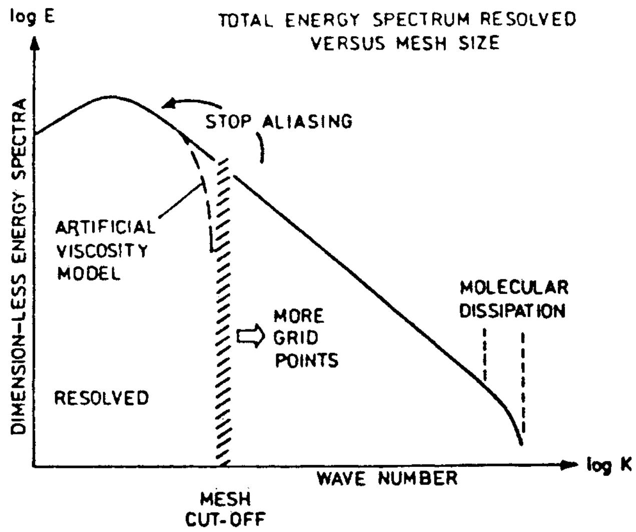

Fig. 1 Energy cascade of high Reynolds number turbulent flow.

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