Technical Notes

TECHNICAL 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).

Effects of Compressibility on Dynamic Stall

K.-Y. Fung*
University of Arizona, Tucson, Arizona 85721
and
L. W. Carr†
NASA Ames Research Center, Moffett Field, California 94035

Introduction

DYNAMIC stall effects are becoming increasingly important to both helicopter and fixed-wing fighter aircraft aerodynamicists as these vehicles are pushed to the limit of their flight envelopes. Many studies of dynamic stall have been made during the last 20 years (see survey by Carr[1]). As the range of the freestream Mach number considered in these studies has kept expanding, mounting evidence has appeared that shows that the dynamic lift that occurs at low Mach number may be significantly limited as the Mach number increases. Mach number effects on dynamic stall have been observed on aircraft[2] where the lift increases due to unsteadiness were found to virtually disappear as the freestream Mach number exceeded 0.6; Dadone[3] found similar results for airfoils oscillating in pitch. This detrimental effect has recently been documented by Lorber and Carta.[4] McCroskey et al.[5] found that a local supersonic region exists in many of the cases in their comprehensive dynamic stall study of various oscillating airfoils but did not observe any shock patterns in shadowgraph studies. Ericsson and Reding[6] argued that a strong shock would soon appear if the Mach number in the experiment were further increased, even though it has not been observed in experiments to date. Based on the assumption of shock-induced stall, they were quite successful in predicting global aerodynamic characteristics. Katary[7] confirmed the existence of a local supersonic zone for pitching motions that exceeded a certain critical pitch angle and frequency. Carr et al.[8] observed that, for low Mach numbers, the flow reversal on an oscillating airfoil could appear at the trailing edge and move upstream, whereas for higher Mach numbers, leading-edge stall would dominate the development of dynamic stall on the same airfoil. Visbal[9] found a local supersonic bubble and shock at $M_\infty = 0.4$ in his calculations using an unsteady Navier-Stokes solver with turbulent modeling. Thus, there is agreement that compressibility can have a significant effect on the dynamic stall loads; however, the actual mechanism by which compressibility causes these effects is not as well defined. The present paper proposes a new interpretation of the experimental results and offers insight into how compressibility can introduce limits on the dynamic overshoot in lift that is so characteristic of dynamic stall.

It will be shown that the effects of frequency on the stall onset are distinctly different, depending on whether the flow has locally exceeded sonic conditions during part of the cycle of oscillation. Based on classical theory, the flow around an airfoil at high angles of attack can reach supersonic conditions, even at low freestream Mach numbers, as long as the flow remains attached. It is suggested that shock-induced effects compete with the dynamic viscous effects occurring in the boundary layer in determining the onset of separation, which can lead to premature dynamic stall and can significantly reduce the maximum dynamic lift that can otherwise be obtained.

Compressibility vs Unsteadiness

A set of unsteady pressure distributions of an NACA-0012 airfoil pitching about the static stall angle chosen from McCroskey et al.[10] to represent typical dynamic stall processes is shown in Fig. 1, where the pressure distribution over the upper surface of the airfoil is plotted at intervals marked by the corresponding instantaneous angles of attack for two different reduced frequencies (defined as $k = \omega C/2U$, where ω is the circular frequency of oscillation, C the chord, and U the freestream speed) and a freestream Mach number of 0.072. The effects of unsteadiness on the onset of stall can be seen in this figure. Starting from a low angle of attack where the flow is attached, one sees that the suction peak rises as the angle of attack increases, the instantaneous pressure distributions resemble their corresponding steady ones, and the static stall angle is passed without the breakdown of the pressure distribution associated with stall. Then, a sudden drop of the pressure peak occurs; the boundary layer has broken away and the flow has separated. Up to the point where the flow separates, the pressure distributions for $k = 0.248$ differ only slightly from those for $k = 0.099$ for the same angle of attack. Note, however, that lower suction peaks occur for the higher frequency at each angle. The separation angle was delayed from 21 to 23 deg due to increased unsteadiness.

The suction peaks that are attained at the leading edge of the oscillating airfoil at three different Mach numbers are presented in Fig. 2 as a function of reduced frequency. Note that at $M_\infty = 0.072$ and 0.184, the $C_{p_{\max}}$, which is proportional to the total lift, increases significantly as the frequency of oscillation increases. However, at $M_\infty = 0.30$, the $C_{p_{\max}}$, for 30 separate cases from the dynamic stall experiments of McCroskey et al.,[10] stays at roughly the same value, -9.0, regardless of pitch rate, amplitude, or mean angle of pitching. Although the total dynamic lift may continue to increase due to the separation vortex as it moves downstream over the airfoil, the maximum suction peak drops rapidly after locally sonic conditions are reached. Figure 3 shows the maximum suction peaks (the symbols) measured in McCroskey et al.[10] as a function of freestream Mach number for three airfoils undergoing the

Received Dec. 7, 1988; revision received Dec. 4, 1989; accepted for publication Dec. 6, 1989.

*Associate Professor, Department of Aerospace and Mechanical Engineering. Member AIAA.

†Group Leader, Unsteady Viscous Flow, U.S. Army Aeroflightdynamics Directorate and Fluid Dynamics Research Branch, NASA. Member AIAA.

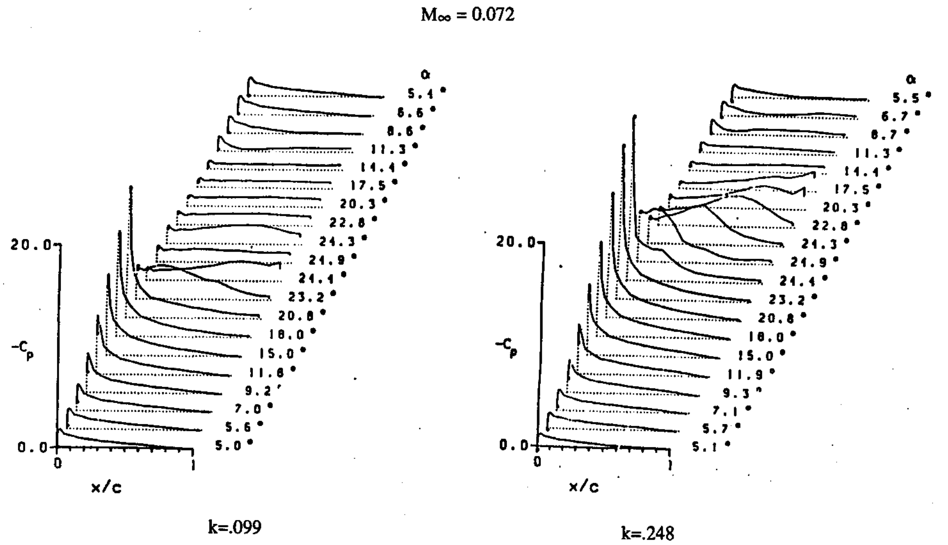

Fig. 1 Dynamic data from McCroskey et al.[10] for two frequencies and M_∞ = 0.072 for the NACA-0012 airfoil.

same oscillatory motion. The solid line represents the critical C_p at which the flow becomes locally sonic. For a given airfoil, e.g., the NACA 0012, the maximum suction peak attained before the onset of stall increases with the Mach number up to the critical C_p, then it decreases rapidly as the critical C_p decreases with M_∞. It becomes clear that a change in the physics of the dynamic stall process is occurring, a change brought on by the increasing impact of compressibility on the flow.

An analysis based on classical theory shows that a 12% Joukowsky airfoil will be critical at α = 9 deg and a freestream Mach number of 0.3. This airfoil will reach critical conditions at M_∞ = 0.2 and α = 17 deg, which is not unusual for a helicopter blade at forward flight conditions. Beyond the critical conditions, any increases in Mach number or angle of attack will result in the development of a local supersonic zone and a shock, albeit weak. Shock-free flows are unlikely according to the Morawetz theorems,[11-13] which state that shock-free flows are isolated and it is impossible to find shock-free flows for any continuous range of angles of attack. For the range of reduced frequencies and Mach numbers considered in dynamic stall, the local inviscid flowfield can be considered frozen because the time scale for the formation of a shock is at least four orders of magnitude smaller than the frequency of oscillation.

For flows of low freestream Mach number and high angle of attack, the location of the shock is estimated within 1% of the chord at the leading edge. The boundary layer at this location is most likely laminar. The boundary layer may become unstable and break up simply because of the high adverse pressure gradient of a steepening compression even before a shock is formed. Our knowledge of shock boundary-layer interaction is limited, particularly for cases where the shock is at the leading edge. Without sufficient experimental data, it would be difficult to ascertain the actual process of separation for this case.

Table 1 lists the case number, angle of attack, maximum suction peak, corresponding C_ℓ value, reduced frequency k, mean angle of oscillation α_0, and freestream Mach number M_∞ of the first 14 tests where flow separation had occurred in part of the pitching cycle reported by McCroskey et al.[10] for the NACA-0012 airfoil. There is a significant consistency of the conditions immediately preceding separation in these cases. The maximum suction peaks were all around −9.0, the corresponding C_ℓ values around 1.5, and the angle of attack around 14 deg, regardless of the differences in frequency, mean angle, and amplitude of oscillation. Taking into account all the uncertainties in the experiments and the discrete nature of data recording, this is a strong indicator that separation is

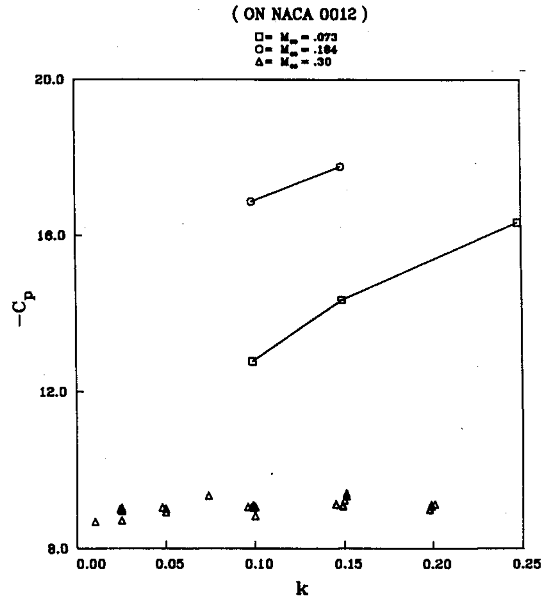

Fig. 2 Effects of frequency on suction peak ($C_{p_{max}}$) at stall.

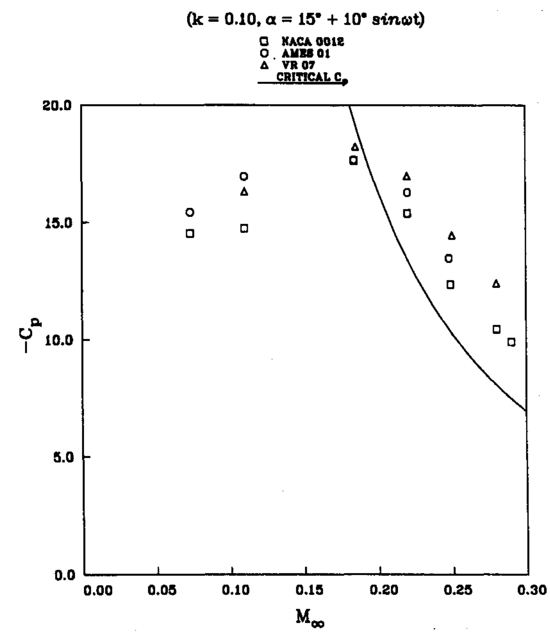

Fig. 3 Effect of M_∞ on maximum suction peak.

Table 1 Maximum suction peaks, corresponding α, and C_ℓ found in McCroskey et al.[10] for the NACA-0012 airfoil stalling over a range of reduced frequencies, mean angles, and amplitudes of oscillation

Case	α	$C_{p_{\max}}$	C_ℓ	k	α_0	M_∞
7021	13.7	−9.06	1.47	0.100	8.99	0.299
7023	13.6	−9.13	1.53	0.201	8.99	0.299
7101	13.8	−9.23	1.51	0.150	8.99	0.301
7112	13.8	−9.03	1.41	0.025	9.97	0.301
7113	13.8	−9.10	1.51	0.099	9.95	0.301
7114	13.8	−9.08	1.56	0.199	9.91	0.301
7117	14.0	−8.98	1.41	0.025	10.92	0.301
7118	13.9	−9.01	1.47	0.050	10.92	0.301
7119	13.9	−9.05	1.52	0.099	10.92	0.301
7120	14.1	−9.08	1.54	0.149	10.96	0.301
7121	14.1	−8.99	1.57	0.198	10.88	0.301
7200	14.0	−8.97	1.40	0.025	11.92	0.301
7205	14.0	−9.07	1.52	0.099	11.92	0.302
7207	14.4	−9.01	1.59	0.198	11.88	0.302

being caused by a new phenomenon not seen at low Mach number.

Conclusions

From the observed behavior of the flow before separation, we have shown that the parametric dependency of separation on frequency for supercritical flows is different from that for subcritical flows. For subcritical flows, increasing the reduced frequency delays separation of the boundary layer and, hence, allows the airfoil to attain higher lift values at higher angles of attack. However, as the airfoil assumes lift values higher than the value at static stall, the flow around it can easily reach supercritical conditions. The formation of a local supersonic region and the associated shock can occur at a location close to the leading edge where the radius of curvature is small and the boundary layer laminar. The vortical content of the flow is intensified due to the relatively short extension of the local supersonic region. The local outer flow and the boundary layer are no longer stable. Hence, compressibility effects pose a limit on lift enhancement by increasing unsteadiness.

Our studies here suggest that it is important to consider compressibility effects for freestream Mach numbers as low as 0.2.

Acknowledgments

Most of the research reported here was carried out during the first author's sabbatical at the NASA Ames Research Center. He would like to thank NASA for support under the NASA Ames University Consortium Grant NCA2-196, monitored by Sandford Davis, and the Air Force Office of Scientific Research (AFOSR) for support under AFOSR Grant 83-0071, monitored by James D. Wilson, and AFOSR Grant 88-0163 under Hank Helin. We are thankful to Thomas Pulliam for his technical advice on using the ARC2D codes and to Joan Thompson for her help in the generation of the graphics presented here and elsewhere.

References

[1]Carr, L. W., "Progress in Analysis and Prediction of Dynamic Stall," *Journal of Aircraft*, Vol. 25, No. 1, 1988, pp. 6–17.

[2]Harper, P. W., and Flanigan, R. E., "The Effect of Change of Angle of Attack on the Maximum Lift of a Small Model," NACA TN-2061, March 1950.

[3]Dadone, L. U., "Two-Dimensional Wind Tunnel Test of an Oscillating Rotor Airfoil," NASA CR-2914, Dec. 1977.

[4]Lorber, P. F., and Carta, F. O., "Airfoil Dynamic Stall at Constant Pitch Rate and High Reynolds Number," AIAA Paper 87-1329, June 1987.

[5]McCroskey, W. J., McAlister, K. W., Carr, L. W., Pucci, S. L., Lambert, O., and Indergrand, R. F., "Dynamic Stall on Advanced Airfoil Sections," *Journal of the American Helicopter Society*, Vol. 26, No. 3, 1981, pp. 40–50.

[6]Ericsson, L. E., and Reding, J. P., "Shock-Induced Dynamic Stall," *Journal of Aircraft*, Vol. 21, No. 5, 1984, pp. 316–321.

[7]Katary, M., "An Experimental Study of the Development of a Supersonic Zone Near the Leading Edge of an Airfoil Oscillating in Subsonic Flow," AIAA Paper 83-2133, Aug. 1983.

[8]Carr, L. W., McCroskey, W. J., McAlister, K. W., Pucci, S. L., and Lambert, O., "An Experimental Study of Dynamic Stall on Advanced Airfoil Sections, Vol. 3, Hot-Wire and Hot-Film Measurements," NASA TM-84245, Dec. 1982.

[9]Visbal, M. R., "Effects of Compressibility on Dynamic Stall of a Pitching Airfoil," AIAA Paper 88-0132, Jan. 1988.

[10]McCroskey, W. J., McAlister, K. W., Carr, L. W., and Pucci, S. L., "An Experimental Study of Dynamic Stall on Advanced Airfoil Sections," NASA TM-84245, Sept. 1982.

[11]Morawetz, C., "On The Non-Existence of Continuous Transonic Flow Past Profiles, I," *Communications on Pure and Applied Mathematics*, Vol. 9, 1956, pp. 45–68.

[12]Morawetz, C., "On The Non-Existence of Continuous Transonic Flow Past Profiles, II," *Communications on Pure and Applied Mathematics*, Vol. 10, 1957, pp. 107–131.

[13]Morawetz, C., "On The Non-Existence of Continuous Transonic Flow Past Profiles, III," *Communications on Pure and Applied Mathematics*, Vol. 11, 1958, pp. 129–144.

Observations of Dynamic Stall Phenomena Using Liquid Crystal Coatings

Daniel C. Reda*
Sandia National Laboratories, Albuquerque, New Mexico 87185

Introduction

UNSTEADY boundary-layer flows occur on airfoil surfaces of wind energy conversion systems, rotary-wing aircraft, and maneuvering fixed-wing aircraft. In such flows, the airfoil angle of attack varies with time, and both boundary-layer transition and turbulent separation locations can undergo extensive and rapid movements, particularly on the airfoil lee surface. For transient angle-of-attack excursions beyond an airfoil's static-stall limit, complex leading-edge-separation/vortex-shedding ("dynamic stall") phenomena occur. An excellent review of the state-of-the-art concerning dynamic stall phenomenology was recently given by Carr.[1]

In the present research, newly formulated (shear-stress-sensitive/temperature-insensitive) liquid crystal coatings[2] were applied to the surface of an oscillating airfoil in order to investigate the unsteady fluid physics associated with the dynamic stall process. Surface-mounted microtufts and laser-sheet/smoke-particle flow visualization were also utilized to complement the liquid crystal technique. Boundary-layer transition and turbulent separation locations were measured as a function of geometric angle of attack, and results are presented in comparison with predictions generated with the Eppler airfoil design code.[3]

Received July 24, 1989; revision received Dec. 30, 1989; accepted for publication Feb. 5, 1990.

*Senior Member Technical Staff; currently Fluid Dynamic Research Branch, MS260-1, NASA Ames, Moffett Field, CA. Associate Fellow AIAA.