AIRFOIL LEADING AND TRAILING EDGE SEPARATION CONTROL
USING SDBD PLASMA ACTUATORS

A Dissertation

Submitted to the Graduate School
of the University of Notre Dame
in Partial Fulfillment of the Requirements
for the Degree of

Doctor of Philosophy

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December 2013
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Abstract

by
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Wind tunnel experiments were conducted to understand boundary layer separation control on airfoils. The objective of this research was to better implement flow control strategies with single dielectric barrier discharge (SDBD) plasma actuator designs for airfoils that exhibit both leading and trailing edge stall characteristics at realistic flight speeds and Reynolds numbers. The NASA energy efficient transport (EET) airfoil was a platform to study the effect of plasma actuators on leading edge stall and boundary layer separation in a strong adverse pressure gradient, while the V-22 Osprey airfoil was used to study trailing edge stall with separation in a weaker adverse pressure gradient.

The EET airfoil was designed to have a spanwise plasma actuator on removable leading edges made from two different dielectric materials: Kapton and Macor. Two different plasma waveforms were also tested with the same electrodes, alternating current (AC) and nanosecond pulse (NP) driven. Aerodynamic force and moment measurements showed that both plasma actuators were effective at increasing the stall angle of attack and maximum lift for the range of Mach numbers tested, 0.1–0.4, and Reynolds numbers of 560,000–2,240,000. This indicated that the shear layer instability was highly receptive to both disturbances: either the body force from AC plasma actuator, or the nondirectional thermal disturbance of the NP plasma
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actuator. The shear layer instability also provided for an opportunity to quantify the effect of unsteady, or duty cycle, operation. The lift to drag ratio of the EET airfoil was improved the most by operating the AC plasma actuator at a reduced frequency of unity and the NP plasma actuator at 2 or higher.

The second part of the experiment examined the efficacy of plasma actuators on a V-22 airfoil for trailing edge separation control in the presence of new factors such as moving separation location, crossflow, turbulent boundary layers, and weaker pressure gradients at the line of vanishing shear. Initial consideration of moving separation location with angle of attack motivates the use of plasma streamwise vortex generators (PSVGs) which take up a larger percent of the chord dimension and produce streamwise vorticity from both crossflow momentum addition and by reorienting spanwise vorticity from the boundary layer. The PSVGs were compared to traditional passive vortex generators (VGs). These devices were installed on the wing section by which the angle of attack could be used to vary the streamwise pressure gradient. The experiment was performed for freestream Mach numbers 0.1–0.2 and a Reynolds number range of 790,000–1,590,000. Three-dimensional velocity components were measured using a 5-hole Pitot probe in the boundary layer. These measurements were used to quantify the production of streamwise vorticity and the magnitude of the reorientation term from the vorticity transport equation. Reductions in drag were well correlated to streamwise vorticity production. For the PSVG, vorticity production was proportional to the residence timescale of freestream momentum and operating voltage. These results indicate that the PSVGs could outperform the passive VGs and provide a suitable alternative for flow control. Finally, a design equation was proposed to create a PSVG equivalent to a VG including design parameters such as Mach number, angle of attack, operating voltage, and electrode length.
To my grandparents Norma, Ralph, Ruth, and Bob
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ACKNOWLEDGMENTS

Graduate school has been a time of personal growth thanks to numerous people who have made the process of getting a Ph.D. challenging, exciting, and fulfilling.

I would like to thank my adviser, Professor Corke, for the opportunity to do research that was genuinely fun and interesting and for your problem solving expertise along the way. Professors Atassi, Nelson, and Dunn, I really valued your opportunities to lecture for your classes because I found out that I really enjoy teaching. And thanks to the other AME faculty at Notre Dame for the graduate courses which taught me new and relevant material important to my understanding of aerospace engineering. I would also like to thank my professors I had at UIUC because I felt well prepared to start graduate school thanks to the quality of your aerospace undergraduate program. I would also like to thank Dr. Sutter and Mr. Gebhard for giving me a passion for science through your teaching in high school with Science Olympiad and memorable physics experiments.

Thanks Professor Pat Dunn for being one of the coolest professors around. Whether discussing Arduino projects or the homogeneity of coffee grounds, I found a friend and unofficial mentor at Notre Dame who valued personal interactions and technical rigor in healthy balance.

Thanks Rob Chlebek for being a huge help in setting up my experiments, calling the power plant for more power, and for your mechanical know-how.

I have great friends that made graduate school more enjoyable. I would like to thank Rick Rateick for encouraging me to become interested in graduate school. This turned out to be great foresight. Thanks Joanna Ho for being such a fun person to
talk to when there were inevitably so many roadblocks as we conducted research. Thanks Jessica Cobb for encouraging creativity as an important aspect of engineering. Thanks for making Hessert Lab fun matlab guru Brian Neiswander, lunch organizer Grady Crahan, and fellow cyclist Mike Waldrop. I also value friends such as Dustin Cain and Adam Coffman who have remained in contact with me from high school and supported me remotely. Thanks Dr. Sawyer Campbell for being my hilarious “brother in science.”

Above all, I would like to thank my parents for your love and support. You are infinitely more important than this piece of paper and I love you very much.
SYMBOLS

English

\( a_\infty \) freestream speed of sound, m/s

\( b \) span of wing, m

\( c \) airfoil chord, m

\( C_d \) section drag coefficient

\( c_f \) skin friction coefficient

\( C_l \) section lift coefficient

\( c_{l_{\text{max}}} \) lift coefficient at stall

\( C_{l_\alpha} \) section lift slope

\( C_{m_{c/4}} \) moment coefficient about quarter-chord

\( C_p \) pressure coefficient

\( f \) unsteady actuator frequency, Hz

\( f^+ \) reduced actuator frequency, \( fL_s/U_\infty \)

\( H \) boundary layer shape factor, \( H = \delta^*/\theta \)

\( l \) length of PSVG electrode parallel to freestream, m

\( L \) total edge length of electrode along which plasma forms, m

\( L_s \) characteristic length of separated flow region, m

\( M \) Mach number, \( \frac{U_\infty}{a_\infty} \)

\( p \) static pressure, Pa

\( p_0 \) stagnation pressure, Pa

\( Re \) Reynolds number, \( U_\infty c/\nu \)

\( S \) wing planform area, m²
\( T_P \)  vorticity production time scale, \( \frac{\delta^*}{V_P} \), s

\( T_{Res} \)  residence time scale, \( \frac{L}{U_\infty} \), s

\( T_{VR} \)  vorticity reorientation time scale, \( \frac{\delta^*}{u_T} \), s

\( U_\infty \)  freestream velocity, m/s

\( u' \)  standard deviation of freestream velocity, m/s

\( u_T \)  shear (friction) velocity, \( \sqrt{\frac{\tau_{wall}}{\rho}} \), m/s

\( V_p \)  peak induced velocity from plasma actuator, m/s

\( V_{p-p} \)  peak-to-peak potential of plasma actuator signal, V

\( x \)  streamwise coordinate along chord line, m

\( y \)  airfoil surface normal coordinate, m

\( z \)  crosstream coordinate along span, m

**Greek**

\( \alpha \)  angle of attack, \( ^\circ \)

\( \alpha_{0L} \)  zero lift angle of attack, \( ^\circ \)

\( \alpha_s \)  stall angle of attack, \( ^\circ \)

\( \delta \)  boundary layer thickness, m

\( \delta^* \)  displacement thickness, m

\( \theta \)  momentum thickness, m

\( \lambda \)  PSVG inter-electrode spacing, m

\( \Lambda \)  Wing sweep, negative is forward swept, \( ^\circ \)

\( \mu \)  dynamic viscosity, kg/(m·s)

\( \nu \)  kinematic viscosity, \( \frac{\mu}{\rho} \), m\(^2\)/s

\( \rho \)  fluid density, kg/m\(^3\)

\( \tau_{wall} \)  shear stress at wall, \( \mu \frac{\partial U}{\partial y} \bigg|_{y=0} \), Pa
Abbreviations

EET  energy efficient transport
NP   nanosecond pulse
PSVG plasma streamwise vortex generator
SDBD single dielectric barrier discharge
VG   vortex generator
CHAPTER 1

INTRODUCTION

1.1 Motivation

Often in fluid mechanics, a flow is decelerated due to viscosity and an adverse pressure gradient and ceases to propagate parallel to a solid surface. When this separation occurs, whether from a cylinder, a flat plate, an airfoil, fuselage ramps, or other abrupt changes in geometry, there are associated losses such as reduced pressure recovery, increased drag, and decreased lift. Due to the wide range of mechanical systems operating in the Earth’s atmosphere, these losses represent increased fuel burn for aircraft and ground vehicles, decreased energy production from wind turbines, greater chemical and sound pollution from turbomachinery, and ultimately less efficient operation of a great deal of machines important to our everyday lives. It should be emphasized though that separation is not always undesirable. For example, one can think of situations where separation is advantageous such as overspeed protection of a wind turbine or an aircraft increasing drag on approach to land. Therefore, boundary layer separation control should be approached from an application specific strategy deciding whether flow control should be used to cause reattachment or separation.

The history of flow control is as old as powered, heavier than air flight with the foundational work on boundary layers by Prandtl in 1904 [49] in which separation control was also demonstrated. Many experiments with steady suction or blowing type flow control have been successfully flight tested since the 1920’s highlighted by Lachmann [25]. And yet still today it seems very novel that a small hydrodynamic
disturbance can be amplified by nature to have a global affect on the fluid flow, pressure distribution, and the associated aerodynamic forces and moments. Nowhere is this more evident, as will be shown in the present paper, than in a stalled airfoil regime where a small addition of momentum into the boundary layer at the leading edge of an airfoil can increase the lift coefficient by 0.9 and the stall angle by 5 degrees.

Plasma actuators have shown promise as a separation control device, replacing the complex and heavy mechanical systems of many other active flow control technologies. But SDBDs are not high momentum devices and are not well suited to be thrusters as common induced velocities are on the order of 5 m/s. So it is natural to question how can these devices have any measurable changes on global performance such as lift, drag, and moment at freestream velocities 25 times faster? Leading up to this research it remains an open question: do SDBD plasma actuators act as effective separation control devices on airfoils at subsonic freestream velocities and order million Reynolds numbers relevant to flight vehicles? And if so how can they be best implemented for the various stalling characteristics of airfoils?

This chapter focuses on motivating the present work with historical perspective of steady and unsteady flow control methods, their differences better understood through a short discussion of boundary layer transition, turbulence, and large coherent structures, and modern separation control experiments which have used similar plasma actuators for active flow control as the present work. Some initial investigations for generating streamwise vorticity with plasma actuators will also be presented. While it is well studied how the design of passive vortex generators can be optimized for various boundary layers, pressure gradients, and expansion angles as reviewed by Lin [28], only preliminary investigations for understanding the flow physics of plasma based vortex generators have been investigated on flat plates [68].
1.2 Background

Methods for controlling separated flows have been categorized and diagrammed in various ways throughout the flow control literature. For example, two primary methods of flow control are passive and active. The distinction here is whether or not the method requires additional energy expenditure when using the specified device because the device can be turned on or off. Another important distinction in separation control methods are steady versus unsteady. From a historical perspective this in general tends to divide steady versus unsteady as old versus new respectively. It can be argued that unsteady methods of separation control only became possible because of better understanding of transition, turbulence, and large coherent structures. This categorization shows that one can and should use flow instabilities if they are present, which is the direction of modern flow control: help amplify the dominant frequency of large coherent structures which is intensified the most by nature since this will require less input power.

1.2.1 Steady Flow Control

Flow control to reduce drag (usually referred to as boundary layer separation control) and increase lift (circulation control) are as old as powered flight. In 1904, Prandtl [49] demonstrated suction through a slot to eliminate the separated flow region in the wake of a cylinder. The slot being located asymmetrically could reattach the separated boundary layer on just one side of the cylinder which created circulation and vectored the wake, creating lift. Prandtl’s development of the boundary layer was of course entirely new, and finally explained the energy loss in fluid flows which was not explained by potential flow theory, solving D’Alembert’s paradox, despite the fact this loss was contained to a small layer region close to the body. Some aeronautical historians have asserted that Prandtl felt his main contribution was separation control and not the concept of a boundary layer [14]. Nonetheless, in
experiments, the condition of the boundary layer is fundamental to understanding when it separates. However it should be noted that separation was treated as a purely steady flow phenomena by Prandtl.

Flow control progressed from separation and circulation augmentation on a cylinder to real flight tests on aircraft wings with slot based suction flow control in 1921 seen in the work of Page and Lachmann [25] with a 60% rise in $C_{l_{\text{max}}}$. A pump pulled air into the flap cove, reenergizing the boundary layer to reattach separated flow over the flap and was visualized with tufts of string on the flap. This proved as early as 1921, the ability to control separated flow in real flight conditions was entirely possible with steady flow control.

By 1947, Taylor had introduced the vane-type passive vortex generators [60]. These consisted of metal tabs extending from the wall with a height on the order of the boundary layer height and are angled relative to the incoming flow to produce a wingtip vortex that is propagated downstream. The streamwise vorticity is the mechanism by which high momentum fluid is entrained downward into the boundary layer, preventing separation in an adverse pressure gradient. They have the advantage of being simple to install and not requiring any internal modifications to the aircraft wing such as pneumatic tubing required for active blowing or suction. However, their spacing and height should be well designed for their specific application to avoid adding form drag [28]. This streamwise vorticity can persist 100 device heights downstream, before turbulence breaks down the large vortical structure. It has been shown that device height can be reduced to 20% the boundary layer height and still reduce the separated flow length region by 90% of that which the vortex generator of height $\delta$ did [27]. These devices, studied extensively by Lin known as low-profile vortex generators, tradeoff vorticity production for less drag by reducing their height. It is with this same design approach plasma streamwise vortex generators will be studied, examining how streamwise vorticity is affected by the design parameter space.
unique to these PSVGs such as operating voltage and electrode length.

1.2.2 Unsteady Flow Control

A dynamic system typically does not equally respond to harmonic forcing at all frequencies. To limit the power and hence efficiency of a flow control device, one seeks to decrease the power of the input disturbance, but at a harmonic forcing frequency that will have the largest amplification due to a fluid instability and hence have the largest effect on the performance parameter of interest, say $C_{t_{\text{max}}}$ or $L/D$ in the case of an airfoil. This whole approach is a bit more sophisticated than steady flow control as one seeks to take advantage of fluid instabilities rather than steady momentum addition to the boundary layer. As early as 1948, the first unsteady boundary layer control experiment was published [55]. Schubauer and Skramstad used a loudspeaker (initially just the ambient noise of the wind tunnel) and also a vibrating ribbon to excite the Tollmien-Schlichting instability and hence cause transition from a laminar to a turbulent boundary layer. By 1978, the first unsteady flow control experiment for separation control on an airfoil was published by Viets et al. [65] and at high (stalled) angles of attack by Huang and Maestrello [21]. However unsteady acoustic excitation is extremely facility dependent as concluded in Greenblatt [15]. This issue is highlighted by the effectiveness of acoustic excitation on aerodynamic changes only at acoustic frequencies which excited the resonant modes of the wind tunnel test section as seen in Zaman et al. [70] and only at sound pressure levels damaging to the human ear (156 dB) as seen in Ahuja et al. [2].

In a separate line of more fundamental unsteady flow control research but still quite relevant to separation control, was the work of Winant and Browand, and Brown and Roshko in the understanding of turbulence. Until this time it was thought that a turbulent shear flow was too chaotic to effectively control. However, Winant and Browand [69] showed in experiment that the large-scale structure in a turbulent
shear layer consists of a vortex pair of a very consistent frequency. The shear layer is relevant to the current work because it exists in a separated flow over an airfoil, and exhibits the same dominant large coherent structures shedding at a very constant frequency. Brown and Roshko [6] also showed that large coherent structures were the main factor of momentum entrainment from the high to low momentum side of a mixing layer. Rather than small turbulent scales “nibbling” at the edges of the shear flow, the large scale structures are “gulping” the high momentum fluid into the low momentum fluid region. Their work concluded that turbulent shear layers do contain a dominant frequency of vortex shedding on the large scale, and before decaying to small scales show why separated flows could be receptive and prefer a single flow control forcing frequency.

Building on this large coherent structure concept, a great deal of application specific unsteady flow control experiments showed success during the 1980s and 1990s. Various experiments in unsteady flow control during this period include the control of coherent structures in the wake of a backward facing step [51], boundary layer control by unsteady vortex generation [66], oscillating leading edge flaps [71], control of separated flow on bluff bodies [24], and postponement of separation on airfoils using oscillatory blowing [56], delta wings [16], circular cylinders [3], and 2D fences [57]. It should be noted that some authors, such as Greenblatt [15], replace steady and unsteady flow control with the synonymous terms boundary layer control and excitation respectively. These experiments highlight success with unsteady flow control, however these studies were all concerned with oscillatory blowing, suction, and zero net mass flux devices. They did not investigate the body force producing action on air of SDBD plasma actuators.

There has been a move away from pneumatic based flow control systems to those that are electrical. One reason being that for pressure tubing, frequency response is limited. Another disadvantage of pneumatics is the mechanical complexity including
solenoids and plumbing. There are a number of modern active flow control devices which are electrically operated. One such technology is the zero mass-flux device (or synthetic jet) very similar to the diaphragm of a loudspeaker. Unlike the earlier loudspeaker experiments which acted from a distance, these devices exist at the interface of the fluid and solid body, so they act directly in the boundary layer. The vibrating diaphragm has been powered by a voice coil seen in the work of Nagib et al. [40] and also McCormick [31] and has also been piezo-electrically driven as seen in Chen et al. [9].

The most modern flow control methods seek the frequencies which amplify the most hence requiring the smallest perturbation in the interest of improving efficiency. However it is not always clear as to the best unsteady flow control frequency. Mittal and Kotapati in 2006 were the first to notice three distinct peaks in the power spectral density of velocity fluctuation in unsteady flow separation from numerical studies [38]. The authors attribute these observations to instabilities associated with three different flow regions. First, in lower angle of attack cases when a separation bubble was seen, there was the Kelvin-Helmholtz instability in the shear layer at the leading edge of the airfoil with frequency $f_{SL}$. This is due to the velocity shear from the very high velocity at peak suction location on the airfoil, and the low velocity of the separated boundary layer bubble. Second, there was the characteristic frequency of the vortex shedding out of the separation bubble $f_{sep}$. Third, the von Karman street evident in the airfoil wake has frequency $f_{wake}$. In cases where there was leading edge separation or at high angles of attack then only two distinct frequency peaks were seen, those corresponding to $f_{SL}$ and $f_{wake}$, meaning that $f_{sep}$ frequency is now associated with the chord length scale and hence is identical to the shear layer rollup frequency, $f_{SL}$. These frequencies are of utmost importance in consideration for forcing frequencies with unsteady flow control since they are already being amplified most by nature.
1.2.3 SDBD Flow Control

Roth at the University of Tennessee [52], and Corke et al. at the University of Notre Dame were building upon these unsteady flow control successes and developing their own niche of expertise using SDBD plasma actuators which is the focus of the current work. Specifically, Post and Corke published successful results showing that plasma actuators could effectively reattach separated flow on an airfoil and increase the stall angle in static operation [47], and dynamic stall control on an oscillating airfoil [48]. The SDBD plasma actuators have an additional benefit of being either a steady or unsteady device. In this way both regimes may be tested by simply changing the input waveform.

The flow control approach is centered on the use of Single Dielectric Barrier Discharge (SDBD) plasma actuators that have been successful in a number of aerodynamic flow control applications. Background on SDBD plasma actuators and their use has been presented in a number of recent review articles by Corke et al. [10, 11].

The SDBD plasma flow control devices consist of two electrodes that are separated by a dielectric material. One of the electrodes is typically exposed to the air. The other electrode is fully covered by the dielectric material. The “plasma” in this case refers to the weakly ionized air that forms over the electrode area covered by the dielectric. Figure [I.1] shows an illustration of a generic SDBD plasma actuator arrangement along with a photograph of the diffuse plasma that forms during its operation.

For the AC plasma actuator, the ionized air in the presence of the electric field produced by the electrode geometry results in a body force vector field that acts on the neutral air. The asymmetric electrode arrangement that is shown in Figure [I.1] leads to a net body force that causes the neutrally charged air to move towards the wall and jet away from the exposed electrode (left to right in the schematic). At atmospheric pressure, the body force produced by the AC plasma actuator scales as
The power-law exponent depends on static pressure. Valerioti and Corke \[64\] documented this for pressures ranging from 0.19 bar to 9 bar.

The body force vector field produced by AC plasma actuators is easily included in the momentum equations of numerical flow solvers. Numerically efficient and validated models for the space-time dependent ionization that lead to the time-dependent body force have been developed \[42, 43, 41, 37, 36\]. The advantage of these flow simulations is that the requirements of the AC plasma actuator such as its location and voltage, can be determined and designed for the application and experiment.

With separation control, periodic pulsing of the plasma actuator has often been found to be more effective than steady operation \[47, 17, 46, 19, 20\]. In these cases, the optimum unsteady frequency, $f$, has corresponded to a dimensionless frequency of $f^+ = fL_s/U_\infty = 1$, where $L_s$ is the length of the separation region, and $U_\infty$ is the local outer mean velocity.

The NP plasma actuator is also a SDBD device. It generally has the same electrode

\begin{center}
\textbf{Figure 1.1.} Schematic illustration of a Single Dielectric Barrier Discharge (SDBD) plasma actuator and photograph of the plasma during operation.
\end{center}
and dielectric layer design that is used with AC plasma actuators and was illustrated in Figure 1.1. The voltage waveform used for the NP actuator is however quite different from that of the AC plasma actuator. That waveform consists of nano-second duration positive voltage pulses.

While AC plasma actuators produce a body force vector field that acts on the neutral air, the NP plasma actuators generate short-duration localized heating that has been shown to produce expanding hemispherical pressure waves [58, 29, 53, 59]. The pressure waves are typically used to excite a fluid instability such as a Kelvin-Helmholtz instability in a shear layer [45]. As a result, NP plasma actuators have been shown to be effective for leading edge separation control.

In the literature, there are two preliminary investigations of slightly higher speed leading edge separation control nanosecond pulse DBD plasma actuators. Little et al. [29] used a leading edge nanosecond pulse plasma actuator on the same EET airfoil as the current investigation and were able to increase stall angle by a few degrees at a Reynolds number of $1 \times 10^6$. Roupassov et al. [53] have a preliminary nanosecond pulse plasma actuator investigation of leading edge separation control for Mach numbers to up 0.85. Pressure tap measurements indicate leading edge reattachment at a few angles of attack, but changes in the lift curve, increases in stall angle, and $C_{l_{max}}$ are not reported, nor is there any mention of lift, drag, or moment effects. In addition, the physical mechanism of flow control in nanosecond pulses is understood to be thermal, rather than a body force with SDBD. Nonetheless, their results indicate through pressure tap measurements that flow reattachment of leading edge separation is possible within a few degrees past stall angle at high subsonic Mach numbers. The modeling of NP plasma actuators involves simulation of the localized heating effects. The recent work [26, 62] has been successful in replicating many of the experimental observations.
1.2.4 Stall Morphology - Separation on Different Geometry Airfoils

The empirical results of airfoil data are well described in classic texts including Abbott and Doenhoff’s *Theory of Wing Sections* [1] and von Mises’ *Theory of Flight* [67]. The thickness to chord ratio of an airfoil is a major factor in determining maximum lift and lift slope of an airfoil. Increasing this ratio reduces the slope of pressure distribution aft of the suction peak (reducing the adverse pressure gradient) thereby extending the percentage of the chord which can sustain an attached boundary layer. Generally then, thinner airfoils have a more abrupt falloff in lift at stall because when the critical angle is reached, the boundary layer separates near the leading edge just aft of the suction peak due to the very large adverse pressure gradient. When this occurs the peak suction is reduced (the dominant contribution to lift) and the pressure is increased over the upper surface greatly reducing lift, increasing form drag, and increasing the nose-down pitching moment. For a thick airfoil, the strength of the adverse pressure gradient is relieved and the boundary layer separation location progresses slowly from the trailing edge to leading edge with increasing angle of attack. For a thick airfoil, reduction in form drag is possible pre-stall by separation control, but would have little affect on lift since separation is aft of the suction peak.

The pressure on the suction surface of both the EET and V-22 airfoil as calculated by XFOIL is presented in Figure 1.2 for three different angles of attack 4°, 6°, and 8°. A steep negative slope is indicative of a strong adverse pressure gradient, very evident for the 12% thin EET airfoil from the suction peak at $\frac{z}{c} = 0.03$ to 0.1. As the angle of attack increases, the suction peak continues to rise until the pressure gradient is large enough to cause separation near the LE. However for the thick V-22 airfoil, as predicted by its larger thickness to chord ratio, the adverse pressure gradient is much smaller as seen in Figure 1.2 by a less steep negative slope causing separation to occur near the TE. The chord Reynolds number is larger then at separation and for the freestream velocities of interest to this experiment would put the chord Reynolds
Figure 1.2. Pressure distribution comparing thin EET airfoil to thick V-22 airfoil at 4°, 6°, and 8° angles of attack as calculated from XFOIL.

number in a transitional or turbulent regime.

1.2.4.1 Leading Edge Stall

The NASA EET (energy efficient transport) airfoil has a profile as seen in Figure 1.3. This airfoil has a thickness to chord ratio, $t/c = 12\%$. This thin of an airfoil combined with the small leading edge nose radius and flattop shape of a supercritical airfoil all contribute to this airfoil exhibiting leading edge boundary layer separation which has been observed in experiment for clean configurations, i.e., no flaps or slats [63, 30]. The entire suction surface stalls abruptly as angle of attack is increased to this critical value. When this stall occurs, the pressure distribution over the entire
suction surface is essentially constant. The suction peak near the leading edge and pressure recovery aft on the airfoil are lost. This different pressure distribution in stall leads to greater aft loading, and hence larger pitching moments, in addition to a rise in form drag.

Due to the separation location being locked to one location, the leading edge separation is well suited for spanwise oriented plasma actuator at the leading edge which will always be close to and just upstream of the separation location for all angles of attack that stall occurs. In addition, the local chord Reynolds numbers is small, indicating the boundary layer is laminar at separation \[30\]. In NASA experiments, this airfoil is often designed with a leading edge slat which makes the airfoil exhibit trailing edge stall \[39\]. However, with the current experiment without a slat it will be shown that the airfoil exhibits leading edge stall and so only two dominant shedding frequencies will be present: one associated with the shear layer rollup and the second with von Kármán street frequency in the wake. In this way a single SDBD plasma actuator will always be located at or close to the source of the shear layer.

1.2.4.2 Trailing Edge Stall

The thick airfoil used in this experiment is 23 % thick. Due to its proprietary nature, the section shape is not presented here. The V-22 wing section trades structural

![NASA EET Coordinates](image)

Figure 1.3. EET Airfoil Coordinates.
rigidity at the penalty of higher drag. Since the Osprey can hover, the entire weight of the aircraft is supported at the wingtips which explains the necessity of a thick and rigid section. The use of passive vane-type vortex generators which produce pairs of counter-rotating vortices can be observed on the aircraft, and one can speculate they are to reduce drag by means of boundary layer separation control. Again, due to the proprietary nature of the section, the exact design of VG spacing and locations are unknown in the literature. Nonetheless this thick airfoil provides an opportunity to study the efficacy of PSVGs on a wing section the exhibits trailing edge stall and in a boundary layer that is turbulent. It is worth noting sometimes trailing edge separation control refers to separation control on the leading edge of the flap element on a wing or simply separation control on a single piece airfoil section with no high lift device. The latter is the focus of this experiment since the flap of the V-22 section was sealed for the entirety of the experiment.

Trailing edge separation control has been a more difficult experimental problem for relatively low momentum flow control devices than leading edge separation control \[35\]. This can be attributed to, first, the lack of a strong velocity difference in the shear layer. This is evident from the pressure distribution in Figure \[1.2\] For the EET airfoil the local velocity at the edge of the boundary layer near the leading edge is much larger than the freestream because of how high the suction peak is. However for the thick V-22 section with separation near the trailing edge, the local flow velocity is roughly the same order as the freestream velocity. The local velocity above the separated flow region represents a source of high momentum fluid which can be brought down into the boundary layer or as explained by Chandrasekhar \[8\], “The source of the Kelvin-Helmholtz instability clearly lies in the energy stored in the kinetic energy of relative motion of the different layers: the tendency towards mixing and instability will be greater, the larger the prevailing shear as measured by \[\frac{\partial U}{\partial y}\].” The second aspect which makes trailing edge separation control more difficult is the
turbulent boundary layer. Since causing transition is itself a method of separation control, this coupled with the weaker local velocity near the trailing edge makes this problem more difficult than leading edge separation control. So the production of streamwise vorticity with plasma actuators will be the basis of this aspect of the experiment.

1.2.4.3 Scaling and Physics of PSVG Operation

The creation of streamwise vorticity with a PSVG is not confined to a small region as with a passive VG. There is both a turning of spanwise vorticity from the boundary layer and an accumulation due to the direct crossflow body force of the PSVG. This process has been well visualized at lower Reynolds numbers experimentally by Jukes and Choi [23].

To ensure that the PSVG electrodes are of sufficient length in producing streamwise vorticity, $\omega_x$, to reduce drag due to separation, one should compare the time of convection of freestream momentum to the two known mechanisms to how PSVGs create streamwise vorticity. As shown in Wicks et al. [68] for streamwise vorticity creation in the boundary layer on a zero pressure gradient flat plate, both direct production of streamwise vorticity due to the electrohydrodynamic wind and a spanwise variation of the freestream component of velocity are sources. The time associated with the convection of freestream momentum over the distance of the actuator length is called the residence time, $T_{Res}$. The timescale of vorticity production, $T_P$, and vorticity redistribution, $T_{VR}$. The effect of freestream velocity, peak induced velocity of the PSVG, and boundary layer parameters such as displacement thickness, shear velocity, and electrode spacing, all determine the relative magnitudes of these timescales. The specific timescales for this experiment, and comparisons to the flat plate experiment will be analyzed in Section 4.2. From a design aspect however, their results highlight that a longer electrode will increase the maximum freestream velocity where the PSVG
is still effective and operating in the regime where streamwise vorticity is proportional to freestream velocity.

The scaling laws as derived by Wicks et al. to explain three important timescales and vorticity increments are reviewed here to explain how the production of streamwise vorticity for PSVGs is different from passive VGs. For reference the scaling laws for vorticity and their associated timescales are summarized in the following equations.

\[ T_{Res} = \mathcal{O} \left( \frac{l}{U_\infty} \right) \]  \hspace{1cm} (1.1)

\[ T_P = \mathcal{O} \left( \frac{\delta^*}{V_P} \right) \]  \hspace{1cm} (1.2)

\[ T_{VR} = \mathcal{O} \left( \frac{\delta^*}{u_\tau} \right) \]  \hspace{1cm} (1.3)

Regime 1: if \( T_{VR} < T_{Res} \) \( \Rightarrow \) \( \Delta \omega_x = \mathcal{O} \left( \frac{2 \delta u_\tau}{\delta^* \lambda} \right) = \mathcal{O} \left( \frac{2 \delta}{\lambda} \cdot \frac{1}{T_{VR}} \right) \) \hspace{1cm} (1.4)

Regime 2: if \( T_{VR} > T_{Res} \) \( \Rightarrow \) \( \Delta \omega_x = \mathcal{O} \left( \frac{4 V_P^2}{\delta^* \lambda} \cdot \frac{l}{U_\infty} \right) = \mathcal{O} \left( \frac{4 V_P}{\lambda} \cdot \frac{T_{Res}}{T_P} \right) \) \hspace{1cm} (1.5)

Wicks et al. found that there are two distinct regimes of operation for the PSVG with increasing freestream velocity, \( U_\infty \). “Regime 1” is characterized by a linear increase in streamwise vorticity with freestream velocity according the Equation 1.4 due to significant contribution from spanwise vorticity reorientation. In this regime, a boundary layer with greater shear at the wall due to increasing freestream velocity leads to a greater distortion in the spanwise variation of freestream velocity, \( \frac{\partial U}{\partial z} \), thus leading to better tilting of spanwise vorticity, \( \omega_z \) to streamwise vorticity, \( \omega_x \). However, as freestream velocity continues to increase for a given length electrode, the residence continues to decrease and becomes the same order of magnitude as \( T_{VR} \). Beyond this saturation velocity, the streamwise vorticity scales inversely with the freestream velocity according to Equation 1.5 named “Regime 2.” Thus ”Regime 1” represents the on design mode of operation for a PSVG such that it can continue
to produce greater streamwise vorticity as freestream velocities increase. For even higher freestream velocities when $T_{Res} < T_P$, the PSVG would no longer be creating any streamwise vorticity according to the scaling laws.

1.3 Objective

Currently there exists no airfoil study in the effectiveness of SDBD plasma actuators in separation control at Reynolds and Mach numbers similar to flight conditions of commercial or military aircraft. It is the goal of these experiments to first evaluate the effectiveness of SDBD plasma actuators for separation control on a thin airfoil (12%) which exhibits leading edge separation at stall. The removable leading edge allows for different dielectric materials to be tested. Second, this experiment will evaluate the effectiveness of PSVG type plasma actuators for separation control on a thick airfoil (23%) which exhibits trailing edge separation even at small angles of attack.

The present work started with leading edge separation control on the NASA EET airfoil to confirm the effectiveness of plasma actuators as seen in the previous experiments at Notre Dame by Post et al. However in that study, leading edge separation was only tested up to a Reynolds number of $0.46 \times 10^6$, while the current study investigated up to $2.24 \times 10^6$, that is 30 m/s versus 136 m/s. On using plasma actuators to generate streamwise vorticity, Wicks et al. only used a flat plate with a zero pressure gradient. Thus there remains the open question of using plasma actuators to suppress trailing edge separation using PSVGs and comparing them to passive VGs. This work presents new experimental data of plasma streamwise vortex generators operating in an adverse pressure gradient with better understood separation control physics through the measurement of streamwise vorticity, and finally proposing how to design such a device that performs equivalently to the traditional passive vane type vortex generator.

The conclusions of these related experiments are promising for the feasibility of
good performance with SDBD plasma actuators at realistic flight conditions. It is also evident in the literature that any plasma based flow control experiments at higher subsonic speeds and Reynolds numbers above of a million are quite preliminary. It is the goal of this research to extend the proven performance of SDBD plasma actuators, based on the expertise and facilities of the FlowPAC initiative at Notre Dame, to higher Mach and Reynolds numbers previously unrealized for airfoil separation control and to propose a set of design criteria for implementing PSVGs and creating the electric equivalent of a vane type passive VG for suppressing trailing edge stall.
CHAPTER 2

EXPERIMENTAL SETUP

This chapter shows the details of the experimental facilities used in the separation control experiments. For the leading edge separation control experiment, the details of the EET leading edge dielectric materials and the various waveforms used are discussed. For the trailing edge separation control experiment, the V-22 PSVG design is shown including decisions about electrode length and spacing. The electrical power requirements is presented as is the effect of electrical noise. Next the details of how the aerodynamic forces and moment were measured are examined. The uncertainty in the measurands is quantified. Finally the various Pitot tubes used for boundary layer measurements are discussed in addition to calibration of a 5-hole probe. Table 2.1 provides a summary of the airfoils and flow control devices used in the separation control experiments.
2.1 Wind Tunnel

The closed-loop White Field Wind Tunnel at the University of Notre Dame is capable of sustained freestream velocities of 200 m/s ($M = 0.6$) with turbulence intensities below $u'/U_\infty \leq 0.05\%$. The test section is 0.91 m × 0.91 m in cross-section and 2.73 m long as seen in Figure 2.1. Removable square windows, 0.61 m to a side, allow easy optical access around the test section, useful in this experiment for PIV. The tunnel is powered by a 1.3 MW (1750 H.P.) AC motor. The fan is 2-stage high-solidity vane-axial fan 2.43 m in diameter. A 440 kW (125 refrigeration tons) chiller creates ice for 4 °C water to flow through the turning vanes, in this case doubling also as a heat exchanger. This allows for closed-loop feedback control to maintain constant wind tunnel temperatures throughout testing.

The test sections are easily removed from the wind tunnel to allow for setup and instrumentation of the experiment with the aid of an air cushion frame. The rubber skirts on the frame are pressurized with shop air and allow for easy movement of the test section around the Whitefield Wind Tunnel laboratory to allow for concurrent experiments.
2.2 EET Airfoil Design

The EET airfoil was sized to have a span of 30.48 cm to fit between the existing splitter plates in the Whitefield Mach 0.6 wind tunnel. A chord of 30.48 cm was chosen to have a Reynolds number of $2 \times 10^6$ at Mach 0.3, while keeping blockage under 5% at 20° angle of attack. Blockage as a function of angle of attack is shown in Figure 2.2. The wing had an $AR$ of 1, but 2-dimensionality of the flow was ensured by the splitter plates.

The EET airfoil was designed to have a removable leading edge as seen in Figure 2.3. A channel running through the center allowed for high voltage leads to the plasma actuator and also allowed to pass Tygon tubing from the 7 suction side pressure taps. Four bolt holes held the airfoil to a steel collar on each side of the airfoil centered exactly at the quarter-chord. Figures of detailed engineering drawings are given in Appendix A.1.
Figure 2.2. Airfoil Blockage in Whitefield Mach 0.6 Wind Tunnel.

Figure 2.3. Fabricated EET airfoil with Macor leading edge.
2.3 EET Spanwise Plasma Actuator Design and Electronics

Two different removable leading edges were fabricated so the effect of dielectric thickness on aerodynamic performance could be investigated.

The first leading edge was made of Macor, a machinable ceramic with a wall thickness of 3.175 mm. This thick dielectric has been shown to increase body force and hence the momentum addition of the plasma actuator. The actuator location is at the leading edge as used by Post et al. \[47\]. The extent of the exposed electrode is 6.35 mm and the buried electrode 19.05 mm as shown in Figure 2.4. Also consistent with previous experiments, the direction of the induced body force is in the same direction as the freestream with the idea that momentum transfer to the boundary layer is enhanced.

Kapton was chosen as the second dielectric material to test because it is the most often used material for proof of concept experiments since Kapton tape is easy to apply to a curved surface and does not require any machining. This leading edge was provided by Boeing, the research sponsor, and had a uniform dielectric thickness of 0.127 mm (25 times thinner than the Macor leading edge). The copper electrode had a thickness of 7 \(\mu\)m.
The dielectric barrier discharge was sustained by a 2.3 kHz sine-waveform generated by an Agilent 33220A Function Generator. This signal was boosted through the use of a commercially available Crown XTi 4000 audio amplifier. This signal was then sent to the primary side of two CMI 5525 transformers with the polarity of the two connections inverted. This allowed the output of the transformers to be $\pi$ radians out of phase, doubling the voltage seen at the plasma actuator. The function generator output was adjusted until the secondary side of the transformer had an output of 30 kV$_{p-p}$ for the Macor leading edge and 15 kV$_{p-p}$ for the Kapton leading edge. This final operating voltage was measured by a LeCroy PPE20KV high voltage probe. For unsteady operation of the plasma actuator, the duty cycle was maintained at 50%, and the frequency of turning on and off the actuator for a range of reduced frequencies was $0.2 \leq f^+ \leq 4.0$.

With the NP plasma actuator, only unsteady operation is relevant since the separation control mechanism is through an unsteady temperature disturbance rather than a body force like the AC plasma actuator. The unsteady NP operation consisted of generating a series of nano-time-scale voltage pulses that were switched on and off at an optimal frequency, $f$. The pulse amplitude was approximately 11.9 kV$_{p-p}$, and the pulse width was approximately 100 ns. Figure 2.5 shows a sample voltage time series of the NP plasma actuator input. The pulse frequency for the NP actuator was nominally 1 kHz. The unsteady operation consisted of forming a train of 1 kHz nano-pulses that were switched on and off at a lower frequency, $f$. With this, the duty cycle was maintained to be approximately 50%.

Tables 2.2 and 2.3 provide a summary of the plasma actuators used in the EET experiment and their operating parameters both in steady and unsteady operation.
Figure 2.5. Sample input voltage time series to NP plasma actuator.

### TABLE 2.2

<table>
<thead>
<tr>
<th>Steady Operation</th>
<th>Thickness [mm]</th>
<th>Waveform</th>
<th>$V_{p-p}$ [kV]</th>
<th>Frequency [kHz]</th>
</tr>
</thead>
<tbody>
<tr>
<td>AC Macor</td>
<td>3.175</td>
<td>sine</td>
<td>30</td>
<td>2.3</td>
</tr>
<tr>
<td>AC Kapton</td>
<td>0.127</td>
<td>sine</td>
<td>12.5</td>
<td>2.3</td>
</tr>
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</table>

### TABLE 2.3

<table>
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<tr>
<th>Unsteady Operation</th>
<th>Thickness [mm]</th>
<th>Waveform</th>
<th>$V_{p-p}$ [kV]</th>
<th>Frequency [kHz]</th>
<th>Duty Cycle %</th>
</tr>
</thead>
<tbody>
<tr>
<td>AC Macor</td>
<td>3.175</td>
<td>sine</td>
<td>30</td>
<td>2.3</td>
<td>50</td>
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<tr>
<td>AC Kapton</td>
<td>0.127</td>
<td>sine</td>
<td>12.5</td>
<td>2.3</td>
<td>50</td>
</tr>
<tr>
<td>NP Kapton</td>
<td>0.127</td>
<td>pulse (Fig. 2.5)</td>
<td>11.9</td>
<td>-</td>
<td>50</td>
</tr>
</tbody>
</table>
Figure 2.6. AC voltage waveform showing high carrier frequency and unsteady operation with 50% duty cycle.
2.4 V-22 Osprey Airfoil Design

For the investigation of trailing edge separation, Bell Helicopter has provided the proprietary V-22 Osprey section shape. The model for this experiment has a span of 30.48 cm and a chord of 35.56 cm, again to fit in the same test section and splitter plates as the EET section in the Whitefield Wind Tunnel. The V-22 airfoil was designed to have a dielectric insert that extended from the chord location \( \frac{x}{c} = 15\% \) and extending to the trailing edge of the main element. This allows the PSVGs to start in the same boundary layer and flow field conditions as the passive vortex generators on the V-22. This also maximizes the area over which PSVGs could be applied. To prevent electrical arcs to the metal body, the final electrodes could not run the entire length of the dielectric, but rather are positioned from \( \frac{x}{c} = 20\% \) to 60\%. To account for the effect of any possible crossflow, and to be similar to the real aircraft, the model has 5\° of forward sweep. An isometric drawing of the final design can be seen in Figure 2.7.

The dielectric epoxy Hysol ES-1902 was used because the volume of the piece was large enough to cost prohibit machining the piece from a solid piece of Macor. It was cast in a CNC mill created aluminum mold. The dielectric insert had a uniform wall thickness of 6.35 mm which allowed for operation of the PSVGs up 48 kV\( p - p \).

2.5 V-22 PSVG Actuator Design

It has been shown that the shear layer’s growth, and hence entrainment of momentum is most sensitive to disturbances at its origin, which on an airfoil is where the shear layer’s source is, at the beginning of separation [44]. However with trailing edge stall, this critical point can vary with angle of attack, Reynolds number, and surface roughness. All of these factors influence the pressure distribution around the airfoil and move the separation location. Ideally you would then have an SDBD plasma
Figure 2.7. V-22 Osprey airfoil with $5^\circ$ forward sweep, flap, and 6.35 mm thick dielectric epoxy insert for PSVGs.

actuator anywhere on the suction surface of the wing. This becomes possible by reorienting the SDBD plasma actuator from having an electrode in the span direction to the chord direction. In this way multiple electrodes spaced across the span all parallel to the freestream flow will now generate streamwise vorticity rather than spanwise vorticity. The work of Wicks et al. have shown the efficacy streamwise vorticity production using PSVG devices on a flat plate with no pressure gradient [68]. There are two important linear dimensions in PSVG design, the inter-electrode spacing, $\lambda$, and the PSVG length, $l$.

There does notcurrently exist a scaling law known between the boundary layer thickness and optimal PSVG spacing. Wicks et al. first found that for a laminar boundary layer on a flat plate, the greatest streamwise vorticity was produced by PSVGs when the the inter-electrode spacing was equal to the boundary layer height, $l = \delta$. However when the boundary layer was tripped by roughness and was turbulent with an 80% increase in $\delta$, the optimal spacing reduced to $l = 0.6\delta$. Knowing that no scaling was found for inter-electrode spacing for the flat plate experiment, and knowing that in the V-22 experiment the pressure gradient and boundary layer thickness would
change, a series of flow visualization videos were recorded with various vortex generator spacings, $\lambda = 15.9$ mm, 19.1 mm, 22.2 mm, and 25.4 mm, at two different angles of attack, $\alpha = 0^\circ$ and $13^\circ$, and three Mach numbers, $M = 0.06$, 0.1, and 0.2. Tufts of yarn were affixed to assess the extent of flow separation on the airfoil. Also, because each electrode of the PSVG produces a pair of counter-rotating vortices, passive vortex generators were used rather than PSVGs to find the optimal spacing because they could much more quickly by removed and replaced further apart. Qualitative results seen in the video, a screen shot of each can be seen in Figure 2.8, indicate that the 19.1 mm was optimal at all the tested Mach numbers and angle of attacks because this spacing reduced the flow separation region the most.

Now that the optimal electrode spacing has been found, the attention is switched to electrode length. The specific timescales for this experiment, and comparisons to the flat plate experiment will be analyzed in the Experimental Results Section 4.2 according to the scaling laws discussed in the Introduction Section 1.2.4.3. From a design aspect however, the scaling laws developed by Wicks et al. highlight that a longer electrode will increase the maximum freestream velocity where the PSVG is operating in “Regime 1.” In this case, the longest electrode length that could be fit onto the dielectric was $l = 14.2$ cm, or 40% of the chord.

<table>
<thead>
<tr>
<th>Dielectric</th>
<th>Thickness [mm]</th>
<th>Waveform</th>
<th>$V_{p-p}$ [kV]</th>
<th>Frequency [kHz]</th>
<th>$\lambda$ [cm]</th>
<th>$l$ [cm]</th>
<th>$\frac{l}{c}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hysol ES1902 Epoxy</td>
<td>3.175</td>
<td>sine</td>
<td>31-48</td>
<td>2.3</td>
<td>2.22</td>
<td>14.2</td>
<td>0.40</td>
</tr>
</tbody>
</table>
Figure 2.8. Optimal PSVG inter-electrode spacing was found to be 19.1 mm from flow visualization.
Figure 2.9. Sample input voltage and current time series and average power to PSVG plasma actuator at $48 \, kV_{p-p}$, 2.3 kHz.

2.6 Electrical Power Requirements

An inductive current coil was used to measure the instantaneous current through the PSVG plasma actuator along with the instantaneous voltage. The key to an accurate power measurement is the resolving of the short timescale current spikes due to microdischarges superimposed over the low frequency current which lags the voltage [4]. This was accomplished by sampling at 10 MHz. These traces can be seen in Fig. 2.9 for the $48 \, kV_{p-p}$ case.

The average power was found by multiplying the instantaneous voltage and current, then integrating over one period according to the following equation.

$$P_{avg} = \frac{1}{T} \int_{0}^{T} v(t) \cdot i(t) \, dt$$  \hspace{1cm} (2.1)

This calculation was repeated for the 4 different voltages tested. The total edge length (L) of the 8 PSVG electrodes along which plasma forms is 1.68 m. Figure 2.10 provides a summary of electrical power required per linear distance of electrode edge.
Figure 2.10. Power required for PSVG at various operating voltages.
2.7 Passive Vortex Generator Design

The vane-type passive vortex generators are based on the counter-rotating configuration. The geometry consists of a VG chord of $0.03c$ and the device height of approximately $6\delta$. The average width of the VG is equal to the PSVG electrode width, 6.35 mm or $12\delta$. The spread angle relative to the centerline is $19^\circ$. In this experiment the VG height was excessive as it has been shown that heights equal to $0.8\delta$ are just as effective at boundary layer reattachment in an adverse pressure gradient and hence some angles of attack might see drag increases from form drag of the device itself. Nonetheless this VG design has similar proportions to those used in the NASA backward facing ramp separation control experiments [28]. The VG spacing is always equal to the PSVG electrode spacing in the experiment, 22.2 mm or about $45\delta$.

Figure 2.11. Vane-type passive vortex generator.
2.8 Force and Moment Measurements

The ends of the model support shaft that passed through the test section ceiling and floor were connected to torque measurement load cells that were a part three-component force measuring platforms. A schematic and photograph of the platform that was on the underside of the test section is shown in Figure 2.12 and 2.13. The force measuring platforms consisted of two stacked sliding assemblies. The sliding assembly allowed motion in the stream-wise and cross-stream directions. The forces in those two directions were measured by translation load cells. The load cell that was sensing the force in the flow direction (drag) is located at the right of the image. The one sensing the cross-flow force (lift) is in the middle-foreground of the image. The torque cell was attached to the outer-most (lower) platform in the stack. Two measuring platforms were fabricated so that the top and bottom support shafts had their reaction loads measured. For a given time series of data, the lift, drag, and moment measurements were summed from both the upper and lower assemblies as a function of time to calculate the total loads at any given time.

The model support design was based on maximum lift, drag and moment coefficients of 1.77, 0.26 and -0.18, respectively. At Mach 0.4, these led to lift and drag forces of 1120 N (252 lbf) and 166 N (37.4 lbf), respectively, and a torque of -21.7 N·m (-16.0 ft·lbf). The shafts holding the model are solid steel bars with an outside diameter of 3.175 cm (1.25 in). These were found to be necessary to minimize vibration amplitudes of the model at the post-stall angles of attack. A finite elements analysis of the model and support shafts with steady loading, indicated that the maximum displacement of the model due to bending would be 0.004 in. The insulated wires to the plasma actuator electrodes and pressure tap tubulations from the model were cable-tied to the aft part of the solid shaft.
Figure 2.12. Schematic of force and moment balance.

Figure 2.13. Photograph of force and moment balance.
2.9 Data Acquisition

Voltage time series proportional to the lift and drag forces, and pitching moment were recorded from the two force measuring platforms that supported the airfoil model. The forces and moment were measured with Transducer Techniques MLP load cells. The voltages from the load cells were acquired at a sampling frequency of 5 kHz for a period of 10 seconds using a digital data acquisition system. These voltage time series were then processed to obtain the time-averaged forces and moments, and the root-mean-square of their fluctuations. The average and fluctuating forces and moments from the two force platforms were summed to obtain the total aerodynamic loads on the model. The aerodynamic drag on the support shafts was not removed from the drag measurements on the EET airfoil. For the V-22, half-cylinder shrouds were placed upstream of the support shafts so that the total drag was just due to the V-22 wing.

2.10 Uncertainty of Measurements

The load cells have a nonlinearity of 0.1%, a hysteresis of 0.1%, a non-repeatability of 0.05%, and a zero balance error of 1% of the full scale range. The combined uncertainty of these is \( \frac{u_{L}}{L} = 1.0112\% \) of the full scale range. The full scale range of the load cells used to measure lift was 667 N (150 lbf). The resolution of the analog to digital conversion was \( 2^{16} \). Therefore the measurement uncertainty due to quantization was negligible, \( \frac{u_{res}}{L} = 2^{-16}/2 = 15.3 \times 10^{-6}\% \). Given this, the uncertainty in the lift measurement was then

\[
    u_{L} = \sqrt{u_{1}^{2} + u_{res}^{2}} = 1.52 \ \text{lbf} = 6.75 \ \text{N}. \tag{2.2}
\]
The uncertainty in the lift coefficient, \( u_{CL} \), was then

\[
  u_{CL} = u_L \frac{\partial C_L}{\partial L} = u_L \frac{1}{2\rho U_{\infty}^2} \frac{1}{c_b} = 0.0289.
\]  

(2.3)

2.11 Effect of Electrical Noise

It is known that SDBD plasma actuators can produce some electromagnetic interference. The effect of electronic noise is explored for its effect on the Scanivalve pressure transducer since these sensors were critical to measuring boundary layer profiles, five-hole probe 3D velocity components, and freestream velocities. The Scanivalve was arranged as it was in the experiment so its proximity was identical to when it was being used with the wind tunnel on. Gage pressures were measured first with the SDBD actuator off and the wind tunnel off so the gage pressure is zero. Next the SDBD plasma actuators were turned on at various operating voltages again with the wind tunnel off so that the only difference in the gage pressure would be due to electronic noise induced by operation of the actuator. The average of each time series was next calculated and the difference with respect to the no actuator is plotted in Figure 2.14. The standard deviation of the time series difference with respect to no actuator is in the right side of Figure 2.14.

This figure shows that even at 48 kV\(_{p-p}\), the effect of electrical noise on the pressure is only -0.5 Pa, or 0.0007% full scale range. The standard deviation of the pressure time series changing due to the actuator was at most 7 Pa, or 0.01% of the full scale range. So the electronic noise from the AC type waveform did not affect the accuracy of the pressure transducer system.

The challenge with the nanosecond pulse system was in dealing with the electronic noise it produced. Some shielding and grounding was explored, but this had a minimal effect on the electronic noise that was picked up in the signals from the force measuring load cells and pressure transducers used in experiment. The voltage levels of electronic
Figure 2.14. Mean and standard deviation of pressure measurements with Scanivalve for various operating voltages.

noise exceeded the ±10 V limit of the A/D converter in the data acquisition system. Analog filtering would not reduce the amplitude of the electronic noise, which possibly indicated that it was entering through the instrument ground.

The electronic noise appeared as narrow voltage spikes that were correlated with the high-voltage pulses sent to the plasma actuator. Normally the pulses were generated by a waveform generator. The workaround solution was to trigger the waveform generator with a TTL-pulse from the digital clock on the A/D converter. A time delay was then set in the waveform generator following the trigger pulse so that voltages were sampled in between the electronic noise spikes. Because of the limits of having discrete frequencies for the A/D clock, this approach required some variation in the pulse frequency and duty cycle in order to vary $f^+$. However, they were nominally kept constant.
2.12 Wake Drag Measurements

A traverse which held a Pitot-static probe was located one chord downstream of
the trailing edge of the V-22 airfoil so that the local velocity could be measured and
can be seen in Figure 2.15. The velocity profile of the wake was integrated according
the momentum deficit method to find the section drag coefficient as first proposed
by Betz [5] and simplified by Jones [22] for wake drag measurements near the airfoil
trailing edge.

\[
C_d = \frac{2}{c} \int_{\text{wake}} \sqrt{\frac{p_0(y) - p(y)}{p_0 - p_\infty}} \left(1 - \sqrt{\frac{p_0(y) - p(y)}{p_0 - p_\infty}}\right) dy
\] (2.4)

The quarter span location was used because mounting holes prevented a PSVG
electrode on the airfoil centerline, and flow visualization showed some wall effects near
the splitter plates at high angles of attack. Rather than use the total drag measured
from the load cells which would include the wall effects, the wake traverse would
reflect the true sectional drag coefficient.
Figure 2.15. Pitot tube, looking upstream, traversed horizontally in wake of airfoil to measure drag.
2.13 Boundary Layer Measurements

The velocity profiles of the boundary layer were measured with a Pitot probe made from a 0.64 mm outer diameter stainless steel tube manufactured by United Sensor. The opening on the front measured total pressure and was crimped to an opening of 0.076 mm as photographed in Figure 2.16. This Pitot tube was mounted in the same micrometer adjusted stages seen in Figure 2.22. The two effects that must be considered for using a Pitot tube near a wall are the effects of shear and wall proximity. By crimping the end, the effect of shear is minimized by reducing the area of spatial integration over which the stagnation pressure is measured \[61\]. By keeping the tube diameter also small, velocity measurements as close to the wall as the tube diameter are accurate \[32\], in this case measurements as close as 0.64 mm. In a boundary layer only the total pressure varies as a function of height. So the total pressure measurements in conjunction with a single point static measurement from the five-hole probe provided the necessary data for local flow speed from the isentropic relations.

![Pitot tube](Figure 2.16. Photograph of Pitot tube for boundary layer velocity profiles.)
2.14 Five-Hole Probe

A five-hole Pitot tube was used to measure all three orthogonal components of velocity to investigate the 3-dimensional flows in the boundary layer. This multi-hole probe was calibrated using the method developed by Bryer and Pankhurst [7]. Details of how streamwise vorticity and vorticity tilting were quantified from the 3D velocity components is also examined in this section.

2.14.1 Five-Hole Probe Velocity Calibration

First the probe is mounted a series of goniometers so that pressures in all 5 orifices can measured at known combinations of pitch and yaw angle of a known flow speed. This assembly is then positioned in a uniform flow in the exhaust of a jet. A photograph of this setup is seen in Fig. 2.17. The port numbering convention is in Fig. 2.18.

For every pitch angle ($\alpha$) and yaw angle ($\beta$), two functionals were tabulated, $F_1$ and $F_2$ defined as

\[
F_1 = \frac{p_4 - p_5}{p_1 - p_m},
\]

(2.5)

\[
F_2 = \frac{p_2 - p_3}{p_1 - p_m},
\]

(2.6)

The mean pressure of the four circumferential ports is $p_m$. Since $p_4$ and $p_5$ are on opposing sides of the probe, $F_1$ is primarily a function of pitch angle, and $F_2$ primarily yaw. This calibration map is shown in Fig. 2.19 with solid lines being curves of constant pitch, and dashed lines being curves of constant yaw. Angles were adjusted in 10° increments up to ±40°.

So when an actual measurement is taken in the wind tunnel with the five hole probe, the corresponding flow angles relative to the probe head can be backed out via interpolation on the calibration map. Next the static and dynamic pressure maps need to be found for each calibrated flow angle so that the local velocity, with the
Figure 2.17. Photograph of 5-hole probe, goniometers, and calibration jet.

Figure 2.18. Numbering convention of pressure ports looking at tip of 5-hole probe.
Figure 2.19. Pitch and yaw angle calibration map.
already known flow angles, will yield $U$, $V$, and $W$.

\[ S = \frac{p_0 - p_1}{p_1 - p_m} \quad (2.7) \]

\[ Q = \frac{p_1 - p_m}{p_0 - p_s}. \quad (2.8) \]

These calibration maps for static and total pressure are seen in Fig. 2.20 and 2.21. Thereby interpolating the known flow angles, the local total pressure, $p_0$ and the local static pressure, $p_s$, can be found from Eqns. 2.7 and 2.8.

The five-hole Pitot probe was then used in the experiment to measure the 3D velocity components in the boundary layer downstream of the PSVGs and the passive VGs at a chord location $\frac{x}{c} = 0.7$. This location also corresponds to the flap cove location and was 0.1 $c$ downstream of the aft edge of the PSVG electrodes. The probe was moved in 0.5 mm increments along the span, $z$, using the micrometer stages seen in Figure 2.22 a total distance of one flow control device spacing, $\lambda = 22.23$ mm (7/8 in.) The pair of electrodes closest to the quarter span was the range of traversing so any 3D flow from the splitter plates was avoided. The centerline was not chosen as the PSVG electrodes spacing here was not 22.23 mm because of Teflon screws securing the dielectric to the airfoil surface.

2.14.2 Quantifying Vorticity

The height of the five-hole probe above the airfoil surface was kept equal to the local boundary layer displacement thickness, $y = \delta^*$, which changed for different angles of attack. Because 3D velocity was measured only along one line, and not in a plane, the vorticity cannot be found from taking the curl of a grid of velocity data. However, for any boundary layer it can be shown that the centroid of spanwise vorticity occurs at the displacement thickness [18]. By measuring at this location, the five-hole probe is traversed in the span direction through the core of each pair of counter-rotating
Figure 2.20. Static pressure calibration map.

Figure 2.21. Dynamic pressure calibration map.
Figure 2.22. Five-hole Pitot traverse micrometer stages.
vortices. When traversing through the vortex centerline, the maximum of the gradient of wall normal velocity in the span direction, $\frac{\partial V}{\partial z}$, is equal to the actual vorticity, $\omega_x$, since the other component, $\frac{\partial W}{\partial y} = 0$ here. So by subtracting the maximum slope from the minimum slope of wall normal velocity, the streamwise vorticity is measured, $$(\frac{\partial V}{\partial z})_{\text{max}} - (\frac{\partial V}{\partial z})_{\text{min}} \equiv \frac{\partial V}{\partial z}_{p-p} = 2\omega_x.$$ At the highest angle of attack, $13^\circ$, it will be seen in Section 4.4.1 that the velocity is not periodic per $\lambda$ so the previous equation would not quantify the maximum vorticity. In this case $2(\frac{\partial V}{\partial z})_{\text{max}}$ is used to quantify streamwise vorticity.

In the vorticity transport equation an important source of streamwise vorticity is the tilting or reorientation term, $\omega_z \frac{\partial U}{\partial z}$. If you have spanwise vorticity, $\omega_z$, it can be reoriented by variation in the stream component of velocity, $\frac{\partial U}{\partial z}$. Due to viscosity at the airfoil surface and the boundary layer there definitely exists streamwise vorticity. So the ability of each flow control device to produce a spanwise gradient of stream component velocity is another quantity measured by the five-hole Pitot probe presented in Section 4.4. Just like the maximum gradient of wall normal velocity was used to quantify streamwise vorticity, the maximum gradient of stream component velocity was calculated to quantify the ability of each actuator to reorient spanwise vorticity, $$(\frac{\partial U}{\partial z})_{\text{max}} - (\frac{\partial U}{\partial z})_{\text{min}} \equiv \frac{\partial U}{\partial z}_{p-p}.$$
2.15 Particle Image Velocimetry

Digital PIV measurements were also performed for Mach numbers up to 0.3 in order to provide further documentation of the flow reattachment produced by the plasma flow control. The laser sheet was introduced through a window in the test section on the suction side of the model and originated from a Litron LDY 300 Series CW Nd:YLF (neodymium-doped yttrium lithium fluoride) laser with Q-switching for pulsed output. This dual cavity laser is well suited for PIV at high flow speeds since the time delay between laser pulse can be very short. Each laser pulse was 150 ns in duration. The digital PIV camera (LaVision High Speed Star 6) was located on top of the test section. Its focal plane was equal to the laser sheet, and perpendicular to the suction surface of the model. The viewing window for the PIV camera was restricted somewhat by the top load measuring platform supporting the model. As a result, the PIV measurement region was restricted to the leading approximately 30% of the model. Seeding became difficult at these speeds limiting the accuracy of quantifying the local flow velocities. Thus only the streamlines are presented as a means of flow visualization.
Figure 2.23. Schematic of test section with EET airfoil and PIV system installed.
This chapter presents all the experimental data including baseline aerodynamics, differences in stall characteristics of the airfoil with flow control, and confirmation of reattachment with surface pressure and PIV measurements to the highest freestream speeds tested. In addition the effect of dielectric thickness and unsteady operation is presented in addition to both AC and NP type waveform generated plasmas and their effect on aerodynamics in stall.

3.1 Baseline Aerodynamics

First the baseline aerodynamic quantities were found for the EET airfoil with the plasma actuator installed but not turned on. This was to include any effects of roughness due to exposed copper electrode and transition from the Macor leading edge to the Aluminum airfoil body. That way, the only change with flow control was powering on the plasma actuator. Data was taken for a range of Mach numbers between 0.10 and 0.40 in 0.05 increments, and a range of angles of attack from $-5^\circ$ to $25^\circ$ in $1^\circ$ increments. With an airfoil model of chord 30.48 cm at atmospheric conditions, this corresponded to a Reynolds number range of 560,000 through 2,240,000.

The results of the baseline measurements are shown in Figures 3.1–3.3 for the Macor leading edge without plasma. These correspond to the mean and standard deviation (fluctuation) levels of the lift, drag, and moment coefficients. The different symbols correspond to the different freestream Mach numbers in the test section.
The Mach number was measured by a Pitot-static probe that was located in the test section approximately 1 m upstream of the airfoil leading edge.

The baseline mean lift coefficient at the different Mach numbers are shown in the left part of Figure 3.1. Each of the distributions have a linear lift versus angle of attack region with a slope, \( C_{l\alpha} = dC_l/d\alpha \). Drawn for reference is a line with \( C_{l\alpha} = 0.11/° \) that corresponds to 2-D thin airfoil theory. Generally, \( C_{l\alpha} \) is a function of Mach number in proportion to \((1 - M^2)^{-1/2}\). Therefore \( C_{l\alpha} \) increases as Mach number increases, with the point of pivot being at the zero-lift angle of attack, \( \alpha_{0L} \). This effect of compressibility was evident in the baseline lift coefficient measurements in Figure 3.1.

A comparison of \( C_{l\alpha} \) and \( \alpha_s \) from the present experiment to that of others for the same EET section shape, are tabulated in Table 3.1. These other experiments are generally at much lower Mach numbers and Reynolds numbers. To account for differences in Mach number between the experiments, the equivalent incompressible lift slope, \((C_{l\alpha})_0\), was computed using the Prandtl-Glauert correction. Based on this, the average incompressible lift slope measured in the present experiment is identical to the average incompressible lift slope measured by Melton at al. and Little et al. \[29\], namely \((C_{l\alpha})_0 = 0.0870/°\). This is a value lower than that given by thin airfoil theory, and likely due to the amount of camber in the EET section shape. The present work and that of Little agree on \( \alpha_s \) and \( C_{l\max} \). In the Melton experiment, the EET airfoil was equipped with a leading edge slat that was not part of the present experiment or that of Little. Roughness at the slat/main element junction undoubtedly led to the higher stall angle and maximum lift coefficient in the Melton experiment compared to the others.

The value of \( \alpha_{0L} \) is approximately \(-4°\) which is close to the published value of \(-3°\). The stall angle of attack, \( \alpha_s \), varies with Mach number and possibly Reynolds number.
TABLE 3.1

COMPARISON OF EXPERIMENTAL LIFT SLOPE, STALL ANGLE,
AND MAXIMUM LIFT FOR EET AIRFOIL

<table>
<thead>
<tr>
<th>Experiment</th>
<th>$Re \times 10^6$</th>
<th>$M$</th>
<th>$C_{t\alpha} [1/^\circ]$</th>
<th>$(C_{t\alpha})_0$</th>
<th>$\alpha_s^\circ$</th>
<th>$C_{l_{\text{max}}}$</th>
<th>$(C_{l_{\text{max}}})_0$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Kelley</td>
<td>0.560</td>
<td>0.1</td>
<td>0.082</td>
<td>0.082</td>
<td>10</td>
<td>1.2</td>
<td>1.19</td>
</tr>
<tr>
<td>Kelley</td>
<td>2.240</td>
<td>0.4</td>
<td>0.100</td>
<td>0.092</td>
<td>10</td>
<td>1.25</td>
<td>1.15</td>
</tr>
<tr>
<td>Melton et al. [33]</td>
<td>0.750</td>
<td>0.08</td>
<td>0.096</td>
<td>0.096</td>
<td>12</td>
<td>1.6</td>
<td>1.59</td>
</tr>
<tr>
<td>Little et al. [29]</td>
<td>0.750</td>
<td>0.13</td>
<td>0.079</td>
<td>0.078</td>
<td>10</td>
<td>1.3</td>
<td>1.29</td>
</tr>
</tbody>
</table>

A sensitive indication of $\alpha_s$ is the sharp increase in the fluctuations in lift. This can be observed in the right plot in Figure 3.1 which shows the standard deviation of the lift coefficient fluctuations, $C'_l$. At the lower Mach numbers, $0.10 \leq M \leq 0.20$, $\alpha_s = 10^\circ$.

For intermediate Mach numbers in the range $0.25 \leq M \leq 0.30$, $\alpha_s$ increased to $11^\circ$. In addition, the post-stall lift coefficient varied considerably and appeared to alternate between the $C_l$ values at the lower Mach numbers ($M \leq 0.2$) and the higher Mach number ($M \geq 0.35$). The supposition is that the boundary layer flow over the airfoil switched from leading edge separated to trailing edge separated as a result of turbulent transition at the higher Reynolds number.

For the higher Mach numbers, $\alpha_s$ was again $10^\circ$. The supposition here is that the boundary layer flow over the airfoil returned to leading edge separated, which was due to a Mach number effect. In this case, the post-stall lift coefficient did not exhibit the sharp drop that occurred at the lower Mach numbers.

The baseline mean and fluctuating drag coefficient distributions at the different Mach numbers are shown in the left and right parts of Figure 3.2 respectively. In the lower Mach number range, $0.10 \leq M \leq 0.20$, the mean drag coefficient shows a
steep increase at $\alpha = 10^\circ$ that agrees with $\alpha_s$ obtained from the mean lift coefficient. This is accompanied by a jump in the drag fluctuations. As also indicated by the mean lift coefficient, based on the mean drag, $\alpha_s$ increases to $11^\circ$ at the intermediate Mach numbers, $0.25 \leq M \leq 0.30$, and decreases to possibly as low as $7^\circ$ at the higher Mach numbers, $M \geq 0.35$. In all cases the steep rise in drag at the stall angle is to be attributed to form drag. When the airfoil stalls, the pressure recovery is greatly reduced over the suction surface of the airfoil, so the aft portion (where $\frac{dz}{dx} < 0$) contributes a significant pressure drag.

The baseline mean and fluctuating moment coefficient distributions at the different Mach numbers are shown in the left and right parts of Figure 3.3 respectively. Negative values are indicative of nose down direction torque. The maximum in the mean moment coefficient is another good indication of the value of $\alpha_s$. Beyond this critical angle, the pressure recovery is greatly reduced over the entire suction surface of the airfoil, contributing the the dramatic increase in the nose down pitching moment (reduction in $C_{m_{\alpha}}$). Again at the lower Mach number range, $0.10 \leq M \leq 0.20$, $\alpha_s = 10^\circ$. This is again observed to increase to $\alpha_s = 11^\circ$ at the intermediate Mach numbers, $0.25 \leq M \leq 0.30$, and then to decrease to $\alpha_s = 10^\circ$ at $M \geq 0.35$. The sharp increase in the moment fluctuations exhibited are a particular concern on a wing since it could couple with aeroelastic motions and lead to flutter.
Figure 3.1. Mean lift coefficient (left) and RMS of lift coefficient fluctuations (right) versus angle of attack for the baseline EET airfoil as a function of Mach number.

Figure 3.2. Mean drag coefficient (left) and RMS of drag coefficient fluctuations (right) versus angle of attack for the baseline EET airfoil as a function of Mach number.
Figure 3.3. Mean $c/4$-moment coefficient (left) and RMS of $c/4$-moment coefficient fluctuations (right) versus angle of attack for the baseline EET airfoil as a function of Mach number.
3.2 Thick Macor Dielectric Leading Edge

The thick Macor leading edge provided the capability to see the effect of high voltage operation of the AC type plasma actuator both in steady and unsteady conditions.

3.2.1 Angle of Attack Dependence

A series of measurements of the mean and unsteady lift, drag and pitching moment were performed for a range of angle of attacks that bracketed $\alpha_s$ at Mach numbers from 0.10 to 0.40. The conditions for the AC plasma actuator were kept fixed throughout the measurements. These were an AC frequency for the actuator of 2.3 kHz, and an AC amplitude of $30\text{kV}_{p-p}$. The measurements included both steady and unsteady operation. For the unsteady operation, an $f^+ = 1$ was used, and the duty cycle was maintained at 50%.

The results for the different Mach numbers are presented in Figures 3.4 to 3.10. They include the mean lift, drag and moment coefficients as a function of angle of attack that are plotted on the left side of the figures, and the standard deviation of the lift, drag, and moment coefficient fluctuations that are plotted on the right side of the figures. For each quantity, the baselines, along with the steady and unsteady AC plasma actuator results are included. A negative moment is defined as the nose-down direction.

At Mach 0.1 in Figure 3.4, both the steady and unsteady plasma actuator perform the same up to approximately 2° past $\alpha_s$. Beyond that angle of attack, the unsteady performs better. This is particularly evident in the lift and moment coefficients. The moment coefficient is a very sensitive indicator of the amount of peak pressure recovery at the leading edge, and as such is a good metric for judging if the plasma actuator attached the boundary layer at the leading edge.

The results in the range of Mach numbers from 0.15 to 0.25 are shown in Figures
Figure 3.4. Mean (left) and RMS (right) of the lift, drag, and moment coefficients versus angle of attack at Mach 0.10 for Macor leading edge.

Figure 3.5. Mean (left) and RMS (right) of the lift, drag, and moment coefficients versus angle of attack at Mach 0.15 for Macor leading edge.
Figure 3.6. Mean (left) and RMS (right) of the lift, drag, and moment coefficients versus angle of attack at Mach 0.20 for Macor leading edge.

3.5–3.7. These show comparable trends with both the steady and unsteady plasma actuators increasing $C_{l_{\text{max}}}$ and $\alpha_s$. As before, the differences between the steady and unsteady operation only manifest themselves at the highest post-stall angles of attack, where again the unsteady performs better. The standard deviations of the time series of the aerodynamic quantities remain flat until the stall angle of attack is reached. Deep in stall, $\alpha > 15^\circ$, the fluctuating quantities eventually exceed the baseline fluctuations, despite having higher steady lift and moment.

The plasma actuator continues to perform well at the higher Mach numbers above 0.3. These results are shown in Figures 3.8 to 3.10. There it continues to increase $C_{l_{\text{max}}}$ above the baseline, and to flatten the lift-stall character, maintaining a significantly higher lift coefficient compared to the baseline as far as $6^\circ$ past $\alpha_s$. Also notable is the significant increase in the pitching moment over the baseline values at the highest angles of attack. This is a strong indication that the plasma actuator is attaching the boundary layer at the leading edge.
Figure 3.7. Mean (left) and RMS (right) of the lift, drag, and moment coefficients versus angle of attack at Mach 0.25 for Macor leading edge.

Figure 3.8. Mean (left) and RMS (right) of the lift, drag, and moment coefficients versus angle of attack at Mach 0.30 for Macor leading edge.
Figure 3.9. Mean (left) and RMS (right) of the lift, drag, and moment coefficients versus angle of attack at Mach 0.35 for Macor leading edge.

Figure 3.10. Mean (left) and RMS (right) of the lift, drag, and moment coefficients versus angle of attack at Mach 0.40 for Macor leading edge.
3.2.2 Unsteady Flow Control

The effect of the AC plasma actuator was evaluated for both steady and unsteady operation. Steady operation consisted of operating the plasma actuator at a constant AC voltage amplitude. Unsteady operation consisted of switching the AC amplitude on and off at a frequency, $f$, that was lower than the AC frequency.

The effect of the unsteady forcing was evaluated by measuring the aerodynamic forces and moment on the airfoil at an angle of attack of $15^\circ$, and a Mach number of 0.25. This angle of attack was then $4^\circ$ past the largest measured stall angle of attack ($11^\circ$) of the baseline airfoil. The AC frequency for the plasma actuator was 2.3 kHz. A 50% duty cycle was maintained as the unsteady frequency was varied. The AC amplitude was $30 \text{kV}_{p-p}$.

The results are shown in Fig. 3.11. For these, the unsteady frequency, $f$, is normalized by the local freestream velocity, $U_\infty$ and the airfoil chord length, $c$, giving a dimensionless frequency, $f^+ \equiv fc/U_\infty$. Two horizontal lines are included in the plots. The solid line corresponds to the respective quantities for the baseline airfoil. The dashed line corresponds to those quantities with the AC plasma actuator with the same AC voltage and frequency, but in steady operation.

The steady plasma actuator operation produced a marked increase in the mean lift and moment coefficients that is consistent with an attached flow at the post-stall angle of attack. Possibly surprisingly, the steady, and unsteady, plasma actuation produced a slight increase in the mean drag coefficient, although the lift-to-drag ratio with the steady plasma actuation increased by 20%.

The unsteady plasma actuator operation showed a clear frequency dependence, particularly in the mean lift and moment coefficients. Based on the mean lift coefficient, the optimum is at $f^+ \approx 1$. However, the drop-off in the mean lift coefficient at higher $f^+$ values is rather minimal. For the pitch-moment coefficient, less negative values indicate a larger suction pressure loading at the leading edge that would be consistent
with attached flow. Based on this, \( f^+ \simeq 1.8 \) is the clear optimum. However, higher \( f^+ \) values above 1.8 produce about the same effect on the pitch moment.

As with the steady operation, the unsteady operation resulted in an increase in the drag coefficient compared to the baseline. The drag increase however decreases with increasing unsteady frequency. Therefore this suggests another criterion based on the lift-to-drag ratio that takes into account the more rapid drop in the drag compared to lift with increasing unsteady frequency. Based on that criterion, the optimum is \( f^+ \simeq 1.5 \), at which \( C_l/C_d = 2.5 \). The optimum \( f^+ \) based on the different criteria are summarized in Table 3.2.

Overall, at the conditions of the evaluation, the unsteady operation at the respective optimum \( f^+ \) values was more effective in enhancing the aerodynamic properties than the steady operation. The following section evaluates the steady and unsteady performance at the full range of angles of attack and Mach numbers.
Figure 3.11. Mean and fluctuating lift, drag, and moment coefficients and lift to drag ratio as a function of normalized unsteady excitation frequency, $f^+$, for Macor leading edge, $\alpha = 15^\circ$, $M = 0.25$. 
3.2.3 The Influence of Mach Number

A general comparison between the steady and unsteady AC plasma actuator can be made on the basis of the maximum overall lift coefficient, and the maximum difference between the lift coefficient with the plasma actuator and that of the baseline. The former is normally the metric of merit for a leading edge flaps which delay separation to higher angles of attack. The latter is a potential metric of merit for flight control at high angles of attack. The maximum lift coefficient produced by the steady and unsteady AC plasma actuator as a function of Mach number is shown in the left part of Figure 3.12. For this, the maximum lift coefficient is normalized by the baseline lift coefficient at the angle of attack of the respective plasma actuator lift maximum, $\alpha_{C_{l_{\text{max}}}}$. The respective $\alpha_{C_{l_{\text{max}}}}$ values are plotted in the right part of the figure, where the baseline stall angle of attack has been subtracted off to represent the post-stall angles of attack where the maximum lift occurred.

Figure 3.12 reveals that the steady and unsteady AC plasma actuator operation produced a very comparable maximum lift enhancement. Up to a freestream Mach number of 0.25, this resulted in an average 15% increase in the maximum lift. For these, the maximum lift occurred at a 2–4° larger angle of attack than the baseline stall angle. This angle of attack of maximum lift also coincides with that of the smallest negative pitch moment, signifying the largest leading edge suction peak. The maximum lift enhancement and stall angle of attack produced by the AC plasma actuators decreased at Mach numbers 0.3–0.4. At Mach 0.4, the maximum lift coefficient still increased by approximately 5%, with an approximate 1° increase in $\alpha_s$. This is significant for the AC SDBD plasma actuator. For example, the EET airfoil with a mechanical leading edge slat was seen to increase the maximum lift coefficient by 17% at a Reynolds number of $0.75 \times 10^6$ [34].

The other measure of the AC plasma actuator performance is addressed in Figure 3.13. This shows the maximum change in the lift coefficient compared to the baseline
Figure 3.12. Maximum lift coefficient ratio (left) and increase in stall angle of attack (right) as functions of freestream Mach number for thick Macor plasma actuator.

Figure 3.13. Maximum lift coefficient difference with respect to the baseline (left) and post-stall angle of attack of maximum lift coefficient difference (right) as functions of freestream Mach number for thick Macor plasma actuator.
during stall in the left plot, and the angle of attack where this occurs past stall in the right plot. The maximum change in the lift coefficient, $\Delta(C_l)_{max}$, generally occurs at larger baseline post-stall angles of attack as a result of the faster drop in the baseline lift compared to that with the plasma actuator. This is evident by the large values of $\alpha \Delta(C_l)_{max}$ that increase with increasing Mach number. In contrast to this, the trend for $\alpha C_l_{max}$ was the opposite, namely decreasing with increasing Mach number. The point of crossing of the two trends in this case occurred at Mach 0.2, where $\alpha \Delta(C_l)_{max} = \alpha C_l_{max}$. At a $5^\circ$ post-stall angle of attack, the AC plasma actuators ultimately increased the lift coefficient by as much as 0.4 at Mach 0.3 and 0.2 at Mach 0.4. This is significant if the application were flight control during high angle of attack flight.

The results in Figures 3.12 and 3.13 are intended to show the trend with Mach number. For the fixed voltages tested Mach 0.2 showed the best performance of the SDBD plasma actuator as a leading edge flap in increasing the maximum lift. The SDBD plasma actuator acting as a flight control at high angle of attack is also maximized at Mach 0.2. At Mach 0.4, the plasma actuator caused the greatest change in lift at the highest angle of attack. The absolute lift coefficient values will depend on the plasma actuator voltage and dielectric material thickness and properties. The 30 kV$_{p-p}$ voltage used in these experiments was arbitrarily chosen, and conservative. The 3.175 mm (0.125 in.) thick ceramic used as the dielectric layer for the plasma actuator was capable of 3-times higher voltages that would translate into a 3$^{3.5}$-times higher body force [10]. The higher voltage capability of the ceramic leading edge would allow the plasma actuator voltage to increase with increasing freestream Mach number to offset the performance decrease at the highest Mach number tested.
3.3 Thin Kapton Dielectric Leading Edge

The objective of this section is to compare the performance of the AC plasma actuator against that of a nanosecond pulse (NP) plasma actuator for the same experimental conditions. For this comparison, the Macor leading edge of the model was replaced with a new leading edge that was made of a non-electrically-conducting material and covered with a 5 mil (0.127 mm) thick Kapton film. The electrodes were a 0.3 mil (0.007 mm) thick copper layer deposited on the exposed and covered Kapton surfaces. The size and locations of the electrodes was the same as with the plasma actuator on the Macor leading edge. The Kapton dielectric for the NP actuators is most commonly used in the literature [58, 53, 29] and therefore was chosen for this comparison. The AC plasma actuator operation was then tailored to the thin Kapton.

Experiments were performed with the wind tunnel off (no flow) with the NP actuator on and off to determine if the electronic noise was producing any DC voltage in the load sensor output. Some DC was detected that amounted to a few pounds of equivalent force when the load sensor calibration constants were applied. This DC offset was removed from the final results.

An AC actuator also made use of the thin Kapton leading edge. The amplitude to the AC plasma actuator was 12.5 kV<sub>p−p</sub>, and the AC frequency was 2.3 kHz. As mentioned, these were tailored to match the characteristics of the thin Kapton dielectric. For unsteady operation, an <i>f</i><sup>+</sup> = 1 was again used. There was no detectable electronic noise in the sensor signals with operation of the AC plasma actuator.

3.3.1 Unsteady Frequency Dependence

The method for determining the optimal unsteady frequency, <i>f</i>, for the NP plasma actuator followed that used with the AC plasma actuator. It again involved measuring the aerodynamic forces and pitch-moment on the airfoil while operating the actuator at different unsteady frequencies. As before, the airfoil was placed at a post-stall
angle of attack of 15°, and at a freestream Mach number of 0.25.

The results are shown in Fig. 3.14. Here again, the unsteady frequency, $f$, is normalized to give a dimensionless frequency, $f^+ \equiv fc/U_\infty$. The horizontal line corresponds to the respective quantities for the baseline airfoil.

Overall, the frequency selection results for the NP actuator look similar to those of the AC plasma actuator that was shown in Fig. 3.11. With regard to the lift coefficient, the optimum is again at $f^+ = 1$. For the pitch-moment coefficient, $f^+ \simeq 1.8$ is a relative break point in the frequency dependence, above which there is less dependence on the unsteady frequency.

As with the AC plasma actuator, the unsteady NP plasma actuator initially resulted in an increase in the drag coefficient compared to the baseline. The drag however decreases with increasing unsteady frequency such that the lift-to-drag ratio increased with increasing unsteady frequency. Based on that criterion, for the NP plasma actuator, the optimum $f^+$ is at least 2 with this ratio continuing to improve up to the highest frequency tested. This agrees well with previous NP experiments in the literature [50]. The optimum $f^+$ for the NP plasma actuator based on the different criteria are summarized in Table 3.3. The following section evaluates the unsteady performance of the NP and AC plasma actuators under the same conditions for the full range of angles of attack and Mach numbers.

3.3.2 Angle of Attack Dependence

A series of measurements of the mean and unsteady lift, drag and pitching moment were performed to directly compare the performance of the AC and NP plasma actuators for a range of angles of attacks that bracketed $\alpha_s$ at freestream Mach numbers from 0.10 to 0.40. Both plasma actuators used the same 5 mil (0.127 mm) thick Kapton film dielectric layer and electrode arrangement. The only difference
Figure 3.14. Mean and fluctuating lift, drag, and moment coefficients and lift to drag ratio as a function of normalized unsteady excitation frequency, $f^+$, for NP plasma actuator on Kapton LE, $\alpha = 15^\circ$, $M = 0.25$. 
was the actuator power system that was designed either for steady and unsteady AC operation, or designed for unsteady NP operation. For both modes of operation, the conditions for the AC and NP plasma actuators were kept fixed throughout the measurements. For the unsteady operation of the AC plasma actuator, and the full-time operation of the NP plasma actuator, an $f^+ = 1$ was used. This was shown to be most effective for both AC and NP actuators to maximize the lift coefficient. The duty cycle for both actuators was maintained at 50%.

The results for the four different Mach numbers are presented in Figs. 3.15 and 3.16. They include the mean lift, drag, and moment coefficients as a function of angle of attack. For each quantity, the plots include the baseline case, the steady and unsteady AC plasma actuator cases, and the NP plasma actuator case. The baseline case corresponds to the leading edge with the Kapton film covering to account for any differences with that of the Macor leading edge that might be due to differences in surface roughness.

With regard to lift at Mach 0.10 shown in the left part of Fig. 3.15, the steady AC plasma actuator performed the best, achieving the largest increase in $C_l$ for the full range of post-stall angles of attack. The unsteady AC plasma actuator had the next best performance, which was then followed by the NP plasma actuator. With
regard to the pitch moment at the highest (16°) angle of attack, the steady AC plasma actuator again performed the best and followed by the other two in the same order.

At Mach 0.20 shown in the right part of Fig. 3.15, the steady and unsteady AC plasma actuators performed the best and nearly identically with regard to lift and pitch moment coefficients. This is similar to the results at this Mach number with the Macor leading edge. The performance of the NP plasma actuator was below that of the AC plasma actuator, although it was still a notable improvement over the baseline post-stall condition.

The results for Mach 0.30 and 0.40 are shown in Fig. 3.16. At these two higher Mach numbers the performance of the three plasma actuator cases was nearly identical.
Figure 3.16. Mean lift, drag, and moment coefficients versus angle of attack at Mach 0.30 and 0.40 for thin Kapton leading edge.
3.3.3 The Influence of Mach Number

Again a general comparison between the steady and unsteady AC plasma actuator and NP plasma actuator is made on the basis of the maximum overall lift coefficient, and the maximum difference between the lift coefficient with the plasma actuators and that of the baseline as a function of the freestream Mach number. The maximum lift coefficient produced by the steady and unsteady AC plasma actuator and NP plasma actuator as a function of Mach number is shown in the left part of Figure 3.17. As before, the maximum lift coefficient is normalized by the baseline lift coefficient at the angle of attack of the respective plasma actuator lift maximum, $\alpha_{C_{l,max}}$. The $\alpha_{C_{l,max}}$ values are plotted in the right part of the figure, where the baseline stall angle of attack has been subtracted off to represent the post-stall angles of attack where the maximum lift occurred.

Figure 3.17 reveals that steady operation was preferred to unsteady for the AC plasma actuator operation and produced maximum lift enhancement as great as 17% for the Kapton leading edge, down from 22% with the Macor leading edge. The NP plasma actuators did not significantly increase the maximum lift. At Mach 0.4 the maximum lift has still been increased by approximately 8%. Overall the maximum lift angle of attack was increased by as much as 5° at Mach 0.1 and 2° at Mach 0.4.

The other measure of the AC plasma actuator performance is addressed in Figure 3.18. This shows the maximum change in the lift coefficient compared to the baseline in the left plot, and the angle of attack where this occurs (beyond stall) in the right plot. As pointed out earlier, the maximum change in the lift coefficient, $\Delta(C_l)_{max}$, generally occurs at larger baseline post-stall angles of attack as a result of the faster drop in the baseline lift compared to that with the plasma actuator. This is again evident in by the large values of $\alpha_{\Delta(C_{l,max})}$ that increase with increasing Mach number. As with the maximum lift, the AC plasma actuators performed better at the two lowest Mach numbers. However, $\Delta(C_l)_{max}$ and $\alpha_{\Delta(C_{l,max})}$ were very comparable for
all three plasma actuators at Mach 0.3 and 0.4, and nearly identical to the previous results with the Macor leading edge.
Figure 3.17. Maximum lift coefficient ratio (left) and increase in stall angle of attack (right) as functions of freestream Mach number for thin Kapton plasma actuator.

Figure 3.18. Maximum lift coefficient difference with respect to the baseline (left) and post-stall angle of attack of maximum lift coefficient difference (right) as functions of freestream Mach number for thin Kapton plasma actuator.
3.3.4 Flow Reattachment

To confirm leading edge flow reattachment, seven pressure taps measured the surface pressure on a small portion of the suction surface. The distribution of pressure taps is not sufficient to provide lift and drag data through pressure integration, but did provide another way to observe the plasma actuators were effective at reattaching the flow. As seen in Figure 3.19 at pre-stall angles of attack, the plasma actuator has little effect on the surface pressure distribution. However, in stall, Figure 3.20 shows that the plasma actuator reattaches the leading edge separation as evidenced by the pressure suction peak near the leading edge and better pressure recovery further downstream.

PIV data was also collected to observe the flow field near the leading edge of the airfoil with and without flow control. The highest speed that could be visualized with sufficient particle seeding densities in the Whitefield wind tunnel was up to Mach 0.3. Figure 3.22 shows flow from left to right around the leading edge of the airfoil. On the left the actuator is not on and the flow is clearly diverging away from the surface. On the right the flow closely follows the airfoil suction surface, and the stagnation point is further downstream indicating that the flow is attached and the lift has increased. In addition the darker contours are higher velocities.
Figure 3.19. Surface pressure distribution pre-stall at Mach 0.4 and angle of attack 7 degrees for Kapton leading edge.

Figure 3.20. Surface pressure distribution post-stall at Mach 0.4 at angle of attack 13 degrees for Kapton leading edge.
Figure 3.21. Surface pressure distribution Mach 0.10, 0.25, and 0.40 and angle of attack 14 degrees for Kapton leading edge.

Figure 3.22. Time averaged PIV streamlines for no flow control (left) and AC steady flow control (right) at Mach 0.3 and angle of attack 13 degrees for Kapton leading edge.
CHAPTER 4
TRAILING EDGE SEPARATION MEASUREMENTS

This chapter presents the results of experiments quantifying the differences in controlling trailing edge separation on the V-22 airfoil with both passive vane-type vortex generators and plasma streamwise vortex generators. First, the general aerodynamic measurements were taken on the Osprey wing section including lift, drag, and moment up to \( M = 0.2, \text{Re} = 1.59 \times 10^6 \). Next, boundary layer measurements were taken to look at both the velocity profile and the three-dimensionality of the boundary layer and its spanwise variation. These were measured with the previously discussed total pressure and 5-hole Pitot tubes respectively. Measurement locations for the these probes were at various spanwise positions at two different chord locations, \( \zeta = 0.2, 0.7 \), the beginning of the PSVG electrodes and at the flap cove downstream of the flow control region, respectively. By repeating these measurements for various electrode length, Mach numbers, and operating voltages, conclusions are drawn for designing a PSVG with the same performance as a passive VG.

4.1 Aerodynamics

Lift and drag, and the quarter chord moment were measured using the same load cells as before. These aerodynamic quantities were all taken with the flap sealed and with no flap deflection. Figures 4.1-4.2 show these results for both Mach 0.1 and 0.2 respectively. The baseline data is with no flow control device, VG is with the passive vortex generators described in Chapter 2.7 and the various voltages range from 31 to 48 kV\(_{p-p}\) is for the plasma streamwise vortex generators designed in Chapter 2.5.
Figure 4.1. V-22 aerodynamic coefficients at $M = 0.10$.

Figure 4.2. V-22 aerodynamic coefficients at $M = 0.20$. 

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At Mach 0.1, the lift data shows a very slow fall off from thin airfoil theory indicative of a thick airfoil which exhibits trailing edge stall, that is one where flow separation moves forward with angle of attack. The drag coefficient between baseline and all PSVG voltages sees very small changes, but the passive VGs have increased the total drag. This is not expected, so in consideration of this drag increase for the VGs and corner flows observed with flow visualization, the sectional drag is not accurately being measured with the load cells. This would also indicate that the PSVGs could also be having an effect on drag and flow separation over a percentage of the span but is being washed out due to the corner flows. This necessitates an alternate drag measurement technique that can measure the sectional drag rather than the total drag force first to better compare drag changes with each device. The moment coefficients remain positive (nose-up) until 9° for no flow control and PSVGs, and 10° for the VGs. This indicates that the VGs were effective at reducing separation at 10°, since better pressure recovery on the aft section of the wing leads to a nose-up torque.

The aerodynamic data at Mach 0.2 is similar to Mach 0.1 where the lift fall-off is slow, and only the VGs show an ability to suppress separation up to 10°. The fact that all drag data is very close in amplitude necessitates an alternative method for measuring drag. Therefore all subsequent drag measurements will be performed by measuring the momentum deficit in the wake of the airfoil. This wake traverse is performed on the quarter span location so that 3D flow structures, observed from oil based flow visualization near the splitter plates at high angles of attack, will not affect the section drag coefficient measurement.
4.2 LE Boundary Layer and Vorticity Timescales

The boundary layer velocity profile was measured at the chord location \( \xi \equiv 0.2 \) just upstream of where the flow control devices are located. The small Pitot tube was traversed in the wall normal direction to measure the variation in total pressure versus height, while the static pressure was found from the five-hole Pitot tube measurements since static pressure does not vary through the boundary layer height. The velocities were normalized by the local freestream velocity \( U_0 \) and are plotted in Figure 4.3. Increased local velocity at higher angles of attack has thinned the boundary layer and increased the shear at the wall.

![Figure 4.3. Boundary layer profile at beginning of flow control region \((\xi = 0.2).\)
Table 4.1 presents the important boundary layer parameters measured and tabulated from the profiles in Figure 4.3 including boundary layer height, $\delta$, displacement thickness, $\delta^*$, momentum thickness, $\theta$, shape factor, $H = \delta^*/\theta$, and the friction velocity, $u_\tau$. These values were found by integrating a spline fit of the velocity profile. The boundary layer thickness is quite small, both around 0.5 mm. The displacement thickness is important for the five-hole Pitot tube measurements, since this height is the location of the centroid of spanwise vorticity. The range of shape factors measured, 2.33–2.19, would indicate a laminar boundary layer as expected close to the airfoil leading edge with some small roughness. If roughness had been further reduced on the leading edge a shape factor of 2.6 would be expected for a Blasius profile. If the boundary layer was turbulent a shape factor closer to 1.4 would be typical \cite{54}. The shear velocities are important for the time scales associated with the production of streamwise vorticity through reorientation of spanwise vorticity in the boundary layer. This increased shear at higher angles of attack is the first indicator of why PSVGs perform better than passive VGs at higher angles of attack since $\Delta \omega_x \propto u_\tau$.

The ability to measure enough data points near the wall for an accurate slope can be difficult. For a Blasius profile at the 20\% chord, the shear velocity would be 1.07 m/s. Shear velocities 1/30th the freestream velocity are common in a turbulent boundary layer \cite{18}, if that were the case, the friction velocity would be 1.13 m/s. Since the measured values range from 1.76–2.31 m/s and are close to those predicted for a laminar or turbulent boundary layer, this indicates that the total pressure probe was adequately close enough to the airfoil to measure the maximum shear.

Now that the incoming boundary layer conditions are quantified, some physics about the operation of the PSVGs can be quantified. As argued in Wicks et al. \cite{68}, a set of scaling laws can describe the relative time scales associated with direct production of streamwise vorticity due (electrohydrodynamic wind), $T_P$, and that
TABLE 4.1

SUMMARY OF BOUNDARY LAYER PROFILE AT BEGINNING OF FLOW CONTROL REGION \((\bar{z} = 0.2)\) AT MACH 0.1

<table>
<thead>
<tr>
<th>(\alpha^\circ)</th>
<th>(\delta) [mm]</th>
<th>(\delta^*) [mm]</th>
<th>(\theta) [mm]</th>
<th>H</th>
<th>(u_r) [m/s]</th>
</tr>
</thead>
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<tr>
<td>0</td>
<td>0.53</td>
<td>0.18</td>
<td>0.08</td>
<td>2.33</td>
<td>1.76</td>
</tr>
<tr>
<td>13</td>
<td>0.49</td>
<td>0.13</td>
<td>0.06</td>
<td>2.19</td>
<td>2.31</td>
</tr>
</tbody>
</table>

TABLE 4.2

TIME SCALES ASSOCIATED WITH VORTICITY PRODUCTION AND REORIENTATION

<table>
<thead>
<tr>
<th>M</th>
<th>(\alpha^\circ)</th>
<th>(T_{Res}) [ms]</th>
<th>(T_p) [ms]</th>
<th>(T_{VR}) [ms]</th>
<th>(\frac{T_p}{T_{VR}})</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.1</td>
<td>0</td>
<td>4.17</td>
<td>0.04</td>
<td>0.10</td>
<td>0.40</td>
</tr>
<tr>
<td></td>
<td>13</td>
<td></td>
<td>0.03</td>
<td>0.06</td>
<td>0.50</td>
</tr>
<tr>
<td>0.2</td>
<td>0</td>
<td>2.10</td>
<td>0.04</td>
<td>0.07</td>
<td>0.57</td>
</tr>
<tr>
<td></td>
<td>13</td>
<td></td>
<td>0.03</td>
<td>0.04</td>
<td>0.75</td>
</tr>
</tbody>
</table>

which is reoriented from the boundary layer, \(T_{VR}\).

Both of these should occur faster than the time it takes for freestream momentum to move the length of the PSVG actuator in order for the streamwise vorticity to increase proportionally with velocity, “regime 1.” Table 4.2 tabulates these three time scales for the range of Mach numbers and angle of attacks tested according to Equations 1.1–1.3.

For the PSVGs as designed with a length of \(l = 14.2\) cm, electrode spacings of
\[ \lambda = 2.22 \text{ cm}, \] and the previously calculated boundary layer parameters, the time scales associated with both streamwise vorticity production and vorticity realigned from the spanwise vorticity in the boundary layer occurs at least 30 times faster than the residence time of freestream momentum. This means that, according to the scaling laws, the PSVG electrodes are of sufficient length to avoid \( 1/U_\infty \) in “regime 2.”

The production timescale, \( T_P = \frac{\delta^*}{v_P} \), depends on the displacement thickness and induced velocity both of which show no change with increased freestream velocity. However, at higher angles of attack, the local flow velocity near the leading edge increases, and thus \( \delta^* \) slightly thinned, and hence the production timescale decreases accordingly. So the production timescale is not affected by the freestream velocity and just slightly decreased by higher angles of attack.

The vorticity reorientation timescale, \( T_{VR} = \frac{\delta^*}{u_\tau} \), depends on the displacement thickness and the shear velocity. Both higher angles of attack and higher freestream velocities increase the shear stress along the wall near the leading edge of the airfoil because of the local suction peak and thus both lead to reduced timescales. Therefore it is an advantage on a thick airfoil that the local velocity increases with angle of attack where the flow control region starts because the boundary layer is thinned despite the adverse pressure gradient increasing over the PSVGs with increasing angle of attack.

Finally, the ratio of production to reorientation timescale is tabulated in Table 4.2. Both freestream velocity and angle of attack is observed to increase this value. This means that the vorticity reorientation timescale is reduced more so than the production timescale for the same increase in Mach number or angle of attack. This also highlights that both of these processes does not occur instantaneously relative to one another and both mechanisms are important for the current application of PSVGs to create streamwise vorticity and aid separation control. Section 4.4 investigates these predictions of the time scales by quantifying the streamwise vorticity.
4.3 TE Boundary Layer

The boundary layer pressure profile was measured at the chord location $\xi/c = 0.7$, 0.1c downstream of where the PSVGs end and at a span location halfway in between electrodes. Flow visualization indicated that separation had extended the length of the flap at $\alpha = 8^\circ$. The boundary layer profile with and without the PSVGs being operated is seen in Figure 4.4. As expected the profile is just barely inflectional indicating that flow separation is imminent at this chord location, consistent with flow visualization. When the PSVGs are turned on, the boundary layer is thickened. Since the measurement location is halfway between electrodes, this is where there is an induced vertical velocity exists thickens the boundary layer. When the angle of attack is further increased to $\alpha = 13^\circ$, flow separation moves forward on the suction surface, approximately to location of maximum thickness, $\xi/c = 0.35$, the middle of the wing main element. The total pressure profile at an angle of attack of $13^\circ$ can be seen in Figure 4.5. Because the profile is inflectional without flow control and only concave left with the PSVGs on, there is good indication that the PSVGs were successful at reattaching the boundary layer at least to $\xi/c = 0.7$ at $M = 0.1$ and $\alpha = 13^\circ$. The ability of the PSVGs reduce form drag at other angles of attack and Mach numbers is discussed in detail in Section 4.5.
Figure 4.4. Boundary layer profile at flap cove, $\alpha = 8^\circ$.

Figure 4.5. Boundary layer profile at flap cove, $\alpha = 13^\circ$. 
4.4 Vorticity Production and Reorientation

This section presents the results of the 5-hole Pitot probe measurements downstream of the PSVGs and VGs. First the 3D velocity components are measured in the boundary layer without and then with the two flow control devices. The 3D flow structures are then reduced to vortical structures by which the streamwise vorticity and the reorientation term of vorticity transport equation are quantified. This vorticity is important to the transport of high momentum fluid down into the boundary layer to prevent separation and reduce drag. Next the effects of different pressure gradients by changing angle of attack is analyzed, as is Mach number on vorticity production. Finally different length electrodes are used with the PSVG to answer the question related to the scaling laws and timescales proposed by Wicks et al.

4.4.1 3D Velocity Components in Boundary Layer

The five-hole Pitot probe was traversed along the span a total distance one electrode spacing, $\lambda = 22$ mm, at a height equal to the the displacement thickness, $y = \delta^*$, at the flap cove, $\frac{c}{\delta} = 0.7$, for a range of angles of attack, $4^\circ$, $6^\circ$, $8^\circ$, and $13^\circ$ at Mach 0.1, and $4^\circ$, $6^\circ$, and $13^\circ$ at Mach 0.2 for Figures 4.6–4.12.

At all angles of attack below $13^\circ$, the freestream component of velocity, $U$, the airfoil surface normal component, $V$, and the spanwise velocity component, $W$, are all periodic with the PSVGs operating as expected by traversing through the center of a pair of counter-rotating streamwise vortices. The edge of each electrode is on the outer bounds of the traversed range however the maximum upwelling created in the boundary layer, when $V$ is most positive is not in the center. However, since there is $5^\circ$ of forward sweep this led to the baseline $W$ not equalling zero especially evident at Mach 0.2. So there is a shift of the vortex pair slightly along the span. The peak is observed at $z = 6$ mm, and so the vortex pair moved 17 mm laterally across the
span, with the center of the adjacent electrodes being at -11.1 mm. So the measured coherent vortical structure originated from the adjacent PSVG electrode pair.

The effect of voltage is clearly visible in changing the amplitude of velocity difference. When the slopes of the trends are calculated this will lead to larger vorticity and reorientation with higher voltage as expected when the body force increases. As the angle of attack is increased to 13°, the flow becomes highly three-dimensional as seen in Figure 4.9. The crossflow component, $W$, has increased to an average 14 m/s of crossflow, almost half the freestream velocity. The periodic velocity distribution has also been eliminated at this highest angle of attack, indicating that turbulence and the building adverse pressure gradient has broken apart the large coherent vortical structures previously seen in the boundary layer at lower angles of attack.

The passive VGs also produce a pair of counter-rotating vortices. However since their streamwise vorticity production is nearly instantaneous along their short length of 0.035c, the pairs of coherent vortical structures are not existing from $\frac{x}{c} = 0.2$ to the measurement location of the five-hole probe at $\frac{x}{c} = 0.7$ at even the smallest angles of attack. However the spatial gradient of velocity is still indicative of the streamwise vorticity produced.
Figure 4.6. 3D velocity components downstream of PSVGs ($\frac{\alpha}{c} = 0.7$), $\alpha = 4^\circ$, at Mach 0.1.

Figure 4.7. 3D velocity components downstream of PSVGs ($\frac{\alpha}{c} = 0.7$), $\alpha = 6^\circ$, at Mach 0.1.
Figure 4.8. 3D velocity components downstream of PSVGs ($\frac{x}{c} = 0.7$),
$\alpha = 8^\circ$, at Mach 0.1.

Figure 4.9. 3D velocity components downstream of PSVGs ($\frac{x}{c} = 0.7$),
$\alpha = 13^\circ$, at Mach 0.1.
Figure 4.10. 3D velocity components downstream of PSVGs \( \left( \frac{x}{c} = 0.7 \right) \), \( \alpha = 4^\circ \), at Mach 0.2.

Figure 4.11. 3D velocity components downstream of PSVGs \( \left( \frac{x}{c} = 0.7 \right) \), \( \alpha = 6^\circ \), at Mach 0.2.
Figure 4.12. 3D velocity components downstream of PSVGs ($\frac{x}{c} = 0.7$), $\alpha = 13^\circ$, at Mach 0.2.
4.4.2 Effect of Angle of Attack

Now that the 3D velocity components have been quantified, the following results examine the vorticity in the boundary layer. The angle of attack of the airfoil was changed to investigate how different pressure gradients affected the operation of each flow control device. One would expect an adverse pressure gradient to reduce the streamwise vorticity since a slowing of velocity in the stream direction causes compression of a vortex filament and a reduction in vorticity, typically explained by angular momentum principles and the vorticity stretching term, \( \frac{D\omega_k}{Dt} = \omega_j \frac{\partial U}{\partial x_j} + \ldots \)

Since the adverse pressure gradient increases with angle of attack, one would then expect both the PSVG and passive VG vorticity production to decrease with angle of attack. This section addresses this preconception and highlights how the PSVG has different flow physics than the passive VG.

The maximum streamwise vorticity per flow control device spacing has vorticity estimated as \((\frac{\partial V}{\partial z})_{max} - (\frac{\partial V}{\partial z})_{min} \equiv \frac{\partial V}{\partial z}_p - p\). This is found from the velocity measurements in Section 4.4.1 at a location downstream of the PSVGs and wing main element, \( \xi_c = 0.7 \). This vorticity is plotted for the baseline flow, with passive VGs, and PSVGs operating at different voltages in Figure 4.13. Since there is an observed increase in baseline vorticity with angle of attack without the use of a flow control device, its value should be subtracted off to see solely the effect of the flow control devices and their ability to change streamwise vorticity, \( \Delta \frac{\partial V}{\partial z}_p - p \). The increasing vorticity with angle of attack is explained by wing sweep and the increase in crossflow with angle of attack.

The ability of each flow control device to produce streamwise vorticity in excess of the baseline flow is plotted for increasing pressure gradients, or similarly angles of attack, in Figure 4.14. For the passive VG, the vorticity stretching analogy holds: as the adverse pressure gradient increases, the streamwise vorticity is reduced. However, the case for the PSVG is actually quite different, it seems there is an increase in
streamwise vorticity as the pressure gradient becomes larger, at least up to $8^\circ$. Overall, the passive VGs could produce the most streamwise vorticity at Mach 0.1 and 0.2 for smaller angles of attack, 4–8°. At highest angle of attack the PSVGs appear to produce more vorticity than the VGs for the given voltages. To answer this question of PSVGs producing greater vorticity with higher adverse pressure gradients, the reorientation term of the vorticity transport equation was measured.

Since $\frac{\partial U}{\partial z}$ multiplies by $\omega_z$ in the x-component of the vorticity transport equation, a spanwise variation of $U$, produces a tilting or reorientation of $\omega_z$ in the boundary layer, to streamwise vorticity $\omega_x$. The vorticity reorientation term $\frac{\partial U}{\partial z} p - p$ was calculated and plotted for various angles of attack in Figure 4.15 according to the formulas discussed in Section 2.14.2 and quantifies how much this contributes to streamwise vorticity. Since there is naturally some spanwise variation in $U$ even without flow control, its value should be subtracted off to see solely the effect of the flow control devices and their ability to reorient spanwise vorticity, $\Delta \frac{\partial U}{\partial z} p - p$ seen in Figure 4.16. At Mach 0.1, the two highest voltages tested show that the PSVGs greater spanwise gradient of the freestream component of velocity. At Mach 0.2, the PSVGs only outperform the passive VGs at the highest angle of attack tested. Nonetheless, this highlights how vorticity reorientation for the PSVGs falls off at a slower rate with angle of attack than the VGs which is perhaps a possible explanation to vorticity increasing with higher angles of attack for the PSVGs and also contributes to the argument that PSVGs are better suited to separation control at higher adverse pressure gradients on an airfoil.
Figure 4.13. Effect of angle of attack on vorticity production at Mach 0.1 (left) and Mach 0.2 (right).

Figure 4.14. Effect of angle of attack on change in vorticity production at Mach 0.1 (left) and Mach 0.2 (right).
Figure 4.15. Effect of angle of attack on vorticity reorientation at Mach 0.1 (left) and Mach 0.2 (right).

Figure 4.16. Effect of angle of attack on change in vorticity reorientation at Mach 0.1 (left) and Mach 0.2 (right).
4.4.3 Effect Mach Number and Electrode Length

By increasing the freestream velocity from Mach 0.1 to 0.2 in 0.05 increments, along with varying the PSVG electrode lengths, the ability to produce streamwise vorticity and create vorticity reorientation are quantified. As seen in Figures 4.17 and 4.18, the overall trend is for the production of streamwise vorticity to linearly decrease with Mach number, indicating that at these low angles of attack (4° and 6°), the PSVG is operating in “Regime 2” mode of operation. This plot shows how a longer electrode can help compensate for the loss in vorticity with Mach number. The data point at Mach 0.1 with the 100% electrode was taken without the five-hole probe adequately braced in the wind tunnel, and hence its vibration is the likely cause of the outlier in the trend. The trends are very similar between the two angles of attack, with just slightly higher vorticity production at 6° versus 4°.

The ability of the PSVGs to reorient spanwise vorticity with increasing Mach numbers is plotted in Figures 4.19 and 4.20 for angles of attack 4° and 6° respectively. It is interesting to see that in this case, a higher freestream Mach number lowers the ability of the PSVGs to reorient vorticity. This is in contrast to the previous section which examined different angles of attack. There, at higher angles of attack, the local velocity at the start of the flow control region and all across the electrodes, caused an increase in the reorientation effectiveness. Therefore it should be concluded that stronger adverse pressure gradients, and not the local fluid velocity allows the PSVGs to better reorient vorticity.
Figure 4.17. Effect of PSVG electrode length on peak to peak vorticity production, $\Delta(\frac{\partial V}{\partial z})_{p-p}$, at $\alpha = 4^\circ$.

Figure 4.18. Effect of PSVG electrode length on peak to peak vorticity production, $\Delta(\frac{\partial V}{\partial z})_{p-p}$, at $\alpha = 6^\circ$. 
Figure 4.19. Effect of PSVG electrode length on peak to peak vorticity reorientation, $\Delta (\partial U / \partial z)_{p-p}$, at $\alpha = 4^\circ$.

Figure 4.20. Effect of PSVG electrode length on peak to peak vorticity reorientation, $\Delta (\partial U / \partial z)_{p-p}$, at $\alpha = 6^\circ$. 
4.4.4 Effect of Residence Timescale

Rather than looking at electrode length and freestream Mach number separately, their ratio, the residence timescale, should now be examined to see if there is a collapse of data as predicted by the scaling laws previously discussed. Figures 4.17–4.20 consist of a range of different freestream speeds and electrode lengths, from 0 ms to 4.17 ms. First the ability to produce streamwise vorticity is plotted versus the various residence timescales for $\alpha = 4^\circ$ and $\alpha = 6^\circ$ in Figures 4.21 and 4.22.

For both Mach 0.1 and 0.15 their is an excellent collapse of vorticity production with increasing residence time. Since residence time is proportional to $1/U_\infty$, this is indicative of “Regime 2” mode of operation, where the primary source of boundary layer energization is directly from the body force and induced ionic wind that becomes a smaller and smaller percentage of the freestream velocity. At Mach 0.2 the residence is shorter and a falloff from the collapsed dataset is observed. This would indicate that vorticity production is damped at an even faster rate than $1/U_\infty$ for Mach 0.2 perhaps due to a Reynolds number effect of increased turbulence and mixing of large vortical scales to small scales that dissipate energy due to viscosity.

The collapse of the vorticity reorientation as a function of the residence timescale, seen in Figures 4.23 and 4.24, is good and shows a proportionality to $T_{res}$ and hence is proportional to $1/U_\infty$. Again this would agree with “Regime 2” mode of operation across all Mach numbers and electrode lengths tested for both angles of attack $\alpha = 4^\circ$ and $\alpha = 6^\circ$.

For the present dataset it has been shown that the PSVGs always operate in the regime where increased freestream velocity would damp both the streamwise vorticity production and the vorticity reorientation. However, this is not expected from the previously calculated timescales for production and reorientation seen in Section 4.2, as even at Mach 0.2 these timescales were more than 20 times faster than the residence timescale. Therefore it must be concluded that in this experiment,
Figure 4.21. Effect of PSVG residence time on peak to peak vorticity production, $\Delta (\frac{\partial V}{\partial z})_{p-p}$, at $\alpha = 4^\circ$.

Figure 4.22. Effect of PSVG residence time on peak to peak vorticity production, $\Delta (\frac{\partial V}{\partial z})_{p-p}$, at $\alpha = 6^\circ$. 

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Figure 4.23. Effect of PSVG residence time on peak to peak vorticity reorientation, $\Delta \left( \frac{\partial U}{\partial z} \right)_{p-p}$, at $\alpha = 4^\circ$.

Figure 4.24. Effect of PSVG residence time on peak to peak vorticity reorientation, $\Delta \left( \frac{\partial U}{\partial z} \right)_{p-p}$, at $\alpha = 6^\circ$. 
the complicating factor of pressure gradients does not allow the comparison of the vorticity reorientation timescale to residence timescale to accurately predict the divide between “Regime 1” and “Regime 2.”

4.4.5 Effect of Voltage

The induced body force is known to change proportionally to electric potential to the $7/2$ power for air at atmospheric pressure. So it is expected that higher voltage should produce more vorticity. This effect on the vorticity in the boundary layer is now presented. Figure 4.25 shows a range of voltages between 31 and 48 kV$_{p-p}$ for a range of angles of attack and at Mach 0.1 and 0.2. At Mach 0.1, for angles of attack up to $8^\circ$, the produced streamwise vorticity is quite linear with operating potential. At Mach 0.2, the produced vorticity is essentially constant, and hence not a function of operating voltage. The ability of the PSVGs to produce vorticity reorientation at different voltages seen in Figure 4.26. The slope of the vorticity reorientation with voltage is reduced by Mach number.
Figure 4.25. Effect of voltage on change in vorticity production at Mach 0.1 (left) and Mach 0.2 (right).

Figure 4.26. Effect of voltage on change in vorticity reorientation at Mach 0.1 (left) and Mach 0.2 (right).
4.5 Wake Measurements and Drag Reduction

This section presents the results of wake measurements to quantify the change in drag from using the both the PSVGs and the passive VGs. The effect of angle of attack and Mach number are investigated.

4.5.1 Velocity Wake Measurements

The test matrix for the velocity wake measurements, and hence drag also, consists of a range of Mach numbers 0.1, 0.15, and 0.2 at the angles of attack 0°, 2°, 4°, 6°, 8°, and 13°. The wake profiles are presented in Figure 4.27–4.32. These wake velocity profiles are all normalized by the freestream velocity and the $y$-location is normalized by the chord length. In all cases $y = 0$ is defined as directly downstream of the airfoil trailing edge where the reduction in velocity is greatest for the baseline case. Positive $y$-values are the upper (suction) half of the airfoil and negative $y$-values are the lower (pressure) side of the airfoil.

Up to $\alpha = 8^\circ$, the wake remains narrower than 20% the chord, which is less than the thickness ratio of the airfoil, 23%, indicating that the flow is not separating at the maximum thickness location. Small changes the local wake velocity are noticeable which will be integrated into the drag coefficient. At the highest angle of attack tested, $\alpha = 13^\circ$, there is a noticeable velocity deficit in the wake of the passive VGs not seen with the PSVGs.
Figure 4.27. Airfoil wake 1 chord downstream of trailing edge at $\alpha = 0^\circ$.

Figure 4.28. Airfoil wake 1 chord downstream of trailing edge at $\alpha = 2^\circ$. 

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Figure 4.29. Airfoil wake 1 chord downstream of trailing edge at $\alpha = 4^\circ$.

Figure 4.30. Airfoil wake 1 chord downstream of trailing edge at $\alpha = 6^\circ$.
Figure 4.31. Airfoil wake 1 chord downstream of trailing edge at $\alpha = 8^\circ$.

Figure 4.32. Airfoil wake 1 chord downstream of trailing edge at $\alpha = 13^\circ$. 
4.5.2 Drag Reduction

Now that the wake profiles have been measured, the drag coefficients were calculated according to the Jones method described by Equation 2.4. The change in drag for each flow control device with respect to no flow control was then calculated for all the angles of attack and Mach numbers. These contour plots are seen in Figure 4.33 and Figure 4.34 for the PSVGs and VGs respectively. Because the change in drag coefficient is a small number, the contours represent change in drag count, which is equal to $\Delta C_d \times 10^4$. A negative number is indicative of a drag reduction, and positive a drag increase with respect to the baseline airfoil with no flow control.

For the PSVGs, drag reduction was only possible at the highest angle of attack and at Mach 0.15. This agreed well with the greatest vorticity production with an attached boundary layer with 100% electrode length, seen in Figure 4.17. Interestingly, the PSVG performed the best in drag reduction when turbulence had both damped the vorticity produced at $\alpha = 13^\circ$. However the ability to increase vorticity reorientation with stronger pressure gradients explains how drag reduction is still possible.

The drag reduction contour seen in Figure 4.34 shows that the passive VGs reduced drag the most between the intermediate angles of attack 6-8°. Also the passive VGs show Mach number independence since all the contours are vertical. And that maximum drag count reduction, 40, was equal between the VGs and PSVGs.
Figure 4.33. Change in drag using PSVGs (48 kV_{p-p}) at various angles of attack and Mach numbers.

Figure 4.34. Change in drag using VGs at various angles of attack and Mach numbers.
4.6 Proposed PSVG Design Equivalent to VG

This section proposes how one could design a PSVG that operates equivalently to a passive VG for various flow conditions such as angle of attack and Mach number. The parameters of design that can be adjusted include electrode length and the operating voltage. So by these, one can design a PSVG that produces the same streamwise vorticity as a vane-type VG for the design Mach number and angle of attack.

The issue of turbulent boundary layer separation remains difficult to predict in both experiments and simulations. The breakdown of the streamwise vortical structures is likely due to either boundary layer separation or increasing range of turbulence length scales that lead to mixing in high Reynolds number flow. In this experiment it seems evident that up to vortex breakdown and flow separation, production of streamwise vorticity and reorientation of spanwise vorticity is quite linear with PSVG operating voltage and residence time. And as seen in the analysis of residence time, the influence of freestream velocity in conjunction with electrode length collapsed well for $\alpha = 4^\circ$ and $6^\circ$ up to Mach 0.15 when the longest residence timescale at Mach 0.1 and and the Mach 0.2 data points are also left out of this linearization due to vortex breakdown and turbulence. The slope of each of the vorticity production to residence timescale ratios increased for higher pressure gradients and for a small range of angle of attack will assumed to be linear. Thus a linear fit is performed on Figure 4.21 and 4.22 as described by $(28.15(\alpha^\circ) + 151.2)\frac{l}{U_\infty}$. The constants come from the model of the behavior, not any yet described theory. This equation can be seen plotted with the data in Figures 4.35 and 4.36. Since Mach 0.2 data was left out, this means the production timescale of vorticity needs to be least 50 times smaller than the residence timescale, $T_P \leq \frac{T_{Res}}{50}$ for this design equation to hold. This is in contrast to the flat plate experiment in Wicks et al. which suggested $T_P \leq T_{Res}$.

Figure 4.25 showed a linear and constant slope for all angles of attack when there was not boundary layer separation at the five-hole probe, $\alpha = 4^\circ$–$8^\circ$ at Mach 0.1.
So the change in vorticity linearized with respect to the maximum voltage tested is described by \(16(kV_{p-p} - 48)\). To summarize, the following criteria should be met to use the PSVG design Equation 4.1.

- One would like to produce the same streamwise vorticity as a vane-type VG on the suction surface of an airfoil where the PSVG electrodes start at the same chord location as the VGs for a given freestream velocity.
- Find optimal VG spacing.
- The boundary layer profile is known at the beginning of the flow control region and the three main timescales can be calculated: \(T_{Res}, T_P\), and \(T_{VR}\).
- \(\frac{T_{Res}}{T_P} \geq 50\) to ensure that vorticity scales linearly with residence timescale.
- No boundary layer separation occurs at the chord location where desired vorticity condition is measured.
- The Reynolds number is not exceedingly high enough to cause vortex breakdown.
- For a known change in streamwise vorticity production by using VGs with respect to the baseline (no flow control case) at a design freestream velocity and angle of attack, calculate the combination of electrode length, \(l\) in mm, and operating potential, \(kV_{p-p}\) to produce just as much streamwise vorticity using Equation 4.1

\[
\Delta \omega_x \approx \Delta \left(\frac{\partial V}{\partial z}\right)_{p-p} \approx (28.15(\alpha^\circ) + 151.2)\frac{l}{U_\infty} + 16(kV_{p-p} - 48)
\] (4.1)
Figure 4.35. Effect of residence time on change in vorticity production $\Delta(\frac{\partial V}{\partial z})_{p-p}$, $48 kV_{p-p}$, at $\alpha = 4^\circ$. Horizontal blue lines are for passive VGs.

Figure 4.36. Effect of residence time on change in vorticity production $\Delta(\frac{\partial V}{\partial z})_{p-p}$, $48 kV_{p-p}$, at $\alpha = 6^\circ$. Horizontal blue lines are for passive VGs.
4.6.1 Design Example

The PSVGs did reduce drag for some design conditions when the VGs did not, specifically when $\alpha = 13^\circ$ at Mach 0.15. However, the PSVGs did not reduce drag at lower angles of attack when the VGs did. So by using the proposed design equation, one can calculate how much the voltage or electrode length should be increased to produce the same amount of streamwise vorticity as the VGs and hence have an equal chance at entraining high momentum fluid down into a separated boundary layer region.

For this example, consider the specific case that one would like to design a PSVG equivalent to a VG in both the wind tunnel model ($c = 0.36$ m) and a full scale flight test ($c \approx 2$ m) on the V-22 Osprey wing. The angle of attack is 6°, the freestream speed is Mach 0.15 ($U_\infty = 51$ m/s), and the VGs produce 2218 1/s of vorticity. Using these values in the proposed linearized design equation, the following equality must hold, $2218 = 6.24 l + 16(kV - 48)$. Combinations of voltage and electrode length that satisfy the design equation are plotted in Figure 4.37.

So at the highest voltage tested, 48 kV, the electrode needs to be lengthened to 0.35 m to produce as much streamwise vorticity at Mach 0.15. This is not possible as the chord is 0.36 m. However since it has been shown that the production of vorticity is linearly dependent of the residence timescale, when considering the full scale V-22 wing with a chord of 2 m, now the electrode length required is a much smaller percent of the chord, $l = 0.18 c$. So it seems that PSVGs are much better suited for full scale tests due to vorticity scaling linearly with the residence timescale, $\frac{l}{U_\infty}$ seen in this experiment. Alternatively the voltage could be increased so that a shorter electrode would be required.
Figure 4.37. Design example for combinations of PSVG electrode length and voltage that produce as much streamwise vorticity as passive VG at 6° and Mach 0.15 ($U_\infty = 51$ m/s)
CHAPTER 5

SUMMARY AND CONCLUSIONS

SDBD plasma actuators have been shown to be very adaptable as a boundary layer separation control technology. They can readily couple with the shear layer instability by operating in an unsteady mode matching the frequency of spanwise large coherent structures and thus reattach separation for even the highest Mach number tested, 0.4. In the alternate PSVG configuration, SDBDs have been shown to produce streamwise vortical structures. In comparing to passive vortex generators, SDBDs have demonstrated the ability to produce more streamwise vorticity, and hence be more effective at reducing drag at the highest angle of attack tested and as the adverse pressure gradient increases.

The spanwise arrangement of a plasma actuator is well suited for boundary layer separation control on thin airfoil which exhibits leading edge stall. Increases in maximum lift coefficient and stall angle for the EET airfoil using the AC SDBD plasma actuator were comparable to leading edge slats seen in the literature. Steady operation is slightly preferred when acting as a leading edge flap to increase maximum lift, whereas unsteady operation is slightly better as a high angle of attack flight control to increase the maximum change in lift coefficient during stall. When unsteady operation was used, the lift to drag ratio was improved the most by operating the AC actuator at a reduced frequency of 1, and operating the NP actuator at a reduced frequency of at least 2.

For the trailing edge separation control experiment, PSVGs were able to reduce drag by an equal amount ($\Delta C_d \times 10^4 = -40$) to traditional vane-type passive VGs.
Reductions in drag for the VGs were independent of freestream Mach number and were limited to pre-stall angles of attack (4°–6°). The greatest reduction in drag for the PSVGs was observed post-stall at $\alpha = 13^\circ$ and Mach 0.15. Vorticity measurements in the boundary layer downstream of both flow control devices showed that VGs produced less vorticity with increasing angle of attack as explained by vortex compression in an adverse pressure gradient. The PSVGs, alternatively, were able to produce more streamwise vorticity with higher angle of attack. This result is not yet fully explained, but the PSVGs were better suited at reorienting spanwise vorticity for all angles of attack tested.

The PSVG flow control physics showed a good collapse of vorticity production proportional to both the residence timescale and operating voltage. These results were incorporated into the proposed linear design equation to help decide what length of plasma actuator electrode or voltage is required to produce equal vorticity to a traditional passive VG design. This equation also highlighted that PSVGs would be easier to implement on a full scale wing rather than a wind tunnel model because the required electrode length would take up a smaller percentage of chord for the same freestream velocity. The current experiment only observed PSVGs operating in the “Regime 2” mode of operation due to the linearity of vorticity with residence timescale. In addition it has been proposed that the ratio of residence to production timescale be greater than 50, that is $\frac{T_{\text{Res}}}{T_{\text{P}}} \geq 50$, to ensure the scaling laws apply for PSVGs in an adverse pressure gradient common to the suction surface of an airfoil. A future experiment could further reduce the residence timescale by reducing the freestream velocity to see if any “Regime 1” mode of operation is observed in an adverse pressure gradient.
APPENDIX A

WIND TUNNEL DRAWING
Figure A.1. Schematic of test section with force transducers.
APPENDIX B

FLOW VISUALIZATION

The presence and location of flow separation is critical for the justification of flow control, and actuator placement. A UV dye was mixed with 10,000 cSt silicone oil and then brushed onto the airfoil surface, and the wind tunnel was run for 10 minutes. The V-22 wing was instrumented with PSVGs. This way any roughness effects of the copper electrodes would be included in this visualization. Initially it was believed the V-22 had separated flow at low angle of attack due to the manufacturer placing passive vortex generators at approximately the \( \frac{c}{10} \) location. However, as seen in Figures B.1–B.2 at freestream Mach numbers of 0.1–0.3 and at an angle of attack of 8\(^\circ\), there is no flow separation until the cove of the flap. In the interest of quicker flow separation visualization, tufts of yarn were affixed along the midspan of the wing every 1 inch and flow visualization was carried for the same flow conditions as the baseline aerodynamic measurements. The yarn was observed and video recorded. In general, the effect of Mach number and flap angles had very little influence of the separation location. Angle of attack had the most influence. When \( \alpha = 9^\circ \), tuft vibration, and hence flow separation was evident just upstream of the flap cove. As angle of attack was increased stall progressed forward. And at \( \alpha = 15^\circ \), the zero-shear line was now at the trailing edge of the Macor insert.

It then seems quite advantageous to use an active flow control technology to reattach separated flow at higher angles of attack to decrease form drag while being able to lay flat and introduce no additional form drag at lower angles of attack, \( \alpha < 9^\circ \), for which no flow separation existed on the upper wing main element surface.
Figure B.1. UV based flow visualization at Mach 0.1, $\alpha = 8^\circ$, $\delta_f = 15^\circ$.

Figure B.2. UV based flow visualization at Mach 0.3, $\alpha = 8^\circ$, $\delta_f = 15^\circ$. 
BIBLIOGRAPHY


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