Sensing and control of flow separation using plasma actuators

BY THOMAS C. CORKE*, PATRICK O. BOWLES, CHUAN HE AND ERIC H. MATLIS

University of Notre Dame, Institute for Flow Physics and Control, Hessert Laboratory for Aerospace Research, Notre Dame, IN 46556, USA

Single dielectric barrier discharge plasma actuators have been used to control flow separation in a large number of applications. An often used configuration involves spanwise-oriented asymmetric electrodes that are arranged to induce a tangential wall jet in the mean flow direction. For the best effect, the plasma actuator is placed just upstream of where the flow separation will occur. This approach is generally more effective when the plasma actuator is periodically pulsed at a frequency that scales with the streamwise length of the separation zone and the free-stream velocity. The optimum frequency produces two coherent spanwise vortices within the separation zone. It has been recently shown that this periodic pulsing of the plasma actuator could be sensed by a surface pressure sensor only when the boundary layer was about to separate, and therefore could provide a flow separation indicator that could be used for feedback control. The paper demonstrates this approach on an aerofoil that is slowly increasing its angle of attack, and on a sinusoidally pitching aerofoil undergoing dynamic stall. Short-time spectral analysis of time series from a static pressure sensor on the aerofoil is used to determine the separation state that ranges from attached, to imminent separation, to fully separated. A feedback control approach is then proposed, and demonstrated on the aerofoil with the slow angle of attack motion.

Keywords: boundary layer separation; detection; control; plasma actuator

1. Introduction

The control of boundary layer flow separation is important to technologically relevant fluid-based systems. Examples include flow separation control on lift-producing geometries ranging from aircraft wings to gas turbine blades, or in internal flows such as the inter-stage passages of gas turbine compressor and turbine stages. In all these examples, successful boundary layer separation control can significantly improve system performance by minimizing drag or pressure losses. As a result of its importance, there have been numerous boundary layer

*Author for correspondence (tcorke@nd.edu).

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flow separation approaches investigated. These are generally designed to direct high-momentum fluid towards the wall in order to increase the fluid inertia to overcome the local adverse pressure gradient. The approaches can generally be categorized as ‘passive’ and ‘active’. Passive approaches can be defined as ones in which there is no energy input supplied to the flow control device. Examples include boundary layer trips (in the cases where the boundary layer is laminar), wall bumps and streamwise vortex generators. Active approaches can be defined as ones in which energy input is provided to the flow control device. Examples of these include blown wall jets, suction surfaces, moving surfaces and plasma actuators.

Because they require no energy input, passive approaches have been preferred. However, they are not adaptable to changing conditions, and produce a parasitic loss if the flow were naturally attached so that the passive devices were not needed. Active approaches can be turned off when they are not needed. However, they then require a means to determine when they need to be activated. This places an emphasis on measuring the precursors of boundary layer separation and incorporating that information into a feedback control strategy.

It recently has been shown by Patel et al. [1,2] that a pulsing plasma actuator could be used in conjunction with a surface pressure sensor to provide a precursor of flow separation for feedback control. More remarkably when separation control was on, the approach could sense when it was no longer required.

Plasma actuators have proven to be extremely effective in boundary layer separation control. This has been demonstrated on the leading edges of fixed wing sections by Corke et al. [3,4] and Post & Corke [5–7], on pitching wing sections undergoing dynamic stall by Post [8] and Post & Corke [5–7], on turbine blades in a linear cascade by Huang [9], Huang et al. [10,11], List et al. [12], Suzen et al. [13], Rizzetta & Visbal [14] and River [15], and on bluff body wake control by Thomas et al. [16,17].

The most common configuration of a plasma actuator for separation control is one with an asymmetric electrode arrangement shown in figure 1a. It consists of two electrodes that are separated by a dielectric layer. The top electrode is
exposed to the air. The bottom electrode is fully encapsulated by the dielectric material. The actuator has the electronic characteristics of a capacitor, and is therefore driven by an AC power source. The AC amplitude is set to be large enough to cause the air to ionize. This occurs over the surface of the dielectric covering the bottom electrode. This is illustrated by the blue region in the actuator schematic.

The effect of the plasma actuator on the neutral flow is through a vector body force. An example of the body force vector field during a portion of the actuator AC cycle corresponding to \( t/T = 0.76 \), where \( T \) is the AC period, is shown in figure 1b. The thick horizontal lines represent the electrodes. This is based on the single dielectric barrier discharge (SDBD) plasma actuator simulations by Orlov [18] and Mertz [19]. The larger magnitude body force vectors that occur near the edge of the exposed electrode are angled in the direction towards the covered electrode. These induce the neutral air to jet along the surface of the dielectric, away from the exposed electrode (from left to right in figure 1). The thrust produced by the tangential jet has been used as a metric of the performance of plasma actuators. Another general feature of the body force distribution is the vectors that are nearly normal to the dielectric surface. This might be considered to produce an effect like suction at the wall, although the no-penetration boundary condition holds. However, it is a strong effect that numerous flow visualization images have documented by the turning of fluid stream tubes towards the surface of the dielectric layer (see [20]). A general review of DBD plasma actuators and their use in aerodynamic flow control is given by Corke et al. [21,22].

An example of a plasma actuator located on the leading edge of an aerofoil to control the leading edge flow separation is shown in figure 2a. The actuator is designed so that the bottom electrode is on the suction side of the aerofoil while at a positive angle of attack. This electrode is covered by a dielectric film (0.15 mm thick Kapton). The top (exposed) electrode is placed so that the downstream edge is on the aerofoil leading edge \((x/C = 0)\). Figure 2b,c shows images of the visualized flow over the aerofoil while at a post-stall angle of attack, \( \alpha \), of 16°. At this angle of attack, the boundary layer separates from the leading edge and does not re-attach on the aerofoil. This is demonstrated in the flow visualization image of figure 2b. Figure 2c shows the visualized flow with the plasma actuator operating. This illustrates that the boundary layer was forced to attach. Further details on this are given by Post & Corke [6,7].

Figure 2. Photograph of a plasma actuator on the leading edge of a NACA 0015 aerofoil section (a) and flow visualization records for the aerofoil at a post-stall angle of attack with the plasma actuator turned off (b) and on (c).
Figure 3. Time sequence of flow visualization images for the NACA 0015 aerofoil at a post-stall angle of attack that shows the initial response of the separated flow to a leading-edge plasma actuator pulsing at a frequency corresponding to $St = 1$. $tU_\infty/C = 1.0, 4.9$ and $5.9$ from (a) to (c).

It has been shown in the literature, for example, by Seifert et al. [23] and Greenblatt & Wygnanski [24], that the introduction of periodic disturbances near the separation location can prevent or delay the onset of flow separation. The optimum forcing frequency, $f$, is generally found to occur at a Strouhal number, $St = fL_{sep}/U_\infty$, near unity. Here $L_{sep}$ is the streamwise extent of the separated flow region. An example of the initial response of the separated flow over the NACA 0015 aerofoil with the leading-edge plasma actuator is shown in Figure 3. For this, the plasma actuator was pulsed on and off at a frequency corresponding to $St = fC/U_\infty = 1$, where $C$ is the aerofoil chord length.

The flow visualization images show the temporal development of spanwise vortical structures that are produced by the pulsing actuator. The optimum Strouhal number results in approximately two vortical structures at a given time in the separated flow region. These cause the flow at the leading edge to first attach with the separation point moving progressively downstream until the flow becomes fully attached everywhere. The final state was shown in the right-most flow visualization image in Figure 2. The dimensionless time from the start of the pulsing in the images is $tU_\infty/C = 1.0, 4.9$ and $5.9$ from left to right.

Even with the flow fully attached, the flow visualization indicates that the attached boundary layer is still responding to the pulsing plasma actuator. This will be important to the separation detection and the closed-loop control approach that is discussed in §3.

2. Experimental set-up

Two different experimental set-ups were used in the analysis of boundary layer separation detection and control on aerofoils. One was used to measure lift and drag on slowly pitching aerofoils at different angles of attack. A schematic of that set-up is shown in Figure 4a. It consisted of a lift–drag force balance that was located on the top of a wind tunnel test section. The wind tunnel was an open-return, draw-down design with a 0.421 m square by 1.8 m long test section. The aerofoil was mounted on the support sting of the force balance. It 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 aerofoil. A hole in the ceiling end-plate accommodated the sting supporting the aerofoil. A hole in the floor end-plate allowed access for the plasma actuator wiring. This hole was aligned with the support sting so that it would
Review. Sensing and control of flow separation

not interfere with angular positioning of the aerofoil 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. Further details are provided by He [25] and He et al. [26].

For the slowly pitching aerofoil, the chord length was 12.7 cm and the span was 30.48 cm. The size of the aerofoil 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 to minimize stall hysteresis in the speed range for the facility.

The angle of attack was steadily increasing and decreasing between 0° and 23° at a rate that was slow enough so that there was no lift-cycle hysteresis. At the largest angle of attack of 23°, the solid blockage was 8.5 per cent, which still ultimately required correction for the blockage in the measured lift and drag coefficients. The experiments were performed at two free-stream velocities of 21 and 30 m s⁻¹. For these, the blockage-corrected chord Reynolds numbers were 0.217 × 10⁶ and 0.307 × 10⁶.

The other experimental set-up was used to study dynamic stall on a periodically pitching aerofoil. The wind tunnel for this set-up was also an open-return, draw-down design, but it had a larger, 0.607 m square test section, which was 1.8 m long. The aerofoil was pitched about its quarter-chord location with a computer-controlled servo motor. The servo motor was mounted outside of the test section. The motor shaft was connected to the aerofoil through a hollow shaft that passed through a side wall of the test section. A ball bearing in the side wall helped support the aerofoil. A rotary position encoder in the servo motor provided instantaneous information on the aerofoil angle of attack. A photograph of the servo motor mounted outside of the test section, and a photograph of the aerofoil model is shown in figure 4b.

Figure 4. Schematic of set-up for slowly pitching aerofoil lift and drag measurements (a) and photographs of computer-controlled servo motor and aerofoil used in periodic pitching dynamic stall experiments (b).
The dimensions of the aerofoil used in the pitching experiments were the same as those of the aerofoil used in the stationary angle of attack experiments. However, the pitching aerofoil had 29 static pressure ports for measuring the pressure distribution. The pressure ports were aligned with the mean flow direction at the centre-span location. The inner diameter of the pressure ports was 0.5 mm. The pressure ports were spaced relatively uniformly around the aerofoil, with a slight decrease in their spacing near the leading edge where the pressure gradient was the largest. Tubulations from the pressure ports passed through the hollow support shaft and then attached to a scanning pressure valve that selectively connected each static port to a pressure transducer. The dynamic response of the total pressure measurement system was determined experimentally to be flat to 100 Hz, which was 25 times larger than the highest pitching frequency used.

The aerofoils used in both experiments had a fast dynamic response pressure transducer that was located inside the aerofoil. One side of the pressure transducer was connected to a static pressure port, which was located at $x/C = 0.05$ on the suction side of the aerofoil. The reference side of the pressure transducer was open to the free-stream static pressure. The frequency response of that system was 500 Hz.

### 3. Boundary layer separation detection

The approach to detecting a precursor to boundary layer separation is based on the receptivity of the flow to unsteady disturbances. Haddad et al. [27] had found, in the analysis of acoustic receptivity of a boundary layer over a parabolic leading edge, that there was a dramatic growth ($\sim 100$ times) in the receptivity coefficient just prior to the formation of a flow separation bubble. Thus, the boundary layer was significantly more responsive to the acoustic pressure disturbances when the flow was on the verge of separation.

Our approach to exploit this as a stall precursor detector was to place a static pressure port near the location where the boundary layer would first separate. For the NACA 0015 aerofoil, this was near the leading edge, at $x/C = 0.05$. Figure 5 shows the location of the pressure port on the aerofoil relative to flow visualization that reveals the initial flow separation bubble that forms at a large pre-stall angle of attack. The pressure port was connected to a pressure transducer that was located inside the aerofoil. The dynamic response of the pressure transducer was fast enough to be able to detect the frequency of the pulsing plasma actuator that was optimal to re-attach the boundary layer ($St = 1$).

As previously discussed in §1, a plasma actuator at the leading edge of the aerofoil could be pulsed at an optimum Strouhal number to maintain an attached flow. Figure 6 shows plots of the power spectral density of pressure time series on the surface of the aerofoil at $x/C = 0.05$ for two fixed angles of attack of 6° and 14°. For the free-stream conditions, the actuator pulsing frequency for $St = 1$ was 80 Hz. The stall angle of attack of the aerofoil is 15°. At stall, the flow over the aerofoil becomes fully separated, and produces a dramatic drop in lift and increase in drag. Both the spectra shown in figure 6 are at pre-stall angles of attack.

At each angle of attack, pressure spectra are shown with the plasma actuator pulsing on and off at 80 Hz. At the 6° angle of attack, the pressure spectra with the plasma actuator on or off are indistinguishable from each other. Therefore,
any disturbances introduced by the plasma actuator at the leading edge are not sensed as static pressure fluctuations at $x/C = 0.05$. However, when the angle of attack was increased to $14^\circ$, one degree before stall, spectral peaks at the plasma actuator pulsing frequency and its harmonics are clearly visible. Flow visualization like that in figure 5 correlated the appearance of the spectral peaks with the formation of a small separation bubble that extended over the location of the pressure port.

In this instance near the leading edge of the aerofoil, the magnitude of the spectral peak detected in the static pressure fluctuations increased with the angle of attack. This is shown in figure 7. The angles of attack have been correlated to the lift distribution and flow visualization to identify three regions corresponding to ‘no flow separation’, imminent flow separation’ and ‘fully separated flow’. The increase in the spectral peak is consistent with the increase in the receptivity of a boundary layer to pressure disturbances as a result of a flow separation that was found by Haddad et al. [27]. The ability to capture the precursor to boundary layer separation on the aerofoil requires that the detection location be as close as possible to the leading edge.

(a) Feedback control

The feedback control schematic is shown in figure 8a. Following some time interval schedule, the leading-edge plasma actuator is turned on to pulse at $St = 1$, which is optimum for maintaining the attached flow. The voltage level to the actuator would be low and only sufficient to produce a pressure disturbance.
Figure 7. Pressure spectral amplitude correlation with leading edge separation condition on a NACA 0015 aerofoil at different angles of attack.

Figure 8. Schematic for feedback flow separation control based on plasma actuator $St = 1$ pulsing (a) and lift coefficient versus angle of attack comparison with feedback control off and on (b). Adapted from He [25]. (b) Triangles, feedback control; squares, baseline.
This is referred to as the ‘sensing’ level. Following this, the time series from the pressure sensor is acquired. A spectral analysis is performed on the time series to determine if there is a spectral peak at $St = 1$. If one is detected, the flow is separated or about to separate, and therefore the plasma actuator voltage is increased to a ‘control’ level that is sufficient to re-attach the flow or prevent it from separating. If a spectral peak at $St = 1$ is not detected, the flow is not separated or about to separate, and the plasma actuator is maintained at the sensing level of operation.

The feedback control approach was applied to the slowly pitching NACA 0015 aerofoil. The result of this is shown through the plot of the lift coefficient versus the angle of attack in figure 8b. The pitching motion is slow enough so that there are no dynamic effects. The free-stream velocity was $21 \text{ m s}^{-1}$, and the actuator pulsing frequency was $180 \text{ Hz} \ (St = 1)$. The results for the baseline aerofoil are shown by the square symbols. This shows a range of angles of attack where the lift versus angle of attack follows linear thin aerofoil theory, without dynamic stall, with $dC_L/d\alpha = 0.11 \text{ deg}^{-1}$. This is followed by a nonlinear lift change that is the precursor to a dramatic drop in the lift (stall) at $\alpha = 15^\circ$. With the feedback control operating, imminent leading-edge separation was detected at $\alpha = 14^\circ$, and the leading-edge plasma actuator voltage was increased to the control level. The ranges of angles of attack where the feedback control determined that the plasma actuator remain at the control level are denoted by the bold arrows labelled ‘sense’ and ‘control’. With the plasma actuator at the control level and pulsing at $St = 1$, the stall angle of attack increased from $15^\circ$ to $17^\circ$, and the maximum lift coefficient increased from 1.28 to 1.40. The exact improvement will depend on the body force generated by the plasma actuator, which is a function of the actuator design, dielectric properties and the AC voltage level, as discussed by Corke et al. [21,22]. The plasma actuator in this example also had the benefit of producing a ‘soft’ stall, where the lift remained near the maximum level for angles of attack well above the stall. In this range compared with the aerofoil without feedback control, the lift-to-drag ratio was 1.6 times higher than the control as given by He [25] and He et al. [26].

(b) Application to dynamic stall

This approach for boundary layer separation detection clearly works in the transition from attached to separated flow in the simple case of a slowly pitching aerofoil. However, there is interest in applying this to the problem of a fast pitching aerofoil undergoing dynamic stall that is relevant to helicopter rotor aerodynamics. For this, the NACA 0015 aerofoil was pitched about its quarter-chord location with a computer-controlled servo motor. The motor was programmed to produce a time-dependent angle of attack, $\alpha(t) = \alpha_{\text{mean}} + \alpha_{\text{amp}} \sin(2\pi ft)$.

Short-time spectral analysis was performed on the time series output from the pressure sensor that was monitoring the unsteady static pressure at $x/C = 0.05$. This was performed using a Gabor short-time Fourier transform that translates an analysis window, or ‘windowing function’, along the time series to capture temporal changes in the spectra. The Gabor transform is defined as

$$G(\tau, \omega) = \int_{-\infty}^{\infty} F(t)g(t - \tau)e^{-i\omega t} \, dt,$$

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Figure 9. Short-time spectral analysis of the pressure time series measured at \( x/C = 0.05 \) on the aerofoil pitching with a mean angle of attack of \( 10^\circ \), and a pitching amplitude of \( 10^\circ \) that is producing light dynamic stall conditions. The plot at the top shows the corresponding angle of attack and pressure time series. Dashed line, \( \alpha \); solid line, pressure. (Online version in colour.)

where \( G \) is the time–frequency coefficient matrix of the mapped time series, \( F(t) \), \( g \) is the user-defined windowing function, which in this case was a Hanning function, \( \tau \) is the time marching increment determined by the size (length) of \( g \) and \( \omega \) represents frequency in rad s\(^{-1}\). The discrete implementation is defined

\[
G(m, \omega) = \sum_{n=-\infty}^{\infty} F[n]g[n-m]e^{-i\omega n},
\]

where \( n \) designates the index of the signal associated with \( t \) and, likewise, \( m \) the index associated with \( \tau \). The squared norm of \( G \), \( |G|^2 \), is the spectrogram. \( G \) restricts the Fourier transform of \( F \) to the vicinity of \( t = b \) by computing the Fourier transform of the product of \( F \) and \( g \), where \( g \) must meet several criteria given by Chui [28].

In the spectral analysis, the frequency resolution was 1 Hz. The maximum resolvable frequency, set by the anti-alias filter, was 120 Hz. This was 30 times the aerofoil pitching frequency, and 1.5 times the plasma actuator pulsing frequency which was 80 Hz. For the analysis, the width of the spectral window corresponded to 24 per cent of the aerofoil pitching period. This would make frequencies below approximately 12 Hz difficult to resolve with the spectrogram. Our interest however was in the 80 Hz frequency, which we could easily resolve with the spectrogram.

Data were taken for different combinations of \( \alpha_{\text{mean}} \) and \( \alpha_{\text{amp}} \). Two examples of the short-time spectral analysis are shown in figures 9 and 10. The first is an example of ‘light’ dynamic stall, with a mean angle of attack of \( 10^\circ \) and a pitching amplitude of \( 10^\circ \). The second is an example of ‘deep’ dynamic stall with a mean angle of attack of \( 15^\circ \) and a pitching amplitude of \( 15^\circ \).
Figure 10. Short-time spectral analysis of the pressure time series measured at $x/C = 0.05$ on the aerofoil pitching with a mean angle of attack of 15°, and a pitching amplitude of 10° that is producing deep dynamic stall conditions. The plot at the top shows the corresponding mean removed angle of attack and pressure time series. Dashed line, $\alpha$; solid line, pressure. (Online version in colour.)

angle of attack of 15° that is beyond the static stall angle of attack, and a pitching amplitude of 10°. The pitching frequency in both cases had a reduced frequency, $k = \pi f C / U_\infty = 0.16$.

The magnitudes of the short-time spectra are presented as contours of constant levels. The horizontal axis is time (ms). The period of the aerofoil pitching cycle is 250 ms ($f = 4$ Hz). The vertical axis is frequency. For reference, a horizontal dashed line has been drawn at 80 Hz. This corresponds to the leading edge plasma actuator pulsing frequency at which, in this case, $St = 1$.

Included with the short-time spectra are the corresponding time series of the aerofoil angle of attack, $\alpha(t)$, and the voltage proportional to the mean removed static pressure, $P(t)$, that was measured on the aerofoil at $x/C = 0.05$.

For the case with light dynamic stall shown in figure 9, it is best to first focus on the pressure time series. The pressure measurement location is on the suction side of the aerofoil. The static pressure is therefore a positive maximum at the minimum angle of attack in the pitching cycle, and a negative maximum at the maximum angle of attack.

Throughout the pitching cycle, the leading edge plasma actuator was pulsing at the lower sense amplitude level, at a frequency of 80 Hz. During the pitch-up portion of the cycle where the angles of attack are lower, the pressure time series is smooth. In this portion of the pitching cycle, flow visualization indicated that the boundary layer over the aerofoil was fully attached.

In contrast to this, during pitch up when the aerofoil reaches larger angles of attack, the pressure time series exhibits high-frequency fluctuations. These fluctuations remain until $\alpha \simeq 5^\circ$ during pitch down. The short-time spectral analysis indicates that the pressure fluctuations are centred at a frequency of
80 Hz, and cover from $\alpha \simeq 16^\circ$ during pitch up to $\alpha \simeq 5^\circ$ during pitch down. The detection of the leading-edge actuator pulsing frequency signifies that the boundary layer has separated during that portion of the pitching cycle. This was confirmed by flow visualization. The short-time spectra clearly show the boundary layer separation and re-attachment that repeats from one cycle to the next of the periodically pitching aerofoil.

The short-time spectral analysis for the conditions of deep dynamic stall is shown in figure 10. Deep stall is characterized by a massive flow separation and the formation and propagation of a leading-edge vortex over the suction side of the aerofoil. Under this condition, there is a noticeable phase lag in the pressure fluctuation cycle with respect to the pitch angle. In contrast to the case with light dynamic stall, with the deep stall there is an initial sharp peak in the pressure time series that is caused by the formation and passage of the leading-edge dynamic stall vortex.

The higher frequency fluctuations are apparent on the pressure time series during the higher angle of attack portion of pitch up in the pitching cycle, which is similar to the light dynamic stall case. The short-time spectral analysis again reveals that these fluctuations are at the 80 Hz pulsing frequency of the leading-edge plasma actuator. This again clearly shows the repeated flow separation and re-attachment that occurs over the pitching aerofoil.

We observe that cycle-to-cycle variations in the short-time spectra around the 80 Hz pulsing frequency are more apparent under the deep stall conditions. This is somewhat expected and reflects the increased flow unsteadiness caused by the dynamic vortex formation and convection over the aerofoil. The delineation of flow states defined in figure 7 provides a margin of uncertainty that can account for such variability. However, further experiments should be performed to determine how factors such as Reynolds number might affect the state map.

A sequence of flow visualization still images taken at different angles of attack during the pitching cycle for the conditions of deep dynamic stall are shown in figure 11. These images can be used to correlate the boundary layer conditions to the spectrogram in figure 10. The pitch angles are indicated on each image. The visualization images begin at $\alpha = 5^\circ$, which is the lowest angle of attack in the cycle.

At the lowest angle of attack in the cycle, the flow visualization indicates that the boundary layer is fully attached. Based on the flow visualization, the boundary layer remains attached during pitch up to approximately the first half of the pitch-up cycle, or $\alpha \simeq 15^\circ$. This is consistent with the spectrogram that first shows energy in the 80 Hz band at about the middle of the pitch-up portion of the cycle, $\alpha = 15^\circ$.

The boundary layer remains separated from the leading edge for the remainder of the pitch-up portion of the cycle and into the pitch-down portion of the cycle to $\alpha \simeq 9^\circ$. This agrees well with the spectrogram that shows energy in the 80 Hz band extending to approximately the middle of the pitch-down portion of the cycle to $\alpha = 9^\circ$. The attached boundary layer progresses towards the trailing edge so that the flow over the aerofoil is fully attached at the start of the next pitching cycle.

Figure 12 shows a similar sequence of flow visualization still images for the conditions in figure 11, except with the leading edge plasma actuator pulsing at $St = 1$ (80 Hz) with a control amplitude that was sufficient to influence the boundary layer separation. In this case, the plasma actuator is pulsing throughout...
the cycle, never turning off in an example of open-loop control. In a feedback control loop using the spectrogram analysis, the actuator would switch to the lower voltage sense condition during the first half of the pitch-up portion and last half of the pitch-down portion of the cycle. Such feedback control is in the process of being implemented.

The open-loop control provides a reference between no control and closed-loop control conditions on which the closed-loop control will eventually be judged. Comparing the flow visualization in figures 11 and 12, both start the pitching

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**Figure 11.** Flow visualization sequence of periodically pitching aerofoil with a mean angle of attack of 15°, and a pitching amplitude of 10° that is producing deep dynamic stall conditions.
cycle from the same initial state where the boundary layer is fully attached on the suction surface of the aerofoil. Without control, a flow separation bubble is then clearly evident at $\alpha = 19^\circ$. With control, this is delayed to $\alpha \simeq 22^\circ$, at which a small separation bubble is first forming.

At the three highest angles of attack during pitch up without control, the flow over the suction surface of the aerofoil is massively separated. In contrast, with control, the flow visualization at $\alpha = 25^\circ$ shows a closed leading edge separation.
bubble. This separation bubble eventually grows and convects over the full suction surface of the aerofoil in the pitch-down portion of the cycle; however, it is never as large as in the uncontrolled case.

During pitch down, the plasma actuator re-attached the boundary layer at the leading edge within a couple of degrees of the maximum angle of attack. In the uncontrolled case, this did not occur until more than half of the pitch-down portion was completed \( (\alpha \approx 13^\circ) \), at which the flow over three-quarters of the suction surface is still separated. At the same angle of attack in the controlled case, more than half of the suction surface of the aerofoil is attached.

Finally, with the controlled case, the boundary layer on the suction surface of the aerofoil during pitch down is fully attached by \( \alpha = 8^\circ \). The vortical structures in the flow visualization image are being driven by the pulsing plasma actuator. Based on the spectrogram, the boundary layer would be naturally attached so that the actuator is no longer needed at this point in the cycle. With closed-loop control, the voltage to the plasma actuator would be lowered to the sense state during this portion of pitch down, and would remain low until approximately the second half of the pitch-up portion of the cycle, where imminent boundary layer separation would be sensed.

Overall, the flow visualization shows clear differences between no control and open-loop control. A quantitative comparison with regards to the cycle lift and pitch moment was performed by Post [8] and Post & Corke [29]. Figure 13 shows a comparison of the cycle lift distribution for the same conditions as the flow visualization in figures 11 and 12. This indicates that the open-loop control
increased the lift in the parts of the cycle corresponding to the first half of pitch up and the last half of pitch down, where the largest differences in the visualized flow between no control and open-loop control were observed. This resulted in a 5 per cent improvement in the cycle-integrated lift as given by Post [8] and Post & Corke [29].

The open-loop control was found to suppress the dynamic stall vortex formation during pitch up near the maximum angle of attack. As a result, the open-loop control reduced the lift in that portion of the cycle. However, the cycle-integrated lift did increase. Post [8] implemented a quasi-closed-loop control that turned the plasma actuator off and on according to a pitching cycle schedule that was based on flow visualizations like those in figure 12. This produced the same actuator schedule that would have independently resulted from the spectrogram in figure 10, with the exception of lowering the power to the actuator near the maximum angle of attack to allow the formation of the dynamic stall vortex. This produced a 13 per cent increase in the cycle-integrated lift, or an almost threefold improvement over the open-loop control. Details on this can be obtained from Post [8] or Post & Corke [29].

This approach was limited to the specific conditions (reduced frequency, pitching amplitudes, etc.) by which the boundary layer conditions over the aerofoil were derived. It used the angular position encoder for feedback rather than an in situ detection of the boundary layer separation state that this paper promotes. Nevertheless, it provides valuable information on the viability and benefits of closed-loop control of the dynamic stall process associated with the unsteady aerodynamics of a pitching aerofoil.

4. Conclusions

A boundary layer separation detection approach that used a plasma actuator designed for boundary layer separation control was presented. The approach is based on the heightened receptivity of a separated boundary layer to local pressure disturbances. For separation detection, these disturbances were produced by periodically pulsing the plasma actuator. The detection of these pulses in fluctuations in the downstream wall static pressure perfectly correlated with the separation state of the boundary layer. This was demonstrated on a slowly pitching aerofoil and on a periodic pitching aerofoil undergoing dynamic stall. If the plasma actuator pulsing was at the optimum frequency to attach the flow, previously shown to be at $St = 1$, then the combination of separation detection and control was ideal for closed-loop control.

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