Unsteady Flow Separation Control over a NACA 0015 using NS-DBD Plasma Actuators

THESIS

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Abstract

Flow field surrounding a moving body is often unsteady. This motion can be linear or rotary, but the latter will be the primary focus of this thesis. Unsteady flows are found in numerous applications, including sharp maneuvers of fixed wing aircraft, biomimetics, wind turbines, and most notably, rotorcraft. Unsteady flows cause unsteady loads on the immersed bodies. This can lead to aerodynamic flutter and mechanical failure in the body. Flow control is hypothesized to reduce the load hysteresis, and is achieved in the present work via nanosecond pulse driven dielectric barrier discharge (NS-DBD) plasma actuators.

To better understand the physics of unsteady flow over an airfoil a new facility was constructed, and new processing codes were developed and implemented. A NACA 0015 airfoil was mounted to oscillating mechanism, and the angle of attack was varied sinusoidally. The Reynolds number was varied from $0.17 \cdot 10^6 - 0.50 \cdot 10^6$, and the reduced frequency of oscillation was varied from 0.025 – 0.075 to gain a better understanding of these parameters on the unsteady flow dynamics. The plasma actuator was mounted at $x/L = 0.01$, just downstream of the airfoil leading edge. It was noted that the construction of the actuator influenced baseline behavior.

Validation of the facility was achieved via qualitative comparisons of the baseline results to the results in a similar experimental setup in literature. After validation,

experimentation in the form of surface pressure and particle image velocimetry measurements were performed.

In the excited experiments, three major conclusions were drawn. The first was that excitation results in the formation of flow structures during the stalled regime of dynamic stall similar to static stall results. Low excitation Strouhal numbers result in the formation of large structures that cause significant unsteadiness in the stalled regime of dynamic stall. As the excitation Strouhal number increases, this unsteadiness increasingly reduces and eventually disappears due to the generation of increasingly smaller vortices.

Secondly, excitation also results in earlier reattachment due to the formation of the structures that momentarily reattach the flow. Once the angle of attack of the airfoil decreases past the static stall angle, the flow remains attached.

Lastly, it was noted that all excitation resulted in the reduction of the lift and moment hysteresis and is reflected in the increased damping coefficient. This is partially due the reattachment of the flow in the stalled regime. The other major factor is the decreased dynamic stall vortex strength. This subsequently results in the decreased magnitudes of the lift and moment peaks. Excitation leads to the formation of leading edge structures that remove vorticity build up at the leading edge. As the excitation Strouhal number increases and the size of structures generated decreases, the vorticity build up becomes insignificant, preventing the formation of the dynamic stall vortex.

Immediate future work should be concentrated on establishing better motion repeatability, while long terms goals should include determining the effects of frequency modulated excitation, and understanding three dimensionality of dynamic stall.

Dedication

Dedicated to my family, friends, and teachers – their guidance and patience encouraged me to pursue higher education.

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Publications

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C. Clifford, **A. Singhal**, and M. Samimy, "Flow Control over an Airfoil in Fully Reversed Condition Using Plasma Actuators", AIAA Journal, Vol. 54, No. 1 (2016), pp. 141-149.

Thesis Publications

A. Singhal. "Flow Control over a Rotorcraft Blade Modeled by a Boeing VR-7 Airfoil", Undergraduate Honors Thesis, Department of Mechanical and Aerospace Engineering, The Ohio State University, Columbus, 2015.

Conference Publications

A. Esfahani, **A. Singhal**, C. Clifford, and M. Samimy, "Flow Separation Control over a Boeing Vertol VR-7 using NS-DBD Plasma Actuators." 54th AIAA Aerospace Sciences Meeting, AIAA Paper 2016-0843, 2016.

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Fields of Study

Major Field: Mechanical Engineering

Studies in: Aerodynamics, Experimental Techniques, Flow Control, Automation and Control, Robotics

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Nomenclature

Latin Symbols

c	airfoil chord, 203	mm
C_D	drag coefficient	—
C_L	lift coefficient	—
C_M	moment coefficient	—
C_p	pressure coefficient	—
ci	confidence interval half width	—
f	frequency	Hz
k	reduced frequency, $\pi f L / u_\infty$	—
L	airfoil chord, 203	mm
p	pressure	Pa
q	dynamic pressure, $1.05 \times (p_o - p_\infty)$	Pa
Re	Reynolds number, $u_\infty L / \nu$	—
St	Strouhal number, $f L / u_\infty$	—
u	streamwise velocity	m/s
v	vertical velocity	m/s
U	velocity vector, (u, v)	m/s
U^*	non-dimensional velocity vector, U / u_∞	—
x	streamwise position	mm
y	vertical position	mm

Greek Symbols

α	angle of attack	°
ϕ	phase	°
λ	swirling strength	Hz
λ^*	non-dimensional swirling strength, $\lambda L/u_\infty$	—
ν	kinematic viscosity	m^2/s
ω	vorticity, $\nabla \times U$	Hz
ω^*	non-dimensional vorticity, $\nabla^* \times U^*$	—

Subscripts

amb	ambient	—
e	excitation	—
o	stagnation	—
∞	freestream	—

Acronyms

AC	alternating current	—
DBD	dielectric barrier discharge	—
NS	nanosecond	—
PIV	particle image velocimetry	—

TKE

turbulent kinetic energy

—

1. Introduction

Rotorcraft represent a unique niche in the aircraft community. While not as fast as their fixed wing counterparts, their ability to hover provides a flexible and desirable platform for conducting operations. These operations, among others, include, fire and rescue, medical transport, and agriculture. This increased capability is not without limitations. On the advancing rotor blades, the tip velocity is constrained to prevent tip losses that occur from compressibility effects, limiting flight speeds [1].

On the retreating side, the blades are positioned at high angle of attacks to maintain lift symmetry, potentially resulting in retreating blade stall [2]. Furthermore, to maintain lift symmetry, the rotorcraft is not able to make use of the potential lift gained by increasing the angle of attack on the advancing side. A graphic illustration is presented in Figure 1.1.

The primary focus of this work is a phenomenon caused by the need to maintain lift symmetry - dynamic stall. Dynamic stall occurs when the rate of change of angle of attack is fast enough to maintain attached flow past the static stall angle. As the airfoil motion slows and begins to pitch down, a vortex is shed from the leading edge. Termed the dynamic stall vortex, this structure momentarily increases lift. Once it has convected past the airfoil, the flow is fully stalled. This results in large oscillatory loads that

significantly reduce the life of the blade [3]. As such it is desirable to eliminate the oscillatory loads, and this problem has attracted much attention in the rotorcraft community.

Figure 1.1: Potential lift versus usable lift due to lift symmetry in conventional rotorcraft [4]. The advancing side of the blade is on the left and has significant potential for lift. The lift produced by the advancing side is limited by the retreating side blade (right).

Previous research has focused on the use of flow control devices to eliminate the dynamic stall vortex. These devices consist of both passive (such as geometric modifications to the airfoil) and active (such as momentum injection) techniques [5, 6, 7, 8, 9]. For practical application, it is pertinent to ensure such a system can be incorporated into the blade of a rotorcraft. Geometric modifications such as vortex generators would result in heavier rotor blades, and may not easily be incorporated into an existing rotorcraft fleet. Furthermore, geometric modifications tend to work well at design conditions, but become ineffective or detrimental at off-design conditions.

Active techniques aim to provide efficacy while maintaining a broad operating range. Active techniques typically involve momentum injection [5]. Momentum injection is traditionally achieved via blowing using jets. This requires the construction of cavities and use of pumps to achieve excitation [10]. While potentially effective, it is not a desirable solution due to the increased mechanical complexity, weight, and cost. As such, dielectric barrier discharge (DBD) plasma actuators have gained more attention. These devices consist of two copper electrodes separated by a dielectric barrier. A high voltage AC signal is applied to the exposed electrode, which generates plasma. This actuator is known as an alternating current dielectric barrier discharge (AC-DBD) plasma actuator and uses the electrohydrodynamic effect to generate a flow. The use of an AC-DBD plasma actuator adds momentum to the flow (similar to jets), but maintains a minimally complex construction.

The primary disadvantage of momentum injection techniques is at high speeds. At high speeds, more momentum is required to maintain efficacy which can be difficult to provide. The focus of this paper will be the use of nanosecond pulse driven dielectric barrier discharge (NS-DBD) plasma actuators. These actuators are constructed identically to AC-DBD plasma actuators. Instead of an AC signal, high voltage and short duration ($\sim$ ns) DC pulses are used to create localized, and rapid heating that results in the formation of a compression wave at the DC pulse frequency as shown in Figure 1.2 [11]. At certain frequencies, the compression wave can be used to excite natural flow instabilities. This is the flow control mechanism for the NS-DBD plasma actuators and

since it is based on excitation on instabilities they require relatively low power input and the actuation is effective at all speeds.

NS-DBD plasma actuators have been shown to reattach flow in stalled airfoils in a variety of airfoils and flow velocities in a static configuration [12, 13, 14, 15, 11]. This work aims to extend the existing body of work on NS-DBD flow control to unsteady flows.

Figure 1.2: Image of compression wave generated by NS-DBD plasma actuators using phase-locked schlieren imaging [11]. The image is taken normal to the span of the airfoil.

2. Background

In rotorcraft, the angle of attack of the rotor blade is varied throughout the rotation of the blade to preserve lift symmetry to the extent it is possible. This gives rise to a phenomenon known as dynamic stall. Dynamic stall results in large unsteady loads, which can cause the failure of the rotor blades. With the increasing demands of fast flight and greater payloads, the loads on rotor blades are only expected to increase. Thus the ability to minimize these unsteady loads is critical, if these vehicles are to keep up with the rising demands.

A. Dynamic Stall

The relative airspeed of the retreating blade in rotorcraft is significantly lower than the airspeed approaching the advancing blade and this problem will become more pronounced as the forward speed is anticipated to increase. The velocity asymmetry is shown in Figure 2.1. Regions of green indicate forward flow, whereas regions of red indicate reverse flow.

Figure 2.1: Local velocity distribution along rotor blade. Red indicates regions of reverse flow on the retreating blade, whereas green indicates forward flow.

To compensate for the ensuing loss of lift due to a reduction in relative wind speed, the angle of attack of the retreating blade must increase so that the lift generated throughout the rotor disk is equally distributed. The quick change in angle of attack (due to the high rotational velocity of the rotor) gives rise to a phenomenon termed dynamic stall. In static stall, the flow will immediately separate after the angle of attack has exceeded the static stall angle of attack. However, in dynamic stall the flow will remain attached beyond the static stall angle of attack. This is a result of airfoil pitching motion, and the formation of a closed separation bubble near the leading edge of the airfoil [1]. In addition to the lift increase caused by the motion of the airfoil, vorticity near the leading edge accumulates during the pitch up motion and results in the formation and shedding of the dynamic stall vortex. This further increases the generated lift. After the dynamic stall vortex convects over the airfoil, the airfoil becomes fully stalled. During the pitch down motion, the flow eventually reattaches.

The process of dynamic stall can be categorized into five stages and is shown in Figure 2.2. The first stage begins at the lowest angle of attack. As the airfoil pitches up the flow remains attached and the lift increases steadily with angle of attack. When the angle of attack exceeds the static stall angle of attack, the airfoil is in the second stage of dynamic stall [5]. In this stage, flow separation is delayed due to the airfoil motion and the formation of a closed separation bubble near the leading edge of the airfoil. In this stage, vorticity accumulates at the leading edge.

The ejection of this vorticity in the form of the dynamic stall vortex is the third stage of dynamic stall. The strength of the vortex is dependent on many factors, including airfoil shape. It has been shown though, that the pitch rate has a strong influence on the size of the vortex – higher reduced frequencies result in a more energetic vortex [16]. There are several mechanisms that may lead to the ejection of the dynamic stall vortex. These include the breakdown of the separation bubble due to turbulent transition [17], and vortex induced separation [18]. As the vortex gets ejected, it results in increased aerodynamic loading (magnitudes of lift, drag, moment increase significantly). When the vortex has convected past the airfoil, the flow is fully stalled. This is the fourth stage of the dynamic stall process and leads to a sharp decrease in the lift force. The final stage of dynamic stall is flow reattachment that begins at the leading edge and traverses downstream. The reattached flow recovers the negative moment coefficient as seen in Figure 2.2.

Figure 2.2: Illustration of dynamic stall phenomenon taken from Corke, 2015 [5]. As the angle of attack increases, the flow remains attached (states 1 and 2). Then the dynamic stall vortex is shed and convects over the airfoil (state 3). After the vortex convects, the airfoil becomes fully stalled (state 4). During the pitch down motion, the flow will reattach (state 5).

Dynamic stall is influenced by a number of parameters, including the maximum angle of attack. The maximum angle of attack attained in the motion of the airfoil results in one of the four regimes of dynamic stall. The first regime occurs when the maximum angle of attack is less than the static stall angle of attack. In this regime the flow remains

attached and the aerodynamic loads are similar to those in the static case. The second regime occurs when the maximum angle of attack equals the static stall angle of attack. Again, the flow remains attached, and the loads are similar to the static case. This regime produces the most lift without large drag and moment penalties [5]. The remaining two regimes occur when the maximum angle of attack is greater than the static stall angle of attack and are termed light stall and deep stall. Light stall results in small aerodynamic hysteresis and is defined by the thickness of the separated region, as shown in Figure 2.3 (adapted from Coleman et al. [19]). In light stall, the thickness of the separated region is on the order of the airfoil thickness. In deep stall, the thickness of the separated region is on the order of the airfoil chord.

Figure 2.3: Light stall flow (left) versus deep stall flow (right) [19].

B. Aerodynamic Flutter

This hysteresis in the aerodynamic loads can lead to aerodynamic flutter. Aerodynamic flutter causes vibrations in the rotor blade which may lead to catastrophic failure. Aerodynamic flutter can be quantified by damping coefficient, Ξ . It is defined as the closed loop integral of the moment coefficient normalized by the multiplication of π and

the amplitude of motion squared. The aerodynamic damping coefficient was derived by Carta et al. using a single degree of freedom system with a uniform freestream [20]. A negative damping ratio indicates that airfoil is extracting energy from the flow and thus has the potential to undergo aerodynamic flutter [5]. Negative damping occurs when the moment coefficient on the upstroke is greater than the moment coefficient on the downstroke. This is best illustrated by a figure adapted from Corke et al. [5], shown in Figure 2.4.

Figure 2.4: Moment coefficient curve adapted from Corke et al. [5]. This illustration indicates the contribution of the moment coefficient to the damping coefficient. Negative damping occurs when the moment coefficient on the upstroke is greater than the moment coefficient on the downstroke.

Active flow control techniques have the potential to provide viable solutions to the transient forces, high drag and potential for flutter associated with dynamic stall.

Plasma-based actuators, due to their wide bandwidth, high amplitude [21, 22, 23] and absence of moving parts, are promising candidates for implementation of active flow on rotorcraft blades.

C. Flow Control

Pitching airfoils result in unsteady aerodynamic loads and potentially vibratory behavior. When the airfoil is in stall during its motion, undesirable effects such as increased drag are also observed. Due to the prominence of unsteady flow in various applications (i.e. wind turbines, and helicopters), unsteady flow may benefit greatly from flow control.

Flow control can be through passive or active means. Passive flow control techniques rely on modifications to the shape of the airfoil. These include vortex generators, leading edge gloves, and surface roughness elements (boundary layer trips). Although passive flow control techniques work well at their design conditions, their design envelope is often limited. Outside of their design envelope, passive flow control techniques often are insufficient or detrimental. One example of passive flow control is a leading edge glove with streamwise vortex generators used by Martin et al. [24]. The researchers were able to demonstrate significant reduction in the moment stall of the airfoil with these passive devices up to a Mach number, $M = 0.4$. At higher Mach numbers, these devices were ineffective.

Active flow control tends to involve momentum addition (such as blowing or pulsed jets), or instability excitation. Momentum addition is a promising technique that shows efficacy in suppressing the dynamic stall vortex, and reducing flow hysteresis [25, 26, 27]. However, there are two main drawbacks of these techniques. First, momentum

addition is limited at high speeds, or requires extensive power to maintain efficacy. Secondly, many momentum injection techniques increase the mechanical complexity of the blade and therefore are undesirable. To address the mechanical complexity aspect, plasma actuators have been given more attention in recent years. Dielectric barrier discharge actuators driven by AC waveforms have been shown to have control authority at relatively low Reynolds numbers and at static angle of attacks as well as in dynamic stall applications [27]. Successful implementation of these plasma actuators in high speed flows requires the use of thicker dielectrics and higher voltages [28, 29, 30, 31] to increase the supplied body force [32]. Even when these changes are implemented there is a limit to the effectiveness of DBD actuators driven by an AC waveform [30].

As such, a new driving waveform has been applied to these actuators. Consisting of nanosecond high voltage DC pulses, these actuators have been shown to possess control authority at higher speeds. The results of several studies [33, 34, 35, 36] indicate that these actuators function by a thermal mechanism. Nanosecond pulse driven dielectric barrier discharge plasma actuators affect the flow via localized Joule heating. This rapid, localized Joule heating, leads to the formation of compression waves that excite naturally occurring flow instabilities.

In static testing, NS-DBD plasma actuators have been shown to be effective with lower energy consumption than their AC-DBD counterparts. Research conducted on the well-studied NACA 0015 airfoil indicates the efficacy of the actuators at high speed flows. Rethmel et al. [12] used NS-DBD plasma actuators mounted at $x/c = 0.01$. Shown in Figure 2.5 [12] are vorticity plots at an angle of attack $\alpha = 18^\circ$ (post stall), and

$Re = 1.15 \times 10^6$ for baseline (top) and excited (bottom) flow. As illustrated in the figures, excitation results in the reattachment of the shear layer. This increases the lift produced by the airfoil. In terms of rotorcraft, this allows the angle of attack of the retreating blade to exceed static stall angles, increasing the overall lift produced by the rotor. Clifford et al. [14] used NS-DBD plasma actuators at the aerodynamic leading edge (or geometric trailing edge) of a NACA 0015 undergoing reverse flow, mimicking flight conditions near the hub of the rotor. Although the region of reverse flow is small in helicopters due the high angular velocity of the rotor, this research demonstrated the flexibility of the actuators. This work was directed at a new class of vehicles known as slowed-rotor compound vehicles, where the region of reverse flow can span the entire retreating blade. Shown in Figure 2.6 [13] is the phase averaged swirling strength for the baseline case (top) and several excited cases. As indicated in the figure, excitation generates stronger and more coherent vortices. As the excitation Strouhal number increases, the structures become smaller, move further upstream on the airfoil, and quickly breakdown and dissipate. Similar results were obtained for a Boeing Vertol VR-7 airfoil in forward flow tested by Esfahani et al. [15]. As such, these actuators are applicable in rotorcraft relevant geometries as well. Esfahani et al. [15] also showed the presence of vortex merging in excited flow as shown in Figure 2.7, which was also observed by Clifford [13].

It is hypothesized that the plasma actuators can function by a similar mechanism in the stalled regime of dynamic stall. This would result in decreased aerodynamic load

hysteresis and an increased damping coefficient (and thus decreased chances of aerodynamic flutter).

Figure 2.5: Vorticity (ω) plots taken from Rethmel et al. [12] for a NACA 0015 airfoil at $\alpha = 18^\circ$ and $Re = 1.15 \times 10^6$. Top figure indicates baseline (no excitation) flow and illustrates fully separated flow. Bottom figure indicates excited flow at $St_e = 2.75$, and flow separation has been shifted downstream.

Figure 2.6: Phase-averaged swirling strength (λ_{ci}^*) taken from Clifford et al. [14] for a NACA 0015 airfoil undergoing reverse flow at $\alpha = 15^\circ$ and $Re = 500,000$. Cases (a)-(f) indicate excitation Strouhal numbers of $St_e = 0.03, 0.08, 0.16, 0.19, 0.88$, and 1.12 , respectively. Unmarked (top) case is the baseline case. Figures indicate excitation organizes wake, which results in reattachment.

Figure 2.7: Phase-averaged swirling strength (λ_{ci}^*) taken from Esfahani et al. [15] for a VR-7 airfoil at $\alpha = 19^\circ$, $Re = 500,000$, and $St_e = 0.61$. Phases are ordered from top to bottom. Phases $\phi = 120^\circ - 180^\circ$ indicate vortex merging.

3. Experimental Facilities and Techniques

Experiments were performed in the recirculating wind tunnel located at the Gas Dynamics and Turbulence Laboratory, within the Aerospace Research Center at The Ohio State University [12, 37]. The tunnel has an optically clear acrylic test section measuring 61×61 cm in cross-section and 122 cm in length. A NACA 0015 airfoil was used in the experimentation. Although not a typical cross section for rotorcraft, it has been well characterized in static flow. The tunnel is capable of producing a continuous range of flow velocities from 3 – 95 m/s. The corresponding Reynolds numbers based on chord length ($Re = u_{\infty}L/\nu$) are $0.04 \cdot 10^6 - 1.20 \cdot 10^6$. The freestream turbulence intensity is on the order of 0.25% of the freestream velocity for the cases presented [38].

Freestream static pressure (p_{∞}) and stagnation pressure (p_o) are measured using piezometer rings consisting of four pressure taps. Located at either end of the converging section of the wind tunnel, these pressure taps are connected to Omega Engineering pressure transducers (models PXS655-25DI and PX655-5DI). Freestream temperature is measured using a thermocouple located downstream of the test section. This thermocouple is also used to obtain the ambient temperature prior to the startup of the tunnel. Ambient pressure data is recorded from the Meteorological Aerodrome Report

(METARs) data reported by OSU Airport (KOSU). Free stream velocity can then be computed using Bernoulli's Equation (Equation 3-2):

$$u_{\infty} = \sqrt{2k_c \frac{p_o - p_{\infty}}{\rho}} \quad \text{Equation 3-1}$$

where $k_c = 1.05$, is the corrective factor and was empirically determined using a hot-wire anemometer [8].

Two coordinate systems are used throughout this paper. The first coordinate systems has its origin at the aerodynamic leading edge (ALE), as shown in Figure 3.1. The first system is a straight line along the chord line normalized by the chord length, denoted x/c , where positive coordinates indicate the aerodynamic suction side and negative coordinates indicate the aerodynamic pressure side. This system is in the airfoil reference frame and is used to indicate on-board instrumentation, actuators, and the flow separation line. The second system is a two-dimensional grid aligned with the test section and normalized by the chord, denoted x/L and y/L . This system is in the test section reference frame and is used to indicate velocity data. The origin of this test section is at the point of rotation, or $x/c = 0.25$. A moment, M , is defined to be positive when it results in the pitch down motion of the nose, as shown in the figure.

Figure 3.1: Schematic of experimental arrangement showing the coordinate origin, and actuator location.

A. Oscillating Mechanism

As a precursor to experimentation, the static facility had to be modified to be able to dynamically oscillate the airfoil. The mechanism was designed to maximize optical access and modeled after that of Greenblatt et al. [39]. As such, it consists of two aluminum disks securing acrylic disks. The ends of the airfoil are mounted to these disks. Thin ball-bearings used to mount the airfoil support rings to the tunnel. The aluminum rings are driven by a servo via timing belts, as shown in Figure 3.2. The large belts (seen in the left image) connects the aluminum rings to a pulley on a central shaft. The central shaft (a close up is shown on the right image) is connected to the servo via another belt. The overall gear ratio between the angle of attack of the airfoil and the servo is 16:63.

Figure 3.2: Photographs of oscillating mechanism. All aluminum parts would later be sandblasted and anodized black (Type II, 0.001" thick, black) to minimize reflections during optical measurements.

The servo itself is driven by a voltage follower program that uses a proportional-integral-derivative controller programmed in MINT, the native programming language of the servo controller. The PID parameters were manually tuned, and due to the lack of an accurate time step, the derivative constant is zero. The MINT program is fed two external voltage signals, a voltage signal for the position of the airfoil, and a voltage signal for the angular velocity of the airfoil. A machine-learning algorithm is used to monitor and adjust the motion of the airfoil. If the output is within a specified tolerance, the external voltage signals are accepted, otherwise they are modified using a PID controller on the amplitude, DC offset, and frequency of the airfoil motion. A total of 10 cycles are read, and the amplitude, DC offset, and frequency are computed using LabVIEW 2012 (using the sub-VIs “Extract Single Tone Information” and “AC & DC Estimator”). From the

current state, new parameters are computed for the subsequent cycle. The PID parameters and convergence tolerances are tabulated in Table 1.

Table 1: Tabulated are the PID parameters for the airfoil motion, and voltage follower code.

	Motion Amplitude	Motion DC Offset	Motion Frequency	Voltage Follower
Proportional Constant	0.75	0.75	0.75	0.6
Integral Constant	0	0	0	10
Derivative Constant	0	0	0	0
Convergence Tolerance	1%	0.2°	1%	N/A

Unfortunately, the motion of the servo is not consistent from cycle to cycle as shown in Figure 3.3. This figure shows the motion for an airfoil oscillating sinusoidally from $\alpha = 8^\circ$ to $\alpha = 18^\circ$ at a physical frequency of 3.48 Hz (this corresponds to $k = 0.1$ at $Re = 300,000$). Although the individual cycles can vary by as much as a degree, the averaged result is very close to the desired motion.

Figure 3.3: Motion of airfoil plotted against time. Plot on the left shows a few individual cycles plotted against the desired motion (blue). Plot on the right shows the averaged motion with 95% confidence intervals against the desired motion (blue).

The motion of the airfoil in the reminder of the paper is described by three parameters, the mean angle (α_m), the amplitude ($\Delta\alpha$), and the reduced frequency ($k = \pi f L / u_\infty$, where f is the physical oscillation frequency of the airfoil). As such the desired motion is described by Equation 3-2:

$$\alpha(t) = \alpha_m - \Delta\alpha \cos\left(\frac{2ku_\infty}{L}t\right) \quad \text{Equation 3-2}$$

B. Data Acquisition System Synchronization

Three primary DAQ systems are utilized in the facility. They are: 1) NI-DAQ systems, 2) Scanivalve Pressure Scanners, 3) ABB Servo Controller. The unsteady behavior of dynamic stall makes synchronization between all three systems critical. Synchronization is performed to the NI-DAQ system.

Synchronization between the NI-DAQ and Scanivalve pressure scanners is achieved using a speaker that continually plays a chirp signal. The speaker (Visaton FRS8-4 with a Sony STR-DH100 audio amplifier) is housed in a sealed enclosure with three static pressure taps. Each pressure tap is connected to a different pressure array. The setup is show in Figure 3.4. The chirp signal is generated in LabVIEW and is recorded by the NI-DAQ system. Subsequently, the output of the speaker is recorded by each pressure array. Shown in Figure 3.5 are two plots. These plots show the time traces of the speaker input voltage and speaker output pressure recorded by the NI-DAQ and the Scanivalve pressure array, respectively. The left plot shows the uncorrected curves and the right curves shows the speaker output pressure after the time correction is applied. The

correction is applied by computing the time lag in each cycle of data, which is calculated by taking the peak of a cross correlation.

Figure 3.4: Physical setup of speaker and pressure transducers. Speaker is placed as close as possible to the pressure transducers to minimize impacts of tubing lengths.

Figure 3.5: Time traces of the speaker input voltage (blue) and the speaker output pressure (orange) recorded by the NI-DAQ system and the Scanivalve pressure array, respectively. The plot on the left shows the uncorrected curves and the right plot shows the speaker output pressure after the time correction is applied.

A pressure transducer (Kulite model XTL-140-25A with a Kemo Benchmaster 21M Filter set to a gain of $10k$ and a low pass filter frequency of 25 kHz) was used to record the output of the speaker. The output voltage of the transducer was measured by the NI-DAQ systems. This transducer was used to determine the constant lag between the speaker input and speaker output. This constant was determined to be 7.50 ms and is factored in when the time correction is applied. While using a pressure transducer during the tests could have eliminated the need for this correction, pressure transducers tend to be very susceptible to electromagnetic interference (EMI) and as such, cannot be used in conjunction with plasma actuation.

Synchronization between the NI-DAQ and ABB servo controller is relatively less complicated. The servo position is recorded both using an analog output on the servo controller (high temporal resolution) and by network communication (high angular resolution). Given the lower resolution of the analog output of the servo, the position given by the network communication is recorded, and then aligned to the analog output of the servo using cross-correlations. The adjustments made by the cross-correlations tend to be small, as shown in Figure 3.6.

Figure 3.6: Lag between servo position and servo position provided by network connection.

The accuracy of the cross correlation is a sample size, or 2.5 ms . As such, in terms of the synchronization, the motion of the airfoil and the pressure data is only accurate to 2 samples or 5 ms . The effect on the accuracy of the data depends on the angular velocity of the airfoil, i.e. lower reduced frequencies will result in better accuracy

as the maximum angle of attack deviation during a 5 ms period is much smaller. The effect of this inaccuracy on the lift coefficient is shown in.

Figure 3.7: Effect of cross correlation accuracy on the angle of attack and lift coefficient. The red and blue curves represent either extreme of the accuracy. The blue curve on the angle of attack vs time plot (left) corresponds to the blue curve on the lift coefficient vs angle of attack curve (right).

C. Plasma Actuator

A single actuator was placed on the airfoil to evaluate the efficacy of excitation. The actuator is constructed of two 0.09 mm thick copper tape electrodes; the exposed high-voltage electrode is 6.35 mm wide and the covered ground electrode is 12.70 mm wide. The dielectric layer is composed of three layers of Kapton tape, each 0.09 mm thick with a dielectric strength of 10 kV . The total thickness of the entire actuator is 0.45 mm . The actuator was placed on the suction side of the airfoil with the electrode junction at $x/c = 0.01$ and covers the entire span of the airfoil.

Although the actuators are thin ($0.0022c$), the physical presence of the actuator does have an impact on the flow. Shown in Figure 3.8 are lift curves for two different cases at $Re = 167,000$. The red curves indicate that the actuator was mounted to the

airfoil, but the actuators were not turned on. The blue curves are the cases without an actuator present, and indicate the clean, baseline case. The plot on the left oscillates with $\alpha_m = 13^\circ$, $\Delta\alpha = 5^\circ$, and $k = 0.050$, whereas the motion of the plot on the right is described by $\alpha_m = 10^\circ$, $\Delta\alpha = 10^\circ$, and $k = 0.050$. As indicated in the plots, the lift peak due the dynamic stall vortex is delayed and weakened in magnitude due to the presence of the actuator. Thus, excited data will be compared to data acquired with the actuator mounted to the airfoil but turned off (referred to as baseline or $St_e = 0$).

Figure 3.8: Lift curves for two cases at a $Re = 167,000$ with and without a plasma actuator. Curves on the left have a motion described by $\alpha_m = 13^\circ$, $\Delta\alpha = 5^\circ$, and $k = 0.050$, whereas curves on the right have a motion described by $\alpha_m = 10^\circ$, $\Delta\alpha = 10^\circ$, and $k = 0.050$. Actuator presence clearly delays and weakens the effects of the dynamic stall vortex.

The actuator is powered by a custom, in-house manufactured pulse generator. The pulse generator utilizes a magnetic compression circuit to create the input waveform for the actuator. A DC power supply with a 450 VDC output is used to power the pulse generator. The specifics of the pulse generator are discussed in previous work [37, 40].

Representative discharge characteristics were acquired for a ~570 millimeter long actuator driven at 100 Hz. Voltage and current traces are acquired using a LeCroy Wavejet oscilloscope (model 324A) simultaneously. A Tektronix high-voltage probe (model P6015A) is placed across the output terminals of the pulse generator and a Pearson current probe (model 2877) is placed on the ground output terminal of pulse generator (which is connected to the ground of the actuator). A total of 16 pulses were acquired and averaged. The voltage and current traces of this input waveform are shown in Figure 3.9, along with the power and instantaneous energy traces.

Figure 3.9: Voltage-Current traces (left) and Power-Energy traces (right).

The peak voltage was 10 kV and the peak current was 36 A. The peak power consumption was 335 kW. However due to the narrow pulse width, the steady state energy consumption was 12.6 mJ per pulse. For the frequencies considered (less than 1800 Hz), this corresponds to a maximum time averaged power of 23 W. At the testing

conditions, this corresponds to a maximum of 6% of the freestream flow energy (based on the cross section of the test section).

D. Static Pressure

Static pressure measurements on the airfoil surface were acquired using three Scanivalve digital pressure sensor arrays (DSA-3217). A total of 35 taps are located on the surface of the airfoil and the tap distribution is shown in Figure 3.10. The pressure coefficient, $C_p = (p - p_\infty)/q_\infty$, was phase averaged over 4 sets of 16384 samples acquired at 400 Hz near the centerline, where p is the static pressure, p_∞ is the freestream static pressure, and q_∞ is the freestream dynamic pressure. The sectional lift coefficient was calculated using the line integral $C_L = - \int_{-1}^1 C_p \sin \theta ds$, where θ is the surface-normal angle and ds is the arc length. Similarly, the sectional pressure drag coefficient was calculated as $C_D = - \int_{-1}^1 C_p \cos \theta ds$. The moment was calculated as $C_M = - \int_{-1}^1 C_p \times (\frac{1}{4} \sin \theta - x \sin \theta - y \cos \theta) ds$. The moment is calculated about the quarter chord of the airfoil, and a positive moment is defined to be a moment that tends to decrease the angle of attack.

Figure 3.10: Static pressure tap distribution for NACA 0015 Airfoil.

Although pressure data from the pressure scanners is acquired at 400 *Hz*, data from the NI-DAQ systems is recorded at 500 *Hz*. Splines are used on data acquired by the NI-DAQ systems to align the data with the pressure scanners. Pressure data is conditionally filtered to remove cycles that are excessively noisy. This is achieved in two steps. First the lift is computed for each cycle and compared to the mean global lift. If the difference between the two is within the magnitude of the mean global lift, the cycle is kept. Second the standard deviation of the cycle lift is compared against the standard deviation of the global lift. If the difference between the two is within the magnitude of the standard deviation of the global lift, the cycle is kept. Furthermore, cross correlations are used to ensure that the lift curves are temporally aligned cycle to cycle. The cycles are binned based on their time lag to the first cycle recorded by the data acquisition systems,

and each bin size is 10 *samples* or 25 *ms* (at 400 *Hz*). The largest bin is used to represent that data set and cycle averaged.

Due to the long pressure tubing lengths, lag/gain compensation is necessary as shown by Bergh et al. [41]. However, the presence of various couplings and adaptors make the direct application of the Bergh-Tijdeman model inaccurate. As such, empirical lag/gain compensation is computed using a speaker and pressure transducer (the same setup used to determine the delay between the speaker input voltage and the speaker output pressure), as well as a static pressure tap. This static pressure tap is connected to a pressure tap on the airfoil and subsequently measured by the pressure scanners. By measuring both the pressure input and output and repeating this process for each pressure tap on the airfoil, empirical lag/gain compensation curves can be computed. The Bergh-Tijdeman model is then fit (using a nonlinear least squares) to these curves (which are distinct for each pressure tap), and the resulting correction is then applied to subsequent data. The model parameters are archived for each pressure tap. A sample gain curve is shown in Figure 3.11.

Figure 3.11: Amplitude ratio as computed using the experimental data (blue) and curve fit (orange) using the Bergh-Tijdeman (B-T) model.

E. Particle Image Velocimetry

Particle image velocimetry (PIV) was the primary diagnostic technique. PIV is a quantitative technique that enables the mapping of the velocity and vorticity fields. PIV data was acquired for both the flow surrounding the suction side of the airfoil and behind

the trailing edge. Maps of the normalized velocity magnitude, $U^* = U/u_\infty$, normalized vorticity, $\omega^* = \nabla^* \times U^*$, and turbulent kinetic energy, $TKE = \frac{1}{2}(u_{rms}^2 + v_{rms}^2)$, are shown to characterize the flow. In the plots, there is an arc of corrupted data. This is caused by the aluminum rings that secure the acrylic windows to the facility as shown in Figure 3.12. It is not representative of any flow phenomenon.

Figure 3.12: Dual camera PIV experimental setup.

The seed particles were injected upstream of the test section. Extra virgin olive oil was atomized using a TSI 6-jet atomizer (model 9306A). The seed particles are illuminated using a Spectra Physics PIV-400 double-pulsed Nd:YAG laser. The laser beam is formed into a sheet with the use of a 1 m focal length spherical lens and one 25 mm focal length cylindrical lens. Various turning optics were used to direct the laser sheet into the wind tunnel. The laser sheet has a thickness of $\sim 2\text{mm}$ and is located at 60.4% (of the span of the airfoil) from the far end of the acrylic test section.

PIV measurements were acquired using two LaVision 12 *bit* 2048×2048 pixels Imager Pro camera bodies, each with a Nikon Nikkor 50 *mm f*/1.2 lens. The cameras were positioned 73.5 *mm* center to center on a horizontal optical rail. The lenses of the cameras were located ~ 980 *mm* away from the laser sheet. The cameras acquired data simultaneously at an acquisition rate matching the frequency of motion. For excited cases, five sets of 100 *image pairs* were taken for a given case, where as in baseline data, a single set of 500 *image pairs* were taken.

Data processing is done in the commercially available DaVis software by LaVision. For each image pair, a multi-pass cross correlation was used. The first window size used was 64x64 pixels with a 50% overlap, followed by two 24x24 pixels windows with a 75% overlap. The final pass was performed using a B-spline-6 reconstruction. The velocity fields were then post-processed to remove spurious vectors using a correlation peak ratio criterion, allowable vector range, and median filter based on nearest neighbor. The resulting velocity field was then smoothed using a 3x3 pixel Gaussian smoothing filter. Afterward, the vector fields from each camera were stitched together and averaged.

Full scale error was computed using $\epsilon_{vel} = \epsilon_{cp}/(S dt)$, where ϵ_{vel} is the full scale error [*m/s*], ϵ_{cp} is the correlation peak estimation error [*pixels*], S is the scaling factor [*pixels/m*], and dt is the laser pulse separation [*s*]. The laser timing error was assumed to be negligible and ϵ_{cp} was taken to be 0.1 *pixels* [42]. As such, this corresponds to an error in instantaneous velocity of 0.74 *m/s* (or at most 6% of the freestream velocity).

Only PIV data with the acquisition phase-locked to the airfoil motion was acquired. Instantaneous images were captured, synchronized with specific phases of the motion, and then averaged. The phases considered vary on a case by case basis based on corresponding static pressure data. Swirling strength was used as the vortex identification technique, as detailed by Adrian et al. [43], and was non-dimensionalized by the freestream velocity divided by the chord.

F. Facility Validation

As a precursor to experimentation, the new facility and DAQ codes had to be validated. First the repeatability of the setup was examined by acquiring multiple runs of various cases. Although many unique cases were acquired, the lift and moment curves for only two cases are presented in Figure 3.13. In both cases shown, the mean angle was $\alpha_m = 13^\circ$ and the amplitude was $\Delta\alpha = 5^\circ$, and the reduced frequency was $k = 0.050$. The plots on the top have a Reynolds number of $Re = 167,000$ and the plots on the bottom have a Reynolds number of $Re = 300,000$. Experimental repeatability was found to be good.

Figure 3.13: Resulting lift coefficient (left) and moment coefficient (right) for two runs with a mean angle of $\alpha_m = 13^\circ$, amplitude of $\Delta\alpha = 5^\circ$, and reduced frequency of $k = 0.050$ at $Re = 167,000$ (top) and $Re = 300,000$ (bottom). Red and blue curves indicate different experiments.

Given the complex nature of unsteady flow, it was difficult to find an exact match in literature for a direct comparison. This is due to the combination of our experimental limitations (low speed facility, and low physical oscillation frequency capability), and the

airfoil. Although the NACA 0015 airfoil is a well-studied airfoil in static testing, it is seldom used in unsteady flow. Data acquired by Greenblatt et al. [44] was similar to facility's capability and experimental setup with a few key differences.

Firstly the $C_L - \alpha$ curves for static testing (see Figure 3.14) do not match. Although both facilities utilized a NACA 0015 airfoil and $Re = 300,000$, the airfoil used in this study has a much sharper stall as compared to Greenblatt et al. [44]. This will result in a much greater lift hysteresis in comparison to Greenblatt et al. [44], when the airfoil motion exceeds the static stall angle. This sharp stall is also observed in literature [12, 45]. The static curves also show the small difference in the lift coefficient prior to stall. This will result in slight differences during the upstroke of the motion. Despite these differences, the stall angle of both airfoils is identical $\alpha_{ss} \cong 13^\circ$.

Figure 3.14: Static $C_L - \alpha$ curves for NACA 0015 at $Re = 300,000$.

The work of Greenblatt et al. [44] also uses two roughness elements (grit #100 sandpaper with a chordwise length of $0.1L$) at the leading edge of the airfoil ($x/c = 0.1$), one on either side of the airfoil. The roughness elements are used throughout testing in the work of Greenblatt et al. [44]. When scaled roughness elements (grit #180 sandpaper mounted at $x/L = 0.1$ with a length of $0.1L$) were mounted to this study's airfoil, the changes in the lift and moment coefficient were not deemed significant as shown in Figure 3.15. As such, the roughness elements were removed from the airfoil.

Figure 3.15: Lift and moment coefficient for two cases. Top plots are taken at $\alpha_m = 13^\circ$, $\Delta\alpha = 5^\circ$, $k = 0.050$, and $Re = 300,000$. Bottom plots are taken at $\alpha_m = 13^\circ$, $\Delta\alpha = 5^\circ$, $k = 0.10$, and $Re = 167,000$. Blue curves are the results for a clean airfoil, and red curves are the results with the roughness elements.

Another noted difference between our arrangement and that of Greenblatt et al.

[44] is the tunnel height to airfoil chord ratio, H/L , (see for Duraisamy et al. [46] for

specifics). Greenblatt et al. [44] utilized a facility with a $H/L = 2$, whereas this study utilized a facility with a $H/L = 1.5$. It is well documented by McAlister [47, 48] and Duraisamy [46] that short tunnels exhibit a more pronounced pressure changes due to the dynamics stall vortex than taller tunnels. Shown in Figure 3.16 is a figure taken from McAlister et al. [48] that shows the lift coefficient for the NACA 0012 undergoing unsteady flow at a $k = 0.10$ in two different facilities. The curve labeled “Present” was taken from a facility with a $H/L = 2.5$ and the curve labeled “Ref. 3” was taken from a facility with a $H/L = 1.25$. As indicated in the figure, the curves are very similar. The exception occurs between $\alpha = 20^\circ$ and $\alpha = 25^\circ$. In the curve labeled “Ref. 3”, there is a much stronger lift peak observed as compared to the curve labeled “Present”. This is attributed to blockage effects [48], and the interference between the boundary layers of the top and bottom walls of the test section and the unsteady boundary layer of the airfoil.

Figure 3.16: Figure taken from McAlister et al. [48]. A NACA 0012 airfoil was used and the experimental parameters were $\alpha_m = 15^\circ$, $\Delta\alpha = 10^\circ$, $k = 0.10$, and $M_\infty = 0.1$. The curve labeled “Present” has a $H/L = 2.5$, and the curve labeled “Ref. 3” has a $H/L = 1.25$.

Lastly, shown in Figure 3.17, is the static $C_M - \alpha$ curve. Unlike the static $C_L - \alpha$ curve which matches reasonably well with Greenblatt et al. [44], the static moment curves do not match well. As such, it is not expected that the dynamic moment curves will align well with Greenblatt et al. [44]. Unsteady flow data is acquired at similar flow conditions ($Re = 300,000$) and motion parameters (vary with case) as Greenblatt et al. [44]. Presented in Figure 3.18, Figure 3.19, Figure 3.20, Figure 3.21, and Figure 3.22 are the lift and moment curves for five cases for comparison with Greenblatt et al. [44]. The captions of these labels also indicate the motion parameters for both cases. For the most part, the moment curves do not match very well with each other. The trends are similar, but the magnitudes are different.

In Figure 3.18 and Figure 3.19, the motion of the airfoil does not exceed the static stall angle of the airfoil. As such, good agreement is seen between the data sets. There does appear to be a slight offset between the curves; but, as previously discussed, this is accounted for by the differences in the static lift curves. In Figure 3.20 and Figure 3.21, the motion of the airfoil does exceed the static stall angle. During the pitch up motion, the lift curves are in good agreement with one another. As the airfoil reaches the maximum angle of attack, and the dynamic stall vortex is shed, the experimental data of this study indicates a stronger signature of the dynamic stall vortex. This is attributed to blockage effects (as discussed previously) and explains the lift peak, that is not present in the data of Greenblatt et al. [44]. During the downstroke of the airfoil of this study, the flow is fully separated and indicates a greater lift hysteresis due to the difference in the stall behavior between the two airfoils. It is also interesting to observe that the flow does not reattach during the downstroke; this is common to these four cases. The last case, Figure 3.22, exhibits similar behavior as the previous case – stronger dynamic stall vortex signature and increased lift hysteresis. However, in this case, the lift curves do not align as well as in previous cases. This is explained by the fact that the flow does not re-attach prior to the start of the motion. As seen in curve of Greenblatt et al. [44], the slope of the lift curve is not constant until the angle attack reaches $\alpha = 14^\circ$ which is well into the upstroke. In the data of this study, a greater lift hysteresis is observed. Thus the re-attachment seems more severe, but close observation indicates that the slope of the lift curve steadies at $\alpha = 14^\circ$. As such, this case also compares well with Greenblatt et al.

[44]. With all of the discrepancies accounted for, it allows for both validation of the facility and the understanding of the influence of the experimental setup on the results.

Figure 3.17: Static $C_M - \alpha$ curves for NACA 0015 at $Re = 300,000$.

Figure 3.18: Lift and moment coefficient taken at $\alpha_m = 5$, $\Delta\alpha = 5^\circ$, $k = 0.10$, and $Re = 300,000$.

Figure 3.19: Lift and moment coefficient taken at $\alpha_m = 9$, $\Delta\alpha = 5^\circ$, $k = 0.10$, and $Re = 300,000$.

Figure 3.20: Lift and moment coefficient taken at $\alpha_m = 13$, $\Delta\alpha = 5^\circ$, $k = 0.10$, and $Re = 300,000$.

Figure 3.21: Lift and moment coefficient taken at $\alpha_m = 13$, $\Delta\alpha = 5^\circ$, $k = 0.050$, and $Re = 300,000$.

Figure 3.22: Lift and moment coefficient taken at $\alpha_m = 17$, $\Delta\alpha = 5^\circ$, $k = 0.10$, and $Re = 300,000$.

4. Results and Discussion

After the facility was validated the effect of excitation on the baseline flow was examined. Flows at three Reynolds numbers were investigated ($Re = 167k, 300k$, and $500k$), at three reduced frequencies ($k = 0.025, 0.050$, and 0.075). For each combination of Reynolds number and reduced frequency, 20 excitation Strouhal numbers were acquired. Guided by pressure data, phased-locked PIV data was acquired at one combination of Reynolds number ($Re = 300k$) and reduced frequency ($k = 0.050$) and for three excitation Strouhal numbers. The only motion considered is a sinusoidal waveform with an amplitude of 10° and a mean of 10° , which is in the regime of deep stall.

A total of 2^{16} (65536) *samples* of pressure data were collected at 400 Hz for each case. In excited cases, these samples were collected in four runs due to the limited run time of the actuators. This allowed a minimum of 50 cycles of data to be obtained at the lower reduced frequencies. This amount of data was sufficient to yield converged lift and moment curves [49].

For phase-locked (to the motion of the airfoil) PIV results, 500 image pairs were taken at the frequency of the motion. Again, due to the limited run time of the actuators, the data was acquired in four runs for the excited cases. The phase of the results is

consistent with the sinusoidal equation – for example, a phase of 0° corresponds to an angle of attack of 10° on the up stroke and a phase of 330° corresponds to an angle of attack of 5° on the up stroke.

When analyzing the phase-locked PIV data, it is important to understand the uncertainty of the data. Shown in Figure 4.1 is the root-mean-square of a particular phase where the flow is completely separated. In this figure, it is observed that the root-mean-square is on the order of the freestream velocity in the separation region. Three distinct processes cause this: turbulence in the separation region, repeatability error in the airfoil motion, and error inherent in the PIV process.

Figure 4.1: Phase-locked PIV results taken at $Re = 300,000$, $k = 0.050$, and $\phi = 90^\circ$ (which corresponds to an angle of attack of 20°). The down arrow indicates that the airfoil is pitching down from the maximum angle of attack. Shown is the root-mean-square of the streamwise velocity normalized by the freestream velocity.

The latter two can be characterized by the 95% confidence interval of the data, which is shown in Figure 4.2. The confidence interval is an order of magnitude smaller than the root-mean-square (Figure 4.1), indicating that the majority of the unsteadiness is indeed flow related, and not an artifact of the experimental setup.

Figure 4.2: Phase-locked PIV results taken at $Re = 300,000$, $k = 0.050$, and $\phi = 90^\circ$ (which corresponds to an angle of attack of 20°). Shown is the confidence interval of the streamwise velocity normalized by the freestream velocity.

A. Baseline Results

Figure 4.3 shows the baseline phase-averaged lift and moment coefficients for the lowest Reynolds number and reduced frequency tested. In this figure, the five stages of dynamic stall, as outlined by Corke et al. [5] and shown in Figure 2.2, are numbered.

The process of dynamic stall begins as the airfoil begins pitching up. In this stage, the flow is attached and the lift increases steadily with angle of attack. When the angle of attack exceeds the static stall angle of attack still increases, the airfoil is in the second

stage of dynamic stall [5]. In this stage, the lift increases due to the delayed separation – a consequence of the airfoil motion and the formation of a closed separation bubble near the leading edge of the airfoil. This is also observed in the C_p distribution, shown in Figure 4.4, where the flow remains attached up to 17° and past the static stall angle of 13° . Although C_L and C_D are measures of the aerodynamic forces and may provide some insight into the state of the flow, the C_p distribution provides a more direct metric for understanding flow attachment.

In the second stage, the vorticity at the leading edge accumulates and the dynamic stall vortex begins to form. This results in unsteady loading [5], and can be observed by the large confidence intervals at 9.6° and 16.7° in Figure 4.4.

Figure 4.3: Baseline phase-averaged lift and moment coefficient at $Re = 300,000$ and $k = 0.050$. Dark colors indicate pitch up motion, whereas lighter colors indicate pitch down motion. Numbering is used to correlate phase of airfoil to the stages of dynamic stall as indicated by Corke et al. [5].

Figure 4.4: Baseline phase-averaged suction side pressure coefficient at $Re = 300,000$ and $k = 0.050$. Angle of attack of the airfoil is displayed on right of plots.

This accumulation of vorticity is better visualized by the PIV results, shown in Figure 4.5. This is more clearly evident in phases, $\phi = 37^\circ, 44.5^\circ, 53^\circ$, and 330° ($\alpha = 10^\circ, 16^\circ, 17^\circ$, and 18° , respectively), near the leading edge of the airfoil. Close up results are provided in Figure 4.6. The growth of this vorticity is significant and relatively rapid as shown in Figure 4.6.

Figure 4.5: Shown here are the baseline phase-averaged vorticity maps at $Re = 300,000$ and $k = 0.050$. Angle of attack of the airfoil is displayed on left of plots, along with the phase. The up arrows indicate that the airfoil is pitching up.

Figure 4.6: Close up of baseline phase-averaged vorticity maps at $Re = 300,000$ and $k = 0.050$. Angle of attack of the airfoil is displayed on left of plots, along with the phase. The up arrows indicate that the airfoil is pitching up.

As the angle of attack increases, the airfoil undergoes the third stage of dynamic stall [5] – the ejection of this vorticity in the form of the dynamic stall vortex. The ejection of this vortex is attributed to vortex induced separation [18]. As observed by Mulleners et al. [18], the dynamic stall vortex formation is accompanied by counter rotating vortices on the airfoil’s surface. These vortices travel upstream, pushed by the dynamic stall vortex. This results in the detachment of the dynamic stall vortex and is termed as vortex induced separation [50]. Vorticity plots shown in Figure 4.5, indicate the presence of two streams of counter-rotating vorticity. Swirling strength is shown in Figure 4.7, and shows the dynamic stall vortex convecting over the airfoil at $\phi = 73.5^\circ$ ($\alpha = 19.6^\circ$). However, the vortex does not seem to be well defined. As indicated in the

swirling strength, there are two streams of vortices, which is expected given the vorticity results shown in Figure 4.5.

Figure 4.7: Baseline phase-averaged normalized swirling strength at $Re = 300,000$ and $k = 0.050$. Angle of attack of the airfoil is displayed on left of plots, along with the phase. The up arrows indicate that the airfoil is pitching up.

In order to highlight vortex formation, as well as the direction of the vorticity, the swirling strength has been multiplied by the sign of the vorticity in Figure 4.9. In this figure, the counter rotating vortices are much more evident.

The shape of the dynamic stall vortex in the PIV results can be explained by looking at Figure 4.8. The figure illustrates the suction side pressure coefficient as it varies chordwise (horizontal axis) and in time (vertical axis). Vortices produce pressure spikes that move along the chord and can be identified [51] by red streaks in the surface pressure. The dynamic stall vortex is shown by the green dashed line on the left plot. On the right plot, the phase $\phi = 73.5^\circ$ ($\alpha = 19.6^\circ$) is shown by the blue dashed line. This

supports the conclusion that the dynamic stall vortex is convecting at this time. However, the width of the red streak indicates that phase at which the dynamic stall vortex convects is not consistent from cycle-to-cycle. This is attributed to both the stochastic nature of dynamic stall, and the variation in the motion profile.

Figure 4.8: Baseline suction side pressure coefficient at $Re = 300,000$ and $k = 0.050$, showing the dynamic stall vortex (left), and PIV phase (right).

The convection of the dynamic stall vortex results in an additional increase in lift, as shown by the lift peak near $\alpha = 19^\circ$ in Figure 4.3. The vortex also results in a sharp decrease in pitching moment, since the vortex moves the airfoil center of pressure downstream as it convects [52]. It is known that moment stall precedes lift stall [5]; however, due to the limited temporal and spatial resolution of the pressure taps, they appear to occur simultaneously.

After the dynamic stall vortex has convected, the airfoil is fully stalled. This marks the fourth stage of the dynamic stall process [5], and corresponds to the sharp drop

in the lift, as seen in Figure 4.3. This drop in lift is due to flow separation as shown in Figure 4.10. This figure shows the streamwise velocity for several phases in the downstroke.

The last stage of dynamic stall is flow reattachment and is shown in the latter phases of the streamwise velocity, Figure 4.10. The flow reattachment is a stochastic process, in which random features of the shear layer result in large cycle to cycle variation in aerodynamic loads during recovery [53, 54, 5].

Figure 4.9: Baseline phase-averaged normalized swirling strength multiplied by the sign of the vorticity at $Re = 300,000$ and $k = 0.050$. Angle of attack of the airfoil is displayed on left of plots, along with the phase. The up arrows indicate that the airfoil is pitching up.

Figure 4.10: Baseline phase-averaged normalized streamwise velocity at $Re = 300,000$ and $k = 0.050$. Angle of attack of the airfoil is displayed on left of plots, along with the phase. The down arrows indicate that the airfoil is pitching down.

B. Excited Results

Shown in Figure 4.11 is the phase averaged lift coefficient for the baseline and excitation results for $Re = 167,000$ and $k = 0.025$. Although a total of 20 different excitation Strouhal numbers were tested (as indicated by the figure), only four ($St_e = 0, 0.3, 0.78, \text{ and } 9.9$) will be shown for clarity for every combination of Reynolds number and reduced frequency. The trends for each combination of Reynolds number and reduced frequency are similar and nothing is lost by the omission of these results. In these figures, the confidence interval will not be shown; instead, the confidence interval half-width is tabulated in Table 2. Since the confidence interval is comparable for a given combination of Reynolds number and reduced frequency, only the mean baseline confidence interval half width is recorded. The figures are labeled Figure 4.12 to Figure 4.20.

Due to time constraints, PIV data was only acquired at one combination of Reynolds number and reduced frequency. This case was $Re = 300,000$ and $k = 0.05$. Only three excitation Strouhal numbers were tested, $St_e = 0$ (*baseline*), 0.3 , and 9.9 . Phase-locked (to the angle of attack of the airfoil) PIV data was collected at 17 phases for the baseline case, and 9 phases for the excitation cases.

The trend of the effects of excitation on the overall progression of dynamic stall appears to be similar for different combinations of Reynolds number and reduced frequencies. Each combination follows three main trends:

1. Low excitation Strouhal numbers ($St_e < 0.5$) result in oscillatory lift and drag after the dynamic stall vortex has shed. This behavior smooths out, as the excitation Strouhal number increases.
2. All excited cases observe earlier flow reattachment as compared to baseline data.
3. All excited cases observed reduced lift and moment hysteresis and decreased dynamic stall vortex strength.

Figure 4.11: Phase-averaged lift coefficient for various excitation Strouhal numbers for $Re = 167,000$ and $k = 0.025$. Dark colors indicate pitch up motion, whereas lighter colors indicate pitch down motion.

Figure 4.12: Phase-averaged lift coefficient and moment coefficient for various excitation Strouhal numbers at $Re = 167,000$ and $k = 0.025$. Dark colors indicate pitch up motion, whereas lighter colors indicate pitch down motion.

Figure 4.13: Phase-averaged lift coefficient and moment coefficient for various excitation Strouhal numbers at $Re = 167,000$ and $k = 0.050$. Dark colors indicate pitch up motion, whereas lighter colors indicate pitch down motion.

Figure 4.14: Phase-averaged lift coefficient and moment coefficient for various excitation Strouhal numbers at $Re = 167,000$ and $k = 0.075$. Dark colors indicate pitch up motion, whereas lighter colors indicate pitch down motion.

Figure 4.15: Phase-averaged lift coefficient and moment coefficient for various excitation Strouhal numbers at $Re = 300,000$ and $k = 0.025$. Dark colors indicate pitch up motion, whereas lighter colors indicate pitch down motion.

Figure 4.16: Phase-averaged lift coefficient and moment coefficient for various excitation Strouhal numbers at $Re = 300,000$ and $k = 0.050$. Dark colors indicate pitch up motion, whereas lighter colors indicate pitch down motion.

Figure 4.17: Phase-averaged lift coefficient and moment coefficient for various excitation Strouhal numbers at $Re = 300,000$ and $k = 0.075$. Dark colors indicate pitch up motion, whereas lighter colors indicate pitch down motion.

Figure 4.18: Phase-averaged lift coefficient and moment coefficient for various excitation Strouhal numbers at $Re = 500,000$ and $k = 0.025$. Dark colors indicate pitch up motion, whereas lighter colors indicate pitch down motion.

Figure 4.19: Phase-averaged lift coefficient and moment coefficient for various excitation Strouhal numbers at $Re = 500,000$ and $k = 0.050$. Dark colors indicate pitch up motion, whereas lighter colors indicate pitch down motion.

Figure 4.20: Phase-averaged lift coefficient and moment coefficient for various excitation Strouhal numbers at $Re = 500,000$ and $k = 0.075$. Dark colors indicate pitch up motion, whereas lighter colors indicate pitch down motion.

Table 2: Confidence interval half width for baseline cases for every combination of Reynolds number and reduced frequency.

Re	k	ci_{c_L}	ci_{c_M}
167,000	0.025	0.027	0.007
167,000	0.050	0.028	0.008
167,000	0.075	0.030	0.009
300,000	0.025	0.020	0.004
300,000	0.050	0.018	0.004
300,000	0.075	0.035	0.007
500,000	0.025	0.028	0.008
500,000	0.050	0.086	0.013
500,000	0.075	0.100	0.014

1. Excitation Effects in the Stalled Regime

The first trend – that excitation introduces oscillatory behavior into the stalled regime of dynamic stall is indicative of the formation of structures. As shown by previous work with the actuators [13, 14, 15, 11], low Strouhal number excitation leads to the formation of coherent structures at post-stall angle of attacks. If the excitation pulse is paired with the moment and lift curves for the low excitation Strouhal number case, it is observed that every pulse results in the formation of a structure. This is shown in Figure 4.21. The large structures were also detected using PIV results. The swirling strength for the stalled phases at low Strouhal number excitation is shown in Figure 4.22. As the excitation Strouhal number increases, the magnitude of the oscillatory behavior decreases, but the frequency scales with the excitation Strouhal number, shown in Figure 4.23. At high excitation Strouhal numbers, this oscillatory behavior is not observed. This is expected given that higher excitation Strouhal numbers result in the formation of more, smaller structures further upstream on the airfoil which quickly develop, disintegrate, and

dissipate [13, 14, 15, 11]. Again, this is well captured by PIV data, shown in Figure 4.24. It should also be mentioned, that due to the limited spatial resolution of pressure measurements and temporal resolutions of the pressure measurement system high frequency events would be filtered out. Thus, while similar signature oscillations may exist for the high-Strouhal number cases, they would not be manifest in this data.

Figure 4.21: Phase-averaged lift coefficient and moment coefficient for $St_e = 0.3$ at $Re = 300,000$ and $k = 0.050$. Dark colors indicate pitch up motion, whereas lighter colors indicate pitch down motion. Magenta stars are used to indicate the firing of the actuator.

Figure 4.22: Phase-averaged normalized swirling strength at $Re = 300,000$, $k = 0.050$, and $St_e = 0.35$. Angle of attack of the airfoil is displayed on left of plots, along with the phase. The down arrows indicate that the airfoil is pitching down.

Figure 4.23: Phase-averaged lift coefficient and moment coefficient at $Re = 300,000$ and $k = 0.050$. Dark colors indicate pitch up motion, whereas lighter colors indicate pitch down motion.

Figure 4.24: Phase-averaged normalized swirling strength at $Re = 300,000$, $k = 0.050$, and $St_e = 9.9$. Angle of attack of the airfoil is displayed on left of plots, along with the phase. The down arrows indicate that the airfoil is pitching down.

2. Excitation Effects on Reattachment

The second trend is easily observed from previous figures (Figure 4.11, Figure 4.12, Figure 4.13, Figure 4.14, Figure 4.15, Figure 4.16, Figure 4.17, Figure 4.18, Figure 4.19, and Figure 4.20) – that excitation results in earlier reattachment. This is related to the first trend. The formation of coherent structures entrains high speed flow and results in momentary reattachment [13, 14, 15, 11]. The formation of small structures (such as those produced by high excitation Strouhal numbers) only results in partial reattachment

[13, 14, 15, 11]. This is also observed in the PIV data presented (Figure 4.25). Regardless of the time of reattachment, once the airfoil reaches an angle of attack less than its static stall angle, α_{ss} , the flow remains attached.

Upon closer examination of the lift and moment coefficient curves, it is seen that there are some variations between the reattachment angle of attack for a given Reynolds number and reduced frequency. In Figure 4.26, the reattachment angle of attack (which is defined as the angle of attack the coefficient of lift is equal on the downstroke and upstroke) is plotted as a function of the excitation Strouhal number. At low excitation Strouhal numbers ($St_e < 2$), there is more discrepancy in the reattachment angle of attack. This is attributed to the timing of the actuator pulse relative to the motion of the airfoil. If vortex-induced flow attachment does not occur before the airfoil angle of attack is less than the critical angle of attack, then reattachment does not occur until the next pulse. Thus timing is critical as at low excitation Strouhal numbers, the period between pulses is relatively large. However, at high excitation Strouhal numbers, the flow remains attached, and thus there is less variation in the reattachment α for the higher Strouhal numbers ($St_e > 6$). Flow attachment is best observed by the suction side pressure coefficient shown in Figure 4.27.

Figure 4.25: Phase-averaged normalized streamwise velocity results at $Re = 300,000$, $k = 0.050$, and $St_e = 9.9$. Angle of attack of the airfoil is displayed on left of plots, along with the phase. The down arrows indicate that the airfoil is pitching down.

Figure 4.26: Reattachment angle of attack versus the excitation Strouhal number at $Re = 300,000$ and $k = 0.050$. Note that by definition, the reattachment angle of attack occurs on the downstroke of the airfoil motion.

Figure 4.27: Suction side pressure distribution at $Re = 300,000$ and $k = 0.050$ for $St_e = 0$ (*baseline*) (left) and $St_e = 9.9$ (right). At $St_e = 9.9$, the flow remains attached and no stalled stage is observed.

3. Excitation Effects on Aerodynamic Hysteresis

The third and final trend is that excitation results in decreased lift and moment hysteresis while decreasing the strength of the dynamic stall vortex. The first part of this trend is related to the generation of structures in the stalled regime of dynamic stall and earlier flow reattachment. The growth of the excitation perturbations into large-scale structures reattach the flow and result in increased lift production, thus reducing the lift hysteresis. These structures entrain high momentum freestream flow into the separated region. At

high excitation Strouhal numbers, when the vortices are small and the induced unsteady loading due to the convection and shedding of vertical structures is small, the moment hysteresis is significantly reduced as shown in Figure 4.15. Note that at small excitation Strouhal numbers, the hysteresis is larger due to the formation of large structures that only momentarily reattach the flow.

An important measure of the aerodynamic hysteresis is the damping coefficient. In particular, regions of negative damping (such as the stalled regime of dynamic stall) can lead to aerodynamic flutter [5]. Negative damping is associated with clockwise loops in the moment coefficient curve [5], as shown in Figure 4.28. The damping coefficient is defined as the closed loop integral of the moment coefficient curve. However, to understand the effect of the actuation on the negative damping, the integration is only performed over the closed loop of the clockwise trajectory of the moment coefficient and is referred to as the negative damping coefficient.

Shown in Figure 4.29 is the negative damping coefficient versus the excitation Strouhal number for $Re = 300,000$ and the three reduced frequencies. Dashed lines indicate the baseline value for the negative damping coefficient. In the figures it is observed that low excitation Strouhal numbers result in worse negative damping coefficient for low reduced frequencies. As indicated by Figure 4.26, low Strouhal number excitation results in the formation of large structures that result in large amplitude unsteadiness on the pitch down of the airfoil motion, hereby increasing the magnitude of the negative damping coefficient. However, in general, as the excitation Strouhal number increases, the magnitude of the negative damping coefficient reduces.

This is attributed to the constant partial reattachment seen in high Strouhal number excitation as well as the decreased strength of the dynamic stall vortex (as shown in Figure 4.11). As the strength of the dynamic stall vortex decreases, the magnitude of the moment peak decreases [55]. This also results in the reduction of the magnitude of the negative damping coefficient (which is best observed in Figure 4.29).

Figure 4.28: Negative damping (red) and positive damping (green) shown on moment coefficient curve for $Re = 300,000$, $k = 0.05$, and $St_e = 0$. As indicated, a counterclockwise trajectory results in positive damping, whereas a clockwise trajectory results in negative damping.

Figure 4.29: Negative damping coefficient, Ξ_- , versus the excitation Strouhal number, St_e for $Re = 300,000$ and various reduced frequencies. The lighter, dashed lines indicate the baseline value for the respective cases.

Phase-locked PIV results were used to understand the causality of the decreased strength of the dynamic stall vortex. Shown in Figure 4.30 and

Figure 4.31 is the normalized swirling strength for various phases at $Re = 300,000$, and $k = 0.05$ for the pitch up and pitch down motion, respectively. The figures are centered around the maximum angle of attack (except for the last row in Figure 4.31). The images are ordered by the excitation Strouhal number column-wise – that is, the left column contains the data for $St_e = 0$, the middle column contains the data for $St_e = 0.35$, and the right column contains the data for $St_e = 9.9$. Figure 4.32 shows when the PIV data

was acquired with respects to the lift and moment coefficients. It provides a good reference of how the individual snapshots relate to the dynamic stall process.

Shown in Figure 4.33 is the suction side pressure coefficient over the airfoil as a function of the airfoil motion for $Re = 300,000$, $k = 0.05$ and $St_e = 0, 0.35, \text{ and } 9.9$. This is particularly useful in understanding convection of the dynamic stall vortex. As any vortex propagates over the surface of the airfoil, it introduces pressure spikes [51]. In the figure, darker streaks indicate the convection of a vortex. For clarity Figure 4.33 is presented again with the vortices marked by dashed lines as Figure 4.34. The dynamic stall vortex is marked in green, whereas vortices due to excitation are marked in purple. In particular, it is easy to observe the formation of large structures that occur at $St_e = 0.35$ in the stalled regime of dynamic stall (as discussed previously – page 74).

As discussed previously, the ejection of the dynamic stall vortex is preceded by the accumulation of vorticity at the leading edge. This is shown by the high swirling strength in Figure 4.30 at $St_e = 0$, and $\phi = 44.5^\circ, 53^\circ, \text{ and } 64.2^\circ$ ($\alpha = 17^\circ, 18^\circ, \text{ and } 19^\circ$). At $\phi = 73.5^\circ$ ($\alpha = 19.6^\circ$), it is seen that the dynamic stall vortex has been ejected and is convecting downstream. Due to the unsteady nature of dynamic stall and the error in the motion repeatability, the vortex is not well defined. However coupled with pressure data (Figure 4.32 to Figure 4.34), it is apparent that the vortex is convecting over the airfoil at this time. When comparing the baseline data with the excited cases, it is observed that the accumulation of vorticity is much smaller in the excited cases, particularly at the high excitation Strouhal number. At $St_e = 9.9$, this accumulation of vorticity is replaced with a stream of small, coherent vortices. These vortices appear to

remove some of the accumulated vorticity. Thus as the number of vortices increases (or as St_e increases) more vorticity is removed from the leading edge and the dynamic stall vortex becomes weaker. The relationship between the dynamic stall vortex strength (indicated by the magnitude of the lift peak) and excitation Strouhal number is best illustrated in Figure 4.11.

To an extent, this is also observed by the phase-locked PIV data in Figure 4.30 at $St_e = 0.35$ and $\phi = 73.5^\circ$ ($\alpha = 19.6^\circ$), where the swirling strength of the dynamic stall vortex is considerably weaker than its baseline counterpart. But at the highest excitation Strouhal number, $St_e = 9.9$, the formation and ejection of the dynamic stall vortex is not observed in Figure 4.30 (pitch up motion) and Figure 4.31 (pitch down motion). Although it is possible that the dynamic stall vortex shed between the captured phases, pressure data (Figure 4.16, Figure 4.33, and Figure 4.34) suggests that this is not the case. Instead, the high Strouhal number excitation seems to have depleted the leading edge vorticity completely, suppressing the dynamic stall vortex. Time resolved PIV or higher phase resolution phase-locked PIV data could be used to further support this.

Figure 4.30: Phase-averaged normalized swirling strength at $Re = 300,000$, and $k = 0.050$. The data is organized by the excitation Strouhal number, column wise. The excitation Strouhal numbers depicted are: $St_e = 0$ (left), $St_e = 0.35$ (middle), $St_e = 9.9$ (right). Angle of attack of the airfoil is displayed on left of plots, along with the phase. The up arrows indicate that the airfoil is pitching up.

Figure 4.31: Phase-averaged normalized swirling strength at $Re = 300,000$, and $k = 0.050$. The data is organized by the excitation Strouhal number, column wise. The excitation Strouhal numbers depicted are: $St_e = 0$ (left), $St_e = 0.35$ (middle), $St_e = 9.9$ (right). Angle of attack of the airfoil is displayed on left of plots, along with the phase. The down arrows indicate that the airfoil is pitching down.

Figure 4.32: PIV phases overlaid with lift and moment coefficient curves. This figure depicts what phases of the flow were captured using PIV with respect to the pressure data. The pressure data shown is for $Re = 300,000$, $k = 0.05$, and $St_e = 0$.

Figure 4.33: Suction side pressure coefficient results for $Re = 300,000$, $k = 0.05$, and various excitation Strouhal numbers. The arrows on the y-axis indicate whether the airfoil is pitching up or down at that instant. This figure provides a time history of the pressure distribution on the suction side of the airfoil. Red streaks indicate the convection of a vortex.

Figure 4.34: This figure is very similar to Figure 4.33, but the vortices have been marked by green (dynamic stall vortex) and purple (vortices due to excitation) dashed lines.

Closer examination of Figure 4.11, indicates that there is fairly monotonic relationship between the excitation Strouhal number and the lift peak. This is consistent with the prior discussion of the formation of vortices that the formation of structures due to excitation removes accumulated vorticity from the leading edge, thereby weakening the dynamic stall vortex and thus reducing the magnitude of the lift and moment peaks.

Due to the poor temporal resolution in the phase-locked PIV data, it is difficult to directly determine the strength of the dynamic stall vortex (as there is not a phase where the vortex is well defined). As a simple alternative, the reduction in the moment coefficient due to excitation normalized by the baseline moment coefficient is shown in Figure 4.35. The figure indicates that in general, as the excitation Strouhal number

increases, the magnitude of the peak moment coefficient decreases significantly. This continues until the formation of the dynamic stall vortex is suppressed and the moment coefficient plateaus with respect to the excitation Strouhal number. However, a more direct metric could be obtained using higher phase resolution phase-locked PIV data in which the dynamic stall vortex and its boundaries are well defined.

Figure 4.35: Reduction in moment coefficient peak versus excitation Strouhal number at $Re = 167,000$ and $k = 0.05$. The change in the moment coefficient is used as a metric for the reduction in strength of the dynamic stall vortex.

5. Conclusion and Future Work

NS-DBD plasma actuators rely on thermal perturbations that manifest as compression waves to excite natural flow instabilities. Static tests of these actuators demonstrate their efficacy to reattach separated flow for a variety of conditions [13, 14, 15, 11]. As such, these actuators could potentially be used to replace leading edge slats and trailing edge flaps to augment lift during take-off and landing of fixed wing aircraft. This represents major savings in weight and mechanical complexity of the aircraft wings. It is desired to expand upon the current work with these actuators into unsteady flow and dynamic stall. Dynamic stall is present in many applications, such rotorcraft and wind turbines.

As a precursor to experimentation, a dynamic stall facility was first designed and built. The design consisted of a single servo that drives two windows attached to the NACA 0015 airfoil via belts and an intermediary shaft. A custom controls system was developed to minimize cycle to cycle variations of the motion. Although the cycle to cycle variation can vary by 1° , the cycle averaged motion profile is very similar to the desired motion. The overall temporal synchronization of the motion and DAQ systems was 5 ms. Due to long pressure tubing lengths, experimental gain and lag curves were used to correct the pressure data.

After the facility was constructed, baseline data was taken and validated against a study whose flow parameters could be replicated. The validation indicated that our results had good run-to-run repeatability, and was consistent with the results in the literature. There were a few differences, including the presence of the lift peak, which is attributed to blockage effects [48] and the increased lift hysteresis, due to the sharp stall of the airfoil used (Figure 3.14).

A NS-DBD plasma actuator was then mounted to the airfoil, and baseline pressure for $\alpha_0 = 10^\circ$ and $\Delta\alpha = 10^\circ$ was taken for three Reynolds numbers and three reduced frequencies. Pressure data was acquired for all nine combinations of Reynolds number and reduced frequency. Due to time constraints, phase-locked PIV measurements were only acquired at $Re = 300,000$ and $k = 0.05$. The baseline pressure and PIV results illustrated the regimes of dynamic stall: attached flow, the accumulation of vorticity at the leading edge, the ejection of the accumulated vorticity in the form of the dynamic stall vortex, stalled flow, and reattachment. PIV results indicated that the ejection of the dynamic stall vortex was due vortex induced separation – a counterclockwise rotating stream of vorticity moves upstream during the pitch up motion and result in the ejection of the dynamic stall vortex.

Although a total of 9 combinations of Reynolds number and reduced frequency were tested in the excited experiments, each combination was remarkably similar. From these results, three major conclusions were drawn. The first was that excitation at low Strouhal number results in the formation of flow structures during the stalled regime of dynamic stall, which result in unsteadiness in the lift and moment curves. As the

excitation Strouhal number increases, this unsteadiness disappears. As observed in static flow control experiments with NS-DBD plasma actuators, low Strouhal number excitation results in the formation of large coherent structures, whereas high Strouhal number excitation results in the formation of small structures that quickly develop, breakdown, dissipate and result in only partial reattachment of the flow. Thus, high Strouhal number excitation results in smoother lift and moment coefficient curves.

Secondly, excitation also results in earlier reattachment due to the formation of the structures that momentarily reattach the flow. Once the angle of attack of the airfoil decreases past the static stall angle, the flow remains attached.

Lastly, excitation results in the reduction of the lift and moment hysteresis and is reflected in the increased damping coefficient. This is partially due the reattachment of the flow in the stalled regime. The other major factor is the decrease dynamic stall vortex strength which subsequently results in the decreased magnitudes of the lift coefficient and moment coefficient. The decreased strength of the dynamic stall vortex is attributed to the decreased vorticity accumulation. Excitation results in the formation of vortices prior to the ejection of the dynamic stall vortex. These vortices take some of the accumulated vorticity. At high excitation Strouhal numbers, this results in the suppression of the dynamic stall vortex.

Although experimental results clearly demonstrates the efficacy of the actuators, the sensor noise and uncertainty in the repeatability of the airfoil motion suggests that a new mechanism needs to be designed prior to acquiring more data. As such, immediate future work will be concentrated on improving the repeatability of the airfoil motion,

while designing a new airfoil with a shorter chord to reduce blockage effects. Higher phase resolution in the phase-locked PIV will provide higher data fidelity and may reveal a well-defined relationship between the excitation Strouhal number and the dynamic stall vortex strength. Phase-locked PIV data collected at other combinations of Reynolds numbers and reduced frequencies will better support the generalization of trends presented in this paper.

Long term goals include studying the effects of varying the excitation Strouhal number throughout the pitching cycle, and examining the three dimensionality of dynamic stall.

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