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CONTROL OF UNSTEADY SEPARATED FLOW ASSOCIATED WITH THE DYNAMIC STALL OF AIRFOILS

Michael C. Wilder

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Control of Unsteady Separated Flow Associated with the Dynamic Stall of Airfoils

M. C. Wilder

Summary

This is the final report for the MCAT Institute research proposal no. MCAT 91-22, covering the period March 2, 1992 - July 15, 1992. The two principal objectives of this research were, (1) achieving an improved understanding of the mechanisms involved in the onset and development of dynamic stall under compressible flow conditions, and (2) the investigation of the feasibility of employing adaptive airfoil geometry as an active flow control device in the dynamic stall regime.

The results of compressible dynamic stall experiments involving both oscillating and transiently pitching airfoils were reported in the MCAT Institute Progress Report no. NCC2-637 by S. Ahmed (ref. 1). The dynamic stall phenomenon was examined by employing schlieren flow visualization, laser Doppler velocimetry (LDV) measurements, and point diffraction interferometry (PDI) for a sinusoidally oscillating airfoil, and by using schlieren flow visualization for a transiently pitching airfoil executing a constant-pitch-rate ramp-up motion from 0 to 60 degrees angle of attack. The results of these studies, along with a discussion of the design methodology, fabrication scheme, and proposed deformation schedule for an adaptive geometry airfoil were presented in the progress report (ref. 1).

Presented in this final report are the results of a quantitative (PDI) study of the compressibility effects on dynamic stall over the transiently pitching airfoil, as well as a discussion of a preliminary technique developed to measure the deformation produced by the adaptive geometry control device, and bench test results obtained using an airfoil equipped with the device.

Nomenclature

c	chord length
M	Mach number
U_∞	freestream velocity
α	angle of attack
$\dot{\alpha}$	pitch rate (radians/sec)
α^+	nondimensional pitch rate = $\frac{\dot{\alpha}\ c}{U_\infty}$

Introduction

The utilization of dynamic stall as a method for increasing the maneuverability and agility of aircraft has received significant attention during the past few years. When an airfoil is rapidly pitched beyond the static stall angle, a dynamic stall vortex forms near the leading edge, resulting in a dynamic lift overshoot. This dynamic lift, unfortunately, is short lived, and the benefits are lost as the vortex propagates past the trailing edge. Several methods, such as leading edge slats (ref. 2), moving walls (ref. 3), suction and blowing (ref. 4), and leading edge deformation (ref. 5) have been investigated for their ability to delay the formation and propagation of the dynamic stall vortex, and with the advent of so called 'smart materials', the concept of dynamically varying the shape of an aerodynamic surface during a maneuver is becoming feasible. A material developed at NASA Ames Research Center (ref. 6) is being investigated for its ability to provide active flow control. Originally developed as an electro-expulsive de-icing device for aircraft wings, the idea here is to employ the material to dynamically adapt the leading edge geometry of the airfoil in such a way as to delay or prevent the onset of stall.

Before attempting to control dynamic stall, it is first necessary to have a thorough understanding of the mechanisms responsible for the formation and development of the dynamic stall vortex. Dynamic stall is a complex phenomenon which has been shown to depend on a variety of parameters, including, the airfoil shape, the leading edge geometry, the degree of unsteadiness, the Reynolds number, and the free stream Mach number to name

a few (ref. 7). Compressibility effects have been shown to change the way that dynamic stall develops, thus a better understanding of these effects has been of importance and interest in the development of supermaneuverable and highly agile aircraft. It is well known (ref. 8) that the effects of compressibility set in at low freestream Mach numbers (M = 0.2 - 0.3) on airfoils operating at high lift levels, due to the development of extremely strong suction peaks near the leading edge. These suction peaks are strong enough, in fact, to accelerate the local flow to supersonic speed. The fact that dynamic lift still persists even when these compressibility effects appear (ref. 9) supports the argument that the benefits of dynamic stall can be exploited in flight systems.

Compressible Dynamic Stall over a Transiently Pitching Airfoil

An extensive investigation of an airfoil undergoing a constant-pitch-rate maneuver was performed, using the technique of point diffraction interferometry (PDI). The investigation was carried out in the Compressible Dynamic Stall Facility (CDSF) of the Fluid Mechanics Laboratory at NASA Ames Research Center. The PDI technique produces constant density interference fringe patterns which are recorded photographically. The interference fringe patterns quantitatively map the global flow characteristics, as well as the surface flow details, and local pressure and Mach number are obtainable using the isentropic flow relations. The PDI technique lends itself especially well to mapping unsteady surface pressure distributions on an airfoil since the number of fringes (and hence, the number of stations at which the pressure is determined) naturally increases as the pressure gradient increases. In this investigation, a 3in. chord length NACA 0012 airfoil was pitched about its quarter chord point from 0 to 60 degrees angle of attack for Mach numbers ranging from 0.2 to 0.45 and for pitch rates between 2000°/sec and 3600°/sec. Tables Ia and Ib indicate the range and combination of parameters investigated. The results indicate that, even though the maximum suction pressure on the airfoil for a given instantaneous angle of attack is dependent upon both pitch rate and Mach number, the maximum coefficient of pressure obtainable (just prior to stall) depends only on the freestream Mach number, and decreases with increasing Mach number (see Figs. 5 and 6 of Appendix A). Pitch rates up to 3600°/sec were examined, and at these high rates locally supersonic flow is obtained over the leading edge, even for moderate freestream Mach numbers.

The observations also showed the formation of multiple shocks on the leading edge at high pitch rates, and the presence of multiple vortices at low pitch rates, confirming the results of earlier schlieren studies (ref. 10).

The results of this investigation are presented in Appendix A in the form of an extended abstract submitted to the 31st AIAA Aerospace Sciences Meeting, to be held January 1993. This Appendix also provides a description of the experimental facilities, the constant pitch-rate apparatus, and the PDI technique.

Adaptive Geometry Flow Control Device

The PDI investigations have revealed that the leading-edge pressure gradients, which ultimately lead to separation, develop less rapidly in the dynamic cases than in the static case. In order to make use of the beneficial effects of the dynamic lift generated, however, the development of these gradients must be delayed still further. An adaptive leading-edge geometry is being examined for its ability to provide this delay. As an active flow control device, it is envisioned that the leading edge will dynamically increase in thickness as the airfoil executes the pitch-up maneuver. This "Dynamically Deforming Leading Edge", or DDLE for short, will be constructed of a material developed at NASA Ames Research Center as an electro-expulsive de-icing device for aircraft (refer to Fig. 1 and ref. 6). The material consists of two conductive strips embedded in elastic sheets. When charged, the electromagnetic force induced within the strips causes them to repel one another making the elastic sheets bulge; the greater the applied charge, the larger the bulge.

A requisite first step in developing this concept was to precisely determine the nature of the deformation produced by this material. A sample of the material, attached to the leading edge of a 10 in. chord NACA 0012 airfoil, was provided by the Civil Technology Office and used in a series of bench tests. In these tests, the deformation was imaged on 3 x 4 in. high speed Polaroid sheet film using an IMA-CON camera. The camera recorded eight to twelve instantaneous images on each sheet of film, at a framing rate of 25,000 frames per second. Figure 2 is a representative Polaroid image with the first frame occurring 1.25ms after the start of the deformation. The camera and a strobe

light were triggered by the same pulse generator which drove the leading edge deformation, however, the camera/strobe trigger pulse was passed through an adjustable delay circuit. This delay circuit allowed the camera to capture any portion of the complete cycle of the deformation in fine detail (the time between images was 0.04ms, while the deformation cycle took approximately 3 ms). The experimental setup is shown in Fig. 3a and Fig. 3b schematically illustrates the experiment. An orthogonal grid of retro-reflective tape, applied to the surface of the airfoil, provided a reference for quantifying the surface deflections. The photographs were digitized and the shape of the grid lines were recorded using image processing software available on the IRIS work station. Shown in Figs. 4 and 5, respectively, are a sequence of the digitized photographic images, and the same sequence shown as line plots. The data for the line plots were obtained via the image processing software. Figure 6 is the profile view of the mid-span grid line showing the maximum and minimum deformation relative to the undeformed (neutral) surface. These data have also been animated using the C-graphics library subroutines available on the Personal IRIS workstation.

The sample of the material utilized in these bench tests contains only one conductive strip, and the deformation is clearly three-dimensional. These results are being employed to improve the material design in order to produce the desired span-wise two-dimensional deformation, which will increase the thickness of the leading 25% of the airfoil chord length.

The dynamically deforming leading edge will be incorporated in a 6 in. chord airfoil, which is twice the length of the airfoil employed in the previous dynamic stall tests performed in the CDSF. As was described in the progress report (ref. 1), design calculations have been performed to check the adaptability of the larger airfoil to the test facility. A stronger model mount was designed to support the increased loads, which necessitated reducing the size of the glass windows in the test section. Replacing the present 6 in. diameter windows will be 2 x 3 in. rectangular windows. Rectangular windows have been chosen to reduce cost and simplify manufacturing and assembly procedures; particularly desirable for a proof-of-concept study such as this. To insure that the rectangular windows will have no adverse effect on the formation of PDI interference fringes, an interferogram was produced with a 2 x 3 in. mask placed over the existing round windows. This interferogram is shown, in Fig. 7,

in comparison with one made under the same conditions but using the round windows. Less light reaches the film plane due to the smaller window area (this will be corrected for by focusing the laser beam to a correspondingly smaller area), but no anomalies were observed in the fringe pattern.

Conclusions

Interferograms taken during the pitch-up of an airfoil in a moderately compressible flow have offered new insight into the character of the dynamic stall vortex occurring under compressible flow conditions. Preliminary analysis suggests that this vortex is significantly different from that seen in incompressible dynamic stall for airfoils undergoing ramp motion. In fact, the later stages of development of the dynamic stall process is clearly affected by the ramp-motion process when compared to oscillating airfoil behavior even in compressible flow.

A material has been examined which may have the potential to be used as an adaptive geometry active flow control device. Incorporated in the leading edge of an airfoil, the material will allow the leading edge thickness to be varied dynamically. A technique for measuring the time varying shape of the dynamically deforming leading edge has been developed. The measurements were employed to suggest refinements to the material design in order to produce a more controllable deformation.

Acknowledgements

The funds for this research were provided, through the NASA contract no. NCC2-637, from the Navy-NASA Joint Institute of Aeronautics, AFOSR, ARO, and NAVAIR. The guidance and valuable expertise provided by Dr. M. S. Chandrasekhara, Associate Director of the Navy-NASA Joint Institute of Aeronautics, and by Dr. L. W. Carr of the U. S. Army AFDD are gratefully appreciated and acknowledged. This research was performed in the Compressible Dynamic Stall Facility of the Fluid Mechanics Laboratory (FML), NASA Ames Research Center. The support of Dr. S. S. Davis, Chief, Fluid Dynamics Research Branch and that of the FML staff is greatly appreciated.

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Table 1a: Experimental Conditions for Global Flowfield Study

M	α^+					
	0	0.02	0.025	0.03	0.035	0.04
0.2	x		x	x	x	
0.25			x	x		
0.3	x	x	x	x	x	x
0.35	x	x	x	x		
0.4	x	x	x	x		
0.45	x	x	x	x	x ($\alpha^+ = 0.0313$)	

Table 1b: Experimental Conditions for Leading Edge Study

M	α^+					
	0	0.02	0.025	0.03	0.035	0.04
0.2				x		
0.25						
0.3	x			x	x	
0.35		x		x		
0.4				x		
0.45	x	x	x	x		

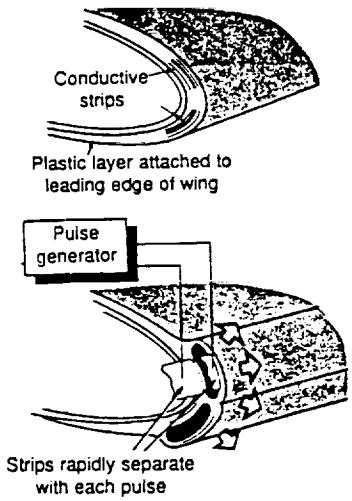

Figure 1. Schematic of the electro-expulsive de-icer system (from ref. 6).

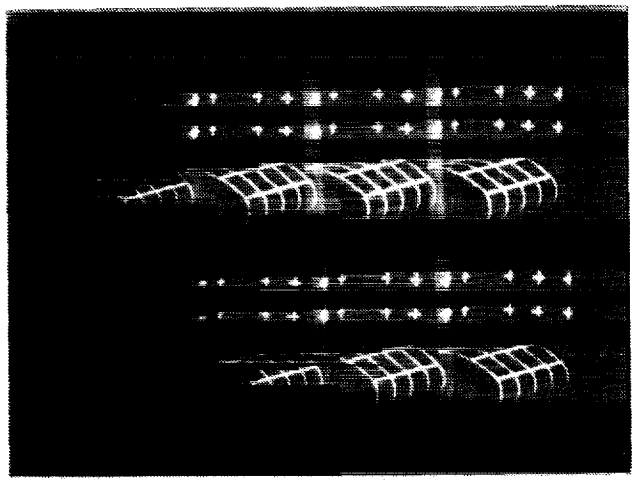

Figure 2. IMA-CON camera Polaroid image showing reflective reference grid and background reference marks (plus marks).
Time between images = 0.04ms, grid cells are 1 x 1 inch.

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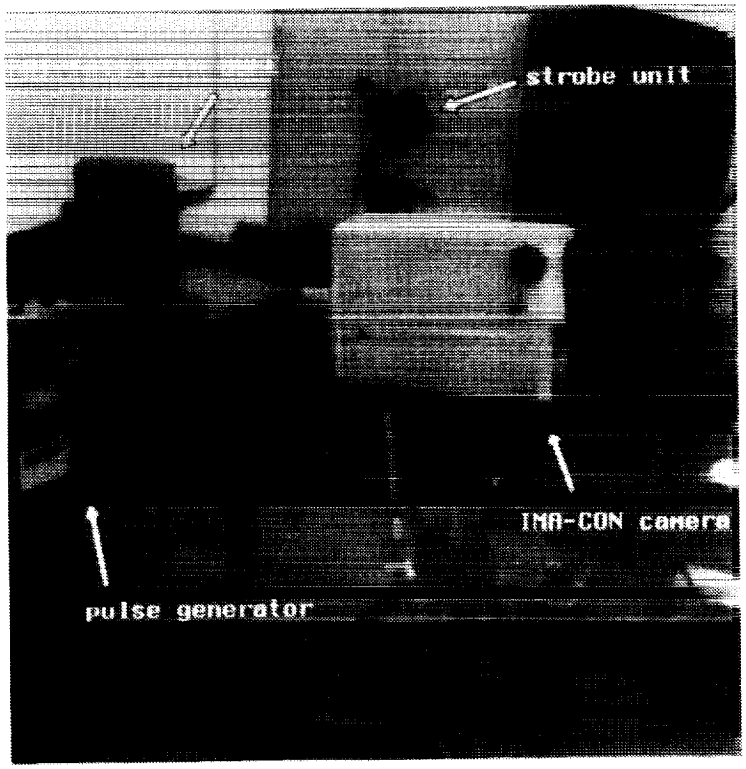

Figure 3a. Dynamically Deforming Leading Edge measurement setup.

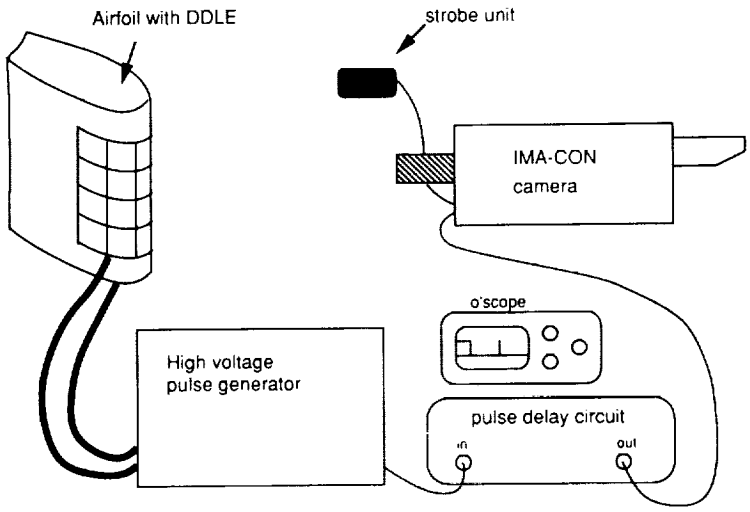

Figure 3b. Schematic of the Dynamically Deforming Leading Edge measurement setup.

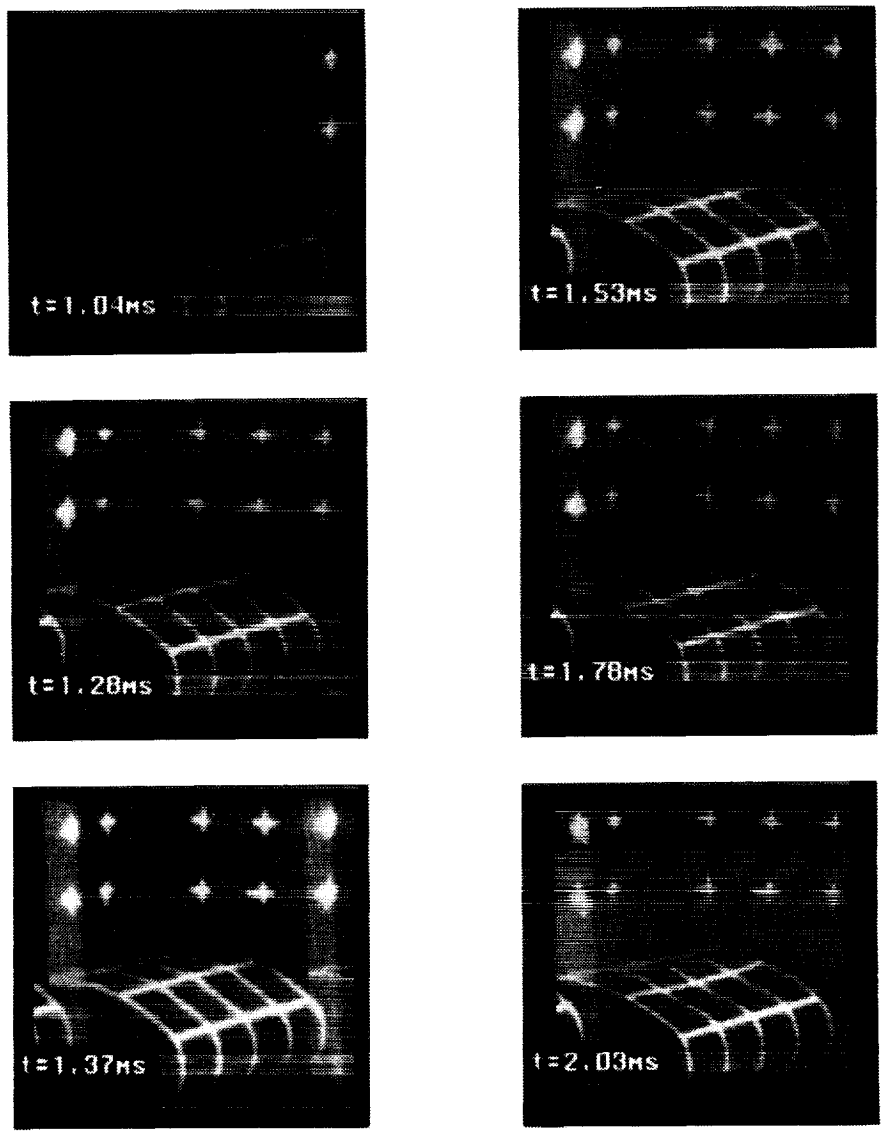

Figure 4. Sequence of digitized IMA-CON images.

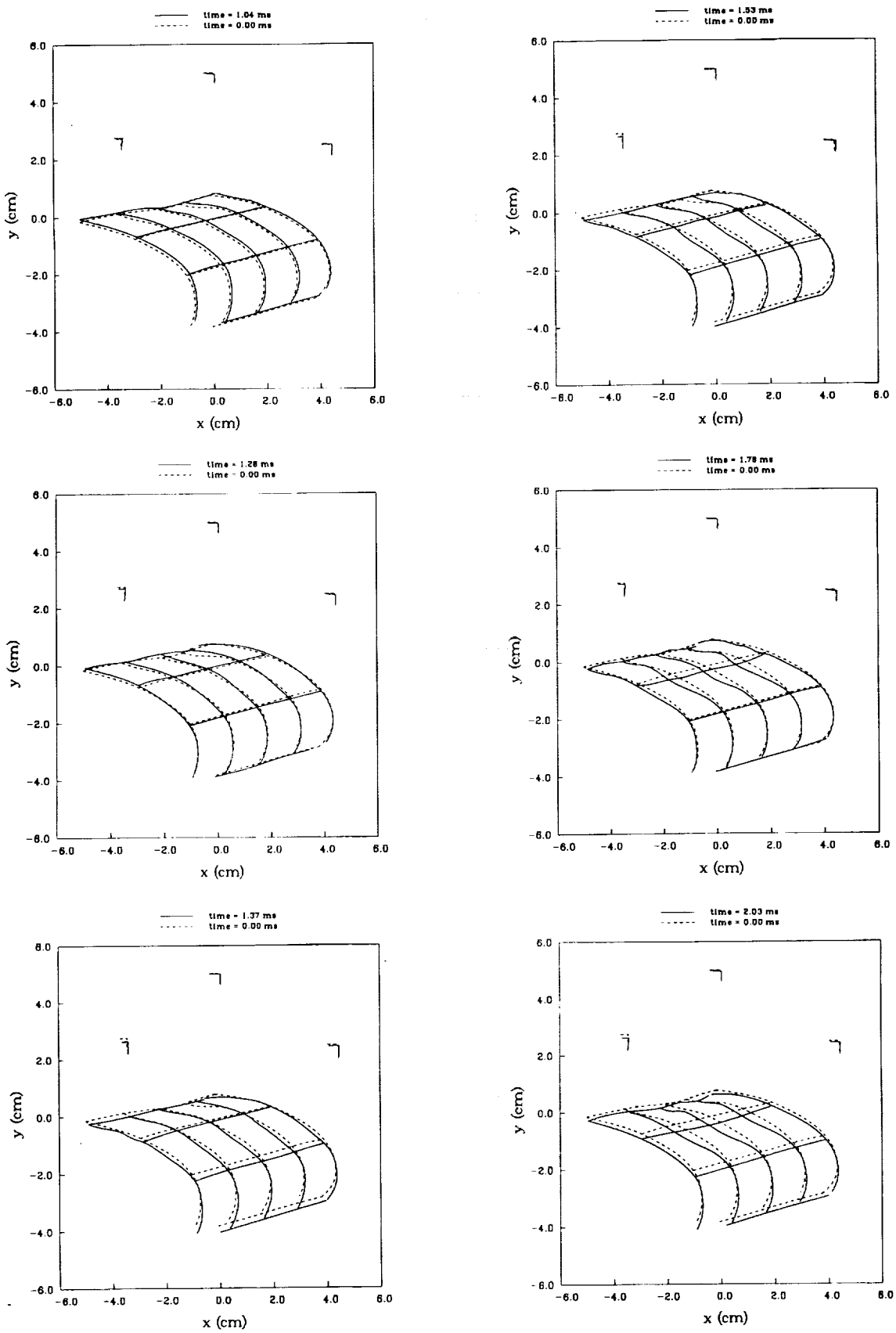

Figure 5. Line plots of data obtained from the digitized images of Fig. 4.

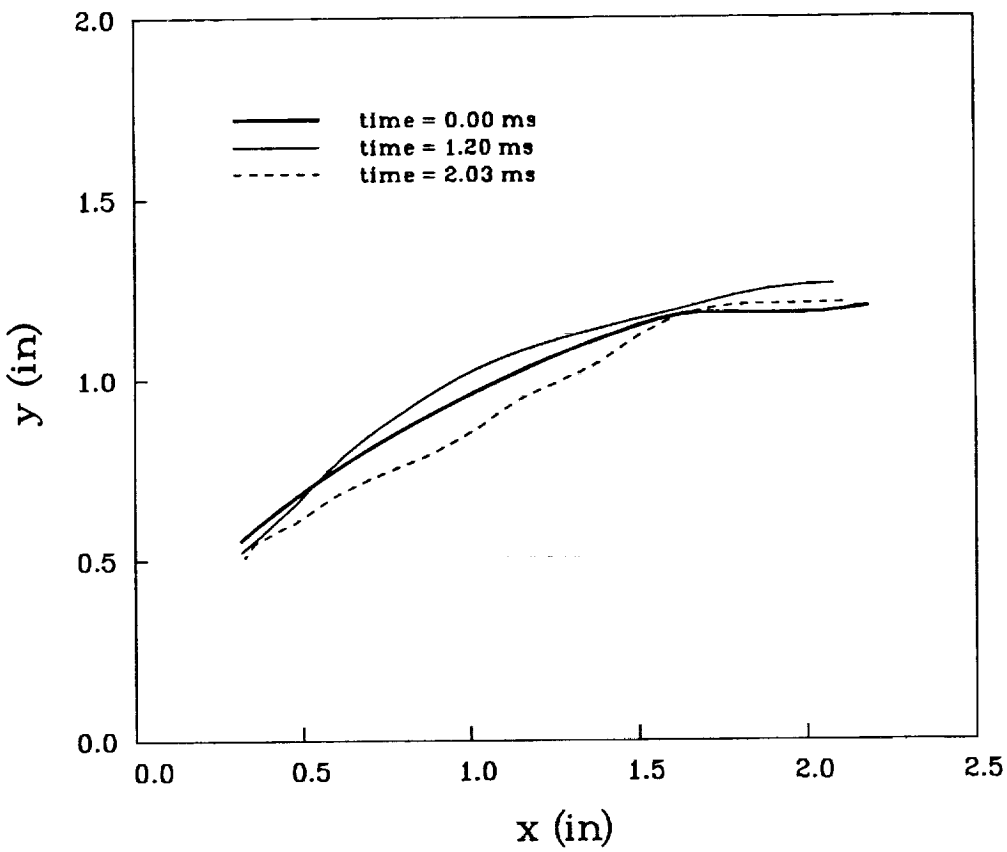

Figure 6. Profile view of the mid-span reference grid line showing maximum deflections from the neutral surface.

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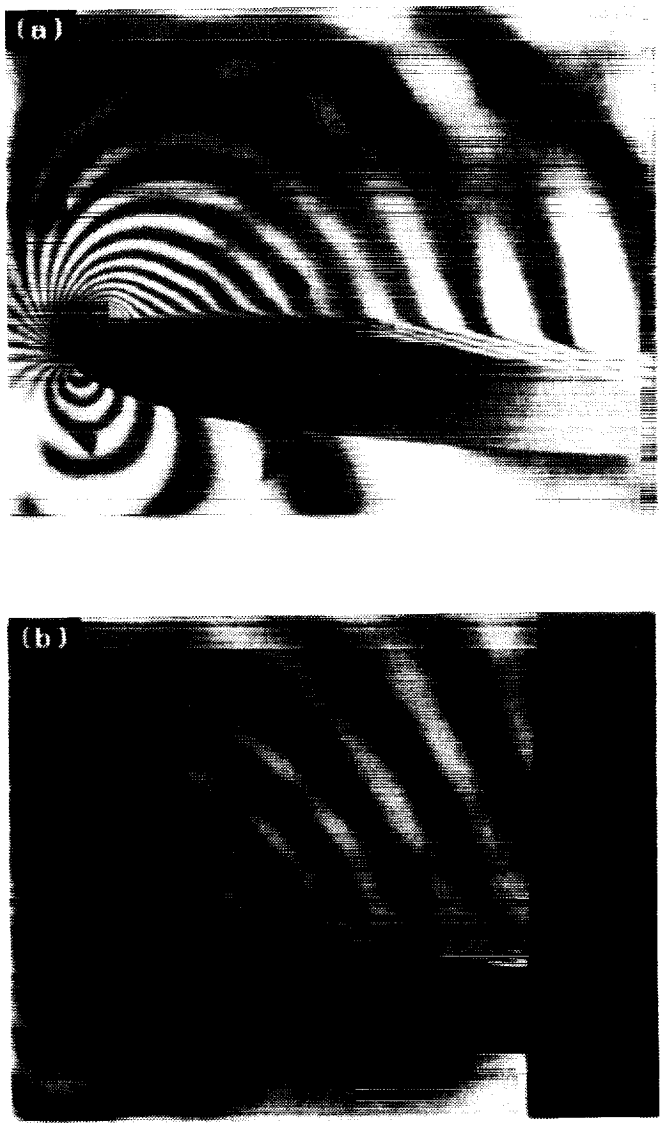

Figure 7. PDI fringe patterns: (a) using 6" diameter windows, (b) using 2" x 3" windows.

APPENDIX A

INTERFEROMETRIC INVESTIGATIONS OF COMPRESSIBLE DYNAMIC STALL OVER A TRANSIENTLY PITCHING AIRFOIL

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Summary

The dynamic stall flow field over a NACA 0012 airfoil pitching transiently from $0 - 60^0$ at a constant rate under compressible flow conditions has been studied using the real-time technique of point diffraction interferometry. This nonintrusive investigation not only provides a quantitative description of the overall flow field, but also of the finer details of dynamic stall vortex formation, its growth and the concomitant changes in the pressure distribution. Several hundred interferograms have been obtained for a range of experimental conditions. Analysis of these interferograms shows that the peak leading edge suction pressure coefficient at stall is nearly constant for a given free stream Mach number at all nondimensional pitch rates. This value is below that seen in steady flow at the static stall angle for the same Mach number, indicating that dynamic effects significantly affect

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the separation behavior. Further, for a given Mach number, the dynamic stall vortex seems to form rapidly at nearly the same angle of attack for all pitch rates studied; its growth and subsequent convection, however, are dependent upon the pitch rate. As the vortex is shed, it induces an anti-clockwise trailing edge vortex, which grows in a manner simliar to that of a starting vortex. The measured peak suction pressure coefficient drops as the free stream Mach number increases, but the value shows locally supersonic flow occurs for Mach numbers greater than 0.35; for free stream Mach numbers above 0.4, several small shocks appear above the leading edge shear layer over a distance covering 0 - 5% chord. The unsteady flow peak suction pressure coefficient at any angle has been experimentally demonstrated to lag that of the steady flow at the same angle of attack. The full paper will address the physical issues associated with the results described above.

Introduction

The utilization of dynamic stall as a method for increasing the maneuverability and agility of aircraft has received significant attention during the past few years. Compressibility effects have been shown to change the way that dynamic stall develops; thus better understanding of these compressiblity effects has been of importance and interest in the development of supermaneuverable and highly agile aircraft. It is well known(Reference 1) that the effects of compressibility set in at very low freestream Mach numbers(M = 0.2-0.3) on airfoils operating at high lift levels, due to the development of extremely strong suction peaks near the leading edge of these airfoils, which cause acceleration of the local flow to supersonic speed. The fact that dynamic lift still persists even when these compressibility effects appear(Reference 2) supports the argument that the benefits of dynamic stall can be exploited in flight systems. However, for these attempts to be successful, detailed study of the effects of compressibility on the developing unsteady flowfield is needed.

Most of the events of dynamic stall onset are concentrated in the leading edge region of an airfoil or wing executing unsteady pitchup motion. These include occurrence of strong suction pressures, rapid movement of the stagnation point, transition of the boundary layer, possible formation of a separation bubble, production of shocks(which can interact with the boundary layer and cause it to separate), generation of large amounts of coherent vorticity(which becomes the dynamic stall vortex), and initial movement of the

dynamic stall vortex over the airfoil. In contrast to dynamic stall onset, the later stages of dynamic stall development requires knowledge about the flowfield away from the surface of the airfoil. Global characteristics of the flowfield are needed in order to understand the interactions that occur as the vortex moves down the airfoil; as the vortex moves past the trailing edge, additional events such as generation of a trailing edge vortex, redistribution of the flow field over the airfoil, etc, occur which need to be documented if this dynamic flowfield is to be controlled and utilized.

Experiments focused on these issues are ongoing in the Compressible Dynamic Stall Facility(CDSF) at the Fluid Mechanics Laboratory(FML) at NASA Ames Research Center. These experiments are directed toward improved understanding of these complex fluid interactions, and toward establishing a benchmark data base for computational studies. These experiments involve quantitative documentation of the dynamic stall flow field of an airfoil executing a constant-pitch-rate maneuver from 0 to 60 degrees angle of attack for a range of pitch rates at freestream Mach numbers varying from 0.20 to 0.45. A real-time point diffraction interferometry technique has been used to obtain global information throughout the full pitching process, as well as details of the flow near the leading edge during the onset of dynamic stall. These interferograms are presently being analysed; some of the results of this analysis are described below. More detailed analysis of the results will be presented in the full paper.

Description of the Facility, Technique and Instrumentation

The *Compressible Dynamic Stall Facility (CDSF)* was established for conducting dynamic stall research in the FML as a part of the Navy-NASA Joint Institute of Aeronautics. This facility is specifically designed for study of dynamic stall over a range of Mach numbers, using non-intrusive optical flow diagnostic techniques. It is operated as a part of the in-draft tunnel complex at the FML (for details see Carr and Chandrasekhara[3]). The CDSF is unique in that the airfoil is supported between two 2.54 cm thick optical quality glass windows by pins that are smaller than the local airfoil thickness. Thus, the entire flow

field including the airfoil surfaces can be viewed unobstructed by any support mechanism. This enables the study of the flow at the surface near the leading edge, where the dynamic stall vortex forms, as well as the flow field away from the airfoil.

The modifications to the Compressible Dynamic Stall Facility to produce the rapid ramp-type pitching of an airfoil at constant pitch rates are shown in Figure 1 (earlier work in this facility was performed on an oscillating airfoil; for details, see Chandrasekhara and Carr[1]). The present system uses a a hydraulic drive to produce the unsteady motion; the specifications are as follows:

angle of attack, α:	0-60°
pitch rate, $\dot{\alpha}$:	0-3600 ^{o}/sec
maximum acceleration rate:	600,000 $^o/sec^2$
change in α during acceleration:	$\leq 6^o$ of pitch
minimum acceleration time:	4 ms
free stream Mach number:	0.1-0.5
airfoil chord:	7.62cm
Reynolds number:	$2x10^5$ - $1x10^6$

The pitch rate of 3600 ^{o}/sec on the 7.62 cm chord airfoil corresponds to a 90°/sec pitch rate of a 3m chord airplane wing at any given Mach number; thus, the rates obtainable from the design are directly applicable to flight conditions. In order to limit or isolate the effects of transients on separation, the change in angle of attack during acceleration and the acceleration time itself were limited to less than 6° and 4 ms, respectively. To properly simulate a maneuver, an angle of attack range of 0- 60° was selected. To provide for reasonable experiment times, the facility has a recycle time of 2 seconds (30 runs/minute). The system uses both the airfoil position and velocity information in its feed back loops to properly perform any required maneuver which can be selected through software. The complete details of the final design are presented in Chandrasekhara and Carr[4].

The airfoil position was read by a digital optical encoder, whose output was input to the digital I/O board of a microVAX II Work Station and timed with its internal clock. Figure 2 shows an example of the actual rates obtained, including the variation of the

angle of attack during the various parts of the pitch-up motion in an experiment. As can be seen, the airfoil goes through its static stall angle with a linear rate of change of angle of attack. For the highest rate, the motion is completed in 18 ms, beyond which the system is settling down (at the highest angle). All the tests were limited to the linear range.

The CDSF is equipped with a wide range of non-intrusive optical flow diagnostic instrumentation such as stroboscopic schlieren, laser Doppler velocimetry, holographic interferometry and point diffraction interferometry systems. The present paper will present results obtained using the point diffraction interferometry system.

Point Diffraction Interferometry Technique

The point diffraction interferometry technique used in this study utilizes the ability of a point discontinuity (in the form of a pin-hole) located at the image of a point source to diffract a portion of the incident light into a spherical reference wave front. In the present application, the primary optics of an existing schlieren system were used (see Reference 5 for details), with a pulsed Nd:YAG laser replacing the conventional spark as the light source, and a specially created point diffractor replacing the usual knife edge. The laser light was expanded through a microscope objective to fill the schlieren mirror, transmitted through the test section, and refocused by another schlieren mirror. The exposed photographic plate used to create the point-diffraction spot was placed at the focus of this second mirror, and the laser was pulsed with enough energy to burn a hole, or spot, in the emulsion located at the focal plane of the second mirror. The spot was created *in situ* by passing light through the test section at a no-flow condition. The spot was precisely tailored to the application under investigation, automatically correcting for nonuniformities in the light source or optics. The tunnel was turned on and the real-time interference fringes were recorded on Polaroid film(ASA 3000), and were available for immediate viewing. More detail about the point diffraction interferometry technique used in this study can be found in References 6 and 7.

Results

Several sequences of interferograms were obtained for a range of Mach numbers and pitch rates, ranging from 0.3 - 0.45 and $2000^o/sec$ - $3600^o/sec$, respectively. Figure 3 presents one such sequence created for M = 0.40, $\alpha^+ = 0.025$ (at a pitch rate of 2585 degrees/sec, where the non-dimensional pitch rate is defined as $\alpha^+ = \frac{\dot{\alpha}c}{U_\infty}$). The laser light source was strobed at the appropriate phase angle (including the correct delay due to the time it takes to lase) by using hardware specially built for this purpose. The actual instantaneous angle of attack at which the laser fired was read by freezing the encoder display using specially built light sensitive (photo-diode based) hardware. These images present instantaneous quantitative measurements of the actual density field associated with the development of dynamic stall on the airfoil and offer the first insight into the compressible field away from the airfoil surface for airfoils undergoing ramp motion.

Figure 3a shows that at $\alpha = 6.02^o$, the flow on the rapidly pitching airfoil is smoothly progressing through the region of high negative pressure; compare this to Figure 3b, at $\alpha = 9.1^o$, where a leading edge bubble has developed. Figure 3c shows that at the angle of attack of 12.5^o the dynamic stall process has begun, as can be seen near the leading edge. The flow over the airfoil is still attached as can be seen by the fringes in the boundary layer on the rear part of the airfoil; in fact, the outer flow passes smoothly around the vortex and blends into the boundary layer without any discontinuities. As the angle of attack is increased to 15.5^o degrees (Figure 3d), the presence of the dynamic stall vortex is clearly observable. It should be noted that up to this angle of attack, the ramp-motion dynamic stall process appears very similar to that which occurs on airfoils oscillating in pitch. However, the ramp-motion stall process becomes signficantly different as the angle continues to increase. Figure 3e, at $\alpha = 20.2^o$, shows the character of the flow when the dynamic stall vortex is at 75% chord; here there is a strong interaction between the dynamic stall vortex and a vortex which is forming at the trailing edge. This interaction is very obvious in Figure 3f, which shows the flowfield as the airfoil dynamically passes 50.4 degrees angle of attack. Here the trailing-edge vortex has moved away from the airfoil; note the very strong pressure gradients in the flow below the airfoil near the trailing edge. These gradients do not appear in the flow directly above the flow line emanating from the trailing edge of the airfoil, demonstrating that the airfoil is still supporting a significant

pressure differential even though the flowfield appears to be separated. Compare this to the corresponding image for steady flow, as shown in Figure 4, taken at 50.4 degrees angle of attack in steady flow. Here, a series of fringes appear on the airfoil, showing that the airfoil is no longer creating lift of significant magnitude.

The interferograms obtained at the various experimental conditions have been analysed, and pressure coefficients have been obtained. Figure 5 shows the development of the maximum suction that appears on the airfoil, determined at the instant that dynamic stall begins. This figure shows that the maximum Cp decreases dramatically as the freestream Mach number is increased. It also shows that the maximum Cp is relatively independent of pitch rate; this characteristic of the flow will be explored in detail in the full paper. Figure 6 shows the variation of Cp with angle of attack for a range of reduced frequencies. It is clear from this figure that pitch rate has a significant effect on the maximum Cp that can be reached for a given angle of attack, although the maximum value of Cp obtained is essentially independent of pitch rate (as was seen in Fig. 5). The impact of local compressibility on this maximum is under investigation at the present time, and will be explored more fully in the full paper.

5. CONCLUSIONS

Interferograms taken during the pitch-up of an airfoil in a moderately compressible flow have offered new insight into the character of the dynamic stall vortex occuring under compressible flow conditions. Preliminary analysis suggests that this vortex is significantly different from that seen in incompressible dynamic stall for airfoils undergoing ramp motion. In fact, the later stages of development of the dynamic stall process is clearly affected by the ramp-motion process when compared to oscillating airfoil behavior even in compressible flow. The final paper will contain a detailed description of the developing flow field, and present quantitative data concerning the instantaneous density field, with comparisons to oscillating airfoil dynamic stall.

6. REFERENCES

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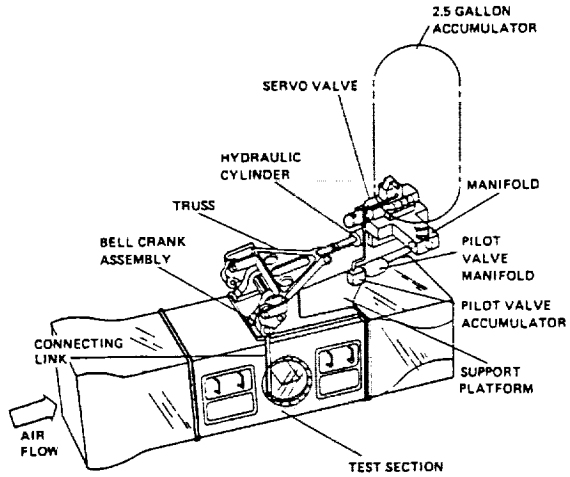

Fig. 1. Diagram of the In-draft Wind Tunnel with the Constant Pitch Rate Mechanism

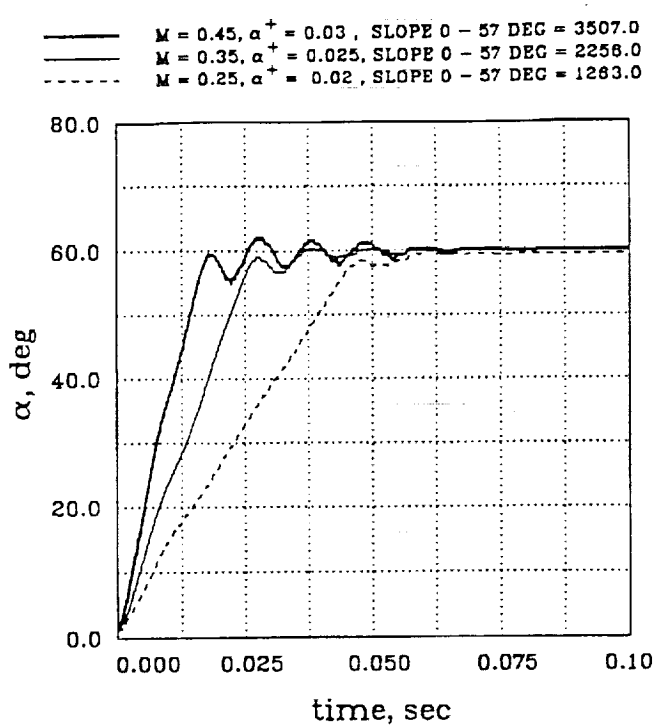

Fig. 2. Time-history of Ramp for Sample Test Conditions

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(a)

(b)

Fig. 3. Interferograms of dynamic stall on rapidly pitching airfoil as shown by Point Diffraction Interferometry, M = 0.40, $\alpha^+ = 0.025$, (a)$\alpha = 6.02^0$; (b) $\alpha = 9.1^0$.

(c)

(d)

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Fig. 3. (continued) Interferograms of dynamic stall on rapidly pitching airfoil as shown by Point Diffraction Interferometry, M = 0.40, $\alpha^+ = 0.025$, (c) $\alpha = 12.5^0$; (d) $\alpha = 15.5^0$.

(e)

(f)

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Fig. 3. (continued) Interferograms of dynamic stall on rapidly pitching airfoil as shown by Point Diffraction Interferometry, M = 0.40, $\alpha^+ = 0.025$, (e) $\alpha = 20.2^0$; (f) $\alpha = 50.4^0$.

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Fig. 4. Interferogram for M=0.40, $\alpha = 50.4^o$, steady angle of attack.

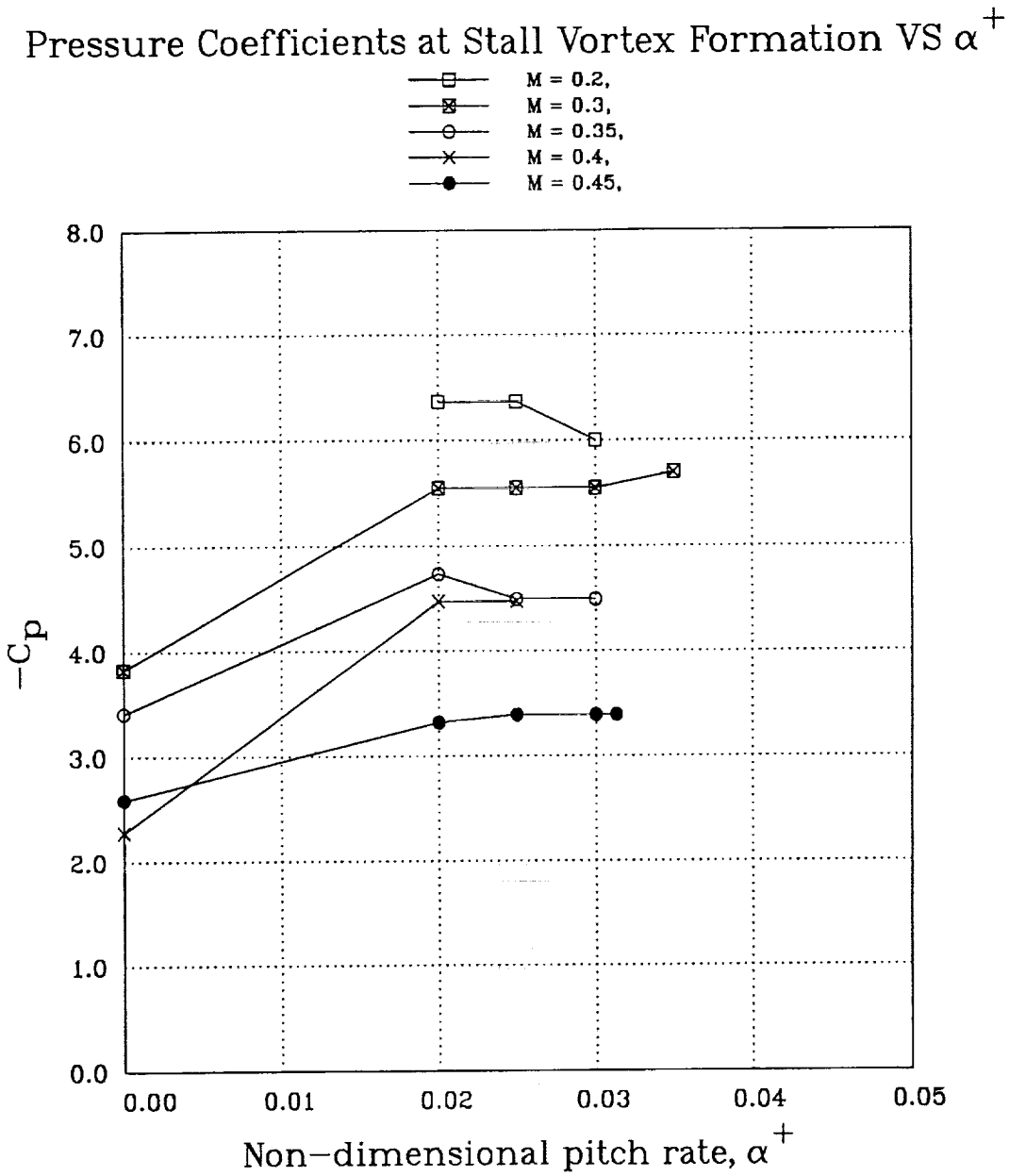

Fig. 5. Variation of Maximum Negative Pressure Coefficient with Nondimensional Pitch Rate at Several Mach Numbers.

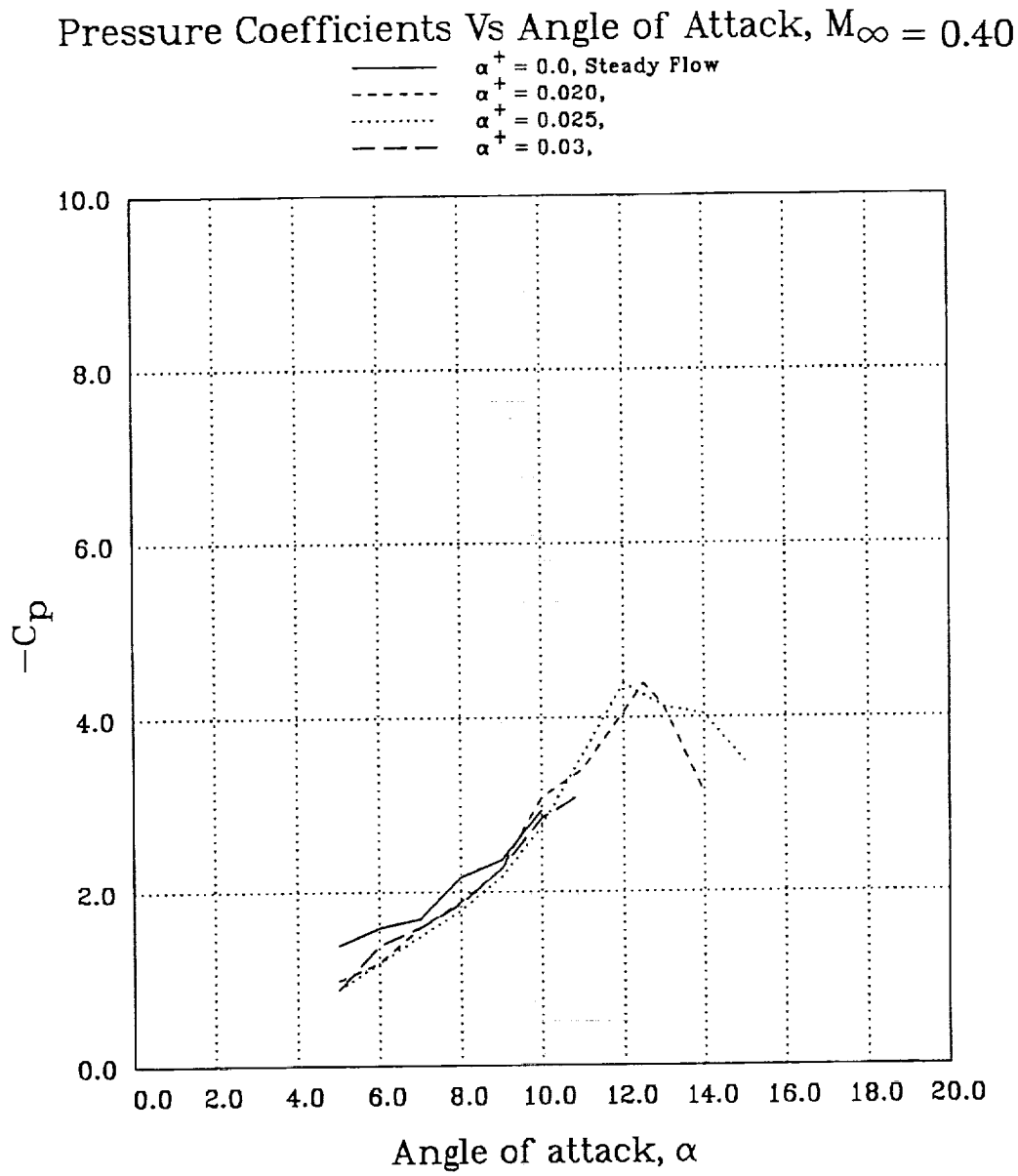

Fig. 6. Variation of Maximum Negative Pressure Coefficient with Angle of Attack for Several Nondimensional Pitch Rates.