PLASMA SLATS AND FLAPS: AN APPLICATION OF PLASMA ACTUATORS FOR HINGELESS AERODYNAMIC CONTROL

A Dissertation

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Abstract

by

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Currently a great deal of interest within the community is to utilize the emerging flow control technology to design revolutionary air vehicles without moving control surfaces while still maintaining controlled flight. This work is focused on the application of plasma actuators to control separation on the wing in a manner that will replace the leading-edge slat and trailing-edge flap.

The experiment examined the use of plasma actuators at the leading edge to control separation, and at the trailing edge, to control lift on NACA 0015 airfoil. At the leading edge, the actuator was operated in both “steady” and “unsteady” manner. The steady actuator was able to reattach the flow for angles of attack up to 19°, which was 4° past the normal stall angle. Even better performance was found with unsteady actuation, which was able to reattach the flow up to a 9° past the normal stall angle. The leading-edge separation control resulted in a increase in both $C_{L_{\text{max}}}$ and $\alpha_{\text{stall}}$. It resulted in an $L/D$ improvement of as much as 340%. The trailing edge actuator was located on the surface of one side of the airfoil at $x/c = 0.9$, and spanned most of its width. When operated in a steady manner, it was found to produce the same effect as plane trailing-edge flap.

The plasma actuators were found to control the separated flow over the V-22 wing section and reduce the drag. At both 10 m/s and 20 m/s the flow separated relatively
evenly over the leading and trailing edges of the wing section. Each of the actuators operating separately, were able to reattach their respective flow regions. When operated together, they gave a combined effect that was approximately the sum of their individual effects. Overall with both actuators operating the drag coefficient was lowered by 44% and 27% respectively.

An experiment was conducted to control the leading-edge vortex breakdown on a 1303 UCAV planform at high angles of attack without moving surfaces using plasma actuators. Optimum lift enhancement was achieved by placing the actuators at a chord-wise location that was close to the leading edge on the suction side at $x/c \simeq 0.03$. The actuators were placed parallel to the leading edge and were operated in the unsteady mode. For these, the actuators on the inboard half of the wing was only effective for angles of attack greater than 20°. The actuator on the outboard half of the wing was, however, effective for angles of attack from 9° up to the largest angle examined, 35°, for which the conventional trailing-edge flaps were ineffective. The results suggests that the application of plasma actuators on a swept UCAV planform can alter the flowfield of the leading-edge vortex in a manner that allows control without the use of hinged control surfaces.

In order to obtain the lift enhancement controllability at low angles of attack, a wall-mounted hump model was selected in this study as a canonical turbulent separated flow field. Both spanwise and streamwise plasma actuator configurations were investigated at a chord Reynolds number of $Re_c = 288$ K. The surface pressure coefficients demonstrated that both configurations increased the pressure level in the separation region, and significantly reduced the size of the separation bubble. For the same conditions, both configurations achieved the similar performance.

Numerical simulations using Reynolds-averaged Navier-Stokes solver were per-
formed to predict the flow field over the wall-mounted hump model with both spanwise and streamwise plasma actuators on and off. For base flow, $SA$, $k-\varepsilon$, and $k-\omega$ turbulence models were used and all of them predicted the surface pressure coefficient in good agreement with the experimental data. $k-\varepsilon$ model was chosen for the controlled cases. The plasma actuator effect was simulated through a body force model. For both spanwise actuation and streamwise actuation, computations agreed with experimental data very well except around the reattachment region. Details of the flow field were also examined through streamlines, vorticity magnitude, velocity vector field, surface static pressure contour, and surface streamlines.
For my father, thank you for your love and support.
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<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
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<tbody>
<tr>
<td>$a.c.$</td>
<td>alternating current</td>
</tr>
<tr>
<td>$c$</td>
<td>chord length</td>
</tr>
<tr>
<td>$C_L$</td>
<td>lift coefficient</td>
</tr>
<tr>
<td>$C_D$</td>
<td>drag coefficient</td>
</tr>
<tr>
<td>$C_p$</td>
<td>surface pressure coefficient</td>
</tr>
<tr>
<td>$C_\mu$</td>
<td>relative momentum addition coefficient</td>
</tr>
<tr>
<td>$d.c.$</td>
<td>direct current</td>
</tr>
<tr>
<td>$e$</td>
<td>electronic charge</td>
</tr>
<tr>
<td>$E$</td>
<td>electronic field strength</td>
</tr>
<tr>
<td>$f$</td>
<td>frequency</td>
</tr>
<tr>
<td>$f_{mod}$</td>
<td>modulation frequency</td>
</tr>
<tr>
<td>$F_B$</td>
<td>body force vector</td>
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<td>$h$</td>
<td>height</td>
</tr>
<tr>
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<td>total test section height</td>
</tr>
<tr>
<td>$k$</td>
<td>Boltmann’s constant</td>
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<td>$k$</td>
<td>reduced frequency</td>
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<tr>
<td>LEV</td>
<td>leading edge vortex</td>
</tr>
<tr>
<td>$n_e$</td>
<td>electron density</td>
</tr>
<tr>
<td>$n_i$</td>
<td>ion density</td>
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$n_o$  
plasma density

$P$  
pressure

$P_{\infty}$  
freestream pressure

PIV  
particle image velocimetry

Re\textsubscript{c}  
chord Reynolds number

St  
Strouhal number

SPPA  
spanwise plasma actuator

STPA  
streamwise plasma actuator

$t$  
time

$T$  
period

$T$  
temperature

$u$  
velocity

$U_{\infty}$  
freestream velocity

$V$  
voltage

$V_{p-p}$  
peak-to-peak voltage

VBD  
vortex breakdown

VG  
vortex generator

$x$  
distance in the x-direction

$y$  
distance in the y-direction

$z$  
distance in the z-direction
Greek symbols

\( \alpha \) angle of attack
\( \alpha \) pitch angle
\( \alpha_{\text{stall}} \) stall angle of attack
\( \beta \) skew angle
\( \delta \) boundary layer thickness
\( \varepsilon_o \) permittivity of free space
\( \lambda_D \) Debye length
\( \rho \) fluid density
\( \rho_c \) charge density
\( \rho_{\infty} \) freestream fluid density
\( \phi \) electric potential
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1.1 Introduction

The maximum lift and stall characteristics of a wing affect many performance aspects of aircraft including take-off and landing distance, maximum and sustained turn rates, climb and glide rates, and flight ceiling[16]. In a 2-D wing, the maximum achievable lift is ultimately limited by the ability of the flow to follow the curvature of the airfoil. When it cannot, the flow separates. In most cases, this occurs at the leading edge.

One solution to prevent leading-edge separation is to increase the leading edge radius. This is the principle effect of a leading-edge flap. An Example is a Krueger flap, which consists of a hinged surface on the lower side of the wing leading edge. A slotted leading-edge flap (slat) is the leading-edge equivalent of the trailing-edge slotted flap. It works by allowing air from the high-pressure lower surface to flow to the upper surface to add momentum to the boundary layer and prevent flow separation.

Conventional multi-element wings and wings with movable control surfaces such as the leading-edge slats and trailing-edge flaps that contain gap regions are a major source of airframe noise and vibration, especially at high deflection angles. Most of the noise originates from the separated flow in the gap region. It is also known that the hinge gaps contribute to the form drag component of the viscous drag on the wing.
In order to improve the aerodynamic performance of the wing, it is desirable to either completely replace the traditional moving surfaces with hingeless control surfaces, or limit the deflections of moving surfaces without compromising the wing’s performance. Both these alternative necessitate other approaches for controlling flow separation over the surface of the wing.

This work is focused on the applications of plasma actuators to control flow separation on the wing in a manner that will potentially replace the leading-edge slat and trailing edge flap of a wing with an array of plasma actuators on the leading edge (plasma slat configuration) and an array of plasma actuators on the trailing edge (plasma flap configuration), respectively.

Eliminating the hinge gaps will have a direct effect on the form drag component of the viscous drag on the wing. By present estimates used in the wing and tail design, this would result in a 10% drag decrease[16]. Then hinge gap is also a source of radar wave reflection; hence by eliminating the gap, the wing radar stealth could be significant improved. In the ultimate application of the plasma actuators on wings, a “virtual section shape” would come about by completely tiling the source of a generic shape with plasma actuators (see Figure 1.1). The choice of the generic shape might come from other considerations such as weight or structural strength. The aerodynamics of the wing would be varied according to the flight plan, e.g., high lift for take-off and landing, virtual drag bucket at the design $C_L$ for efficient cruise and endurance.

1.2 Flow Separation

The classical concept of flow separation from a continuous surface is governed by two factors, adverse pressure gradient and fluid viscosity. In order to remain attached to surface, the fluid in a boundary layer must have enough momentum to overcome
Figure 1.1. Schematic of technical difficulty for application of plasma actuators.

the kinetic energy loss associated with this adverse pressure gradient and the viscous dissipation along its flow path. This loss has a more pronounced effect near the wall where the momentum is much less than in the outer part of the shear layer. If the flow retardation is such that further advancement of the fluid is no longer possible, then the surface streamline nearest to the wall leaves the bounding surface at this point and the boundary layer is said to separate (Maskell). At separation, the rotational flow region next the wall abruptly thickens, the normal velocity component increases, and the boundary-layer approximations are no longer valid.

1.2.1 Flow Separation on Airfoils and Wings

As shown in Figure 1.2 and Figure 1.3 two types of boundary layer separation are generally possible, namely, laminar separation at or near the leading edge and turbulent
separation from the trailing edge, depending on camber, ratio of thickness to chord, leading-edge radius, and Reynolds number. As indicated in Figure 1.3, separation from the leading edge of swept wings is usually manifested in the form of a vortex sheet with its axis inclined at an angle to the free-stream direction.

Laminar separation is usually accompanied by a subsequent reattachment following transition in the free shear layer to the turbulent state, forming a region of local separation or bubble. The laminar separation bubble is either, “short” or “long” depending on the magnitude of Reynolds number (based on velocity and momentum thickness of boundary layer at separation) and pressure gradient.

If the bubble is short and the angle of attack is then increased, the separation point moves forward to a region of increasing surface curvature. Eventually turbulent reattachment of the full shear fails to take place, and there is a consequent sudden loss of lift and increase drag. Separation involving formation of a short bubble can occur on most conventional airfoil sections of moderate thickness to chord ratio in the range of $0.09 < t/c < 0.15$.

If the bubble is long, then an increase in angle of attack produces a progressive rearward movement of the reattachment, thus increasing the length of the bubble until it coincides with the trailing edge. The separation is reached at about this angle of attack; any further increase results in a gradual reduction of lift, as shown in Figure 1.4(b). This separation process can also occur on most conventional thin airfoil in a range of $t/c$ up to about 0.09 and is usually called thin-airfoil stall. Aerodynamically, whereas the formation and development of a long bubble has a considerable adverse effect on drag via the pressure distribution, the existence of a short bubble has a negligible effect right up to the harmful stall condition.

The turbulent separation from the trailing-edge as illustrated in Figure 1.2 (a) is
Figure 1.2. Separation on 2-D airfoil: (a) Separation bubble and trailing-edge separation; (b) Close view of laminar separation bubble.
Figure 1.3. Separation on swept wing (adapted from Anderson[7]).
characteristic of most conventional thick airfoil sections in a range of $t/c$ greater than 0.12. An increase in angle of attack produces a gradual forward movement of the point of separation and steady and gradual decrease in lift (Figure 1.4 (c)).

It is possible for both short-bubble and trailing-edge separation to exist on the same families of airfoils at the same time over a certain range of Reynolds numbers. The former generally starts to develop at a lower angle of attack than does the latter; This combination of separation consequently displays characteristics of both short-bubble and the trailing-edge separation, with the possibility of either a semi-rounded lift curve peak followed by an abrupt decrease in lift or a relative sharp lift curve peak followed by a relatively rapid decrease in lift (Figure 1.4 (d)).

Figure 1.4. Types of airfoil stall:(a) short-bubble stall; (b) long-bubble stall; (c) trailing-edge stall; (d) combined stall (from Chang[12])
Usually, a low-frequency oscillation in the flow over airfoil is observed near stall conditions. Boreren et al. [10] performed an experiment to study the low-frequency oscillation on the LRN-1007 airfoil. The author provided surface oil-flow visualization results prior to the onset of the unsteady flow. There was a leading edge separation bubble that grew in size on the upper surface as the angle of attack was increased. The data also showed that there was significant boundary layer separation from a point downstream of the separation bubble reattachment. The role of the separation bubble and turbulent boundary layer separation was investigated more detail in another experiment conducted by Broeren and Bragg [11]. LDV measurements were performed on the LRN-1007 airfoil upper surface for $\alpha = 15$ deg and $Re = 3 \times 10^5$. The results showed the development and growth of a leading-edge separation bubble that merged with the turbulent boundary-layer separation causing a completely separated or stalled condition. The separation bubble was also found to play a key role in the oscillation, as its elimination caused the low-frequency oscillation vanish.

1.3 Separation Control

Due to the large energy losses associated with boundary-layer separation, the performance of many practical devices is often deteriorated by separation. Hence, separation flow control is of extreme importance for many technological applications of fluid mechanics. Up to this point, hundreds of methods have been explored and generally can be divided into three categories:

- Boundary Layer Mixing
- Traditional Boundary Layer Control (Blowing/Suction)
- Periodic Excitation
1.3.1 Boundary Layer Mixing

1.3.1.1 Vortex Generators

Vortex generators are passive devices used to enhance momentum transport from the free-stream to the wall region. Fluid particles with high streamwise momentum are swept along helical paths toward the surface to mix with the slower near-wall flow. These streamwise vortices bring high momentum fluid into the boundary layer to prevent flow separation. The aerodynamics applications include airfoil/wing performance improvements through increased lift and reduced drag for a low-Reynolds number airfoil, high-lift airfoil, highly swept wings, and a transonic airfoil. The non-airfoil aerodynamics applications include aircraft interior noise reduction at transonic cruise, inlet flow distortion reduction within compact ducts, and more efficient overwing fairing. These applications significantly benefit many civil transport aircraft as well as maneuverable and stealthy combat aircraft over a wide range of speeds from subsonic to supersonic.

Vane-type vortex generators were introduced first by Taylor[65] in the late 1940s. These devices consisted of a row of small plates or airfoil which were mounted on the surface and set at an angle of incident to the local flow. Figure 1.5 shows various types of vortex generators. The VG height, $h$, is typically on the order of the boundary layer thickness, $\delta$. These conventional VGs have been successfully employed for separation control by increasing the near-wall momentum through the momentum transfer from
the outer flow to the wall region. However, Taylor-type vortex generators produce a relative large device drag, which causes a reduction in aerodynamic performance.

In the past two decades, the optimum low-profile vortex generators were developed to control boundary layer separation. Their height is only a fraction of the conventional vane-type VGs, but still can generate sufficient momentum transfer over a region several times their own height for effective separation control and lower the parasitic drag at the same time. Figure 1.5 shows two relatively new types of low-profile vortex generators. The Wheeler doublet vortex generators consisted of a double row of triangular, ramp-shaped devices resembling overlapping arrowheads. The purpose of the second row was to reinforce the vortices produced by the first row. The wishbone vortex generators
consisted a single row of v-shaped ramps with their apexes pointing downstream.

Kerho et al.[34] performed an experiment to reduce the laminar separation bubble on a Liebeck LA2573A low-Reynolds number airfoil using low-profile VGs. Typically, for the intermediate Reynolds number ($10^4 \sim 10^6$), the separated flow proceeds along the direction of the tangent to the surface and transition to turbulence takes place in the free-shear layer. Subsequent turbulent entrainment of high-speed fluid causes the flow return to the surface, thus forming a laminar separation bubble. Although separation bubble has little effect on lift, it can make the boundary layer become thicker and thus the drag increases significantly. In his experiment, the designed Reynolds number, $Re_c$, is $2.35 \times 10^5$ at $\alpha$ below the stall angle of attack, which represent typical operating conditions for a low-Reynolds number airfoil. The VGs were placed directly downstream of the airfoil’s pressure peak and contained completely within the boundary layer. Two different types of low-profile VGs were tested, Wishbone VGs with $h/\delta \sim 0.3$ and ramp cone VGs with $h/\delta \sim 0.4$. The results indicate that both types of VGs provide a significant drag reduction at the design condition of $Re_c = 2.35 \times 10^5$ and $\alpha = 4^\circ$. The $h/\delta \sim 0.4$ ramped cone VGs ($\Delta z/h \sim 39.7$) yields a 35% drag reduction, and the $h/\delta \sim 0.3$ wishbone VGs ($\Delta z/h \sim 64.6$) yields 38% drag reduction. Drag reduction over a range of lift coefficients was also investigated. The results showed that both types of VGs yield a substantial reduction in drag without any adverse effect on lift.

Lin et al.[37] conducted tests during the early 1990s in the Low-Turbulence Pressure Tunnel at NASA Langley Research Center for separation control using low-profile vortex generators on a single-flap three-element airfoil at $Re_c$ of 5 and $9 \times 10^6$. At a typical landing approach condition ($\alpha \sim 8^\circ$), baseline separation on the flap occurs at approximately 45% of the flap chord. In this case, counter-rotating, trapezoidal-wing VGs ($h/c = 0.0018, e/h = 7, \beta = \pm 23^\circ$) located at 25% of the flap chord were used.
The benefit of this particular placement is that the VGs can be hidden in the flap during cruise, but exposed when needed during take-off and landing as illustrated in Figure 1.6. Lift coefficient versus angle of attack and drag polar for the VG configuration are shown in Figure 1.7. At typical $\alpha$ of $4^\circ \sim 8^\circ$, the generator-induced attached flow on the flap could increase the lift by 10%, reduce the drag by 50%, and increase the lift-to-drag ratio, $L/D$, by 100%.

1.3.1.2 Vortex Generator Jets

Boundary layer mixing is an effective way to prevent separation from the study of vortex generator. In the mixing, streamwise vortices are generated to increase longitudinal momentum near the wall and suppress separation. Another method for generating streamwise vortices is through the use of jets blown through holes in a solid surface. The holes in the surface are inclined at an angle to surface, skewed with respect to the free-stream direction, and arrayed along the surface much like classical vortex generator. Streamwise vortices are generated by the interaction between the jets and the
Vortex Generator Jet (VGJ) is an active control method which has several advantages over the conventional VGs. VGJ can adjust the strength of streamwise vortices by varying the jet speed and have the potential of displaying short response time in situations where rapid deployment is required for separation control. Furthermore, compared with conventional vortex generator, the drag penalty is negligible when the jet flow is turned off. However, the VGJ need some source of air, and the limited availability of air makes it less attractive.

A schematic of a single VGJ is shown in Figure 1.8. The key parameters include the jet-to-freestream velocity ratio, $VR = V_j/U_\infty$, the ratio of jet diameter to the local boundary layer thickness, $D/\delta$, and the orientation of the jet (pitch angle $\alpha$ and skew angle $\beta$) with respect to the free-stream. If pulsed injection is applied to the VGJ, the pulsed frequency $f$ and duty cycle are also important. Wallis[71] is the first to propose and study the VGJs as a method to delay the separation of turbulent boundary layer. Recently, experimental studies of VGJs have been presented by Johnston[33].

Figure 1.7. Effect of low-profile VGs on lift and drag of a high-lift airfoil[37].
and Compton[14] to prove that the VGJs produce streamwise vortices in a turbulent boundary layer which is similar to a weak vortex formed by a solid vortex generator. The maximum vorticity level is produced when skew angle is between 45 deg and 90 deg and jet pitch angle is 45 deg. The vortex strength increases with the jet-to-freestream velocity ratio of $0.7 \leq VR \leq 1.3$. Zhang[76] studied the vortices produced by rectangular and round nozzles and proved that a rectangular nozzle produced a stronger vortex than a circular nozzle with the same mass flow rate.

Physical features of VGJ associated with the key parameters are a crucial element for design as well. Rixon et al.[54] investigated the development of a single steady VGJ in a turbulent boundary layer when the jet was pitched 45 deg and skewed 90 deg. The velocity field was measured at four stations downstream of the jet exit, $x/D = 5, 10, 20$ and 30. The jet generated a pair of streamwise vortices, one of which was stronger and dominated the flow field. The circulation, peak vorticity, and wall-normal position of the primary vortex increased linearly with the jet velocity. The circulation and peak

Figure 1.8. Schematic of a vortex generator jet and associated coordinate system.
vorticity decreased exponentially with the distance from the jet source.

1.3.2 Conventional Boundary Layer Control (Blowing/Suction)

Suction was the first method proposed for the control of separation by Prandtl[57]. The principle is to remove the low-momentum fluid from the boundary layer and deflect the high-momentum free-stream fluid towards the surface. When the fluid near the wall is sucked away, a new boundary layer is formed downstream of the suction area. This eliminates the viscosity effect and prevents boundary growth. By providing suction upstream of either the separation line or the transition region, flow separation may be prevented and laminar flow maintained to avoid turbulent flow. This technique can be applicable to subsonic, transonic, and supersonic flows. However, to date, this method has not been applied to the wings of operational aircrafts. The major reason is the mechanical complexity and additional weight. Any aerodynamic gains made by suction are offset by the power required to operate the suction devices. Another reason is the physical surface contamination. Contamination, caused by ice, rain, dust, pollen, and insects, may block suction slots or holes which results in a consequent loss of suction.

Steady blowing tangentially along the surface is an effective technique for suppressing separation. The basic principle is to add high-momentum fluid to decelerated boundary layer and thus delay the separation. The steady blowing is particularly attractive because compressed air from the jet engine compressor may be used for blowing, and the high pressure air bleed can be bled to a choked blowing slots or holes.

Poisson-Quinton[48] investigated the effect of blowing on lift and pressure distribution for different positions as shown in Figure 1.9. For case (a), blowing at the leading edge shifts the separation line downstream and increases the maximum lift coefficient at higher angle of attack. However, this effect is small at lower angle of attack. Case
Figure 1.9. Influence of steady blowing on boundary layer; the upper series of sketches indicate lift coefficient as a function of angle of attack and lower series the pressure distributions on the upper side of the airfoil; from Poisson-Quinton[48].

(b) is blowing at the knee of a movable airfoil nose. The most effective position of blowing must be selected to around the separation line. For case (c), blowing over the flap delays the separation but a greater intensity of blowing is required. The angle of attack for maximum lift coefficient decreases due to separation at the leading edge. Case (d) is blowing from the under side of the airfoil. It creates a jet flap which increases circulation.

The standard measure of relative momentum addition is the total momentum, which was first introduced by Poisson-Quinton and given the symbol $c_\mu$. This fundamental parameter is defined by the equation:

$$c_\mu = \frac{\rho j U_j^2 H}{0.5 \rho_\infty U_\infty^2 L}$$
Figure 1.10. Lift coefficient vs. momentum coefficient (left) and critical value of momentum coefficient for boundary layer control[8].

where the subscript \( j \) refers to a 2-D jet, \( H \) is the slot height, and \( L \) is a reference length.

In some circumstances, when the momentum coefficient is less than 2%, blowing can be detrimental. Thus many investigators care more about \( c_\mu > 3\% \). Attinello[8] shows that blowing over the flap results in phenomena described as both boundary layer control and supercirculation. As illustrated in Figure 1.10, a relatively rapid increase in lift coefficients is achieved when the momentum coefficient is less than 5%. Beyond this critical value, the boundary layer control is superseded by circulation control which is less effective. The increment in lift coefficient realized by each increment in the momentum coefficient is reduced significantly.
1.3.3 Periodic Excitation

1.3.3.1 Acoustic Excitation

Flow control using periodic excitation starts from studies of the influence of sound on boundary layer transition performed by Schubauer and Skramstad[58], Knapp and Roache[35] and Spangler et al.[64]. According to their conclusions, acoustic wave can trigger premature transition to turbulence which is less susceptible to separation. Collins and Zelenevitz[13] realized that this technique could apply on boundary layer control and demonstrated that external sound can cause partial reattachment of the flow on a stalled airfoil, greatly increasing the lift and decreasing the drag. Following this experiment, acoustic excitation was demonstrated on a variety of airfoils at low Reynolds Numbers.

Zaman et al.[74] experimentally investigated the effect of acoustic excitation on flow over a two-dimensional airfoil in different ranges of angle of attack at low Reynolds

Figure 1.11. Smoke-wire flow visualization pictures for \( \alpha = 15^\circ \) and \( 18^\circ \) with and without excitation; \( Re_c = 4 \times 10^4 \)[74].
Figure 1.12. Acoustic control of leading-edge separation at $\alpha = 14^\circ$ [43].

number ($4 \times 10^4 \sim 1.4 \times 10^5$). For $\alpha \leq 8^\circ$, Small-amplitude excitation in a wide, low-frequency range was found to eliminate the laminar separation effectively. The data were supported by flow visualization (see Figure 1.11). For $\alpha \geq 18^\circ$, during post-stall, significant increase in lift was achieved by large-amplitude excitation. From stability analysis, the authors inferred that the excitation mechanism must lie in the instability of the separated shear layer.

Nishioka et al. [43] reported that leading-edge separation control on a flat-plate airfoil by means of acoustic excitation were studied experimentally and theoretically on the basis of the linear stability theory, at a chord Reynolds number $Re_c = 4 \times 10^4$. The results showed that the separated shear layer is extremely sensitive to the small-amplitude perturbation and rolls up to form discrete vortices (see Figure 1.12). The rolled-up vortices with matched scales and frequencies work well to enhance the entrainment and maintain the necessary influx of momentum toward the wall region to suppress the separated shear layer.

Ahuja [2, 3], Marchmann et al. [38] and Hsiao [28, 29] also conducted experiments on separation control by acoustic excitation on different airfoil at low Reynolds num-
bers and observed the similar effects. The optimum Strouhal number varies widely, from $O(1)$ to $O(100)$. In order to get some significant effects, very high levels of excitation are required, which limits the practical application of the method. However, acoustic excitation illustrates the receptivity of the separated shear layer to the periodic excitation, which is the principle for separation control.

1.3.3.2 Hydrodynamic Excitation

Hydrodynamic oscillation has been investigated extensively in the last decade. The source of oscillation usually is located on the surface before the separation line. This perturbation can be amplified and modulate the formation of large scale, phase-locked coherent vortex structure, which are considered to be the most unstable modes in shear layer. The existence of these large vortex structures enhance the mixing between the outer and inner regions, thus in turn enable the flow to withstand the adverse pressure gradient without separation. Hydrodynamic oscillation can be achieved by oscillatory blowing system[59–61] or piezoelectric actuator (synthetic jets[63]) or pulsed vortex generator jets[39, 40], etc.

Seifert et al.[59] investigated the effects of oscillatory blowing on flapped NACA 0015 airfoil. The flap extended over 25% of the chord and was deflected at angles as high as 40 deg. The oscillatory blowing was emanated from a two-dimensional slot over the hinge of the flap. The total jet momentum was separated into two components: a steady component based on the steady blowing only and an oscillatory component based on the maximum amplitude of the velocity oscillations. The steady blowing momentum could be varied independently of the amplitudes and frequencies of the superimposed oscillations. The efficiency of the airfoil could be increased significantly by low momentum oscillations superimposed on a small amount of steady blowing. This
efficiency was measured by the enhancement of lift and reduction in drag at all angles of attack. There are many parameters governing the flow and they all have to be optimized to keep the flow attached at minimum input of momentum. The Strouhal number ($St$) is probably the most important one. The dependence of lift ($C_L$) on Strouhal number ($St$) was tested at fixed flap deflection, Reynolds number and blowing momentum coefficients. Optimum performance was obtained at Strouhal number, based on the flap chord, of the order of unity.

Another experiment was carried out by Seifert et al.[60] on different airfoils to investigate the parameters such as the location of the blowing slot, the shape and incidence of the particular airfoil. It was observed that the location of oscillatory blowing was much more effective within the vicinity of separation than far away from it. This technique was applied on different airfoils and the similar improvement was obtained. The optimum reduced frequency was reaffirmed, however, the sensitivity of the flow to the reduced frequency was not large and it diminished with increasing angle of attack. Phase-locked pressure measurements at optimum reduced frequencies revealed that at least two eddies swept over the surface at any instant time and their size increased when moving downstream.

Application of oscillatory blowing at high Reynolds numbers was also investigated by Seifert et al.[61]. Stall was delayed and poststall characteristics were improved when oscillatory blowing was located at the leading-edge region, however, the flap efficiency was improved when control was applied at the flap shoulder. One to three vortices over the upper surface of the airfoil at optimum frequencies was observed, which was consistent with the low-Reynolds-number experiments.

Synthetic jets[24, 63] use the similar principle to control separation. A synthetic jet in the absence of a cross flow is produced by the interactions of a train of vortices that
are typically formed by alternating momentary ejection and suction of fluid across an orifice such that the net mass flux is zero. During the suction stroke may be thought of as similar to the flow induced by a sink that is coincident with the jet orifice, the flow during the ejection stroke is primarily confined to a finite narrow domain in the vicinity of jet centerline. During the momentary ejection, the flow separates at the sharp edges of the orifice and forms a vortex sheet that typically rolls into a vortex (see Figure 1.13). The utility of synthetic jets for separation control was demonstrated by Amitay et al.[4–6].

Amitay et al.[4] reported a investigation of the manipulation of the global aerodynamic forces on a thick airfoil using surface-mounted synthetic jet actuators. The effects of the actuation was tested at two ranges of jet actuation frequencies $St \sim O(1)$ and $O(10)$. Flow reattachment over the stalled airfoil began with the advection of a strong clockwise vortex past the measurement station indicating a reduction in lift that is followed closely by a stronger counter-clockwise vortex indicating the reestablish-
ment of lift.

The application of synthetic jet actuators on an uninhabited air vehicle with 50-deg leading-edge sweep was investigated by Amitay[5]. He found that excitation with a sinusoidal waveform resulted in a sharp suction peak near the leading edge, while pulse modulation generated a larger and wider suction peak. As the modulation frequency decreased, the size of the suction peak increased where the highest magnitude was achieved when modulation frequency reached the instability frequency of the separated flow over the wing.

Pulsed vortex generator jets have been studied by McManus et al.[39, 40] to enhance the efficiency of VGJs in delaying the boundary layer separation. It has been found that pulsing can significantly improve the performance of VGJs for separation control. With the same mass flow rate, a pulsed jets delayed flow separation on an airfoil to a higher angle of attack as compared to steady VGJs. Moreover, pressure recovery on significantly larger areas are affected by the pulsed jets.

Bons et al.[9] reported the application of pulsed vortex generator jets on suction surface of a low-pressure turbine blade for separation control. Blade static pressure distributions show that separation is almost completely eliminated by the pulsed jets. Experimental evidence suggests that the mechanism for pulsed jets may lie in the starting and ending transitions of the pulsing cycle rather than the injected jet itself.

1.3.3.3 Mechanical Excitation

The flow control mechanism is to generate the periodic vortices in a separating flow at or near an optimal frequency. This periodic vortices initiate the instability resides in the separated layer and become stronger. The newly formed vortices exhibit high momentum and energize the separated boundary layer thus delaying or eliminating the
separation. The unsteady excitation can be produced by injecting air in an oscillatory way as discussed in previous part. It can also produced by moving a mechanical element.

Osborn et al.[46] presented a device - high frequency micro-vortex generator for separation control. Figure 1.14 shows the concept of second-generation HiMVG device. The device consists of a VG blade and a mechanical system to drive the oscillation. The VG blade is flush with the aerodynamic surface when retraced. It can extend into the boundary layer and oscillate at a certain frequency when flow separate.

1.3.3.4 Electrohydrodynamic and Magnetohydrodynamic Excitation

Plasma actuators 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. A schematic illustration is shown in Figure 1.15.
The plasma actuators are low power devices, with typical power levels of approximately $2 \sim 40$ Watts per linear foot of actuator span. They can be operated either in a “steady” or unsteady manner. In the “steady” operation, the input driving a.c. frequency is well above the fluid response frequency and therefore the flow senses a constant body force. In unsteady operation, the higher driving a.c. frequency is switched on and off at low frequencies down to a fraction of a Hertz. The latter can be used to excite fluid instabilities that act to further amplify the actuator effect. In the unsteady operation, very short duty cycles are possible, which reduce the actuator power significantly. For example, in application of separation control in which unsteady forcing is the most effective, a 10% duty cycle is sufficient, which lowers the power requirement by 90% compared to steady operation[19, 51].

The important property of the single dielectric barrier discharge is that it is self-limiting and therefore stable at atmospheric pressure (see Figure 1.16). During the a.c. cycle, the electrons and ions move according to the electric potential, in an attempt to cancel it. In one-half of the cycle, electrons move from the exposed electrode to the surface of the dielectric. The buildup of charge on the dielectric will eventually balance the a.c. potential so the plasma generation will stop. This is the self-limiting aspect of
Figure 1.16. The self-limiting behavior of dielectric barrier discharge[21, 22]: (a) Electrons emitted from the exposed electrode buildup on the dielectric surface. (b) Electrons go back to the exposed electrode. The electrons available to the discharge are limited to the deposited on the dielectric surface in the previous half-cycle.
the dielectric barrier that prevents a cascade of charges that would cause an electric arc. In the other half of the a.c. cycle, the plasma reforms and electrons that were deposited on the dielectric travel back to the exposed electrode. The electrons do not leave the dielectric as readily as they do the exposed electrode, and the volume of the plasma is reduced. Because the body force on the external flow is proportional to the volume of the plasma, its space-time characteristic is essential for quantitative prediction of the actuator effect[21, 22].

Plasma actuators for leading-edge separation control of the dynamic stall vortex on an oscillating NACA 0015 airfoil were investigated for the application of control of retreating blade stall on a helicopter rotor[51]. The effectiveness was based on a combination of flow visualization and pressure measurements on the surface of the airfoil (see Figure 1.17). The angle of attack is given by $\alpha(t) = 15^\circ + 10^\circ \sin \omega t$ with
a reduced frequency of \( k = 0.08 \). 20Hz excitation are used in this case. The rational of the 20Hz excitation is that this frequency would excite approximately two vortices for each half cycle of the pitching motion. Having two vortices within the separation zone is the original basis for \( St = 1 \) that was successful with the stationary airfoil. At \( \alpha = 17^\circ \) during pitch-up, two vortical structures resulting from the actuator are clearly evident. At \( \alpha = 23^\circ \) during pitch-up, 2 ~ 3 structures are evident. Three structures are visible during pitch-up at \( \alpha = 7^\circ \). The \( St = 0.25 \) excitation results in a significant improvement of the lift cycle. In particular, there was a higher lift over the entire pitch-down portion of the cycle, most noticeably in the range from \( 25^\circ \geq \alpha \geq 13^\circ \); a higher lift at the bottom of the pitch-down phase, that persisted for the first half of the pitch-up phase; and a more gradual dynamic stall.

Huang et al.[31, 32] investigated the separation control using plasma actuators over turbine blades in the low-pressure turbine (LPT) stage of gas turbine engines at low blade-chord Reynolds numbers typical of high altitude cruise. Pratt & Whitney “Pak B” shaped blades were used. The location of actuators was at \( x/c = 0.67 \), which was just upstream of the flow separation line. Mean velocity profiles over the blade and surface pressure measurements were used to define the region of flow separation and re-attachment. Figure 1.18 shows results at an intermediate chord Reynolds number of 50K. The left part of the figure shows mean velocity profiles with the actuator off (open symbols) and on (closed symbols) in “steady” operation. The mean velocity profiles with actuator off indicate a large separation bubble that extends from \( x/c = 0.7 \) to 0.95. The pressure distributions in the right part of Figure 1.18 are used to further define the separation and re-attachment locations and effects of the plasma actuators. The effect of the “steady” plasma actuator is shown as the triangle symbols. This shows a smaller plateau in the \( C_p \) distribution indicating a small separation bubble remaining,
with re-attachment at $x/c \simeq 0.85$. The effect of the unsteady actuator is shown as square symbols. It shows clearly that the plateau in the $C_p$ distribution is completely removed. This indicates that the flow remained attached.

1.4 Objectives

Due to the importance of the separation control, hundreds of devices and methods have been studied for decades. The approach to separation control in this work is plasma actuator, and the objectives are:

1. To investigate the application of plasma actuators to control flow separation on the wing in a manner that will potentially replace the leading-edge slat and trailing-edge flap.

2. To examine the use of plasma actuators in a variety of applications ranging from
two-dimensional flows to three-dimensional flows, and from laminar separation flows to turbulent separation flows.

3. To examine a means of separation flow detection used in feedback control.

4. To compare simulations with experiments for a turbulent separation control over a NASA hump model.
CHAPTER 2

LEADING-EDGE SEPARATION CONTROL OVER NACA 0015 AIRFOIL

2.1 Experimental Setup

2.1.1 Wind Tunnel

The experiments were conducted in one of the subsonic wind tunnels in the Center for Flow Physics and Control (FlowPAC) in the Hessert Laboratory at the University of Notre Dame. The facility is a open-return draw-down wind tunnel with a 0.421 m (1.39 ft square) by 1.8 m (6 ft) long test section. The tunnel consists of a removable inlet having a series of 12 screens followed by an 24 : 1 contraction that attaches to the test section. The test section was equipped with a clear Plexiglas side-wall that allows optical access to view the model. The back wall of the test section had removable panels to allow access into the test section.

The airfoil used in the study was mounted on a support sting on a lift-drag force balance. The force balance was mounted on the top of the test section. A photograph of the force balance with the airfoil is in Figure 2.1. The airfoil was suspended between end-plates that were attached to the ceiling and floor of the test section. The end-plates were designed to produce a two-dimensional flow around the airfoil. A hole in the ceiling end plate accommodated the sting supporting the airfoil. A hole in the floor end plate allowed access for the actuator wiring. This hole was aligned with the support sting so that it would not interfere with angular positioning of the airfoil.
when setting different angle of attack. A stepper motor on the force balance drove the angular position of the support sting. Its motion was controlled by the data acquisition computer through software. With this, the angular position was repeatable to within 0.005 degrees.

The force balance used a strain-gage bridge to provide voltage outputs proportional to the respective lift and drag forces. Calibration of the force balance was done by applying known weights to a cable pulley system attached to the support sting. An example of the force calibration is shown in Figure 2.2. The curve through the calibration points is the result of a straight line fit that was used to convert the force balance voltages to lift and drag values.
2.1.2 Airfoil

The airfoil used in this study was a NACA 0015. This generic shape was chosen because its steady characteristics are well-known and documented in the literature. This airfoil was also the subject of an experiment on the control of dynamics stall using plasma actuators[50–52], which provided flow visualization records.

The airfoil had a 12.7 cm (5 in) chord and a 30.48 cm (12 in) span. The size of the airfoil was a balance between minimizing blockage effects, especially at the large angles of attack that were investigated, and maintaining a large enough chord Reynolds number. At the largest angle of attack 23°, the blockage was 8.5%, which still ultimately required correction for the blockage in the measured lift and drag coefficients.
The airfoil was cast using an epoxy-based polymer in a two-piece mold. The mold was precisely machined using a numerical-controlled milling machine. Photographs of the airfoil are shown in Figure 2.3.

Two free-stream speeds of 21 and 30 m/s were used in the experiments. The primary criteria for their selection was that the chord Reynolds number be above that value where stall hysteresis is known to occur. At these velocities the chord Reynolds numbers are 180 K and 257 K, which both satisfy our criterion. As a result of blockage, the corrected average Reynolds numbers for the two free-stream speeds was 217 K and 307 K.
2.1.3 Plasma Actuators

The plasma actuator consisted of two copper electrodes separated by two 4-mil thick Kapton film layers. The electrodes were made from 0.0254 mm thick copper foil tape. The electrodes were arranged in the asymmetric arrangement illustrated in Figure 2.4. They were overlapped by a small amount (of the order of 1 mm), in order to ensure a uniform plasma in the full spanwise direction. A high voltage a.c. potential is supplied to the electrodes. When the a.c. amplitude is large enough, the air ionizes in the region of the largest electric potential. This generally begins at the edge of the electrode that is exposed to the air, and spreads out over the area projected by the covered electrode. A photograph of the plasma for the electrode arrangement in Figure 2.4, is shown in Figure 2.5.

The process of ionizing the air in this configuration is classically known as a single dielectric barrier discharge. The ionized air in the presence of an electric field gradient produces a body force on the ambient air, inducing a virtual aerodynamic shape that causes a change in the pressure distribution over the surface on which the actuator is placed. The air near the electrodes is weakly ionized, and there is little or no heating of the air.
The body force per volume of plasma is a vector, given as
\[
\vec{F}_B = \left( -\frac{\varepsilon_0}{\lambda_D^2} \phi \right) \vec{E}
\]
where \(\varepsilon_0\) is the permittivity of free space \((8.854 \times 10^{-12} \text{ F/m})\), \(\lambda_D\) is the Debye length, \(\phi\) is the electric potential, and \(\vec{E}\) is the electric field vector, where
\[
\vec{E} = -\vec{\nabla}\Phi
\]

The Debye length is the characteristic length for electrostatic shielding in a plasma. It varies with plasma density and temperature as
\[
\frac{1}{\lambda_D^2} = \frac{e^2 n_o}{\varepsilon_0} \left( \frac{1}{kT_i} + \frac{1}{kT_e} \right)
\]
where \(kT_i\) and \(kT_e\) are the ion and electron temperatures, \(n_o\) is the plasma density, and \(e\) is the elementary charge. This body force vector can be tailored through the design.

Figure 2.5. Photograph of plasma (blue) for asymmetric electrode arrangement shown in Figure 2.4.
of the electrode arrangement and dielectric material, which control the spatial electric field.

The plasma actuator was bonded directly to the surface of the airfoil. At the leading edge, where the flow is sensitive to the nose-radius, a 4-mil. recess was molded into the model to accept the actuator and produce a smooth, flush surface with the NACA 0015 airfoil shape. The lower photograph in Figure 2.3 shows the dark colored Kapton sheet that fills this recess at the leading edge. The two copper foil electrodes were aligned in the spanwise direction. These were positioned so that the junction between the exposed and covered electrodes was precisely at the leading edge \((x/c = 0)\). The asymmetric arrangement was oriented so that at positive angles of attack, the covered electrode was on the suction side of the airfoil, and the exposed electrode was on the pressure side. With this orientation, the actuator would induce a velocity component in the mean free-stream direction over the suction surface of the airfoil. The actuator at the leading edge spanned the width of the airfoil corresponding to the Kapton (dark) covering seen in the photograph in Figure 2.3.

The trailing edge actuator was located on the suction side of the airfoil for positive angles of attack, at \(x/c = 0.90\). This can be seen near the trailing edge of the airfoil in the bottom photograph in Figure 2.3. The design of the actuator is similar to that at the leading edge. It is oriented so that the exposed electrode is upstream of the covered electrode in order to induce a flow in the mean downstream direction. The junction between the exposed and covered electrodes is located at \(x/c = 0.90\).

The two actuators were controlled individually. The operating frequency of the a.c. voltage supplied to the electrodes was typically 3 to 5 kHz. The actual frequency was tuned to minimize the overall power in the actuator electronics. The a.c. voltage amplitude to the electrodes ranged from 7 to 11 kV\(_{p-p}\). The power used by the actuator...
varied depending on if it were steady or unsteady operation. For steady operation, it was estimated to be 20 Watts per foot span.

The unsteady operation, is illustrated in Figure 2.6. It consisted of cycling the a.c. voltage off and on with an unsteady period, $T$. The percentage of time (duty) within the period that the a.c. was on was controllable. For the unsteady actuator cases, a duty cycle of only 10% was found to be sufficient for leading edge separation control. Thus in these cases, the power was only one-tenth that for steady actuation, or approximately 2 Watts per foot span.

2.2 Results

2.2.1 Leading-Edge Plasma Actuators (Plasma Slat Configuration)

Leading edge separation control effectiveness was evaluated on the basis of lift enhancement and drag reduction. The results presented here document the ability of the plasma actuators to reattach the flow over a stationary airfoil at high angles of attack, beyond natural static stall.

The lift and drag characteristics of the airfoil for the two free-stream speeds with the plasma actuator off, are presented in Figure 2.7. This shows the lift coefficient
Figure 2.7. Lift coefficient versus angle of attack and drag polar for the airfoil at both free-stream speeds with the plasma actuator off. Note airfoil support rod drag not subtracted from total drag.
versus angle of attack in the top plot, and the drag polar in the bottom plot. The lift and drag coefficients have been corrected by accounting for the solid blockage and wake blockage effects. The linear theory slope, $dC_L/d\alpha = 0.11$, is shown for reference in the plot of $C_L$ versus $\alpha$. The good agreement is an indication that the results are consistent with thin airfoil theory.

For the drag polar, the drag of the wing support rod was not subtracted from the total drag. Therefore the drag coefficients are larger than one would expect for this airfoil. However, the shape of the drag bucket in the linear lift region agrees well with archival results for this airfoil[1].

The characteristics of the airfoil are virtually identical at the two free-stream speeds. The flow separates at an angle of attack of 14°. This is observed as a sharp decrease in lift and increase in drag.

Flow visualization was used to verify that the drop in lift at the large angles of attack
were due to a separation of the flow at the leading edge. A sample of this taken from Post[52], is shown in the top photograph in Figure 2.8. For these, smoke streaklines were introduced in the contraction upstream of the test section at the spanwise center-line of the airfoil. The top photograph shows the flow with the airfoil at $\alpha = 16^\circ$. This shows a large separation region that covers the total upper surface of the airfoil.

The bottom photograph in Figure 2.8 shows the visualized flow when the leading-edge plasma actuator was on in “steady” operation. With the actuator on, the flow in observed to be attached all along the upper surface. The actuator conditions leading to this state at angles of attack above the natural stall angle are of particular interest.

With steady operation of the actuator, the only input parameter is the actuator amplitude needed to reattach the flow. We therefore examined this for different angles of attack. An example for $\alpha = 16^\circ$ at the lower free-stream speed is shown in Figure 2.9. This documents the lift and drag coefficients normalized by their maximum values as a function of the actuator voltage that is normalized by a reference minimum value.

At $\alpha = 16^\circ$, the flow over the airfoil was naturally separated, and the lift coefficient was low and the drag coefficient was high. When the actuator voltages were too low, the flow remained separated and there was no improvement in the lift and drag. This corresponds to the left edge of the plot. However once a threshold voltage to the actuator was reached, the flow dramatically reattached. This is observed as a large increase in the lift and decrease in the drag. The voltage condition where this occurred is marked by the dashed line in the plot. Above this voltage, there was little change in the lift and drag coefficients.

Similar experiments were done for different angles of attack and the two free-stream speeds. These were used to determine the voltage required to reattach the flow. The results are compiled in Figure 2.10. Focusing first on the results for the higher velocity,
the voltage required to reattach the flow essentially independent of the angle of attack. This was also the case for the higher angles, $\geq 17^\circ$, at the lower free-stream speed. Therefore based on these results, a single “steady” actuator voltage was selected that would assure that the flow would reattach at least up to $\alpha = 19^\circ$. The selected voltage is indicated by the dashed line.

The result of the leading-edge actuators on the lift versus angle of attack and drag polar for the two free-stream speeds is presented in Figure 2.11 and Figure 2.12. The data for the “steady” operation are shown as the circle symbols. At both free-stream speeds, the steady actuation is observed to maintain lift up to $\alpha \simeq 18^\circ$. In addition, the eventual drop in lift that occurs above this angle is more gradual than without the
leading-edge flow control.

Also included in Figure 2.11 and Figure 2.12 are the results for the “unsteady” actuator operation. This is shown as the triangle symbols. As described earlier, the unsteady operation involves turning the actuator a.c. input on and off at a lower frequency (see illustration in Figure 2.6).

It has been shown in the literature that the introduction of unsteady disturbance near the separation location can cause the generation of large coherent vortical structures that could prevent or delay the onset of the separation. These structures are thought to intermittently bring high momentum fluid to the surface, enabling the flow to withstand the adverse pressure gradient without separating. Periodic excitation by oscillatory
Figure 2.11. Lift coefficient versus angle of attack and drag polar for the airfoil at 21 m/s with the plasma actuator off (squares), and “steady” (circles) and “unsteady” (triangles) operation.
Figure 2.12. Lift coefficient versus angle of attack and drag polar for the airfoil at 30 m/s with the plasma actuator off (squares), and “steady” (circles) and “unsteady” (triangles) operation.
blowing for use in separation control has been documented extensively by Seifert et al.[59–62] and in the review by Greenblatt and Wygnanski[26].

The forcing frequency for the unsteady disturbances is believed to be optimum when the Strouhal number, \( St = f c / U_\infty \), is near unity. Here \( f \) is the actuator forcing frequency, \( c \) is separation length which in the case of the full leading-edge separation is the airfoil chord length, and \( U_\infty \) is the free-stream velocity. Table 2.1 lists the physical frequencies based on \( St = 1 \) for our conditions.

\[
\begin{array}{|c|c|}
\hline
U_\infty (\text{m/s}) & f (\text{Hz}) \\
\hline
21 & 166 \\
30 & 237 \\
\hline
\end{array}
\]

A sensitivity study was performed to determine if an optimum frequency existed for the unsteady actuator in our case. This involved the airfoil at \( \alpha = 16^\circ \) and \( U_\infty = 21 \text{ m/s} \). The results are presented in Figure 2.13. These show that a clear minimum in the voltage required to reattach the flow exists at an unsteady frequency that is close to that for which \( St = 1 \). This was then applied for the “unsteady” operation that gave the results in \( C_L \) and \( C_D \) in Figure 2.11 and Figure 2.12.

At both free-stream speeds, the unsteady actuator at \( St = 1 \) gave significantly better results than the steady operation. At the lower free-stream speed (Figure 2.11) it sig-
Figure 2.13. Lift coefficient versus angle of attack and drag polar for the airfoil at 21 m/s with the plasma actuator off (squares), and “steady” (circles) and “unsteady” (triangles) operation.

Significantly increase $C_{L_{max}}$ and $\alpha_{stall}$, and maintained lift to $\alpha = 22^\circ$, which was $7^\circ$ past the natural stall angle of attack. Similar improvements were also found for the higher free-stream speed. These results were obtained while using a 10% duty of the unsteady cycle. As a result the power to the actuator was only 10% that of the “steady” operation. The power in these cases was then only approximately 2 Watts.

We can further compare between the steady and unsteady actuation by examining the improvement they provide in the lift-to-drag ratios at high angles of attack. This is presented for the two free-stream speeds in Figure 2.14.

A substantial improvement in the lift-to-drag ratio was obtained with the leading-edge actuator at both free-stream speeds. At its maximum, this ranged from approxi-
Figure 2.14. Lift coefficient versus angle of attack and drag polar for the airfoil at 21 m/s with the plasma actuator off (squares), and “steady” (circles) and “unsteady” (triangles) operation.
mately a 2.8 times improvement at the lower speed, to approximately a 3.4 times improvement at the higher free-stream speed.

Overall, the unsteady actuator produced a greater improvement that extended over a larger range of angles of attack. When this is factored with 90% lower power required by the unsteady actuator, the system gains are substantially better.

2.2.2 Trailing-Edge Plasma Actuators (Plasma Flap Configuration)

This section presents results of operating a plasma actuator that was placed near the trailing edge of the airfoil at $x/c = 0.9$. The intention was to examine its ability to control the lift at angles of attack that were in the linear $C_L$ versus $\alpha$ region. This follows a similar study by Corke et al.[18] that examined the effect of from one to four plasma actuators located between $x/c = 0.72$ to 0.9 on a NACA 0009 airfoil.

As was shown in the low photograph in Figure 2.3, the plasma actuator spanned most of the suction side of the airfoil for positive angles of attack. The leading-edge actuator was not operated when investigating the trailing-edge actuator.

The lift coefficient versus angle of attack, and drag polar, for the cases with the trailing-edge actuator are presented in Figure 2.15. These correspond to the lower free-stream speed of 21 m/s. The chord Reynolds number in this case corresponds to one of those used by Corke et al.[18].

The solid curve corresponds to when the actuator was off. This represents the base condition for the airfoil. Again for reference, the $2\pi$ slope from linear airfoil theory is shown.

The results for two actuator amplitudes are shown. The lowest amplitude corresponding to 5 kV, is shown as the dashed line. The higher amplitude corresponding to 7 kV, is shown as the symbols.
Figure 2.15. Lift coefficient versus angle of attack and drag polar for the airfoil at 21 m/s with the trailing-edge plasma actuator off(solid curve), and on in “steady” operation at two different voltage levels.
The effect of the plasma actuator was to uniformly increase the lift coefficient at a given angle of attack. The amount of the lift increase varied with the actuator voltage. For the 7 kV input, the change, $\Delta C_L$, was approximately 0.051. This corresponded to an added lift force of 0.549 N (0.12 lbs). For reference, this was nearly identical to that found by Corke et al.[18] in experiments using a different facility, at the U.S. Air Force Academy.

The drag polars at the bottom of Figure 2.15 indicate that the effect of the trailing-edge actuator is to shift the drag bucket to the right, that is in the direction of higher $C_L$. The amount of shift depends on the actuator voltage. There was also a slight decrease in the minimum drag coefficient. This is in contrast to Corke et al.[18] in which the single actuator at $x/c = 0.72$ resulted in an increase in minimum drag coefficient.

The increase in lift and shift of the drag bucket towards higher $C_L$ that was produced by the actuator is identical to the behavior of a plane trailing edge flap. Thus we have termed this a “plasma flap”. The obvious advantages of this approach are that for simplicity there are no moving parts, and there is no hinge gap that adds drag and reduces stealth.

2.3 Close-loop Control of Flow Separation

Owing to the rapid growth in instrumentation, materials, and control technologies, the roles and capabilities of flow control and aerostructures are evolving. The use of smart aerostructures that can react rapidly to changing flow conditions to improve the aerodynamic and structural efficiencies of aircraft is gaining momentum. It is envisioned that future air vehicle designs will involve surfaces that shelter an integrated system of sensors, flow control actuators, and feedback controllers that are able to adapt to unpredictable conditions (structural damage, wind shear, stall/spin, etc.) and recon-
figure themselves in flight to regain/enhance control. This is the theme of the present
work - the design of a smart aerostructure (slat) that can be used as an intelligent high-
lift device with no moving parts.

This work presents the concept and experimental evaluation of a smart plasma slat; a low-drag, hingeless, high-lift device which uses a sensor, an aerodynamic plasma actuator, and a feedback controller for autonomous sense and control of leading-edge flow separation and wing stall. This work follows on the effects of an “open-loop” plasma slat on the aerodynamic performance of a NACA 0015 which were discussed previously. The present work is focused on formulating self-governing method to enable “close-loop” operation of a plasma slat. Much of this work is focused on reducing the power levels of the plasma actuator for practical air vehicle applications, hence due consideration was given to the design of feedback control approaches that enable a continuous self-governing plasma actuator using a simple commercial off-the-shelf pressure sensor.

Wind tunnel experiments were conducted on a slowly pitching 2-D NACA 0015 airfoil to validate the feedback control technique designed for autonomous control. The designs are generic enough to be applied to any flow-control application where smart sensing and control of incipient flow separation is desired.

2.3.1 Experimental Setup

The diagram in Figure 2.16 represents the experimental setup with closed-loop control. A pressure sensor is added based on the open-loop control, which forms the feedback control loop. Figure 2.17 shows the photograph and schematic of the airfoil model and the pressure sensor used in the experiments. As shown in Figure 2.17 (a), a slot was machined into the pressure side of the airfoil, which was used to accommodate
Figure 2.16. Diagram of experimental system. The block circled by dash line is newly added to the experimental setup based on open-loop control.

the sensor. The slot was sealed by clear tape. The static pressure port was located at $x/c = 5\%$ on suction side.

2.3.2 Results

In order to find the rule for feedback control, the characteristic of static pressure at $x/c = 5\%$ was investigated at each angle of attack when the plasma was on and off. Figure 2.18 shows the FFT analysis of discrete sampled static pressure data at some critical angle of attack. At lower angle of attack ($\alpha \leq 12^\circ$), there is no difference between the spectrum with actuator off and on. When the angle of attack reaches $13^\circ$, which is one degree before the flow separates at the leading edge, a dominant frequency and its harmonic appear in the spectrum when plasma actuator is on. This frequency corresponds to the unsteady forcing frequency of plasma actuator, which is $166$ Hz in this case. At $14.5^\circ$ angle of attack, which is immediately after $\alpha_{stall}$, a low frequency dominates the flow when actuator is off. This low frequency was investigated by Broeren and
Figure 2.17. (a) photograph of NACA 0015 airfoil with a pressure sensor; (b) close view of the pressure sensor; (c) Schematic of NACA 0015 airfoil with a pressure sensor and location of static pressure port.

[Diagram of NACA 0015 airfoil with pressure sensor]

Bragg[10]. The results showed that the development and growth of leading-edge separation bubble that merges with the trailing-edge turbulent boundary layer separation plays a key role in the low-frequency oscillation. When the plasma actuator is turned on, this low frequency vanishes and a spectral peak appears again at 166 Hz. A similar spectral peak in the measured pressure spectrum is observed all the way up to at least $\alpha = 23^\circ$, which was the highest angle of attack investigated.

At a low angle of attack ($\alpha < 13^\circ$), just before the leading-edge stall, a small separation region (bubble) begins to form downstream of the sensing port at $x/c = 0.05$. It is speculated that this location is just past the maximum thickness point of the airfoil ($x/c = 0.3$), as illustrated in Figure 2.19. The plasma actuator design and fre-
Figure 2.18: Power Spectrum of discrete sampled static pressure at angle of attack $8^\circ$, $13^\circ$, $14.5^\circ$, and $22^\circ$ when plasma actuator is turn on and off.
Figure 2.19. (a) Schematic of separation bubble position at $\alpha < 13^\circ$ and vortex generated by the plasma actuator; (b) Schematic of interaction between separation bubble and vortex generated by plasma actuator.
quency operation were meant to force spanwise vortices that efficiently mix outer high-momentum fluid with the low-momentum fluid near the surface, causing the flow to reattach. At lower angles of attack, the flow at the leading edge is attached. When the plasma actuator is on, the strong favorable pressure gradient around the leading-edge nose damps the unsteady input from the actuator so that even close to the leading edge, just downstream of the actuator \( (x/c = 0.05) \), the pressure fluctuations due to the actuator are not sensed.

As \( \alpha \) increases, the small bubble gradually moves forward to the pressure sensor location in the case of the present experiment, at \( \alpha = 13^\circ \). The small separation bubble is very receptive to the unsteady condition produced by the plasma actuator. As a result, the pressure sensor now shows a spectral peak at the actuator unsteady frequency. Note that this small separation bubble is a precursor of the full leading-edge separation on this airfoil at low Reynolds number. Therefore, having it appear in the spectrum at prestall angles of attack provides a feedback signal to keep the actuator on.

The evidence of a separation bubble that occurs near the leading edge comes from two bits of information. The first is the observation that when the unsteady pressure was measured at the 10\% c location, a larger \( \alpha (1.5^\circ - 2^\circ) \) precursor was obtained with the unsteady pressure measurement measured at the 5\% c location. This indicates that the separation bubble was moving forward as the \( \alpha \) increased. The second indication of our interpretation was the prediction of a separation point between 5 – 10\% c at a 13\° obtained using the X-Foil program.

A more remarkable feature of this method of separation detection is that even after the flow has been reattached by the unsteady plasma actuator, the pressure sensor near the leading edge still senses a peak in the spectrum at the unsteady frequency as long as the flow will not attach naturally. However, even with the actuator on, if \( \alpha \) is low
enough for the flow at the leading edge to be naturally attached, the spectral peak at the actuator frequency is not visible. Thus, this characteristic provides a method to both sense incipient separation to turn the actuator on, and sense when the actuator no longer needs to be on, saving actuator power.

Based on this explanation, a control procedure shown in Figure 2.20 is implemented. First, at any given $\alpha$ of the airfoil, the unsteady plasma actuator is turned on, the pressure sensor time series is sampled, and the frequency spectrum is computed. If a spectral peak is found at the actuator unsteady forcing frequency, the flow is sensed to be close to separation, or at a large $\alpha$ at which if the actuator were turned off, the flow would separate. Thus, the unsteady plasma actuator stays on. If the peak at the unsteady actuator frequency does not appear in the pressure spectra, the airfoil $\alpha$ is low enough for the flow to be far from separating. In this case, the plasma actuator is turned off. This control loop is exercised every time $\alpha$ is changed in the laboratory experiment. In a flight scenario, it would be operating autonomously in the control loop to always sense and control incipient separation.

A demonstration of the control procedure is presented in Figure 2.21. This shows the lift coefficient versus $\alpha$, and drag polar for the baseline airfoil condition (actuator off) and for the feedback control of the actuator using this approach. The results are indistinguishable from the open-loop forcing for the same airfoil conditions with the same increase in $\alpha_{stall}$ and $L/D$ in Figure 2.11. However, in this case, the actuator is only operating when necessary, where in flight scenarios will ultimately use less energy compared to open-loop flow control. When the vortex generated by a plasma actuator goes through the separation bubble, it collects the energy that resides in small eddies in the separated turbulent boundary layer and becomes strong enough to be sensed by the pressure sensor. That causes the dominant frequency to appear in the power
spectrum when the actuator is on. The stronger vortex brings high momentum fluid
to the surface and keeps the leading-edge separation bubble from bursting when the
trailing-edge boundary layer separation moves forward and emerges with the separation
bubble as $\alpha$ increases. After the stall, the lift coefficient continues to increase to $C_{L_{\text{max}}}$
and then drops very gradually instead of an abrupt decrease (see Figure 2.21).
Figure 2.20. Flow chart for the amplitude peak sense-and-control feedback control method.
Figure 2.21. Lift coefficient versus angle of attack and drag polar for the airfoil at 21 m/s with feedback control.
2.4 Leading-edge Separation Control Scaling

In order to study the scaling of leading edge separation control using plasma actuators, a series of airfoil models were constructed. These models were all NACA 4-digit profiles having maximum thickness-to-chord, $t/c$, ratios of 6, 15 and 21 percent. The profiles all had zero camber. Thus the airfoils were NACA 0006, 0015 and 0021 profiles. The advantage of using NACA 4-series airfoils was that there was a direct relation between the $t/c$ and the leading-edge radius, $r_{LE}$, which was

$$\frac{r_{LE}}{c} = 1.1019 \left( \frac{t}{c} \right)^2$$

This allowed us to investigate the effect of the leading-edge radius, and thereby the leading edge Reynolds number, as a parameter affecting leading-edge flow reattachment with the plasma actuator.

2.4.1 Experimental Approach

Baseline measurements of the lift and drag characteristics of each of the airfoils were first performed. For these, the plasma actuator was installed on the leading edge but it was not operated. There were two purposes for these measurements. The first was to determine that the airfoils displayed the proper linear lift versus angle of attack region at lower angles and that the lift-coefficient slope, $dC_l/d\alpha = 2\pi$. This indicated that the airfoils were accurately molded. The second purpose of the baseline measurements was to determine the angle of attack were the airfoils reached the maximum lift and subsequently stalled, $\alpha_s$.

The baseline measurements were conducted at six free-stream velocities of 17, 20, 22, 25, 27, and 30 m/s. The values of the stall angle of attack were found to depend on the airfoil $t/c$ as well as to increase slightly with increasing Reynolds number for the
same airfoil. Therefore the characteristics of the plasma actuator to reattach the flow were referenced to the post-stall angle of attack, $\alpha_s + \alpha$ for each profile and Reynolds number.

Once the baseline characteristics of each airfoil had been established, the following procedure was performed:

1. One of the airfoils was mounted on the lift-drag balance.

2. A free-stream speed was set and the airfoil was placed at a post-stall angle of attack, $\alpha_s + \alpha$.

3. The actuator was operated at an unsteady frequency, $f$, that corresponded to $St = fc/U_\infty = 1$, and the amplitude to actuator was increased until the separated leading-edge flow reattached. The flow reattachment was verified real-time by monitoring the voltage output proportional to the lift. The flow reattachment appeared as a step-change increase in lift. The voltage supplied to the plasma actuator when the flow first reattached was defined as $V_{reattachment}$.

4. The free-stream speed was changed and the process was repeated.

The procedure was performed for three airfoils at the six free-stream speeds. The results were then accumulated.

2.4.2 Results

Figure 2.22 shows the results for the NACA 0006 airfoil. The curves are a smooth spline fit through the measured points. The symbols signify which curves apply to the different post-stall angles of attack. Similar results for the NACA 0015 airfoil are presented in Figure 2.23. and for the NACA 0021 airfoil which are presented in Figure 2.24. From these we observe three general features:
1. The minimum actuator voltage, $V_{reattachment}$, required to reattach the flow increases with an increase in the post-stall angle of attack.

2. The change in voltage required to reattach the flow with increasing post-stall angle of attack, increases with increasing $t/c$, with the voltage required for the NACA 0021 being approximately twice these of the NACA 0006.

3. Over the range of chord Reynolds numbers investigated, there is at most a weak dependence of Reynolds number on $V_{reattachment}$.

Given the weak dependence of $V_{reattachment}$ on Reynolds number, the data were plotted to highlight the dependence on the post-stall angle of attack. This is shown in Figure 2.25 to 2.27.

Figure 2.25 shows the results for the NACA 0006 airfoil. At each post-stall angle of attack, the symbols correspond to the different chord Reynolds. In order to obtain the trend of $V_{reattachment}$ as a function of the post-stall angle of attack, a least-square curve fit was performed between points made up of the average of the $V_{reattachment}$ values for the Reynolds numbers at each $\alpha_s + \alpha$ point. The curves shown in the Figure 2.25 to 2.27 represent linear best-fit curves of the respective minimum voltage to reattach the flow as a function of the post-stall angle of attack. These linear fits have been replotted for three thickness-to-chord ratio airfoil in Figure 2.28.

Figure 2.28 indicates that the minimum voltage at the lowest post-stall angle of attack is approximately the same for the three $t/c$ ratios, but the change with angle of attack increases as $t/c$ increases.

The premise is that $V_{reattachment}$ scales with the leading edge radius, $r_{LE}$. To check this, the slope of the line for the NACA 0015 was scaled to that of the NACA 0021 based on the ratio of their leading-edge radii. The result suggests that the difference
in the slope is proportional to the square of the ratio of the leading-edge radii. This is shown as the line segment labelled as

\[ (\text{Slope 0015}) \cdot \left(\frac{r_{LE}^{21}}{r_{LE}^{15}}\right)^2 \]  

(2.2)

where the ratio of the leading-edge radii are based on Equation 2.1.

If the same scaling is applied between the NACA 0015 and 006 airfoils, the slope labelled as

\[ (\text{Slope 0015}) \cdot \left(\frac{r_{LE}^{6}}{r_{LE}^{15}}\right)^2 \]  

(2.3)

is obtained. This slope is somewhat smaller than shown for the 0006 airfoil in the figure. We suspect that the physically very small radius on this airfoil was increased slightly by the addition of the Kapton film for the actuator. Therefore the behavior was more like an airfoil with a slightly larger leading-edge radius. This radius was still smaller than that of the 0015 airfoil since its slope is smaller.

Summarizing the results for leading-edge plasma actuators designed to increase the stall angle of attack, for \(0.380 \times 10^6 \leq Re_c \leq 0.680 \times 10^6\) and \(2^\circ \leq (\alpha_s + \alpha) \leq 10^\circ\), the voltage required to reattach the flow was

1. only weakly dependent on chord Reynolds number,
2. strongly dependent on the post-stall angle of attack,
3. scaled with the square of the radius of the leading edge.
Figure 2.22. Minimum plasma actuator voltage required to re-attach leading-edge flow separation of NACA 0006 airfoil as a function of chord Reynolds number.
Figure 2.23. Minimum plasma actuator voltage required to re-attach leading-edge flow separation of NACA 0015 airfoil as a function of chord Reynolds number.
Figure 2.24. Minimum plasma actuator voltage required to re-attach leading-edge flow separation of NACA 0021 airfoil as a function of chord Reynolds number.
Figure 2.25. Minimum plasma actuator voltage required to re-attach leading-edge flow separation of NACA 0006 airfoil as a function of post-stall angles of attack.
Figure 2.26. Minimum plasma actuator voltage required to re-attach leading-edge flow separation of NACA 0015 airfoil as a function of post-stall angles of attack.
Figure 2.27. Minimum plasma actuator voltage required to re-attach leading-edge flow separation of NACA 0021 airfoil as a function of post-stall angles of attack.
Figure 2.28. Minimum plasma actuator voltage required to re-attach leading-edge flow separation as a function of post-stall angles of attack for three thickness-to-chord ratios.
CHAPTER 3

V22 WING SEPARATION CONTROL USING PLASMA ACTUATOR

3.1 Experimental Setup

The experiments were conducted in one of the subsonic wind tunnels in the Hessert Laboratory at the University of Notre Dame. The facility is an open-return wind tunnel with 0.61 m by 0.61 m (2 ft square) by 1.8 m (6 ft) long test section. The tunnel consists of a removable inlet having a series of 12 screens followed by an 18:1 contraction that attaches to the test section. The test section is equipped with a Plexiglas side-wall and a glass floor that allows optical access when flow visualization is performed. The back wall of the test section is used to mount the airfoil. Removable hatch doors in the back wall provides access into the test section. In addition, a slot in the test section ceiling provide access for a probe connected to a motorized traverse system. A schematic of the wind tunnel is shown in Figure 3.1.

3.1.1 Airfoil

The airfoil used in this study was a V-22 section shape based on coordinates supplied by Bell Helicopter. It was fabricated in two pieces using a rapid-prototyping process in acrylic resin. A cross-section drawing of the airfoil is shown in Figure 3.2.

The chord length of the airfoil model was 15.24 cm (6 in). In the rapid prototyping process, wing section was fabricated in 7.62 cm (3 in) wide segments. Three of
Figure 3.1: Schematic of open-return wind tunnel used in the experiments.
these segments were placed together to make a total wing span of 22.86 cm (9 in). The segments were held together by threaded rods that passed through holes that were fabricated in the airfoil sections. A hollow support tube was inserted into the largest fabricated hole located at the maximum thickness point on the airfoil (see Figure 3.2).

The airfoil was mounted in the tunnel by passing the support tube through the back wall of the test section. The tube was held in place by a chuck that allowed angular motion for setting the angle of attack. Round 40 cm (15.75 in) diameter end plates were used to minimize 3-D end effects on the airfoil. These were constructed from clear Plexiglas to allow visual access for flow visualization over the surface of the airfoil. Figure 3.3 shows the flap at a 70° deflection which is the schedule hover position provided by Bell. All of the measurements were performed with this configuration. Also indicated in this view is the hinge-gap cover. This was positioned according the drawing provided for the 70° flap deflection.

3.1.2 Flow Visualization

Flow visualization was done by introducing continuous smoke streaklines upstream of the wind tunnel screens and contraction. The smoke emanated from a rake of tubes as low-speed laminar jets. The tubes were aligned in the vertical direction and located at the spanwise centerline. The smoke streaks were drawn in the tunnel inlet
Figure 3.3. Location of plasma actuator on 70° deflected trailing edge flap of V22 wing.

and converged into a closely spaced, vertical aligned group following the contracting flow. Figure 3.5 and Figure 3.6 show schematic drawings of the smoke rake and smoke generator respectively.

The smoke steaks were illuminated by a steady, high-intensity light source located below the tunnel test section (see Figure 3.4). The light entered through a glass slot in the floor of the test section. A mirror on the ceiling of the test section reflected some of the light to provide illumination in the shadow that formed upstream of the wing section.

The flow visualization records were made using a Panasonic analog video camera. The video tape was digitalized through a video capture card where individual frames were extracted.
Figure 3.4: Schematic of test section.
Figure 3.5. Schematic of smoke rake used for subsonic wind tunnel.

Figure 3.6. Schematic of smoke generator used for subsonic wind tunnel.
3.1.3 Wake Velocity and Drag Measurements

Velocity measurements in the wake of the wing section were made using a pitot-static probe. The probe was attached to a motorized traversing mechanism that was on top of the test section. The probe entered the test section through a slot in the ceiling that was located on the spanwise centerline (see Figure 3.4).

The pitot probe was located three chord length 45.73 cm (18 in) downstream of the wing section. The profiles encompassed approximately 45 cm (18 in) in the cross-flow direction. This was sufficient to reach the free-stream on either side of the wake. A total of 46 equally spaced spatial points were taken. This then corresponded to a spacing of 1.02 cm (0.4 in) between points.

The pitot-static probe was connected to a differential pressure transducer that provided a voltage proportional to pressure. The pressure transducer had been calibrated against a known pressure reference. The output from the pressure transducer was acquired through the analog-to-digital converter of a digital acquisition board in a laboratory computer. The data acquisition included sampling the voltages from the pressure transducer, and moving of the traversing mechanism. These were controlled by in-house written software that ran under the Linux operating system of the computer. At each spatial point, the pressure transducer voltage was sampled and a mean value was calculated until the variation in the mean was within a specified tolerance.

A separate pitot-static probe in the free-stream was located at the entrance to the test section to monitor the free-stream velocity, $U_\infty$, for the experiments. Two free-stream speeds of 10 and 20 m/s were used.

The drag on the airfoil was calculated using the momentum equation for a control volume that encompassed the region upstream and downstream of the model. This involved the mean velocity profile in wake, as well as the free-stream static pressure.
measured upstream and downstream of the airfoil. The drag was determined as

\[
\frac{D}{c} = \int_0^h (P_0 + \rho U_0^2)dy - \int_0^h (P_1 + \rho U_1^2)dy
\]

(3.1)

where subscripts 0 and 1 refer to respective measurement stations upstream and downstream of the airfoil, \(P\) is the static pressure, \(U\) is the mean velocity, and \(y\) is the cross-stream direction with a total length of \(h\) that totally encompassed the width of the wake of the airfoil.

Upstream of the airfoil, the mean velocity profile is uniform. Also, the static pressure is not a function of the cross-stream location. Therefore the equation for drag reduces to

\[
\frac{D}{c} = h(P_0 - P_1) + h\rho U_0^2 - \int_0^h \rho U_1^2 dy
\]

(3.2)

3.1.4 Plasma actuators

Asymmetric plasma actuators were used to control the separated flow on the V-22 wing section. For this, actuators were located at two locations: one at the 17 percent chord location of the trailing-edge flap, and the other on the leading edge of the main wing section body. The former is illustrated in Figure 3.7.

The plasma actuators use a 5 kHz a.c. to ionize the air and produce the electric field that results in the body force on the ambient flow. For separation control, we have found that a periodic excitation at a lower frequency corresponding to a Strouhal number, \(St = f c / U_\infty \simeq 1\) was most effective. This lower frequency operation consisted of cycling the a.c. voltage off and on with an unsteady period, such as illustrated in Figure 2.6. The percentage of time (duty) within the period that the a.c. was on was also controllable. For separation control a duty cycle of only 10% was found to be
3.2 Results

3.2.1 10 m/s Free-stream Velocity

The first results correspond to the lower free-stream speed of 10 m/s. The visualized flow around the wing section at this speed is shown in Figure 3.8. The top photograph corresponds to the base condition when the actuators are off. The bottom photograph corresponds to when both the leading edge and trailing edge (flap) actuators were on. It is apparent from these photographs that the actuators reduce the width of the wake. In addition vortex indicates that the streamwise extent of the wake is reduced.

The flow visualization provides a qualitative indication of the effectiveness of the actuators. Quantitative results of their effect comes from mean velocity measurements.
Figure 3.8. Visualized flow around V22 airfoil in hover configuration for 10 m/s free-stream velocity with actuators off (top), and leading and trailing edge actuators on (bottom).
Figure 3.9. Comparison of mean profiles in the wake of the V22 wing section in the hover configuration with the actuator off and the trailing-edge (flap) actuator on. $U_\infty = 10 \text{ m/s}$ in the wake of the wing section. These are shown in Figure 3.9 to 3.12 for the 10 m/s free-stream speed.

Figure 3.9 shows the effect of trailing-edge actuator on the mean velocity profile downstream of the wing section. In this and all the other figures of this type, the square symbols correspond to the condition with actuators off. This shows a well defined wake which is fairly symmetric about the minimum velocity location. In the orientation of the plot, the trailing edge actuator is at the top. When the trailing edge actuator is on, there is a significant increase in the velocity in the wake that signifies a reduction in drag. The mean profile is also no longer symmetric, with the velocity increasing in the upper part of the profile, where the actuator is located.

Figure 3.10 shows the effect of leading-edge actuator on the mean velocity profile
downstream of the wing section. In the orientation of the plot, the leading edge actuator is at the bottom. When the leading edge actuator is on, there is again a significant increase in the velocity in the wake that signifies a reduction in drag. The mean profile is also no longer symmetric, with the velocity increasing now in the lower part on the profile, where the actuator is located.

For reference, the individual effects of the two actuators are cross plotted in Figure 3.11. The effect of operating both actuators together is shown in Figure 3.12. This condition corresponds to the flow visualization image in the bottom part of Figure 3.8. With both actuator on, the wake profile embodies the sum of the effect of the individual actuators. In terms of the drag, the drag coefficient, $C_D = D/(1/2 \rho U_{\infty}^2 c)$, was compared. The result is shown in Table 3.1. At 10 m/s free-stream velocity the actuators
Figure 3.11. Comparison of mean velocity profiles in the wake of the V22 wing section in the hover configuration with the actuator off and the trailing-edge or leading-edge actuators on separately. $U_\infty = 10$ m/s
Figure 3.12. Comparison of mean velocity profiles in the wake of the V22 wing section in the hover configuration with the actuator off and both the trailing-edge and leading-edge actuators on simultaneously. $U_\infty = 10 \text{ m/s}$
TABLE 3.1

DRAG COEFFICIENT

<table>
<thead>
<tr>
<th>$U_\infty$</th>
<th>Actuator</th>
<th>$St$</th>
<th>$C_D$</th>
<th>Improvement</th>
</tr>
</thead>
<tbody>
<tr>
<td>10 m/s</td>
<td>Off</td>
<td>-</td>
<td>0.561</td>
<td>-</td>
</tr>
<tr>
<td>10 m/s</td>
<td>TE &amp; LE On</td>
<td>1.0</td>
<td>0.313</td>
<td>44%</td>
</tr>
<tr>
<td>20 m/s</td>
<td>Off</td>
<td>-</td>
<td>0.487</td>
<td>-</td>
</tr>
<tr>
<td>20 m/s</td>
<td>TE &amp; LE On</td>
<td>1.3</td>
<td>0.354</td>
<td>27%</td>
</tr>
</tbody>
</table>

resulted in a 44% drag reduction.

3.2.2 20 m/s Free-stream Velocity

The next set of results correspond to the higher free-stream of 20 m/s. The visualized flow around the wing section at this speed is shown in Figure 3.13. The top photograph corresponds to the base condition when the actuators are off. The bottom photograph corresponds to when both the leading edge and trailing edge (flap) actuators were on. As in the lower velocity case, it is apparent from these photographs that the actuators reduce the width and streamwise extent of the wake.

The Strouhal number of the periodic forcing that was found to give the best results at the 20 m/s free-stream speed was $St = 1.3$. This is opposed to the lower velocity case in which $St = 1.0$ was optimum. In both cases, the airfoil chord length (6 in) was used as the reference length. This may remain an open parameter for achieving optimum control for other free-stream speeds.

Quantitative results of the effect of the actuators at the 20 m/s free-stream speed...
Figure 3.13. Visualized flow around V22 airfoil in hover configuration for 20 m/s free-stream velocity with actuators off (top), and leading and trailing edge actuators on (bottom).
comes from mean velocity measurements in the wake of the wing section. These are shown in Figure 3.14 to 3.17.

Figure 3.14 shows the effect of trailing-edge actuator on the mean velocity profile. As in the lower velocity cases, when the trailing edge actuator was on, there was a significant increase in the velocity in the wake that signified a reduction in drag.

Figure 3.15 shows the effect of leading-edge actuator on the mean velocity profile downstream of the wing section. Although the mean velocity profile with the actuator off indicates that the flow may not be separated at the leading edge, when the leading edge actuator is on, we note a significant increase in the velocity in the wake that signifies a reduction in drag.
Figure 3.15. Comparison of mean velocity profiles in the wake of the V22 wing section in the hover configuration with the actuator off and the leading-edge actuator on. $U_\infty = 20$ m/s
Figure 3.16. Comparison of mean velocity profiles in the wake of the V22 wing section in the hover configuration with the actuator off and the leading-edge or trailing-edge actuators on separately. $U_\infty = 20 \text{ m/s}$
Figure 3.17. Comparison of mean velocity profiles in the wake of the V22 wing section in the hover configuration with the actuator off and both the leading-edge and trailing-edge actuators on simultaneously. $U_\infty = 20$ m/s
Figure 3.18. Schematic showing a modified flap-gap cover for 70° deflected flap on V22 airfoil section.

The individual effects of the two actuators are cross plotted together in Figure 3.16. The effect of operating both actuators together is shown in Figure 3.17. This condition corresponds to the flow visualization image in the bottom part of Figure 3.13. With both actuators on, the wake profile again embodies the sum of the effect of the individual actuators. In terms of the drag, Table 3.1 documents a 27% decrease in the drag coefficient at this free-stream speed.

It became apparent in the flow visualization studies that the flow over the trailing-edge flap was very sensitive to the flap-gap cover. In particular, we investigated an alternate design which represents a “passive” separation control. This modification is illustrated in Figure 3.18.

The rational for the modified flap-gap cover was to provide a smoother transition from the shorter cover onto the flap surface. The flow visualization had indicated that
Figure 3.19. Comparison of mean velocity profiles in the wake of the V22 wing section in the hover configuration with the original and modified hinge-gap cover. $U_\infty = 20$ m/s.

the flow separated downstream of the “bump” that was left just downstream of the end of the original cover.

Evidence of the improvement obtained from the modified cover is presented in Figure 3.19. This compares mean velocity profiles downstream of the wing section with the original flap cover (square symbols) and with the modified cover (diamond symbols). With the modified cover, the wake profile became symmetric about the minimum velocity location.

Although the “passive” improvement of the modified flap-gap cover was evident, how did it compare to the “active” control using the plasma actuators? To illustrate this, the mean velocity profile in the wake for the case with the original gap cover. This is shown in Figure 3.20.
Figure 3.20. Comparison of mean velocity profiles in the wake of the V22 wing section in the hover configuration with the modified hinge-gap cover and for both actuators on with the original gap cover. $U_{\infty} = 20$ m/s

In both cases, the mean velocity profiles are symmetric about the minimum velocity locations. However it is clear that the active control increased the velocity in the wake, signifying a decrease in the drag compared to the passive control.
CHAPTER 4

PLASMA ACTUATORS FOR HINGELESS AERODYNAMIC CONTROL OF UCAV

4.1 Introduction

The present work explores the application of a DBD plasma actuator for controlling the longitudinal dynamics of a three dimensional UCAV with a 47° leading-edge sweep. The UCAV configuration chosen for this study is based on a previously examined U.S. Air Force-Boeing 1303 UCAV design. The vehicle is basically a blended wing body on which the fuselage is blended smoothly with the wing, with a varying cross-section along the span and ±30° trailing-edge sweep angle. In its conventional configuration, the 1303 UCAV features movable flap and split aileron at the trailing edge to control the vehicle. The goal of this research is to demonstrate the feasibility of a plasma wing: a flying wing that uses plasma flow control technology to create aerodynamic control moments of sufficient magnitude so that conventional moving aerodynamic controls could be eliminated. Because the 1303 UCAV contains a 47° leading-edge sweep, a discussion on the leading-edge vortex (LEV) is relevant to touch upon.

Swept wings of low aspect ratio are commonly used on high-speed aircraft because of their favorable wave-drag characteristics. The LEV is the main feature of the flow over swept wings that provide lift for flight control at high angles of attack. At low angles of attack and lower speeds, however, the aerodynamic behavior of swept wings
is vastly different from that of the high-aspect ratio wings. The performance of swept wings outside the high-speed, high-alpha envelope is crucial, because the mission roles of modern aircraft require them to operate at low-speed and low-alpha conditions during different portions in flight (e.g., takeoff, landing, etc.). The formation of the LEV and subsequent vortex breakdown (VBD) phenomena over a swept wing are highly influenced by a number of parameters including angle of attack, leading-edge design, and adverse pressure gradients, which present unique challenges in controlling the vehicle dynamics at different flow conditions. For example, at low angles of attack, the flow remains attached to the surface and the location of the (weak) VBD is usually associated with a loss in vortex lift, which has been shown to cause changes in the lift, drag, and pitching moments of the swept air vehicle [36, 53, 69]. At large angles of attack, the upper wing surfaces show the presence of complex vortex systems that dominate the leeward flowfield and cause the wingtip separation [68].

In the past decade, several researchers have employed flow control methods to control the LEV and VBD phenomena for improved aerodynamics of swept wings. For example, Moeller and Rediniotis [41] demonstrated control of the pitching moment of a 60° swept delta-wing model at high angles of attack using a series of surface mounted pneumatic vortex control actuators. Control was achieved by altering the vortex breakdown phenomena that affected the chordwise lift distribution over the wing, ultimately resulting in an induced pitching moment. Amitay et al. [4] reported an experimental study on the use of synthetic jet actuators on a 1301 UCAV design (nickname Stingray). The design of Stingray [4] and the present 1303 UCVA share some similarity in that the leading-edge sweep angle is approximately 50°, leading to similar three dimensional flow patterns over the wing. Amitay et al. [4] showed that at conditions in which the flow was normally separated from the leading edge, between 14° and 24° angles of
attack, the zero-mass jets were able to produce significant forces and moments on the vehicle. Visser et al. [66, 67, 70] employed steady spanwise blowing to control leading-edge vortex breakdown and asymmetric roll-moment conditions.

In a more recent effort, a computational study on the aerodynamic performance of a 1303 UCAV design for different leading-edge designs was reported by Zhang et al. [75]. The effects of three leading-edge designs (a basic profile, a rounded leading edge similar to the one used in our study, and a sharp leading edge) were investigated using the NPARC code for a Mach number of 0.25 and angles of attack ranging from $\alpha = -5^\circ$ to $20^\circ$. It was found that there were only minor differences among pressure distributions with the three configurations for both the computed and experimental data. The predicted pressure distributions compared favorably with the wind tunnel measurements for all regions except near the wingtip, for which the computations did not consistently predict the separations. At small angles of attack, flowfield studies showed attached, smooth, and well-behaved flow.

In general, the 1303 UCAV that have been developed and successfully flown rely on multiple control surfaces distributed across the wing to provide control moments for trim and maneuvering. Each control surface is essentially a trailing-edge flap that when deflected, changes the lift, drag, and pitching moment over that portion of the wing. By suitably arranging multiple flap across the wing, one can create moments to pitch, roll, or yaw the wing, and moments to trim the wing at a particular flight condition. The ultimate objective of the present work is to demonstrate hingeless flight control with limited or no use of conventional control surfaces.
4.2 Experimental Setup

The UCAV planform used in this study is based on a 1303 design with varying cross sections, a 47° leading-edge sweep, and a ±30° trailing-edge sweep (shown in Figure 4.1). The design was originally developed by the U.S. Air Force Research Laboratory (AFRL) in conjunction with the Boeing Phantom Works, and was recently used as a benchmark for a joint computational fluid dynamics code validation effort by The Technical Cooperation Program (TTCP). TTCP involved a consortium of governmental interests in five countries to study the performance predictions of various leading edges. In the present work, a 1303 configuration with a relatively blunt leading edge was used.

A photograph of a half-span model of the scaled 1303 UCAV used for wind tunnel tests is shown in Figure 4.1. The model has a root chord of 40.64 cm (16 in) and a half-span dimension of 13.97 cm (13.375 in). The same model with traditional control surfaces - flap and split ailerons (shown in Figure 4.4) was also tested in this study. The models were cast from a numerically machined two-piece aluminum mold. The
casting material was a mixture of epoxy and microglass beads that resulted in a very rigid model that accurately duplicated the mold shape.

Wind-tunnel experiments were conducted for angles of attack ranging from 0 to 35°. Many of the tests were performed from \( \alpha = 0^\circ \) to 25°, however additional tests were later conducted for angles of attack up to 35°. Lift and drag measurements on the half-span models were conducted in the 0.42 m (1.39 ft) square by 1.8 m (6 ft) long cross-section, low-speed wind tunnel. All experiments were conducted at a chord Reynolds number, \( Re_c \), of \( 4.12 \times 10^5 \) based on the mean aerodynamic chord, which corresponds to a free-stream velocity, \( U_\infty \), of 15 m/s.

The half-span model was mounted vertically on the support sting of a lift-drag force balance that was mounted on the top of the test section. The model was suspended below a splitter plate that was attached to the ceiling of the test section. The splitter plate was designed to produce a two-dimensional flow with a thin boundary layer leading up to the model. A hole in the ceiling splitter plate accommodated the sting supporting the model. Wiring for the plasma actuators also entered through the hole. This hole was aligned with the support sting so that it would not interfere with angular positioning of the model when setting different angles of attack. A stepper motor on the force balance drove the angular position of the support sting. Its motion was controlled by the data acquisition computer through software; with this, the angular position was repeatable to \( \pm 0.005 \) degrees.

The force balance consisted of independent lift and drag platforms. The lift platform was supported on the drag platform by two vertical plates that flex only in the lift direction. The drag platform was supported by two plates that flex only in the drag direction and hang from two more vertical plates attached to the fixed base of the force balance. Both the lift and drag platforms were connected to separate flexures on which
foil strain-gauge bridges were mounted. The strain-gauge bridges provided voltage outputs proportional to the respective lift and drag forces. The voltages were amplified using custom-designed operational amplifier circuits that minimized offset drift and sensitivity to external electronic noise. Calibration of the force balance was done by applying known weights to a cable pulley system attached to the support sting. The average uncertainty in the force measurements was 0.63% in lift and 0.9% in drag.

The experiment was controlled by a digital computer with a programmable analog-to-digital converter (ADC) and digital input-output (DIO) interface. The minimum voltage resolution of the ADC was 0.6 mV. The voltages proportional to the lift and drag forces were acquired along with a voltage proportional to the velocity at the entrance to the test section. The acquisition software was programmed to acquire 10,000 voltage samples over 10 s. This was found to provide repeatable time-averaged statistics that varied by less than 0.1%. The angular position of the airfoil was controlled by voltage pulse from the DIO into the stepper motor controller; with this, the angular position was repeatable to within ±0.005 degrees. A mechanical readout that was geared to the stepper motor shaft provided positive feedback on the angular position.

Before making lift-drag measurements, values of the lift and drag voltages were first acquired at different angles of attack without flow. Any difference from the zero-force voltage that was due to eccentric loading was recorded and subtracted from the results at the same angles of attack with flow. This process was repeated any time the model was removed from the force balance. The free-stream speed at the entrance to the test section was measured with a pitot-static probe connected to a pressure transducer. The output of the transducer was monitored on a DC volt meter and simultaneously acquired by the data acquisition computer when the voltages proportional to the lift and drag forces were acquired. Based on the pressure transducer calibration, the accuracy of
the free-stream speed measurement was 0.01 m/s. The combination of the uncertainties in the force measurements and voltages resulted in an average uncertainty in the lift and drag coefficients of approximately 1%.

The high voltage leads to the plasma actuator were well shielded to minimize electronic noise effects, if any. The output from the lift and drag channels from the force balance were connected to an oscilloscope to check for RF noise interference when the plasma actuator is turned on. No noise interference was noticed on the scope. This confirmed that the plasma actuator did not corrupt the force data.

These experiments began with lift and drag measurements of the model with traditional control surfaces. Then lift and drag measurements were performed on the half-span model to identify the optimal location for plasma actuators for lift control. These involved plasma actuators on the lee side and wind side of the wing leading edge.

In the present work, the plasma actuator consisted of two 0.05 mm (0.002 in) thick copper electrodes separated by two layers of 0.05 mm (0.002 in) thick Kapton film. The Kapton film has a breakdown voltage of approximately 7 kV per $10^{-3}$ in thickness and a dielectric constant of 3.3, which provide good electrical properties. The electrodes were arranged in an asymmetric arrangement. They were overlapped by a small amount (on the order of 0.5 mm) to ensure a uniform plasma in the full spanwise direction. The plasma actuator was bonded directly to the surface of the wing. When the a.c. voltage amplitude was large enough, the air ionized in the region over the covered electrode. A 0.1 mm recess was molded into the wing model to secure the actuator flush to the surface. The two copper-foil electrodes were aligned parallel with the leading edge. The spanwise length of the actuators was 90% of the wing span. With this arrangement of electrodes, the body force produced by the actuator would introduce a velocity component in the direction from the exposed electrode toward the covered...
electrode. With the actuator oriented on the leading edge, it includes a flow around the leading edge of the wing. Many different actuator arrangements were examined on both the lee-side and wind-side portion of the wing. Figure 4.2 shows an example of a plasma wing configuration with multiple plasma actuators at the wing leading edge.

![Figure 4.2. Photograph of 4.16%-scale half-span wing model used for plasma actuator experiments.](image)

In all of the plasma control experiments presented here, the actuators were operated in an unsteady manner, as shown in Figure 4.3. The a.c. carrier frequency supplied to the electrodes was 5 kHz and the a.c. voltage supplied to the electrodes was on the order of 3-12 kV_{p−p}. The power used by the actuator was approximately 2-4 W per linear foot of actuator span. In the unsteady mode, very short duty cycles are possible, which reduces the actuator power requirements significantly. For example, a 10% duty cycle provided results better than those of the steady operation, which used a 100% duty cycle. The unsteady actuator frequency, f_{mod}, was determined based on Strouhal
number $St = f_{mod} c / U_\infty = 1$, where $f_{mod}$ is the modulation frequency, $c$ is the mean aerodynamic chord, and $U_\infty$ is the free-stream velocity. Previous measurements had shown that $St = 1$ caused the optimum conditions to reattach leading-edge separation. For all cases presented here, the unsteady modulation frequency of the actuator was $\sim 166$ Hz and the actuator was operated at a 10% duty cycle.

Figure 4.3. Example of short-duty-cycle a.c. input for the unsteady operation of plasma actuators.

4.3 Experimental Results

Figure 4.4(top) shows a photograph of the wing trailing edge with conventional control surfaces, flap, and split ailerons. Figure 4.4(bottom) shows a photograph of passive devices with $1^\circ$, $3^\circ$, and $5^\circ$ deflections used in the wind-tunnel testing for the conventional control surfaces. The dimensions of the (inboard) flap and (outboard) split aileron (hereafter, aileron) are $0.1 \times 0.04$ m and $0.13 \times 0.038$ m, respectively, as shown in Figure 4.4. The effects of individual and combined deflection of the flap
Figure 4.4. Photograph showing the trailing edge of the wing model with conventional flap/aileron control surfaces.
and aileron by 1°, 3° and 5° on the lift forces of the wing were measured for $\alpha = 0°$ to 25° with 1° increments. These tests were conducted primarily to provide a basis for comparison with the plasma actuator effects in providing control at high angles of attack and, ultimately, to demonstrate improvements in control authority and the operational flight envelope of the wing using the plasma actuators. The results from these tests are highlighted in Figure 4.5 to 4.7.

Figure 4.5 shows the effects of deflecting the aileron (indicated as A) by 1°, 3° and 5°, respectively, while holding the flap (indicated as F) at 0° (no deflection) on the lift coefficient for $\alpha = 0°$ to 25°. In a similar fashion, Figure 4.6 shows the effects of deflecting the flap by 1°, 3° and 5°, and Figure 4.7 shows the effects of deflecting both the flap and aileron together by 1°, 3° and 5°, respectively. No appreciable effects are observed in the aerodynamic forces for aileron deflection of 1° and 3° compared with the baseline case (see Figure 4.7). For aileron deflection of 5°, shown in Figure 4.5, a noticeable shift in the lift curve is observed for angles of attack up to 15°. This is the classical response of a plane flap that corresponds to an increase in the zero-lift angle of attack that is equivalent to an increase in the wing camber.

Comparisons between Figure 4.5 and Figure 4.6 shows that the effect of the inboard flap, which is smaller in size than the aileron, is stronger than the effect of the outboard aileron on the lift generated. Because the flow remains nominally attached at low angles of attack, the effects of the flap and aileron are considerably stronger than those with high angles, at which flow separation and LEV become dominant. This limits the effectiveness of both the flap and aileron at high angles of attack. The maximum effect is observed when both the flap and aileron are deflected simultaneously (see Figure 4.7). The lift augmentation effects of both control surfaces are roughly additive. In general, a linear shift in the lift coefficient corresponding to an increase in the zero-lift angle of attack.
attack is observed at low angles of attack by deflecting the flap/aileron, whereas at high angles of attack, their effects are nonexistent. Therefore, for flight control at higher angles of attack, conventional trailing-edge devices are not suitable. This is the basis for applying plasma actuators on the 1303 wing; that is, to demonstrate how the control authority can be greatly enhanced using a hingeless plasma flow control system.

Surface flow visualization was performed on the half-span wing by Patel et al. [47]. The results are shown in Figure 4.8. These surface visualizations captured the detail of the flow pattern associated with the leading-edge flow separation and reattachment. At an angle of attack of 10-deg, the primary vortex core and vortex at the wingtip are observed clearly. At 16-deg, the breakdown location of the primary vortex moved inboard to approximately half of the wing span. The vortex at the wingtip was observed to expanded to cover all the region that would be occupied by an aileron.

The results from different plasma actuator experiments are present in Figure 4.9 to 4.14. Figure 4.9 to 4.13 highlight the results from the plasma actuator experiments conducted on the half-span wing model for achieving control at high angles of attack. The details of the specific plasma actuator configurations examined (e.g. P9, P10, P15, P18 and P19) are included in the captions for Figure 4.9 to 4.13. In general, the letter P indicates a case with the plasma actuator on, and the number flowing P indicate a specific plasma actuator configuration, the details of which are provided in the corresponding figure captions. Figure 4.9 to 4.11 show the effects of P9, P10 and P15 plasma configurations on the lift coefficient and drag polar, respectively for angles of attack ranging from $0^\circ$ - $35^\circ$. A comparison of the effects of P9, P10 and P15 with the baseline reveals noticeable effects of the plasma actuators for $\alpha > 15^\circ$, at which the VBD occupies most of the leading edge. There is negligible difference between the effects of P9 and P10 for all angles of attack tested (see Figure 4.9 to 4.10), which
Figure 4.5: Effects of deflecting aileron by 1°, 3°, and 5° on the lift coefficient for \( \alpha = 0^\circ \) to 25°.
Figure 4.6: Effects of deflecting flap by $1^\circ$, $3^\circ$, and $5^\circ$ on the lift coefficient for $\alpha = 0^\circ$ to $25^\circ$. 
Figure 4.7: Effects of deflecting both flap and aileron by $1^\circ$, $3^\circ$, and $5^\circ$ on the lift coefficient for $\alpha = 0^\circ$ to $25^\circ$. 
Figure 4.8: Wind tunnel flow visualization photographs of the suction side of a 47-deg swept wing with a blunt leading edge[47].
suggests that the effects of plasma control on the leeward surface are similar for the inboard and outboard sections of the wing. However, Figure 4.11 demonstrates a significant rise in the lift coefficient and a shift in the drag bucket, respectively. Plasma configuration P15 uses two unsteady actuators at the leading edge, placed slightly on the windward side of the leading edge. Studies have shown that at higher angles of attack, the primary separation location moves upstream, closer to the windward side of the wing, which seems to explain why P15 is more effective than P9 and P10.

Additional tests were conducted to verify this speculation and results from the new actuator configurations P18 and P19, which also used actuators slightly on the windward side. Results shown in Figure 4.12 and 4.13 confirmed the effects of the leading-edge flow control observed earlier. Figure 4.14 shows a comparison of the plasma actuator effects with the conventional flap/aileron cases and the control-off case. This demonstrates the significant control forces that were generated using plasma actuators at high angles of attack.
Figure 4.9. The effects of plasma actuators on the lift coefficient (top) and drag polar (bottom) of the 47° swept wing of a range of angles of attack. P9: unsteady actuator located on the inboard section of the leading edge slightly on the lee side covering 40% of wing span.
Figure 4.10. The effects of plasma actuators on the lift coefficient (top) and drag polar (bottom) of the 47° swept wing of a range of angles of attack. P10: unsteady actuator located at the outboard leading edge slightly on the lee side covering 40% of wing span.
Figure 4.11. The effects of plasma actuators on the lift coefficient (top) and drag polar (bottom) of the 47° swept wing of a range of angles of attack. P15: a pair of unsteady actuators located at the inboard and outboard sections of the leading-edge slightly on the wingward side covering 80% of wing span.
Figure 4.12. The effects of plasma actuators on the lift coefficient (top) and drag polar (bottom) of the 47° swept wing of a range of angles of attack. P18: a pair of unsteady actuators located at the inboard section (one on the windward side and the other on the leeward side).
Figure 4.13. The effects of plasma actuators on the lift coefficient (top) and drag polar (bottom) of the 47° swept wing of a range of angles of attack. P19: a pair of unsteady actuators located at the inboard section (one on the inboard windward side and the other on the outboard leeward side)
Figure 4.14: Comparison of plasma effects with conventional control surfaces at higher angles of attack.
Figure 4.15. Schematic of 20° separation ramp on the suction side of the wing.

To achieve control at low angles of attack, wind tunnel experiments were conducted on a wing model with a slightly modified trailing edge incorporating a separation ramp on the suction side, see Figure 4.15. The main objective of incorporating a ramp was to enhance the effect of plasma actuator at low angles of attack by manipulating the local flow separation region caused by the ramp, and by affecting circulation around the wing. A pair of plasma actuators were placed on the suction side of the wing; one on the inboard section and the other on the outboard section. The actuators were positioned such that the junction of the copper electrodes lined up perfectly with the reflex of the ramp.

The results from the most effective actuator configuration tested (P25) are shown in Figure 4.16. The square symbols represent the actuator-off case and circle symbols represent the actuator-on case. Figure 4.16 shows the lift coefficient and drag polar for the -25° to 10° α-range for the P25 and actuator-off cases. An increase in the lift coefficient at negative angles of attack (from -18° to 3°) is achieved. Changes in the drag coefficient are negligible.
Figure 4.16. The effects of unsteady plasma actuators located at the onset of the 20-deg separation ramp on the suction side.
The following investigates the potential for the use of the plasma actuators to replace the moving conventional control surfaces for roll control on aircraft. Figure 4.17 shows an illustration of a half-span UCAV wing with wing span, $b$, and root chord, $c$. Assuming for simplification, that the lift force is uniform in span over the area $S_a$, the magnitude of the roll moment is

$$L' = \triangle C_L q S_a r_a \quad (4.1)$$

The roll moment coefficient is

$$C_{L'} = \frac{L'}{q S_w b} = \frac{\triangle C_L S_a r_a}{S_w b} \quad (4.2)$$

where $S_w$ is the wing area. The moment arm $r_a$ is the distance from the spanwise center of lift to the roll axis location. In this case, $S_a/S_w = 0.8$, $r_a/b = 0.5$, $\triangle C_L$ for
leading-edge plasma actuators and trailing-edge plasma actuators obtained from experiments. Substituting these values in Equation 4.2, the result is shown in Figure 4.18 and Figure 4.19. Clearly, the plasma actuators generated much larger roll moment than the conventional control surfaces at higher angles of attack. For trailing-edge plasma actuators, they increased the roll moment from $-18^\circ$ to $3^\circ$. 

Figure 4.18. Comparison of plasma effects with conventional control surfaces on roll moment at higher angles of attack.
Figure 4.19. The effects of plasma actuators on roll moment at low angles of attack.
A summery of the results for different configurations is the following:

- The leading-edge actuator on the inboard half of the wing was only effective for angles of attack greater than 20°. The leading-edge actuator on the outboard half of the wing was effective for angles of attack from 9° up to the largest angle examined.

- The trailing-edge actuator with a 20° separation ramp provided lift control at low angles of attack from -18° to 3°.

- The combination of these could cover and expand the flight envelope offered by conventional flaps.
CHAPTER 5

NUMERICAL AND EXPERIMENTAL ANALYSIS OF PLASMA FLOW CONTROL OVER A HUMP MODEL

5.1 Introduction

Currently a great deal of interest within the community is to utilize the emerging flow control technology to design revolutionary Uninhabited Air Vehicles without moving control surfaces while still maintaining controlled flight. Corke et al.[19] proposed a plasma flow control optimized airfoil concept. It uses a laminar airfoil design that maintains a favorable pressure gradient over as much as the upper surface as possible and incorporate a separation ramp at the trailing edge that can be manipulated by a plasma actuator in order to control lift. At lower angle of attack such as cruise, laminar boundary layer maintains downstream until it separates over the separation ramp. The separation is a laminar separation which can easily be controlled with an unsteady plasma actuator at very low power levels. However at higher angle of attack such as take-off and landing, transition to turbulence may occur somewhere before the ramp due to large adverse pressure gradients. Thus the separation over the ramp may be a turbulent boundary layer separation. Plasma actuators for turbulent separation control are therefore needed for full range of flight plans.

A wall-mounted hump model was selected in this study. It consists of a relatively long fore body and a short separation concave ramp near the aft part of the model.
This shape was chosen because the separation location is not sensitive to Reynolds number, and the detailed experimental data are well-documented in the literature[25]. It also was used in NASA Langley CFD validation workshop for synthetic jets and turbulent separation control in 2004[56]. A broad range of numerical techniques such as RANS, LES and DNS have been performed to predict the separated flow fields and its control[73]. So with all this information, it is easy for us to make comparisons.

Turbulent separation control can be achieved either by injection of high momentum fluids, or mixing of outer and inner boundary layer fluid. The spanwise electrode arrangement which is often used in laminar separation control meets the former requirement. The body force generated by this arrangement accelerates the low momentum fluids around the separation location to overcome the adverse pressure gradient. To achieve the boundary layer mixing, a streamwise electrode arrangement was designed in order to generate a pair of counter-rotating vortices which introduce the high momentum fluids from outer boundary layer into the inner boundary layer to delay or suppress the separation.

5.2 Experimental Setup

5.2.1 Wind Tunnel and Hump Model

The experiments were conducted in one of the subsonic wind tunnels in the Center for Flow Physics and Control (FlowPAC) in the Hessert Laboratory at the University of Notre Dame. The facility is an open-return blow-down wind tunnel with a 40.64 cm (16 in) wide by 40.64 cm (16 in) high by 182.88 cm (6 ft) long test section. The tunnel consists of a removable inlet having a series of 12 screens followed by a 24:1 contraction that attaches to the test section. The test section is equipped with a clear Plexiglas side-walls that allow optical access to view the model for flow visualization. The back
A 38.1 cm wide by 162.56 cm long flat Plexiglas plate was mounted 10.16 cm above the floor of the test section, so the test section height became 30.48 cm. The hump model was mounted on the top of the flat plate. The flat plate extended 82 cm upstream of the leading-edge and 42 cm downstream of the trailing-edge of the hump model. A strip of sand paper was attached to the elliptical leading-edge of flat plate in order to trip the flow into a fully-developed turbulent boundary layer. Two end-mounted plates were designed to produce two dimensional flow field over the hump. Each end plate was 82 cm long and 23.5 cm high.

A cross-section drawing of the hump model is show in Figure 5.2. It consists of a relatively long fore body and a relatively short concave ramp at the aft part of the model. The slopes of the leading-edge and trailing-edge are very close to zero, which makes a smooth connection with the flat plate. The hump model has a 42 cm chord and
Figure 5.2. Cross-section drawing of wall-mounted modified Glauert hump model.

a 35.6 cm span. Its maximum thickness is 5.37 cm. The model was precisely machined from an aluminum plate using a numerical-controlled machine.

5.2.2 Static Pressure Measurements

The hump model was equipped with 48 center-span static pressure ports. Each pressure port was 0.5 mm in diameter. The locations of the pressure ports are listed in Table 5.1.

A JS4-48 scanivalve was used for all static pressure measurements. The scanivalve was controlled by a CTLR2/S2-S6 solenoid stepper driver. Figure 5.3 shows a flow chart of pressure measurements. Pressure data acquisition were automated and executed using a C program running under Linux on a PC. The sampling frequency for pressure measurements was 1000 Hz. In order to filter out the effect of unsteadiness, the convergency of the data was checked during its acquisition. The scanivalve was stepped to the next pressure port only after the variation in the mean value was within a pre-defined value, $\varepsilon$. The pressure data were written to a file at the end of each acquisition, and the pressure coefficients were calculated from the mean pressure values.
<table>
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<td>7.09</td>
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<td>119.30</td>
<td>126.00</td>
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Figure 5.3. The flow chart of pressure measurements[30].
5.2.3 Plasma Actuator Configurations

Both spanwise and streamwise plasma actuator configurations were investigated in this study. The spanwise plasma actuation is shown in Figure 5.4. The copper electrodes separated by three layers of 5-mil dark colored Kapton film were aligned in spanwise direction. The electrodes were made from 0.0254 mm thick copper foil tape. They were overlapped by a small amount (of the order of 1 mm), in order to ensure a uniform plasma in the full spanwise direction. The junction of the exposed electrode and the covered electrode was placed precisely on the separation location \(x/c = 66\%\).

The streamwise plasma actuator configuration was also investigated. The photograph (top) and schematic illustration (bottom) of the streamwise plasma actuators are shown in Figure 5.5. The covered electrode was positioned at 1.75 in upstream of separation location \(x = 1.75\) in). It ran across almost the whole span of hump model. The width of the covered electrode was 0.5 in. Several spacings between the exposed electrodes were studied. It ranged from 0.5 in to 3.0 in. Three layers of 5-mil dark colored Kapton film was used to separate the exposed electrodes and the covered electrode.

5.2.4 Flow Visualization

Flow visualization was performed with the streamwise plasma actuators on a flat wall. This was done by introducing discrete smoke filaments (streak lines) upstream of the plasma actuators using the smoke-wire technique. The smoke was produced by vaporizing oil from a fine wire (0.002 in) when the wire was resistive heated. Figure 5.6 shows the setup for the flow visualization. The smoke-wire was suspended 0.25 in above the floor of the test section and perpendicular to the flow direction. Plasma actuators were placed on the floor, 2 in downstream of the smoke-wire in order to avoid any arc between the actuators and the smoke-wire. A laser sheet was projected from the...
Figure 5.4. Photograph of a hump model with a spanwise plasma actuator (top) and schematic of a spanwise plasma actuator (bottom).
Figure 5.5. Photograph of a hump model with streamwise plasma actuators and schematic (top) of a streamwise plasma actuator (bottom).
Figure 5.6. Schematic of experimental setup for flow visualization.

side window of the test section to illuminate the smoke streaks. The flow visualization images were obtained using a Panasonic analog video camera through a mirror on the top of the test section. The video tape was digitalized through a video capture card whereby individual frames were extracted.

5.2.5 Boundary Layer Measurements

In this study, the turbulent boundary layer approaching the hump model was measured at $x/c = -1$. The data would be used as the inflow condition for the numerical simulations. The setup is shown in Figure 5.7. A single tungsten hot-wire of 0.00015 in diameter was attached to a traverse system, which was mounted on the top of the test section. A stepper motor was used to drive a linear rail which held the hot-wire probe. The linear rail was aligned perpendicular to the flat plate. The hot-wire was connected to a DANTEC 56C17 anemometer and operated in the constant temperature mode (CTA). The output signal of the anemometer was passed through a low-pass filter
which was set with a cutoff of 1500 Hz. The signal was then sent into a PC and acquired through a PowerDAQ A/C board. The sampling frequency was chosen to be 3000 Hz.

5.3 Experimental Results

All the experiments were conducted at 10 m/s. This gave a chord Reynolds number of 288 K. Figure 5.8 shows the comparison of $C_p$ distributions over the hump model without plasma actuators and with spanwise or streamwise plasma actuators mounted but off. This verifies that the passive plasma actuators has no effect on the $C_p$ distribution.

The surface pressure coefficient distribution reflects the overall features of the flow field. The flow decelerates when approaching the hump’s leading edge. Immediately downstream of the leading edge, the flow accelerates up to mid-chord region due to a favorable pressure gradient. Afterwards, the sharp drop in the $-C_p$ over the hump near $x/c = 0.66$ indicates a flow separation. The flow remains separated over the concave
Figure 5.8. Comparison of $C_p$ distributions on the hump model without plasma actuators and with spanwise or streamwise plasma actuator off.

For the spanwise plasma actuator, the actuator ran in steady operation. The frequency of the a.c. voltage supplied to the electrodes was about 1.3 kHz. The actual frequency was tuned to minimize the overall power in the actuator electronics. The a.c. voltage amplitude to the electrodes was $34\,\text{kV}_{p-p}$. The results are presented in Figure 5.9. The square symbols represent the base flow, and the circle symbols represent the controlled case. The spanwise plasma actuator improved the separation recovery significantly in the separation region.

For the streamwise plasma actuator, it was designed to generate streamwise counter-rotating vortices like passive vortex generators. Flow visualization was used to verify...
Figure 5.9. Surface pressure distribution with spanwise plasma actuator off and on.

the idea. The picture in Figure 5.10 (top) was taken when the laser sheet and smoke sheet were in the same plane. There were two large streamwise vortices generated by the outside edges of plasma actuators. Two streamwise vortices were also generated by the inside edges. Because the inside edges were close, the inside vortices were smaller making them less clear in the picture.

The picture in Figure 5.11 (top) was taken when the laser sheet was raised above the smoke sheet. In this case three long smoke streaks were captured, because the rotating vortices brought the smoke from lower level to higher level where it could be illuminated by the laser sheet[27].

The spacing between the exposed electrodes determines the size of the generated vortices. The issue then was what size of vortex works best for turbulent separation
Figure 5.10. Photograph of visualized flow around the streamwise plasma actuators (top) and schematic of corresponding experimental setup (bottom) - laser sheet and the smoke wire are in the same plane.
Figure 5.11. Photograph of visualized flow around the streamwise plasma actuators (top) and schematic of corresponding experimental setup (bottom) - laser sheet is above the smoke wire.
Figure 5.12. Surface pressure distribution at different spacings.
control? This was investigated by measuring the surface pressure coefficient over the hump as a function of the spanwise spacing of the electrodes. A range of spanwise spacing from 0.5 in to 3.0 in was investigated. The free-stream speed was 10 m/s. The actuator was run in a steady mode. The frequency of the a.c. voltage supplied to the electrodes was 2.1 kHz. The actual frequency was tuned to minimize the overall power in the actuator electronics. The a.c. voltage amplitude to the electrodes was 28 kV<sub>p−p</sub>. The results are presented in Figure 5.12. In all the cases, the streamwise plasma actuators improved the pressure recovery in the separation region, and the reattachment location moved upstream indicating a reduction in the size of separation bubble. These results indicated that the spacing of 1.25 in was the most effective. This optimum spacing was about two times of the incoming turbulent boundary layer thickness.

5.4 Computational Methodology

A segregated solver was used to solve two-dimensional Reynolds-averaged Navier-Stokes equations that included a model for the plasma actuator effects. This was done using Fluent, a CFD modeling software. The \( u \) and \( v \) momentum equations are each solved sequentially in order to update the velocity field. Since the obtained velocities may not satisfy the continuity equation locally, a “Poisson-type” equation for the pressure correction is derived from the continuity equation and the linearized momentum equations. This pressure correction equation is then solved to obtain the necessary corrections to the pressure and velocity fields and the face mass fluxes such that continuity is satisfied. In the segregated solution method each discrete governing equation is linearized implicitly with respect to that equation’s dependent variable. This results in a system of linear equations with one equation for each cell in the domain. A point implicit (Gauss-Seidel) linear equation solver used in conjunction with an algebraic
multigrid (AMG) method to solve the system equations for the dependent variable in each cell. Fluent uses a control-volume-based technique to discretize the governing equations to algebraic equations that can be solved numerically. A second order upwind scheme was chosen for the convection terms and the viscous terms. The SIMPLE algorithm was used for pressure-velocity coupling[23].

The flow configuration and boundary conditions are shown in Figure 5.13. The inlet is located at $x/c = -2.5$ where the free-stream velocity is set to be $U_{ref} = 10$ m/s. The outlet is located at $x/c = 4.0$ where the pressure is set at $p/p_{ref} = 0.99962$. The top-wall is located at $y/c = 0.5627$ with a slight modification which accounts for the
blockage due to the presence of the end plates. Solid wall conditions are applied at the top-wall, floor and hump surfaces. For the spanwise plasma actuator, the computation domain is two dimensional. The plasma actuator was placed at $x/c = 0.66$, near where the flow was predicted to separate. For the streamwise plasma actuators, the computational domain is three dimensional. The spanwise domain size is $z/c = 0.075$, which corresponds to the optimum spacing of streamwise plasma actuator. Periodic boundary condition is applied to both sides of the domain. The plasma actuators were placed at $x/c = 57\%$. The placement of the plasma actuators reflects the experiments.

The computational grid was generated using Gambit, Fluent’s geometry and mesh generation software. For spanwise plasma actuator simulation, the grid is a two dimensional structured grid as shown in Figure 5.14 (left). It has $316 \times 45$ ($x \times y$) grid points. For streamwise plasma actuator simulation, the grid has 40 points in the spanwise direction.

In order to simulate the plasma actuator effect, a body force model was created[45]. The body force per volume of plasma is a vector, given as

$$\vec{F}_B = \left(-\frac{\varepsilon_0}{\lambda_D}\phi\right)\vec{E}$$  \hspace{1cm} (5.1)

where $\varepsilon_0$ is the permittivity of free space ($8.854 \times 10^{-12}$ F/m), $\lambda_D$ is the Debye length, $\phi$ is the electric potential, and $\vec{E}$ is the electric field vector, where

$$\vec{E} = -\nabla\phi$$  \hspace{1cm} (5.2)

The Debye length is the characteristic length for electrostatic shielding in a plasma.
It varies with plasma density and temperature as

\[
\frac{1}{\lambda_D^2} = \frac{e^2 n_o}{\varepsilon_0} \left( \frac{1}{kT_i} + \frac{1}{kT_e} \right) \tag{5.3}
\]

where \(kT_i\) and \(kT_e\) are the ion and electron temperature, \(n_o\) is the plasma density, and \(e\) is the elementary charge.

The electrostatic potential \(\phi\) is given in the following equation, which is derived from one of Maxwell equations.

\[
\nabla (\varepsilon \nabla \phi) = s(x,y) \frac{1}{\lambda_D} \phi \tag{5.4}
\]

where \(s(x,y)\) is a rectangular subdomain in the air over the plasma actuator that is 1 at the intersection of the exposed and covered electrode, and decrease exponentially to 0 at the downstream edge of the covered electrode and 0 over the exposed electrode. A full description of the body force simulations are given by Orlov[44].

The solution of the spatial distribution for the voltage potential is substituted into Equation 5.1 and Equation 5.2 to obtain the spatial body force, \(\vec{F}_B\). This added body force is applied to the computational grid points at the location of the actuator in the computational domain. For the spanwise plasma actuator, the body force vector field is shown in Figure 5.15. For the streamwise plasma actuator, a y-z slice of body force vector field at \(x/c = 60\%\) is shown in Figure 5.16.
Figure 5.15. Steady body force vector field over the hump model for spanwise oriented plasma actuators.

Figure 5.16. Steady body force vector field for streamwise plasma actuators as viewed at $x/c = 60\%$.
5.5 Numerical Results

5.5.1 Spanwise Plasma Actuator Results

The turbulent boundary layer development over the flat plate provided the inflow condition for the simulations. The inflow condition extracted from the simulation at $x/c = -1.0$ and experimental mean velocity profile are cross plotted in Figure 5.17. This shows that the numerical prediction agrees well with the experimental data.

![Figure 5.17. Comparison of the inflow mean velocity profiles for the experiment and the simulation.](image-url)
Figure 5.18. Surface pressure distribution predicted by turbulence models for no-flow-control condition.

A number of turbulence models were used in the numerical simulations. These included $k$-$\varepsilon$ model, SST $k$-$\omega$ model and SA models. A comparison of the results for the different models are shown in Figure 5.18. In the separation region, the $k$-$\varepsilon$ and SST $k$-$\omega$ models agree with the experimental results very well. The SA model predicted a slightly higher pressure than the others. In the reattachment region, all the turbulent models slightly miss the pressure level found in the experiments. Overall, all the turbulence models captured the flow features of the hump model reasonably well. Based on this and given that the $k$-$\varepsilon$ model is easy to implement in the code, $k$-$\varepsilon$ model was used in all of the later simulations.

A comparison of the surface pressure coefficient distribution for the base flow and
the controlled case with the spanwise plasma actuator is shown in Figure 5.19. When
the steady actuator was on, the body force accelerated the ambient air locally, and
causd a local pressure drop. The peak in $-C_p$ at $x/c = 0.66$ is indicative of this effect.
The plasma actuator reduced the strength of the separation, as seen by the larger positive
$C_p$ values in the concave region of the hump. In addition, the overall agreement between
the simulation and experiment was very good.

Streamlines colored by streamwise velocity in Figure 5.20 shows the effectiveness
of the spanwise plasma actuator. The red and blue colors represent high and low veloc-
ity, respectively. Clearly the steady actuation reduced the bubble size significantly, and
the reattachment location as well as the center of circulation moved upstream.
5.5.2 Streamwise Plasma Actuator Results

Since the streamwise plasma actuators generate counter-rotating vortices, the flow over the hump model is no longer two dimensional. In order to simulate this case, the grid for spanwise plasma actuator was extended in the spanwise direction as shown in Figure 5.14. The width of the computational domain was determined by the optimum spacing obtain from the experiments. For the controlled case, the bodyforce was placed in the middle of the domain.

The comparison of surface pressure distribution for base flow and controlled case is shown in Figure 5.21. The peak in the $-C_p$ values at $x/c = 0.66$ that was observed with the spanwise plasma actuator is not present for the streamwise plasma actuators. In this case, the flow is not accelerated in the flow direction but rather streamwise vorticity is created. In the separation region from $x/c = 0.66$ to $x/c = 0.92$, the simulation predicts the static pressure coefficient distribution very well. After the reattachment location,
the simulation tends to slightly under-predict the pressure recovery produced by the plasma actuators.

Figure 5.22 shows the general characteristics of streamwise plasma actuator control. The hump surface is colored based on the shear stress. The red color represents high shear stress. This clearly indicates the location of the streamwise plasma actuator. A slice of the vorticity field at $x/c = 0.65$ shows that a pair of counter-rotating vortices have taken shape. Two streamlines colored by the streamwise velocity magnitude are also shown in Figure 5.20. A closer view of counter-rotating vortices at $x/c = 0.65$ is shown in the vorticity field at the top of Figure 5.23, and in projected velocity vector field at the bottom of Figure 5.23.

The spatial development of the counter-rotating vortices is shown in Figure 5.24
Figure 5.22. Streamlines with streamwise plasma actuator on and a slice of vorticity field at $x/c = 65\%$. 
Figure 5.23. Close view of the vorticity field at $x/c = 65\%$ (top) and velocity field at the same location (bottom).
and 5.25. The streamwise plasma actuator starts at $x/c = 0.57$ and ends at $x/c = 0.62$. Since there is no influence of the plasma actuator at $x/c = 0.5$, the flow field over the hump model is two dimensional. After encountering the plasma actuator, the counter-rotating vortices start to form and subsequently grow stronger as they are convecting downstream. The strength of the vortices reach their maximum right after the actuator at $x/c = 0.62$. The vortices continue to convect downstream and interact with the separated flow at $x/c = 0.67$. Since the vortices redirect the outer flow towards the wall, we observe a large mean flow distortion with an alternate thickening and thinning of the separated region.

Static pressure contours from the simulation are shown in Figure 5.26. There are two streamwise plasma actuators in this figure. The dash lines indicate the position of the covered electrode. After the plasma actuator, there is a semi-circular region dominated by low pressures. In the separation region, the pressure is relatively uniform. In order to confirm this observation, surface pressure coefficients at three locations between two streamwise plasma actuator was extracted and plotted in Figure 5.27. The peaks of $-C_p$ for A and B locations correspond to the semi-circle region in Figure 5.26. From $x/c = 0.67$ to $x/c = 1.0$ the pressure level in the spanwise direction almost doesn’t change.

Surface streamlines and the flow separation pattern for the base flow and the controlled case are shown in Figure 5.28. There are two saddle points on the surface associated with the flow separation. Emanating from the saddle points of separation are lines of separation (red lines), and intersecting the hump surface at these lines are separation sheets which extend into the flow. This separation pattern is quite different from that of the base flow (see Figure 5.28). For the base flow, the separation sheet is two-dimensional, starting from the separation line and ending at the reattachment
line. But for this case, the separation sheets start from the separation lines (red lines in Figure 5.28 (b)) and extend into the flow. Therefore it is three-dimensional.
Figure 5.24: Vorticity fields at different locations with streamwise plasma actuator on, showing the development of counter-rotating vortices.
Figure 5.25: Velocity vector fields at different locations with streamwise plasma actuator on, showing the development of counter-rotating vortices.
Figure 5.26. Surface static pressure contour.
Figure 5.27. Surface pressure coefficient distributions at three locations between two streamwise plasma actuators. $\Delta z$ is the distance between two exposed electrodes.
Figure 5.28. (a) Surface streamlines for the base flow; (b) Surface streamlines for the streamwise plasma actuators showing the effect of the counter-rotating vortices.
CHAPTER 6

CONCLUSIONS & RECOMMENDATIONS

6.1 Conclusions

Plasma actuators have been successfully applied to two-dimensional leading-edge separation control on NACA series airfoils, two-dimensional trailing-edge separation control on a V22 airfoil, three-dimensional leading-edge and trailing-edge ramp separation control on UCAV 1303 model, and for turbulent boundary layer separation control on a NASA hump model. Most of the work has emphasized the experimentation. The last one includes both experiments and numerical simulations. A large effort has been made to develop and optimize a new streamwise plasma actuator configuration designed to generate counter-rotating vortices similar to passive vortex generators.

6.1.1 Leading-edge Separation Control

The experiment examined the use of plasma actuators at the leading edge to control separation, and at the trailing edge, to control lift. At the leading edge, the actuator was operated in both “steady” and “unsteady” manner. The steady actuator was able to reattach the flow for angles of attack up to 19°, which was 4° past the normal stall angle. Even better performance was found with unsteady actuation, which was able to reattach the flow up to a 9° past the normal stall angle. An optimum unsteady actuation
frequency corresponding to $St = f c / U_\infty = 1$ was used in these cases. The power to the actuator in these cases was approximately 2 Watts.

The leading-edge separation control resulted in an increase in both $C_{L_{max}}$ and $\alpha_{stall}$. It resulted in an $L/D$ improvement of as much as 340%.

The trailing edge actuator was located on the surface of one side of the airfoil at $x/c = 0.9$, and spanned most of its width. When operated in a steady manner, it was found to produce the same effect as a plane trailing-edge flap. This included a uniform shift towards higher $C_L$ of the drag bucket. In addition, there were slight decrease in the minimum $C_D$.

A spectral amplitude peak sense-and-control method was investigated based on the frequency and time domain analysis of the pressure data at $x/c = 0.05$. The remarkable feature of this method was that imminent separation could be detected in a repeatable way to signal if the leading-edge plasma actuator should be operated to maintain an attached flow. Then even after the flow has been reattached by the plasma actuator, the approach could determine if the conditions still required the actuator to be on or if the flow would naturally stay attached. Although there are numerous ways of detecting flow separation in the literature, this is the first that we are aware of that can detect while the flow control is on, that the flow control is no longer needed. Thus this can save actuator power.

6.1.2 V22 Wing Separation Control Using Plasma Actuators

The plasma actuators were found to control the separated flow over the wing section and reduce the drag in a hover configuration. At 10 m/s, the flow separated relatively evenly over the leading and trailing edges of the wing section. Each of the actuators operating separately, were able to reattach their respective flow regions. When operated
together, they gave a combined effect that was approximately the sum of their individual effects. Overall with both actuators operating the drag coefficient was lowered by 44%.

At 20 m/s the flow separation over the leading edge was less than that over the trailing-edge flap. A passive modification of the flap gap cover could eliminate the separation over the flap at this free-stream speed. However, active control with the plasma actuators showed the best improvement, with an overall reduction in the drag coefficient of 27%.

6.1.3 Aerodynamic Control of an Unmanned Air Vehicle Using Plasma Actuators

An experimental investigation was conducted to investigate the use of DBD plasma actuators for longitudinal control of a 1303 UCAV planform at high angles of attack with no moving surfaces. Control was implemented at the wing leading edge to provide longitudinal control without the use of hinged control surfaces. The concept employed the use of lee-side and wind-side plasma actuators at the wing leading edge to alter the aerodynamic lift and drag distribution over the 1303 planform by manipulating the flow field over the lee side of the wing section.

Force-balance results showed considerable changes in the lift and drag characteristics of the wing for the plasma-controlled cases compared with the baseline cases. When plotted with effects of conventional trailing-edge devices, the plasma actuators demonstrated a significant improvement in the control authority and, therefore, the operational flight envelope of the wing. Optimum lift enhancement was achieved by placing the actuators at a chordwise location that was close to the leading edge on the suction side at $x/c \approx 0.03$. The actuators were placed parallel to the leading edge and were operated in the unsteady mode. For these, the actuators on the inboard half of the wing was only effective for angles of attack greater than 20°. The actuator on the
outboard half of the wing was, however, effective for angles of attack from $9^\circ$ up to the largest angle examined, $35^\circ$, for which the conventional trailing-edge flaps were ineffective. The results suggests that the application of plasma actuators on a swept UCAV planform can alter the flowfield of the leading-edge vortex in a manner that allows control without the use of hinged control surfaces. The study demonstrated the feasibility of a plasma wing concept for hingeless flight control of air vehicle.

In order to achieve control at low angles of attack comparable to conventional control surfaces, wind tunnel experiments were conducted on the same half wing model with a slightly modified trailing edge incorporating a separation ramp on the suction side. An increase in the lift coefficient at negative angles of attack, from -$15^\circ$ to $0^\circ$, is achieved. Changes in the drag coefficient are negligible. It was realized that the $20^\circ$ separation ramp angle was too aggressive to achieve ideal flow condition in the low positive angle of attack range. The effectiveness was also possibly linked with the degree of the cross-flow velocity component, which made the trailing-edge ramp less effective.

6.1.4 Turbulent Separation Control over a Hump Model

A wall-mounted hump model was selected in this study as a canonical turbulent separated flow field. This shape was chosen because the separation location is insensitive to Reynolds number, and the shape was part of a NASA CFD workshop that provided the baseline flow data.

Both spanwise and streamwise plasma actuator configurations were investigated at a chord Reynolds number of $Re_c = 288\,K$. The surface pressure coefficients demonstrated that both configurations provided a pressure recovery in the separation region, and significantly reduced the size of the separation bubble. For the same conditions,
both configurations achieved the similar performance.

The spanwise plasma actuator can be used in either laminar separation control (unsteady operation) or turbulent separation control (steady operation) depending on the state of the flow field. Such a change in operation is fully electronic without any physical modification of the actuator.

For the streamwise plasma actuator, it can be placed upstream of the separation location unlike the spanwise plasma actuator. Therefore it may be more useful for separated flows in which the separation location changes with flight conditions.

Numerical simulations using Reynolds-averaged Navier-Stokes solver were performed to predict the flow field over the wall-mounted hump model with both spanwise and streamwise plasma actuators on and off. For base flow, SA, $k$-$\varepsilon$, and $k$-$\omega$ turbulence models were used and all of them predicted the surface pressure coefficient in good agreement with the experimental data. $k$-$\varepsilon$ model was chosen for the controlled cases. The plasma actuator effect was simulated through a body force model. For both spanwise actuation and streamwise actuation, computations agreed with experimental results very well. Details of the flow field for the streamwise plasma actuator were also examined through streamlines, vorticity magnitude, velocity vector field, surface static pressure contour, and surface streamlines. For spanwise actuation, the separation type (horseshoe-type separation) did not change, but for streamwise actuation, the separation type changed from horseshoe-type separation to a three-dimensional Werle-type separation.

The ultimate goal of this work is to incorporate a separation region near the trailing-edge of an airfoil so that it can be manipulated by plasma actuators for flight control. The benchmark of the simulation tools for the flow field and plasma actuator effect that was presented for the NASA hump model is a necessary step in the design of such
control-optimized airfoil shapes.

6.2 Recommendations

This work has focused on the application of plasma actuators to control separation on the wing in a manner that is intended to replace the leading-edge slat and trailing-edge flap. Results have showed that plasma actuators successfully reattached the separation flow over a wide range of AoAs after stall on NACA series airfoils, V-22 wing, and UCAV 1303 wing. In order to replace the flap, a separation ramp concept was introduced in this work. Results demonstrated that the plasma actuators were capable of turbulent separation control over the ramp at the aft part of the wall-mounted hump model. As stated earlier, the ultimate goal is to incorporated a separation ramp near the trailing-edge of an airfoil so that it can be manipulated by plasma actuators for flight control. There are further investigations that can be done. Here are some recommendations.

The HSNLF(1)-0213 airfoil is well known as a natural laminar flow airfoil. It has a relatively long favorable pressure gradient region which maintains the flow to be laminar. The concave type pressure recovery is used at the aft part of the airfoil to help lessen the severity of the pressure recovery with respect to separation. But in order to obtain lift enhancement controllability, a separation at the aft part of the airfoil is needed. So the next step is to redesign the airfoil by incorporating a separation ramp at the trailing edge based on the hump model.

For the streamwise plasma actuator, the development of the vortices in the turbulent boundary layer is predicted by the CFD, but the corresponding experiments have not been done yet. PIV measurements can be used to document the instantaneous flow field for further understanding of the mechanism of the plasma actuator. These data also can
be used to improve the modeling of the plasma actuator for even better simulations.

The size (or strength) of the counter-rotating vortices produced by the streamwise plasma actuator is a critical parameter for the separation control. It is a function of the plasma actuator geometry, size, orientation, and location. Systematic investigations of these parameters on vortex characterization are need. On the other hand, the turbulent boundary thickness may have an impact on the effectiveness of the vortices, since for the thicker boundary layer, the vortices may not be strong enough to transfer the high momentum flow to the wall.

Since the body force vector can be tailored through the design of the electrode arrangement, other types of flow, like vectored jets, and co-rotating vortices can be produced by the plasma actuator. It would be interesting to investigate these designs.
APPENDIX A

UDF FOR FLUENT

%This UDF is used to load the bodyfore file and
%pass the value to the source term in N-S equations
%in order to simulate the plasma actuator effects

#include "udf.h"

DEFINE_ON_DEMAND(copy_uds_to_udm)
{
  Domain* d = Get_Domain(1);
  Thread* t;
  cell_t c;

  thread_loop_c(t,d)
  {
    begin_c_loop(c,t)
    {
      C_UDMI(c,t,0) = C_UDSI(c,t,0);
      C_UDMI(c,t,1) = C_UDSI(c,t,1);
    }
  }
}
end_c_loop(c,t);
}

return;
}

DEFINE_SOURCE(source_x,c,t,dS,eqn)
{
real source;
dS[eqn] = 0.0;
source = C_UDMI(c,t,0);
return source;
}

DEFINE_SOURCE(source_y,c,t,dS,eqn)
{
real source;
dS[eqn] = 0.0;
source = C_UDMI(c,t,1);
return source;
}

================================
%Data File Input Format

168
2 (for fluent 6.2)
2 (for 2D)
n (the number of points)
2 (the number of scalar functions)
uds-0 (the name of the scalar function)
uds-1 (the name of the scalar function)

x1
x2
...
xn

y1
y2
...
yn

f1
f2
...
f_n

g1
g2
…

gn
BIBLIOGRAPHY


