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Nanosecond dielectric barrier discharge plasma actuators flow control of compressible dynamic stall

ABSTRACT

See the thesis

**Nanosecond Dielectric Barrier Discharge Plasma Actuator Flow Control of
Compressible Dynamic Stall**

THESIS

Presented in Partial Fulfillment of the Requirements for the Degree Master of Science in
the Graduate School of The Ohio State University

By

Matthew William Frankhouser

Graduate Program in Aeronautical and Astronautical Engineering

The Ohio State University

2015

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Abstract

Dynamic stall is a performance-limiting phenomenon experienced by rotorcraft in directional and maneuvering flight. Dynamic stall occurs on the retreating blade due to the high angles of attack that are experienced by the blades. Increasing the angle of attack is required to overcome the asymmetry of lift across the rotor disk that is a result from the velocity disparities between the advancing and retreating blade. This work sets out to study and improve the performance of a dynamically pitching NACA 0015 airfoil. The airfoil is subjected to both an incompressible and compressible flow field to simulate the dynamics of a rotor blade with cyclic pitching. In this experimental investigation of dynamic stall flow control, the effectiveness of nanosecond dielectric barrier discharge (NS-DBD) plasma actuation will be evaluated as a means to exert control authority. The NS-DBD plasma actuation is generated by a high-voltage magnetic compression pulsed power supply that was designed and built at The Ohio State University. To measure the influence of plasma actuation on the flow, surface pressures on the airfoil were measured through discrete pressure taps located on both the suction and pressure surfaces. The surface pressures are used to calculate the lift and moment during the dynamic pitching cycle. To visualize the compressibility effects in the outer flow, shadowgraph imagery was used to capture features in the flow around the leading edge of the test article.

Tests were conducted at static and oscillating angles of attack at both Mach 0.2 and 0.4, and Reynolds numbers of 1.2 million and 2.2 million respectively. Pitch

oscillations were conducted at reduced frequencies of 0.05 ($k = 0.05$). Actuation frequencies varied from non-dimensional frequencies (F^+) of 0.78 to 6.09. Surface pressures acquired at Mach 0.2 without actuation applied agreed with historical data at static angles of attack, validating that the application of the actuator had limited intrusiveness to the flow. When subjected to pitch oscillations, plasma actuation reduced the severity of lift and moment stall by altering the development of the dynamic stall vortex at Mach 0.2.

At Mach 0.4, marginal improvements were gained through actuation. Excitation resulted in a strong dynamic stall vortex that convected slower in comparison to the baseline case. Shadowgraph imagery revealed lambda shock waves forming over the first 15 percent of the airfoil chord, in the same proximity of the actuator. The shocks can lead to separation and diminished control authority.

Dedication

This document is dedicated to my wife, family, and friends.

Acknowledgments

It brings me great pleasure to express my gratitude and appreciation to the numerous people who have supported me during my tenure at The Ohio State University. First and foremost I would like to thank my family and friends for the steadfast support during my studies and for the encouragement to keep me inspired.

It has been a pleasure to work under the advisements and mentorship of Dr. James Gregory, whom without, this work would not have been possible. A special thanks is owed to the research group at the Aeronautical Research Center who supported and assisted with this work. Special thanks to Dr. Jeffery Bons for the guidance and discussions during this research. I would also like to acknowledge Dr. Bryan Glaz and Dr. Matthew Munson from the Army Research Office for the grant that allowed this study to be conducted.

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- Hird, K., Frankhouser, M. W., Gregory T. W., and Bons, J. P., "Compressible Dynamic Stall of an SSC-A09 Airfoil Subjected to Coupled Pitch and Freestream Mach Oscillations," in 70th American Helicopter Society Annual Forum & Technology Display, Montréal, 2014
- Hird, K., Frankhouser, M. W., Naigle, S., Gregory T. W., and Bons, J. P., "Study of an SSC-A09 Airfoil in Compressible Dynamic Stall with Freestream Mach Oscillations," in 71st American Helicopter Society Annual Forum and Technology Display, Virginia Beach, 2015
- Naigle, S., Frankhouser, M. W., Hird, K., Gregory T. W., and Bons, J. P., "Blowing Flow Control of Dynamic Stall under Coupled Pitch and Freestream Oscillations," in 71st American Helicopter Society Annual Forum and Technology Display, Virginia Beach, 2015
- Frankhouser, M. W., Hird, K., Naigle, S., Gregory T. W., and Bons, J. P., "Nanosecond Dielectric Barrier Discharge Plasma Actuator Flow Control of Compressible Dynamic Stall," in AIAA 45th Fluid Dynamics Conference/46th AIAA Plasmadynamics and Laser Conference, Dallas, 2015

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Nomenclature

A	= Test section area
A^*	= Throat area
α	= Angle of attack
$\Delta\alpha$	= Change in angle of attack
α_0	= Mean angle of attack
α_Δ	= Angle of attack amplitude
$AC\text{-}DBD$	= Alternating current – dielectric barrier discharge
c	= Chord length
C_M	= Moment coefficient
C_L	= Lift coefficient
C_P	= Coefficient of pressure
ΔC_{Pmax}	= Change in peak C_P
DSV	= Dynamic Stall Vortex
F^+	= Reduced pitching frequency based on model chord, fc/U_∞
f	= Actuation repetition rate
γ	= Ratio of specific heat
Γ	= Lift Excursion
k	= Reduced frequency, $\omega c/2U_\infty$
M	= Mach number
$NS\text{-}DBD$	= nanosecond – dielectric barrier discharge
ω	= Pitching frequency

p	= pressure
ϕ	= Phase angle
$\Delta\phi$	= Change in phase angle
Ψ	= Azimuthal angle
x	= Chordwise position
ζ	= Moment Excursion

Subscripts

∞	= Free stream
F	= Forcing or actuation on
T	= Total

Chapter 1: Introduction

Dynamic stall is a well-known helicopter performance limiting phenomenon that usually occurs on the rotor blades at high forward flight speeds or during maneuvers with high load factors. Some of the detrimental consequences associated with dynamic stall are the onset of large torsional air loads and high amplitude vibrations on both the retreating and advancing blades either due to cyclic pitching or shock induced flow separation respectively.

Thus, the operational envelope of a rotorcraft is directly related to the aerodynamic performance of the rotor blades. This thesis sets out to take a dynamically pitching airfoil and implement active flow control to improve the aerodynamic performance. The active flow control method evaluated in this project is nanosecond dielectric barrier discharge (NS-DBD) plasma actuation. Plasma actuation has advantages over other flow control mechanisms which are desirable for rotorcraft systems including surface mounted actuators which do not alter the blade geometry and can be retroactively applied to current systems with only simple electrical wiring to negotiate the rotor blade. In theory, the connection to the rotor system is made via a simple rotation contact plate that can deliver the required voltage from an onboard power supply to the rotor system. The simple design of the actuators makes them highly desirable due to their light weight easy of application, maintenance, as well as lack of mechanical components.

NS-DBD actuators are a form of plasma actuation which allow for flow control to scale from low Mach and Reynolds numbers to high Mach and Reynolds numbers. The

use of NS-DBD actuators on a dynamically pitching airfoil in a compressible flow is a novel concept that, to the knowledge of the author, has yet to be fully analyzed.

Section 1.1: Helicopters in Forward Flight

For a traditional helicopter (a helicopter with a single main rotor and tail rotor), the main rotor blades experience a difference in blade-relative velocities between the advancing and retreating sides of the disk during forward flight. A schematic of a traditional helicopter in forward flight is illustrated in Figure 1. The lift generated by an individual blade is proportional to the square of the velocity

$$L = \frac{1}{2}\rho S C_L v^2$$

The discrepancy in relative velocity from the advancing blade to the retreating blade results in an asymmetric lift distribution about the rotor disk. To balance the lift across the rotor disk, cyclic blade pitching is enacted by the hub linkages. Cyclic pitching imposes a sinusoidal angle of attack oscillation of the blade in a manner that results in the lowest angle of attack on the advancing blade and the highest angle of attack on the retreating blade. At relatively high airspeeds the advancing and retreating blade begin to operate close to the limits where flow cannot easily remain attached to the rotor blade surface. The advancing blade, operating at low angles of attack and high airspeeds, is near the region where shock induced flow separation occurs. In contrast, the retreating blade is operating at high angles of attack and low airspeeds in the region where critical

angle induced stall occurs. For a traditional helicopter, the maximum forward flight speed is limited by this highly time-dependent, dynamic stall phenomenon.

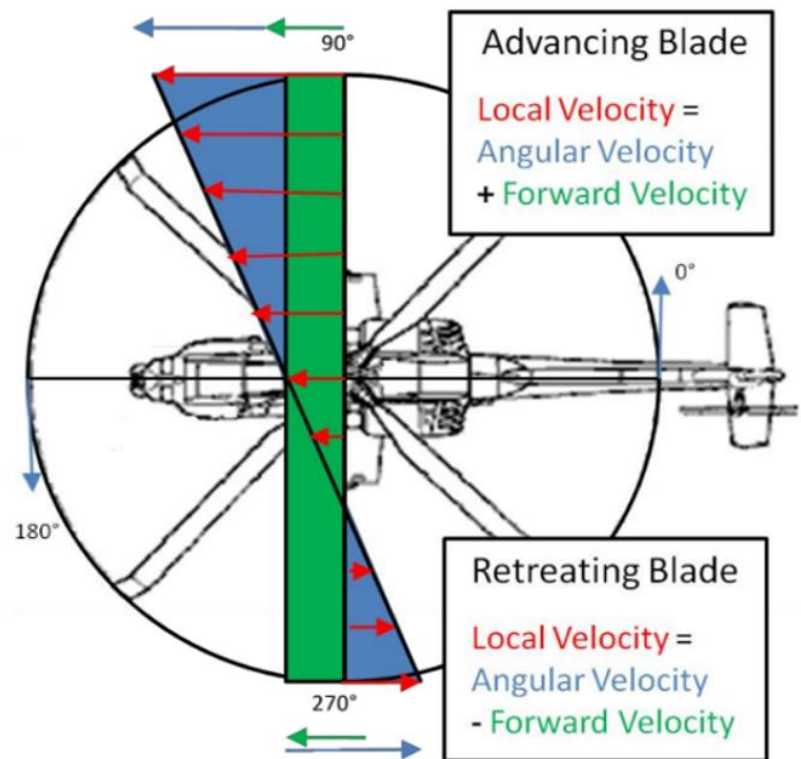


Figure 1: Schematic of a helicopter in forward flight illustrating blade-relative velocity at various azimuthal positions¹.

Dynamic stall on rotorcraft blades is a direct result of the cyclic pitching to increase lift on the retreating blade. The process of dynamic stall occurs when a certain

critical angle of attack is reached. As the blade pitches to higher angles of attack, the flow breaks down in a process known as stalling. For a rotorcraft blade experiencing rapid angle of attack oscillations, the onset of stall can be delayed to angles of attack beyond the static stall angle. Once dynamic stall has occurred, it is often more severe compared to static stall. As the angle of attack increases, a separation bubble forms near the leading edge. Circulation begins to build as the shear layer near the leading edge rolls up and forms the leading-edge vortex (LEV). As the LEV convects downstream, the increased suction associated with the vortex core results in lift augmentation and stall delay. During its passage over the airfoil, a sudden increase in nose-down pitching moment occurs as the LEV propagates downstream past the aerodynamic-center resulting in moment stall. After moment stall the LEV continues to convect and eventually becomes unstable and detaches from the surface resulting in a dramatic decrease in lift.

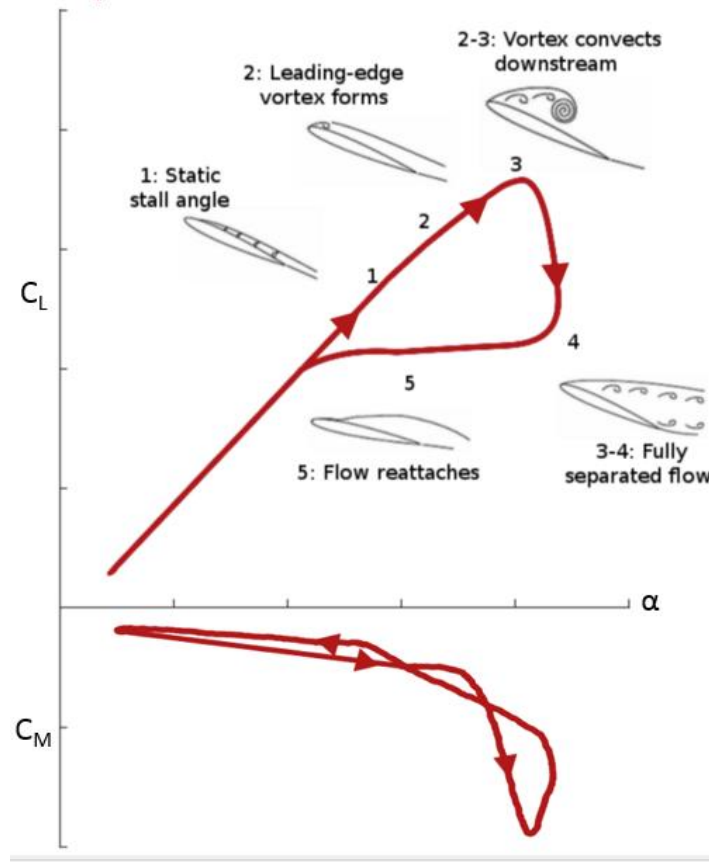


Figure 2: Dynamic stall process, Insets: Yu *et al*².

The unsteady aerodynamic effects due to dynamic stall have a major impact on the maneuverability and control of the rotorcraft by limiting performance and maximum forward flight speed. The process of dynamic stall vortex shedding and downstream convection induces lift and moment stall that can produce significant fluctuations in the pitching moment. The high pitch link loads that result from moment stall require structurally-robust pitch links and swashplates, which increase aircraft weight and limit the operational envelope of the rotorcraft.

Due to dynamic stall limitations, attempts have been made to mitigate and prevent dynamic stall. These dynamic stall preventative methods come in the form of passive flow control and active flow control techniques. Some passive techniques include modification to the leading edge of the airfoil, and vortex generators applied near the leading edge. Active flow control techniques include Vortex Generator Jets (VGJ's), tangential and normal blowing, synthetic jets, and plasma actuation. Besides flow control techniques, new forms of helicopters have been tested that implement coaxial contra-rotating blades. Use of coaxial contra-rotating blades alleviates the need for cyclic pitching; the rotor blade angle of attack can remain fixed throughout each revolution. This is possible because the asymmetric lift on one rotor disk is balanced by the asymmetric lift on the second counter rotating disk. Although dynamic stall can be bypassed by using coaxial rotor systems, the helicopters performance remains limited by shock-induced stall.

Section 1.2: Fluid Dynamics and Experimental Background

Numerous investigations have been performed to analyze the flow physics of dynamic stall; relevant studies of dynamic stall stretch over four decades. These studies have provided insight into the basic phenomenon of dynamic stall under static freestream conditions and are thoroughly documented³⁻⁹. Typically these studies have been conducted at low-speeds, within the incompressible flow regime¹⁰⁻¹². The primary focus

of these investigations was on the development and convection of the leading edge vortex under pitch oscillations or freestream oscillations.

In particular, airfoils oscillating in a steady freestream have been given considerable attention as the performance of rotorcraft has increased. A common proxy for fundamental aerodynamic studies of dynamic stall consist of an airfoil either pitching or plunging at rotorcraft-relevant reduced frequencies (k ranging from 0.05 to 0.2). Such experiments are designed to capture the flow changes accompanying the dynamic blade pitching. Various pitch schedules have been prescribed and studied, including a simplified ramp motion in which pitch increases linearly with time, as well as sinusoidal variation at relevant frequencies and amplitudes^{13,14}. These investigations determined that a pitching airfoil encounters varying temporal and spatial pressure gradients associated with unsteady flow separation.

Chandrasekhara *et al.*¹⁵ conducted an investigation in which schlieren photography was used to visualize the flow around a NACA 0012 airfoil undergoing dynamic pitching. From this study, it was concluded that dynamic stall delay was intensified by increasing the rate of pitching. Through the schlieren photographs, it was evident that increasing the pitch rate resulted in the vortex remaining attached to the surface longer during convection.

Periodic excitation is one form of active flow control that has been used to modify the flow field behavior in separated flows – this can take the form of blowing jets or plasma actuation^{16,17}. Often this is accomplished by positioning such a flow control device near the separation location or leading edge on the surface of the airfoil. With the

flow control mechanism located near the leading edge, the excitation can manipulate natural flow instabilities with maximum receptivity to prevent or delay flow separation.

Zero-net-mass-flux oscillatory jets are one from of active flow control that has proven to be effective. Seifert *et al.*¹⁸ implemented such jets via blowing slots located at 10% chord, with almost tangential streamwise excitation. Experiments were conducted at Mach 0.28 and 0.4 at a Reynolds number based on chord of 12.7×10^6 . The experiments demonstrated that under incompressible flow conditions, the maximum coefficient of lift could be increased by 15%, and post-stall lift could be increased by as much as 50% along with a reduction in post-stall drag of up to 50%. At the higher Mach numbers, when excitation was applied upstream of the shock wave, detrimental effects on both lift and drag were observed. The reduction in performance with excitation in front of the naturally-occurring shock was believed to be from localized disturbances created at the blowing slot. When excitation was introduced immediately upstream of the shocks, an increase of the lift to drag ratio was observed. It was concluded that when control excitation was applied close enough to the separation location beneficial results were obtained due to the receptivity of the flow.

In 2014, Beahan *et al.*¹⁹ conducted a study on a pitching NACA 0015 airfoil by implementing 400 microjets from 0% to 10% of the chord. The airfoil was subjected to freestream velocities of Mach 0.3 and 0.4 with Reynolds numbers of 1.03 and 1.40 million respectively. The airfoil was pitched at reduced frequencies of 0.05 and 0.10 with a mean angle of attack of 10° and an amplitude of 10° . Interferometry was used to visualize the flow around the leading edge at various angles of attack during the pitch

cycle. The use of these microjets was able to delay the shedding of the dynamic stall vortex, and thus delaying the onset of dynamic stall. In the experiments at Mach 0.4, the microjets exhibited the ability to prevent naturally occurring shocks thus eliminating shock induced separation.

Section 1.3: Plasma Actuator Flow Control

Flow control via plasma actuator excitation has grown in popularity in recent years. Plasma actuators are highly desirable forms of flow control because they are surface mounted with no mechanical components making them lightweight and easy to maintain. Current plasma actuators can be classified into two types: alternating-current dielectric barrier discharge (AC-DBD) and nanosecond dielectric barrier discharge (NS-DBD). The primary difference between AC-DBD and NS-DBD plasma actuators is that NS-DBD actuators transfer very little momentum to the surrounding air. Instead, control of the flow is achieved through thermal effects, where compression waves add energy to the flow^{20,21}.

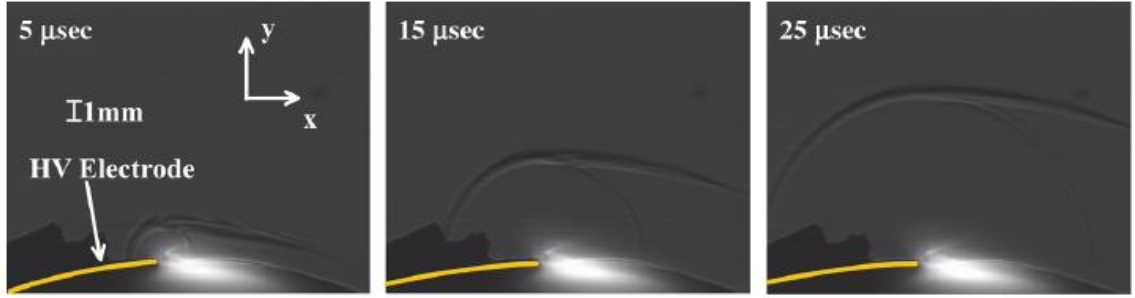


Figure 3: Compression waves generated by NS-DBD actuation. From Little *et al.*²².

Post *et al.*²³ implemented AC-DBD plasma actuators on a NACA 0015 airfoil at 0% chord. The airfoil was subjected to a freestream of 10 m/s and was pitched at a reduced frequency of 0.08. The airfoil was pitched sinusoidally with a mean angle of attack of 10° and an amplitude of 15° . The actuators were pulsed at frequencies of 20 Hz and 80 Hz utilizing open-loop steady, open-loop unsteady, and closed-loop control. From their study, it was discovered that actuation at the leading edge improved the lift cycle. The steady plasma actuation resulted in increased lift over most of the cycle except at the peak angle of attack. Actuation suppressed formation of the dynamic stall vortex, while unsteady actuation produced significant improvements in lift during the downstroke of the pitch cycle. It was also discovered that the optimum forcing frequency to delay moment stall and reduce the negative peak moment was a dimensionless frequency (F^+) of unity.

The use of NS-DBD actuators on airfoils to date has largely been performed under steady flow conditions. Little *et al.*²² demonstrated the ability for NS-DBD actuators to achieve flow control on a NASA EET airfoil. The actuator was placed near the leading edge at approximately 0% chord. The airfoil was subjected to an

incompressible flow with Reynolds numbers up to 1×10^6 . The high-voltage side of the actuator was supplied with 16 kV and a repetitive nanosecond pulse of 60 ns. Under excitation, particle image velocimetry (PIV) results indicated that the shear layer is less intense, and the vorticity in the flow is closer to the suction surface. This demonstrates the NS-DBD leading-edge actuation was able to force the flow to attach to the suction surface in an otherwise separated flow when actuation is not applied.

NS-DBD actuators were also applied to a NACA 0015 airfoil by Rethmel *et al.*²⁴. The actuator was positioned near the leading-edge at approximately 1% chord. The airfoil was subjected to freestream speeds up to Mach 0.26 and Reynolds number up to 1.16×10^6 . Various excitation frequencies were tested to determine the optimum forcing F^+ . From the study, it was discovered that localized heating from the actuation formed a compression wave that propagated into the flow with each pulse. AC-DBD and NS-DBD actuator configurations were tested to aid direct comparison of their effects. It was found that the effectiveness of the AC-DBD actuators decreased as Reynolds number was raised, while the NS-DBD actuators demonstrated similar control authority across the corresponding Reynolds number range. During nanosecond-pulsed excitation, the magnitude of the suction pressure on the upper surface of the airfoil was increased above the baseline case over the first 50% of the chord resulting in an increase in lift at static angles of attack for both pre-stall and post-stall.

At pre-stall angles of attack, the NS-DBD actuators act as an active trip; while near stall and in post-stall the actuator manipulates flow instabilities^{22,24}. While the control authority of NS-DBD actuation has been demonstrated in low Reynolds number and low

pressure environments, experimental investigations of NS-DBD actuation on a dynamically pitching airfoil at high Mach and Reynolds numbers are limited.

The purpose of this investigation was to perform a study of the effects and control authority of NS-DBD plasma actuators under compressible dynamic stall conditions. The Ohio State 6"x22" unsteady transonic tunnel was used to study compressible dynamic stall flow control with NS-DBD actuation under steady freestream conditions and pitching airfoil oscillations.

Chapter 2: Experimental Methodology

This experimental investigation was conducted in a transonic dynamic stall facility capable of both time-varying and steady conditions. Presented data were generated from pressure measurements integrated over the suction and pressure surfaces over the 5 inch chord of a NACA 0015 airfoil. NS-DBD generated plasma occurred at 4% chord and was powered and controlled via an in-house design which is capable of delivering 10 kV to the 6 inch span-wise mounted dielectric strips. The contents of Chapter 2 describe the facility and experimental setup in detail. Each section describes the various components and how each device is implemented into the system. To obtain highly time-accurate and synchronized data, each device must be precisely incorporated in the data acquisition system. This chapter outlines how the devices are implemented into the system to fully depict the complexity of the setup.

Section 2.1: Experimental Facility

The dynamic stall experiments were conducted in a transonic wind tunnel located in the Aerospace Research Center (ARC) at The Ohio State University. The facility was designed specifically for testing two-dimensional airfoils. The history of the transonic wind tunnel starts in 1976 and was pivotal for experiments for the development of

general aviation airfoil analysis and design. Aside from being used by major aircraft manufacturers and testing some of the more commonly used airfoil cross-sections, it has been used in dynamic stall investigations²⁵. These investigations include dynamically pitching of the airfoil while holding a constant freestream Mach number, and more recently, coupled pitch and Mach oscillations to simulate a rotorcraft in forward flight^{26,27,28}.

A schematic of the 6''×22'' unsteady transonic wind tunnel is depicted in Figure 4. The tunnel is a blowdown facility, with air supplied through a 20cm (8in) supply line from two 42.5 m³ (1500 ft³) air storage tanks. The storage tanks can be pressurized to a maximum 17 MPa (2500 psi) with in-line air dryers to maintain gas purity. The high pressure air flow to the transonic wind tunnel is controlled by two valves. The first valve in series is a fast-acting control valve that is used to start and stop the flow. The second valve in series is a control/positioning valve that is used to set the total pressure and Reynolds number in the test section of the wind tunnel. The positioning valve is controlled through LabVIEW software and is used to regulate the test section pressure below the maximum operating pressure of 350kPa (50psia).

The tunnel was designed as a low turbulence facility to primarily operate in the transonic regime. This is achieved through the settling/stagnation chamber. The stagnation chamber is designed to condition the flow to a test-section turbulence level of less than 0.5% of the freestream velocity under steady flow conditions. To accomplish this, the flow passes through a perforated plate, honeycomb, and eight screens of 60-mesh located in the settling chamber to condition the flow. The flow enters the test section by

passing through a subsonic nozzle with a contraction ratio of 15:1 which provides excellent flow uniformity. The test section is 6”×22” and is 1.1m long. Perforated spanwise ceiling and floor walls with 3.2 mm holes that yield an effective porosity of 6% are inserting inside the test section. These perforated plates create isolation cavities that reduce Mach wave reflections in transonic flow.

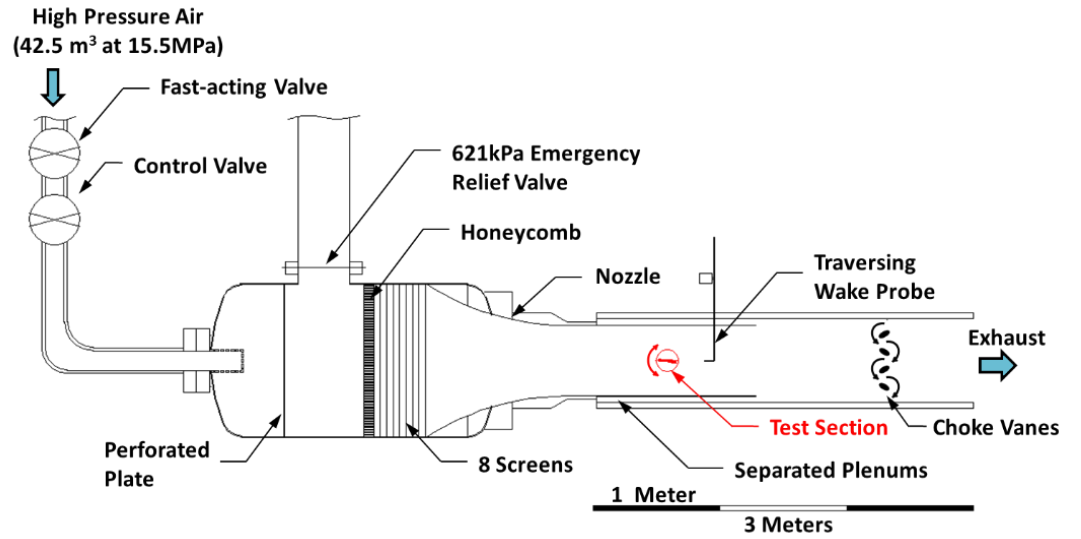


Figure 4: Schematic of OSU 6”x22” tunnel²⁹.

The tunnel was originally designed to operate at a time-invariant (steady) Mach number by adjusting the throat area (A^*) downstream of the test section.

$$\frac{A}{A^*} = \left(\frac{\gamma+1}{2}\right)^{-\frac{\gamma+1}{2(\gamma-1)}} \frac{\left(1 + \frac{\gamma-1}{2} M^2\right)^{\frac{\gamma+1}{2(\gamma-1)}}}{M}$$

This allows the Mach number to be uniquely set by the throat area independent of the stagnation pressure and Reynolds number. This is achieved by maintaining a choked flow

during the run. The throat area is established through the installation of choke bars of various diameter and can be arranged in various patterns. The Reynolds number is set independently from the Mach number through the positioning valve that sets the stagnation chamber pressure and controls the flow density. Use of the choke bars and positioning valve allow for the Mach number and Reynolds number to vary over a considerable range.

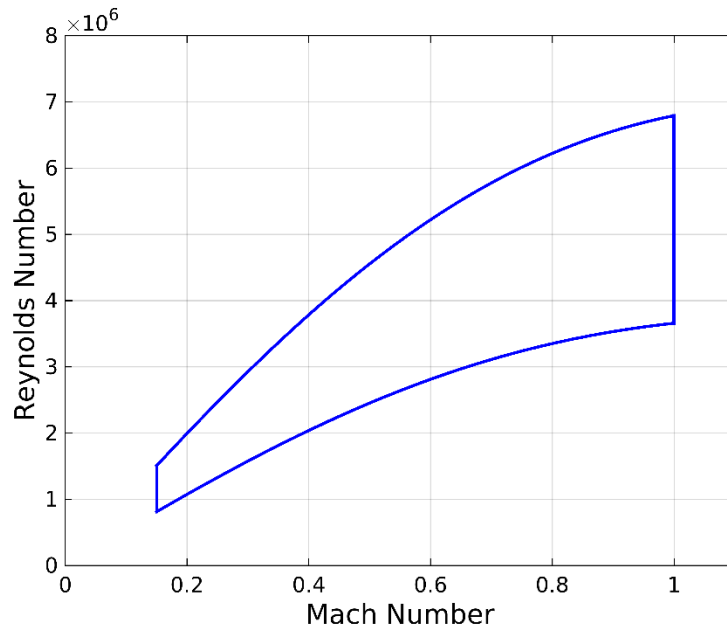


Figure 5: 6"x22" transonic tunnel operation limits for 5" characteristic length.

Modifications to the 6"x22" transonic tunnel enables operation in several dynamic modes. This allows the flow to simulate compressible dynamic stall flow fields. The first dynamic mode allows for the airfoil to be dynamically pitched throughout the run. An oscillation mechanism is powered by a 5-hp motor which drives a face cam and linkage

arm connected to the airfoil. The mechanism can be operated at frequencies up to 21 Hz with typical oscillation amplitudes of 5° and 10°. Figure 6 and Figure 7 illustrate the pitch oscillation assembly.

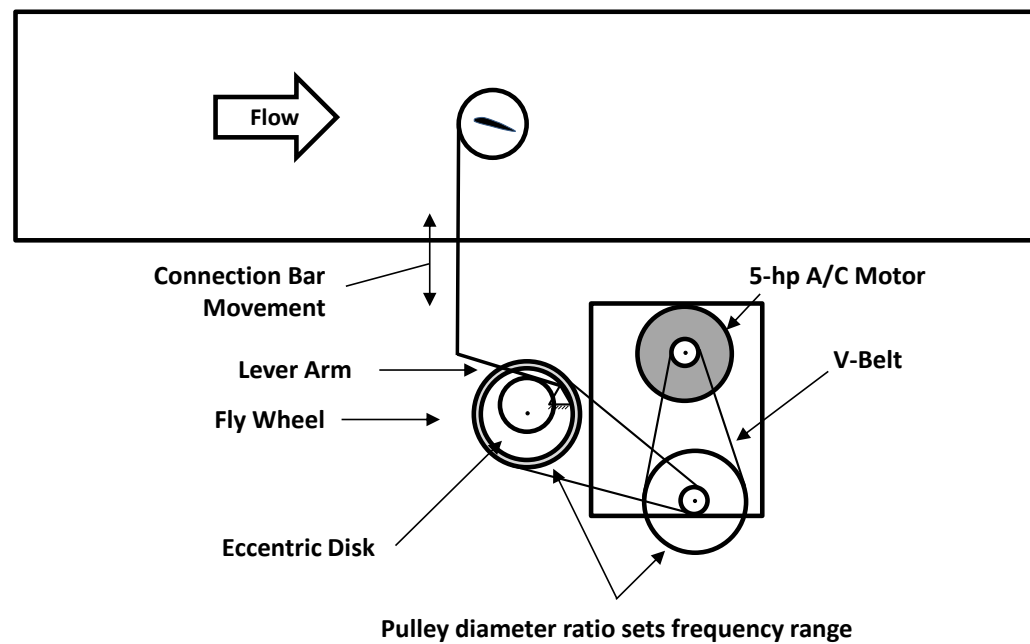


Figure 6: Airfoil oscillation assembly

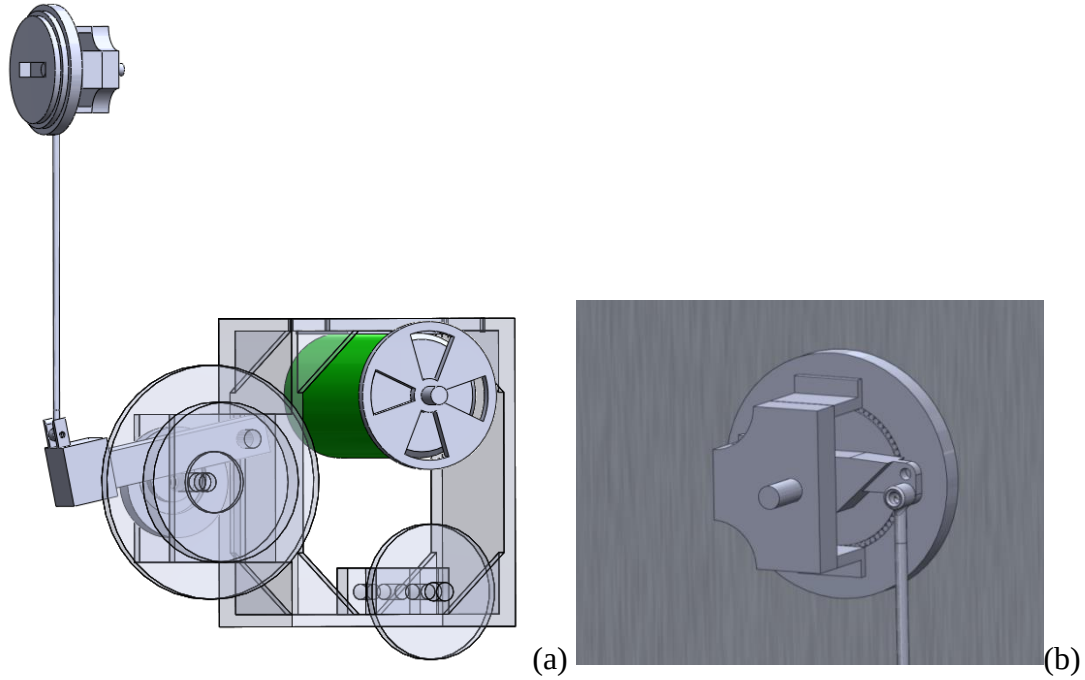


Figure 7: Detailed schematic of eccentric disk and oscillation mechanism (a), and thrust bearing for pitch oscillation (b).

The second dynamic mode allows for Mach oscillations within the test section. Mach oscillations are achieved through elliptical rotating choke vanes that are installed at the throat. The vanes are powered by a servo-motor that rotates the choke vanes through a drive chain up to 20 Hz in the test section.

The third dynamic mode allows for coupled pitch and Mach oscillations. Under this configuration, the airfoil can be pitched at various phase delays relative to the Mach number oscillations. The coupled oscillations are achieved by slaving the Mach oscillation servo-motor to an optical encoder connected to the shaft of the 5-hp motor on the pitch oscillation mechanism. Operation in coupled oscillation mode allows for a flow

field that has the time varying velocity and an airfoil with dynamic pitching motion to more accurately model a rotorcraft blade in forward flight.

For all three dynamic tunnel operations, for a typical airfoil used in experiments with a 152.5mm (6”) chord, the Mach number operation ranges from Mach 0.2 to 1.0. Reynolds number can range from 1-7 million. These ranges allow for experimental studies with relevant reduced frequencies for rotorcraft dynamics.

Section 2.2: Data Acquisition

Data Acquisition of time-dependent information during these dynamic transonic wind tunnel tests is critical to obtain accurate results. A series of three computers are utilized to control, synchronize, and record analog (temperature, angle of attack, etc.) and pressure data in the 6”x22” transonic facility.

The first computer is the main control system that operates the positioning valve. The position of the valve is set by supplying an analog signal from a terminal Data Acquisition Board (DAQ Board) to the electronic positioner that opens the valve to the correct setting to obtain the user defined pressure and Reynolds number throughout the run. This computer also sends a digital signal to the fast-acting pneumatic valve to open a path to the high pressure storage tanks which acts as the start and stop to the run. In addition, this computer monitors the stagnation chamber pressure in real time to allow for an emergency shutoff if a tunnel abnormality occurs. During the monitoring of the stagnation pressure, once the tunnel has passed through the transient of pressurization, the

DAQ Board outputs a trigger signal that is used by the remaining data acquisition computers to begin the recording process.

The second computer is purely a data acquisition computer that records the analog signals from the various devices installed on the tunnel. Two DAQ Boards are used to record signals, a National Instruments BNC-2120 with PCI-6251 card, and a National Instruments BNC-2121 Quadrature board with a PCI-6602 card that records the optical encoders and records data at 200 kHz. The BNC-2120, after receiving the trigger signal from the control computer records various instruments including wake position, pressure transducers, trigger signals, encoder once per rev signals, and can also output digital and analog signals for triggering and timing of various pieces of equipment. The BNC-2121 Quadrature board records three different optical encoders that are used during the different dynamic tunnel configurations. One encoder is used to measure the angle of attack of the airfoil during dynamic pitching, and is connected directly to the airfoil. The second encoder measures the phase position of the airfoil and is connected to the motor shaft of the pitching mechanism. The third encoder measures the angular position of the elliptical rotating choke vanes when operating in an unsteady freestream Mach number configuration (In this experiment unsteady freestream tunnel operation was not conducted).

The third computer is part of a stand-alone system that operates the ESP Pressure Scanner System. The ESP system allows for multiplexed sampling of 64 pressure signals at a rate of 1000 Hz. This is used to record airfoil surface pressures as well as total and static tunnel pressures from various locations within the tunnel.

Both the data acquisition computer system and ESP Pressure system computers are connected to a Quantum Composer Model 9514 Pulse Delay Generator that is used as an external clock. This ensures that both systems record data at a known and highly accurate rate of 200 kHz. Use of an external clock is crucial to obtaining the time-critical information during dynamic experiments to allow for proper alignment of all recorded signals.

Section 2.3: Airfoil Instrumentation

Airfoil surface pressure signals are recorded with pressure scanners and then integrated in post-processing to obtain accurate lift, moment, and pressure drag measurements. The pressure scanners used for this experiment are well suited for the dynamic flow features that will be present on the airfoil. The pressure scanners are capable of multiplexing at high rates, can be triggered externally, and the individual pressure sensors have thermal compensation to minimize zero and span shifts with temperature.

The surface pressure taps are connected to the ESP Pressure Scanner System to obtain surface static pressures along the airfoils perimeter. The pressure-coefficient is calculated at each tap location by means of the following equation.

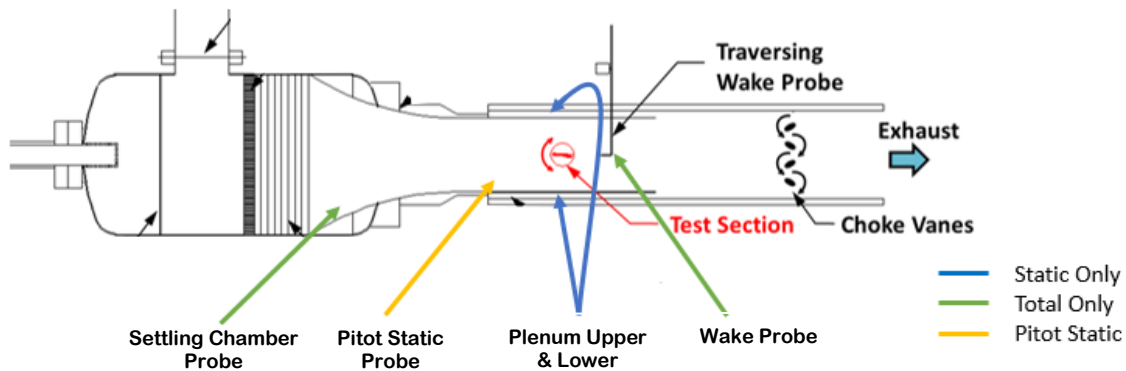
$$C_p = \frac{\frac{P}{P_\infty} - 1}{\frac{1}{2} \gamma M_\infty^2}$$

A trapezoidal rule is used to integrate between the discrete pressure tap locations to obtain the coefficient of pressure (C_p) curve, which is used to calculate the sectional lift (C_L) and moment (C_M) coefficients. It is important to note that C_L and C_M measurements do not include the contributions of skin friction forces³⁰.

Two ESP 32HD Pressure Scanners are mounted outside of the test section, in-line with the mid-chord of the airfoil, and the test article pressure taps are connected to the pressure scanner ports via plastic tubing of 1.4mm diameter (0.06in). The ESP 32HD scanners are miniature electronic differential pressure units consisting of an array of 32 silicon piezoresistive pressure sensors with a differential of +/-210kPa per scanner port. The electrical outputs from the pressure sensors are multiplexed through a single onboard instrumentation amplifier. The multiplexed amplified output signal is connected to a DTC Initium scanner interface and passed to the dedicated computer via Ethernet cable. A Quantum Composer Pulse Delay Generator which is triggered by the DAQ control computer is used as an external TTL pulse train to ensure a stable sample interval. By means of TTL triggering, a maximum pressure scanning rate of 1,200 samples/sec per sensor is capable but a scanning rate of 100 Hz was used in this experiment. In addition to a stable sample interval, triggering assures accurate temporal correlation with the various analog signals which are recorded by the DAQ computer.

Section 2.4: Tunnel Instrumentation

The tunnel is outfitted with pressure ports that are recorded by the ESP Pressure Scanners. These pressure ports are located at various tunnel locations and record both total and static pressure measurements. The total pressure measurements include one total pressure located in front of the subsonic nozzle, another total pressure from a pitot-static probe located forward 18 inches and down 8 inches from the airfoil centerline, and another total pressure located on a wake rake that is 18 inches downstream of the airfoil. The static pressures are measured at the pitot static probe located forward 18 inches and down 8 inches from the airfoils quarter-chord, two static pressures are located at the airfoils quarter-chord and are mounted in the ceiling (upper plenum) and floor (lower plenum) of the test section. Locations are depicted in Figure 8. Each port is connected to the piezoelectric transducer on the ESP Scanners via a short length of plastic tubing. Pneumatic corrections were applied to compensate for amplification attenuation and phase lag and the details are outlined in Section 2.5



An Omega pressure transducer is used to measure the total pressure inside the stagnation chamber. This pressure is monitored during the pressurization of the tunnel during the transient and used to determine if over pressurization occurs.

The Mach number is calculated using the total and static pressure using the isentropic relation for pressure ratio.

Figure 8: Tunnel pressure measurement locations.

$$M = \sqrt{\frac{2}{\gamma - 1} \left[\frac{P}{P_T}^{\frac{\gamma - 1}{\gamma}} - 1 \right]}$$

In addition, the Mach number can be calculated by using the average of the upper and lower static plenum pressures and the wake probe total pressure (when wake is not in use) or settling chamber pitot probe. Both methods of calculating the freestream Mach number result in identical Mach number values.

Over the course of a run, the flow temperature decreases at a rate of approximately 0.5K/s (1°F/s) depending on Mach number and Reynolds number as the air is discharged from the constant-volume high pressure storage tanks. As a consequence, the speed of sound can decrease as much as 9 m/s during a 30 second run. A Omega J-Type thermocouple is used to measure the stagnation chamber temperature. The total temperature is recorded at 10 Hz to the Data Acquisition computer. From the stagnation chamber temperature, the static temperature in the test section can be calculated via the isentropic temperature relation using the freestream Mach number.

$$\frac{T}{T_t} = \left(1 + \frac{\gamma - 1}{2} M_\infty^2 \right)^{-1}$$

Section 2.5: Dynamic Compensation

Pneumatic distortion of the signal was taken into consideration for these experiments. Plastic tubing was used to connect the airfoil to the ESP pressure scanners, and transmit the surface pressure. The tubing length and diameter, combined with the small sensor volume of the internal transducers of the ESP scanner and the high natural frequencies associated with the tunnel, can result in pneumatic distortion. To minimize the distortion of the pressure signals, tube lengths with minimal distances were used.

During the collection of dynamic pressure measurements, specific steps were carried out to address the attenuation of the measured pressure signals due to the viscous effects with the tubing and sensors of the ESP pressure scanners. In 1965 Bergh and Tijdeman developed an analytic model which corrects for the attenuation and phase lag associated with pneumatic tubing³¹. The model that was developed implements the tubing geometry (diameter and length), transducer volume, and the ambient conditions to develop a function to characterize the dynamic response for each tube segment and sensor. The Bergh Tijdeman model (B-T model) is a transfer function ratio of the pressure measured by the sensor with the pressure at the surface pressure tap in the frequency domain. The transfer function has the capability to include discontinuities in

tubing length and radius to account for the difference in internal pressure ports within the airfoil, and the tubing that connects the ports to the ESP pressure scanners.

Section 2.6: Nanosecond – Dielectric Barrier Discharge Pulse Generator

The nanosecond - dielectric barrier discharge is achieved by use of a high-voltage pulse-generator. This generator was designed and built at The Ohio State University. The high voltage unit consists of an internal DC switching circuit and a magnetic pulse compression (MPC) unit. The pulse-generator has a peak voltage of 10 kV, and a pulse width of 90 ns, and an output power coupling up to 20 mJ per pulse. The input DC voltage is limited to 600 V and is split at a high-power Insulated Gate Bipolar Transistor (IGBT) switch to store and input charge up to 20 μC and 100 mJ of input energy in the capacitors. This stored charged is transmitted to the DBD actuators by means of a single-stage low current oscillation circuit in the pulse-generator unit. By compressing the pulse width, the peak voltage is increased.

This particular pulse-generator is equipped with a liquid cooling system to meet the thermal dissipation requirements of the individual internal components. At the frequency of excitation and power levels used in this study, many of the internal components are operating near their maximum temperature range. Thus, cooling of these components is vital for the longevity of the pulse-generator. A radiator and water blocks that are typically used for cooling high-powered computers are fitted into the pulse generator system. The radiator is equipped with three CPU fans to remove heat from the

cooling loop, and distilled water was used as the coolant. One water block was fitted to the IGBT and the high-power resistors. The magnetic output switch was fully submerged in the distilled water to increase heat conduction.

Section 2.7: Shadowgraph

Shadowgraph photography was used as a method to visualize the flow around the leading edge of the NACA 0015 airfoil. This optical technique was employed to further understand physical phenomena about the flow beyond what was obtained by the surface pressure measurements. To allow for the pressure tubing to be passed from the airfoil to the exterior of the tunnel and to permit pitch oscillations, the airfoil is fitted with a circular boss. The boss is located at mid-chord, which results in optical access of the flow from a quarter-chord in front of the airfoil's leading edge to the quarter-chord of the airfoil.

Shadowgraph is an optical technique that reveals non-uniformities in the flow due to temperature and density gradients. The temperature and density gradients act as a disturbance and refract the light rays as the light passes through the medium and casts a shadow. This produces a simple method to detect non-uniformities in the flow that is non-intrusive and simple to use.

The setup consists of a 120V LED array that passes through a pinhole to create a point source of light. The point source of light is directed to a 6 inch concave mirror that is used to collimate the light. The collimated light is directed towards the airfoil in a

manner that resulted in the light being perpendicular to the chord. Obtaining collimated light that is perpendicular to the chord of the airfoil is critical to ensure an average of the flow features along the span of the airfoil. A smooth white sheet of plastic was applied to the opposite side of the test section to serve as the backdrop for imaging. It is important to note that the white sheet was applied to only the rotating window on the opposite side of the light source. Any non-uniformities or marks on the white backdrop will stay fixed relative to the airfoil during pitching. Any differences in the light intensity are proportional to the first spatial derivative of the refractive index in the flow. A high speed Phantom V1210 CMOS camera was used to image the refractive light. Figure 9 and Figure 10 show the shadowgraph setup employed in the dynamic studies.

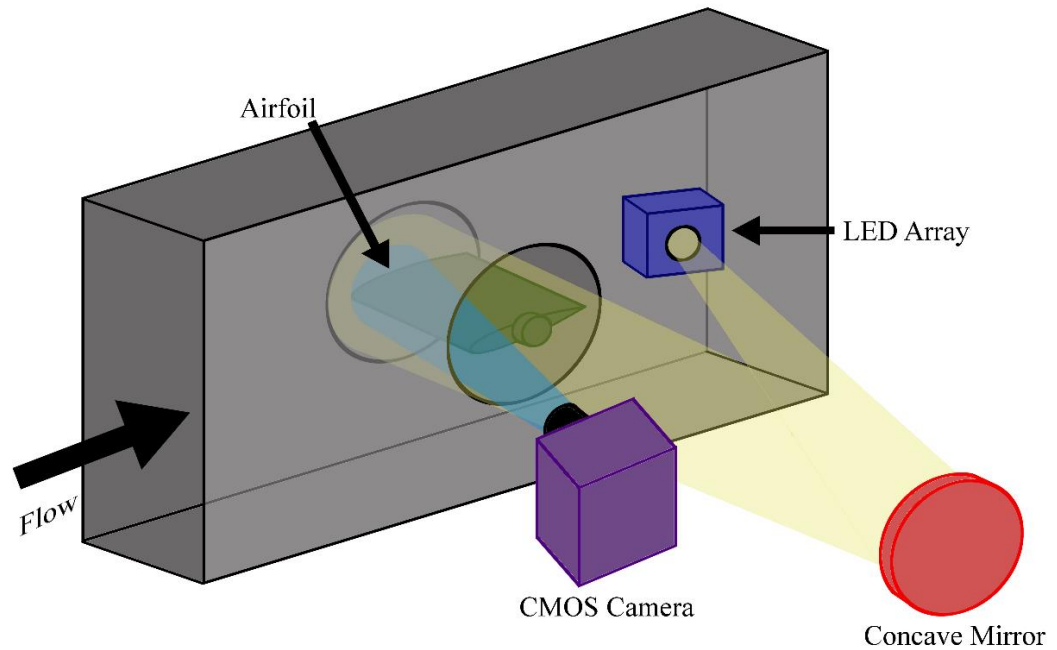


Figure 9: Shadowgraph setup, isometric view.

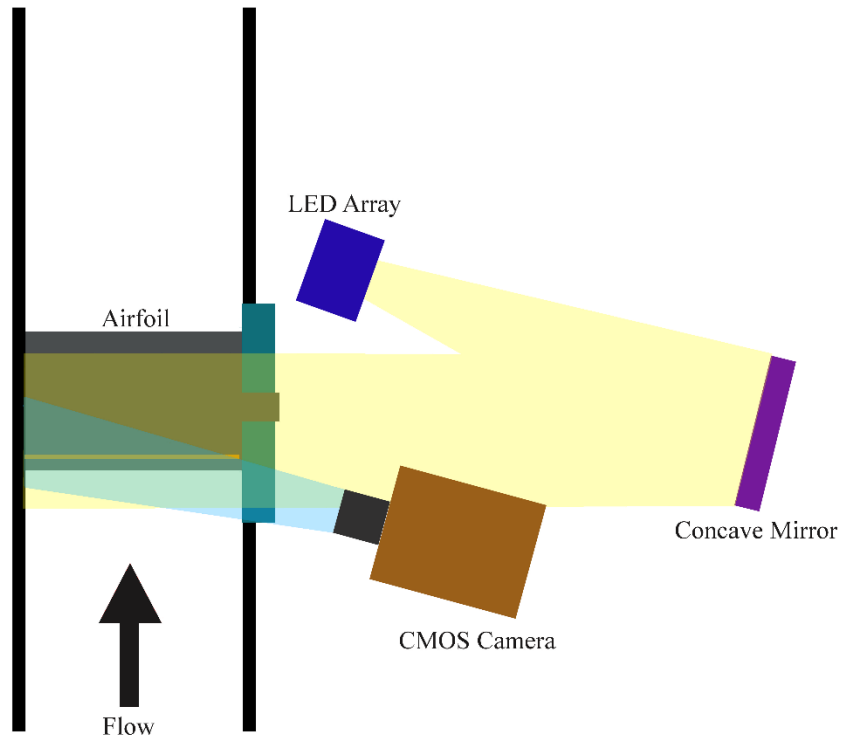


Figure 10: Shadowgraph setup, top-view.

Section 2.8: Test Article

Various airfoils were tested to determine the best candidate for this study. Initially, an aluminum SSC-A09 airfoil was used due to its relevancy to current airfoils in use on rotorcraft as well as the large sample of data that was already acquired with that test article in the 6"×22" facility. To prevent arcing to the surface of the model, three layers of Kapton tape of 0.09mm thickness were wrapped around the surface of the model to act as a dielectric barrier between the model and the actuator. This configuration

resulted in three layers of Kapton, the grounded electrode, and the high voltage electrode with three layers of Kapton separating the high voltage and grounded electrode.

With the three additional layers of Kapton to separate the airfoil from the actuator, it was assumed that the additional thickness was substantial to prevent arcing but would result in a drastic modification to the performance characteristics compared to a clean SSC-A09. Testing of the airfoil indicated that the additional thickness had minor changes to the performance, however there was dramatically depreciated actuator performance. The conductivity of the aluminum model produced an increased loading across the actuator. As a result, less than half the voltage could be supplied to the actuator when installed on the aluminum SSC-A09 compared to installation of the actuator on an acrylic (non-conductive) sheet.

Composite airfoils were considered due to the conductivity of metallic models. Of the models tested, a NACA 4415, an S-809, and NACA 0015 airfoil were used. The composite models that were made for the 6"×22" wind tunnel are formed from an epoxy resin mixed with aluminum powder as a binder.

The NACA 4415 airfoil was not a suitable candidate for this study. With the application of the actuator, the baseline data indicated this test article exhibited trailing edge stall. Trailing edge stall coupled with a sharp leading-edge are non-desirable characteristics. Under trailing edge stall, the stall line advances forward from the trailing edge to the leading edge. To properly control the flow, the actuator would need to be applied directly in front of the initial stall/separation line. This would require a new actuator configuration for various test conditions or an array of actuators over the surface

which would obstruct surface pressure taps. The additional thickness at the leading edge due to the application of the actuator resulted in drastic changes to the performance of the airfoil compared to the clean NACA 4415. The modification to the performance characteristics indicated that the thickness of the surface mounted actuator was intrusive to the flow.

The S-809 airfoil was outfitted with surface mounted plasma actuators at the leading-edge and subjected to Mach 0.2 flow. Under actuation, minimal control authority was demonstrated. Historically, NS-DBD plasma actuators have been studied on symmetric models such as the NACA 0015, or on airfoils with large leading edge radii. The geometry appears to be a critical characteristic of the test article for NS-DBD actuation. Operating the pulse-generator up to the maximum range resulted in marginal changes to the airfoils performance. In contrast, actuation on a NACA 0015 airfoil at low to moderate ranges of the pulse generator range resulted in considerable control in incompressible flows.

Geometry of the test article is of primary importance. Patel *et al* conducted an investigation that tested a NACA 0006, NACA 0012, and NACA 0021 with leading-edge plasma actuation and found that the power required to reattach the flow scaled with leading-edge radius³². The results from the NACA 4415, S-809, and NACA 0015 would seem to suggest the opposite trend, that larger leading-edge radius results in greater control authority. However, Greenblatt *et al* performed an investigation that analyzed the effects of leading-edge actuation on a NACA 0012 with leading-edge stall, and a NACA 0015 with trailing-edge stall³³. The experiment concluded that with leading-edge stall

high frequency (short wave-length) perturbations amplified close to the excitation location (short distances downstream from the actuator), whereas low frequency (long wave-length) perturbations amplified further downstream. These findings suggest that with the NACA 0015 airfoil, low frequency excitation would provide greater control authority due to the inherent trailing-edge stall. However, high frequency excitation would provide greater control authority on naturally occurring leading-edge stall, such as on the S-809 airfoil.

Two main observations came from the preliminary study conducted with the NACA 4415, S-809, and NACA 0015. First, scalability may be an issue with these short chord and span airfoils. The additional thickness and discontinuous surface that is a result of the application of the surface mounted actuator may lead to a severe trip at the leading-edge and a significant change to the geometry of the airfoil that altered the performance characteristics such as the type of stall. Second, with leading-edge stall high frequencies are more effective at establishing flow control. The frequencies that are required may not have been tested and could be outside of the operational limit of the pulse-generator to reattach the flow on the NACA 4415 and S-809. The additional thickness from the actuators may be a less significant geometry alteration on larger leading-edge radius of the NACA 0015; the low frequencies that were tested in the preliminary study were better suited for downstream amplification where stall naturally occurs on a NACA 0015.

The preliminary study of test articles resulted in the selection of a composite NACA 0015 airfoil. This was an existing model at The Ohio State University that met the requirements for this study. The desirable characteristics were: large leading edge radius

where application of the actuator resulted in minimal increase to the airfoil thickness, trailing-edge stall, symmetric geometry, composite substrate, capability of dynamic pitch oscillations, vast historical data, and has recently been used in static angle of attack NS-DBD studies^{22,24}.

The test article used for this experiment was a composite NACA 0015 airfoil with 0.127 m (5 in) chord and 0.152 m (6 in) span, and is pitched about mid-chord. Pressure taps on the airfoil (21 suction surface and 16 pressure surface taps) allow for surface pressure measurements that are used to calculate the lift and moment forces experienced by the airfoil. All pressure tap measurements were recorded using an ESP 32HD pressure scanner utilizing a DTC Initium interface and multiplexed at an effective sample rate of 1 kHz per channel. All data herein presented is calculated from pressure measurements. As this tunnel is a transient tunnel, computations are an average of 20 to 40 cycles depending on pitching frequency.

The nanosecond pulse voltage waveform for the NS-DBD actuators is produced by a high-voltage magnetic compression pulsed power supply. The pulse-generator is discussed in Section 2.6. The pulsed power supply connects to the airfoil via two 1-m long highly insulated 18 gauge continuous-flex wires rated to 20 Amps and 42 kVDC. The actuator is composed of two 0.09 mm thick copper strips. The exposed high-voltage electrode was made with one 6.35 mm (0.25 in) wide copper strip and the covered grounded electrode was made of one 12.70 mm (0.5 in) wide copper strip. The dielectric layer is composed of three layers of Kapton tape, each layer is 0.09 mm thick with a dielectric strength of 10 kV. The actuator was not recessed resulting in a local airfoil

thickness increase of 0.36 mm at the exposed, high-voltage electrode compared to 0.09 mm thick with flush mounted actuators. The actuator is affixed along the spanwise direction near the leading edge with the electrode junction at $x/c = 0.04$ which was left constant for all test conditions. The resultant actuator application method covered the pressure taps from $x/c = 0$ to $x/c = 0.12$ on the suction surface and from $x/c = 0$ to $x/c = 0.2$ on the pressure surface, as illustrated in Figure 12 and Figure 13. A consequence of this reduction in taps at the leading edge is an inability to resolve the suction peak, which leads to substantial errors in calculation of integrated quantities such as lift coefficient or moment coefficient. Despite the obscured suction peak, the ability of the actuators to impact the flow is evident by the remaining pressure taps.

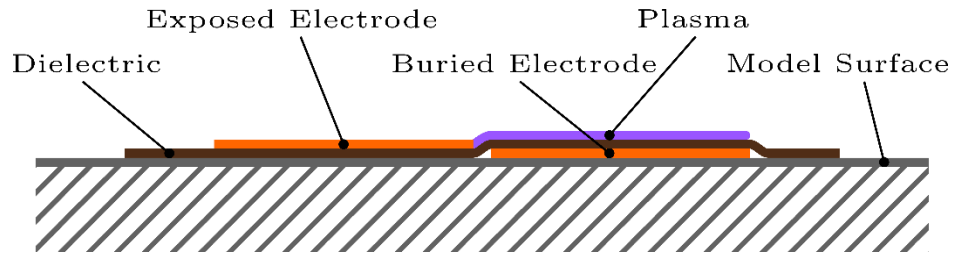


Figure 11: Asymmetric DBD plasma actuator schematic.

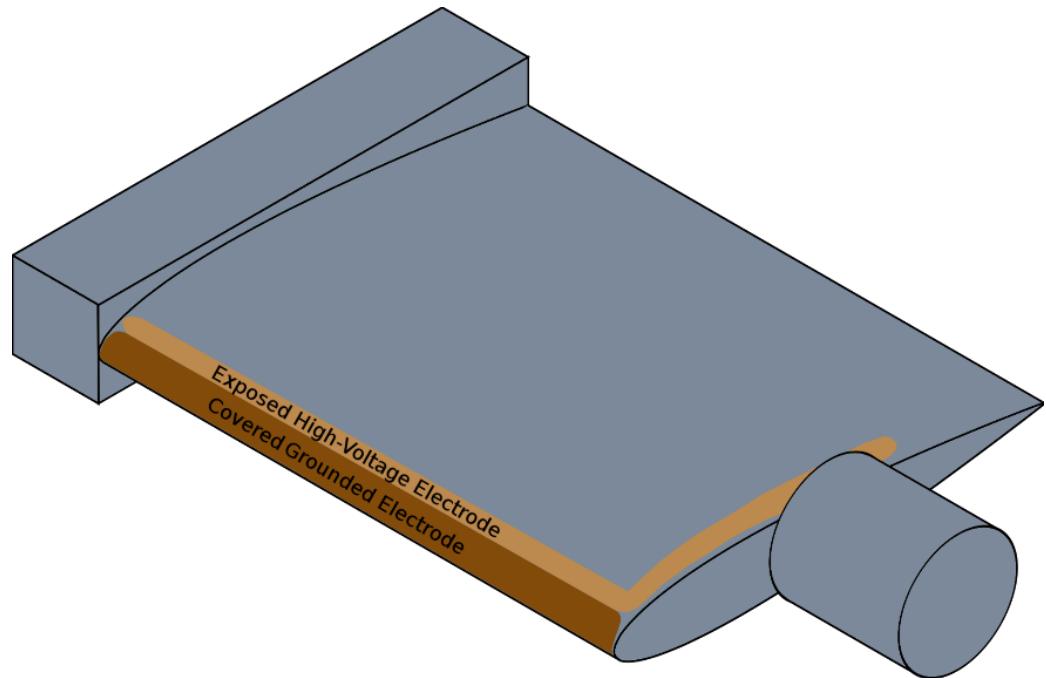


Figure 12: NACA 0015 with actuator at 4% chord.

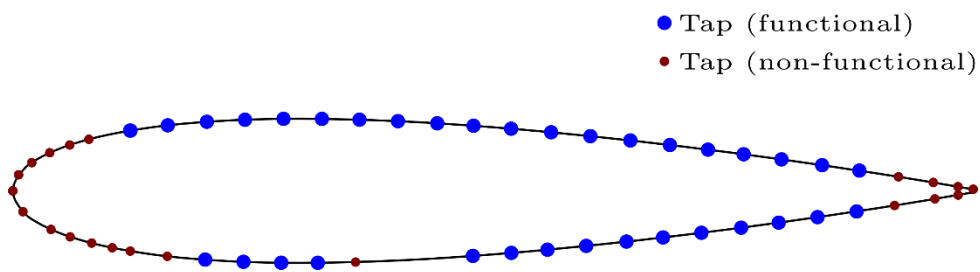


Figure 13: NACA 0015 pressure tap locations.

Section 2.9: Effects of Leading Edge Tap Loss

Even though the poor tap resolution at the leading edge misses the suction peak, there remains sufficient residual pressure data to capture key details of the lift and moment hysteresis loops (in a qualitative sense). In support of this contention, Figure 14 illustrates the changes introduced in the lift and moment orbits by excluding the affected data at a reduced frequency of 0.05, Mach 0.4, and Reynolds number of 2.8 million. The predominant effect is a change in the lift curve slope when the suction peak is not resolved. Similarly, the moment curve is significantly altered, especially immediately prior to stall, resulting in the absence of two self-intersections in the orbit. Discussion of the results will thus be made on a qualitative basis when two or more test conditions are compared.

Significant features of the pressure contours are nonetheless still measurable, even with the loss of such an important subset of the data. The convective velocity of the dynamic stall vortex can be calculated by determining the spatial location of the vortex center, manifested as a region of low pressure, at successive phase angles, and performing the chain rule as follows:

$$\frac{d}{dt} x_{DSV} = \frac{d\Phi}{dt} x \frac{d}{d\Phi} x_{DSV}$$

Further, it is possible to estimate the location of the vortex when it breaks down or is ejected from the surface in separated flow and the phase angle at which it does so. Under some conditions, the low pressure region appears to dissipate before reaching the trailing edge; this is a consequence of surface pressure taps providing no information about the behavior of the flow away from the suction surface.

Figure 15 shows a comparison of C_P contours for the SSC-A09 with the suction peak resolved (a) and with the taps on the first 12% chord removed (b). The abscissa indicates chordwise location, x/c , with the leading edge on the left. The ordinate is the phase cycle with the beginning of the phase at the origin and progressing vertically through the field such that α_{max} is at $\phi=180^\circ$ or midway up the plot. The field color scales with a more negative C_P (increased suction) denoted by deep blue and less negative C_P (decreased suction) depicted in dark red.

The loss of the suction peak due to the removal of taps results in the minimum pressure not being resolved. The contour for critical pressure coefficient on the suction surface is indicated by a bold black line in Figure 15 (a) at $\phi = 135^\circ$, extending from 5 to 10 percent chord. Since the critical C_p occurs near the leading edge, insight into the region of supercritical flow is lost by the lack of leading edge tap resolution.

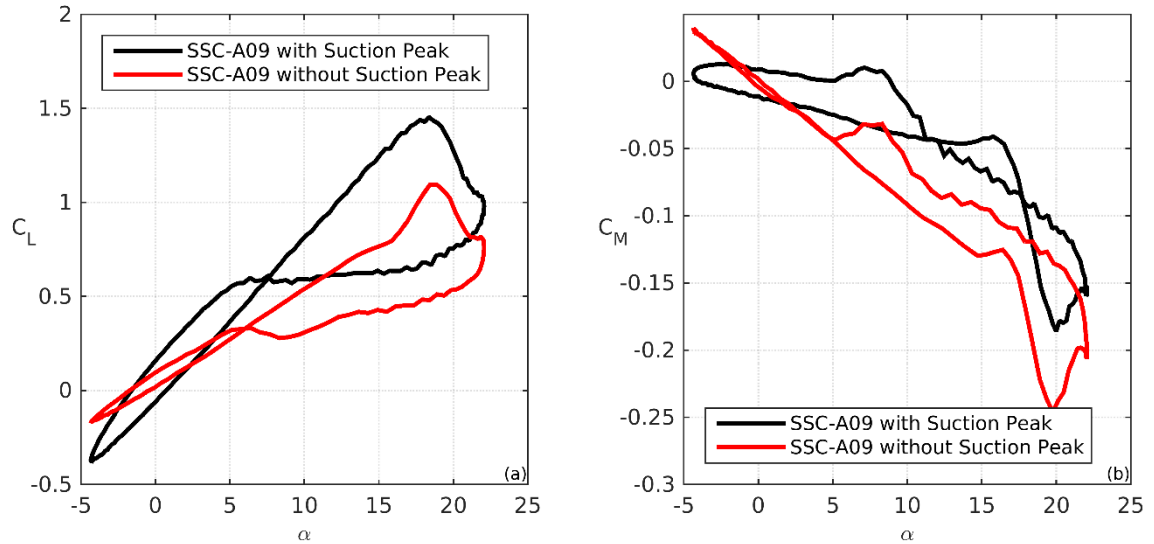


Figure 14: Subsampled SSC-A09 (a) lift hysteresis, (b) moment hysteresis.

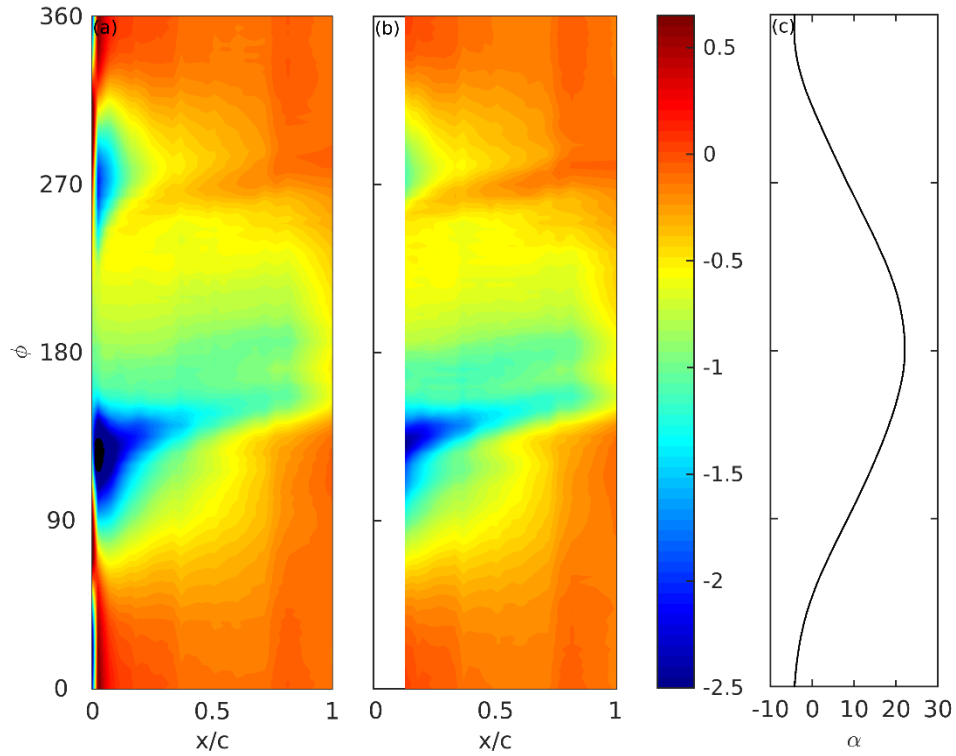


Figure 15: Subsampled SSC-A09 to represent data with loss of the suction peak (a) C_p contour, (b) C_p contour with suction peak removed, (c) α schedule.

Chapter 3: Results

For this experiment, static results were obtained at Mach 0.2 at angles of attack ranging from 0 to 20 degrees. The static results were obtained to demonstrate the actuator effectiveness in a high Reynolds number, high pressure environment. Pitch oscillation results were then obtained at Mach 0.2 and 0.4 with sinusoidal angle of attack oscillations with a mean angle of approximately 10° and amplitude of approximately 10° . Mach 0.2

test conditions allow for the demonstration of flow control during pitch oscillations at high pressure and Reynolds number, while the Mach 0.4 test conditions represent an oscillating compressible flow that is more rotorcraft relevant.

Section 3.1: Static Results Validation

To validate results obtained on the NACA 0015 airfoil used in this experiment, a direct comparison is made between data collected by Rethmel *et al.*²⁴. Rethmel conducted an experiment using a NACA 0015 airfoil with leading edge NS-DBD plasma actuation. Tests were conducted a Mach 0.06, with a Reynolds number of 0.25×10^6 at atmospheric conditions. Due to the 6''×22'' transonic facility designed as a blow down facility, the validation runs were conducted at Mach 0.2, with a Reynolds number of 1.2×10^6 at pressures more than two atmospheres. Even with variations in testing conditions, similar trends were obtained.

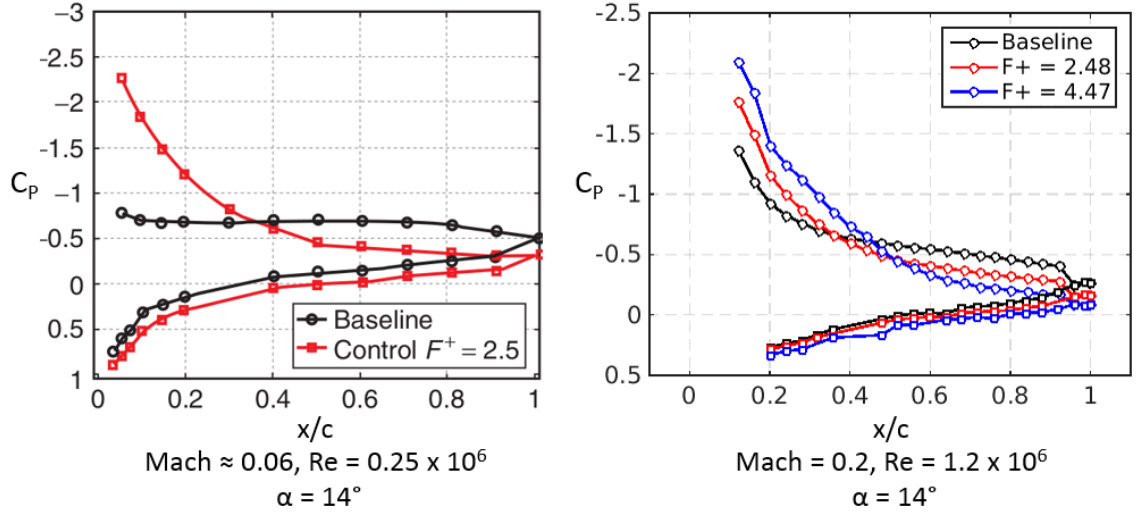


Figure 16: Static comparison of NS-DBD actuation, Rethmel *et al.*²⁴ (left), current study (right).

As indicated by Figure 16, a similar trend occurs for an F^+ of 2.5 (red data set) compared to the baseline (black data set). Due to the higher Mach number and Reynolds number, the suction surface pressure is greater and a stronger suction peak is obtained compared to Rethmel *et al.* in the baseline study. Under actuation, the suction surface pressure is increased and a stronger suction peak is obtained in both studies. This is indicated by the increase in the suction surface pressure from the first available tap to approximately 40% chord. After 40% chord, both studies resulted in a lower suction surface pressure compared to the baseline. However, the pressure surface has higher pressures under actuation. This is an indication that with actuation, stronger circulation was generated aiding in the increased lift.

Comparison with this study to the experiments conducted by Rethmel *et al.* show good agreement between baseline and actuations experiments at similar F^+ . This

indicated that NS-DBD plasma actuation does scale, and is a valid technique to apply in higher Mach number and Reynolds number flow. In addition to the changes in the freestream conditions, the actuation was still capable of increasing the circulation and lift in a high pressure environment.

Section 3.2: Static Results

Static experiments were conducted at Mach 0.2 and Mach 0.4 with Reynolds numbers of 1.2×10^6 and 2.2×10^6 respectively. Excitation of the actuator was applied with various F^+ values to identify the optimum static actuation frequency. Figure 17 shows the lift curve slopes for Mach 0.2 and Mach 0.4.

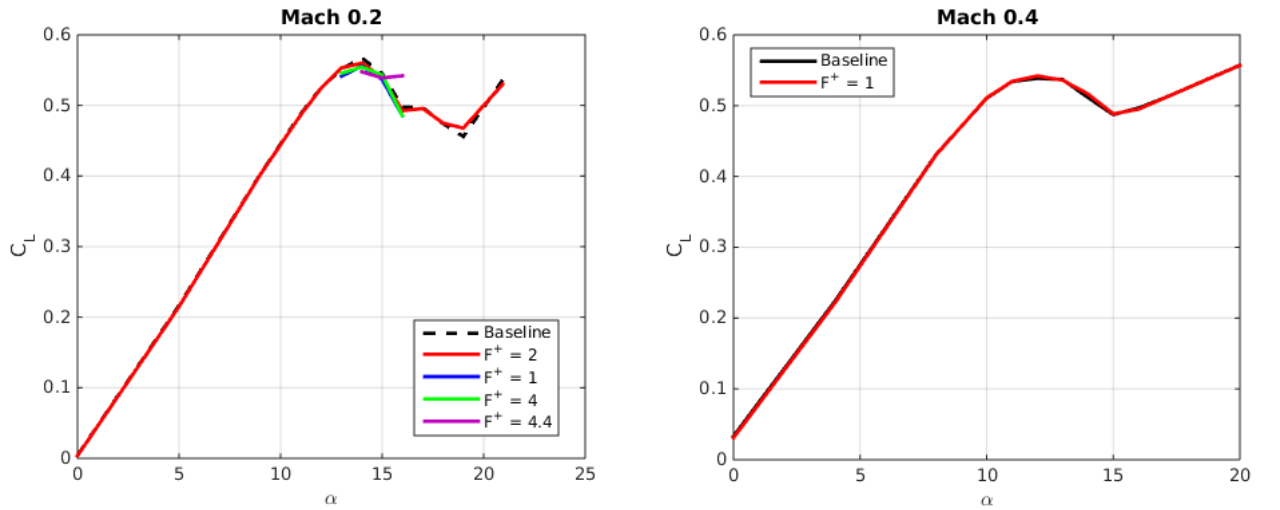


Figure 17: Static lift curve slope, (left) Mach 0.2 at a Reynolds number of 1.2×10^6 , (right) Mach 0.4 at a Reynolds number of 2.2×10^6 .

As evident by the lift curve slopes, actuation had very little effect and minimal changes were observed in the coefficient of lift which is attributed to the limited access of pressure at the suction peak. This is not an unexpected result. As previously indicated in Figure 16, at the high Reynolds numbers used in this study, the baseline has a large suction peak compared to low-speed NS-DBD studies. The elevated baseline suction peak pressures diminish the overall increase of actuation.

In addition to the increased suction pressure, as stated in Section 2.8, the actuators obstruct pressure taps. Due to the small chord of the test article, the actuator covers the first 12% of chord on the suction surface, and first 20% of chord on the pressure surface. The use of leading edge actuation will increase the suction surface and pressure surface pressures most in the region where the actuators are applied and no pressure data is available. As a result, the coefficient of lift is moderately effected from actuation.

Another way to investigate static actuation is by comparing the surface pressure from the non-actuation (baseline) and actuation cases. In particular, the effects of actuation at pre and post stall angles of attack are critical. Figure 18 provides the surface pressures at pre and post stall angles of attack for Mach 0.2.

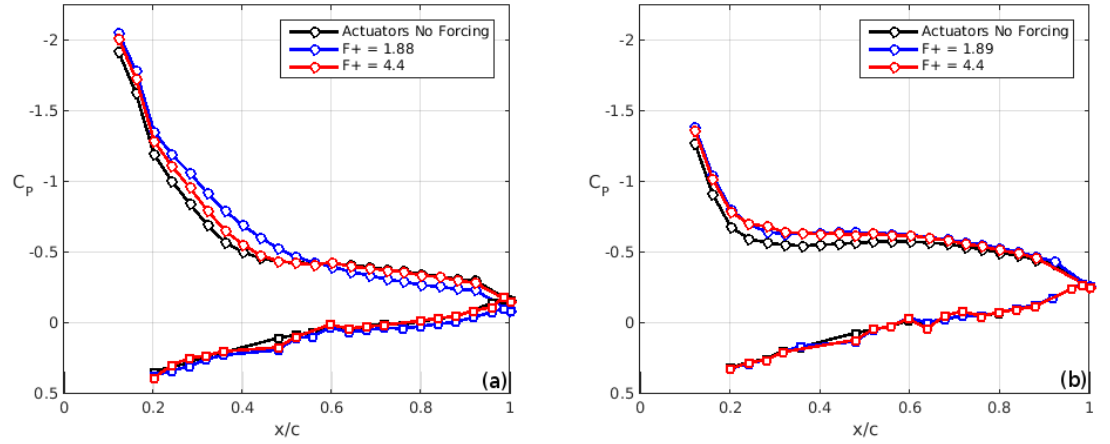


Figure 18: Static angle of attack at Mach 0.2 and Reynolds number of 1.2×10^6 ; (a) $\alpha = 15^\circ$ pre-stall, (b) $\alpha = 16^\circ$ post-stall

As indicated by the surface pressures, actuation does impact the pressure distribution on both the suction and pressure surfaces of the airfoil. The high Mach number and Reynolds number flow results in a high suction peak in the baseline case at 15° and 16° . In comparison to the best actuation frequency, F^+ of 1.88, actuation does have an impact on the surface pressures. From the first available tap to approximately 60% chord at both pre and post stall angles of attack, the suction surface pressure shows a considerable increase compared to the baseline. At the pre-stall condition, $\alpha = 15^\circ$, the suction surface pressure is lower than the baseline case from 60% chord to the trailing-edge of the airfoil. This indicates that the circulation was increased due to actuation. The physical effects of circulation growth produce a decrease trailing-edge C_p and an increase in leading-edge C_p . Therefore it can be presumed that the C_p from the leading edge to 12% chord must have increased to a higher suction pressure than the baseline case.

However without the suction peak C_P available, the integration of the surface pressures for the static actuation case is moderately effected compared to the baseline case.

Section 3.3: Mach 0.2 Pitch Oscillation

This section considers the effects of NS-DBD plasma actuation on pitch oscillations in a steady freestream. Data were acquired at Mach 0.2 (67 m/s) with a reduced frequency of $k = 0.05$ and Reynolds number of 1.2 million. The airfoil pitch schedule, described by the equation

$$\alpha = \alpha_0 + \alpha_{\Delta} \cos(\omega t)$$

was controlled by setting the parameters to $\alpha_0 = 9^\circ$ and $\alpha_{\Delta} = 12^\circ$. Multiple sets of data were collected for F^+ values ranging from 0.78 to 6.09. Of the F^+ values collected, two that showed the greatest control authority are discussed here.

With NS-DBD plasma actuation at $F^+ = 0.78$, the forcing modified the flow and changed the post-stall and reattachment process, as shown in Figure 19 and Figure 20. Characteristics of the dynamic stall vortex were inferred by determining where the vortex begins to convect based on the reduced pressure along the suction surface. The convection speed, peak vortex pressure, and reattachment locations can be determined based on the angular and phase locations of pressure features within the pitching schedule. Using the pressure and angular measurements for the actuator-on case, the dynamic stall vortex is shed at a phase position ($\Delta\phi$) 3.8° earlier than the uncontrolled

case, corresponding to $\Delta\alpha = 0.5^\circ$ in the angle of attack schedule. The actuation also caused pressure recovery or flow reattachment to occur at an earlier phase angle and angle of attack by 3.7° and 0.8° , respectively. This earlier vortex shedding in the actuation case had a reduced peak C_p of 8.4% and travels downstream more slowly at $U_{DSV} / U_\infty = 0.29$, a reduction in speed of 25.6% compared to the non-actuation case which advected at $U_{DSV} / U_\infty = 0.39$.

During the downstroke, after the vortex is shed, the non-actuation case shows a secondary vortex that is shed shortly after the primary vortex at a phase angle of approximately 200° . The actuation case shows a distinct difference in suction surface pressure compared to the non-actuation case. Actuation resulted in a smooth uniform pressure distribution (Figure 19) from a phase angle of 180 to 240 degrees. This more uniform and higher pressure present on the suction surface during the downstroke is thought to be a consequence of the coherent vortical structures that are generated by the NS-DBD actuation²². In this post-stall region, NS-DBD actuation can manipulate flow instabilities which generate these coherent vortices that remain attached to the surface. As these vortices move along the chord, they induce pressure spikes that are displayed as the uniform high pressure during the downstroke in the phase-averaged surface pressure (Figure 19).

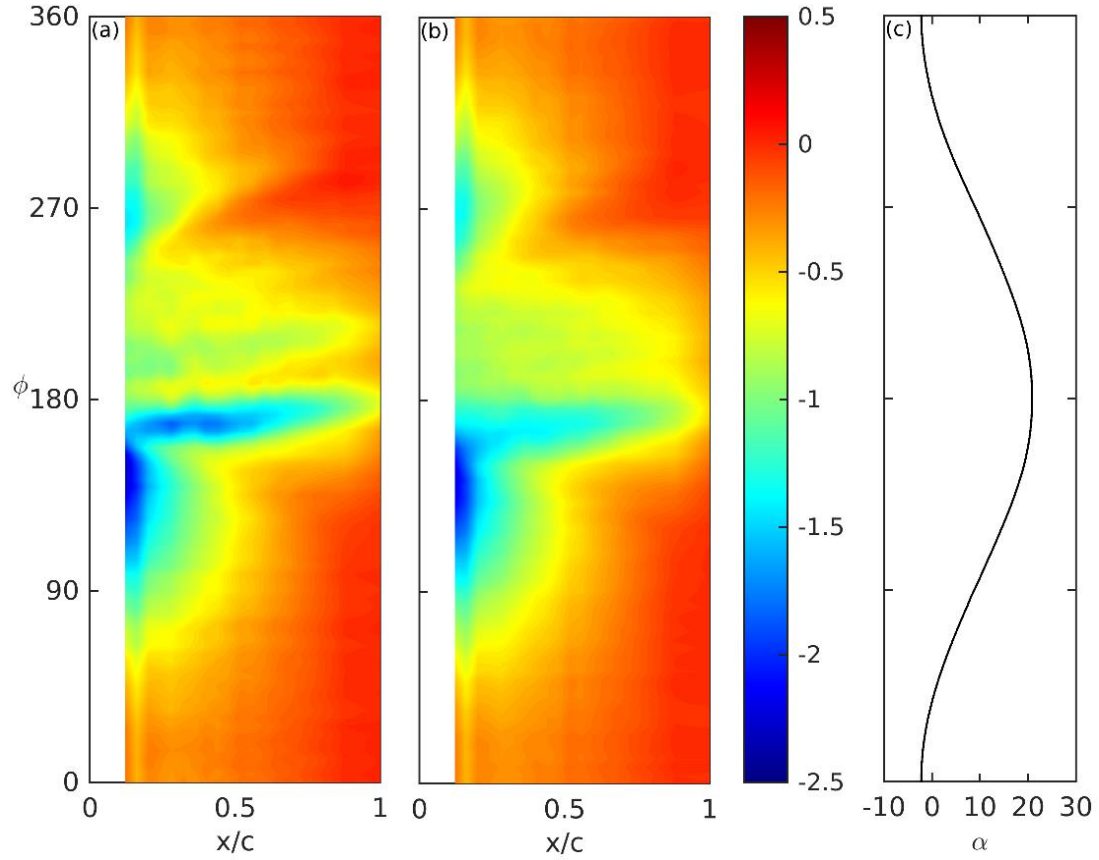


Figure 19: Mach 0.2, $k = 0.05$, $\text{Re} = 1.2 \times 10^6$, C_P contour (a) non-actuation, (b) actuation $F^+ = 0.78$, (c) α schedule.

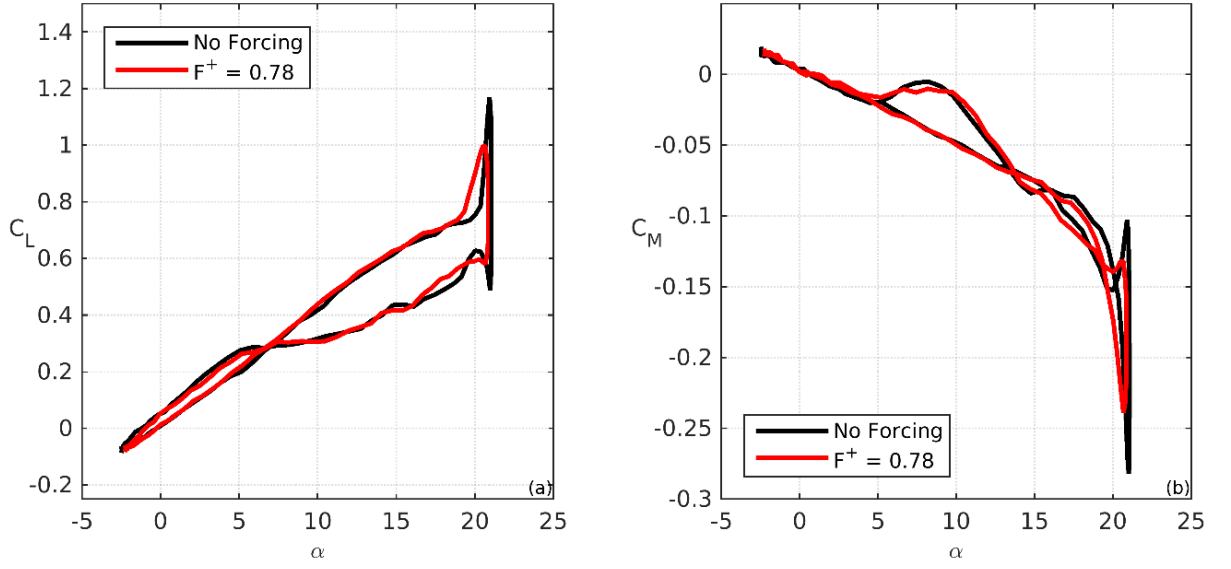


Figure 20: Mach 0.2, $k = 0.05$, $F^+ = 0.78$, $Re = 1.2 \times 10^6$ (a) C_L comparison, (b) C_M comparison.

The removal of the secondary vortex with actuation also improves the aerodynamic damping and pitching moment (Figure 19). Shedding of the secondary vortex induces a positive spike in lift and a negative spike in moment (Figure 20) that are undesirable, resulting in additional loading and vibrations in the rotor system. By eliminating the secondary vortex through flow control, the negative aerodynamic damping has been reduced in the post stall region, which was also observed in a study by Glaz *et al.*³⁴. In the actuation case, the weaker, slower vortex and more uniform surface pressure indicates a more stable shear layer in the post-stall region.

Operating at the same tunnel conditions at Mach 0.2 but increasing the actuation frequency to F^+ of 4.06, similar results are obtained as the case for $F^+ = 0.78$. The results of actuation at an F^+ of 4.06 are shown in Figure 21 and Figure 22. The dynamic stall

vortex generated under actuation begins to shed at a $\Delta\alpha = 0.2^\circ$ and $\Delta\phi = 3.8^\circ$ earlier than the non-actuation cycle. The vortex convection speed is significantly slowed from $U_{DSV} / U_\infty = 0.39$ in the non-actuation condition to $U_{DSV} / U_\infty = 0.23$, a reduction of 40.8%. In addition to the earlier shed dynamic stall vortex, the vortex is slightly weaker with a peak vortex C_p decrease of 2.7%. The downstroke pressure recovery occurred at the same phase and angle of attack position of the pitch schedule. The dynamic stall vortex and post-stall flow modifications at different F^+ values indicate that under various forcing frequencies at these test conditions, NS-DBD actuation is able to effectively control the flow.

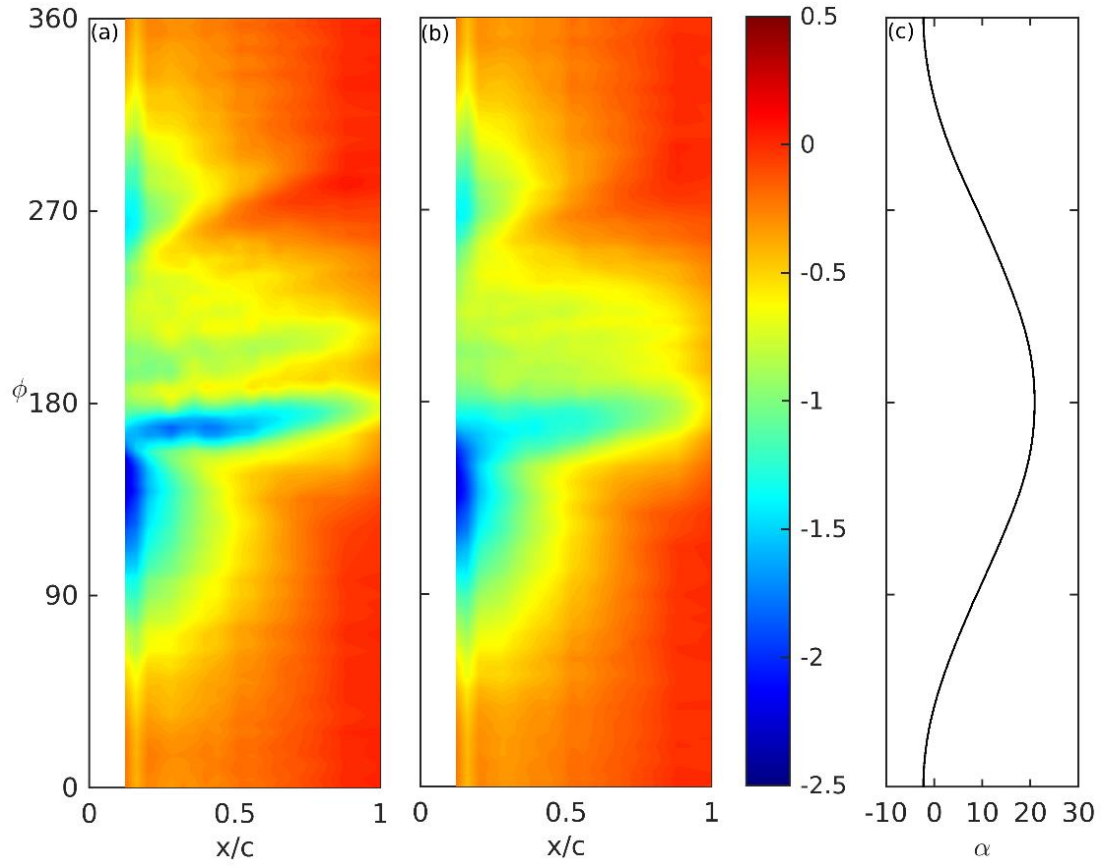


Figure 21: Mach 0.2, $k = 0.05$, $\text{Re} = 1.2 \times 10^6$, C_P contour (a) non-actuation, (b) actuation $F^+ = 4.06$, (c) α schedule.

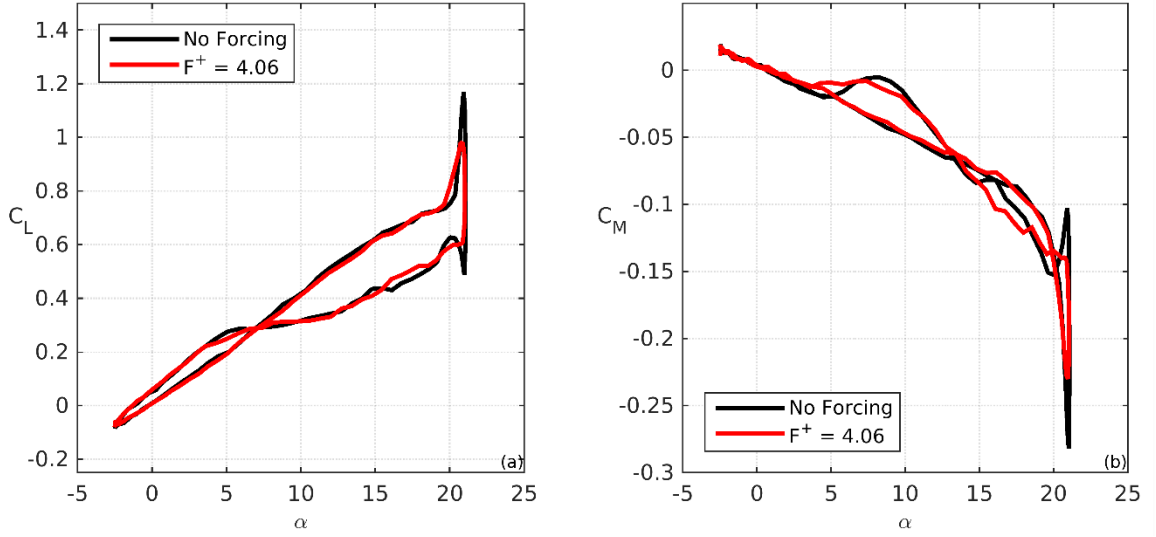


Figure 22: Mach 0.2, $k = 0.05$, $F^+ = 4.06$, $Re = 1.2 \times 10^6$ (a) C_L comparison, (b) C_M comparison.

In contrast, actuation at other frequencies can result in minor changes to the flow. Operating at the same tunnel conditions at Mach 0.2 (67 m/s) with an actuation frequency F^+ of 3.03 demonstrates that the frequency is critical to obtaining flow control/modification. Figure 23 and Figure 24 show the results for F^+ of 3.03. The dynamic stall vortex sheds earlier during the actuation case than the non-actuation case by a $\Delta\alpha = 0.3^\circ$ and $\Delta\phi = 3.8^\circ$. The dynamic stall vortex in the actuation case has a reduced $C_{P_{max}}$ of 4.2% but appears to retain its strength or remain closer to the surface until 50% chord. This is evident by the lower pressure associated with the DSV that extends further over the surface compared to the other actuation cases. Pressure recovery on the downstroke transpires at the same phase and angle of attack as the non-actuation cycle. However, the vortex convects over the suction surface at a slower speed of $U_{DSV} /$

$U_\infty = 0.23$, a decrease of 40.8%. This slower vortex shed is then followed by a weaker secondary vortex that is shed at an identical phase and angle of attack as the secondary vortex shed in the non-actuation case.

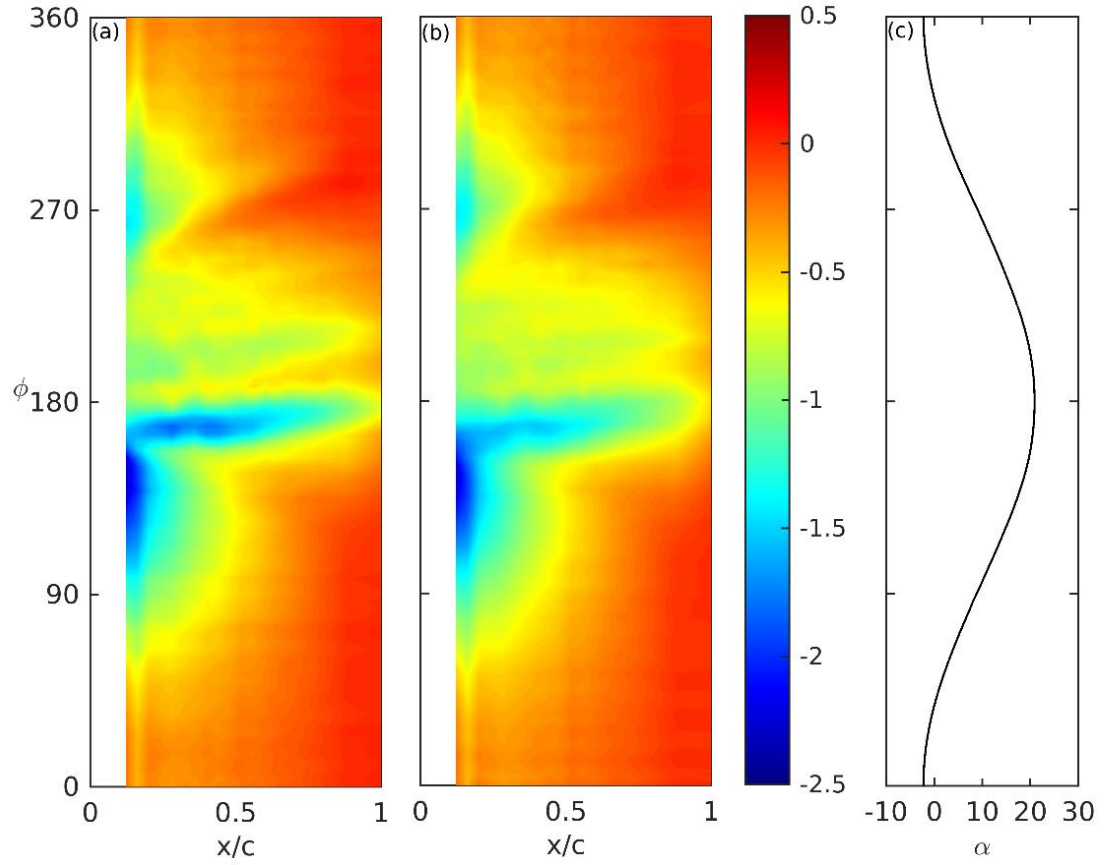


Figure 23: Mach 0.2, $k = 0.05$, $Re = 1.2 \times 10^6$, C_p contour (a) non-actuation, (b) actuation $F^+ = 3.03$, (c) α schedule.

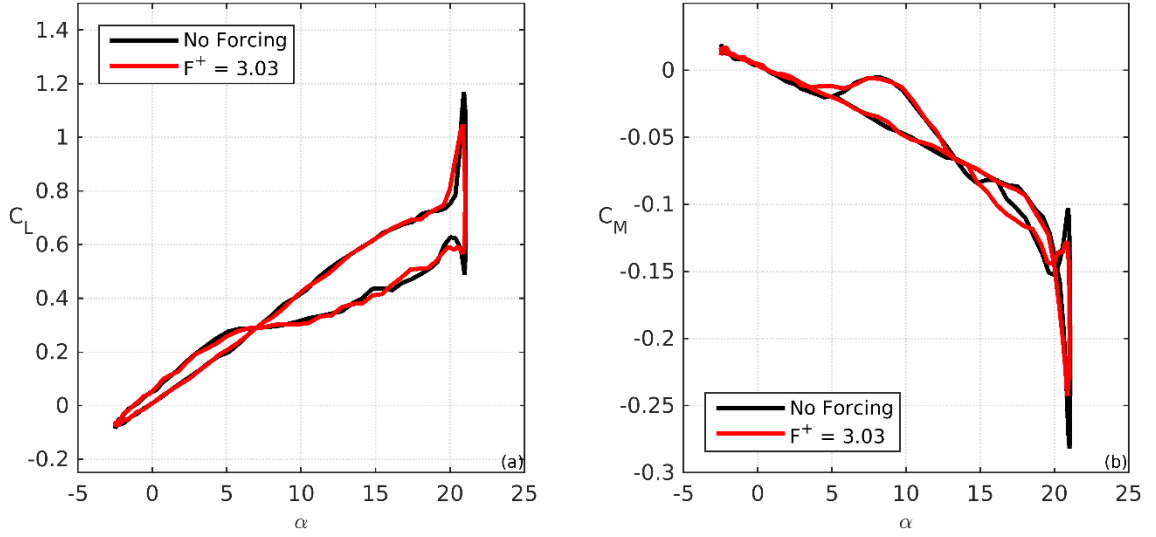


Figure 24: Mach 0.2, $k = 0.05$, $F^+ = 3.03$, $Re = 1.2 \times 10^6$ (a) C_L comparison, (b) C_M comparison.

To accurately compare specific values of the baseline case to the actuation cases is non-trivial due to the loss of the suction peak. Instead, a comparison will be made between the lift excursion (Γ) and the moment excursion (ζ). The lift excursion will be representing the sudden lose in lift associated with lift stall, and the moment excursion will represent the sudden increase in negative (nose-down) pitching moment after moment stall. Comparison between the baseline and actuation Γ and ζ allows for a qualitative assessment. To determine the impact of actuation on the flow, the change in the DSV convection speed (ΔU_{DSV}) and DSV strength, measured by the maximum coefficient of pressure of the DSV ($\Delta C_{p,core}$), were calculated. In addition, the angle of attack at which the DSV begins to convect ($\Delta \alpha_{DSV \text{ shed}}$) and the angle of attack that reattachment occurs during the down stroke ($\Delta \alpha_{Recovery}$) were calculated to compare the

effects of actuation. Figure 25 shows what the parameters Γ and ζ represent in the lift and moment hysteresis loops respectively.

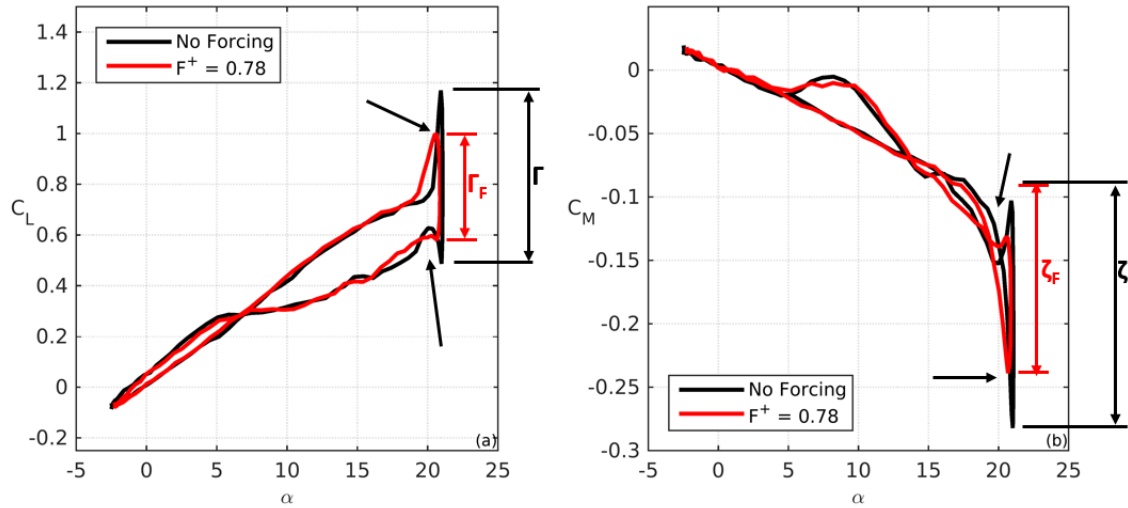


Figure 25: Representation of lift excursion Γ (left), and moment excursion ζ (right).

The key observations from the Mach 0.2 actuation study have been combined in Table 1. Table 1 tabulates the parameters that are used to compare the actuation results to the baseline pitch oscillation case. The table corresponds to the data points that are tabulated in Figure 25. The data is color coded so that green and red represent favorable and unfavorable changes respectively. From the tabulated data, it is desirable to reduce the DSV strength or DSV peak coefficient of pressure. Reduction of the DSV strength helps to reduce the severity of the lift excursion associated with the detachment of the DSV from the suction surface. In addition to reducing the lift excursion, a weaker DSV correlates to an earlier pressure recovery during the down stroke. In a similar manner, the

reduction in DSV convection speed is also desirable. A reduction in convection speed directly correlated to a delay in stall and reduces the severity of the moment excursion that is associated with moment stall.

Table 1: Mach 0.2, Reynolds number 1.2×10^6 Tabulated parameters

F^+	Γ_F/Γ	ζ_F/ζ	ΔU_{DSV} (%)	$\Delta C_{P,core}$ (%)	$\Delta \alpha_{DSV}$ Shed	$\Delta \alpha_{Recovery}$
0.00	-	-	-	-	-	-
0.78	0.61	0.56	-25.64	-8.4	0.5	-0.8
0.85	0.64	0.73	-24.83	-1.8	0.3	-0.4
0.93	0.80	0.70	-24.83	5.4	0.1	-0.6
1.02	0.59	0.43	-50.42	-4.7	0.2	-1.5
1.13	0.79	0.72	-40.13	-3.2	0.2	-0.3
1.22	0.63	0.79	-40.39	-2.2	0.4	-0.7
1.32	0.52	0.61	-40.20	-9.9	0.2	-0.8
1.42	0.84	0.56	-40.39	-6.0	0.3	-0.8
1.53	0.63	0.55	-25.73	-7.5	0.3	0.0
3.03	0.70	0.64	-40.81	-4.2	0.3	0.1
4.06	0.54	0.48	-40.81	-2.7	0.2	0.0
6.09	0.64	0.56	-40.96	2.4	0.2	0.0

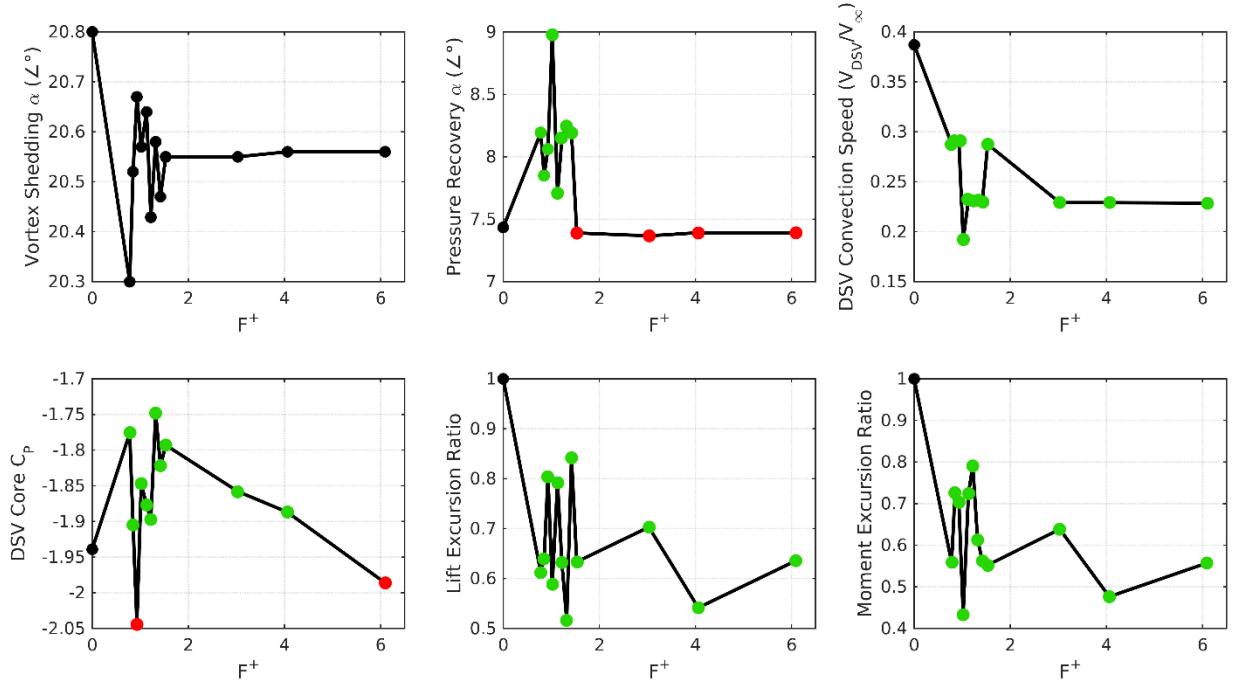


Figure 26: Mach 0.2, $Re = 1.2 \times 10^6$ effects of actuation (a) α_{DSV} shed, (b) angle of attack of pressure recovery, (c) maximum lift coefficient, (d) normalized vortex convection speed vortex convection speed U_{DSV}/U_∞ , (e) peak vortex pressure, (f) minimum moment coefficient.

Figure 26 and Table 1 show that for a freestream Mach number of 0.2, actuation at all F^+ values tested resulted in a reduction in the lift excursion and moment excursion. In all of the actuation cases, the vortex begins to shed earlier in the pitch cycle than in the non-actuation case ($F^+ = 0$). The earlier shed vortex may be caused by the compression wave of the pulse train essentially disturbing the boundary layer, and caused the dynamic stall vortex to lift or pinch off prematurely rather than delay the shedding of the vortex. The prematurely vortex shed is weaker and convects at a significantly slower speed than the non-actuation dynamic stall vortex, with the most reduction in DSV convection speed

of 50% at an F^+ of 1.02. The strength of the DSV was reduced for all F^+ values except 0.93 and 6.09. The angle of attack at which pressure recovery occurs had an interesting feature occur. For F^+ values less than an F^+ of approximately 1.50 resulted in pressure recovery to occur later in the pitch cycle (at a lower angle of attack during the downstroke) than the baseline case; at F^+ values greater than 1.50 results in an earlier pressure recovery (at a higher angle of attack during the downstroke).

It is important to note that the forcing frequencies that resulted in the most improvements were near an F^+ value of unity for the incompressible study at Mach 0.2. In the studies that used non-dimensional frequencies to represent the actuation excitation found that the optimum forcing frequency was unity²³. These studies established that forcing at unity resulted in a delay in moment stall as well as reducing the negative peak moment. In this current study, Table 1 and Figure 26 indicate fluctuations in performance rather than a smooth trend towards an optimum frequency at unity. This is likely attributed to the unknown freestream temperature, which is a consequence of the blowdown facility. Additional, throughout the run the static temperature drops as air is discharged from the constant-volume storage tank. The dimensional forcing frequency has to be selected based on the guess for the initial temperature during the run. Even with the approximate linear temperature drop throughout the run, the temperature that corresponds to a perfect F^+ of unity may not coincide with the specific Reynolds number sought in this experiment since the Reynolds number increases throughout the run. The F^+ values that are expressed for this data are the average F^+ over the entire run. For these reasons associated with the inability to accurately acquire an initial temperature could

explain the oscillations in performance as F^+ approaches unity. However, the results agree with previous studies that indicate the optimum non-dimensional forcing frequency of unity for incompressible flows.

Section 3.4: Mach 0.4 Pitch Oscillations

This section presents the results of NS-DBD plasma actuation on pitch oscillations in a steady freestream in compressible flow. Data were acquired at Mach 0.4 (134 m/s) with a reduced frequency of $k = 0.05$ and a Reynolds number of 2.2 million. The airfoil pitch oscillation schedule, described by the equation

$$\alpha = \alpha_0 + \alpha_{\Delta} \cos(\omega t)$$

was controlled by setting the parameters to $\alpha_0 = 9^\circ$ and $\alpha_{\Delta} = 16^\circ$. The mean angle of attack increases at the higher Mach number due to the higher aerodynamic loading (caused by an increased dynamic pressure), increased physical frequency of pitching oscillation, and structural elastic deformation. Multiple sets of data were collected for F^+ values ranging from 0.39 to 2.03. It is important to note that with the increased Mach number and the introduction of compressible flow, the mechanisms of dynamic stall change³⁵. As the free stream velocity is increased, the velocity about the leading-edge is proportionally increased, eventually becoming locally supersonic even when the free stream is unquestionably subsonic. The shock terminating this supercritical flow can interact with any separation bubble present or cause the boundary layer to separate,

initiating dynamic stall; these mechanisms are of course not available to lower-speed flows in the absence of locally supercritical flow. Such a change in the mechanism of stall affects the subsequent behavior of the DSV, and may cause its size, strength, and trajectory to differ from its incompressible counterpart.

Actuation at F^+ of 1.03 strengthened the vortex and ultimately increased the severity of the dynamic stall event. The evidence of the stronger vortex is present in the C_P contour plot in Figure 27 and hysteresis loops in Figure 28. During the upstroke the lift hysteresis loop manifests a positive spike that is not present in the non-actuation case as the vortex convects over the suction surface. As the stronger vortex detaches from the surface at the trailing-edge, moment stall is delayed by 4° but reaches a sharper, more negative peak in moment

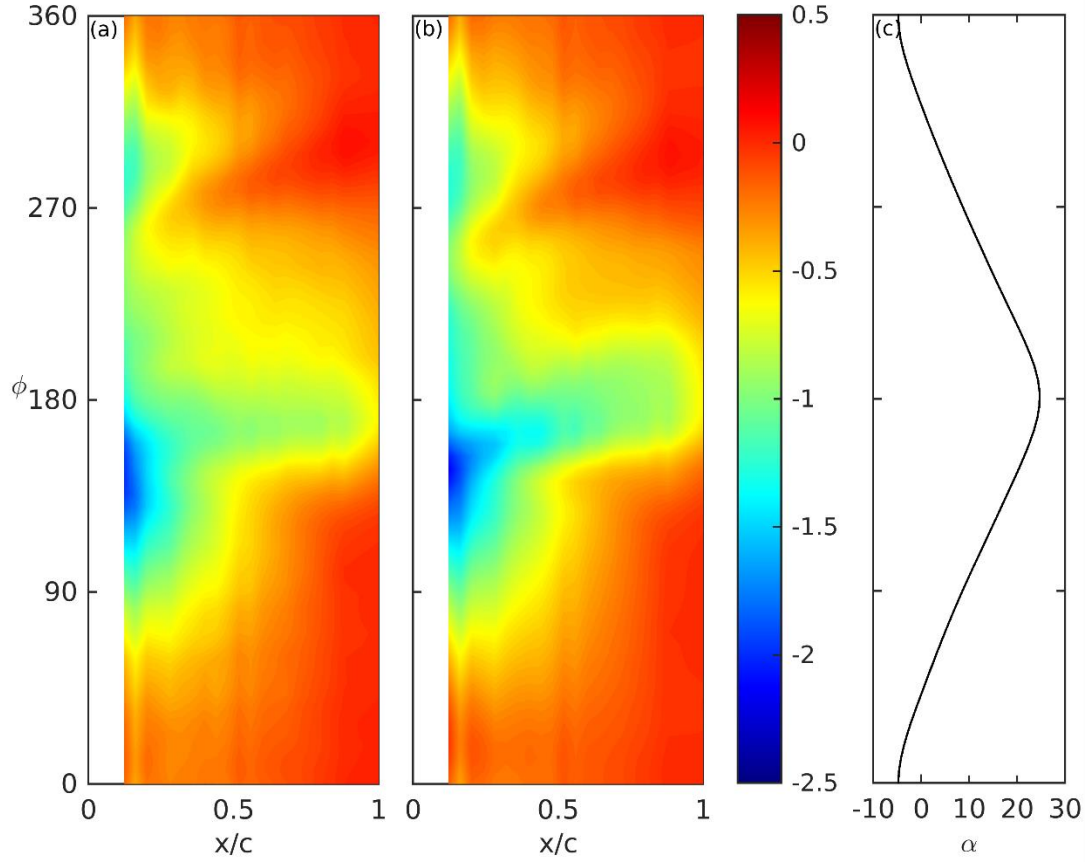


Figure 27: Mach 0.4, $k = 0.05$, $\text{Re} = 2.2 \times 10^6$ C_P contour (a) non-actuation, (b) actuation $F^+ = 1.03$, (c) α schedule.

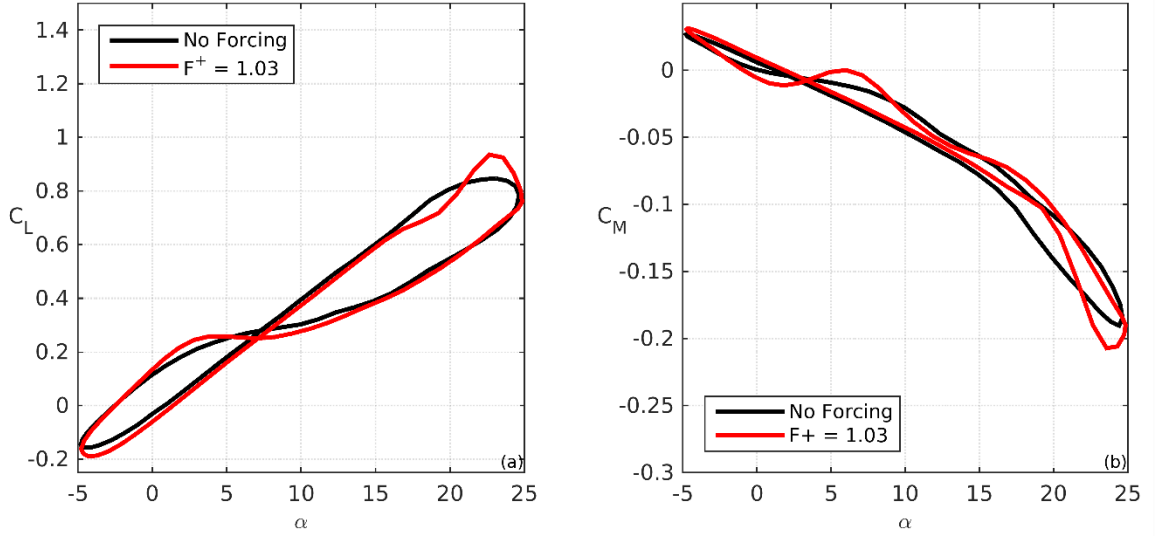


Figure 28: Mach 0.4, $k = 0.05$, $F^+ = 1.03$, $Re = 2.2 \times 10^6$ (a) C_L comparison, (b) C_M comparison.

At Mach 0.4 with an $F^+ = 1.03$, actuation resulted in a stronger dynamic stall vortex. The vortex began to shed from the leading edge at a $\Delta\alpha = 0.4^\circ$ and $\Delta\phi = 5.9^\circ$ sooner than the non-actuation case, but with a higher peak C_P of -2.1, representing an increase in magnitude of 8.2% over the non-actuation case. Pressure recovery did occur $\Delta\alpha = 0.4^\circ$ earlier for the actuation case. This stronger vortex advected over the airfoil at a much slower convection speed of $U_{DSV} / U_\infty = 0.14$, a decrease of 39.2% relative to the unforced case.

Compressibility can cause premature separation due to shocks and flow controlled stall delay is strongly affected by the presence or absence of these shock interactions³⁵. The compression wave generated by the NS-DBD actuators is located at 4% chord, but naturally occurring shocks are documented to develop between 0% to 10% chord on a

NACA 0015^{18,19}. The locations of the shock and NS-DBD-induced compression waves, relative to one another, may determine the effectiveness of the actuator's ability to exert control authority.

Actuation with $F^+ = 1.03$ at these conditions excites instabilities in the bulk flow or shear layers. Modification of these naturally occurring flow features with the NS-DBD compression waves has intensified the dynamic stall vortex as indicated by the decreased surface pressure located at $\phi = 150^\circ$ in the actuated case in Figure 27. Under identical tunnel operating conditions at Mach 0.4 (134 m/s), but increased actuation frequency of $F^+ = 2.03$, the actuators can achieve slight observable flow improvements, Figure 29 and Figure 30. The actuation promoted earlier shedding of the vortex. The shedding event advanced by $\Delta\alpha = 0.5^\circ$ and $\Delta\phi = 5.7^\circ$ compared to the non-actuation case. The earlier DSV convects at a slower rate of $U_{DSV} / U_\infty = 0.12$, a decrease of 49.4% compared to the non-actuation convection of $U_{DSV} / U_\infty = 0.24$. In addition, the DSV strength was marginally reduced with a peak C_p reduction of 4.2%. During the downstroke, actuation leads to earlier reattachment by $\Delta\alpha = 2.5^\circ$ and $\Delta\phi = 12.0^\circ$ degrees compared to the non-actuation case.

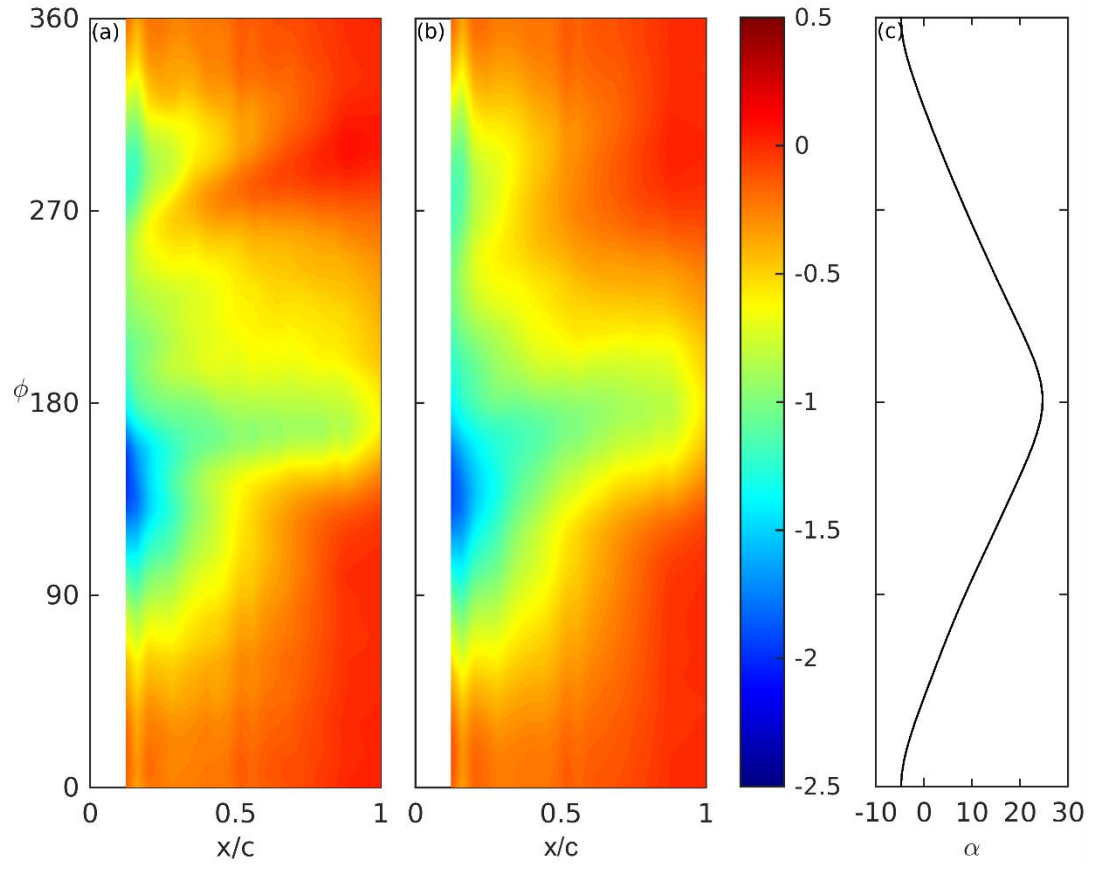


Figure 29: Mach 0.4, $k = 0.05$, $\text{Re} = 2.2 \times 10^6$ C_P contour (a) non-actuation, (b) actuation $F^+ = 2.03$, (c) α schedule.

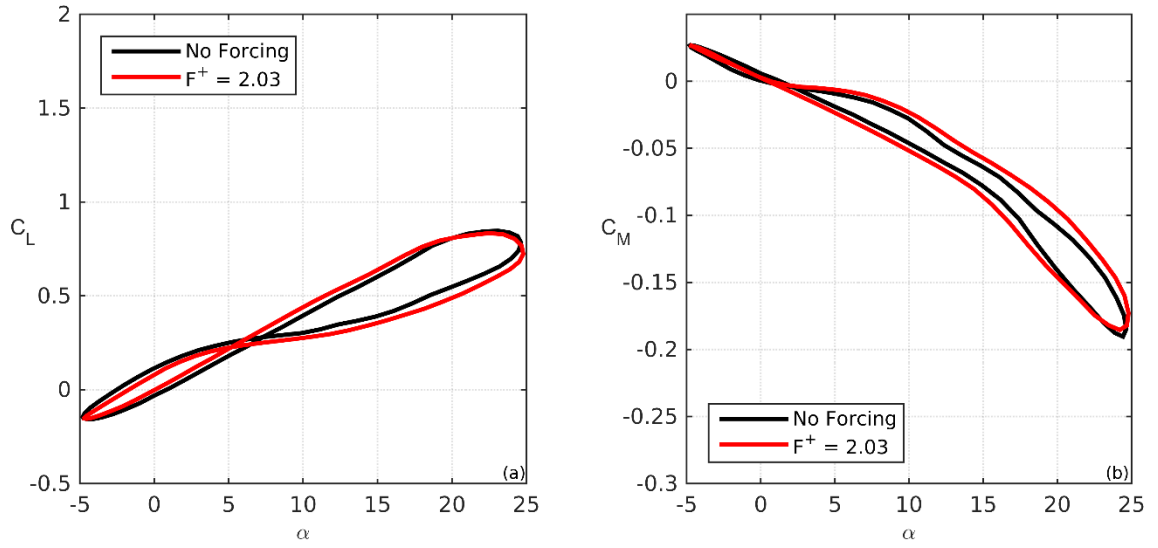


Figure 30: Mach 0.4, $k = 0.05$, $F^+ = 2.03$, $Re = 2.2 \times 10^6$ (a) C_L comparison, (b) C_M comparison

Additional observations and calculations have been summarized below by plotting the various characteristics versus F^+ value. The results for the lift and moment excursions, change in DSV convection speed and strength, and angle of attack of DSV shedding and pressure recovery are tabulated and plotted in Table 2 and Figure 31 respectively with favorable flow changes and unfavorable changes color coded in green and red respectively.

Table 2: Mach 0.4, Reynolds number 2.2×10^6 tabulated parameters

F^+	Γ_F/Γ	ζ_F/ζ	ΔU_{DSV}	$\Delta C_{P,core}(\%)$	$\Delta \alpha_{DSV} \text{ Shed}(\%)$	$\Delta \alpha_{Recovery}$
0	-	-	-	-	-	-
0.39	1.19	1.04	-32.14	10.87	-0.7	0.5
0.76	1.13	1.01	-39.25	13.92	0.5	0.3
1.03	1.37	1.03	-56.62	-8.23	0.4	0.4
1.56	1.19	0.94	-49.38	-3.15	0.4	1.5
2.03	1.03	0.92	-49.38	4.22	0.5	2.5

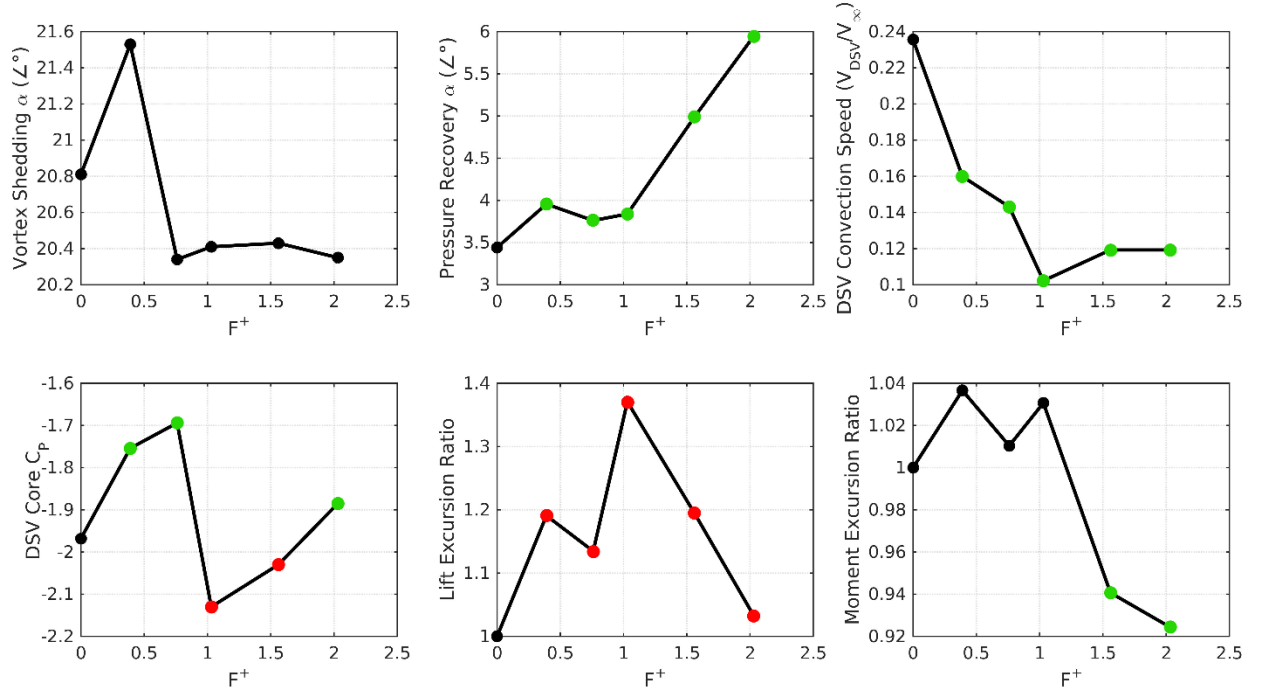


Figure 31: Mach 0.4, $Re = 2.2 \times 10^6$ effect of actuation (a) vortex angle of attack shedding angle, (b) angle of attack of pressure recovery, (c) maximum lift coefficient, (d) normalized vortex convection speed vortex convection speed U_{DSV} / U_∞ , (e) peak vortex pressure, (f) minimum moment coefficient.

In contrast to the Mach 0.2 results, actuation with a freestream Mach of 0.4, the pressure recovery occurred earlier for all F^+ values tested. This may be attributed to the periodic excitation of the actuators redistributing the higher momentum from the bulk

flow, resulting in earlier pressure recovery. Actuation at Mach 0.4 also resulted in an increase in the lift excursion (Γ) for all F^+ values tested. The moment excursion (ζ) was increased for F^+ values below 1.50, but reduced for F^+ values above 1.50. However increasing F^+ results in both the lift and moment excursions reducing towards favorable benefits. All F^+ values reduced the convection speed dramatically. The strength of the DSV does not have a definitive trend, but appears to reduce as F^+ is increased. Similarly, the angle of attack of pressure recovery tends to occur earlier in the pitch cycle during the downstroke with increasing F^+ .

It is important to note that the trend at Mach 0.4 of increasing the actuation frequency beyond F^+ of 2.03 would lead to further reduction of the lift and moment excursions resulting in further favorable benefits and greater improvements in performance. However, as actuation frequencies were increased beyond 2.03 the output current from the pulse-generator approached the maximum allowable current limit. In addition to reaching the limitation of the pulse-generator, the increased power output at F^+ values above 2.03 resulted in an increase in the levels of EMI. These increased levels of EMI began to effect the data acquisition system. The effects included false triggering of the ESP scanners, interruption of the external clock TTL signal resulting in a non-constant sample frequency, and distortion to the optical encoder signal. This resulted in the inability to accurately determine the angular position of the airfoil and to precisely synchronize the various components of the data acquisition system.

The inability of the actuators to exert flow control at Mach 0.4 can be attributed to several factors. The first is the high pressure, high Reynolds numbers environment.

Previous investigations have shown that control of flow separation is limited, and the efficiency of NS-DBD actuation is highly dependent on the forcing frequency at Reynolds numbers greater than 1 million³⁶. In addition, as pressure increases, the plasma formation is altered from a uniform distribution to a filamentary distribution³⁷. When filamentary plasma distribution occurs, for the same input voltage to the pulse generator a high current is output to the actuator. The high pressure freestream essentially increases actuator resistance, or loading across the actuator.

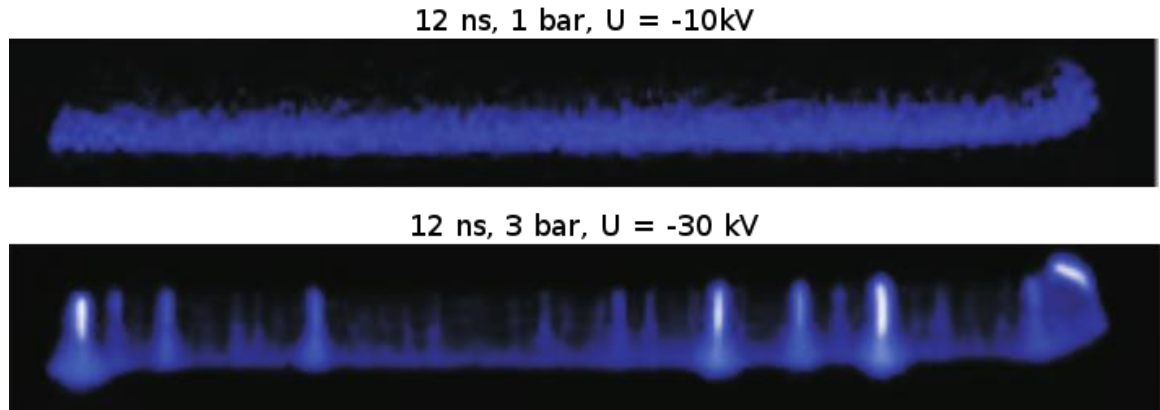


Figure 32: Uniform and filamentary plasma distribution at low and high pressures respectively Stepanyan *et al.*³⁷.

Studies conducted by Beahan *et al.*¹⁹ and Seifert *et al.*¹⁸. showed that at Mach 0.4 with a Reynolds number of 1.5 million, at a static angle of attack between 12° and 15°, shocks were present between 0 and 10 percent chord on a NACA 0015 airfoil. For a dynamically pitching NACA 0015 airfoil at a reduced frequency of 0.05, Reynolds number of 1.5 million, and Mach number of 0.4, shocks were again present on the first 10 percent of the airfoil¹⁹. With shocks occurring in the same region as the actuators, the

compression wave could have less effect on the flow. Naturally occurring shocks can induce and promote early flow separation. The interaction of these shocks and other compressible flow features at the leading edge may result in a separated shear layer that has already lifted away from the surface, out of the influence of the compression waves. If the shock occurs downstream of the actuator, the large pressure discontinuity may interrupt the effects of flow control or cause separation. Therefore, at a Mach number of 0.4, the actuator junction at 4% chord may not be optimally positioned for flow control in the presence of leading edge shocks.

As the Mach number is increased from 0.2 to 0.4, the actual physical oscillation frequency is also doubled to maintain a constant reduced frequency. The temporal relationship between the oscillation cycle and the boundary layer, shear layer, and actuation time scales may be ineffective at achieving flow control.

Section 3.5: Shadowgraph

Shadowgraph was used to visualize the outer flow and determine the presence of naturally occurring shocks. The shadowgraph setup is discussed in Section 2.7. Beahan *et al.*¹⁹ found that on a NACA 0015 at Mach 0.4 and a Reynolds number of 1.5×10^6 at static angles of attack, lambda shocks occur within the first 10% of chord at angles of attack between 12° and 15° . An interferometry image from Beahan's study is provided in Figure 33.

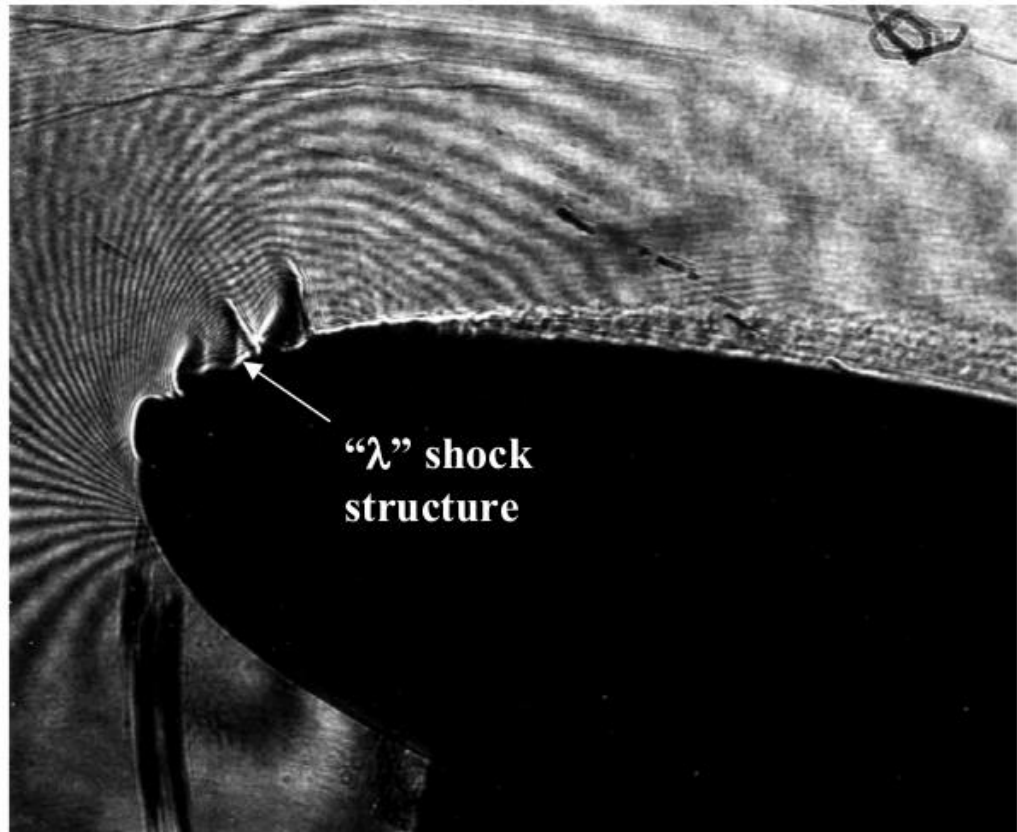


Figure 33: Interferometry image at Mach 0.4, Reynolds number 1.5×10^6 , of a NACA 0015 airfoil. Figure from Beahan *et al.*¹⁹.

These naturally occur shocks are in the same proximity on the airfoil as the actuator position. Since the actuator is non-recessed but surface mounted, it is critical to get a better understanding of the flow in this region. The shadowgraph technique was employed due to the limitation of optical access from one side. The shadowgraph images that will be presented are from pitch oscillation runs at Mach 0.4 and a Reynolds number of 2.2×10^6 . The actuation frequency was set to an F^+ of 2.03, which corresponded to the

best optimum frequency attainable at Mach 0.4. Both actuation and non-actuation runs were captured at a frame rate of 3000 frames per second to make a direct comparison.

In the non-actuation run, the earliest shock occurred at an angle of attack of 11.5° . In contrast, under actuation, the earliest shock occurred at an angle of attack of 11.8° . However this slight variation in the first formation of a shock is not particularly definitive to the effects of actuation. The angle of attack at which shocks occurred during the upstroke changed with each cycle during a run. Figure 34 shows the first lambda shock that was captured during a cycle in the non-actuation run. An important observation is that the first shock appears to have formed with the foot of the lambda shock located directly at the actuator junction.

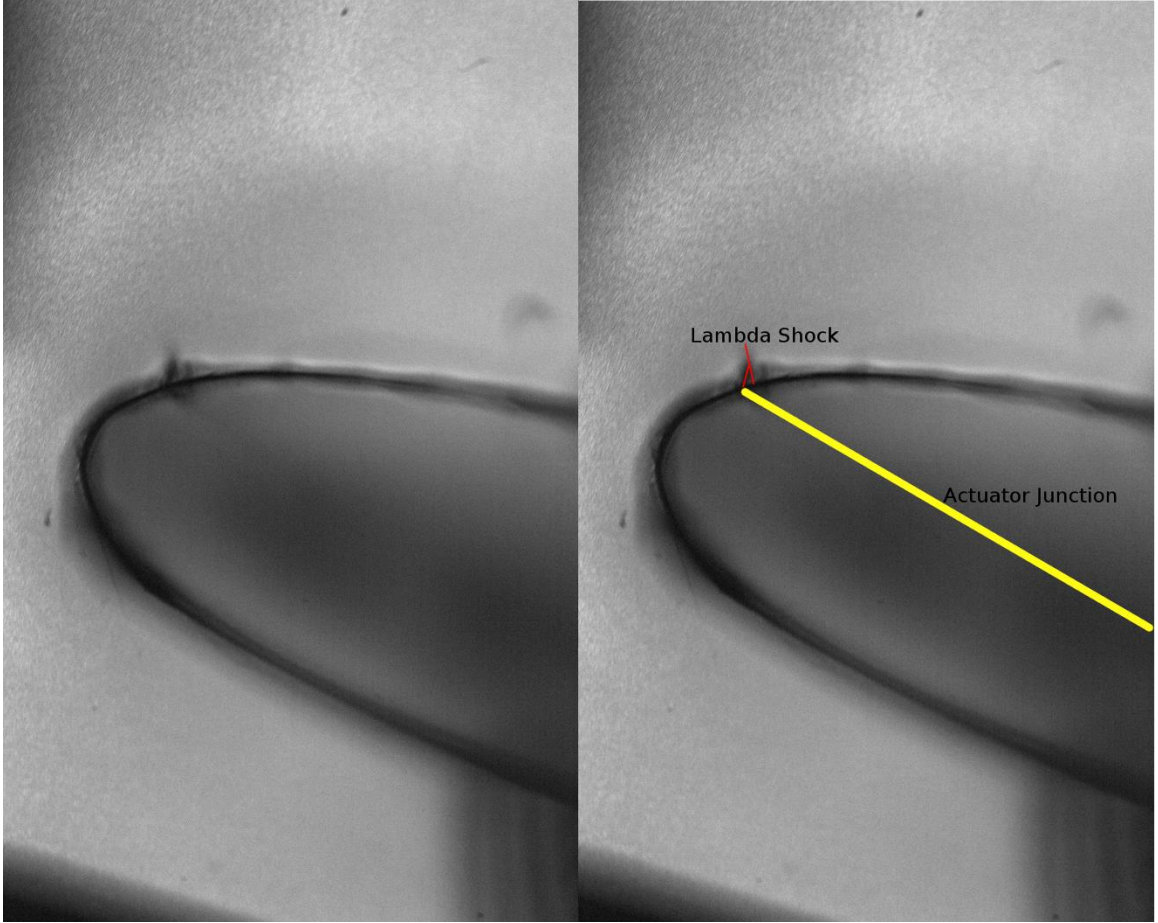


Figure 34: Shadowgraph image of lambda shock occurring at $\alpha = 11.5^\circ$ at Mach 0.4 with a $Re = 2.2 \times 10^6$ with no actuation.

A direct comparison can be made between the actuation and non-actuation case during one cycle by viewing the images side by side. Figure 35 through Figure 38 show the non-actuation shadowgraph images aside the actuation images at Mach 0.4.

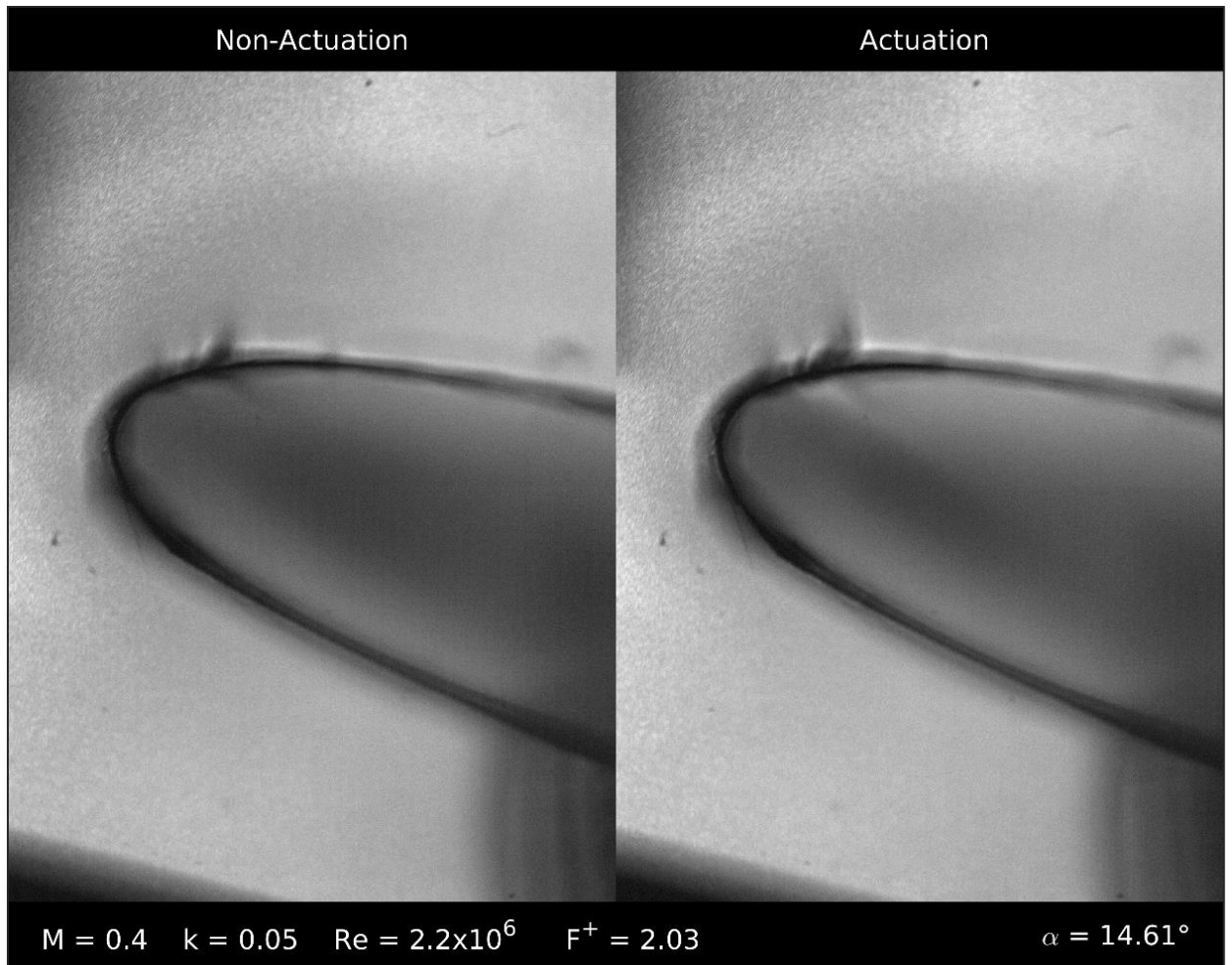


Figure 35: Shadowgraph Mach 0.4, $Re = 2.2 \times 10^6$, $\alpha = 14.61^\circ$, (left) non-actuation, (right) actuation

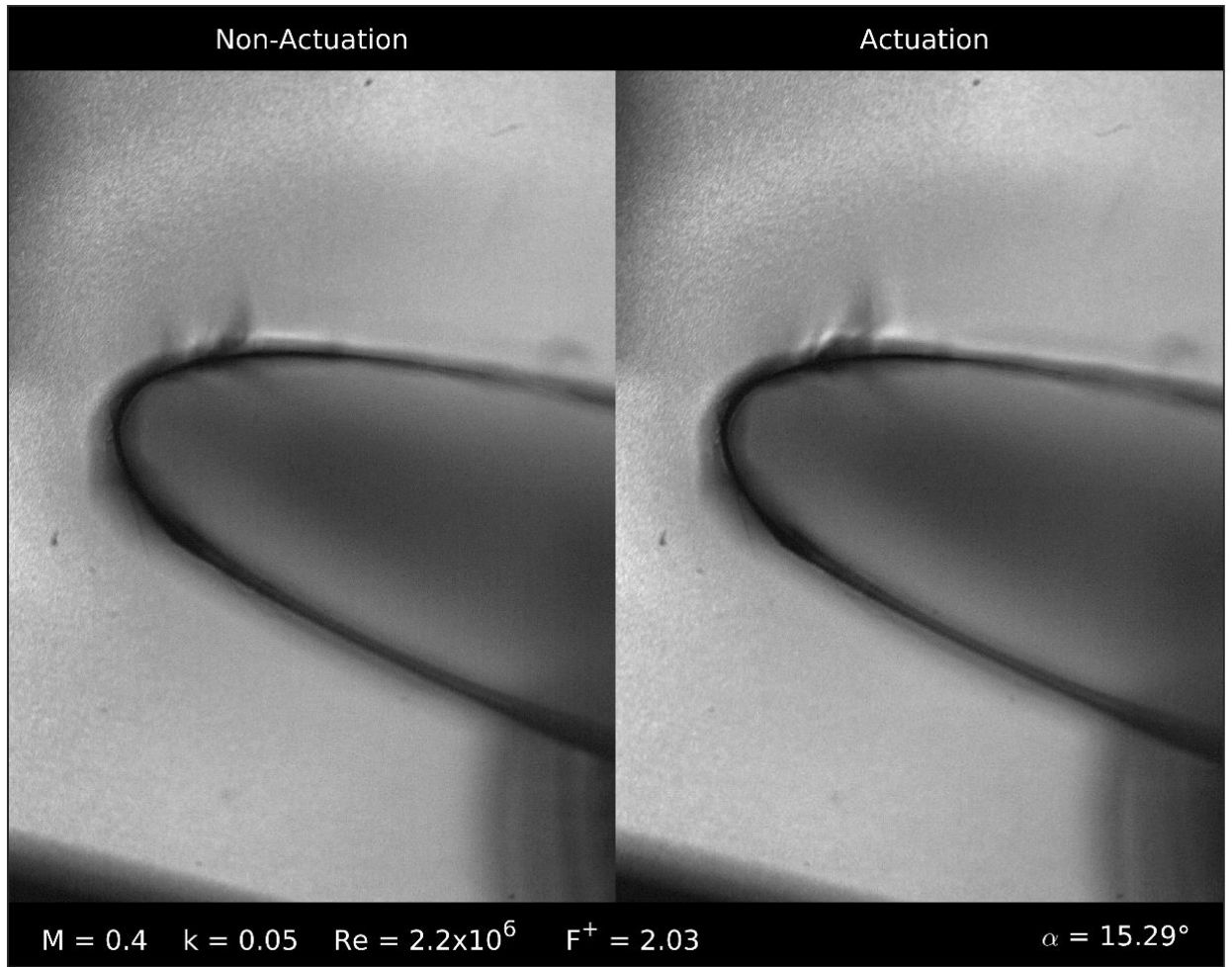


Figure 36: Shadowgraph Mach 0.4, $Re = 2.2 \times 10^6$, $\alpha = 15.29^\circ$, (left) non-actuation, (right) actuation

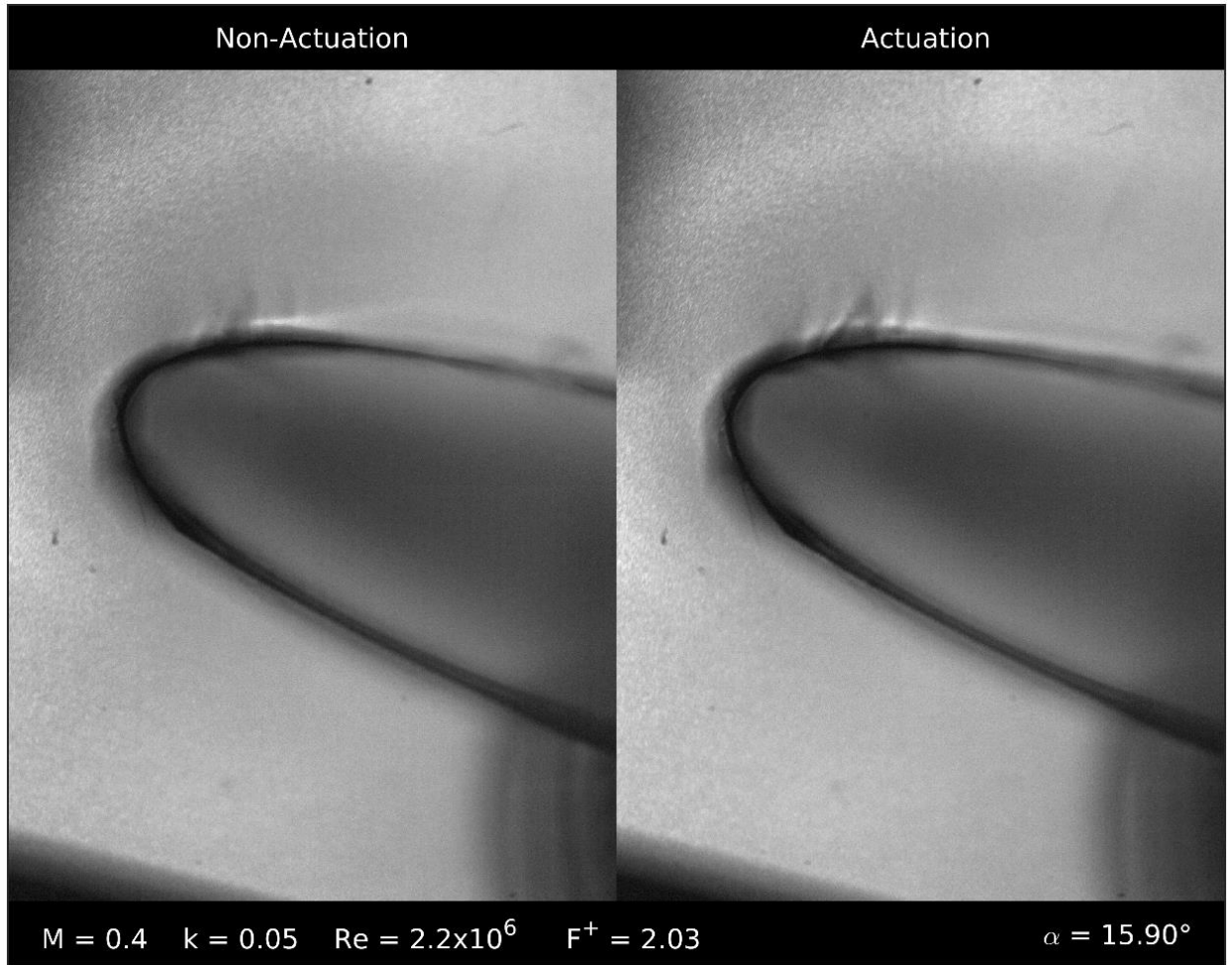


Figure 37 Shadowgraph Mach 0.4, $Re = 2.2 \times 10^6$, $\alpha = 15.90^\circ$, (left) non-actuation, (right) actuation

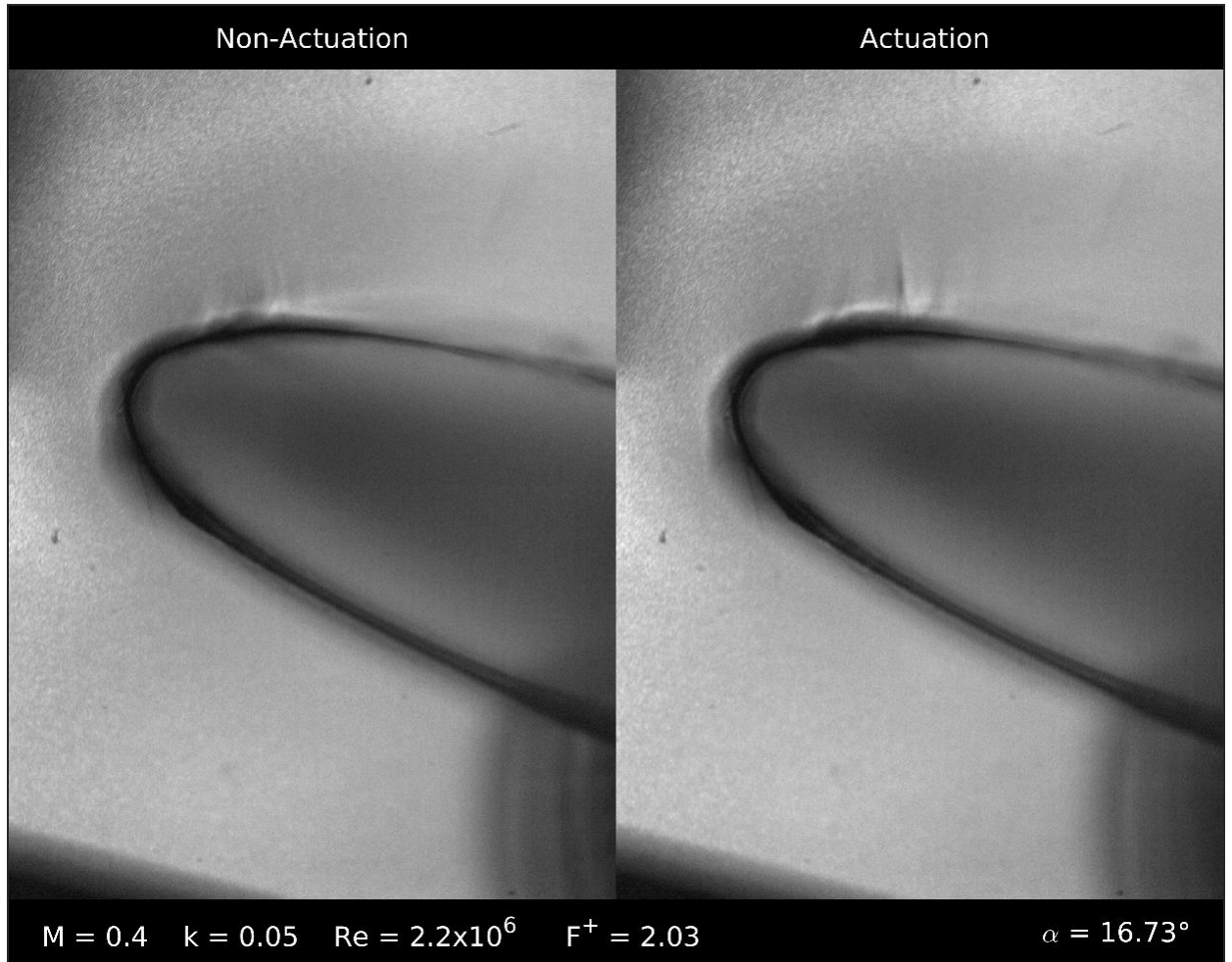


Figure 38: Shadowgraph Mach 0.4, $Re = 2.2 \times 10^6$, $\alpha = 16.73^\circ$, (left) non-actuation, (right) actuation

As the angle of attack increases the lambda shock becomes larger indicating a stronger shock. In Figure 35 at $\alpha = 14.61^\circ$ a shock appears to be “fixed” at the actuator junction, however there appears to be one or two smaller shocks in front of the actuator junction. The appearance of multiple shocks is not uncommon, and agrees with the finds of Beahan *et al.*¹⁹.

As the pitch cycle continues, at $\alpha = 15.29^\circ$, the shock has become more defined. At $\alpha = 15.90^\circ$ the actuation case has a strong lambda shock with a foot anchored on the actuator junction. Both in front and behind the shock, there appears to be multiple shock waves. In the non-actuation case, at $\alpha = 15.90^\circ$ (Figure 37) the shear layer is clearly visible now that it has lifted sufficiently far enough from the surface and is evident by the white streak that extends from the leading-edge over the suction surface. As the angle of attack is increased to 16.73° , multiple shocks are present in both cases with one anchored at the actuator junction. The shear layer is visible in both the actuation and non-actuation case.

One feature that is not depicted in the few images presented in this document is the unsteadiness of the shocks that appear near the leading edge. Progressing through the individual images shows that the first shock that appears on the surface occurs at the actuator junction. As the angle of attack is increased, lambda shocks develop and fluctuate in strength and location. However the presence of a shock located at the actuator junction is common in each individual image. Additional shadowgraph images with the presence of shocks on the suction surface are provided in Appendix E.

The lambda shock anchored at the actuator junction is understood to be a direct result of the application of the exposed high-voltage electrode. At the junction, the copper used as the exposed electrode has a thickness of 0.09mm. This step (forward facing step) results in an unfavorable pressure gradient that in itself could result in the generation of a shock or flow separation. At Mach 0.4, the flow acceleration round the leading edge results in supersonic flow. The abrupt change in thickness from the high-voltage

electrode results in a lambda shock that has the potential to create shock-induced flow separation. As indicated by historical studies the optimum location for plasma actuation is slightly upstream of the separation region. At a slightly advanced upstream location, minimal power is needed because the flow is most susceptible to the excitation immediately prior to separation. With the shock and possible shock-induced flow separation occurring at the point of excitation, the flow is less susceptible to perturbations generated by the actuator. In particular, the NS-DBD actuation generates thermal expansion waves (localized shocks) that disturb the flow. The ability for the thermal expansion waves to disturb the flow is diminished with the presence of a shock located in the same proximity. To achieve complete flow control, the power requirements to generate a strong enough expansion wave from the plasma excitation was not achievable with this experimental setup.

Chapter 4: Conclusion

Chapter 4.1: Summary of Experimental Results

Experiments were performed using NS-DBD plasma actuators on a NACA 0015 airfoil installed in an unsteady transonic blowdown wind tunnel. The airfoil was subjected to sinusoidal pitch oscillations at rotorcraft relevant reduced frequency ($k = 0.05$) at Mach 0.2 and 0.4. The tests were conducted in high pressure and Reynolds number flow that are a result of the blowdown facility. Plasma actuation ranged over F^+ values from 0.78 to 6.06 for Mach 0.2 and 0.39 to 2.03 for Mach 0.4. At these high Reynolds numbers and pressures, the actuation was able to demonstrate flow control authority.

At Mach 0.2, the shedding of the dynamic stall vortex was altered, resulting in a weaker vortex that advected earlier at a reduced convection speed compared to the non-actuation case. The severity of the lift (Γ) and moment (ζ) stall excursions were significantly reduced with NS-DBD actuation. This resulted in a benefit in the post-stall region with higher surface pressures and elimination of secondary vortices. The removal of the secondary vortex reduced the negative damping in the moment hysteresis. As a result of the high surface pressures during the post-stall regime, the pressure recovery occurred earlier in both the phase and angle of attack schedule. The optimum non-dimensional forcing frequency (F^+) was found to be approximately unity for both static and dynamically pitching angles of attack.

Increasing the Mach number to 0.4 had different results from those observed in the Mach 0.2 study. By increasing the Mach number, the Reynolds number was increased by 1 million and compressibility effects were introduced. Past studies have noted that shock waves naturally develop within the first 10% of chord on a NACA 0015 at Mach 0.4 with Reynolds numbers greater than 1.5 million. The presence of shocks may reduce the effectiveness of the actuation at developing coherent structures to influence stall with a dynamically pitching airfoil. The optimum actuator location requires plasma generation to occur directly upstream of separation. In the shadowgraph studies, the presence of a lambda shock anchored at the actuator junction may result in the inability to provide the correct amount of excitation to achieve flow control. Under the right frequency of actuation, the dynamic stall vortex was weakened and shed earlier in the angle of attack schedule. This resulted in a slightly improved pressure recovery on the downstroke of the pitch cycle.

Chapter 4.2: Recommendations for Future Experiments

A possible area to improve this experiment is with the actuator application. One easy alteration would be to move the actuator to a new location rather than having the actuator junction at 4% chord for both Mach 0.2 and 0.4. Aside from moving the actuator, the physical geometry (widths of the grounded and high-voltage electrode) of the actuator could be adjusted. Increasing the widths of the electrodes can decrease the

resistance and the loading across the actuator, allowing for higher voltages to be attained. This would require extensive testing to determine the actuator location. However, a new actuator location for each test condition is unrealistic as this is not possible on an actual rotor system.

As shown by the shadowgraph images, the surface mounted actuator results in additional thickness and a forward facing step at the actuator junction. A consequence of the surface mounted actuator is the high-voltage electrode can act as a trip, and lead to the development of shocks at Mach 0.4. If a recess were to be added to the surface and the actuators were flush mounted a nearly smooth surface would be created, and potentially prevent lambda shocks and flow separation.

To more accurately model current rotorcraft systems, future experimental studies could implement a rotorcraft relevant airfoil. For this experiment, funding for a new composite airfoil was outside of the project budget. The NACA 0015 airfoil that was used in the experiment was the best test article candidate available. Although many previous NS-DBD studies have been conducted using a NACA 0015 airfoil, a rotorcraft relevant airfoil would be more suited for this dynamic pitching experiment.

The last improvement to the current study would be to operate the tunnel in couple pitch and Mach oscillations. Coupled oscillations capture both the dynamic pitching motion ($\partial\alpha/\partial t$) as well as the dynamic freestream velocity component ($\partial M/\partial t$). This would be a more accurate representation of a helicopter in forward flight.

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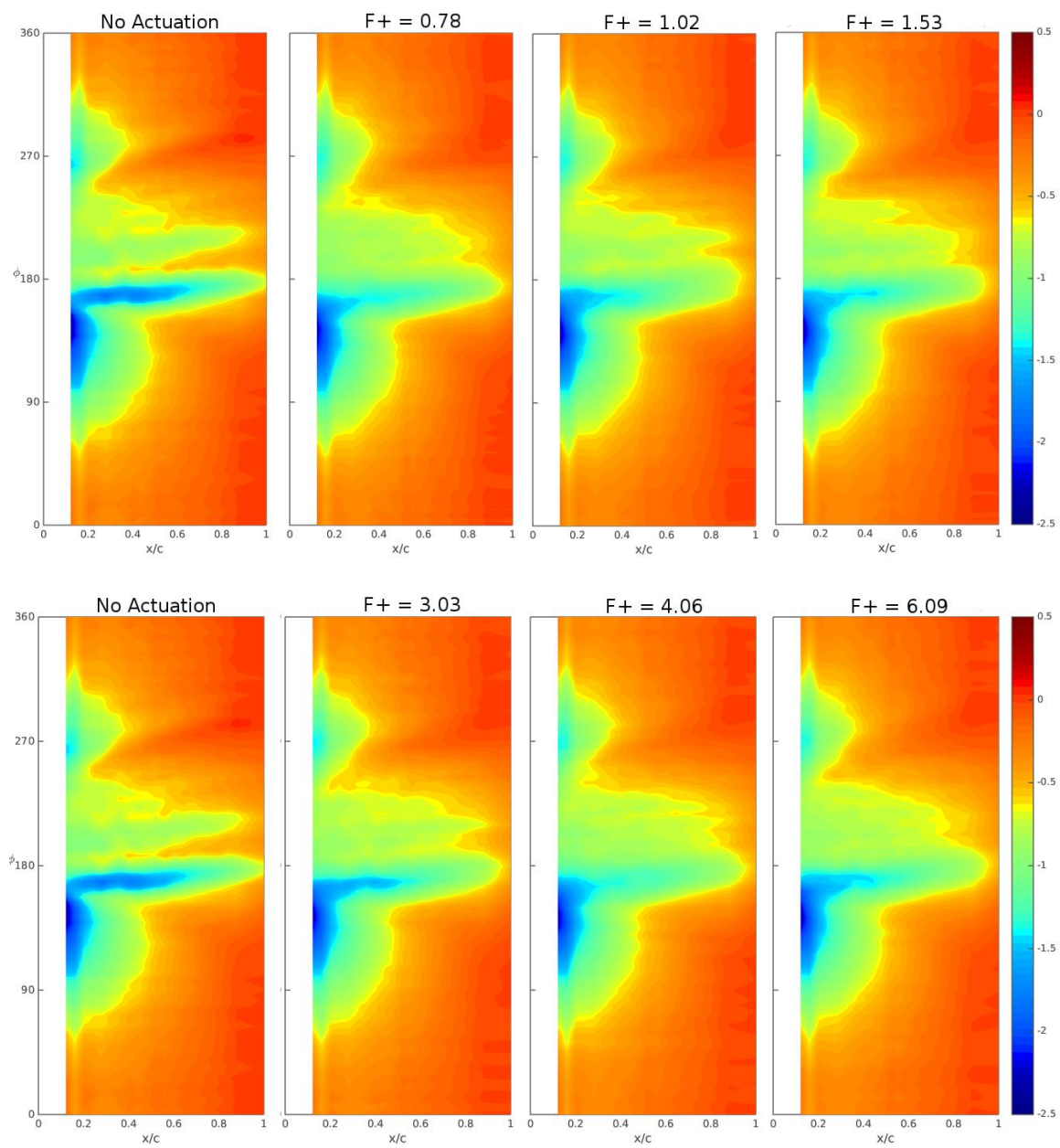
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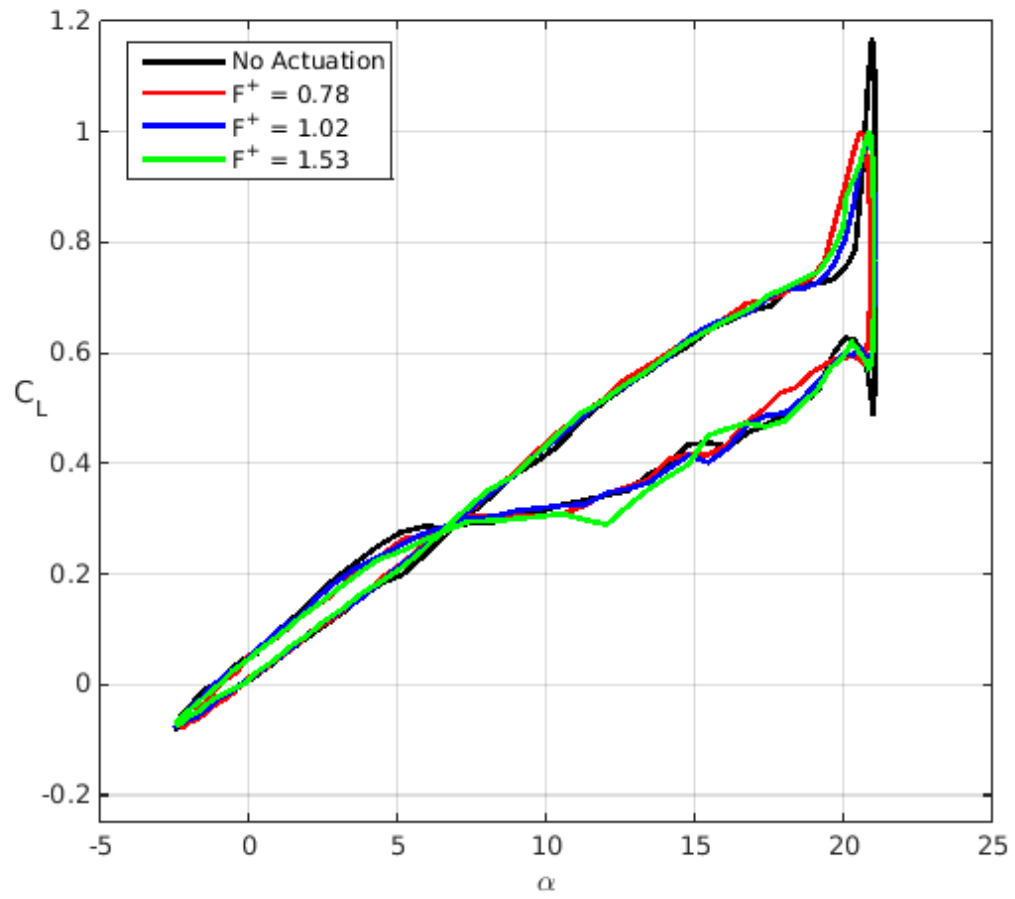
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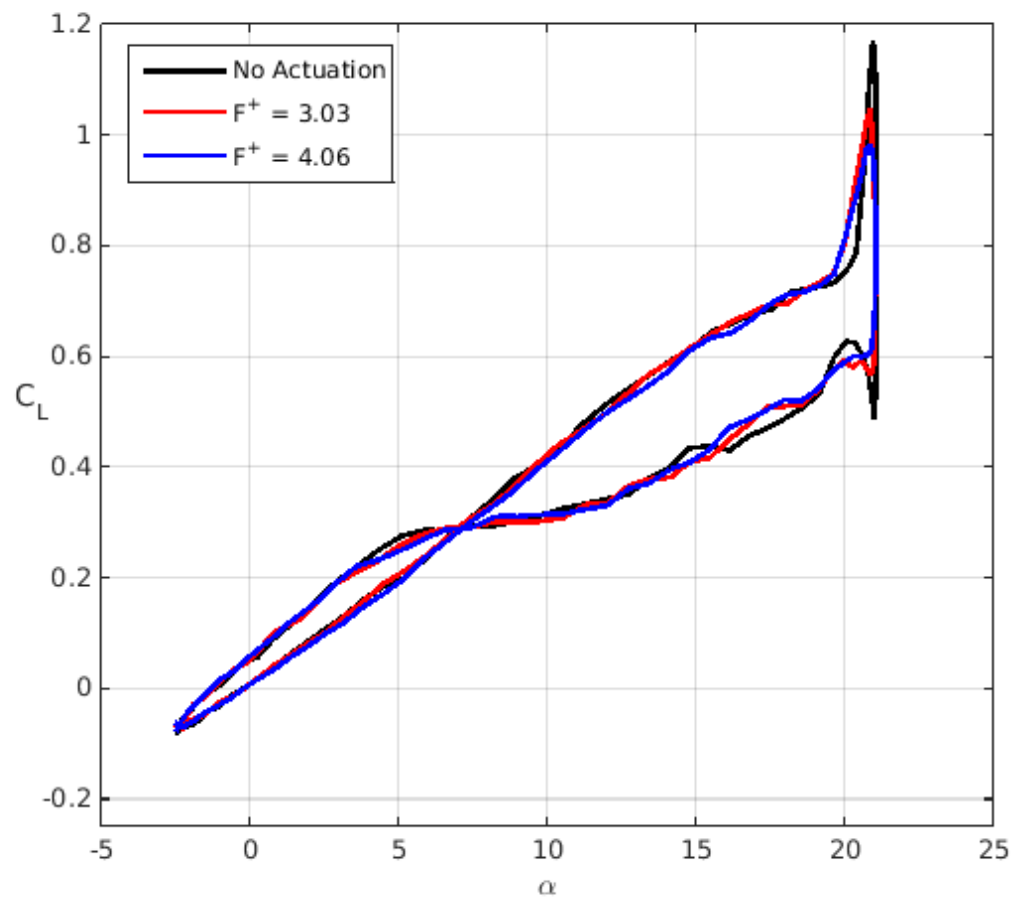
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Appendix A: Mach 0.2 C_p Contours

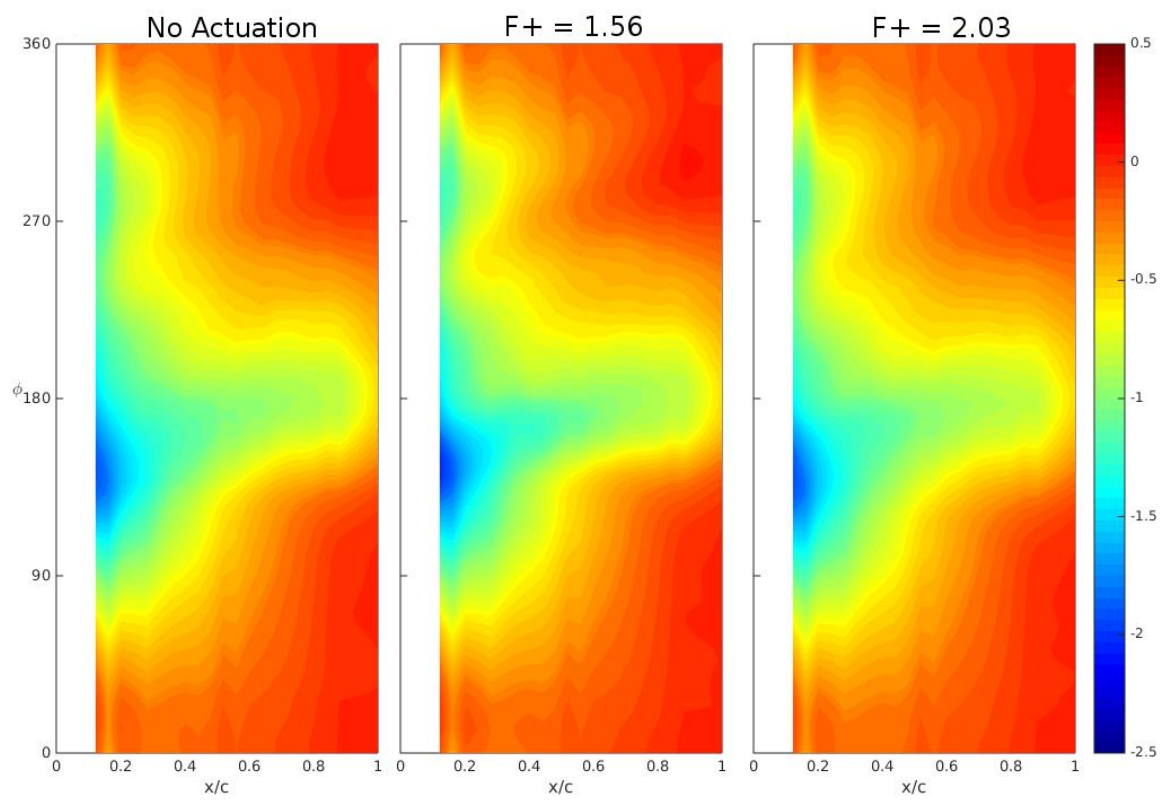
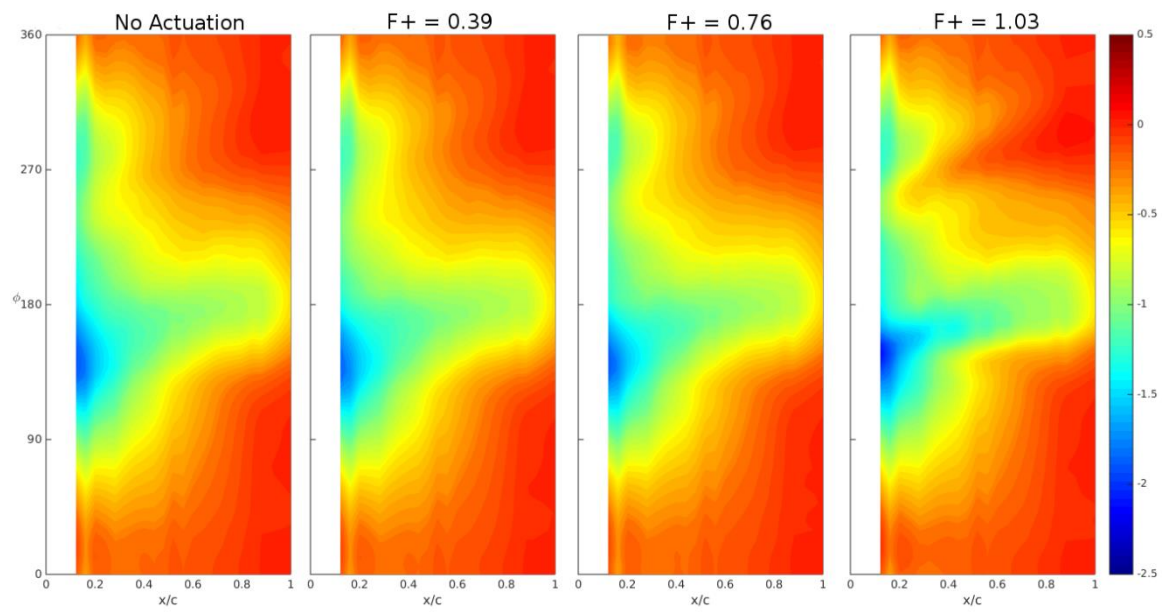


Appendix B: Mach 0.2 C_L Hysteresis Loops

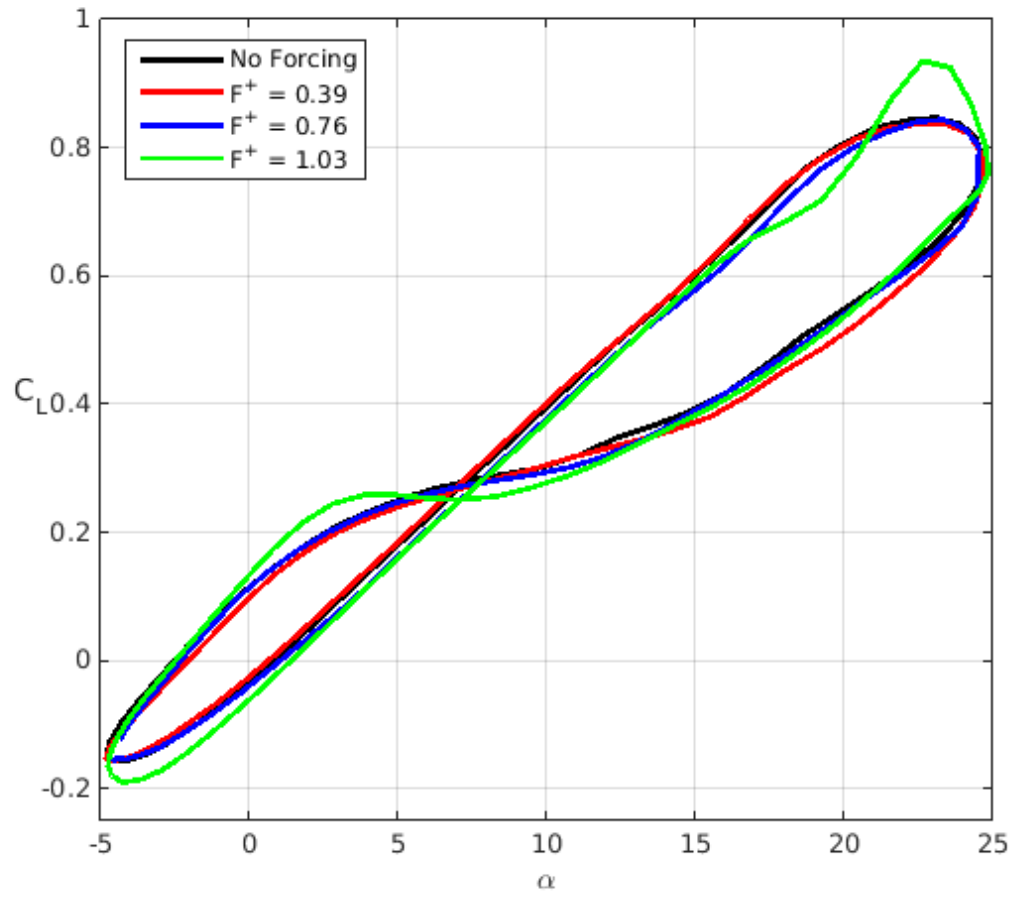


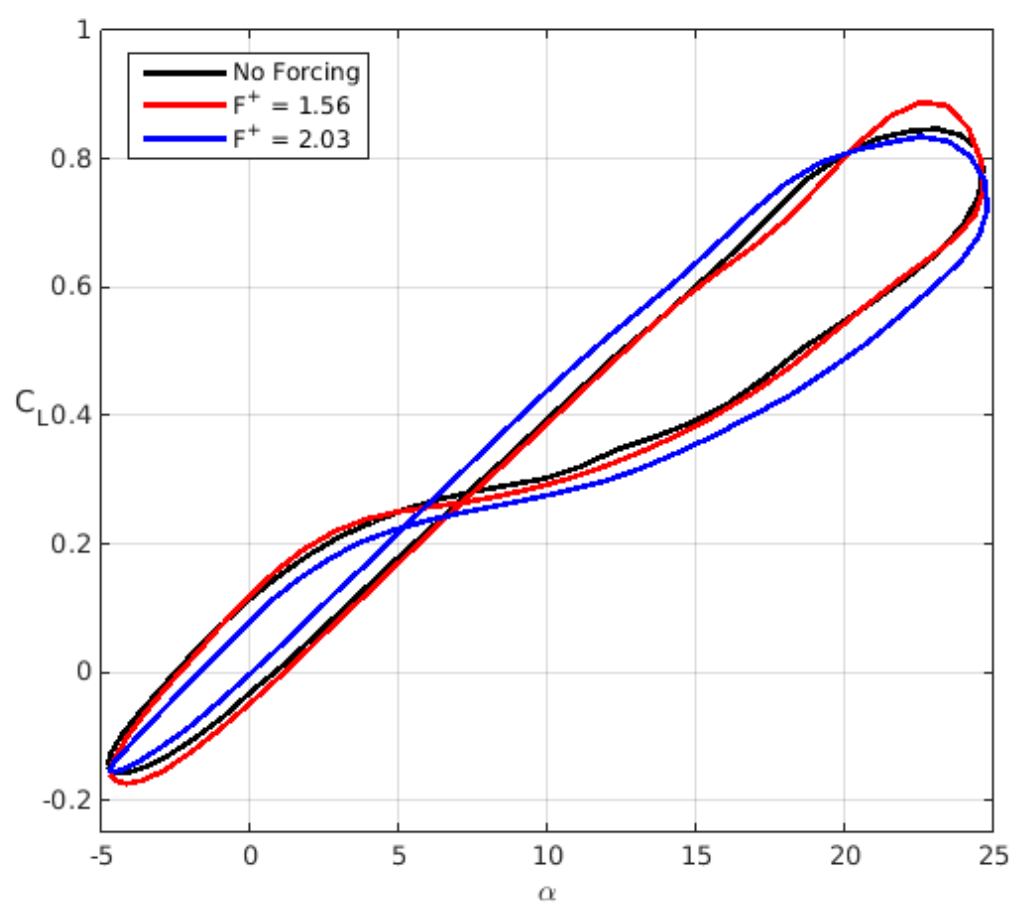


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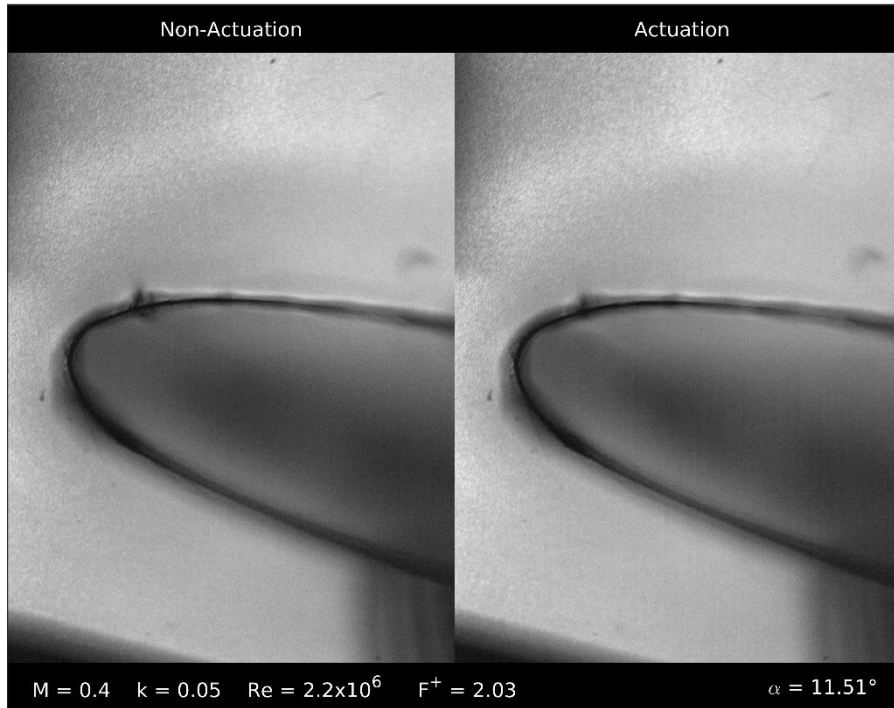


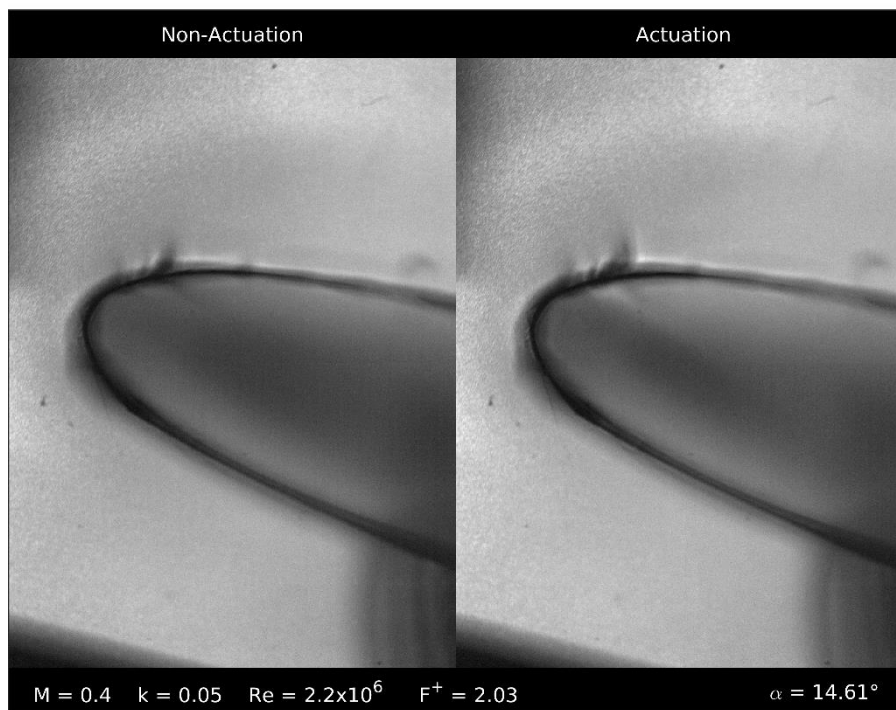
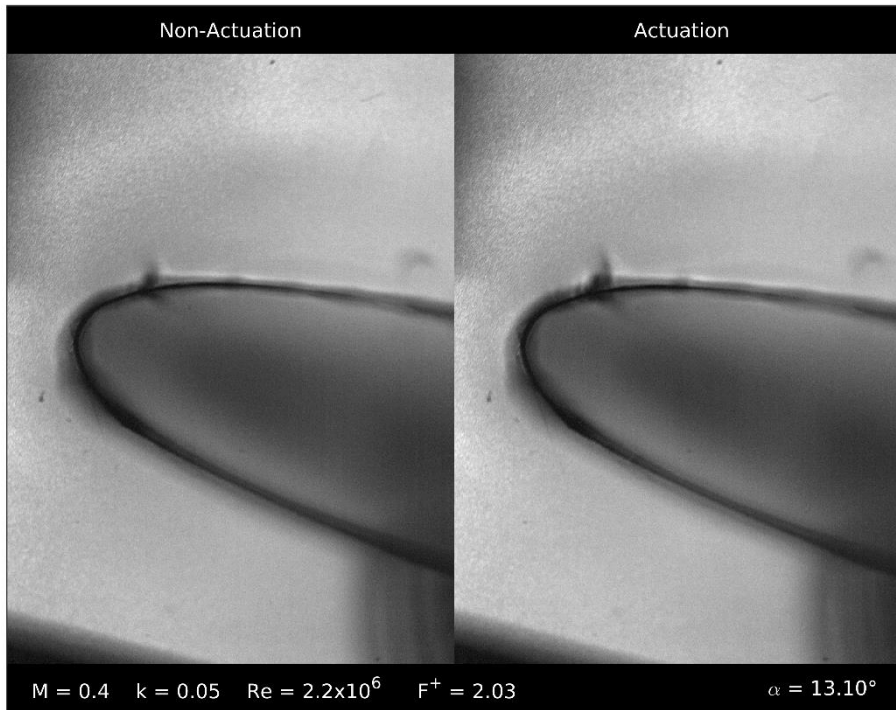
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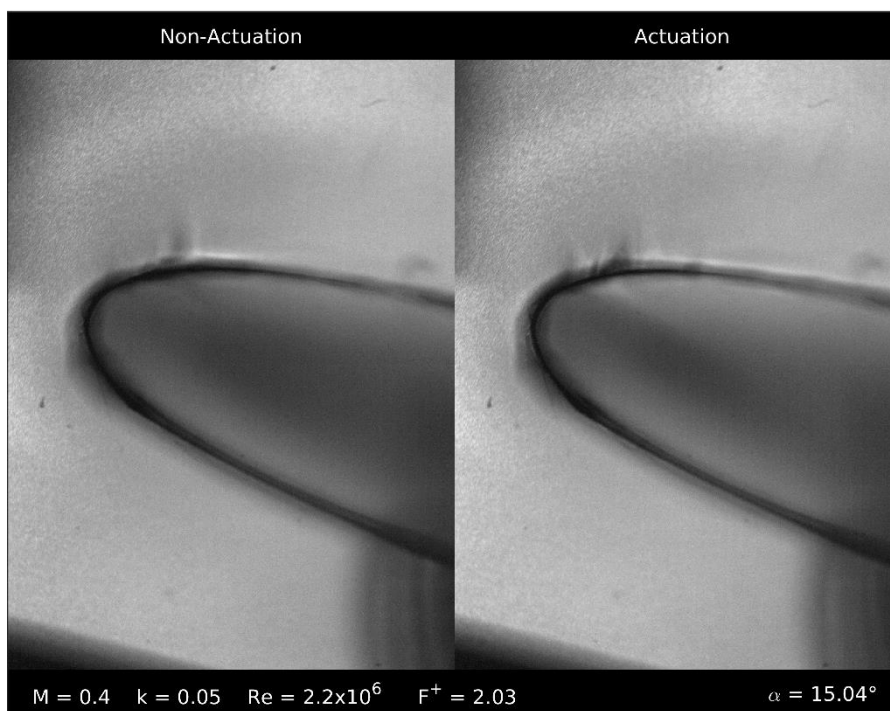
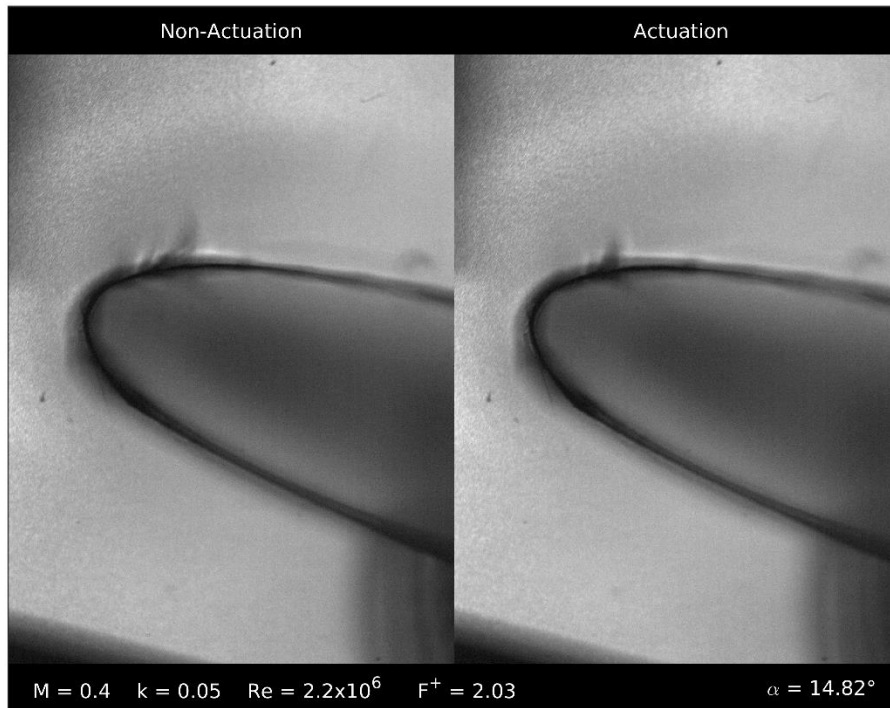


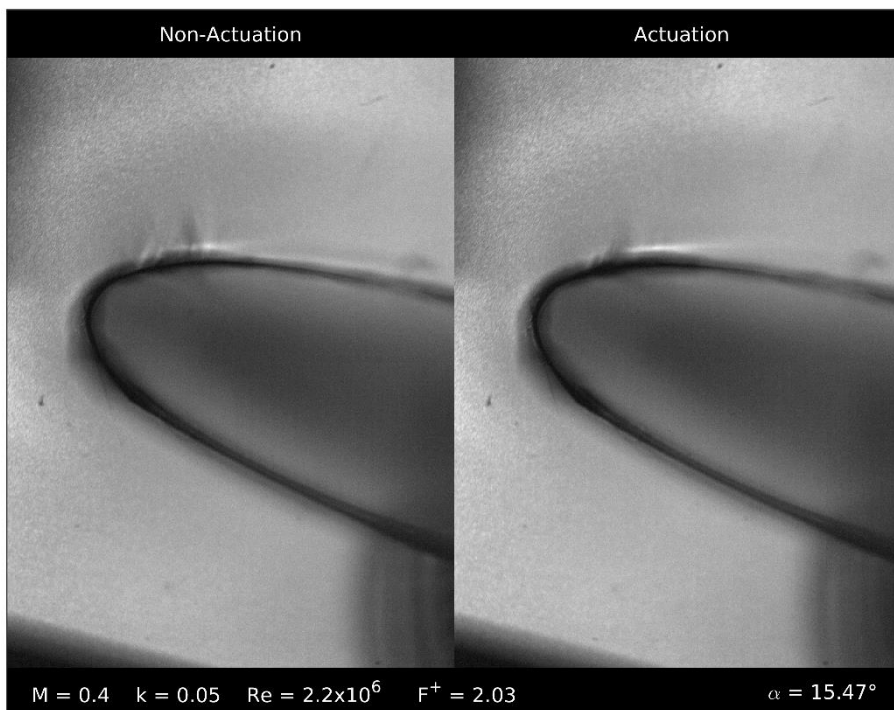
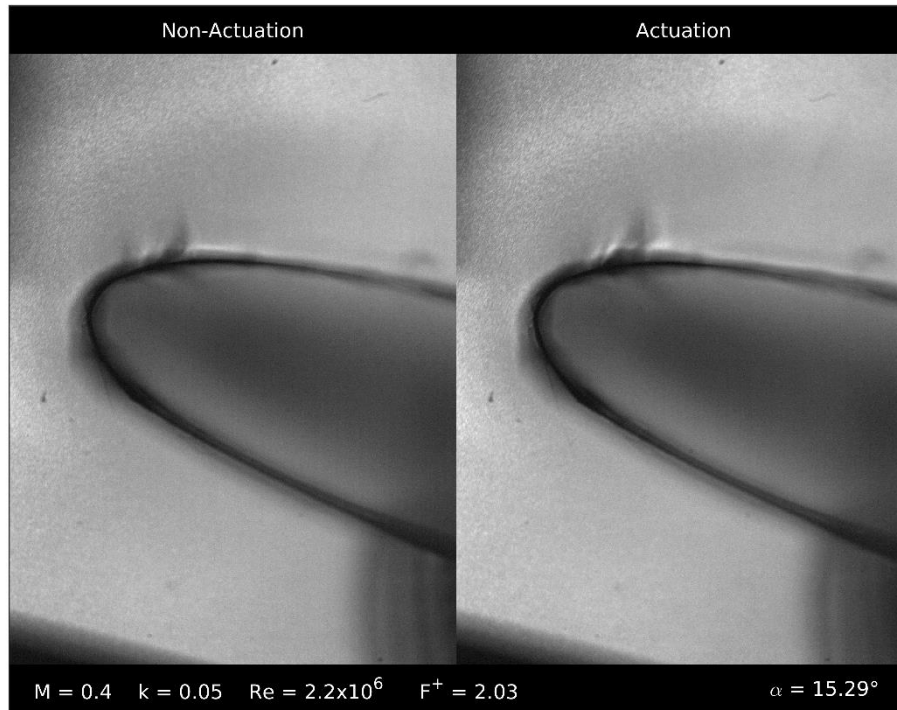


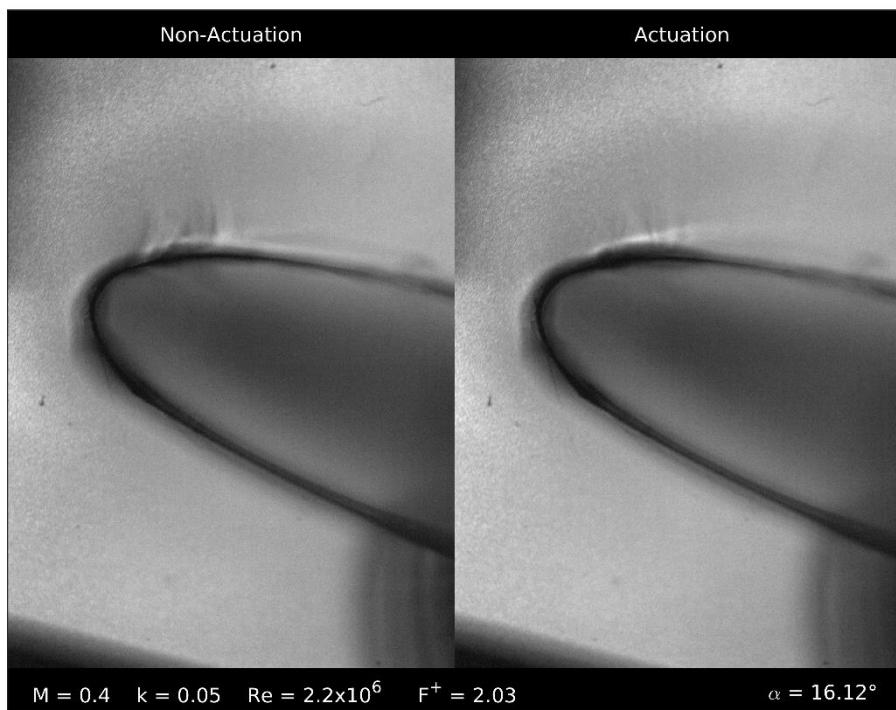
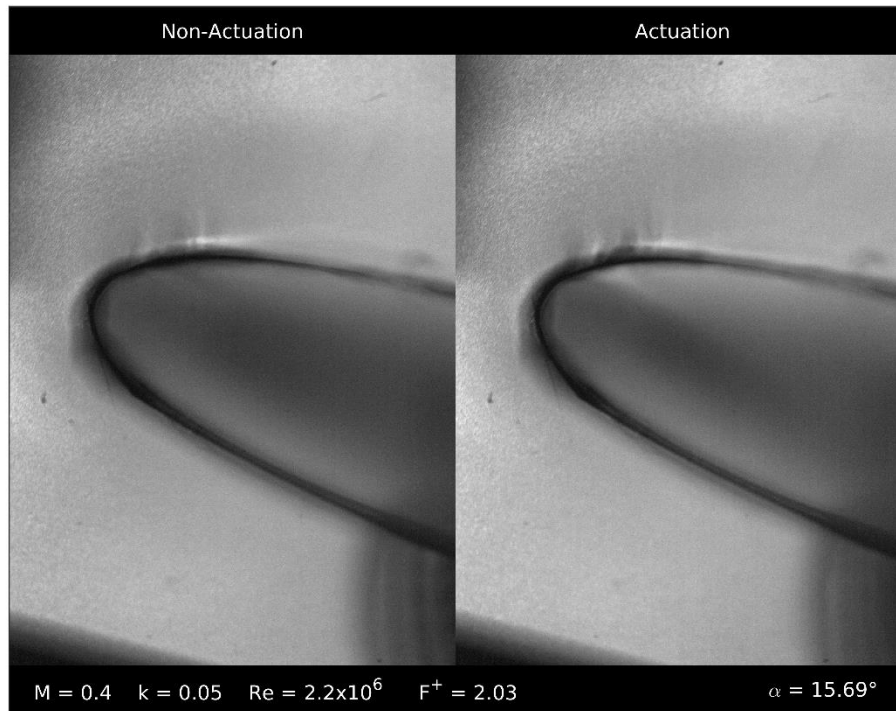
Appendix E: Shadowgraph Images

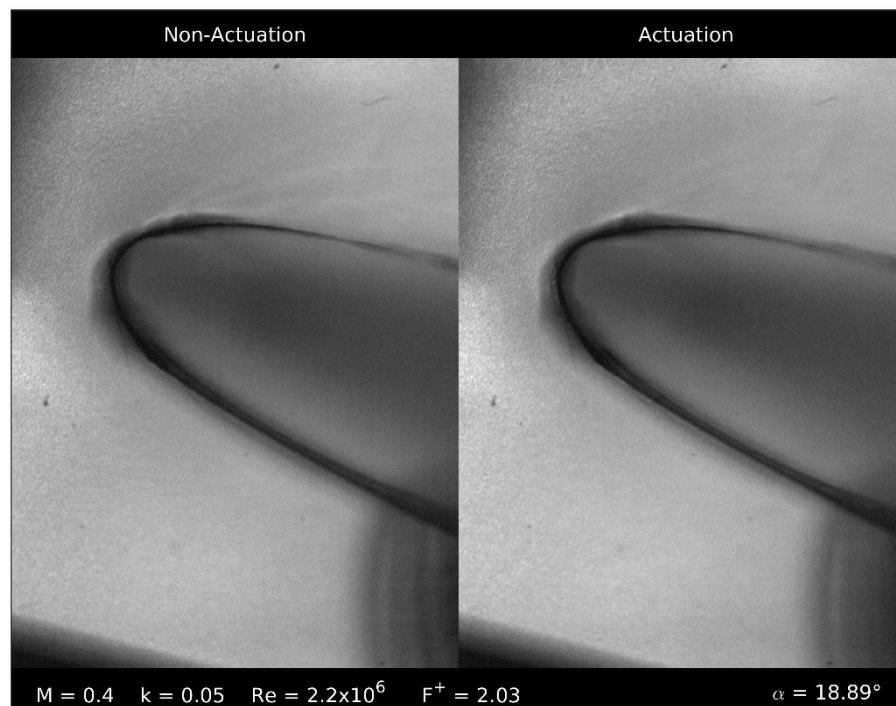
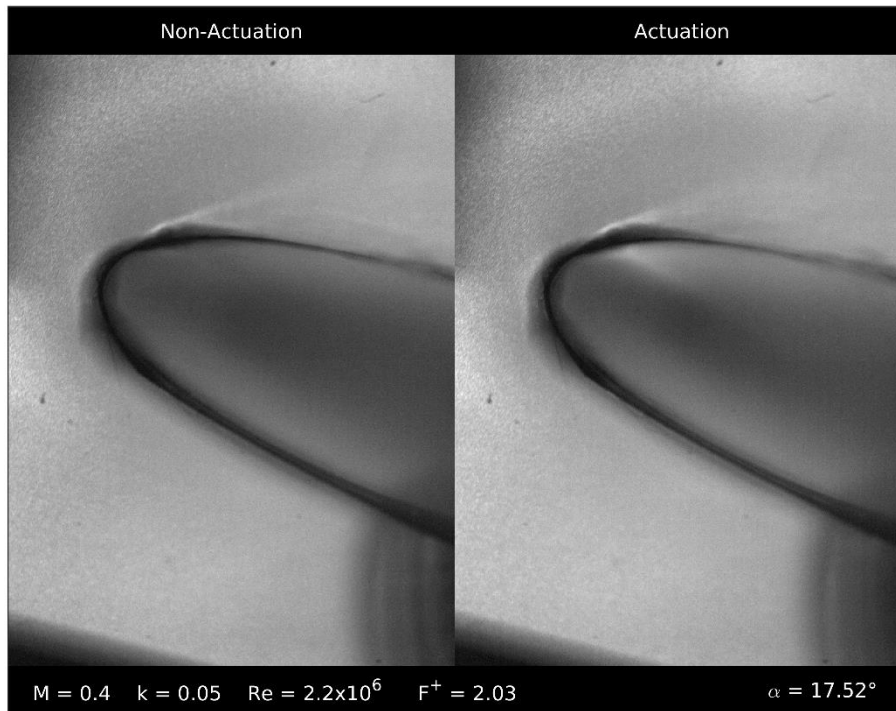


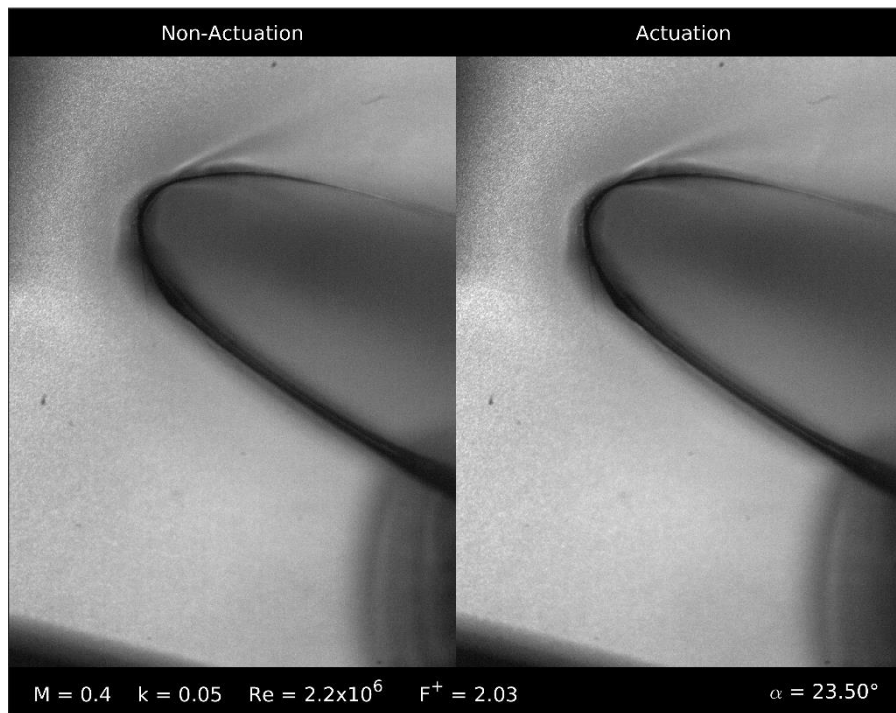












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