Open-Loop Flow Control At Low Reynolds Numbers Using Periodic Airfoil Morphing

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This paper investigates the application of a periodically deforming airfoil surface for the purpose of flow control at low Reynolds numbers. A physical model has been fabricated by bonding Macro Fiber Composite actuators to the underside of an airfoil's suction surface. This model is actuated using a high voltage amplifier and has been tested in a closed-loop wind tunnel at $Re_c = 5 \times 10^4$. It was found that at high enough actuation frequencies such a control technique reduces drag and simultaneously increases lift – thus achieving significant improvements in performance in a flight regime notorious for poor airfoil behavior. Furthermore, by delaying the onset of stall, actuation was able to increase the maximum lift achievable by this airfoil section at $Re_c = 5 \times 10^4$ which can of benefit to small aircraft at take-off and landing where high lift coefficients are required.

Nomenclature

x	Streamwise coordinate of the wind tunnel
z	Spanwise coordinate of the wind tunnel
y	Vertical coordinate of the wind tunnel
c	Model chord
s	Model span
α	Angle of Attack
U_∞	Freestream velocity
Re_c	Reynolds number based on the chord
D	Drag force
L	Lift force
C_D	Drag coefficient
C_L	Lift coefficient
V_A	Amplitude of sinusoidal actuation
V_f	Frequency of sinusoidal actuation

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I. Introduction

BOTH military and civilian interest in Unmanned Aerial Vehicles (UAV) has increased significantly over recent years with real and proposed applications ranging from surveillance and telecommunications to precision delivery services and even a potential system for exploring Mars.¹ Also, the availability of very small sensors, video cameras, listening devices and control hardware has meant Micro Air Vehicles (MAV) capable of complex missions are currently the subject of much research and development to meet projected military and environmental needs.² A combination of small geometric dimensions, low speeds or high altitudes (low densities) results in these aircraft operating at low chord-based Reynolds numbers, Re_c — a flight regime notoriously problematic for conventional airfoil geometries.

At sufficiently low Reynolds numbers — which Lissaman's³ classical review suggests can be anything below $Re_c = 500,000$ — laminar boundary layers forming on an airfoil's upper surface will persist beyond the suction peak and into the pressure recovery region whereupon an adverse pressure gradient will be encountered. Viscous effects close to the airfoil's surface slow a fluid element down thereby reducing its kinetic energy. Unlike turbulent boundary layers — which can compensate for this by mixing the low momentum fluid with higher momentum fluid at the edge of the boundary layer — a laminar boundary layer has no mechanism for re-energizing the near-wall flow, leaving them incapable of overcoming even modest adverse pressure gradients. As a result, when present on an airfoil, laminar boundary layers are prone to separation even at low angles of attack and consequently low lift and high drag coefficients are inherent in flight at low Re_c . Flow control techniques could counter such unfavorable conditions and potentially lead to considerable performance improvements.

Passive flow control techniques, which are more easily implemented than active control, have proven to be very effective in delaying separation by increasing the near-wall momentum, primarily via boundary layer tripping or vortex generation. Mueller & Batil⁴ for example found that the C_L -vs- α slope of a NACA 663-018 airfoil at $Re_c = 40,000$ was improved when surface roughness elements were placed at the leading edge. However, experiments performed by Greenblatt & Wygnanski⁵ on a different airfoil at $Re_c = 50,000$ found that 6 mm diameter $\times$ 1 mm high circular tabs were ineffective as boundary layer trips at such low Reynolds numbers resulting in little to no improvement when compared with their clean airfoil data. This suggests that passive control at very low Reynolds numbers can be difficult to implement.

Furthermore, such methods often deteriorate aerodynamic performance for flow conditions for which they were not designed.⁶ Indeed, the aforementioned study by Mueller & Batil⁴ found that at higher Reynolds numbers, where laminar flow existed over a shorter portion of the surface, surface roughness simply increased drag.

In contrast, active control approaches offer the significant advantage of being largely innocuous except when activated and thus removes the drawback associated with passive control at off-design conditions. Also, the possibility of coupling the control input to flow instabilities associated with separation and transition could enable substantial control authority over a wide range of flow conditions.⁷

The drawback of active control, with respect to passive, is that an energy input is required. A primary concern when employing such methods is that the amount of energy input is more than offset by the energy saved. This imposes practical constraints on control methods and actuation systems. Steady suction and blowing techniques were among the first to be investigated and proved very effective at enhancing airfoil performance. However, they require heavy, complex pneumatic systems and a relatively large amount of power to achieve significant aerodynamic benefits. By exploiting the belief that Large Coherent Structures (LCS) were responsible for transporting momentum across a shear layer and controllable using periodic motion Seifert et al.,⁸ discovered that by superimposing a periodic motion on top of steady blowing, the efficiency of such control methods could be greatly increased.

More recently, the development of new actuation devices and material systems has enabled novel approaches to flow control to be explored and the concept of periodic motion has been further investigated with methods including pulsed vortex generators,⁹ synthetic jets⁷ and oscillating surfaces.¹⁰ Munday et al.¹⁰ used a thin, flexible piezoelectric THUNDER actuator, developed at NASA, to morph the surface of an airfoil. When embedded in a surface or attached to flexible structures such actuators provide a distributed force with little power consumption. They are also very light and easy to integrate to the surface of an airfoil thus maximizing their possible aerodynamic gains. Munday et al. performed both static and dynamic morphing tests at angles of attack from 0° to 9° at Re_c of 2.5×10^4 and 5×10^4 . While their static tests did not prove very successful, dynamic actuation was found to significantly reduce flow separation.

Macro fiber composite (MFC) actuators, also developed at NASA, are a more advanced piezoelectric

actuator, similar to THUNDER. It consists of three main components: 1) a sheet of aligned piezoceramic fibers, 2) a pair of thin polymer films etched with a conductive electrode pattern and 3) an adhesive matrix material.¹¹ The piezoceramic fibers provide a coupling between mechanical and electrical fields so that when they experience an electrical field provided by the electrode pattern, mechanical deformations of the fibers will occur. As a result, MFCs can provide distributed deflection for very little power consumption.

The scope of this paper is to explore the use of MFCs for dynamic actuation of an airfoil surface. Their ability to perform the dynamic motion will be presented along with the effect on aerodynamic performance that such motion has at low Reynolds numbers. Throughout this paper a Reynolds number of 5×10^4 was investigated.

II. Experimental Setup

II.A. Fabrication

An airfoil model based on a NACA 4415 has been fabricated for investigating the effect of dynamic surface morphing by MFC actuators. The leading edge, trailing edge and pressure surface were machined from PVC to true NACA 4415 coordinates with a tolerance of ± 0.1 mm, while the suction surface was fabricated separately to accommodate the desired deformations. To achieve a deformable surface, two MFC actuator patches (Smart Material M-8557-P1) were bonded to a 0.25 mm thick sheet of Titanium with a slow drying epoxy resin. A vacuum bag was placed over the skin while the epoxy set to remove any pockets of air between the Titanium and MFCs, ensuring a strong, clean bond. The resulting skin can be seen in figure 1a. The fully formed model has a chord, c , of 150 mm and span, s , of 158 mm and can be seen in figure 1b.

Figure 1: Airfoil model for testing: a) inner side of a titanium skin with MFC patches; b) assembled model.

The skin was rigidly attached to the body of the aerofoil at the leading edge. At the trailing edge a thin slide joint was used to allow for small displacements in the longitudinal direction that occurred when the surface was deformed.¹² In slight variation to previous models presented by Debiase et al.^{12–14} which were mainly meant for static deformations, the slide on this model was located closer to the trailing edge offering a larger deformable area and therefore increasing the dynamic range.

Figure 2 shows a comparison between true NACA 4415 coordinates and the non-actuated airfoil geometry. The leading edge of the model has a slightly larger radius than the reference geometry and the suction surface appears slightly flatter, but on the whole the model appears to be representative of a true NACA 4415.

Figure 2: Comparison between the physical model and NACA 4415 coordinates

A voltage is applied to the MFCs via a Smart Material HVA 1500/50-2 high-voltage amplifier which has a gain of 200 V/V and can accept input signals from DC to 10 kHz in the range of -2.5 V to 7.5 V. These values are amplified to -500 V and 1500 V respectively and correspond to the operational range of the actuators. Applying a negative voltage to the MFC will cause an outward deformation and a positive voltage will deform the skin inward. The maximum achievable displacement of the skin in the outward direction occurs when the actuators are driven with a voltage of -500 V and the maximum inward displacement occurs at 1500 V. The model geometry for these extreme cases can be seen in figure 3. The non-actuated surface is considered as the origin and a displacement in the outward direction is considered positive.

Figure 3: Model geometry with maximum a) negative voltage; b) positive voltage.

Static deformations, such as those in figure 3 were achieved by driving the MFCs with a DC signal. As already mentioned, the amplifier can accept alternating signals up to 10 kHz, so by providing a sinusoidal input signal the surface will oscillate.

II.B. Experimental Apparatus

The aerodynamic characteristics of the model have been tested in the closed-loop, subsonic wind tunnel at the NUS Temasek Laboratories, figure 4a, which has a 600 mm square test section with a length of 2 m. The test section is connected to the exit of the wind-tunnel nozzle which has a 12:1 contraction ratio. The turbulence intensity level of the wind-tunnel freestream is less than 0.25% at velocities of less than 15 m/s – which is the range of interest in this paper. Since the span of the model is smaller than the width of the test section, the model was mounted vertically on a turntable in the test section floor and a splitter plate was installed 160 mm above it. The turntable allows precise positioning (within $\pm 0.05^\circ$) of the angle of attack, α , of the model. A boundary-layer ingestion slot with a sharp leading edge spanning 72% of the test-section width was utilized to maintain a floor boundary layer roughly as thin as the one on the surface of the splitter plate. The leading edge of the model was positioned 400 mm downstream of the leading edge of the ingestion slot at which location the boundary-layer thickness of the empty test section with $U_\infty = 10$ m/s has been measured to be less than 3 mm. The mounting can be seen in figure 4b.

Figure 4: Experimental facilities. a) closed-loop, subsonic wind tunnel; b) model mounting.

The turntable on which the model was mounted incorporated a force balance consisting of a Gamma ATI SI-32-2.5 piezoelectric gauge capable of measuring the forces and moments along three perpendicular axes. The two axes aligned with the axial (chordwise) and the normal coordinates of the airfoil were used to measure the axial and the normal forces generated by the model. The third axis, coinciding with the axis of rotation of the turntable and aligned in the vertical direction of the tunnel, passed through the airfoil mid-chord point ($c/2$) and was used to measure the pitching moment. The balance was factory calibrated and the corresponding conversion factors are stored in the acquisition unit used with it so the values of the forces and moments obtained are already corrected. The range (and resolution) of the measured forces and moment are $32 (\pm 6 \times 10^{-3})$ N and $2.5 (\pm 5 \times 10^{-4})$ Nm, respectively. For each measurement, 6250 samples were acquired at 1.25 kHz. The lift and the drag coefficients were calculated at each angle of attack from the corresponding values of the chordwise and normal forces as follows:

$$C_L = \frac{F_\eta \cos(\alpha) - F_\zeta \sin(\alpha)}{\frac{1}{2}\rho U_\infty^2 cs}, \quad C_D = \frac{F_\zeta \cos(\alpha) + F_\eta \sin(\alpha)}{\frac{1}{2}\rho U_\infty^2 cs} \quad (1)$$

The subscripts in equation (1) relate to the axis convention described in figure 5. Cartesian coordinates x , y and z refer to the streamwise, cross-stream and spanwise directions respectively with the origin at the aerofoil's leading edge. A rotating coordinate system with an origin at the mid-chord corresponds to the axial (ζ) and normal (η) coordinates of the aerofoil as α changes. Furthermore, a local coordinate system is defined on the aerofoil surface to aid post-processing — n and l being the normal and longitudinal directions respectively.

Figure 5: Axis convention and coordinate notation for mounted aerofoil.

Prior to the aerodynamic investigation, a series of tests were carried out to understand the dynamic response on the skin at different voltage amplitudes and frequencies. The deformation of the skin was measured with a Micro-Epsilon optoNCDT 1710-50 laser displacement sensor. This unit has a measuring resolution of $50 \mu\text{m}$ ($\pm 5 \mu\text{m}$) and its measuring range allows it to be placed outside the wind-tunnel thus enabling measurements of the skin deformation in the flow. These were acquired at 1.25 kHz simultaneously to the corresponding values of the actuation voltage.

Flow-field velocity measurements were obtained using a two-velocity-component PIV system. The flow was uniformly seeded with water-based particles from a SAFEX fog generator. Droplets were produced in the average size SMD of $1\text{ }\mu\text{m}$ whose reflections correspond to no more than 3 pixels in the captured images, allowing a good resolution of the particle displacement when cross-correlation methods are adopted. A dual-head Litron DualPower 200-15 Nd:YAG laser operating at the second harmonic (532 nm) at approximately 150 mJ per pulse was used in conjunction with sheet-forming optics to form a thin sheet ($\approx 1\text{ mm}$) on the x - y plane at 70% along the airfoil span. Two images corresponding to the pulses from the laser were acquired by a 2048×2048 pixels HiSense 620 camera which viewed the streamwise laser sheet orthogonally over the entire field of view. A computer with dual Intel Core processors was used for data acquisition. The acquired images were divided into 32×32 pixel interrogation windows which contain at least 3 seeding particles each. For each image, subregions were adaptively cross-correlated using multi-pass processing with a final 50% overlap that gives a final interrogation area of 16×16 pixels after processing. Based on the flow velocity and the size of the interrogation area, the time separation between the two laser flashes was varied between 100 and 250 μs such that the maximum displacement of a particle in the region of interest is no more than 25% of the interrogation window between pulses. This is optimum for the PIV processing software to calculate accurately the particle velocity. The resulting vector fields were post-processed to remove remaining spurious vectors. A Zeiss 50 mm f/2.0 macro lens which provided a $225 \times 225\text{ mm}$ field of view corresponding to a velocity vector grid of 127×127 points with resolution of approximately $110\text{ }\mu\text{m}$ per pixel. For each acquisition 300 images were taken for statistical averaging.

Profiles of the wake velocity were also obtained at the mid-span location, $2c$ downstream of the leading edge using a single, miniature wire hotwire probe. 2^{17} samples at 6 kHz were acquired at 32 different cross-stream locations between $\pm 0.73c$ in the y -direction. The velocity was calibrated with an 1% error using a dedicated Dantec Dynamics velocity calibrator.

III. Results

III.A. Surface Measurements

The dynamic behavior of the skin was investigated by driving the MFCs with a series of sinusoidal voltage signals of different amplitudes, V_A and frequencies, V_f , defined as:

$$V(t) = V_A \cdot \sin(2\pi V_f t) \quad (2)$$

The fixed points close to the leading and trailing edge of the skin are known to be located $0.07c$ and $0.93c$ along the chord respectively and the time-dependent displacement of the skin was recorded at more 4 locations; $0.25c$, $0.4c$, $0.5c$ and $0.7c$. Figure 6 shows the peak-to-peak displacements at the aforementioned locations. It was important to know how a pressure distribution over the surface affected the response of the skin when immersed in a flow. Therefore measurements were taken in both still and moving air at $Re_c = 5 \times 10^4$.

Figure 6: Peak-to-peak displacement of the skin with flow (Dashed) and without flow (Solid).

Both amplitude *and* frequency of the input signal have significant effect on the dynamic behavior of the skin but the effect of flow appears to be small. As a general rule, the higher the amplitude of sinusoidal voltage, the greater the peak-to-peak displacement. There are a further 100 volts available in the operating range of the MFCs but, with the setup as it is, a current limit of 10mA is reached at 400 V and therefore higher voltages could not be investigated. At frequencies above 30 Hz the amplitude of displacement drops drastically at all measured locations. This was expected and had been observed in previous models.¹³ However, the amplitude begins increasing again at around 90 Hz. From figure 7 this can be attributed to a change in the nature of the skins motion at the highest frequencies, possibly due to a different vibration mode.

Figure 7: Maximum displacement of actuated surface with $V_A = 400V$ (Displacement has been multiplied by a factor of 10 for clarity).

In this paper, the effect of frequency that is to be investigated. However, from figure 6 it is clear that this cannot be achieved by simply varying V_f because this causes large changes in the peak-to-peak displacement

of the surface. Therefore V_A was also modified with V_f , according to figure 6, to keep the peak-to-peak surface displacement at roughly 0.5 mm at a location of $0.4c$ along the chord, thus isolating the effect of frequency. Furthermore, V_f was kept below 90 Hz to ensure that the same vibration mode was being investigated in each case.

III.B. Aerodynamic Measurements

Time-averaged flow fields obtained from PIV data are displayed in figures 8-10. The baseline data was taken when the airfoil surface was stationary and flow fields with $\alpha = 0^\circ, 5^\circ, 10^\circ$ and 15° can be seen in figure 8. At all angles a region of separation is visible in the aft portion of the airfoil where pressure recovery occurs. When $\alpha = 0^\circ$ separation occurs at roughly $0.6c$ and moves upstream to $0.5c, 0.4c$ and $0.2c$ when α is increased to $5^\circ, 10^\circ$ and 15° respectively. This agrees with previous studies of a laminar separation.^{4,15,16} It is also a good demonstration of the inability of a laminar boundary layer to overcome even modest adverse pressure gradients as separation occurs without reattachment even when $\alpha = 0^\circ$. A much larger region of separation is present when $\alpha = 15^\circ$ where the airfoil appears to be stalled. This is confirmed in figure 12 where an abrupt loss of lift is experienced for the stationary airfoil at roughly 12° .

Figure 8: Iso-contours of $|U|/U_\infty$ over the non-actuated airfoil.

The effect of actuation frequency was investigated by testing three different values of V_f : 10 Hz, 40 Hz and 70 Hz, which can be non-dimensionalized with c and U_∞ as defined in equation (3):

$$V_{f+} = \frac{V_f c}{U_\infty} \quad (3)$$

This provided a set of reduced frequencies, V_{f+} , of 0.3, 1.1 and 1.9 respectively.

At the lowest tested frequency, $V_{f+} = 0.3$, the flow fields seen in figure 9 appear very similar to the baseline case, suggesting that actuation at this reduced frequency is incapable of triggering any changes to the flow development. However, from figure 10 it can be seen when $V_{f+} = 1.9$, the separation regions at all angles are significantly smaller when compared with figures 8 and 9, most noticeably when $\alpha = 15^\circ$. The intermediate value of $V_{f+} = 1.1$ was more complicated though. The separation regions in figures 11a-b show

a very minor reduction in the size when compared to the baseline case, however at 10° and 15° this reduction is far more significant thus suggesting that this particular frequency has a greater effect at higher angles of attack.

The effect of actuation on lift can be seen from the C_L -vs- α plot in figure 12 and compares well with what was described above. The curves belonging to the non-actuated and $V_{f+} = 0.3$ cases are very similar, as were the corresponding flow fields, and at higher values of V_{f+} the previously reported reduction in the size of the separated regions has translated into an increase in C_L . For example, when $V_{f+} = 1.9$ the regions of separation in figure 10 were found to have been reduced significantly at all values of α investigated and correspondingly a sizable increase in C_L is found at all values of α in figure 12. Furthermore, figure 12 shows that when $V_{f+} = 1.1$ the effect on C_L increases with α . This would support the greater reduction in separation at $\alpha = 10^\circ$ and 15° when compared with the lower angles seen in figure 11.

Figure 9: Iso-contours of $|U|/U_\infty$ over the airfoil when actuated with $V_{f+} = 0.3$.

Figure 10: Iso-contours of $|U|/U_\infty$ over the airfoil when actuated with $V_{f+} = 1.9$.

Figure 11: Iso-contours of $|U|/U_\infty$ over the airfoil when actuated with $V_{f+} = 1.1$.

Figure 12: Lift coefficient of airfoil model with and without actuation.

The results at normally post-stall angles are particularly promising and merit special attention. From flow field data in figure 8 and the lift curve in figure 12 at $\alpha = 15^\circ$ the non-actuated airfoil experiences stall. The vorticity contours at this angle in figure 13 clearly show that when $V_{f+} = 0$ and 0.3 a shear layer of strong vorticity separates from the airfoil near the leading edge and is accompanied by a large recirculating region, as shown by the local velocity vectors. Such flow conditions would result in an abrupt loss of lift which according to figure 12 occurs at around $\alpha = 11^\circ$ for both the non-actuated airfoil and $V_{f+} = 0.3$.

In their experimental survey of high Reynolds number airfoil performance in the 1930s, Jacobs & Pinkerton¹⁷ observed the onset of stall on a NACA 4415 to occur at $\alpha = 15^\circ$ when $Re_c = 3.1 \times 10^6$ and Hoffmann et al.¹⁸ also found C_L to increase with α for $\alpha \leq 15^\circ$ when $Re_c = 0.75 \times 10^6$. In the uncontrolled case examined here $C_{L_{max}}$ occurs at $\alpha = 10^\circ$ (5° sooner than at higher Re_c) suggesting that the lack of energy in a laminar boundary layer results in an earlier onset of stall, which has had a profound effect on the maximum achievable lift. $C_{L_{max}}$ for the uncontrolled case in figure 12 was found to be 0.91 , compared with 1.38 and 1.57 at $Re_c = 0.75 \times 10^6$ and $Re_c = 3.1 \times 10^6$ respectively. This has implications for the minimum take-off and landing speed for small aircraft for example where high lift coefficients are required.

It was noted earlier that when $V_{f+} = 1.1$ and 1.9 the large separated region at $\alpha = 15^\circ$ was noticeably reduced — in the latter case it has disappeared altogether. The vorticity contours in figure 13c show that when $V_{f+} = 1.1$ the shear layer still separates but the size of the recirculation zone is substantially reduced. From the C_L -vs- α plot in figure 12, it is found that this is a consequence of the actuation delaying the onset of stall by around 2° , to $\alpha = 13^\circ$. Figure 13d shows that there is no recirculation zone present when $V_{f+} = 1.9$ and the shear layer remains attached to the airfoil surface, so there is no loss of lift with increasing α within the ranges of angles tested, as can be seen in figure 12. This means $V_{f+} = 1.9$ is indeed capable of suppressing stall up to, and possibly beyond, $\alpha = 16^\circ$.

By delaying the onset of stall, dynamic surface actuation has increased $C_{L_{max}}$ to 1.10 when $V_{f+} = 1.1$ and 1.19 when $V_{f+} = 1.9$ — a 30% increase on the baseline case.

Figure 13: Iso-contours of $\overline{\Omega_z}$ over the airfoil when $\alpha = 15^\circ$.

Owing to the low speeds at which these experiments were conducted, the aerodynamic forces were low. This was a problem when trying to obtain reliable drag measurements. Instead, an indication of the effect that dynamic surface actuation has on drag is presented in figure 14 in the form of profiles of U_∞ in the wake $2c$ downstream of the leading edge. The velocity deficit observed in such profiles provides information of the profile drag.

Again, the previously described reduction in separation has translated into a reduction in the velocity deficit, and therefore drag. Actuation at $V_{f+} = 1.9$ which has already been found to have the greatest effect on lift, has a considerable effect on drag. Just as the separated region was reduced and lift was increased at all angles when compared with the baseline case, the velocity deficit caused by the airfoil when actuated with $V_{f+} = 1.9$ is smaller when compared with the non-actuated airfoil at all angles. It also further highlights the benefits delaying stall, with the airfoil's wake at 15° being much smaller when $V_{f+} = 1.1$ and 1.9 then when $V_{f+} = 0$. The position of the peak in velocity deficit is also shifted down as a result of the flow separating further downstream on the airfoil surface when compared with the uncontrolled case. This suggests that dynamically actuating the surface at these frequencies not only increases $C_{L_{max}}$, but also reduces the drag significantly at normally post-stall angles. Together, this would result in a significant improvement in the L/D ratio at high angles of attack.

Figure 14: Profiles of $|U|/U_\infty$ at a distance of $2c$ downstream of the leading edge.

IV. Conclusion

An airfoil model has been designed and constructed with an upper skin that can be dynamically actuated by applying a voltage signal to Macro Fiber Composite actuators (MFC) bonded to its under side. These piezoelectric actuators are very thin, light and robust, and have low power consumption. The feasibility of applying these actuators for dynamic surface morphing was first investigated with a series of surface displacement measurements. It was found that the displacement depends on both frequency and voltage but the addition of flow was not found to drastically change the skins behavior. A variety of voltage amplitudes and frequencies were tested and the MFCs proved to be capable of accepting frequencies of at least 120 Hz at voltages of at least 400 V. However, at all values of V_A the peak-to-peak displacement was found to drop significantly at values of V_f above 30 Hz and at V_f greater than 100 Hz the nature of the displacement appears to change – possibly triggering a different vibration mode – so for the purposes of this paper the frequency of actuation was kept below 90 Hz.

The aerodynamic effect of the actuation was investigated with corresponding force balance, PIV and hotwire measurements. An effort was made to keep the amplitudes of displacement comparable by varying V_A so the effect of frequency could be studied. It was found that at reduced frequencies above 1 the dynamically actuated airfoil exhibits a higher lift coefficient at all positive angles of attack and actuation also proved successful in delaying stall. When $V_{f+} = 1.9$ stall was avoided entirely within the range of angles investigated and a 30% increase of $C_{L_{max}}$ was achieved.

Owing to the very small forces involved, reliable drag measurements have proven elusive thus far, however measurements of the velocity deficit in the wake shows that the increase in lift is accompanied by a reduction

in velocity deficit and therefore in drag. This would have a profound impact on the performance — measured at the ratio of L/D — and can be attributed to a reduction in the size of the separated region.

V. Ongoing Work

Further experiments will be carried out with the goal of quantifying the drag reduction — possibly through a combination of more detailed wake surveys and control volume analysis. Also, more detailed behavior of the surface will be sought through the use of photogrammetry. It is hoped that this will not only provide a better resolution of the surface motion than provided in figure 7 but will also capture any 3 dimensional motion. Most significantly, Direct Numerical Simulations (DNS) a currently being developed that will complement the experiments. Once complete and verified against experimental results, they will enable a detailed investigation of the flow development around the controlled and uncontrolled airfoils and the physical processes that explain the findings in this paper.

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