

Control Of Flow Around An Aerofoil At Low Reynolds Numbers Using Periodic Surface Morphing

by

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*In loving memory of
Kenneth Ward
'Popsey'
28/02/1936 – 03/03/2016*

Declaration

I declare that the work within this thesis which I present as my own was carried out by myself during my candidature. Any research that has been referred to but not my own has been properly acknowledged.

Gareth Jones

March 2016

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Abstract

This thesis combines experimental and computational methods to investigate the low Reynolds number flow ($Re_c = 50,000$) around a NACA 4415 aerofoil, and its control using periodic surface motion. A physical model was fabricated and tested in a closed-loop wind tunnel and a good comparison between the experiments and computations was achieved.

Time-resolved measurements of the surface reveal that the peak-to-peak displacement is a function of both the amplitude and frequency of the input voltage signal but the addition of aerodynamic forces does not cause significant changes in surface behaviour. The vibration mode shape exhibits a single peak and is uniform in the spanwise direction at frequencies below 80 Hz, above which a change in the vibration mode occurs.

The flow around the actuated aerofoil was compared with the baseline (i.e. unactuated) flow. The latter exhibits a large separation region and, as a result, produces relatively high drag and low lift forces. By analysing the experimental and computational data, the large separation zone was found to be the result of laminar separation without reattachment. Transition to turbulence does occur but too close to the trailing edge, and far from the wall, for sufficient pressure recovery to take place for reattachment.

When actuated at 70 Hz, the frequency spectra in the vicinity of the trailing edge and near-wake was found to be dominated by the actuation frequency. Sharp peaks suggest the production of Large Coherent Structures at this frequency. In agreement with the experiments, the computations revealed that the vortex shedding from the shear layer was ‘locked-on’ to the surface motion and spanwise coherent vortices were produced during each actuation cycle. The increased momentum entrainment associated with them enabled a large suppression of the separated region, which was seen in both the experiments and computations. The result was a simultaneous increase in C_L and decrease in C_D and therefore a large increase in the L/D ratio.

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Chapter 1

Introduction

1.1 Motivation

The ability to fly relies on the satisfactory performance of lifting surfaces and the proper functioning of such a surface is heavily dependent on the geometry of its cross-section, known as an aerofoil. The aerofoil is therefore of fundamental importance and its ideal geometry depends profoundly on the conditions in which it is expected to operate [1]. A combination of small geometric dimensions, low speed and/or a low density environment — such as high altitude — results in a low Reynolds number flight regime. This is considered to be the case when $Re_c < 500,000$ [6], in which Re denotes the Reynolds number and the subscript c signifies that Re is based on aerofoil chord length. Such flight conditions are notoriously problematic for conventional aerofoils.

The Reynolds number is the ratio of inertial to viscous forces, so that when Re_c is low, the influence of viscosity will be relatively high. The increased effect of viscosity in the low Re_c flight regime ensures that natural disturbances, from freestream turbulence or surface imperfections for example, are strongly damped. As a result, a laminar boundary layer forming on the aerofoil's upper surface can persist beyond the suction peak and into the pressure recovery region whereupon it encounters an adverse pressure gradient (APG). Viscous effects close to the aerofoil's surface slow a fluid element down thereby reducing its kinetic energy. Turbulent boundary layers can compensate for this by mixing the low

momentum fluid with higher momentum fluid further away from the surface. However, laminar boundary layers have no mechanism for re-energising the near-wall flow, leaving them incapable of overcoming even modest APG. Consequently, when present on an aerofoil, laminar boundary layers are prone to separation even at low angles of attack, α , meaning low lift coefficients, C_L , and high drag coefficients, C_D , are inherent in flight at low Re_c .

Engineering applications involving low Re_c include rotor blades, hydrofoils and wind turbines [7] although most of the recent interest has concentrated on the operation of small Unmanned Aerial Vehicles (UAV) and Micro Air Vehicles (MAV). In addition to their relatively small size, UAV and MAV can also be required to perform at low speeds and/or high altitudes [7] putting them firmly in low Re_c realm. Both military and civilian interest in 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 [8]. Also the availability of very small sensors, video cameras, listening devices and control hardware has meant that the capability of MAV for complex missions are currently the subject of much research and development to meet projected military and environmental needs [9].

1.2 Scope of Research

The performance of aerofoils at low Re_c will be reviewed in detail in the following chapter (Chapter 2). To summarise briefly, the flow development around aerofoils at low Re_c can exhibit a combination of flow features such as flow separation, free shear layers, wall-bounded shear layers, laminar-turbulent transition and reattachment. The aerodynamic forces produced by the aerofoil are highly sensitive to the development of these features, which are sensitive to Re_c and angle of attack, α , as well as the external environment and aerofoil geometry. In summary, the aerodynamics of low Re_c aerofoils is extremely diverse and flow control methods claiming to improve the associated poor performance must therefore be flexible.

Methods pertaining to mitigate these problems will also be reviewed in Chapter 2. Again, to

summarise briefly, passive flow control techniques, which are more easily implemented than their active counterparts, have proved to be very effective in delaying separation by increasing the near-wall momentum. However, such methods often deteriorate aerodynamic performance at off-design conditions [10]. Passive flow control must therefore be ruled out as a viable approach since an aerofoil could encounter several different characteristic flow modes at a given α , Re_c or background environment, and it is inconceivable that a passive device can improve performance at all conditions. In contrast, active control approaches offer the significant advantage of being largely innocuous except when activated, and thus remove 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 [11].

The drawback of active control with respect to passive is that 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. By exploiting the fact that Large Coherent Structures (LCS) are responsible for transporting momentum across a shear layer and are controllable using periodic motion, Seifert *et al.* [12] discovered that by superimposing a periodic motion on top of steady blowing, the efficiency of blowing methods could be greatly increased.

More recently, the development of new actuation devices and material systems has enabled novel approaches for flow control to be explored and the concept of periodic motion has been further investigated. This thesis will exploit the properties of the Macro Fiber Composite (MFC) actuators, developed at NASA. These are piezoelectric actuators which consist of three main components: a) a sheet of aligned piezoceramic fibres, b) a pair of thin polymer films etched with a conductive electrode pattern and c) an adhesive matrix material [13]. The piezoceramic fibres provide a coupling between mechanical and electrical fields so that when they experience an electrical field provided by the electrode pattern, mechanical deformation of the fibres occurs. As a result, MFC can provide distributed deflection for very little power consumption. By integrating them into an aerofoil model and applying a sinusoidal voltage signal, a novel active flow control method, based on the periodic motion of the aerofoil can be explored. In this thesis it will be ascertained if this concept can be practically implemented

and the effect it has on aerofoil performance at low Reynolds numbers will be investigated. The investigation will be both experimental (Chapters 3-5) and computational (Chapters 6 and 7).

Chapter 2

Literature Review

This literature review is separated into two main sections: (i) low Reynolds number aerodynamics and (ii) application of active flow control to mitigate the associated poor performance.

2.1 Low Reynolds Number Aerodynamics

Carmichael’s 1981 encyclopedic work [14] on the performance of two-dimensional aerofoils surveyed a comprehensive collection of low Re_c experimental data from $Re_c = 2 \times 10^4$ to 2×10^5 and presented it as a subset of a much larger range from $10^0 < Re_c < 10^9$. Lissaman’s 1983 [1] review of Carmichael’s work includes a flight Re_c spectrum, reproduced in Figure 2.1, to highlight the huge range in which flight has been observed. In Tennekes’ book ‘The Simple Science of Flight’ it is stated that only the Hertzsprung-Russell diagram, which describes the relationship between luminosity and effective temperature in stars, covers so wide a range [15]. The upper and lower extremes of Figure 2.1 are occupied exclusively by man-made flying machines and nature respectively. Above $Re_c = 10^7$ very large aircraft appear and it is in this extensively studied range that the highest-performance aerofoils have been developed [1]. In the range of $10^6 < Re_c < 10^7$ we find, as Carmichael put it, “man and nature together in flight [14].” However, the drop in aerofoil performance relative to that observed when $Re_c > 10^7$ restricts the man-made contribution to simple model aircraft. In contrast,

countless species of birds and insects are capable of remarkable aerial acrobatics [16] proving that high performance lifting surfaces can exist even at very low Re_c .

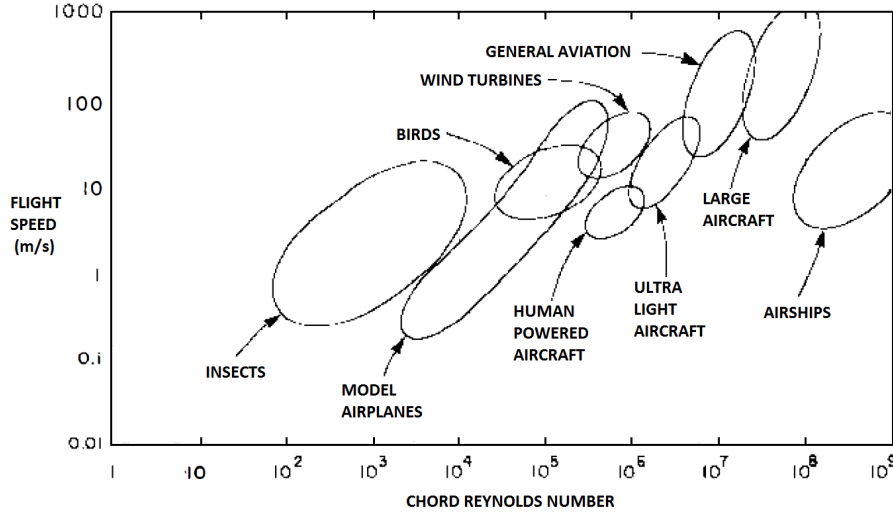


Figure 2.1: Flight Reynolds number spectrum from [1].

2.1.1 Aerodynamic Forces

The performance of aerofoils can be quantified by the aerodynamic forces they produce. Lift, L , and drag, D , are the integrated effect of pressure and viscous forces acting in the streamwise and cross-stream directions respectively and are usually expressed using the non-dimensional coefficients, C_L and C_D . The ratio between the two coefficients provide the glide factor of an aerofoil (C_L/C_D), which is a good measure of efficiency. The endurance factor is defined as $(C_L^{1.5}/C_D)$ [17, 18].

The performance of aerofoils designed for Re_c greater than 5×10^5 deteriorates rapidly as the Re_c decreases below this value [2]. In 1979 McMasters and Henderson [2] made use of available computational tools to clarify and extend the existing experimental database on aerofoils at low Re_c . From their findings they were able to communicate the extent of this deterioration through the diagram in Figure 2.2.

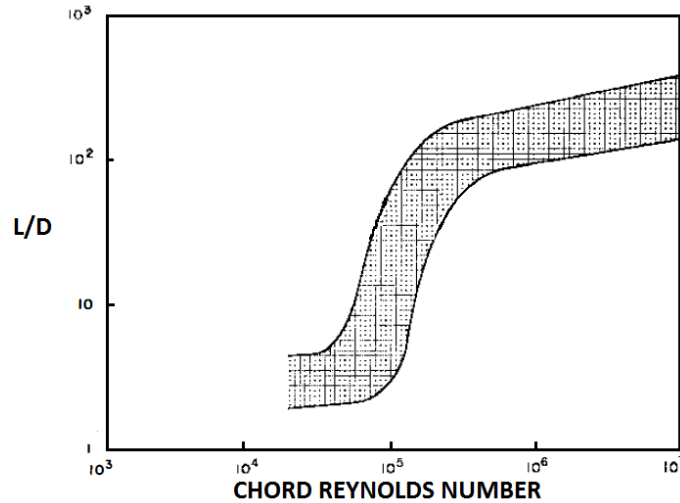


Figure 2.2: Aerofoil Performance vs Reynolds Number from [2].

The exact onset of the performance loss depends, as will be discussed in this section, on many factors. However, what can be deduced is that a significant drop in the glide factor occurs at $Re_c < 10^6$.

Mueller and Batill [19], Laitone [20], Marchman and Abtahi [21] and, more recently, Wang *et al.* [22], all measured the aerodynamic forces produced by different aerofoils at conditions within the $Re_c < 10^6$ range. Despite the different geometries and flow conditions, similar characteristics were observed in the relationship between Re_c , α and C_L .

When $Re_c = 4 \times 10^4$, Mueller and Batill [19] observed a modest increase in C_L with increasing α at low angles, before an abrupt increase occurred at 8° . This agrees with the finding of Laitone [20] and Wang *et al.* [22]. Laitone [20] found the lift-curve gradient is not only low at low Re_c but also highly dependent on Re_c . A slope of 0.041 when $Re_c = 2.07 \times 10^4$ increased to 0.061 and 0.08 as Re_c was increased to 4.21×10^4 and 7.0×10^4 respectively. These values are small compared to a slope of 0.110 (or 2π when α is expressed in radians) predicted by classical lifting line theory [18]. Laitone [20] also found that the severity of the discontinuity in the lift curve decreased with increasing Re_c until it had disappeared altogether when $Re_c = 7 \times 10^4$. This agrees with data of Marchman and Abtahi [21] who also observed no discontinuity when $Re_c = 7 \times 10^4$. The jump in C_L is reflected in a corresponding jump in C_L/C_D [20].

Marchman and Abtahi [21], who also presented data at $Re_c = 7 \times 10^4$, agree with Laitone [20] that the lift-curve exhibited no non-linear jump at this Re_c . Additionally, they found that once $Re_c > 1 \times 10^5$ the lift-curve was not only linear, but also independent of Re_c . Wang *et al.* [22] and Hsiao *et al.* [23] reinforce this and show that for this range of Re_c , the Re_c -independent lift-curve does exhibit a 2π gradient. It is worth noting that the lift-curve slope of 2π is found by Wang *et al.* at 5×10^5 , the quoted Re_c below which aerofoils start to exhibit performance losses [2, 1].

To summarise, when Re_c is low the C_L – vs. – α slope is dependent on Re_c . In general, a lower C_L is achieved for a given value of α when Re_c is decreased. The lift-curve slope – initially much lower than those associated with high Re_c conditions – increases with increasing Re_c and the appearance of discontinuities is reduced. Further increase in Re_c eventually removes discontinuities altogether and the lift-curve becomes linear. When Re_c is sufficiently high the lift-curve slope also becomes independent of Re_c , however this does not coincide with the disappearance of the discontinuity: the lift-curve can be linear and still dependent on Re_c . It can therefore be concluded that the C_L/α dependency on Re_c and the appearance of a discontinuity in C_L/α are controlled by different mechanisms.

The above discussion is concerned with pre-stall α . Stall is characterised by a precipitous loss of lift and increase in drag [18] and the angle at which this occurs is known as the stall angle, α_{stall} . Despite the pre-stall independence of Re_c in the results of Marchmann and Abtahi [21] and Hsiao *et al.* [23], it can be seen in the same results that the approach to stall remains non-linear and α_{stall} is dependent on Re_c , even for $Re_c > 1 \times 10^5$. Both Marchmann and Abtahi [21] and Wang *et al.* [22] observed α_{stall} to increase with an increase in Re_c when $Re_c < 10^6$, although this has been found to become less significant for $Re_c > 10^6$ [24, 21, 25].

Another phenomenon associated with stall at low Re_c is hysteresis. Hsiao *et al.* [23] found a clockwise hysteresis in the lift-curve in the region of stall, meaning that α_{stall} at increasing α is higher than it is at decreasing α . The hysteresis loop becomes less prominent as Re_c increases and Hsiao *et al.* [23] found hysteresis to be insignificant when $Re_c = 7.74 \times 10^5$. This is important when considering aerofoil performance because α_{stall} affects the maximum lift an aerofoil can produce, C_L^{max} , and the hysteresis is relevant to stall recovery [23].

In conclusion, even when C_L/α is linear and independent of Re_c at low angles, the lift-curve can still be dependent on Re_c as stall is approached. Adding to this already complicated picture the fact that at very low $Re_c < 1 \times 10^4$ stall has not been observed to occur [25, 22], it becomes clear that the performance of aerofoils at low Re_c is a complex phenomenon. In the following section it will be shown that the changing C_L – vs. – α gradients, discontinuous jumps, and appearance — or lack thereof — of stall described in this section /are all a direct result of the characteristic flow patterns that develop and interact.

2.1.2 Flow Development

Hsiao *et al.* [23] and Yarusevych *et al.* [26] made use of the smoke flow visualisation technique, Huang *et al.* [27] the Particle Tracking Flow Visualisation (PTFV) and Particle Image Velocimetry (PIV) methods and Wang *et al.* [22] used Laser-Induced Fluorescence (LIF) to study the flow development around aerofoils in the Re_c range of $1 \times 10^3 - 5 \times 10^5$ and a variety of α . These authors found different characteristic flow modes that depend on Re_c and α [27]. Furthermore, these modes are associated with distinctly different wake characteristics [26].

Wang *et al.* [22] reviewed experiments around a NACA 0012 at a large range of Re_c and observed the formation of eight distinctive flow patterns:

- A. Fully attached laminar boundary layer
- B. Laminar boundary layer separation near the trailing edge
- C. Laminar boundary layer separation close to leading edge
- D. Laminar bubble, i.e. laminar flow from separation to reattachment
- E. Partially laminar bubble, i.e. transition prior to reattachment
- F. Fully attached turbulent boundary layer
- G. Turbulent boundary layer separation near the trailing edge
- H. Turbulent boundary layer separation from the leading edge

By considering the range of Re_c at which these eight modes occurred it can be seen that five of them (A, B, C, D and E) are observed in the range of Re_c which McMasters and

Henderson [2] associated with the loss of aerofoil performance. Wang *et al.* [22] divided this range of Re_c , thus far referred to as low Re_c , into three different regimes: ultra-low Re_c , low Re_c and moderate Re_c . Each of these regimes have distinct characteristics making it sensible to use Wang *et al.*'s [22] terminology, rather than referring them all as low Re_c . Consequently, the terms ultra-low Re_c , low Re_c and moderate Re_c will be used henceforth.

Ultra-Low Reynolds Numbers ($Re_c < 1 \times 10^4$)

The ultra-low Re_c regime is considered by Zhou *et al.* [25] to occur at $Re_c < 1 \times 10^4$. As mentioned in Section 2.1.1, the abrupt drop in C_L indicative of stall is not present in the lift-curve at such Re_c and this defines the ultra-low Re_c regime.

Huang *et al.* [27] investigated the start-up flow field around a NACA 0012 at Re_c of the order of 10^3 , firmly in the ultra-low Re_c regime, at angles ranging from 0° to 90° . By analysing the pathlines, instantaneous streamlines and vorticity at increasing time instances they visualised the evolution of the near surface flow and they too came to the conclusion that there existed five characteristic flow modes:

1. **Attached flow mode:** observed only when both α and Re_c are extremely low. The near surface flow remains attached everywhere at all time instances, so no separation or vortical structures are found.
2. **Trailing-edge vortex mode:** observed when $\alpha_{att} < \alpha < \alpha_{stall}$. An initially attached boundary layer, separates from the trailing edge and the separation point moves upstream with time. A trailing-edge vortex (with anti-clockwise rotation) wraps around from the pressure surface to the suction surface near the trailing edge and grows with time. As this vortex departs from the surface, a surface vortex (with clockwise rotation) is formed in the shear layer and advects towards the trailing edge. Downstream shedding of a trailing edge vortex alternates with surface vortex shedding. This process eventually becomes periodic.
3. **Separation Vortex Mode:** observed when $\alpha > \alpha_{stall}$. The boundary layer separates from the trailing edge and the separation point moves upstream as time evolves.

Alternating shedding of trailing-edge vortices and surface vortices is again observed, which eventually becomes periodic. The distinction between (2) and (3) is that an increase in vorticity induced by the reverse flow enables the surface vortex to form prior to the trailing-edge vortex in this mode. The trailing-edge vortex is then induced by the low pressure associated with the separation vortex.

4. **Leading-edge vortex mode:** a start-up vortex is formed at the leading edge but does not propagate immediately to the wake region. Instead, it rolls downstream along the suction surface and forms a large surface vortex which induces a trailing edge vortex when it reaches the trailing edge. As the large surface vortex sheds into the wake, the trailing-edge vortex grows in size and merges with one of two counter rotating surface vortices that have formed near the leading edge. The vortex resulting from this merger then sheds into the wake before the process repeats with the remaining leading-edge vortex.
5. **Bluff-body mode:** observed when α becomes very high. A large low pressure region forms behind the aerofoil, similar to that occurring in bluff body flows. As a result, both the leading- and trailing-edge vortices are sucked into the low pressure region. The trailing-edge vortex then sheds into the wake, followed by the leading edge vortex and then the cycle repeats.

Flow modes (1), (2), (3), (4) and (5) described by Huang *et al.* [27] can be compared with the previously introduced modes (A), (B), (C), (D) and (E) from [22]. When $\alpha = 0^\circ$ the boundary layer is fully attached meaning flow mode (A) is present. This agrees with both the description and flow parameters of Huang *et al.*'s [27] attached flow mode (1). However, for $2^\circ < \alpha < 5^\circ$ the boundary layer is seen to separate, with the separation point seeming to originate from close to the trailing edge and moving upstream with increasing α . Separation occurring at such low angles suggests that the laminar boundary layer cannot withstand even a slight adverse pressure gradient [22] and causes the low C_L/α gradient described in Section 2.1.1 for low values of Re_c . Additionally, when $\alpha > 5^\circ$ separation appears to occur very close to the leading edge and the resulting shear layer rolls up over the aerofoil surface to form a surface vortex. The latter two of the three α ranges, i.e. $2^\circ < \alpha < 5^\circ$ and $\alpha > 5^\circ$,

display characteristics of Wang *et al.*'s [22] modes (B) and (C) respectively. When considering the Re_c and α under investigation, they are also reminiscent of Huang *et al.*'s [27] trailing-edge vortex mode (2), separation vortex mode (3) and the leading-edge vortex mode (4). It is important to note that Wang *et al.*'s [22] flow visualisation did not look at the time evolution of the start-up flow but rather at the fully developed flow. Therefore, bearing in mind that they were distinguished only by the order in which the surface vortex and trailing-edge vortex were initially formed, Huang *et al.*'s [27] trailing-edge vortex and separation vortex modes would appear as the same flow mode in Wang *et al.*'s [22] descriptions. Hence, the trailing-edge vortex mode (2) and separation vortex mode (3) described by Huang *et al.* [27] equate to Wang *et al.*'s [22] mode (B) and the laminar separation from close to the leading edge, which occurred for the α range of 8° to 20° , or mode (C), is equivalent to the leading-edge vortex mode (4).

The remaining bluff body mode observed by Huang *et al.* [27] occurred at very high α , which were not investigated by Wang *et al.* in [22]. Therefore, for the range of ultra-low Re_c and moderate α , three fully developed characteristic flow modes can be observed when α (or Re_c) is increased: an attached laminar boundary layer, a laminar boundary layer with trailing edge separation and finally a completely separated laminar shear layer. The equivalence of the start-up flow modes described in [27] and the fully developed flow modes found in [22] is summarised in Table 2.1.

Ultra-Low Re_c Modes		
Huang <i>et al.</i> [27]		Wang <i>et al.</i> [22]
1	→	A
2	→	B
3	→	B
4	→	C
5	→	N/A

Table 2.1: Flow mode equivalence at ultra-low Re_c

It has already been stated that the low lift-curve gradient associated with ultra-low Re_c is a result of separation occurring at low α . The other characteristic of ultra-low Re_c is the absence of stall. The reason for this is that the transition between the aforementioned flow modes is gradual and successive. A fully separated flow mode (C) is observed but it is

preceded by a continuous upstream movement of separation point from the trailing edge (mode B), rather than an abrupt change in the flow field. Consequently, at ultra-low Re_c there is an absence of the abrupt loss of C_L associated with stall.

Low Reynolds Numbers ($Re_c > 1 \times 10^4$)

As mentioned previously, the fully attached flow mode can only exist at negligible α within the ultra-low Re_c regime. There are however some clear similarities between the two regimes.

When $Re_c = 2 \times 10^4$ – the low Re_c regime – Wang *et al.* [22] observed trailing edge boundary layer separation is already present when $\alpha = 0^\circ$. This is similar to that which occurred at $\alpha = 2^\circ$ at ultra-low Re_c [22]. When $\alpha = 2^\circ$ in the low Re_c case, separation is followed by downstream amplification of instabilities causing the shear layer to roll up prior to the trailing edge [26]. This is similar what occurred for $\alpha > 8^\circ$ at ultra-low Re_c . The shear layer roll-up forms large-scale vortices which are shed into the wake. A large separated region, which extends beyond the aerofoil’s trailing edge, causes a wide wake containing large-scale vortices [23, 7, 26]. Moreover, when $\alpha > 24^\circ$ Hsiao *et al.*’s [23] smoke flow visualisations reveal a leading-edge separation which had thus far also been considered an ultra-low Re_c flow mode — mode (C).

In spite of the apparent similarities in flow development, the two regimes can be distinguished by the presence of stall. As mentioned earlier, in the ultra-low Re_c regime the transition between flow modes (B) and (C) is gradual and successive and as a result there are no abrupt changes in the flow field or aerodynamic performance. In contrast, while modes (B) and (C) have been observed in the low Re_c regime, they are separated by intermediate flow modes (D) and (E).

As seen in the Re_c flow visualisations of Hsiao *et al.* [23], Wang *et al.* [22] and Yarusevych *et al.* [26], a small change in α or Re_c within this regime can cause the previously separated shear layer to reattach before the trailing edge. Consequently, a much narrower wake forms with smaller structures and less coherence [26]. This sudden reattachment is responsible for the discontinuous jump in the C_L – vs. – α of Mueller and Batill [19], Laitone [20] and Wang *et al.* [22], and described in Section 2.1.1. The resulting closed region between the separation

and reattachment points is known as a Laminar Separation Bubble (LSB) and is well documented in classical aerodynamics.

Although much research has been done over the years on both separation and transition, the LSB is still not well understood [28]. The appearance of these bubbles has a profound effect on aerofoil performance at low Re_c . As will be explained below, they essentially control stall and are the defining feature in the low Re_c regime; the flow physics involved is so rich that they deserve special attention.

Moderate Reynolds Numbers

Like low Re_c , the characteristic flow mode (E) occurs at higher Reynolds numbers, within the moderate regime. With increasing α the location of transition shifts towards the separation point and the separation bubble is shortened. This is terminated with increasing α by leading edge stall. According to Wang *et al.* [22] this regime occurs in Reynolds number range from 3×10^5 to 5×10^6 - more than an order of magnitude greater than the one encountered in this thesis.

2.2 Laminar Separation Bubble

Laminar Separation Bubbles were first observed in the 1930s [29] but received little attention until interest was aroused through the use of thin aerofoil sections for reducing the effects of compressibility in the late 1950s and 1960s [30]. During this period a series of experiments were carried out in which the influence of Re_c and pressure gradient on the formation of LSB was investigated. These are summarised by Tani [31] and Horton [3]. Considering the effects they have on performance, as described above, the pressure gradient and Re_c are particularly relevant to aerofoils — α being explicitly responsible for changes in pressure gradient. The experimental facilities available at the time limited these classical experiments to time-averaged results and Horton [3] was able to describe what has become the classical steady-state structure of the LSB, shown in Figure 2.3. Two main features were introduced: a **free shear layer** formed following the boundary layer separation and contained between the

outer edge of the viscous region and the mean dividing streamline, and a **recirculation bubble** contained between the mean dividing streamline and the aerofoil surface. The formation of these two regions depends on the location of transition. In the laminar portion, underneath the separated flow, a more-or-less quiescent or slowly circulating area develops [31] and creates a region of near-constant surface pressure [26]. This is commonly referred to as the **dead-air region** and the constant pressure as a **pressure plateau**. After transition, the turbulent entrainment energises the shear layer so that pressure is able to rise and reattachment takes place. As a result of the pressure rise, a recirculating flow is set up inside the bubble. Since virtually all the pressure rise takes place in the turbulent part of the bubble the recirculation is concentrated mainly in this region, known as the **reverse-flow vortex** [3].

It was thought that an LSB could exist in a ‘short’ or ‘long’ form, terms first coined by Owen and Klanfer [32] in 1953. They used the ratio of total bubble length to boundary-layer displacement thickness to distinguish between the two, but it quickly became more appropriate to consider their effect on the overall pressure distribution [3]. Horton [3] found that at moderate α a bubble would exist whose presence would have very little effect on the overall aerofoil characteristics and pressure distribution. However, by increasing α — and hence the adverse pressure gradient — a point was reached where a sudden expansion in the bubble was observed. This process of sudden expansion became known as **bursting** and results in pressure distributions far different from their inviscid counterparts.

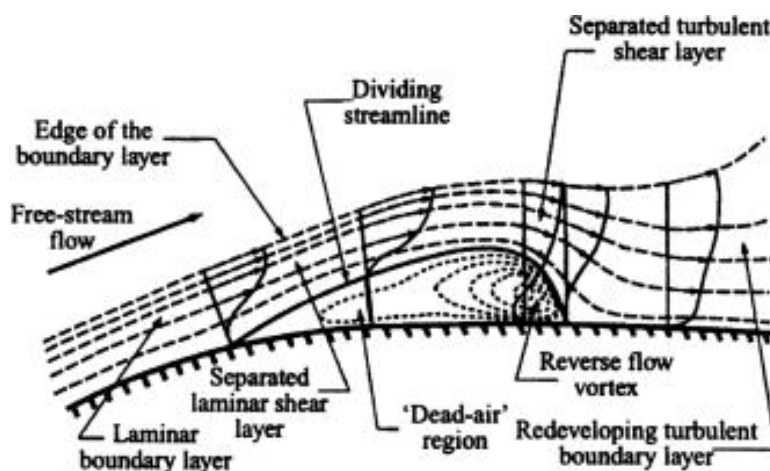


Figure 2.3: Steady-state structure of a LSB from [3].

2.2.1 Laminar Separation Bubbles and Aerofoil Performance

More recent studies by Boutilier and Yarusevych [33], Mueller and O’Meara [34] and Yarusevych *et al.* [26] investigated the effect of Re_c and α on the size of LSB on aerofoils. The time-averaged separation, transition and reattachment locations were obtained from plots of surface pressure distribution based on the description of the LSB structure in the previous paragraph. Separation and transition could be determined from the onset and termination of a pressure plateau respectively and reattachment was taken as the point at which the enhanced rate of pressure recovery following transition decreased. Mueller and O’Meara [34] and Boutilier and Yarusevych [33] confirmed these locations from local velocity profiles and turbulence intensities obtained using hotwire anemometry. All three studies agreed that both Re_c and α affect the average location and size of an LSB. Mueller and O’Meara [34] observed that for a given α , the separation location remains fairly constant with changing Re_c but when Re_c is kept constant the separation point moves upstream with an increase in α . This was confirmed by Boutilier and Yarusevych [33] who plotted the location of separation against α for four different values of Re_c between 1×10^5 and 2.5×10^5 . The upstream movement with increasing α is attributed to the more severe APG which would occur at a higher α [34]. The upstream movement of the reattachment location is also much more rapid than the movement of the separation point when α is increased. The reduction in the size of the LSB with increasing α therefore results predominantly from the upstream movement of the reattachment location [35]. Overall, it appears that α has a more severe effect on the formation of an LSB than Re_c does [34, 35].

Comparing pressure distributions exhibiting LSB with their inviscid counterparts, as in Tani [31] and Yarusevych *et al.* [26], reveals the effect that the presence of an LSB can have on the entire flow over an aerofoil, not just in the proximity of the bubble. Both Tani and Yarusevych *et al.* observe the previously discussed changes in size and location of the LSB with changes in Re_c and α but, in addition, the pressure distributions reveal a significant broadening and reduction of the suction peak upstream of the bubble when compared with the inviscid case. They also both observe a sudden collapse of the pressure distribution, Tani [31] at high α and Yarusevych *et al.* [26] at very low Re_c . In such cases, the pressure plateau extends to the trailing edge, suggesting no reattachment has occurred. Yarusevych *et al.* [26]

found that once this had occurred, further decrease in Re_c had little effect.

An increased in α or a reduction in Re_c can lead to bursting and the formation of a long LSB or an unattached free shear layer from a short LSB. Bursting and long separation bubbles are therefore considered a precursor to thin aerofoil stall [36]. In fact, this is precisely what distinguishes ultra-low Re_c regime from the low Re_c . If stall is preceded by the bursting of an LSB, then if an LSB does not occur, neither can stall. The ultra-low Re_c flow modes do not contain an LSB. As a result, stall does not occur on aerofoils at ultra-low Re_c [22]. On the other hand, in the low Re_c regime, LSB do occur and their bursting leads to an abrupt change in the flow field and consequently an abrupt change in the lift curve.

2.2.2 Stability of Laminar Separation Bubbles

Advances in analysis techniques enabled much more in-depth studies to be carried out, albeit many of them on the simpler flat plate geometry. Since the vortex formation process is essentially two-dimensional, early computational investigations of LSB assumed a strictly two-dimensional flow. For relatively weak pressure gradients Pauley *et al.* [37] did observe separation to build up into a steady, closed bubble structure reminiscent of that seen in Figure 2.3. As the adverse pressure gradient increased, the separated region lengthened and small oscillations in skin friction formed near the reattachment point. The magnitude of the oscillations increased with the pressure gradient but the overall bubble structure initially remained steady. However, beyond a critical pressure gradient skin friction curves became periodically negative and this was classed as **unsteady separation**. By time-averaging over a number of shedding cycles Pauley *et al.* [38] compared their results to those of the aforementioned classical experiments. They found a time-averaged unsteady separation structure similar to that which Gaster [30] and Horton [3] called ‘short’ bubbles – a region of constant pressure followed by a strong adverse pressure gradient at reattachment. On the other hand, the steady structure found at weak pressure gradients did not exhibit a distinct strong adverse pressure gradient – much like the Gaster [30] and Horton [3] observed for ‘long’ separation bubbles. These similarities suggested that ‘long’ and ‘short’ bubbles were in fact steady and unsteady bubbles respectively and the bursting was simply the “demise of

unsteady separation” [37].

The scope of these numerical studies was limited by their two-dimensionality. Transition is an inherently three-dimensional phenomenon and is not captured in a two-dimensional simulation. Alam and Sandham [39] compared two- and three-dimensional results to establish the relevance of two-dimensional results. They discovered that a two-dimensional bubble was 40% longer than the corresponding three-dimensional one and the minimum skin friction was half that of the three-dimensional case. This suggests that transition to turbulence causes a fundamental change in three-dimensional bubbles when compared to two-dimensional ones [39]. Recent three-dimensional simulations of laminar-turbulent transition in LSB highlight the importance of large-scale coherent structures for bubble size and the reattachment process [40, 41, 28]. The breakdown to turbulence during the laminar-turbulent transition process in boundary layers, mixing-layers and bluff body wakes is often preceded by the formation of large-scale coherent structures. Two-dimensional spanwise vortices are the most prominent coherent structures forming during transition in these shear flows and two-dimensional processes may strongly affect the reattachment following transition.

When a boundary layer separates from a surface, the free shear layer that is formed as a result exhibits local velocity profiles with inflection points, which are known to be inviscidly unstable [42]. The separated shear layer in the laminar part of an LSB can therefore act as an amplifier and transition takes place owing to the non-linear breakdown of amplified disturbances [42]. It is also important to distinguish whether the transition is driven by a convective instability, where disturbances grow in space, or an absolute instability, where disturbances grow in time. Nieuw [43], by performing a wave packet analysis on a backward facing step, showed that for velocity profiles with very strong reverse flow of more than 20% an absolute instability did seem to be present. Hammond and Redekopp [44] found an even more severe 30% was necessary. Alam and Sandham [39] used the same analysis method as Nieuw [43] to confirm the theoretical possibility of an absolute instability and in agreement with Nieuw, they concluded that a reverse flow of roughly 20% was required to sustain an absolute instability at typical Re_c associated with LSB. However, values of only 4% to 8% were predicted in their DNS results and a maximum of just 3% was found by Yang and Voke [42], who used LES to study separation and transition. This led Alam and Sandham [39] and

Marxen *et al.* [40] to conclude that an LSB was convectively, rather than absolutely, unstable. This idea was also supported by Spalart and Strelets [45].

Despite agreeing on the instability mechanism, the results of Alam and Sandham [39] and Spalart and Strelets [45] displayed different breakdown behaviour. Alam and Sandham [39] introduced disturbances into the boundary layer upstream of separation to simulate naturally occurring Tollmien-Schlichting (T-S) waves. They then observed spatial amplification of these disturbances followed by transition via the breakdown of Λ -vortices associated with boundary layer transition. Conversely, in the study of Spalart and Strelets [45] upstream disturbances were not introduced but transition still occurred. They did not observe any Λ -vortices and instead of the relatively slow transition associated with them, they found the three-dimensionality to set in rapidly. This led Spalart and Strelets to conclude that entry-region disturbances (T-S waves) could not be the cause of transition. This was supported by Wilson and Pauley [46] who also observed a separation bubble to form in the absence of input disturbances in their LES simulation on a flat plate. From their frequency spectrum in the leading edge of their LSB, they found both a low frequency peak and a higher frequency peak. With increasing streamwise distance from the leading edge, the low frequency peak loses energy and the higher frequency peak becomes more dominant. The higher frequency component is associated with vortex shedding while the low frequency one is associated with a phenomenon known as flapping which has been observed to occur in shear layers in the absence of upstream disturbances.

In an attempt to resolve these disagreements Jones *et al.* [47] performed a DNS study of the development of a LSB on an aerofoil with and without boundary layer forcing. They found that the forcing magnitude had an effect of the size of the separation bubble but nevertheless, transition still occurred upon removal of forcing. Haggmark *et al.* [48] found that instability waves initially resemble T-S waves but that they change upon entering the separated region. Marxen *et al.* [40] agreed with this by stating that the first part of an LSB is dominated by two-dimensional T-S waves but noted that further downstream the instability assumes the characteristic shape of an inviscid free shear-layer instability.

This leads to the current understanding that following separation, the inviscidly unstable separated shear layer amplifies Kelvin-Helmholtz (K-H) instabilities, which arise from the

distortion of an interface between two fluids due to the shear forces between them [42]. This leads to the development of three-dimensional vortices in the shear layer and hence a breakdown to turbulence. The T-S waves, which are amplified in the attached flow do not directly cause transition but their magnitude will affect the size of the bubble. If input disturbances are absent then a so-called ‘flapping’ of the boundary layer will occur.

A further observation made by Jones *et al.* [47] was a slight increase in C_L when the forcing was removed. This was due to the increased length of the pressure plateau but was far outweighed by the increase in C_D attributed to the increase in length of the pressure plateau. The net result was that the lift-to-drag ratio dropped nearly 20% when forcing was removed. It appears that the presence of forcing significantly improves the aerodynamic performance of the aerofoil while requiring little energy input — an observation that introduces the concept of laminar boundary-layer control.

2.3 Three Dimensional Separation

The separation from aerofoils, as discussed in this review, is conventionally approximated with a uniform separation front [49, 50]. However, as argued by Ragni and Ferraira [49], this assumption is perhaps too simplistic. Moss and Mordin [51] were among the first to observe that a steady 2D separation pattern was not possible to obtain in reality. Instead, it has been discovered that when separation occurs from the trailing edge of a wing, the flow is reorganised into three dimensional cellular patterns [52]. Winkelmann and Barlow [53] carried out an oil-flow visualisation of the separated flow over a wing and described that these mushroom-shaped features were the result of a pair of counter-rotating vortices with axis perpendicular to the wing planform. By testing various aspect ratios they found that the size of the individual cells was independent of aspect ratio, but that the number of cells present increases with integer increments.

It is understood that the physical origin of stall-cells is not related to the separation typically occurring at the wing tips [49, 54, 55]. Several hypotheses as to their origin have been proposed over recent years. Weihs and Katz [52] suggested they are the result of a Crow-type

instability, where two counter-rotating vortices mutually interact amplifying small oscillations in their shape, up to the formation of a chain of vortex rings. However, the inviscid nature of Crow instabilities and the presence of a viscous boundary layer, means this cannot fully explain the pattern [49]. More recently their onset has been attributed to both amplifications of a 2D instability in the flow separation line [56] and 3D spanwise instabilities [57].

In addition, there is no agreement yet on their presence at low Re_c . Broeren and Bragg [58] observed no spanwise time fluctuations in time at $Re_c = 3 \times 10^5$ while Zarutskaya and Arieli [59] did find evidence of stall cells at the same Re_c . Note that this is 6 times greater than the Re_c studied in this thesis. The closest literature found to the current test conditions was from Karthikeyan *et al.* [60] who studied flow over a NACA 4415 at $\alpha = 6^\circ$ with leading edge tubercles. They found that the tubercles induced three-dimensional ‘bubble pockets following separation but their baseline case (i.e. without tubercles) showed no such three-dimensional features.

It is clear that this is a rich area of research and considerable disagreement remains over subject of stall cells. A dedicated study of the phenomenon would be required to fully understand their role and implications. Therefore this thesis, like the other literature cited within, will not investigate the three-dimensionality of separation but rather consider separation control in terms of reducing the size of recirculation.

2.4 Flow Control

The ability to enforce a desired change to a flow field has the potential to save billions of dollars in annual fuel costs thus achieving more economically viable and environmentally friendly systems [61]. For this reason, the field of Flow Control has received a great deal of attention – according to Gad El Hak it “undoubtedly accounts for the subject being more hotly pursued by scientists and engineers than any other topic in fluid mechanics [61].” Flow control can trace its routes back to Prandtl’s introduction of the boundary layer in 1904. In his speech at the Third International Congress of Mathematicians, Prandtl introduced the now widely accepted concept of the boundary layer and described the use of suction to delay

its separation from a cylinder [61]. Since then a variety of methods have been investigated, with varying degrees of success.

A control method is chosen based on the desired change in the flow field. Typically the end goal is to reduce the drag, enhance lift, suppress flow-induced noise, increase mixing of mass, momentum, or energy, or a combination of the above. To achieve any of these aims, transition from laminar to turbulent flow may have to be delayed or advanced, flow separation may have to be prevented or provoked, or turbulence levels may have to be suppressed or enhanced. None of these goals is particularly difficult if taken in isolation but they are often adversely interrelated when considered in combination [61]. The challenge then is in achieving a goal with a minimum of possible side effects.

In the previous section in this chapter it was confirmed that the poor performance of aerofoils at low Re_c is a result of boundary layer separation. Given an imposed pressure field, the principle strategy in separation control is to add momentum to the very near-wall region. Passive flow control techniques, which are more easily implemented than active control, have proved to be very effective at increasing the near-wall momentum by introducing inert devices.

2.4.1 Passive Control

A popular passive method is the use of Vortex Generators (VG). These are simply small aspect ratio aerofoils mounted normally to the suction surface ahead of the flow separation point [62]. Tip vortices, shed from the tips of the VG, entrain high momentum fluid from the free stream closer to the solid surface as they convect downstream, re-energizing the near wall flow. Conventional VG have a height of the same order of magnitude as boundary layer thickness. Low-profile VG have a smaller height, such that they remain submerged in the boundary layer [63, 62].

The first use of such a method on an aerofoil at low Re_c was by Kerho *et al.* [64] when they investigated low-profile VG on a Liebeck LA2573A aerofoil. Their aim was to reduce or eliminate the separation bubble and consequently reduce the drag. They investigated different geometries of the VG and found a maximum of 38% drag reduction at the most favourable

parameters. Manolesos and Voutsinas [65] also investigated VG on aerofoils, but with the height of the VG equal to the boundary-layer thickness. Their presence leads to lift enhancement at higher values of α but a drag penalty for $\alpha < 6^\circ$. Considering this, it can be concluded that the effectiveness of VG depends on their height relative to local boundary layer thickness. This is supported by the results in Lin's 2002 review of VG [63]. Moreover, since this is a passive device, the height is prescribed meaning that once mounted on an aerofoil surface, a passive VG will only improve performance while the boundary layer thickness is greater than its height. As a result, it turns out that VG are found to be best suited for applications where the separation locations are fixed so the generators can be placed in a location close to, but upstream of, the separation point [62].

Another popular passive control method is surface roughness. Perturbations in the laminar boundary layer resulting from the presence of surface imperfections can advance transition to turbulence thereby making the boundary layer resistant to adverse pressure gradients. In this way, the separation can be delayed or avoided [66]. Zhou and Wang numerically investigated the use of bumps in the leading edge surface of a SD7003 aerofoil at $\alpha = 4^\circ$ and $Re_c = 6 \times 10^4$. It was found that the large disturbances trigger vortex breakdown and the LSB are reduced or avoided. As a result, the pressure drag was significantly reduced resulting in an overall drag reduction. Also, Mueller and Batil [19] found that the C_L -vs- α slope at $Re_c = 40,000$ of a NACA 663-018 aerofoil with surface roughness elements at the leading edge had a higher gradient than that of the curve corresponding to their smooth surfaced aerofoil mentioned in Section 2.1.1.

However, experiments performed by Greenblatt and Wygnanski [67] on a different airfoil at $Re_c = 50,000$ found that 6 mm diameter by 1 mm high circular tabs were ineffective as boundary layer trips at such low Re_c resulting in little to no improvement when compared with their clean airfoil data. This suggests that this form of control at very low Re_c can be difficult to implement. Furthermore, the aforementioned study by Mueller and Batil [19] found that at higher Re_c , where laminar flow existed over a shorter portion of the surface, surface roughness also increased drag.

Therein lies the drawback with passive control. A technique that is effective at low Re_c would likely increase drag at higher (non-separating) Re_c [68]. Furthermore, as was seen in

Section 2.1.2, flow fields at low Re_c can take several forms, with the location of separation changing drastically with changes in α and turbulence intensity. Considering the fact that an aircraft would encounter a variety of flow conditions during a single mission and a passive device can only be optimised for a single flow field, it is inconceivable that such control strategies can have an positive effect at low Re_c aerofoils in practice.

2.4.2 Active Control

In contrast, active control approaches offer the significant advantage of being largely innocuous except when activated and therefore remove the drawback associated with passive control at off-design conditions. Traditionally, this form of control has been associated with the injection of fluid to, or the removal of a fluid from, a boundary layer. The first method proposed was suction, the basic principle of which is to remove decelerated fluid near a surface and draw the high-momentum free-stream fluid towards the surface [67].

Blowing has also proved a successful method for lift enhancement. This method uses a jet to increase the mean kinetic energy in the boundary layer so that the latter can advance without separation into adverse pressure region [12]. Wyganski and Newman [69] found that the entrainment of external fluid by the jet caused an effective increase in the aerofoil camber. Using momentum coefficient, C_μ , as a measure of actuator performance, they found the benefits of this method to be substantial if C_μ was large.

Both suction and blowing devices, while in principle being capable of improving performance, require complex auxiliary compressors and extensive plumbing systems for operation. The need for a large momentum coefficient means that the power consumption of these devices is large, offsetting the aerodynamic benefits [67]. This example highlights the main disadvantage of using active methods compared with passive — the addition of power.

2.4.3 Periodic Active Flow Control

The basis of periodic forcing for flow control was discovered by Schubauer and Skramstad [70] who introduced periodic perturbations in a laminar boundary layer to trigger a known

instability (the T-S wave).

More recently, the most widely researched periodic active flow control methods are Synthetic Jets (or Zero-Net-Mass-Flux jets) [71, 72, 73] and Direct-Barrier-Discharge (DBD) Plasma actuators [74]. In contrast to periodic-based control's origins, experimental investigations have shown that forced transition does not play a pivotal role in separation control. The formation of Large Coherent Structures (LCS) in the separated shear layer has been widely observed [71], with the periodic motion believed to be capable of accelerating and regulating their production. Moreover, when subjected to excitation, it is understood that these structures are the essential 'building blocks' of the mixing layer [67] and responsible for transposing momentum across it. The triggering of LCS is therefore an efficient method for control of mixing in the separated shear layer and consequently, periodic motion can increase the transfer of high momentum fluid, enhancing entrainment. This principle forms the basis of separation control by periodic excitation.

Since the effectiveness of the method is largely determined by the receptivity of the flow to the imposed disturbances, these have to be of the right scale and be introduced at the right location. The control authority of periodic excitation has been found to exhibit a highly non-monotonic variation with actuation frequency which suggests the presence of rich flow physics [71].

Benard and Moreau [75] — who experimentally investigated the use of DBD plasma actuators for separation control — found when activated in an initially quiescent flow field (i.e. with no flow), the actuator produced a large counter-clockwise rotating vortex. However, when the actuator was used in the actual flow, no such vortex was observed during the reattachment process. Instead an amplification of a clockwise rotating LCS was found. In addition, despite being capable of initiating a dramatic change in flow development, the velocity of the gas produced by the actuator alone was one order of magnitude lower than the external velocity. In agreement with what was stated above, this led the authors to conclude that the momentum transfer did not come from the actuator itself but rather the actuator acts as a catalyser, possibly reinforcing an already existing flow structure although they could not confirm this.

Postl *et al.* [28] — whose Direct Numerical Simulations investigated pulsed vortex generator jets (pVGJ) for separation control — also observed the production LCS and, owing to their fully three-dimensional data, were able to confirm the spanwise coherence of the LCS. The control mechanism associated with pVGJ was identified as being related to the formation of these spanwise LCS. This is in contrast with steady forcing, in which longitudinal vortices were responsible for enhanced entrainment [28]. The authors argued that the pVGJ are able to generate the spanwise LCS as a result of the hydrodynamic instability mechanisms of the separated shear layer. Since an inflection point is present in the velocity profile, the flow is inviscidly unstable (the K-H instability). More recent studies by Marxen *et al.* [76], Sato *et al.* [74] and Buchmann *et al.* [72] also observe spanwise LCS and attribute them to the K-H instability mechanism.

When steady forcing and pulsed forcing were compared, Postl *et al.* [28] found that pulsed blowing is significantly more effective when the same momentum coefficient was used for the actuation. This agrees with the findings of Seifert *et al.* [12] who investigated the possibility of superimposing a periodic motion on top of the steady blowing. It was discovered that a steady blowing with $C_\mu = 0.1$ was required to eliminate pressure drag in the operating range of $1.0 < C_L < 1.6$. However, by modulating a steady jet with $C_\mu = 0.008$ with a periodic jet with $\langle C_\mu \rangle = 0.008$ and a non-dimensional frequency, f^+ , of 2, a similar pressure drag distribution was obtained. The maximum lift coefficient attained in both cases was also similar. This corresponds to a total saving of 84% in C_μ and the saving in the energy is much more significant [12]. In other words, a very low level of steady blowing, when combined with a periodic modulation, could achieve the same degree of control authority as a much larger level of steady blowing alone, but with just a fraction of the power requirement. Since the control authority of the method is largely determined by the receptivity of the flow to the imposed disturbances, they have to be introduced at the right location [12, 71].

Despite the improved effectiveness and efficiency of periodic-based active control methods with respect to both steady active control and passive control methods, it has been difficult to apply the control to actual applications owing to the complexity of the control devices [74]

However, the development of new actuation devices and material systems has enabled novel approaches to periodic flow control to be explored. Munday *et al.* [77] 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 enabling their practical use and maximizing their possible aerodynamic gains. Munday *et al.* [77] performed both static and dynamic morphing tests. While their static tests did not prove very successful, dynamic actuation was found to significantly reduce flow separation.

2.5 Objectives of the Thesis

This thesis is concerned with low Re_c flows around aerofoils and its control by periodic-based active flow control. Much of the current research in this area has been based on synthetic jets and plasma actuators but in this thesis an alternative approach to periodic forcing will be explored, based on dynamic surface morphing. It is believed that this concept has the significant advantage over others of being simpler to implement. Furthermore, the distributed nature of the motion will negate the requirement for the actuator to be placed in a particular position, making it more flexible to changing flow conditions highlighted in this chapter.

The research has the following objectives:

1. To exploit the properties of the MFC actuator and fabricate a physical aerofoil model that is capable of dynamic shape change. Three-dimensional, time-resolved measurements of the surface motion are performed to quantify the ability of the MFC to perform the required shape changes (*Chapter 3*).
2. To determine experimentally the effectiveness of this concept as a flow control method. Both unactuated (baseline) experiments and actuated (controlled) experiments are performed (*Chapter 5*).
3. To analyse in detail the time-averaged and instantaneous flow around the aerofoil model at low Re_c conditions using Direct Numerical Simulations (*Chapter 6*).

4. To investigate the effect of surface actuation and shed light on the interaction between periodic motion and the flow structures (*Chapter 7*).
5. To highlight the major findings are highlighted and make recommendations for future work (*Chapter 8*).

Chapter 3

Physical Model: Fabrication, Actuation and Testing

A physical model capable of performing dynamic surface morphing was fabricated. The primary purpose of this model was to be used for experimental aerodynamic testing. However, it also provides an insight into the practicality of implementing the proposed flow control method. The following section explains how the model was fabricated and how the surface motion is implemented. Furthermore, the results of a series of tests are presented in which the surface motion was examined.

3.1 Aerofoil Model Fabrication

The physical model is based on a NACA 4415 aerofoil section, which is known to suffer from laminar separation at low Re_c [77]. The leading edge, trailing edge and pressure surface were machined from PVC to true NACA 4415 coordinates with a tolerance of ± 0.1 mm (the technical drawings can be seen in Appendix A.1). The suction surface was fabricated separately to accommodate the desired deformations. To achieve a deformable surface, two Macro Fiber Composite 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 then placed over the skin while the epoxy set to remove any pockets of air between the Titanium

and the actuators thus ensuring a strong, clean bond. This was left for twelve hours to set in a specially designed mould with appropriate curvature for the aerofoil's suction surface. Electrical connections were soldered and then insulated using a fast drying epoxy. The resulting skin can be seen in Figure 3.1a.

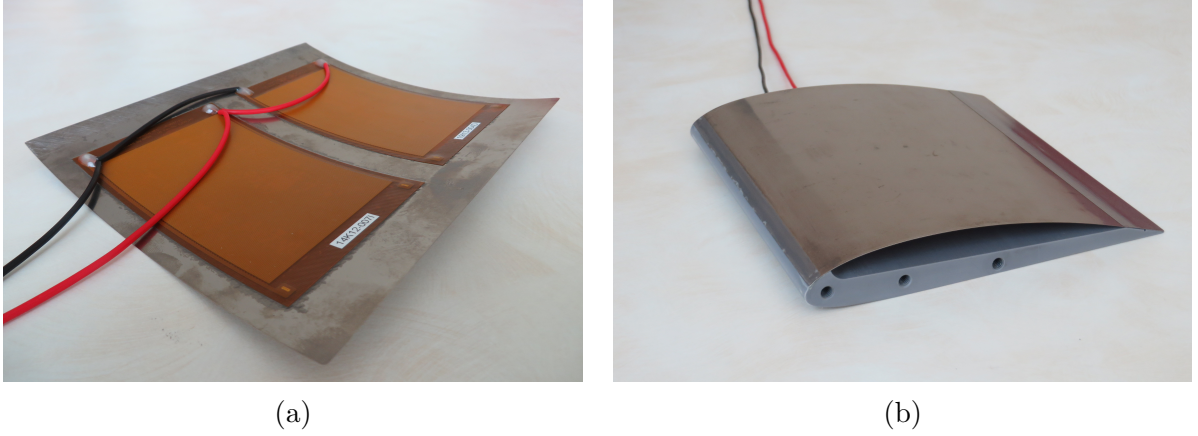


Figure 3.1: Physical model for wind tunnel testing: a) Underside of the suction surface; b) Complete model.

The skin is rigidly attached to the body of the aerofoil at the leading edge and forms the suction surface. At the trailing edge a thin slide joint is used to allow for small displacements in the longitudinal direction that occur when the surface is deformed. The fully-formed model has a chord, c , of 150 mm and span, s , of 158 mm and can be seen in Figure 3.1b. The MFC patches have an active length of 85 mm, which is centred at the mid-chord location. As a result, the active portion of the MFC occupies a region from $0.22c - 0.78c$. Figure 3.2 shows the cross-sectional view of the non-actuated model with true NACA 4415 coordinates plotted with a dashed line for comparison. Separately fabricating the suction surface appears to have resulted in the aft $c/4$ region being slightly thicker — at $0.6c$ the model thickness is $\approx 3\%$ greater than the reference — however the model appears to be a good representation of a true NACA 4415.

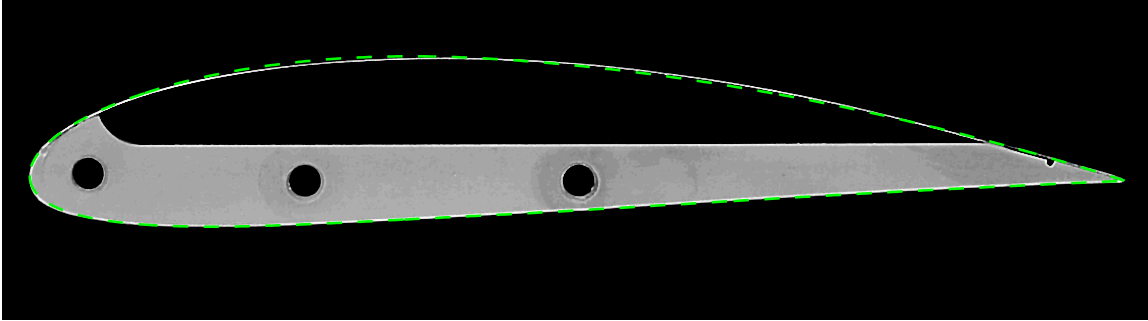


Figure 3.2: Physical model compared with NACA 4415 coordinates (dashed line).

3.2 Actuation Procedure

The surface morphing is provided by piezoelectric patches known as Macro Fibre Composites (MFC), which are bonded to the underside of the suction surface. The patches have three main components: 1) a sheet of aligned fibres of piezoelectric material, 2) a pair of thin polymer films etched with a conductive electrode pattern and 3) an adhesive matrix material [13]. A piezoelectric material couples mechanical and electrical fields so that the piezoceramic fibres will deform when they experience an electrical field [78]. Within an MFC the electrical field is provided by the electrodes so — owing to the strong bond between the MFC and the Titanium skin — supplying the electrodes with a voltage will cause the aerofoil's surface to deform. The voltage is applied 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 (Direct Current) to 10 kHz AC (Alternating Current) 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.

In this thesis, the non-actuated surface is considered as the origin and a displacement in the outward direction is considered positive.

3.2.1 Effect of Non-linear Actuators

Owing to their high electro-mechanical coupling factor and piezoelectric coefficients, piezoceramic materials are commonly employed to form actuators [79]. Specifically, MFC use Lead Zirconate Titanate (PZT) because of its outstanding piezoelectric properties. The drawback of PZT is that as well as being piezoelectric it is also ferroelectric. This is a result of requirements on the electrical polarisation for macroscopic piezoelectric effects [80] and means that MFC exhibit non-linear behaviour such as hysteresis and creep. A stark illustration of this can be seen in Figure 3.3.

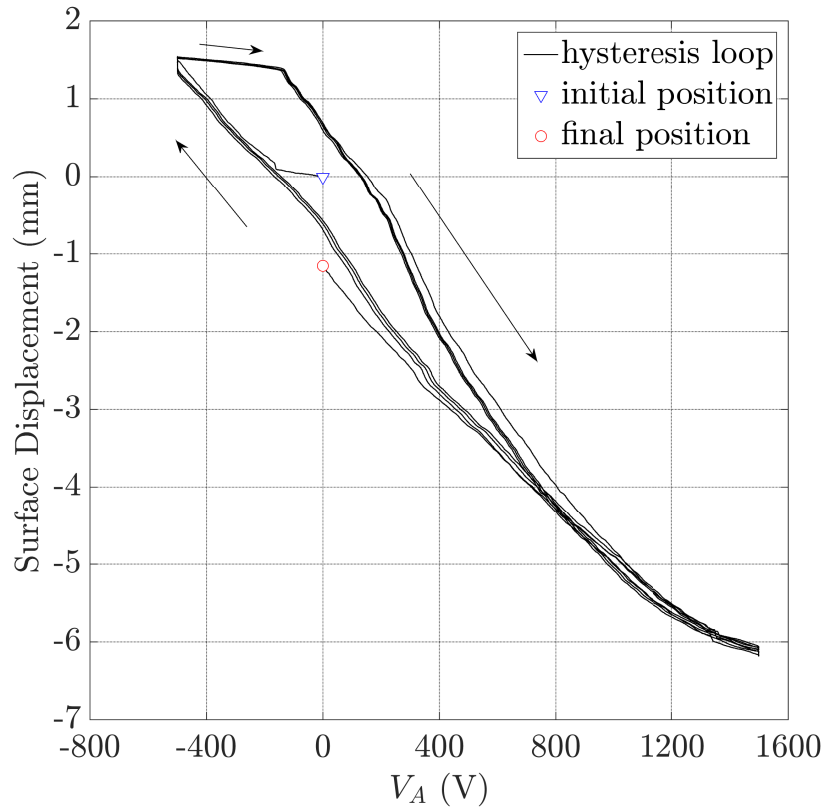


Figure 3.3: Surface Displacement-vs-Voltage for 4 cycles from maximum to minimum MFC voltage bounds.

The MFC were driven with the voltage signal shown in Figure 3.4, which cycled through their operational range four times, pausing at the maximum positive and negative voltages for one second as it did so. The surface displacement was determined from the time-resolved distance

between the surface (at $x/c = 0.4$, $z/s = 0.5$) and a laser sensor's receiver (to be introduced in Section 3.3.1). The origin in Figure 3.3 (shown by a triangle) denotes the distance of the non-actuated surface from the laser sensor before the cycles began (i.e. at 0 V). It is clear from the figure that the position of the surface at 0 V varies significantly depending on how 0 V is approached. Throughout the four cycles the 0 V position varies by approximately 1.2 mm, from +0.6 mm when the voltage grows from negative to positive and -0.6 mm when the voltage is reduced from positive to negative. Furthermore, the final resting place of the surface (shown by a circle) – which follows a more gradual approach to 0 V than during the loops to avoid an abrupt halt – is a further 0.5 mm away from the origin. This means the non-actuated surface is approximately 1.1 mm further away from the sensor post-cycle compared to pre-cycle.

Another non-linear effect noticeable in Figure 3.3 is creep. The voltage remains at the maximum positive and negative values for one second during each cycle but the surface is observed to move 0.13 mm and 0.02 mm during this time at -500 V and 1500 V respectively. Since these piezoceramic non-linearities affect the surface displacement, they change the overall unactuated aerofoil geometry.

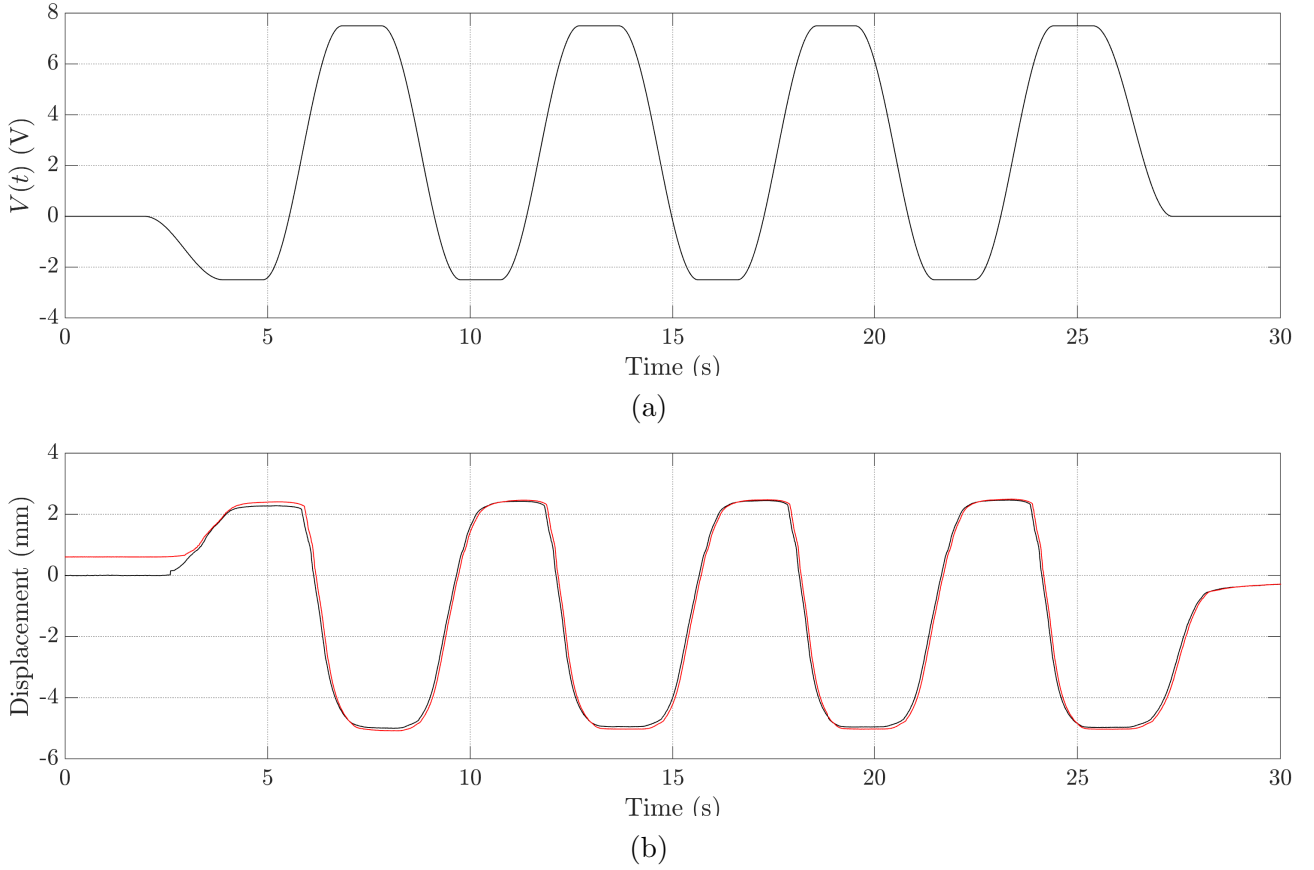


Figure 3.4: Four cycles through the operational range of the MFC: a) input signal; b) two different examples of surface response (note the different starting positions converge to the same value).

Figure 3.4b shows two separate examples of the time-resolved displacement of the surface during the hysteresis cycles (both acquired at $x/c = 0.4$ and $z/s = 0.5$). The measurement point, despite being located at the same position on the surface in both experiments, is displaced by roughly 0.6 mm at the beginning of the cycles. Such an occurrence could result from a different approach to 0 V in a previous experiment, such as a different input frequency, or because of surface creep if the MFC have been left for a period of time without actuation between experiments. As the cycles progress, the position of the measurement points converge. Following the 4 cycles both signals are very close to each other (within the resolution of the laser sensor) and therefore the 0 V position of the surface can be made independent of the preceding history. The drawback is that, as Figure 3.5 shows, the post-cycle surface geometry is slightly thinner than the NACA 4415 geometry presented in Figure 3.2.

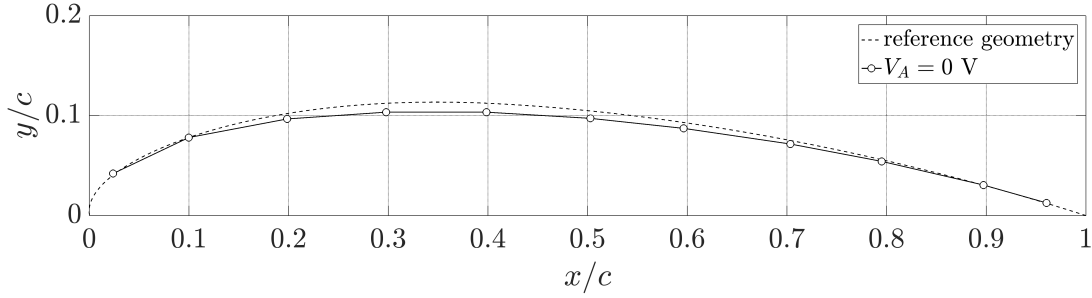


Figure 3.5: Post-cycle aerofoil geometry compared with true NACA 4415 geometry.

A NACA 4415 has maximum thickness of 15% of the chord at $x/c = 0.4$; by contrast, the model has a thickness of 13.6% at the same location. As a result of the non-linearity of the actuators, the non-actuated aerofoil model is therefore 1.4% thinner than a true NACA 4415. Also, by investigation of Figure 3.2, the leading edge has a marginally greater radius. The coordinates for the tested aerofoil in Figure 3.5 were simulated using XFOIL's viscous solver and the resulting C_P distribution is compared with that of a true NACA 4415 for $Re_c = 50,000$ and $\alpha = 0^\circ$ in Figure 3.6.

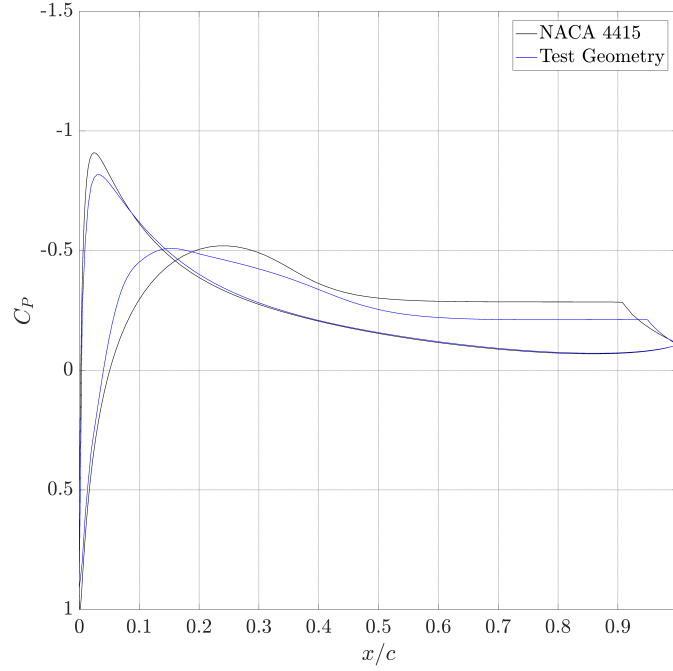


Figure 3.6: Comparison of C_P distributions for the tested aerofoil geometry and a NACA 4415, obtained from XFOIL data.

Since the aerofoil is thinner (and the aft camber is therefore reduced) the suction of the aft portion of the tested aerofoil is reduced compared with a true NACA 4415. Nevertheless, the size of the pressure plateau is more-or-less the same. Also, since the leading edge is slightly more rounded (i.e. greater forward camber), the suction peak is further upstream, although it remains roughly the same the magnitude. These differences will affect the aerodynamic forces generated by the aerofoil but the general flow characteristics remain the same, i.e. laminar separation followed by transition towards the trailing edge without reattachment. This is supported by Figure 3.7 which shows the C_L vs. α curves for the tested geometry a NACA4415. It can be seen, for pre-stall values of α , the curves are simply offset from each other – the tested aerofoil giving lower C_L at each α as a result of the reduced aft suction.

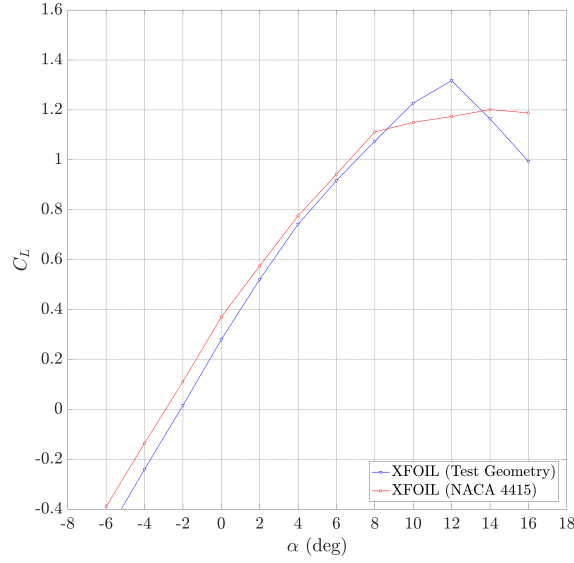


Figure 3.7: Comparisons of the C_L curves obtained from XFOIL.

It is essential that the aerofoil shape is the same between experiments and as such, these hysteresis cycles must be included in the experimental preparation of the model. The procedure to implement this is explained in the next section.

3.2.2 Input Signal

Static deformations were achieved by driving the MFC with a signal of constant voltage. The amplifier can also accept alternating signals up to 10 kHz so with a sinusoidal input voltage signal the surface will oscillate. The input signal was provided as a user-specified text file containing discrete voltage values via an analogue-to-digital converter. This allowed for more flexibility in the signal's form, which is necessary if, for example, the hysteresis cycles are to be included. An example of an input signal for dynamic actuation at 2 Hz is shown in Figure 3.8.

The model will first go through the hysteresis cycle to ensure that the aerofoil geometry is always the same. Following these cycles, the signal will oscillate at the desired frequency. The gradual increase and decrease of the sinusoidal amplitude at the beginning and end of the actuation was included to make the transition to an oscillating surface as smooth as possible in an effort to reduce possible damage to the model. Any data acquisition, whether

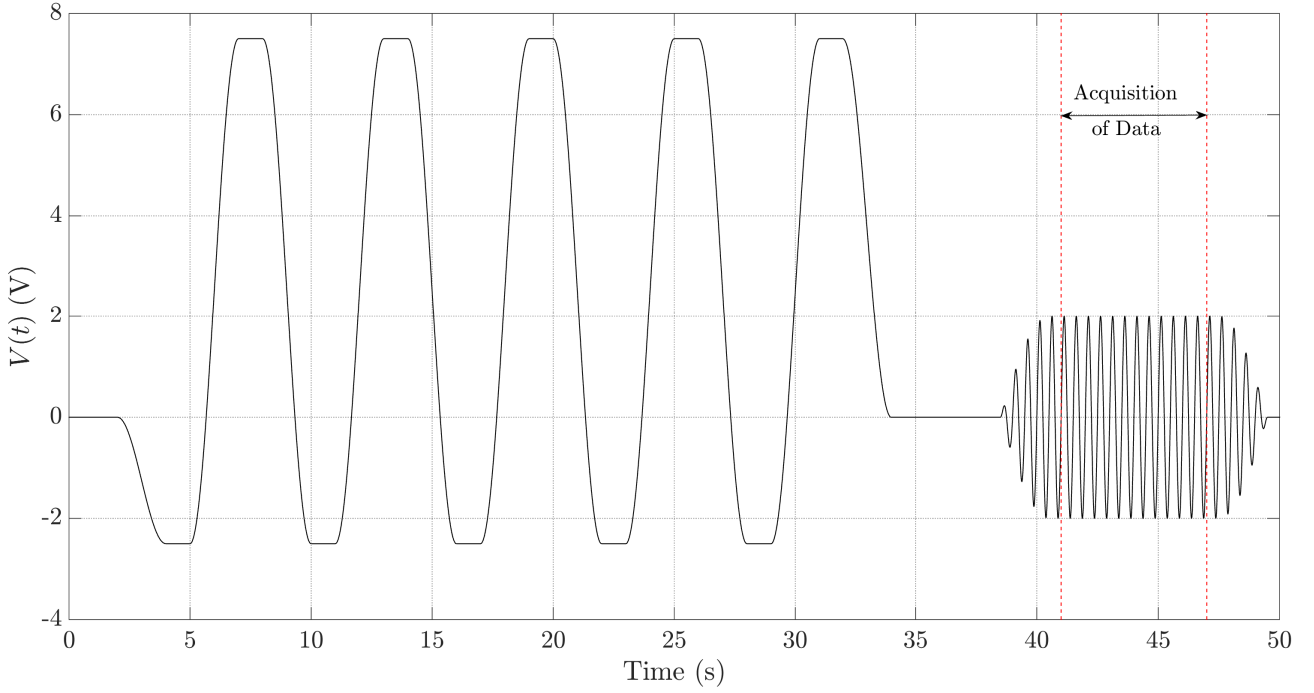


Figure 3.8: Sample of complete input signal — initial model preparation and oscillating phase.

aerodynamic or structural, was obtained during the period where the surface is oscillating with a constant amplitude. This section of the input signal is described by Equation 3.1.

$$V(t) = V_A \sin(\omega t) \quad (3.1)$$

where V_A is the amplitude, measured in Volts, and the angular frequency, ω , is defined as $2\pi V_f$ where V_f is the actuation frequency, measured in Hertz.

The input signal was controlled and monitored using a National Instruments LabVIEW Real-Time Module. The module can only follow a sampling frequency of up to 1.25 kHz and consequently this was the maximum sampling rate used to generate the input signal. This sampling frequency is between 5.2 and 62.5 times higher than the frequency of the input signal, which ranges from 10-120 Hz, and therefore all frequencies were well captured and no aliasing appeared. This discrete signal was then converted to analogue prior to amplification by a National Instruments M-Card which has 16-bit analogue output. With the maximum, pre-amplified, sinusoidal voltage range being ± 2.5 V (5 V peak-to-peak) a 16-bit card has a

resolution of 7.6×10^{-5} V ($5/2^{16}$ V) — more than adequate to correctly perform the conversion.

3.3 Measurement of Surface Displacement

Two techniques were used to measure the surface displacement: a laser displacement sensor and photogrammetry. The setup of the laser displacement sensor and photogrammetry, along with some background to the photogrammetry process, is explained below.

3.3.1 Laser Displacement Sensor

A series of tests were carried out to understand the response of the skin at different V_A and V_f . A Micro-Epsilon optoNCDT 1710-50 laser displacement sensor was used to measure the deformation of the skin at a point. This unit has a measuring resolution (i.e. the smallest displacement it can measure) of $50 \mu\text{m}$ ($\pm 5 \mu\text{m}$) and is optimally positioned at a distance of 550 mm to 600 mm from its target. This range enables it to be placed outside the wind tunnel thus allowing measurements of the skin deformation in a flow as well as in still air (see Figure 3.9a). Markings, as seen in Figure 3.9b, were made at $0.25c$, $0.4c$, $0.5c$ and $0.7c$ on the aerofoil surface as targets for the measurements. These were acquired at 1.25 kHz, providing time-resolved information of the surface motion.

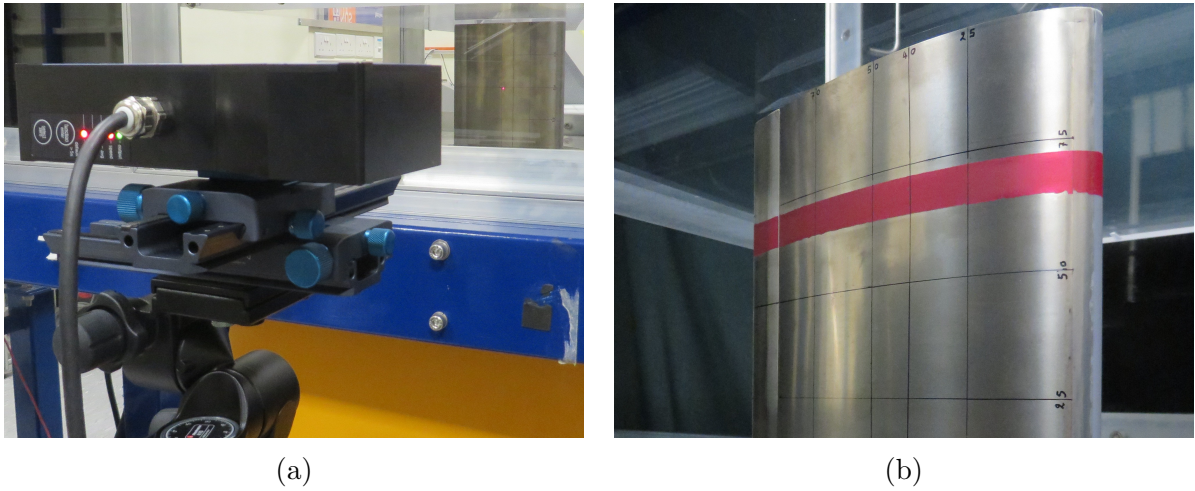


Figure 3.9: Setup of Laser Displacement sensor for surface measurements: a) the measuring unit placed outside the tunnel; b) the mounted aerofoil with surface markings at measurement positions.

3.3.2 Photogrammetry

The laser displacement sensor can only provide information at a single point in one dimension. This limits the amount of analysis that can be done on the surface motion. Photographs can provide a view of the whole surface and, if these are captured at high speed, a time resolved surface motion can be obtained. A Photron FASTCAM SA-Z capable of producing mega-pixel (1024×1024) images at up to 20,000 frames per second (fps), was used to capture snapshots of the aerofoil undergoing both static and dynamic actuation. Examples of such photographs are provided in Figure 3.10.

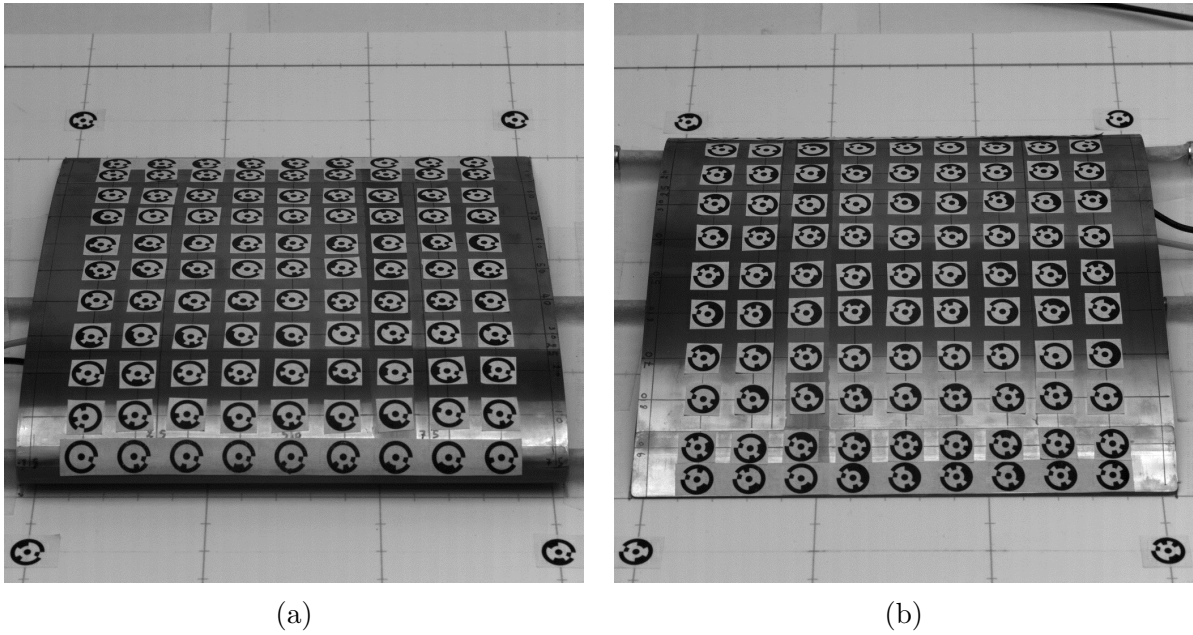


Figure 3.10: 12-bit monochrome photographs captured by the Photron SA-Z fastcam.

Photos such as those in Figure 3.10 can provide information of the static surface deformations since the changes in shape would be visible. Furthermore, when the frames captured at high speed during dynamic actuation are assembled into an animation, information about how the surface behaves dynamically can be obtained. Such information can be revealing but it is only qualitative, which again limits the analysis. To measure the surface motion in time and three-dimensional space, photogrammetry is used to extract quantitative information from the photographs. This was performed using ‘Photomodeler,’ a proprietary software package published by Eos Systems Inc.

Feature Extraction

The images in Figure 3.10 were captured by a CCD sensor with 1 mega-pixel resolution and 12-bit depth. As a result, they contain a huge amount of information, much of which is not relevant to the measurements. Therefore, the first stage of the photogrammetry process is to reduce the amount of data, whilst preserving the relevant information. To achieve this, 99 binary coded targets were placed on the surface of the aerofoil (9 across the span and 11 in the streamwise direction) which can be seen in Figure 3.10. Rather than interpreting the entire image, Photomodeler can just track the motion of these targets, which are individually

distinguished by their binary coding. Picking out these targets was achieved by a pre-processing step called ‘feature extraction.’

The 12-bit CCD sensor in the camera stores the monochrome image as a 1024×1024 array of light intensity values from 0 (black) to 4096 (white). The backgrounds of the coded targets are white while targets themselves are black. When considering the 12-bit depth of these images, the presence of a target would appear as a large step change in the values contained in the light intensity array. Therefore, the coded targets can be extracted by searching the arrays for large intensity gradients and considering the locations of these large gradients as the target’s edges. This is known as edge detection and Photomodeler employs a particular edge detection algorithm known as the ‘Canny Algorithm’ [81]. Simple application of MATLAB’s in-built ‘canny’ function, shown in Figure 3.11, displays the effect of this algorithm on the photos in Figure 3.10.

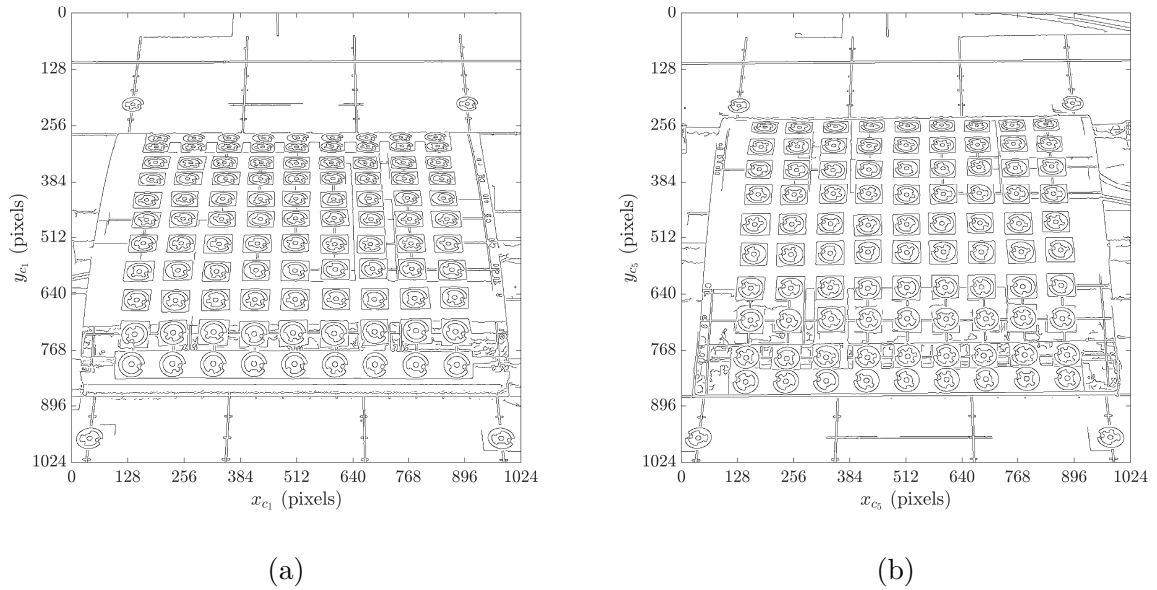


Figure 3.11: Image data reduced by Canny edge detection algorithm: a) leading edge; b) trailing edge

It is clear that the amount of data in Figure 3.11 is greatly reduced when compared to Figure 3.10 but valuable information has been retained since the coded targets can still be seen – the edges have been extracted. However, it is also clear that extraction by light intensity gradients has also caused the extraction of many other ‘edges’ that do not belong to

the coded targets. The most troublesome are along the span in the region around $y_c = 768$ pixels in both images. A great number of edges appear in and around targets in this region which is undesirable as it will inhibit Photomodeler's ability to correctly interpret the target's binary code or, in the worst case, cause it to miss the target altogether.

Comparing Figure 3.11 with Figure 3.10 it can be concluded that these spurious edges are a consequence of the shiny Titanium surface. A smooth finish is required to ensure surface roughness does not interfere with the development of the laminar boundary layer but that has resulted in a highly reflective surface. Reflections also cause step changes in light intensity and these have been interpreted as feature edges by the Canny algorithm.

In Photomodeler, the user can define a threshold in the Canny Algorithm that is used to refine the search for edges. Figure 3.12 shows a subsection of the leading edge photo with changing threshold values.

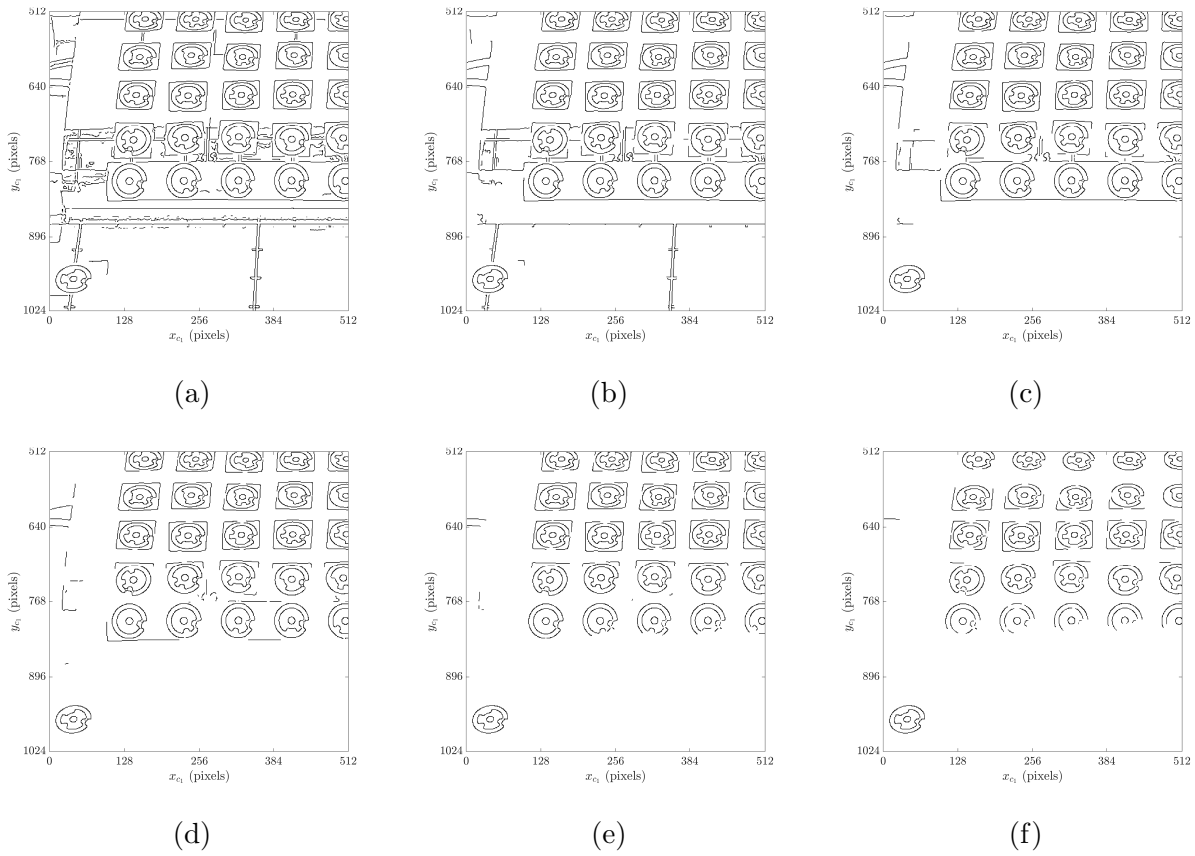


Figure 3.12: Canny edge detection algorithm with changing threshold values [min, max]: a) [0.05 0.1]; b) [0.15 0.2]; c) [0.25 0.3]; d) [0.35 0.4]; e) [0.45 0.5]; f) [0.55 0.6].

As the threshold values are increased, the edges resulting from reflections significantly decrease. However, when the threshold values are greater than $[0.35, 0.4]$ some of the features of the coded targets are lost. Since the aim of edge detection was to reduce the data in the images without losing relevant information, this is undesirable. For this reason threshold values of $[0.35, 0.4]$ were used when processing the images.

Noise in the images can also be improved by applying a Gaussian filter prior to processing. As with the threshold, the standard deviation, σ , in the Gaussian filter is a trade-off between reducing noise and losing relevant information. Figure 3.13 shows the same subsection as in Figure 3.12, with the chosen threshold values included, with increasing values of σ .

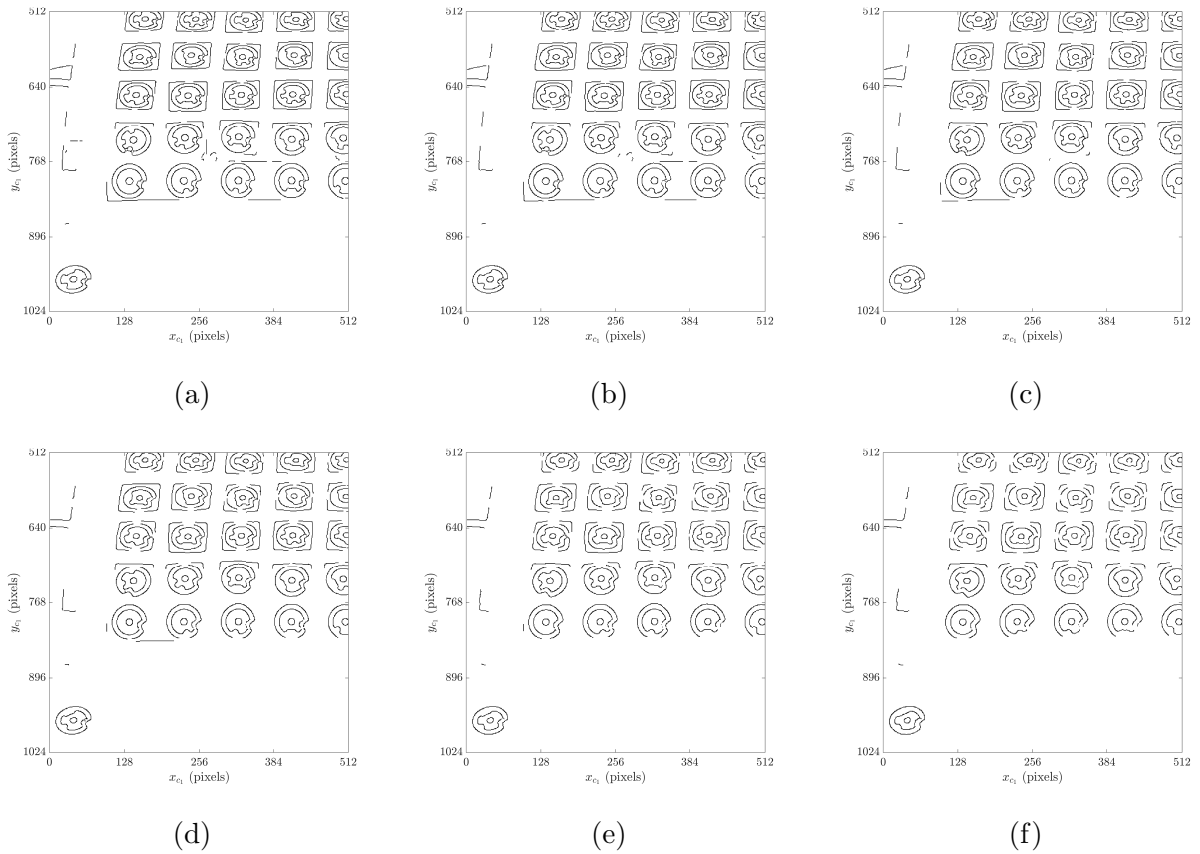


Figure 3.13: Canny edge detection algorithm with constant threshold value $[0.35, 0.4]$ and different filter, σ : a) $\sigma = 0.5$; b) $\sigma = 1.0$; c) $\sigma = 1.5$; d) $\sigma = 2.0$; e) $\sigma = 2.5$; f) $\sigma = 3.0$.

Spurious edges disappear with increasing σ , although at the higher values there is a noticeable distortion of the targets. For this reason a value of σ equal to 1.5 was used in the processing.

This pre-processing step, performed in MATLAB, enabled appropriate parameters for the

Canny algorithm to be determined. These were then used in Photomodeler to make the processing more efficient.

Camera Calibration

To convert data from the CCD sensor to meaningful 3-dimensional information, the data must be mapped from the sensor's pixel coordinates to the coordinate system of the real world, via an intermediate mapping to $\mathbf{x}$, the 2D Cartesian system of the image plane. There are two real world coordinate systems involved; one is with reference to the camera, $\mathbf{X}_c$, and one is with reference to the physical aerofoil model, $\mathbf{X}$. It is the latter of the two that is of interest. The process of mapping CCD sensor coordinates to the physical aerofoil coordinates is described in Appendix A.2. See Figure 3.14 for the definition of the system of axes.

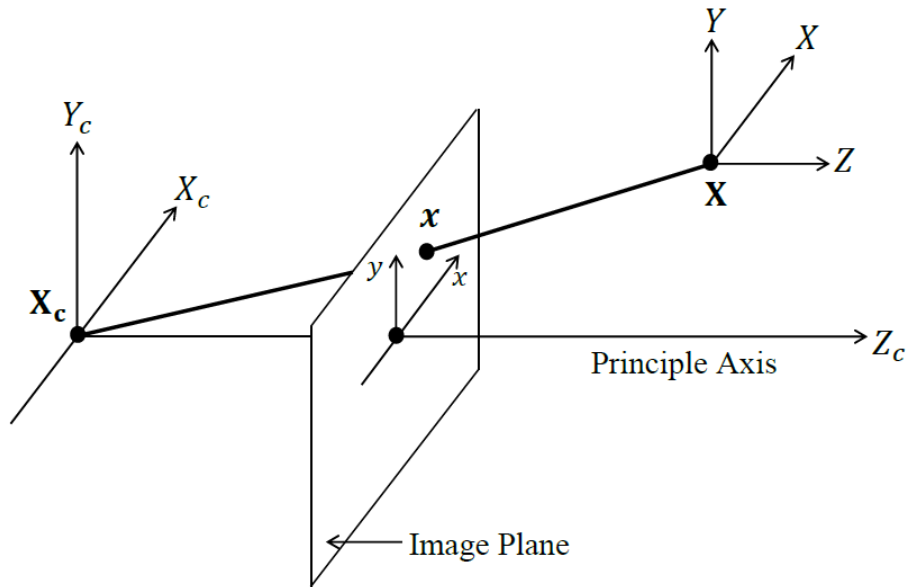


Figure 3.14: Pinhole camera geometry describing the relationship between the coordinate systems involved in photogrammetry.

Image Reconstruction

It is not possible to obtain 3D information from a single photo, since no information about depth is provided. The coded targets in Figure 3.10 are captured by the software which can solve for a particular light ray on which these points lie, but it cannot determine how far along

that ray they are. If a point is captured in more than one photo, an algebraic representation of its corresponding light rays can be found on each image. Solving for the intersection of these rays then provides the 3D information. It was therefore necessary to ensure that each one of the 103 points in this experimental setup was imaged multiple times from different angles. For this reason, 8 separate sets of images were captured at angles roughly 45° apart from each other. These are shown in Figure 3.15. The four points in the scene not on the aerofoil are used in the post processing to provide the world coordinate system, $\mathbf{X}$.

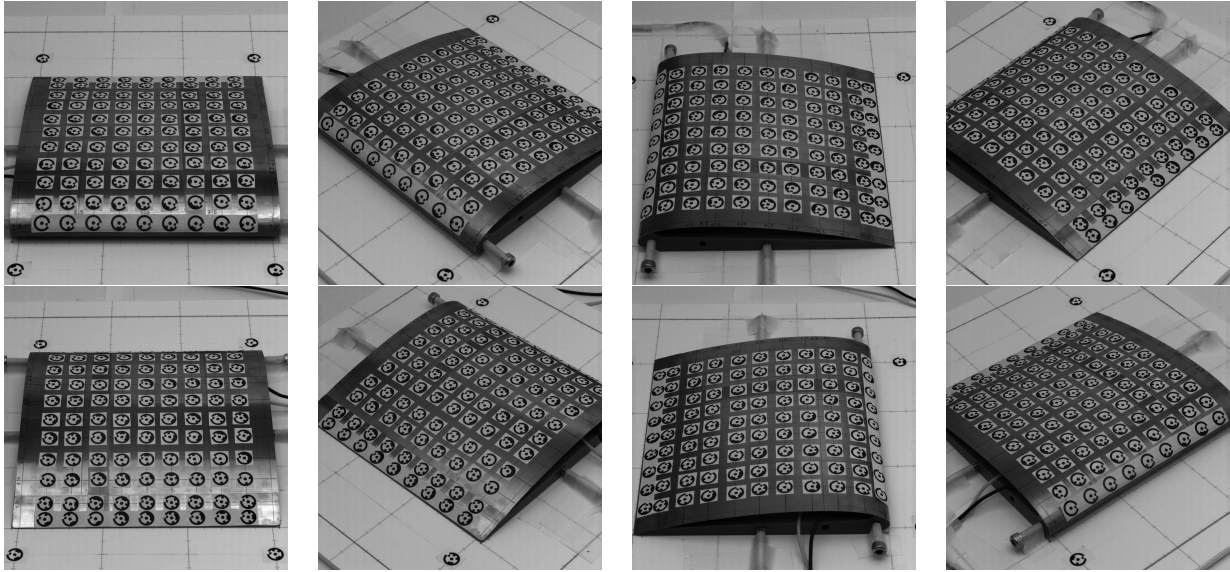


Figure 3.15: 8 views of the model used for 3D reconstruction.

When a single point in 3D space is captured in two separate images the intersection of the light rays can be determined by an epipolar geometry [82]. The locations at which the line connecting the two cameras, or *baseline*, intersects the images planes are called epipoles. The baseline and the point 3D point form an *epipolar plane*. When the location of the 3D point moves, the epipolar plane rotates around the baseline. The intersection of the epipolar plane and the image planes are known as epipolar lines. The epipolar geometry is independent of the scene structure and depends only on the camera properties – which are known from the QR decomposition of P (defined in Appendix A.2) – and relative pose [82]. A 3D point located at x , in image coordinates, in one image and x' in a second image are related by:

$$x'E x = 0 \quad (3.2)$$

where E contains the algebraic representation of the epipolar geometry and is known as the *Essential Matrix*.

If a point is captured on one image plane at x , in image-plane coordinates, that same point must be on the epipolar line on the second image plane. Since the internal parameters of the camera are known, the relative pose of the camera at each viewing angle can be found from E [82] and is performed during postprocessing in the software. The mathematics is beyond the scope of this thesis and the reader can consult Multiple View Geometry in Computer Vision by Hartley & Zisserman[82] for details.

3.4 Static Actuation

The surface could be statically deformed by applying a DC signal. The hysteresis loop was applied prior to the acquisition of data, as explained above. To gain a qualitative appreciation of the shape change the model was capable of, photos of the deformed model are provided in Figure 3.16. The green line represents the 0 V, non-actuated geometry.

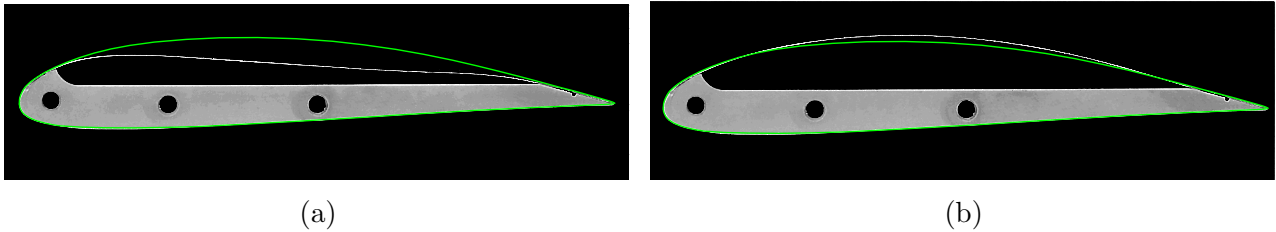


Figure 3.16: Deformed model: a) 1500 V DC; b) -500 V DC.

Photogrammetry was used to capture quantitative details about the static morphing.

Figure 3.17 provides a comparison of the maximum static deformations. The 11 points in the chordwise direction are the average of the 9 spanwise distributed coded targets seen in Figure 3.10. The coded targets are correctly positioned at every $0.1c$ with a maximum error of 0.9% (≈ 1.35 mm), well within a reasonable error associated with the coded targets being positioned by hand. This gives confidence that the photogrammetry is correctly capturing the 3D points.

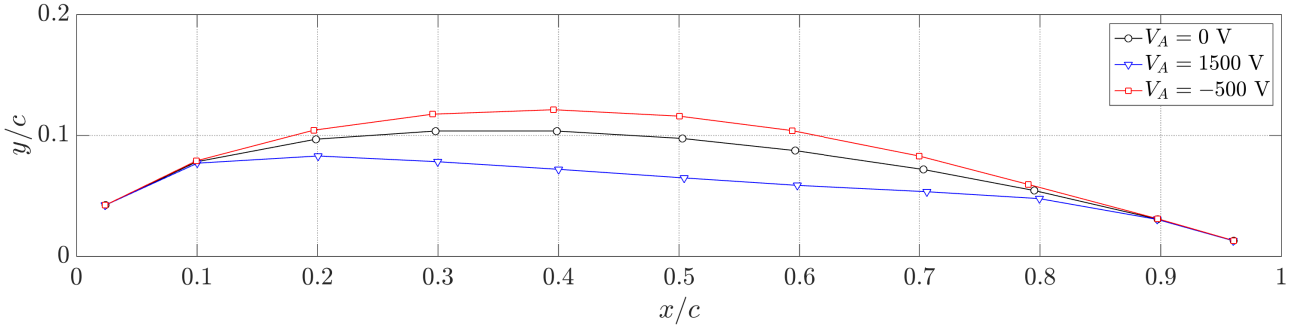


Figure 3.17: Geometry of deformed suction surface.

As already mentioned, the non-actuated model is 1.4% thinner than a true NACA 4415, with a thickness of 13.6% of the chord at $x/c = 0.4$. By applying a voltage of -500 V the surface was deformed by -2.65 mm ($\approx 1.8\%$ of the chord) which resulted in a 15.4% thickness at $x/c = 0.4$ — thicker than a true NACA 4415. It would therefore be possible to compensate for the non-linear effect of the MFC, if desired, by statically deforming the surface by a negative offset voltage.

By applying a positive voltage the aerofoil is made thinner. The maximum positive voltage acceptable by the MFC is 3 times the magnitude of the maximum negative voltage. As a result the inward deformation is greater than that in outward direction. With 1500 V, the aerofoil at $x/c = 0.4$ is 4.75 mm thinner than the non-actuated aerofoil (roughly 3.2% of the chord). In summary, the aerofoil thickness at $x/c = 0.4$ can be varied between 25.15 mm (+4.75 mm at 1500 V) and 17.75 mm (-2.65 mm at -500 V). When compared with the unactuated thickness of 20.4 mm, this means the aerofoil can be made up to 13% thicker or 23% thinner by static actuation of the MFC.

As well as spanwise averaged cross-sections, the photogrammetry data also provide a three-dimensional view of the model. Figure 3.18 shows that the deformation is the same in the spanwise direction i.e. the motion is two-dimensional. This means that the two separate actuators are working well together and the aerofoil geometries in Figure 3.17 are representative of the surface deformation.

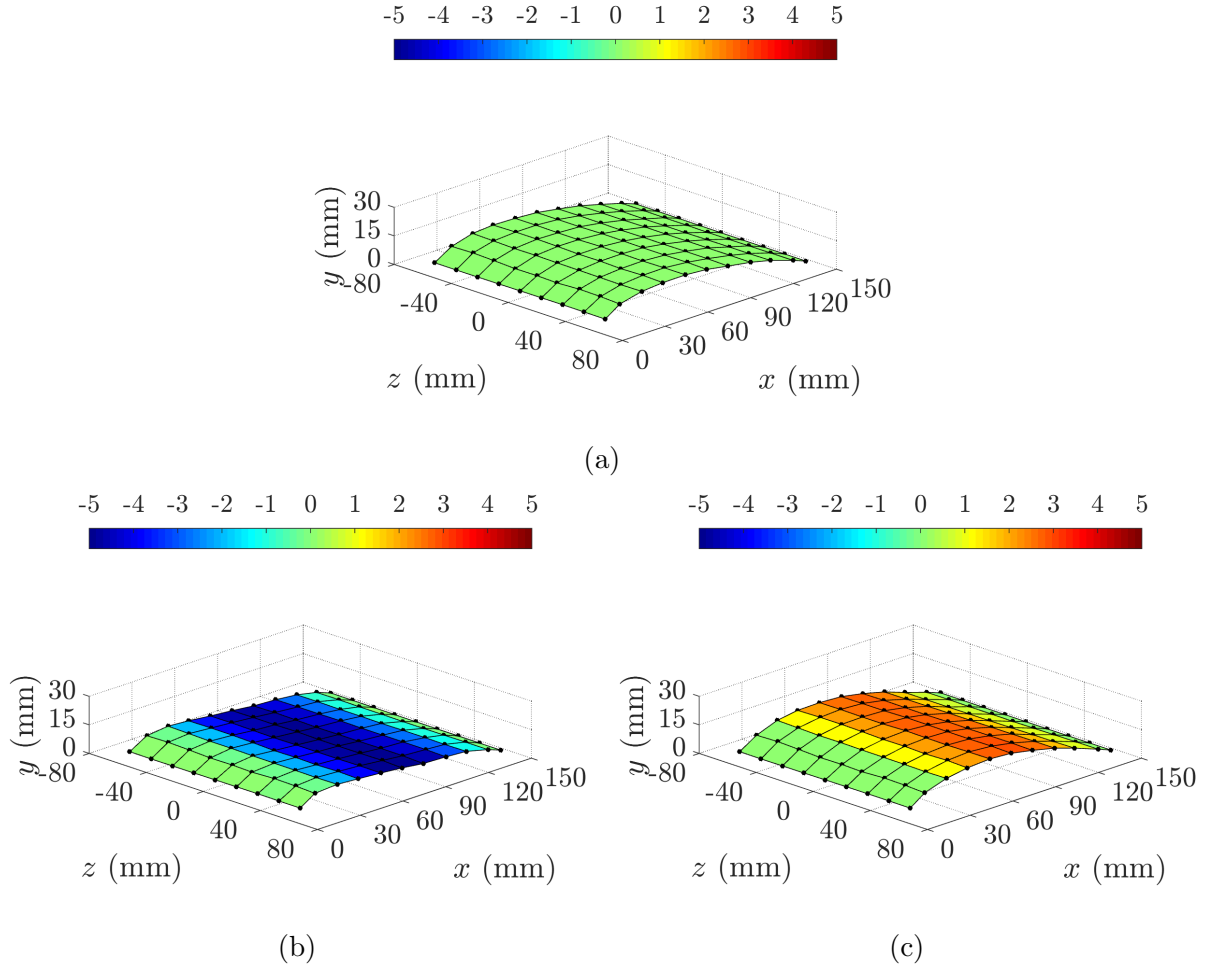


Figure 3.18: 3D reconstruction of statically deformed model: a) $V_A = 0$ V; b) $V_A = 1500$ V; c) $V_A = -500$ V. The contours indicate displacement from non-actuated surface (mm).

3.5 Dynamic Actuation

The procedure for the implementation of dynamic surface actuation has been outlined in Section 3.2 and the methods for measuring this surface motion have been described in Section 3.3. The results of these measurements will now be presented. First, a comparison between the laser and photogrammetry measurements will be used to verify the quality of the photogrammetry results. Following that, the three-dimensional, time-resolved photogrammetry data will explain the surface behaviour under actuation.

3.5.1 Laser Sensor vs. Photogrammetry

Although photogrammetry provides a more comprehensive description of the surface motion, it cannot be performed in the wind tunnel since there was not sufficient optical access. On the other hand, the laser displacement sensor, despite only providing a one-dimensional point measurement, had the advantage of acquiring measurements without impediment through the tunnel's windows. Therefore, the surface was first measured with the laser *in situ* to check its response in still and moving air. This is an important consideration from an aerodynamic point of view as it shows how susceptible the surface motion is to the aerodynamic forces. Peak-to-peak displacements at 4 chordwise locations in both still and moving air as a function of actuation frequency are shown in Figure 3.19.

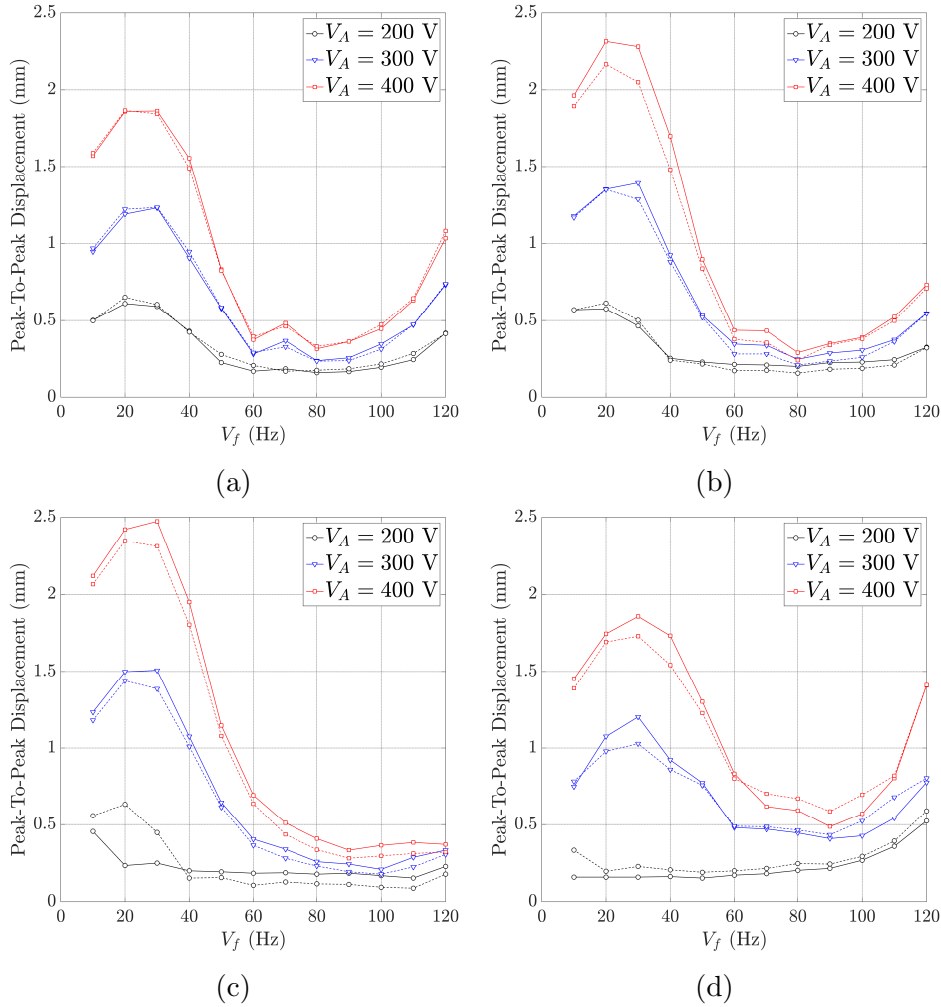


Figure 3.19: Peak-to-peak displacements of the skin with flow (Dashed) and without flow (Solid): a) $x/c = 0.25$; b) $x/c = 0.40$; c) $x/c = 0.50$; d) $x/c = 0.70$;

Both amplitude and frequency of the input signal have a significant effect on the dynamic behaviour of the skin but the aerodynamic forces in moving air appear to have little impact on the surface motion when $Re_c = 5 \times 10^4$. As a general rule, the higher the amplitude of sinusoidal voltage, the greater the peak-to-peak displacement. At frequencies above 30 Hz the amplitude of displacement drops significantly at all measured locations. This was expected and has been observed in previous models. It is attributed to resonance occurring at around 30 Hz. However, due to a change in the vibration mode, the amplitude begins increasing again at around 80 Hz.

To check how accurately the photogrammetry captures the dynamic surface motion, time traces of 10 cycles at 10 Hz, 20 Hz, 40 Hz and 80 Hz are compared with the laser displacement data in still air in Figure 3.20. The laser sensor and photogrammetry results are in very good agreement. The only small difference appears at the negative peak at 40 Hz. From Figure 3.17 it can be noticed that the inward displacement flattens the surface. It is possible that at 40 Hz, where the displacement is large, the flatter surface has caused a larger than expected angle between the surface and the sensor, causing a small error in the laser measurement.

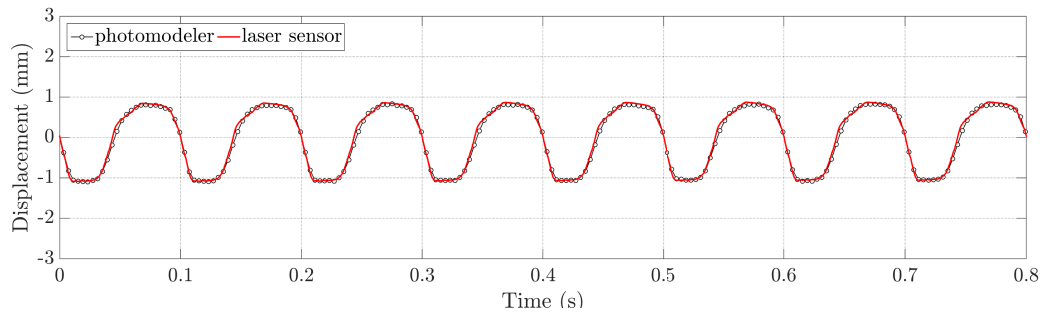
In all cases the surface moves slightly further in the negative (inward) direction than in the positive direction. The motion appears periodic in all cases but not perfectly symmetric, particularly at lower frequencies. The peak-to-peak amplitudes increase from 10 Hz to 40 Hz and is much lower at 80 Hz, in agreement with Figure 3.19.

Dynamic Modes

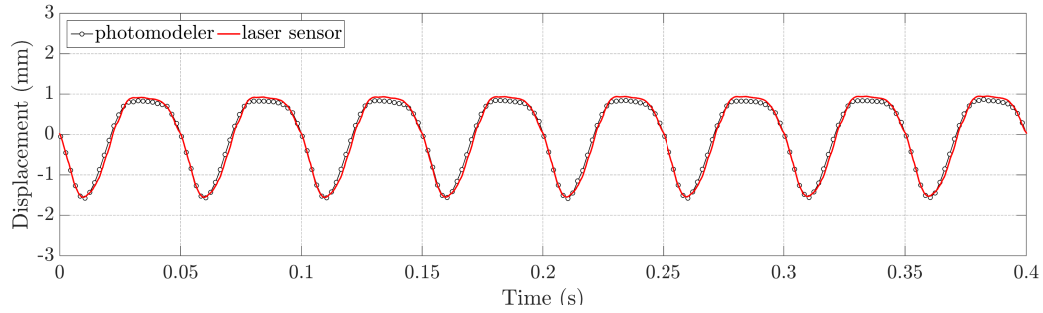
Having verified that the photogrammetry data was accurate, the three-dimensional behaviour of the surface was investigated at a range of frequencies between 10 Hz and 100 Hz. The camera frame rates are proved in Table 3.1

V_f (Hz)	10	20	40	50	80	100
frame/second	250	500	1000	1000	2000	2000
frame/cycle	25	25	25	20	25	20

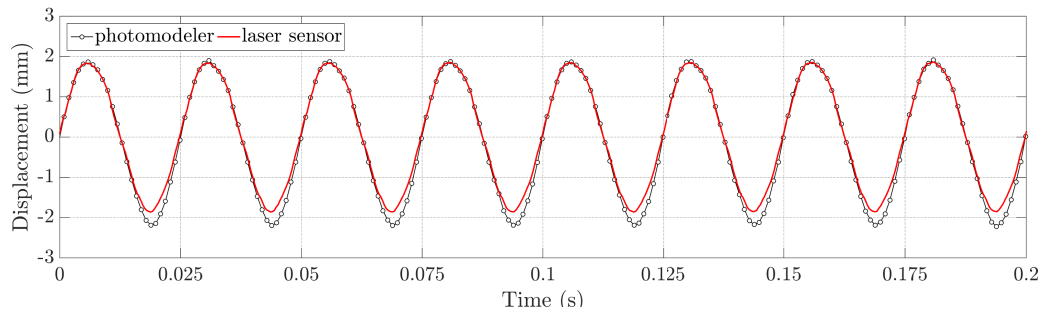
Table 3.1: Camera frame rates at different frequencies.



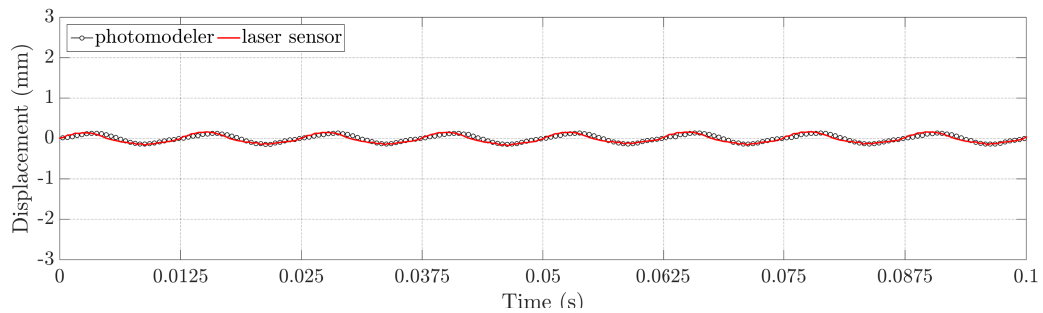
(a)



(b)



(c)



(d)

Figure 3.20: Comparison between Laser Sensor and Photogrammetry measurements at $x/c = 0.4$; $z/s = 0.5$: a) 10 Hz; b) 20 Hz; c) 40 Hz; d) 80 Hz.

Figures 3.21-3.24 show the wall-normal modal behaviour obtained by Proper Orthogonal Decomposition (POD) of the displacement snapshots from 64 cycles. POD provides the most efficient way of capturing the dominant components from an ensemble of data [83].

Consequently, when applied to the photogrammetry results, it reveals relevant structures hidden in the data [84]. Only the wall-normal component of the data was considered since it was the dominant one and the most important from the point of view of the application to flow control. The POD was performed on the snapshots of the z -coordinates and the resulting modes were projected onto the undeformed x and y -coordinates. For an explanation of POD the reader is referred to Kerschen *et al.* [84].

At 10 Hz, almost all of the energy – 99.6% – is contained within the first vibration mode. The mode shapes, represented in 2D and 3D in Figures 3.21b and 3.21c respectively, show that the motion is symmetric about the centre-chord and largely 2D in the spanwise direction.

Figure 3.21b reveals that there is a slightly greater displacement at the far side of the aerofoil, towards $z = 60$ mm, but Figure 3.21c suggests this effect is minor and can be ignored.

When V_f is doubled to 20 Hz (Figure 3.22) and again to 40 Hz (Figure 3.23), the motion is still spanwise two-dimensional and symmetric about the mid-chord. The energy in the first mode decreases very slightly as V_f increases but remains very high at 99.4% and 99.2% respectively.

However, a further doubling of V_f to 80 Hz (Figure 3.24) causes a large change in the modal behaviour. Firstly, the shape of the first mode is quite different. It is no longer symmetric since the location of maximum displacement has moved rearward, towards the slide joint at the trailing edge. This is consistent with the increase in peak-to-peak displacements at $x/c = 0.7$ at high frequencies observed in Figure 3.19. Furthermore, the motion appears more three-dimensional in the spanwise direction. Again it is $z = 60$ mm side of the aerofoil that has the greater displacement. This suggests that the MFC are either not perfectly aligned or not behaving in exactly the same way. It is worth bearing in mind that the axes in Figure 3.24c exaggerates this trend and high speed animation at this frequency reveals the three-dimensionality to still be very small. Aside from the change in the shape of mode 1, there has also been a change in the energy content, which has dropped to 88.3%. This is still a large portion of the total energy but with the secondary vibration mode now containing

more than 10% of the energy, the surface motion has clearly changed. A second vibration mode has been encountered at this frequency. This mode shape exhibits two peaks and is skewed in the z -direction since the peaks have a maximum at opposite sides of the aerofoil in the spanwise direction.

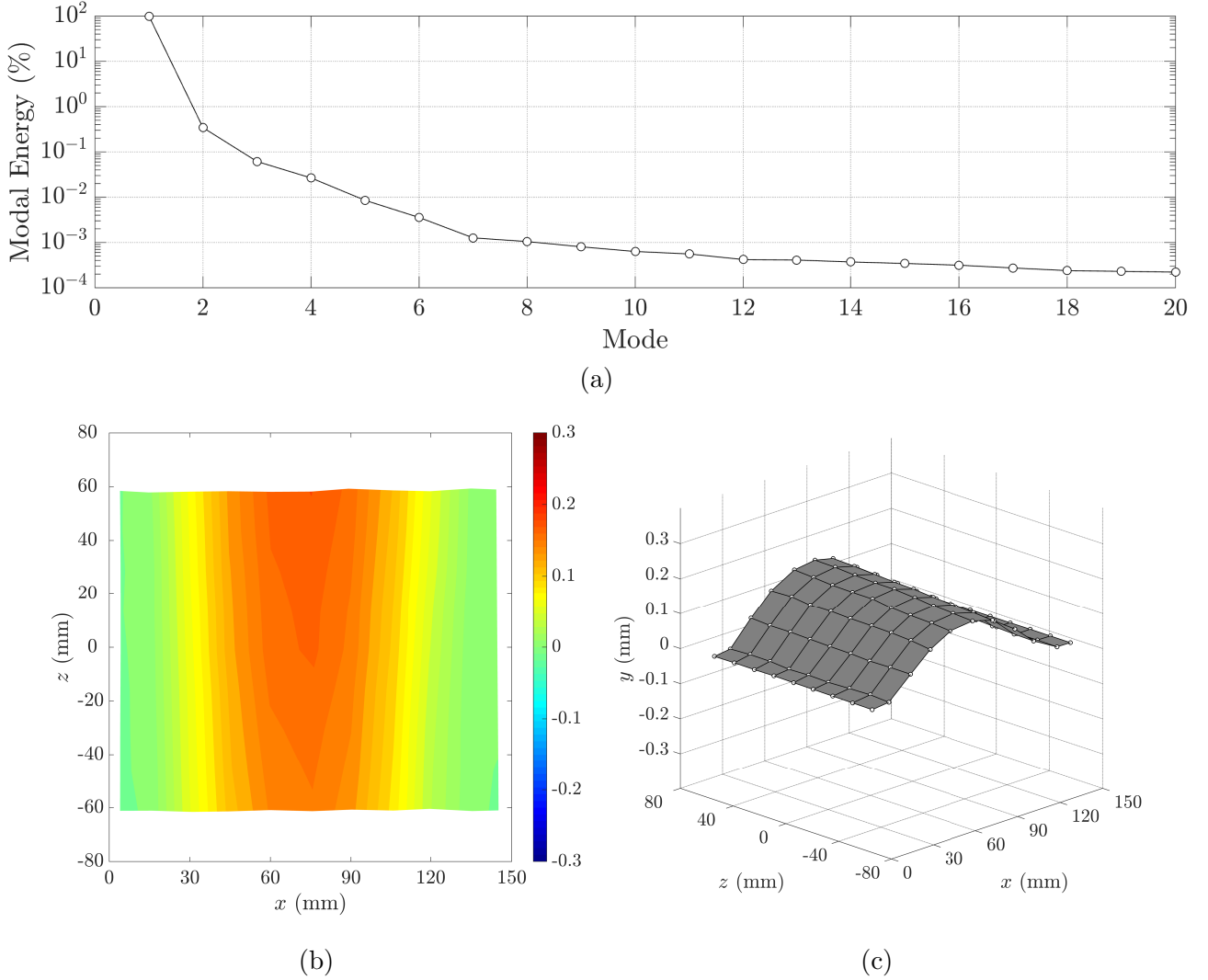


Figure 3.21: Vibration modes of the dynamically actuated surface at 10 Hz: a) energy content of first the 20 modes; b) 2D representation of first mode shape; c) 3D representation of first mode shape.

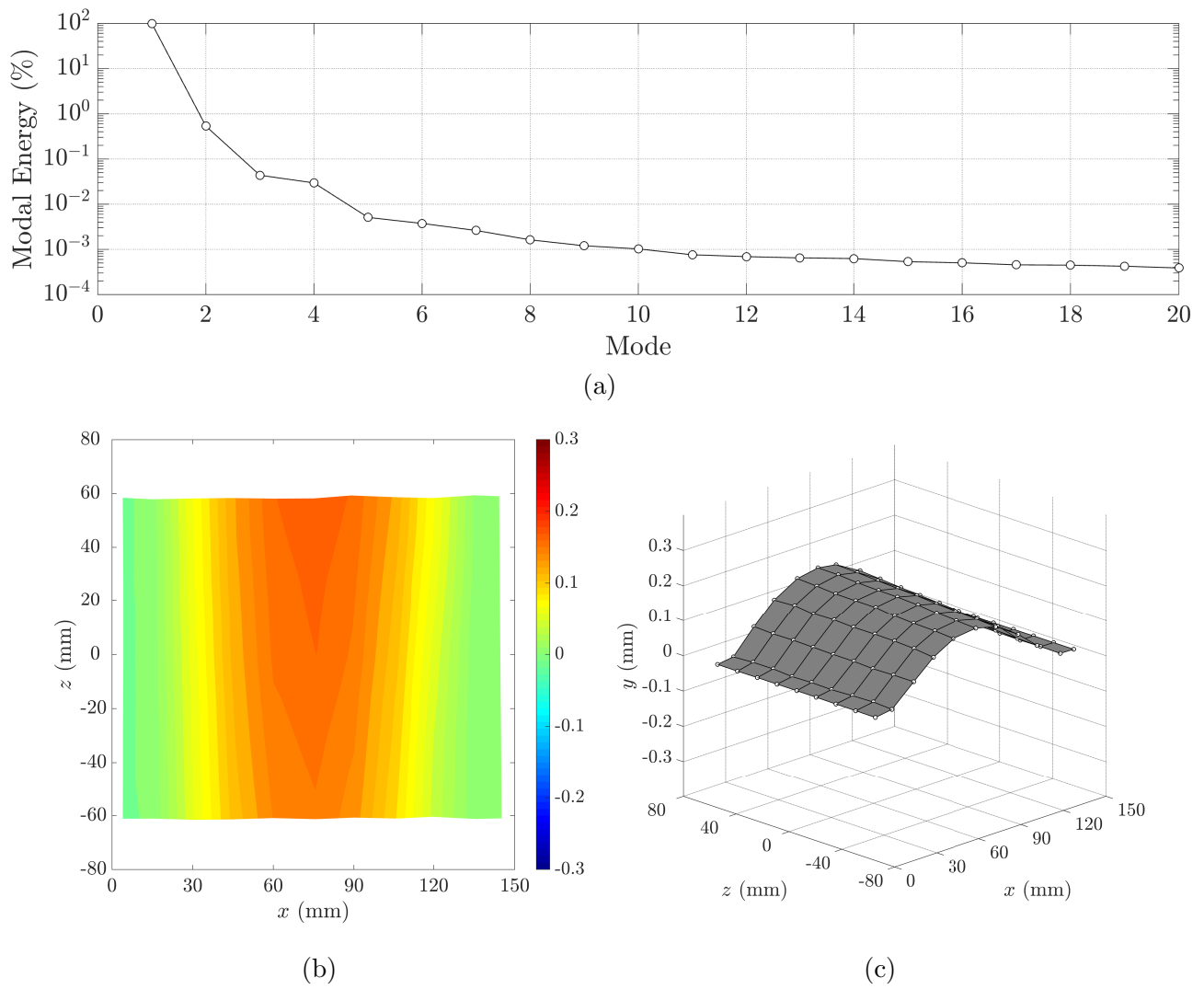


Figure 3.22: Vibration modes of the dynamically actuated surface at 20 Hz: a) energy content of the first 20 modes; b) 2D representation of first mode shape; c) 3D representation of first mode shape.

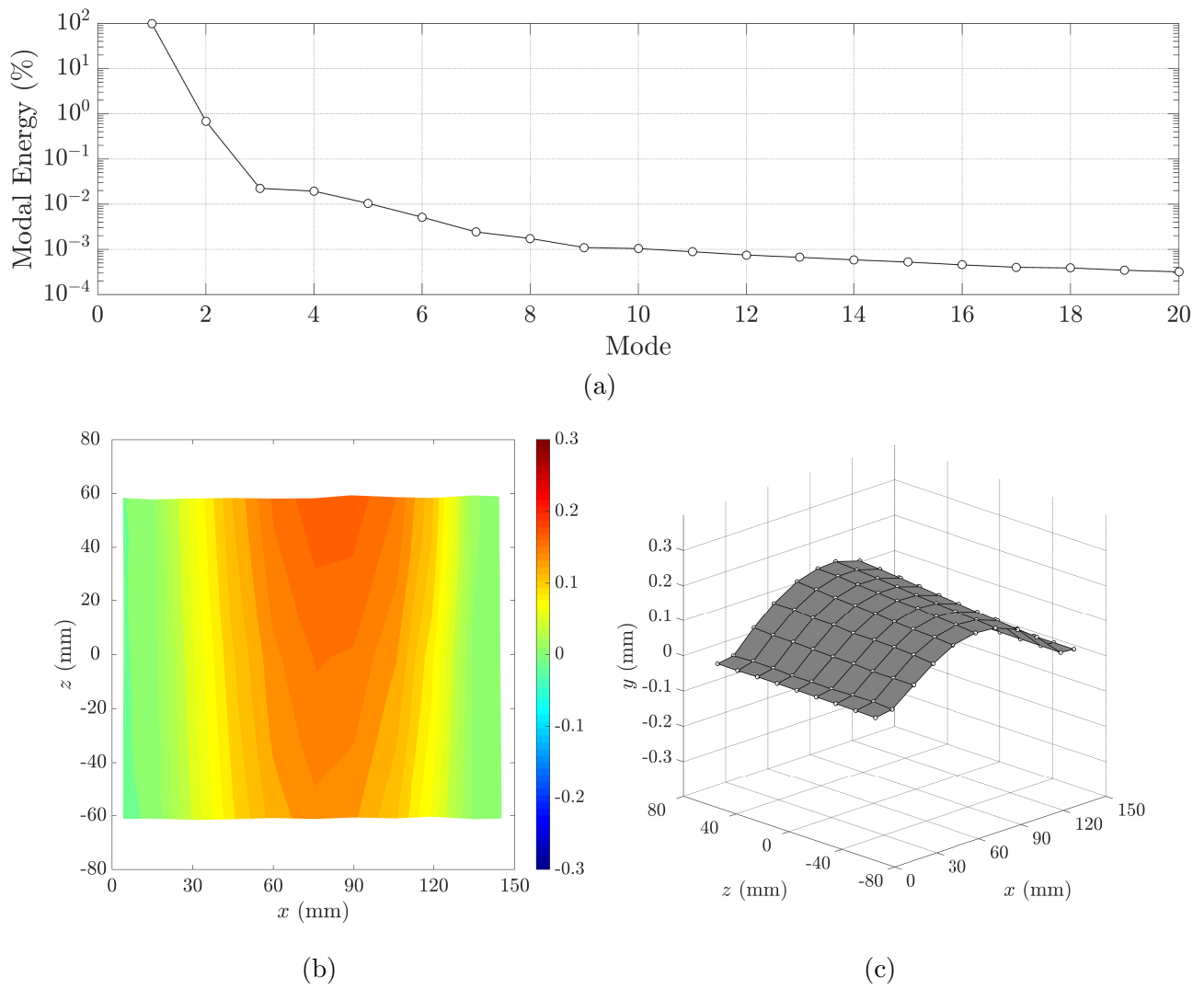
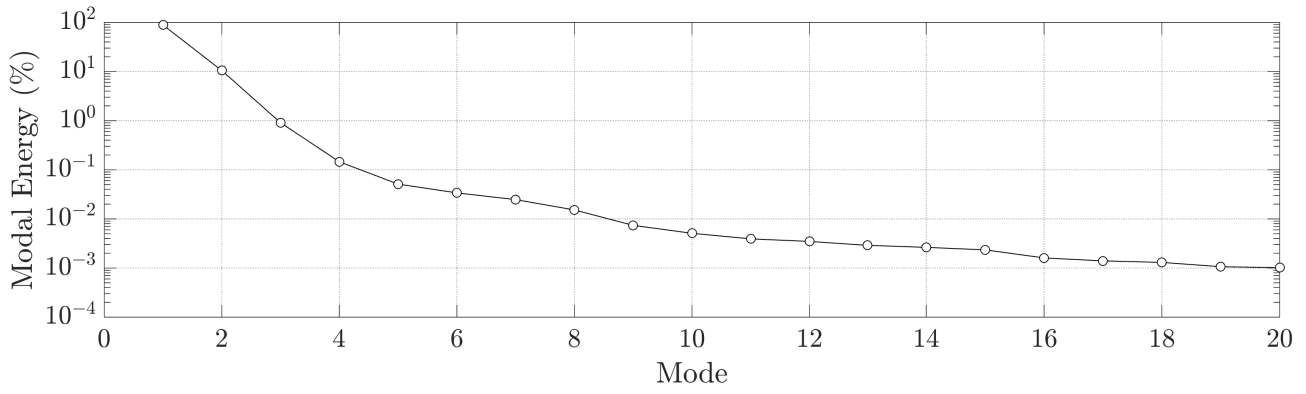
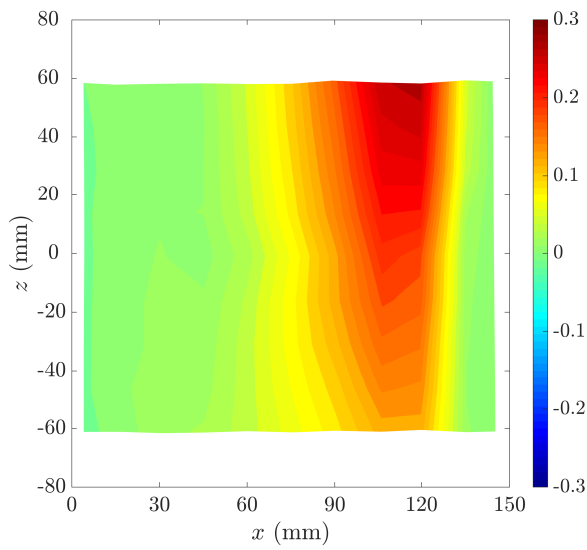


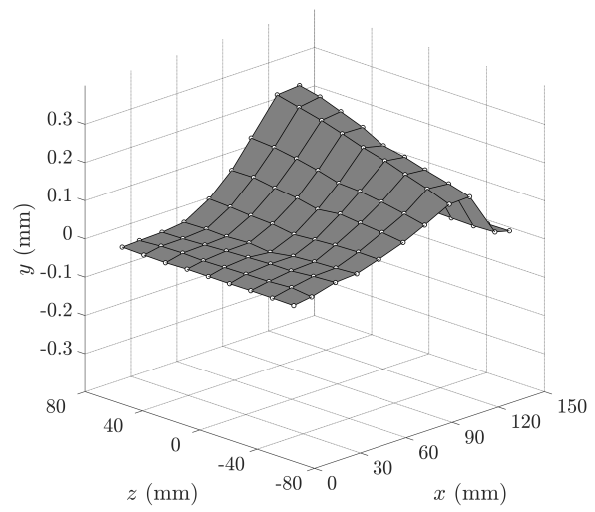
Figure 3.23: Vibration modes of the dynamically actuated surface at 40 Hz: a) energy content of the first 20 modes; b) 2D representation of first mode shape; c) 3D representation of first mode shape.



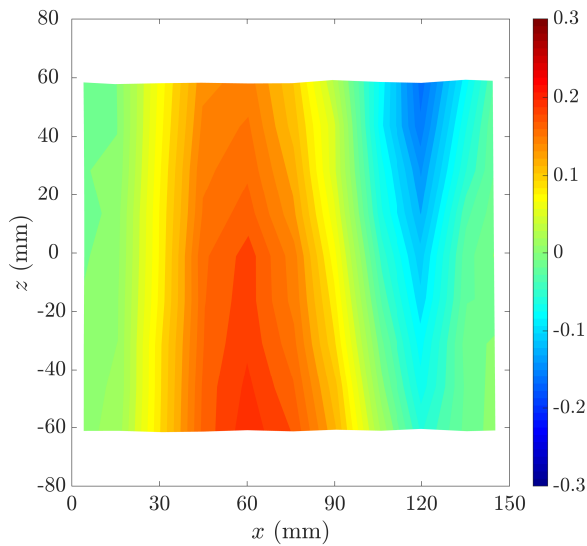
(a)



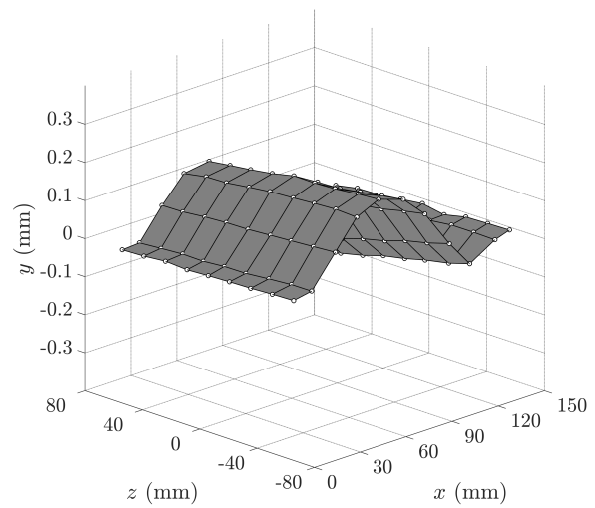
(b)



(c)



(d)



(e)

Figure 3.24: Vibration modes of the dynamically actuated surface at 80 Hz: a) energy content of the first 20 modes; b) 2D representation of first mode shape; c) 3D representation of first mode shape; d) 2D representation of second mode shape; e) 3D representation of second mode shape.

The effect of V_f on modal energy content can be seen in Figure 3.25. The energy in the second mode grows as the energy in the first mode decays, signalling a switching of vibration modes. The growth/decay appears to begin at $V_f > 40$ Hz although the energy in the first mode remains greater than 90% until $V_f = 70$ Hz.

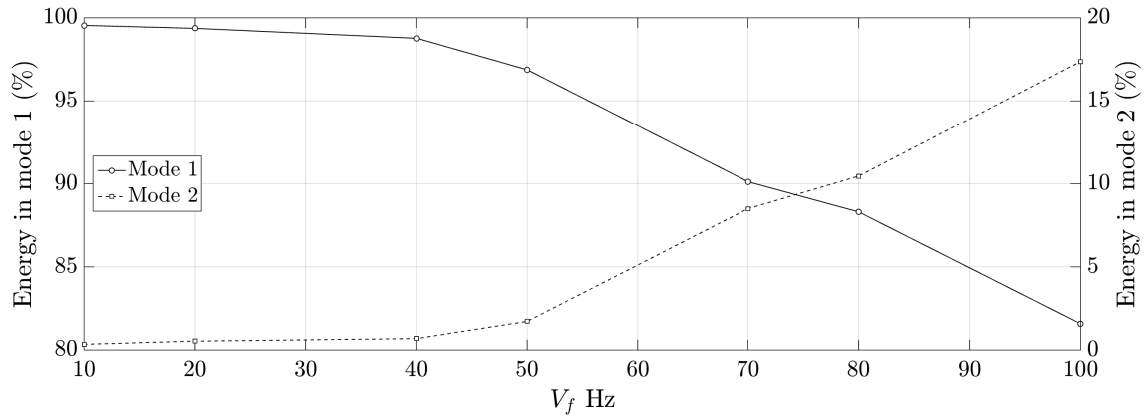


Figure 3.25: Energy content in the first 2 modes with increasing V_f .

3.6 Summary

Using a combination of point (laser sensor) and surface (photogrammetry) measurement techniques, the ability of the MFC to perform both static and dynamic surface morphing was explored in this chapter. A hysteresis cycle was applied prior to all experiments to account for the non-linear effects of the piezoceramic material. The unactuated aerofoil geometry is 1.4% thinner than the reference NACA 4415. By driving the MFC with a DC signal, at the bounds of their operation, the aerofoil can be made up to 23% thinner or 13% thicker.

The primary concern of this thesis is the dynamic behaviour of the surface. It has been shown that both amplitude and frequency have a large effect on the peak-to-peak displacement that the surface can achieve. On the other hand, the aerodynamic forces only have a small effect. Having verified that the photogrammetry was in very good agreement with the laser sensor,

time-resolved 3D data was used to confirm that the dynamic surface motion behaved well up to $V_f = 80$ Hz, where a change in vibration mode occurred.

The remainder of this thesis will be concerned with how the surface motion described in this chapter affects the flow development at low Re_c .

Chapter 4

Experimental Setup

The aerodynamic characteristics of the model were tested in the closed-loop, subsonic wind tunnel at the Temasek Laboratories, National University of Singapore. This chapter describes the primary test facility, the wind tunnel, and the acquisition equipment (hotwire, PIV, force balance) and explains the procedures undertaken to set these systems up.

4.1 NUS Temasek Laboratories Closed-loop wind tunnel

The primary test facility was the closed-loop wind tunnel seen in Figure 4.1. Its test section, which is connected to the exit of the wind tunnel's nozzle, is 600 mm×600 mm and 2 m long. Honeycomb screens are positioned upstream of the test section to reduce turbulence and provide a uniform freestream velocity profile. The 12:1 contraction ratio of the nozzle, positioned immediately downstream of the screens, allows the screens to act on a low velocity fluid before it is accelerated to the desired speed upon entering the test section. This ensures that the flow straightening process suffers from minimal pressure losses. Being a closed-loop type tunnel, the flow will encounter four 90° turns during one loop. To limit separation and the associated recirculation, which would introduce undesirable cross-stream fluctuations, each corner contains a cascade of thin, curved aerofoils, known as turning vanes. As a result,

the tunnel can produce flow velocities from 2 m/s to 90 m/s through its test section with low turbulence intensity, T_u , of less than 0.5%.



Figure 4.1: Closed-loop, sub-sonic wind tunnel at Temasek Laboratories, National University of Singapore.

The model was mounted vertically on a turntable in the test section floor which allowed precise positioning (to within $\pm 0.05^\circ$) of the angle of attack, α . Since the model's span is less than the width of the test section, a splitter plate was installed at 160 mm above the tunnel floor. A boundary-layer ingestion slot with a sharp leading edge spanning 72% of the test-section width was utilised to restart the development of the floor boundary layer and occurred at the same streamwise location as the leading edge of the splitter plate. This meant the boundary layers on both floor and splitter plate were started from the same location. The leading edge of the model was positioned 400 mm downstream of the leading edge of the ingestion slot. The mounting of the aerofoil can be seen in Figure 4.2.

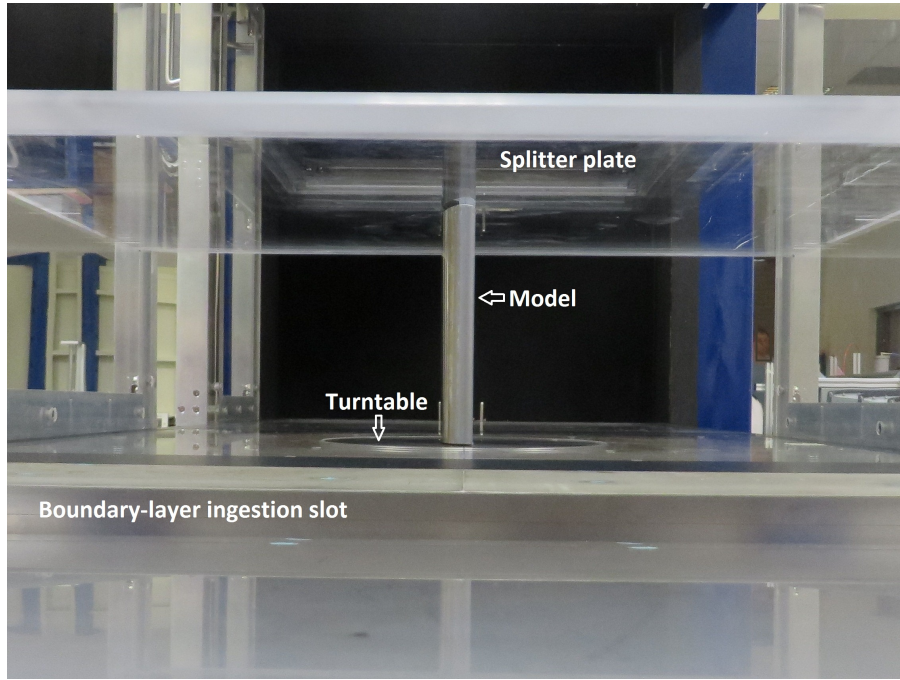


Figure 4.2: Vertical mounting of the aerofoil model beneath a splitter plate.

For α ranging from 0° to 16° , the front area has a minimum of $0.0035m^2$ and a maximum of $0.0077m^2$. The channel between the splitter and the test section floor has an area of $0.096m^2$. This means the blockage ratio ranges from 3.6% to 7.9%.

4.1.1 Characterisation of the Test Section Conditions

Since the experiments were carried out exclusively at low speeds, a verification of the test section conditions was necessary. Hotwire measurements were used to investigate the flow development in the channel between the splitter plate and tunnel floor when $U_\infty = 5$ m/s (the speed at which the experiments are carried out) where U denotes total velocity and the subscript ∞ signifies freestream conditions. To investigate the characteristics and extent of the freestream conditions, vertical sweeps of the entire channel were performed at $x/c = 1.5$ and 2 (measured from the position of the aerofoil's leading edge, had the aerofoil been present). These are presented in Figure 4.3. The floor boundary layers in Figure 4.3a are of the order of 10 mm, or roughly 6% of the channel cross section, while the boundary layers on the splitter plate is thicker at roughly 15 mm thickness. Nevertheless, a constant freestream

velocity does exist over 80% of the channel width and, from Figure 4.3b, it has a freestream T_u of less than 0.5%.

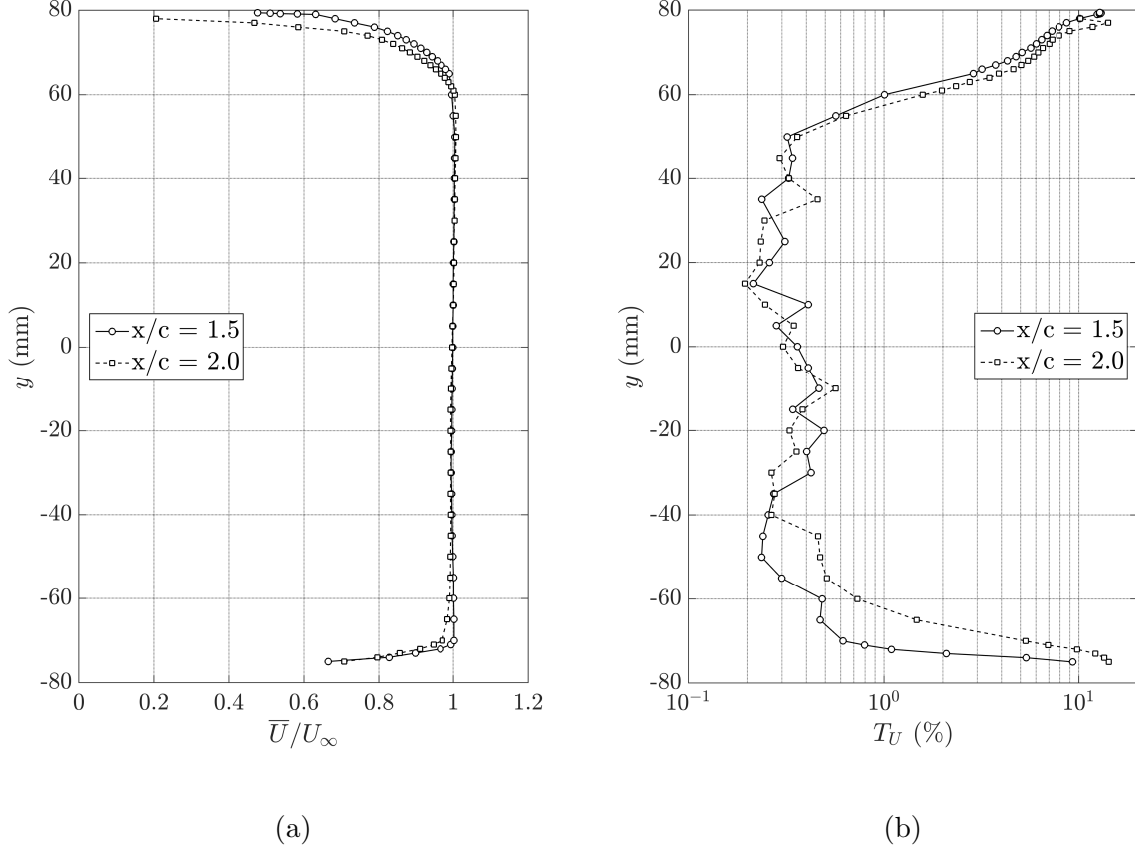


Figure 4.3: Velocity profiles resulting from a cross-stream (i.e. spanwise) sweep of test section channel: a) mean velocity; b) turbulence intensity.

4.1.2 Axis Convention

Figure 4.4 denotes the axis notation that will be used throughout this thesis. Cartesian coordinates x , y and z refer to the streamwise, cross-stream and spanwise directions respectively with the origin at the mid-span position on the aerofoil's leading edge when $\alpha = 0$. A second coordinate system with axes η and ζ that rotates with α and has its origin at the mid-chord is suitable for the measurement of forces. Coordinate ζ is along the chord and η is normal to the chord. When $\alpha = 0^\circ$, (x, y) and (ζ, η) are parallel. Furthermore, a local coordinate system (l, n) is defined in the aerofoil surface to aid post-processing — n and l

being the normal and longitudinal directions respectively, as shown in the figure.

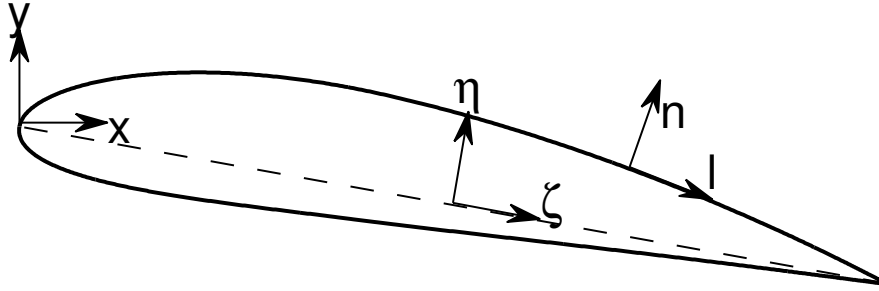


Figure 4.4: Axes convention and coordinate systems for the mounted aerofoil.

4.2 Force Balance

The turntable, on which the model was mounted, incorporated a force balance consisting of a Gamma ATI SI-32-2.5 piezoelectric gauge (see Figure 4.5) capable of measuring the forces and moments along three perpendicular axes. The two axes parallel and normal to the chord (ζ, η) 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 aerofoil mid-chord point ($x/c = 0.5$) and was used to measure the pitching moment. The balance was factory calibrated and the corresponding conversion factors were stored in the acquisition unit. The range (and resolution) of the measured forces and moments are $32 (\pm 6 \times 10^{-3})$ N and $2.5 (\pm 5 \times 10^{-4})$ Nm, respectively.

For each measurement, 9 seconds worth of data (see Figure 3.8) was acquired from the force balance at a minimum sampling rate of 16 samples per cycle. The exact sampling rate will be specified when the results are presented. The beginning of each experiment was externally triggered using LabVIEW to ensure the data acquisition was synchronised with the motion of the surface.



Figure 4.5: Piezoelectric gauge mounted beneath the aerofoil

The lift and the drag coefficients were calculated at each α from the corresponding values of the chordwise and normal forces as

$$C_L = \frac{F_\eta \cos(\alpha) - F_\zeta \sin(\alpha)}{\frac{1}{2}\rho U_\infty^2 cs}, \quad C_D = \frac{F_\zeta \cos(\alpha) + F_\eta \sin(\alpha)}{\frac{1}{2}\rho U_\infty^2 cs} \quad (4.1)$$

in which F , refers to the force the direction of the subscript and ρ is the density of air.

4.3 Constant Temperature Anemometry

Constant Temperature Anemometry (CTA) can provide time-resolved velocity measurements at a point by making use of convective heat transfer from a heated wire to the atmosphere and allows investigation of the unsteady features of the flow around the aerofoil. These systems provide excellent sensitivity at low velocities [85] which makes them suitable for the present research. The system's sensor comes in the form of a miniature wire probe and forms part of a *Wheatstone Bridge* – an electrical circuit capable of measuring unknown resistances. When the resistances of the bridge's arms are equal, the voltage across the bridge is zero. However, air passing over the wire will cool it down and consequently change its resistance

thereby unbalancing the bridge and generating a voltage. This voltage signal, E , is a measure of heat transfer rate from the wire and once calibrated can be used to measure the velocity of the fluid element that cooled it. A servo-amplifier connected in a feedback loop monitors the voltage signal across the bridge and, by controlling the current in circuit, it can restore the wire's temperature and thus re-balance the bridge. This discussion will be concentrated only on the setup relevant to the experiments. A complete description of the theory can be found in Lomas (1986) [85] and Perry (1982) [86].

4.3.1 Hardware Setup

The relationship between length and diameter of the wire is an important one in CTA [85]. A sensor that is too long would be susceptible to breakage by vibrations or impact. A shorter wire will experience a greater proportion of its heat transfer as conduction through the support needles meaning the temperature profile across the sensor is less uniform. This is undesirable since one of the key simplifications in these measurements, introduced in the next subsection, will assume a wire of infinite length. According to Brunn (1971) [87], the optimum aspect ratio is between 200 and 400. For this experiment a single, miniature platinum-plated tungsten wire sensor with diameter of $5\text{ }\mu\text{m}$ and length of 1.25 mm was used. This is an aspect ratio of 250 and is therefore within the optimum range and the effect of convective heat transfer, rather than conductive heat transfer, is dominant.

The whole system was comprised of the probe, an analogue-to-digital converter and the CTA which are connected using a 4 m BNC-BNC cable with a resistance of $0.64\text{ }\Omega$. The CTA unit contains the Wheatstone bridge, servo-amplifier and a signal conditioner. The operation of the system relies on the balancing of a Wheatstone bridge and its performance depends on the response of the feedback loop. To ensure that the static balancing of the bridge was correct, an overheat adjustment procedure was performed to set the resistance of the bridge's passive arm to a value that would balance the wire's resistance at operating temperature, T_w . The operating conditions were determined by setting the overheat ratio, a , to 0.8. This value defines the resistance of the heated wire to the resistance of the wire in ambient conditions. The difference between the wire's operating temperature and the ambient air temperature

was calculated as 222.2 K.

As a passing fluid element cools the wire and unbalances the bridge, a voltage is produced that provides information about the velocity of that fluid element. To obtain time-resolved information about the flow it is the job of the feedback loop to then restore the temperature of the wire to T_w quickly enough such that information of the subsequent fluid element passing the wire can also be recorded. The system therefore requires a high frequency response but it is also important that the re-heating does not overshoot T_w . Calibrating the CTA's frequency response was achieved by applying a square wave to the system from an in-built square wave generator and adjusting the servo-amplifier's filter and gain settings to 3 and 8 respectively. When subject to a square wave test the system took 0.02 ms to return to a balanced state. This corresponds to a cut-off frequency, f_c , of 37 kHz which is far greater than any frequencies of interest in this research. To reduce the effect of unwanted high-frequency noise a signal conditioner is used to low-pass filter the signal at 3 kHz before it is digitalised by the analogue-to-digital converter.

4.3.2 Velocity Calibration

The signal provided by the CTA is in terms of bridge voltage, not velocity. A calibration was therefore carried out for each experiment using a dedicated velocity calibration unit to enable a conversion of the voltage signal to a velocity signal. When supplied with pressurised air, this calibrator is capable of producing a free jet from 0.5 m/s to 90 m/s. By recording the pressure drop across the nozzle and employing a feedback system, the calibrator could iteratively approach a user-defined flow speed. The probe is placed at the tip of the calibrator's nozzle and the voltage experienced by the CTA was recorded when the calibrator produced 20 different velocities within the range of 0.5 m/s up to 6.5 m/s. These points were distributed logarithmically so that the calibration curve at low velocities, where its shape changes most rapidly, was well represented. The relationship between voltage and velocity can then be described by a transfer function given by heat transfer laws. The most widely used heat transfer law was used in this research, namely King's Law.

King's Law describes the heat transfer from infinite cylinders in a fluid due to forced

convection [88]. When applied to hotwire anemometers the law is expressed as

$$E^2 = A + BU^n \quad (4.2)$$

where E is the output voltage from the anemometer and U is the fluid velocity. A and B are constants, as is exponent, n , and are determined during the calibration process. An example of this process is explained in Figure 4.6. It is noted that King's Law describes the effect of forced convection and ignores the contribution of natural (buoyant) convection. At very low velocities the effect of the latter becomes of equal magnitude of the former, however, the velocities in this case are large enough for this not to be the case.

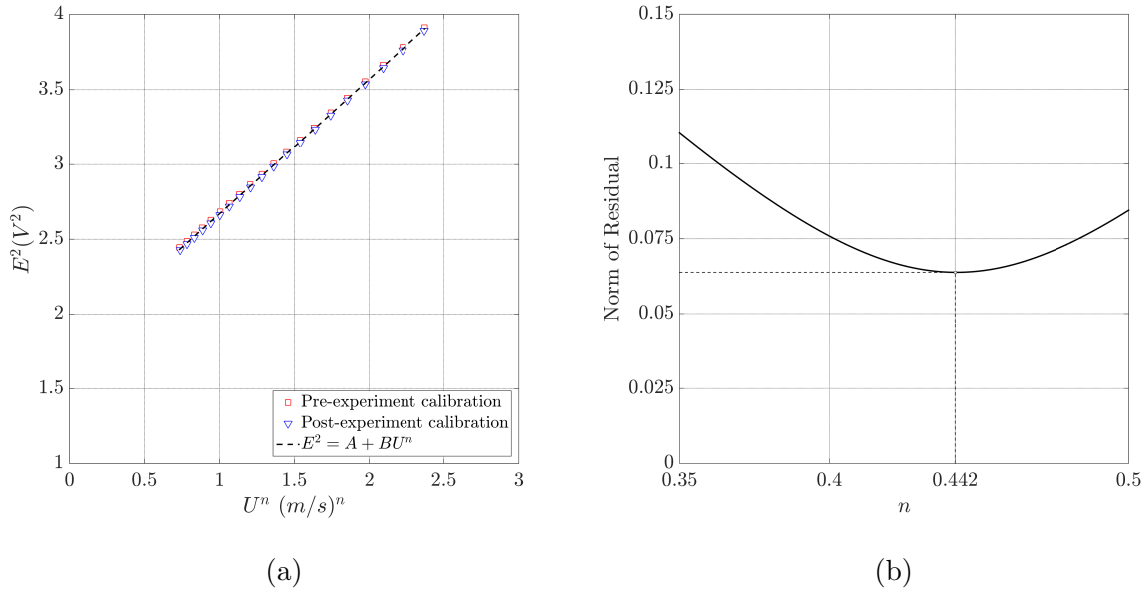


Figure 4.6: Velocity Calibration using King's Law a) calibration curve; b) selection of exponent n

The calibration process was run before and after each experiment to account for any thermal drift during that period. From Figure 4.6a it is clear that this effect is negligible since the pre- and post-experiment points are almost the same. The value of exponent n was chosen by fitting a linear polynomial to E^2 and U^n as n was varied. The value that provided the best fit (i.e. smallest error in Figure 4.6b) was chosen with the corresponding values of A and B to provide the transfer function in equation 4.2.

4.3.3 Procedure

The probe was mounted to the arm of the traverse and placed at a vertical location of half-span and different streamwise locations behind the aerofoil's leading edge. The probe was then traversed in the y -direction, pausing at several different locations to acquire 2^{17} samples of the velocity at a sampling rate of 6 kHz, before moving on to the next point. The velocity signals recorded at each of these locations could therefore provide mean and fluctuating velocity profiles. The frequency content of the time signal can be obtained via a fast-Fourier transform that allows the identification of dominant frequencies in the wake and separated shear layers.

4.4 Particle Image Velocimetry

In contrast to CTA, which provides data at a single point, Particle Image Velocimetry (PIV) is capable of providing full flow field data in an almost entirely non-intrusive manner, making it a very useful tool for investigating the flow development around an aerofoil. The basic idea of PIV is to seed the flow with tracer particles and measure their displacement during a fixed time interval, Δt . In practice, if the concentration of particles is high enough, the flow field is divided into smaller *interrogation regions* and the average displacement of all particles within each region provides a single displacement vector for that particular interrogation window, during that particular time interval. The velocity field is then calculated from the displacement vector and the time interval, Δt .

4.4.1 Flow Seeding

The type of seeding particle is important in PIV techniques since the flow field velocity is actually determined indirectly from the velocity of these particles. If the particle motion does not closely match that of the fluid, the calculated fluid velocity will not be accurate. However, the purpose of the particles is not just to trace the fluid but also to reflect light so their motion can be captured by a camera. Decreasing a particle's diameter will increase their

ability to trace the fluid but it will also reduce the amount of light they reflect [89]. Therefore, the particle selection is a compromise between the two conflicting requirements. For this experiment, water-based particles from a SAFEX fog generator were chosen. These particles have lower density than oil-based or solid particles and could be produced with an average size of SMD $1\text{ }\mu\text{m}$. Their reflections corresponded to approximately 3 pixels in the captured images which gave a good resolution.

The light, which the particles are intended to scatter, was provided by a pulsed Litron DualPower 200-15 Nd:YAG laser which is operated in its second harmonic to generate a light at the visible wavelength of 532 nm and approximately 150 mJ per pulse. Since the population inversion of Nd:YAG-based lasers is too long for successive pulses to capture the motion of the fluid, a dual-head laser is used with a second laser head pulsing at a time Δt after the first. The reflections were captured by a HiSense 620 double exposure CCD camera which has a resolution of 2048×2048 pixels. A Zeiss 50 mm f/2.0 micro lens was used to capture a $225\text{ mm} \times 225\text{ mm}$ field of view and the flow around the complete aerofoil as seen in Figure 4.7.



Figure 4.7: Field of view ($225 \times 225\text{ mm}$) for PIV investigation of the whole flow around the aerofoil.

Synchronisation and control of the timing of the laser and camera are achieved through the DynamicStudio software from Dantec Dynamics, which is also used to process the resulting

images. The timing interval between laser pulses could be varied and was set so that the majority of the particles in the illuminated plane moved by approximately a 25% of the interrogation cell, which is optimum for this software.

Interrogation Window Size

A 225×225 mm field of view captured on a 2048×2048 pixel camera provides an image resolution of approximately $110 \mu\text{m}$ per pixel. According to Adrian [90], PIV modes of operation fall into 3 main categories: 1) Laser-Speckle Mode, when the concentration of seeding particles is so high that images of individual particles appear to overlap in the captured image, 2) Particle-Tracking mode, when the concentration of seeding particles is so low that individual particles can be traced from image-to-image and 3) High-Density PIV mode, when the concentration of seeding particles is high enough that each interrogation window can be guaranteed to have numerous particles within them, but that their images will not overlap [90]. Evidently, from Figure 4.7, the set-up used in this experiment is suitable for the High-Density PIV mode as the seeding is highly concentrated in the field of view but individual particles are still distinguishable. This field is subdivided into interrogation regions that can correctly capture the average particle displacement for calculation of the velocity field while the time interval, Δt , between two frames was set to $220 \mu\text{s}$. As each window will correspond to just one displacement vector, the finer the resolution of the flow field the greater the spatial resolution of the results. However, a smaller interrogation window introduces more uncertainty in the reconstruction of the flow field.

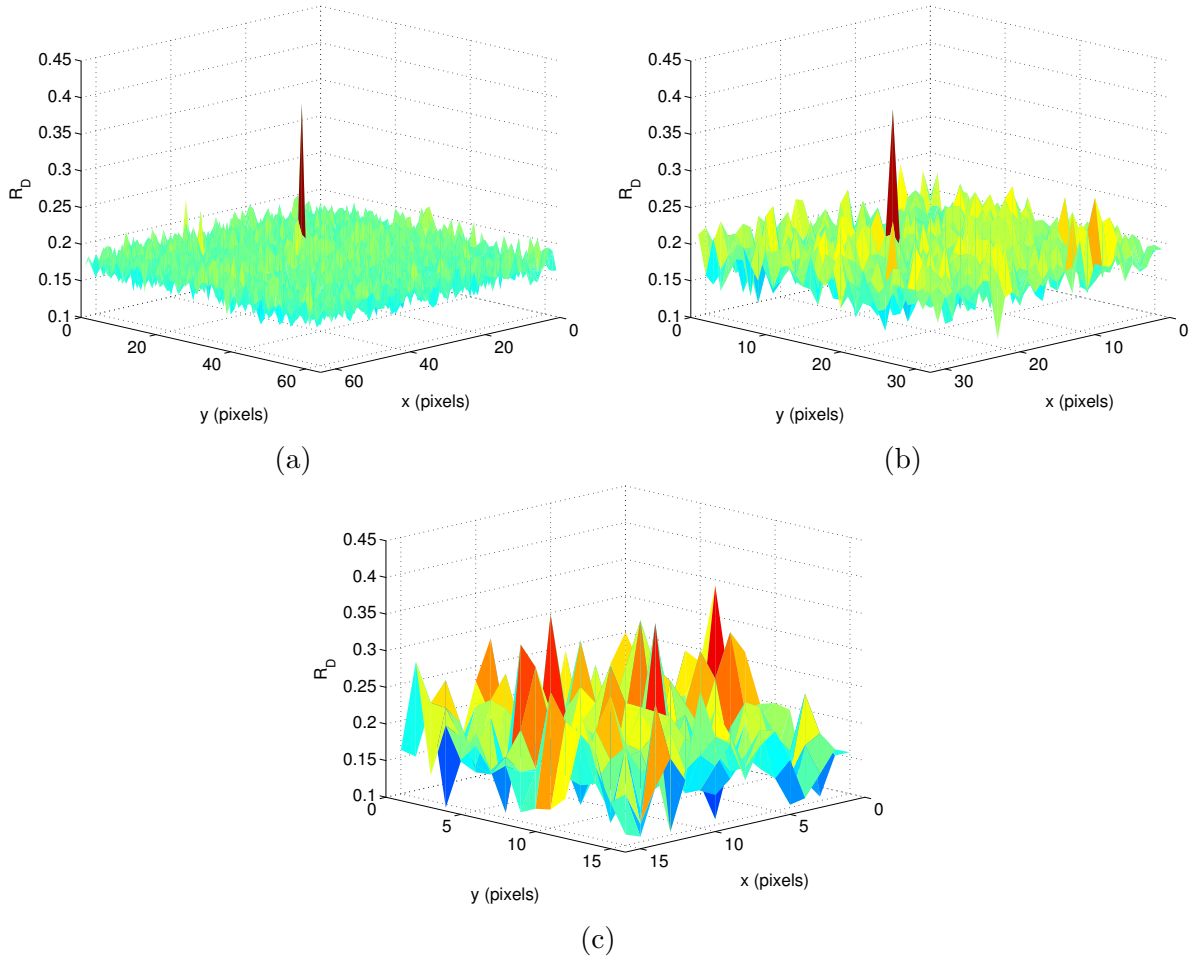


Figure 4.8: Cross correlations of single PIV image with interrogation window size of (a) 64×64 (b) 32×32 (c) 16×16 pixels.

Figure 4.8 shows the cross-correlations of typical interrogation windows of different sizes. Windows of 64×64 and 32×32 pixels provide a single correlation peak meaning the displacement vector and flow velocity in these cases can be calculated with confidence. This cannot be said for the 16×16 pixel case which exhibits several correlation peaks. A compromise between resolution and accuracy of the velocity field must therefore be sought. For this reason, a 32×32 pixel window size was preferred. To improve the accuracy of interrogation, a 50% overlap was included between adjacent windows to account for particles passing across window boundaries between subsequent time intervals. As a result 127×127 velocity vectors were provided in total across the complete field of view.

Procedure

The wind tunnel was seeded with water-based particles upstream of the fan and the tunnel was run until a satisfactory seeding density and uniformity was qualitatively observed in the test section. The laser, camera and data acquisition were all controlled through Dantec Dynamics' DynamicStudio software and connected to an external trigger which ensured that the timing of the laser pulses and image capturing were synchronised. As the laser was only capable of a maximum frequency of 7 Hz, it was not possible to obtain a time resolved velocity field and instead a series of snapshots of the flow field were captured and averaged in time. To ensure convergence of the time-average flow, enough snapshots must be captured such that the average field becomes independent of the number of snapshots used. Furthermore, these snapshots must be obtained at sufficient time intervals that each snapshot is uncorrelated from the preceding snapshot. To check that the suitability of the acquisition of snapshots for time-averaging, the running mean and r.m.s. velocities of the flow at 3 points (indicated with symbols in Figure 4.9) were calculated for 300 images. It is clear from Figure 4.10 that 300 images are more than sufficient to calculate the mean and r.m.s. values of the velocity field.

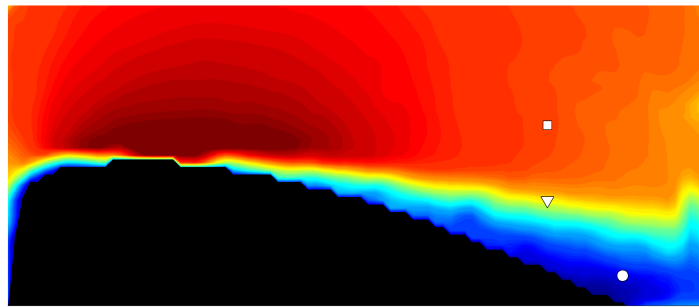
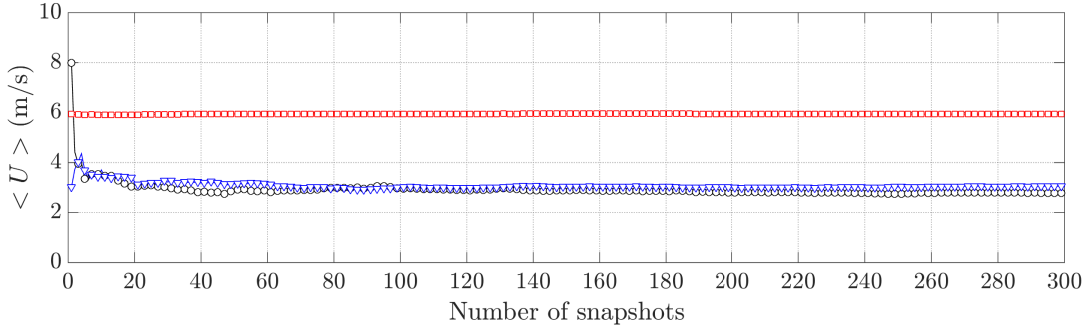
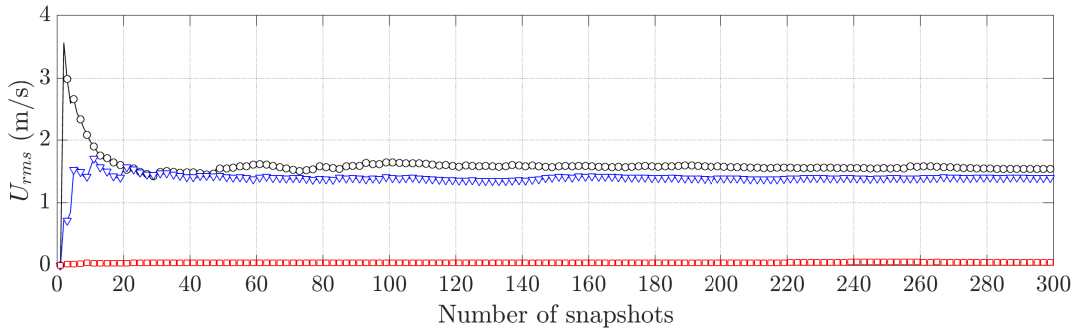


Figure 4.9: Locations of points with the PIV field used for convergence check.



(a)



(b)

Figure 4.10: Statistical convergence of the average velocity field: a) Mean; b) r.m.s. The symbols correspond to points in Figure 4.9.

4.5 Summary

The experimental apparatus that will be used to perform the aerodynamic testing in the next chapter has been introduced. The aerofoil model is vertically mounted, below a splitter plate, in a closed-loop wind tunnel at the National University of Singapore. A constant freestream velocity was found to exist over 80% of the channel between the tunnel floor and splitter plate, with a T_u less than 0.5%. All hotwire and PIV measurements will be taken on a cross-stream plane that lies within this freestream region. These two experimental techniques will be used to study the flow development over both the unactuated and actuated aerofoil. A piezoelectric gauge will be used to measure the aerodynamic force coefficients.

Chapter 5

Experimental Results

A series of experiments were conducted at $Re_c = 50,000$ and α ranging from -6° to 16° . In this chapter, results of the flow development around the aerofoil and its wake – comprising of flow field measurements, velocity profiles and frequency spectra – will be presented together with corresponding measurements of the aerodynamic forces. The non-actuated case will be presented first and the effect of surface motion will be explored later.

5.1 Non-actuated Aerofoil

The purposes of the non-actuated aerofoil experiments were threefold. First, the presence of laminar separation and its development is examined with reference to the literature. Second, the baseline data will be used to verify the validity of the computational simulations. Third, the data will provide a reference case for comparison when the surface is actuated.

5.1.1 Aerodynamic Forces

It was explained in the literature review in Section 2.1.1 that at low Re_c an aerofoil suffers significant losses of aerodynamic performance when compared with the same aerofoil at high Re_c . This has been attributed to the low energy laminar boundary layer separating much

sooner than a turbulent one at high Re_c . In Figure 5.1 the aerodynamic force coefficients are compared with NACA 4415 coefficients from Hoffmann *et al.* [4] and Jacobs and Pinkerton [5], who conducted their experiments at higher Re_c .

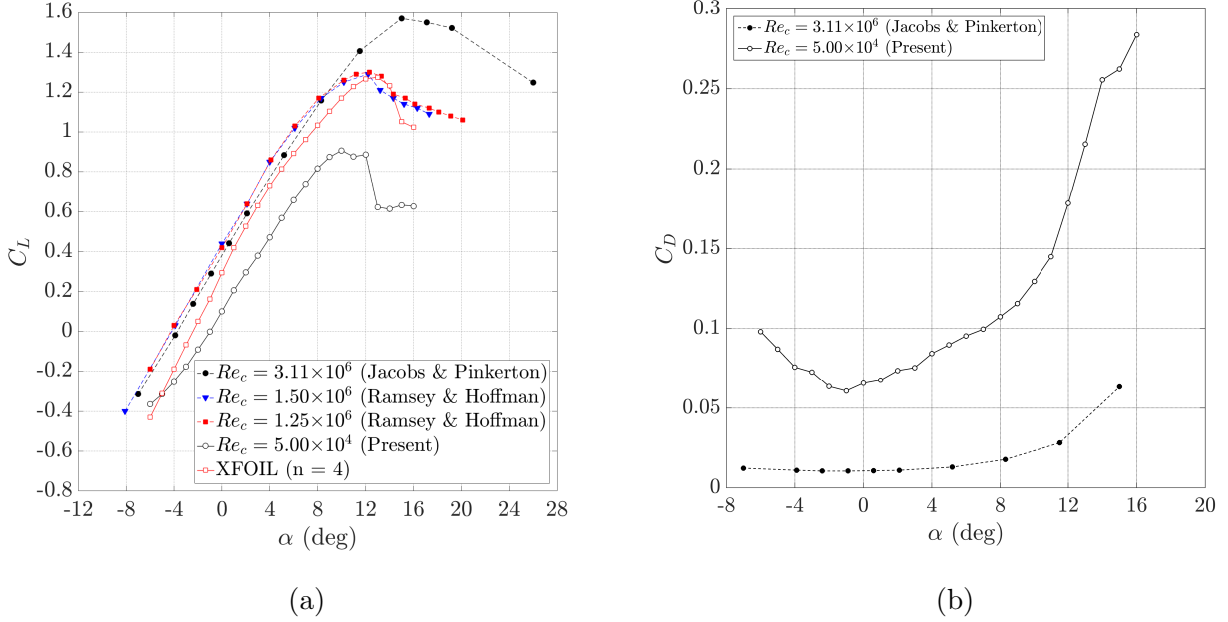


Figure 5.1: Aerodynamic force coefficients against α obtained from the non-actuated aerofoil compared with moderate and high Re_c data from Ramsey and Hoffman [4] and Jacobs and Pinkerton [5] respectively: a) C_L ; b) C_D .

With reference to Figure 2.2 [2] and the description of flow modes in Wang *et al.* [22] in Section 2.1.2, $Re_c = 3.11 \times 10^6$ is within the high Re_c regime while $Re_c = 1.50 \times 10^6$ and 1.25×10^6 are within the moderate regime. In addition, it is recalled that the pre-stall coefficients become independent of Re_c for $Re_c > 5 \times 10^5$ [21], which is indeed observed in the figure (to within 5% due to different test facilities).

In agreement with the literature, a substantial drop in C_L for all α is observed for the current experiments when compared with high Re_c . The C_L at $\alpha = 0^\circ$ is more than 4 times larger in the high Re_c cases compared to the value in the current experiments. The onset of stall occurs at $\alpha = 10^\circ$ limiting $C_{L_{max}}$ to 0.91. This value is 73% and 43% lower compared to $Re_c = 3.11 \times 10^6$ and 1.50×10^6 respectively. In addition, Figure 5.1b shows very large increases in C_D for all α when $Re_c = 50,000$; 6 times larger at 0° and 4 times larger at 15° . A simultaneous increase in C_D and decrease C_L causes an order of magnitude decrease in the

L/D ratio, as can be seen in Figure 5.2.

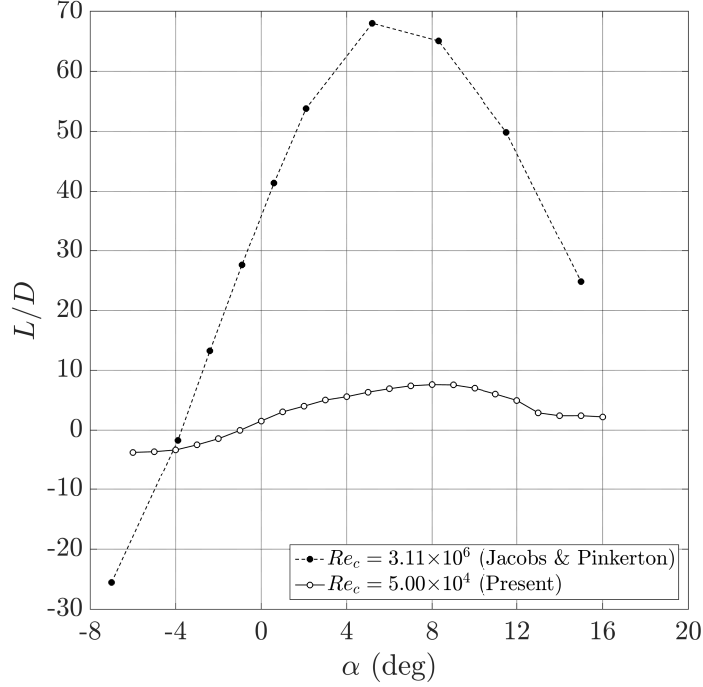


Figure 5.2: L/D ratio of the non-actuated aerofoil compared with the value at high Re_c by Jacobs and Pinkerton [5].

A linear lift-curve for $\alpha < 6^\circ$ is obtained in Figure 5.1a, with a more gradual approach to stall. This is in good agreement with Gerakopoulos *et al.* [91] who studied separated aerofoils at low Re_c . The XFOIL lift curve for the test geometry which is plotted in Figure 3.7 (with $n = 4$) to try and match the tunnel T_U) is also included in Figure 5.1a. It can be seen that XFOIL gives a greater lift curve slope at low angles than the experiment does. This is likely due to the low aspect ratio of the model (which would change the slope) and the difficulty in accurately capturing both the test section conditions and the complex flow physics with such a crude simulation. It is acknowledged that the experimental setup in this case, particularly the model's aspect ratio, mean the absolute values of the coefficients are rather unique to the current parameters. However, it is the effect of actuation relative to this baseline that is of interest in this thesis. The lift-curve is investigated in more detail in Appendix A.3.

5.1.2 Flow Field Development

To understand the aerofoil performance characteristics outlined in the previous section, flow fields at various angles were examined. Iso-contours of time-averaged $\bar{u}/U_\infty$ from PIV data at 4 different values of α are shown in Figure 5.3. The over-line denotes the time-average of the streamwise velocity component. They all show large regions of slow-moving fluid towards the trailing edge of the aerofoil, indicative of separation. The increasing upstream extent of this region with increasing α is evidence of the upstream movement of the separation point. This agrees with previous studies of laminar separation [34, 33, 92]. The flow does not appear to reattach since the separation zones persist beyond the trailing edge. Flow development characterised by laminar separation without reattachment was observed at similar flow conditions by Yarusevych *et al.* [7]. It is expected that a large separation region over the upper surface of the aerofoil results in a dramatic decrease in lift and a significant drag penalty [7]. The presence of such separated regions is the cause of the poor aerodynamic performance presented in the previous section.

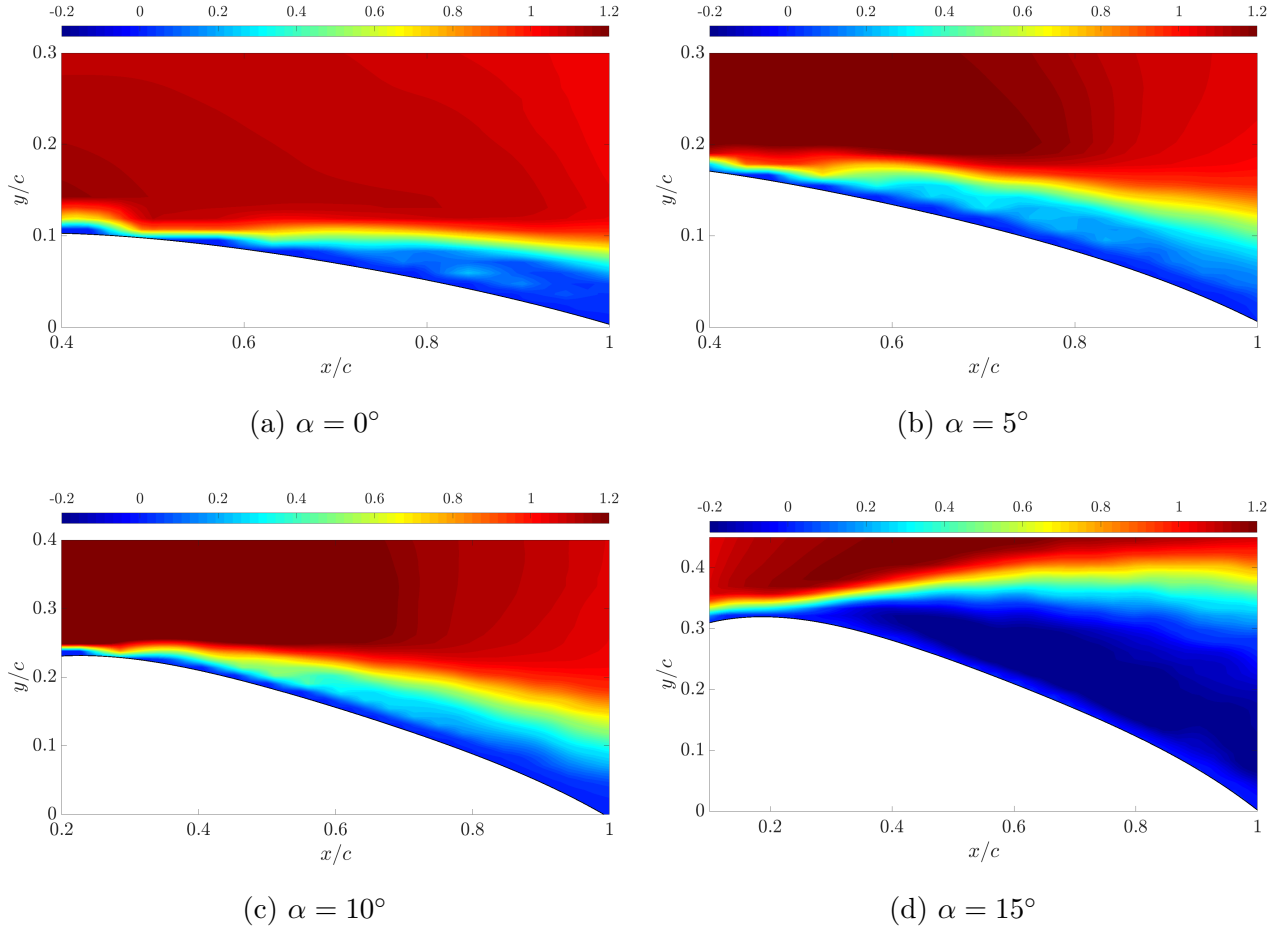


Figure 5.3: Iso-contours of $\bar{u}/U_\infty$ for 4 angles of attack obtained from PIV data for the non-actuated aerofoil.

Owing to the relatively small regions of separated flow in the pre-stall cases, coupled with poor spatial resolution and errors from reflections, the small velocities in the reverse flow region are not well captured in the field of view used in Figure 5.3. Therefore, the camera was zoomed in on the trailing edge, providing a smaller field of view of 97×97 mm and an improved resolution of $47 \mu\text{m}$ per pixel is presented for the $\alpha = 0^\circ$ case in Figure 5.4a. Figure 5.4b shows $\alpha = 15^\circ$ replotted from Figure 5.3d with the alternative colourmap for clarity. The separation region in this case was large enough to capture the reverse flow in the origin field of view required. The recirculation zones are now clearly displayed.

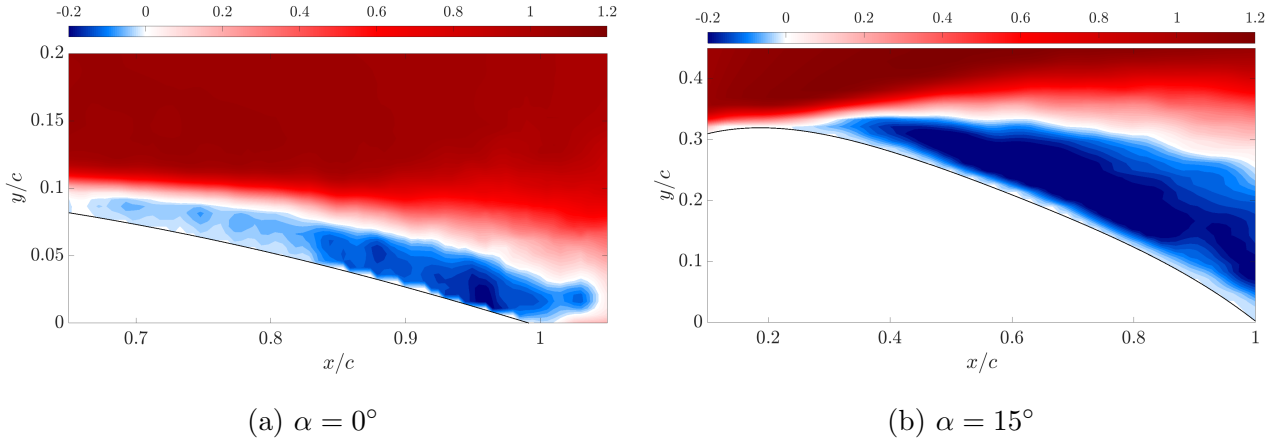


Figure 5.4: Iso-contours of $\bar{u}/U_\infty$ obtained from a field of view focused on the separated region of the non-actuated aerofoil: a) $\alpha = 0^\circ$; b) $\alpha = 15^\circ$.

More insight into the separation characteristics is provided in the form of contours of the time-averaged, spanwise vorticity, $\overline{\Omega_z}$. At 0° the separation zone is much smaller than that at 15° . As seen in Figure 5.5, $\overline{\Omega_z}$ is concentrated exclusively in a thin, separated shear layer between $x/c = 0.65$ and $x/c = 0.8$. The fact that this occurs in a time-average sense indicates the region is steady and corresponds to the laminar portion of the shear layer. From the literature, a more-or-less quiescent flow develops below the shear layer in the laminar separated region [31] which creates a pressure plateau [26]. In agreement with this, very low recirculating velocities are present between $x/c = 0.65$ and $x/c = 0.8$ in Figure 5.4a. While the flow fields in Figures 5.3a and 5.5 are indicative of a laminar separated region, its presence cannot be absolutely confirmed from these plots alone. Further downstream, the magnitude of $\overline{\Omega_z}$ decreases and the region of high vorticity becomes more dispersed. This is interpreted as unsteadiness setting in beyond $x/c = 0.8$ resulting from the breakdown of the shear layer.

The local velocity vectors in Figure 5.5 and the streamwise velocity contours in Figure 5.4a reveal a recirculating vortex is present in, and confined to, this region. The magnitude of the arrows is proportional to $\sqrt{|\overline{U}|}/U_\infty$ so that those in the low velocity region near the wall can be more easily seen. This recirculation zone was also observed in the experiments by Yarusevych *et al.* [26] as upstream entrainment of smoke, emanating from a smoke wire at the trailing edge. The recirculating flow structure occurring in the same region as the termination of a laminar shear layer resembles the well-known reverse-flow vortex found in the turbulent

portion of an LSB [3]. This suggests the breakdown and dispersion of high vorticity in the shear layer is a result of turbulent transition and a pressure recovery would occur in the region of the recirculating fluid.

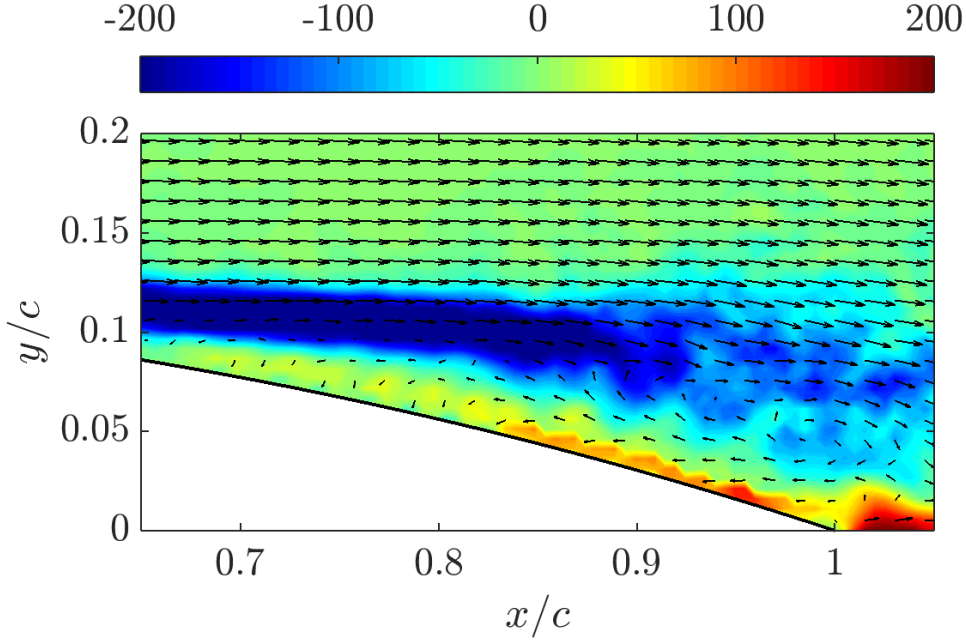


Figure 5.5: Iso-contours of $\overline{\Omega_z}$ when $\alpha = 0^\circ$. The arrows indicate local velocity vectors (the magnitude of the vector is proportional to $\sqrt{|U|}/U_\infty$ so that small velocities can be seen).

It was mentioned in the previous section that the C_D penalty, when compared with the $Re_c = 3.11 \times 10^6$ case, is more severe at $\alpha = 0^\circ$ than it is at $\alpha = 15^\circ$. According to Wang *et al.* [22], stall at high Re_c is preceded by turbulent separation from the trailing edge. The lift curve for aerofoils that experience this so-called ‘trailing-edge stall’ [93, 31] are characterised by a continuous increase in C_L with α until a rounded peak when stall is approached [31]. This is exactly what Figure 5.1a shows for the high Re_c case. It can be concluded then that separation is present in both the high Re_c reference case and the current low Re_c when $\alpha = 15^\circ$.

While separation is also present at $\alpha = 0^\circ$ for low Re_c , the high Re_c reference case will be fully attached at low angles. This explains why the relative drag penalties are worse at lower angles than they are at 15° . At 15° both flows are separated — albeit the separation zone

high Re_c will be relatively small. At $\alpha = 0^\circ$ a separated flow (at low Re_c) is compared with an attached flow (at high Re_c).

When $\alpha = 15^\circ$ the shear layer separates close to the leading edge, seen in Figure 5.6. A large recirculating vortex with size of the same order as the aerofoil chord can be seen after the separation point. This is a clear indication of stall [94, 95] which explains the abrupt drop in C_L in Figure 5.1a.

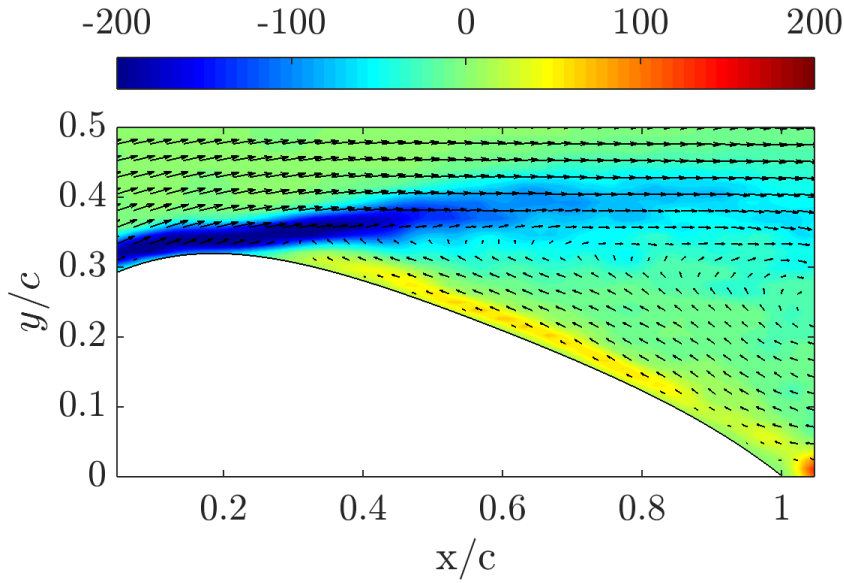


Figure 5.6: Iso-contours of $\overline{\Omega_z}$ when $\alpha = 15^\circ$. The arrows indicate local velocity vectors (the magnitude of the vector is proportional to $\sqrt{|U|}/U_\infty$ so that small velocities can be seen).

5.1.3 Velocity Profiles

Profiles of the time-averaged velocity in the wake at 3 locations: $x/c = 1.25, 1.50$ and 1.70 are shown in Figure 5.7. They reveal the cross stream extent of the wake and the magnitude of the velocity deficit. The latter is related to the profile drag. The width of the wake is comparable to the aerofoil thickness ($0.13c$). This is qualitatively the same as the wide wake observed by Yarusevych *et al.* [26] in their case of separation without reattachment. Their smoke flow visualisations found this wide wake to be formed by large, coherent, vortical structures which were shed downstream.

Profiles of the time-averaged velocity in the wake at 3 locations: $x/c = 1.25, 1.50$ and 1.70 are shown in Figure 5.7. Since this is a single-wire hotwire probe, no sense of the flow direction is provided. Therefore the profiles in Figure 5.7 correspond to the velocity flowing perpendicular to the wire or the ‘effective cooling velocity,’ U_{eff} . They reveal the cross stream extent of the wake and the magnitude of the velocity deficit. The latter is related to the profile drag. The width of the wake is comparable to the aerofoil thickness ($0.13c$). This is qualitatively the same as the wide wake observed by Yarusevych *et al.* [26] in their case of separation without reattachment. Their smoke flow visualisations found this wide wake to be formed by large, coherent, vortical structures which were shed downstream. It can also be seen that the maximum velocity deficit (and the majority of the velocity deficit overall) occurs for positive values y/c . This is a result of the separation from the suction surface.

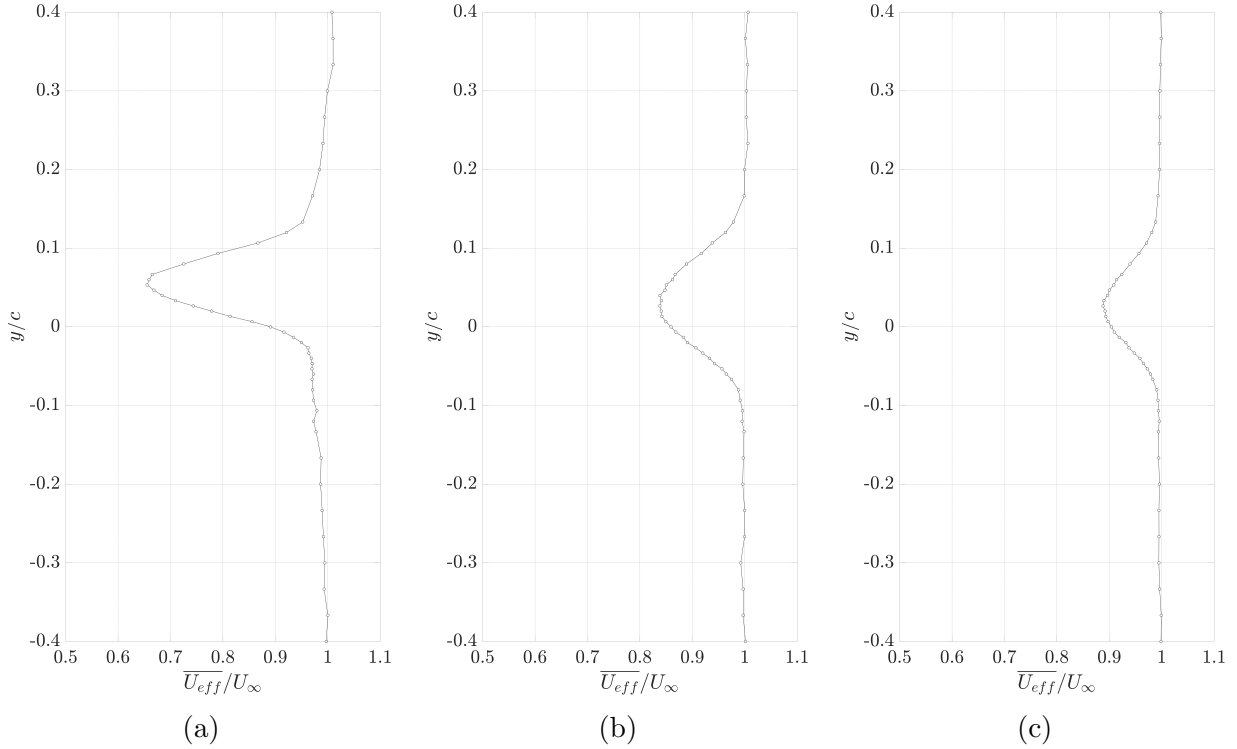


Figure 5.7: Profiles of $\overline{U_{eff}}/U_\infty$ in the wake of the non-actuated aerofoil: a) $1.25c$, b) $1.50c$, c) $1.70c$.

Insight into the evolution of coherent structures can be found from frequency spectra of the flow at different streamwise locations. The velocity signal was obtained by the hotwire at the y/c location corresponding to half the velocity deficit at a particular streamwise location. This signal of 2^{17} samples, sampled at 6 kHz, was divided into signals of 2^{13} samples using a

Hanning window with 50% overlap. This gives a spectral resolution, Δf , of 0.73. The average of the spectra at the three streamwise locations shown in Figure 5.7 are plotted in Figure 5.8b. The spectra were displaced by a factor of 10 in the vertical direction for clarity of presentation. A very strong peak is observed, centred at 85 Hz. This frequency is considered to be the vortex shedding frequency in the wake. The fact that the peak is so distinct indicates a strong coherence [7].

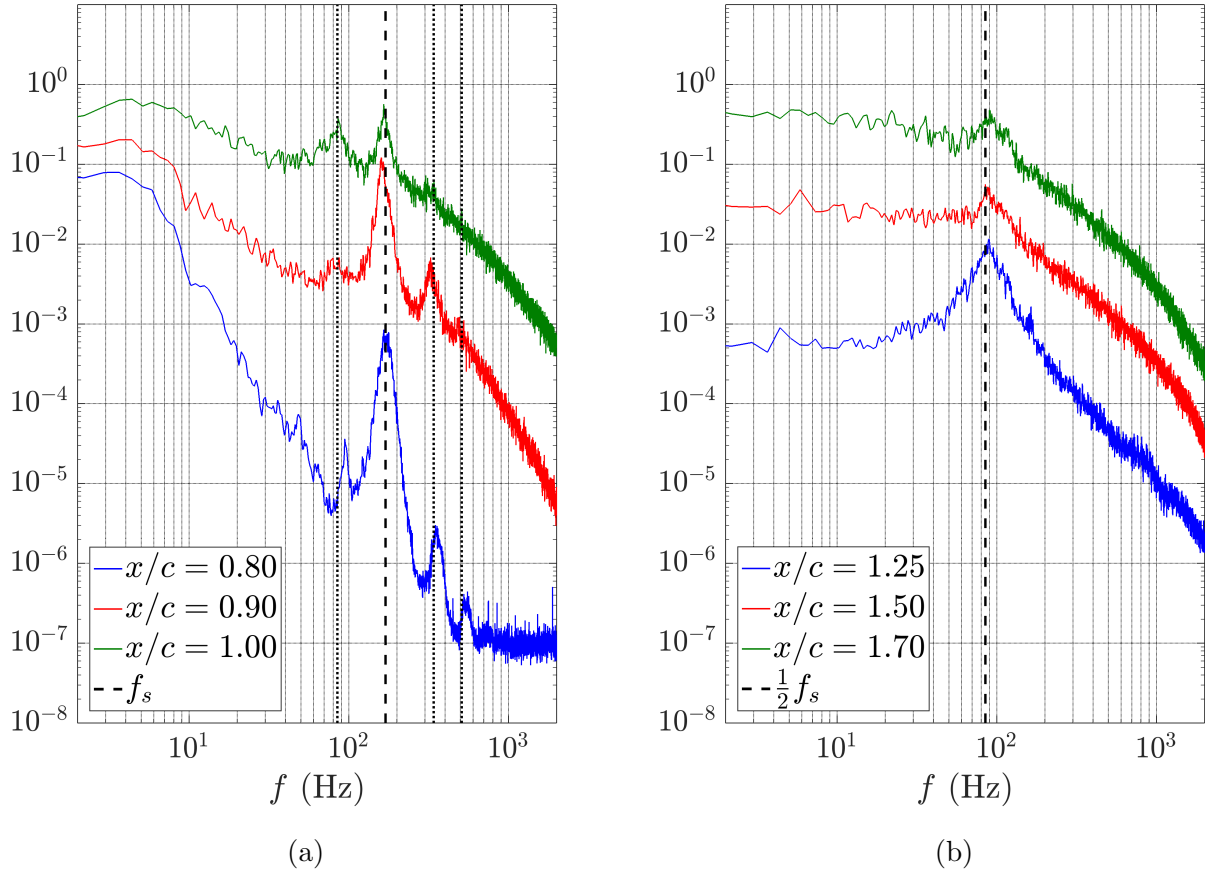


Figure 5.8: Frequency spectra of the non-actuated aerofoil at $\alpha = 0^\circ$: a) 3 locations superclose to the trailing edge (the thick dotted lines signify the first subharmonic and first two superharmonics); b) 3 locations in the near wake.

Coherent structures have also been identified in the shear layer [26, 96, 97]. The frequency spectra at the y/c location with largest r.m.s. velocity are plotted in Figure 5.8a at 3 points close to the trailing edge. Owing to the length of the hotwire probe support, $x/c = 0.8$ is as far upstream as the measurements could be taken. This means that it was not possible to obtain measurements in the purely laminar portion of the separated region.

A distinct peak at roughly 170 Hz appears at $x/c = 0.8, 0.9$ and 1.0 . Such distinct peaks in the shear layer have been attributed to vortex shedding due to shear layer roll up [7, 26]. The appearance of subharmonics and superharmonics (shown in figure 5.8a by dashed lines) indicate non-linear interactions which result in transition to turbulence [98]. Shear layer transition is associated with the decay of roll-up vortices [26, 99] and consistent with this is the disappearance of the spectral peak at $x/c = 1$.

The spectra in Figure 5.8a tie in well with the observations in Figure 5.5. The termination of the region of strong, concentrated vorticity – the shear layer – coincides with the appearance of harmonics in the spectra, which indicates non-linear interactions and consequently the onset of transition. The typical turbulent spectra at the trailing edge are consistent with the location of a reverse-flow vortex in Figure 5.5 which, according to Horton [3], should be confined to the turbulent portion of the flow. In addition, the evolution of the spectral peaks from $x/c = 0.8$ to $x/c = 1.70$ reveals an interesting result. The frequency of 85 Hz that is found in the wake, and attributed to the shedding of coherent vortex structures is actually the first subharmonic of the shear layer frequency, f_s , which results from shear layer roll-up. Yarusevych *et al.* [26] observed similar behaviour and attributed the subharmonic to the merging of the shear layer vortices.

5.2 Actuated Aerofoil

It was shown in Chapter 3 that the peak-to-peak displacement of the surface was a function of both V_f and V_A . Since only the effect of frequency is investigated, V_A was modified for each V_f examined to maintain a constant peak-to-peak displacement of 0.4 mm at $x/c = 0.4$. This is comparable to the value of Munday and Jacob [77] of 0.2% of the chord (corresponding to 0.3 mm in the current setup). Furthermore, it was shown in Section 3.5.1 that a second vibration mode takes effect when $V_f > 80$ Hz so the actuation frequencies were kept below this value.

Effect of Actuation on Flow Development

In Section 5.1.1, it was found that the aerodynamic force coefficients produced by the model were very poor in comparison with a NACA 4415 at moderate [4] and high [5] Re_c . Separation without reattachment was observed at all angles of attack and this explained the poor aerodynamic performance [34, 33, 92, 22]. From Gad El Hak's [100] categorisation, the control objective pursued in this thesis therefore falls into the category of 'separation control.'

The flow fields in Figure 5.9 shows time-averaged contours of $\bar{u}/U_\infty$ when $\alpha = 0^\circ$ for four frequencies. Figure 5.10 shows the same frequencies, with the smaller field of view mentioned in Section 5.1.2 to provide an appreciation of the reverse flow regions. The non-actuated case in Figures 5.9a and 5.10a are the same as the baseline cases in Figures 5.3a and 5.4a. This should be used as reference when comparing with the actuated cases from b-d, which display the flow fields when the surface was actuated at 10 Hz, 40 Hz and 70 Hz respectively.

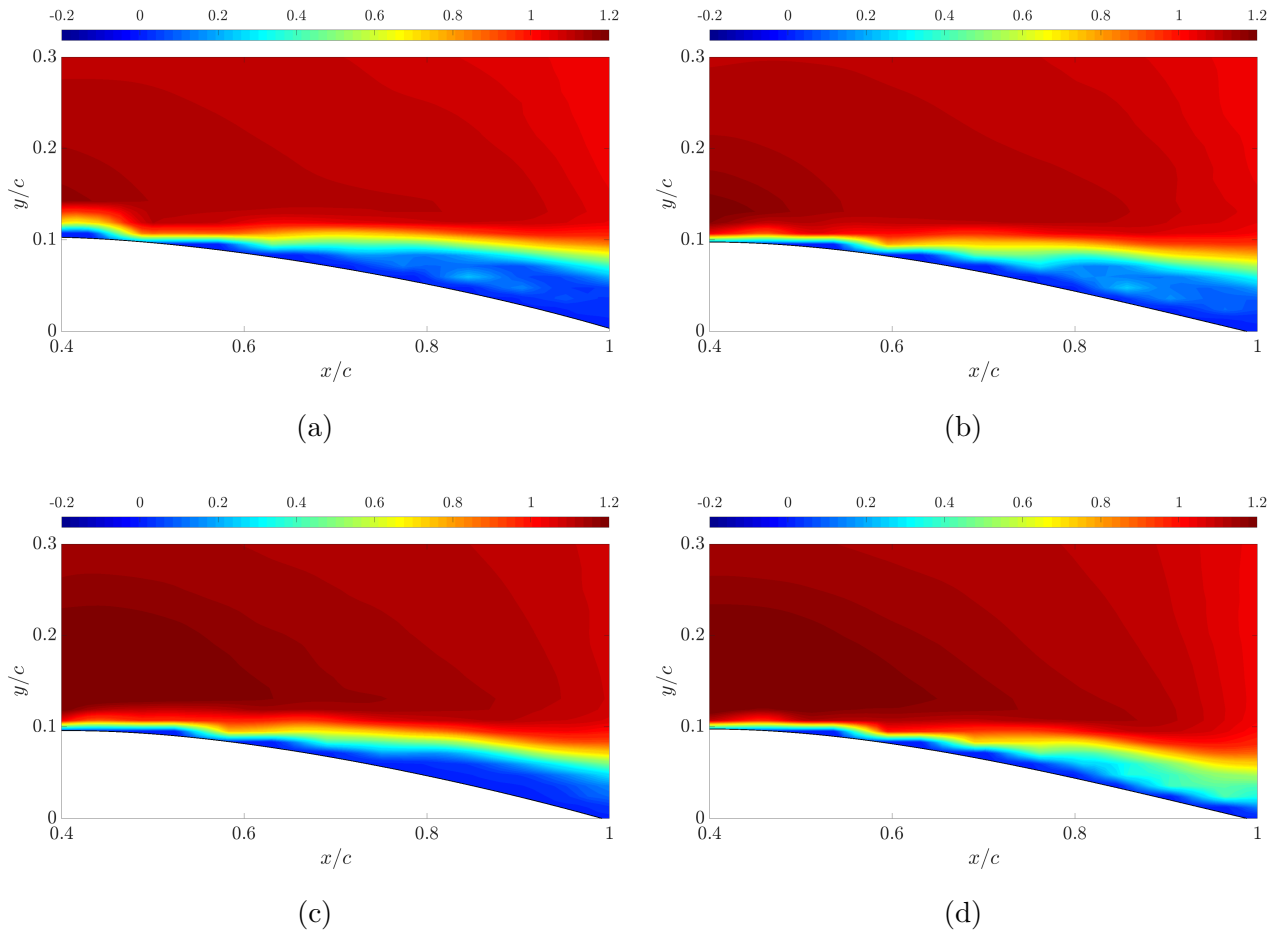


Figure 5.9: Iso-contours of $\bar{u}/U_\infty$ around the aerofoil when $\alpha = 0^\circ$: a) $V_f = 0$ Hz; b) $V_f = 10$ Hz; c) $V_f = 40$ Hz; d) $V_f = 70$ Hz.

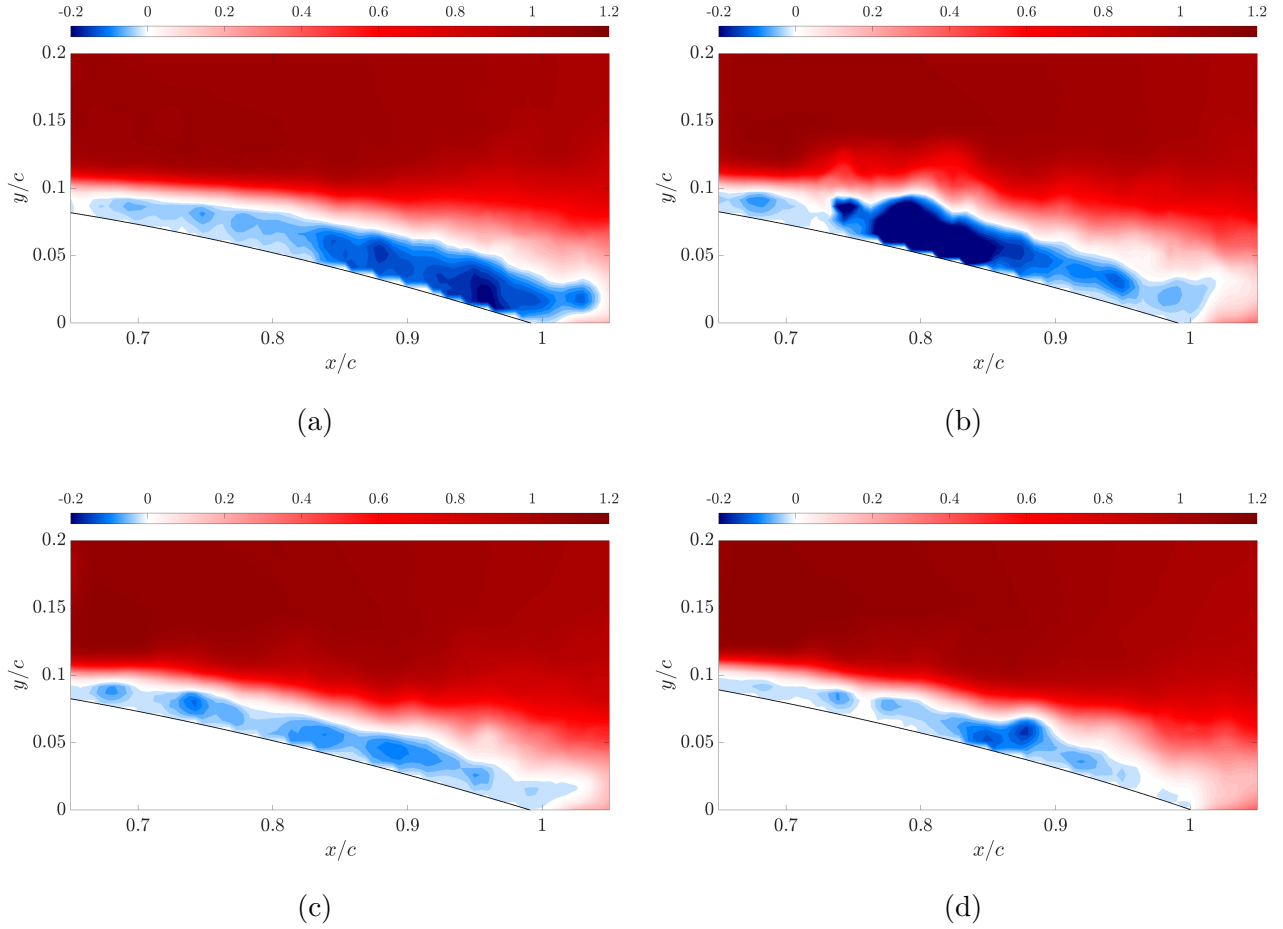


Figure 5.10: Iso-contours of $\bar{u}/U_\infty$ around the aerofoil when $\alpha = 0^\circ$ with field of view focused on separation region: a) $V_f = 0$ Hz; b) $V_f = 10$ Hz; c) $V_f = 40$ Hz; d) $V_f = 70$ Hz.

By comparing the actuated cases with the baseline case, a noticeable reduction in the size of the separation region occurs when $V_f = 70$ Hz. However, there is little difference when $V_f = 10$ Hz. A small reduction can be seen when $V_f = 40$ Hz but it is not as clear as at $V_f = 70$ Hz. It is expected then that since the size of the separation region is more-or-less the same as the baseline case, actuation at 10 Hz will have little effect on the performance of the aerofoil. Also a significant reduction in separation at $V_f = 70$ Hz is expected to result in a significant performance recovery. Further analysis to ascertain whether any control effectiveness is achieved at $V_f = 40$ Hz is required.

A wake with width roughly equal to the aerofoil thickness was presented in Figure 5.7. Since the wake profile is directly linked with the profile drag, such a wide wake with a large velocity deficit indicates a high drag value, which is exactly what is shown in Figure 5.1b. Wake

profiles were obtained at the same locations for the actuated cases and are presented in Figure 5.11.

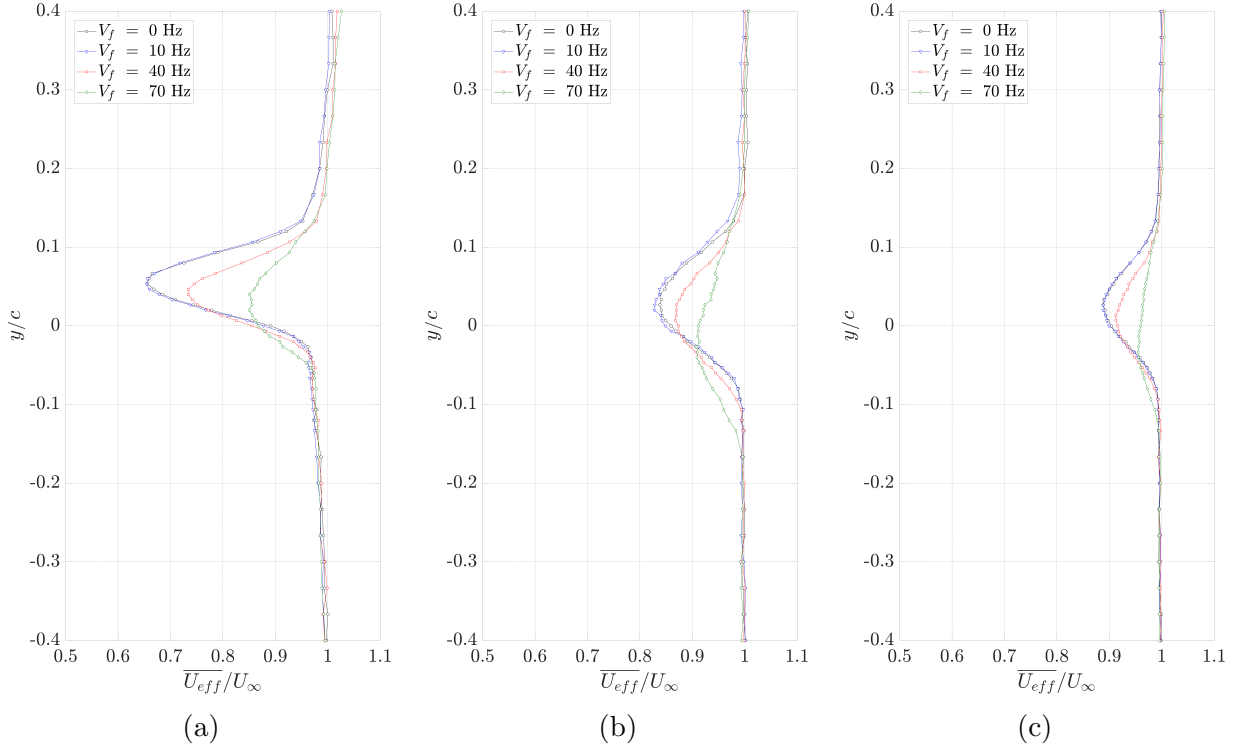


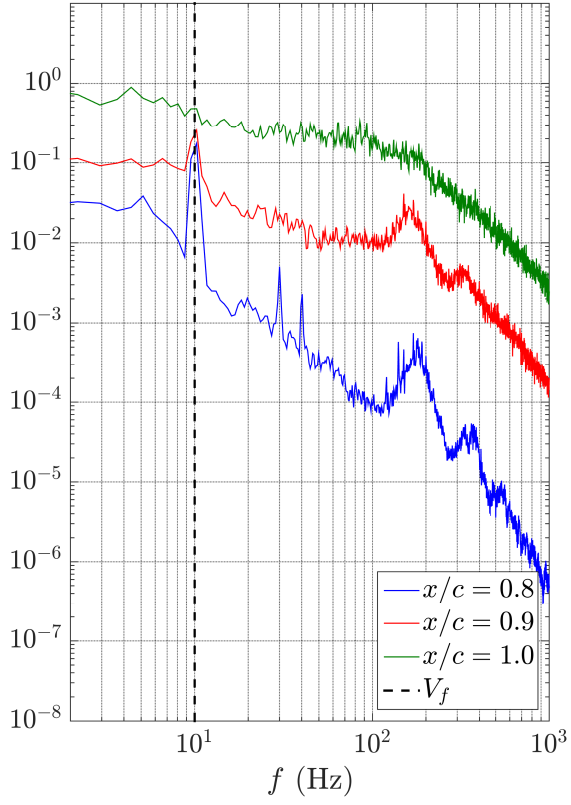
Figure 5.11: Profiles of $\overline{U_{eff}}/U_\infty$ in the wake of the aerofoil at 4 actuation frequencies: a) $1.25c$, b) $1.50c$, c) $1.70c$.

The wake profile is unchanged when $V_f = 10$ Hz, while a significant reduction in the velocity deficit at all streamwise locations occurs when $V_f = 70$ Hz. Actuation at 70 Hz also produces a thinner wake. This ties in with the observations from the time-average flow fields in Figure 5.9 and adds weight to the hypothesis that $V_f = 70$ Hz is capable of significant performance improvements while $V_f = 10$ Hz has little affect. When $V_f = 40$ Hz a small reduction the separation region occurred, shown in Figure 5.9c, but was not as substantial as $V_f = 70$ Hz. This is consistent with the wake profiles that indeed reveal a reduction in the velocity deficit and wake width when $V_f = 40$ Hz but neither is as significant as found for $V_f = 70$ Hz.

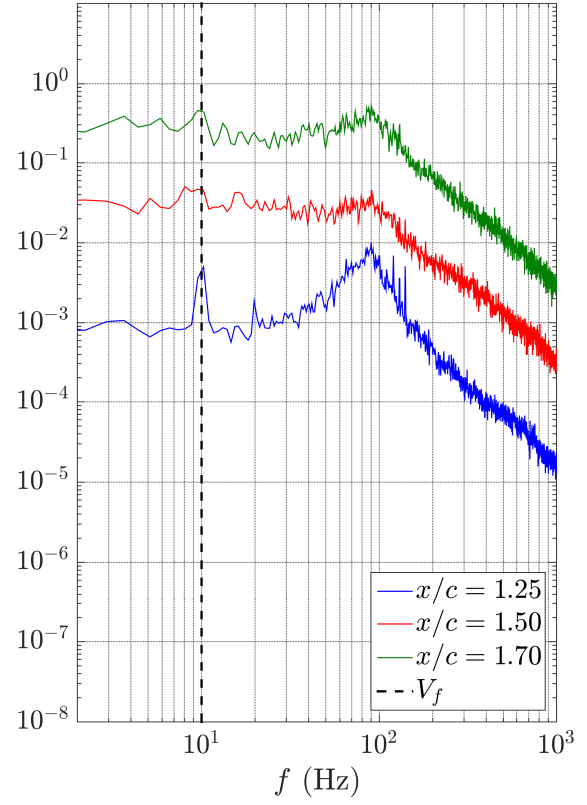
5.2.1 Coherent Structures

In Section 5.1.3, the spectra obtained from the hotwire signals provided an insight into the presence of coherent structures in the shear layer and wake. These are important features since they promote the entrainment of momentum into the near-wall region. Spectral information on the natural development of such structures was provided in Figure 5.8. Figures 5.12-5.14 show the equivalent for the actuated aerofoil.

When $V_f = 10$ Hz (Figure 5.12), a strong peak is present at 10 Hz in the shear layer and in the near wake. Nevertheless, the natural peaks at shear layer frequency and vortex shedding frequency remain, meaning that the flow still has a strong tendency to develop in its natural (i.e. non-actuated) way. This would agree with the observation that the flow field and wake profile are more-or-less the same when $V_f = 0$ and 10 Hz. When $V_f = 40$ Hz (Figure 5.13), strong peaks at 40 Hz are present at all locations, as are its harmonics. Evidence of activity at the naturally occurring frequencies remains, with corresponding peaks in the shear layer upstream of turbulent transition. The harmonics are particularly amplified around these natural frequencies. Such a phenomenon was observed by Sato *et al.* [74] and attributed to the presence of linear unstable modes. It can be considered that actuation at 40 Hz promotes the development of large coherent structures (LCS) at the actuation frequency, V_f , and therefore modifies the flow development. Natural frequencies are still strongly amplified in the shear layer but the wake is clearly dominated by the V_f . This agrees with the reduction in the size of the wake at 40 Hz in Figure 5.11 but only modest changes in the flow field around the aerofoil in Figure 5.9c.

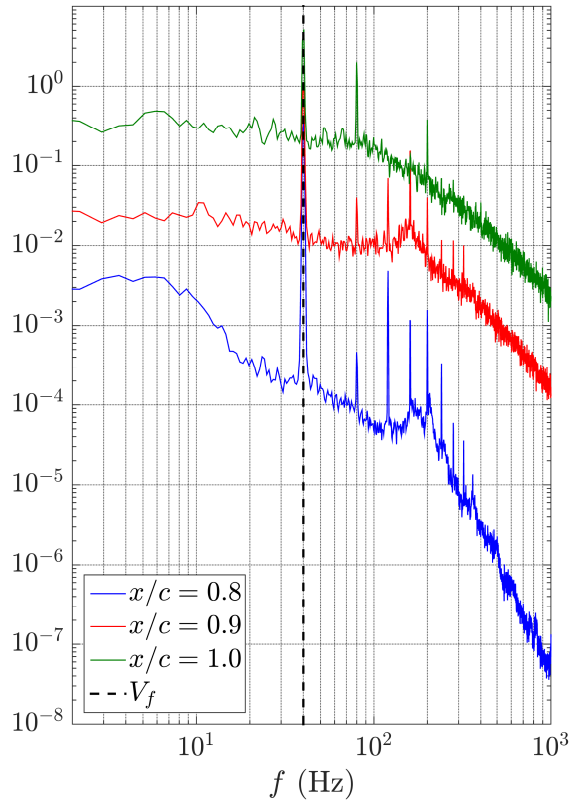


(a)

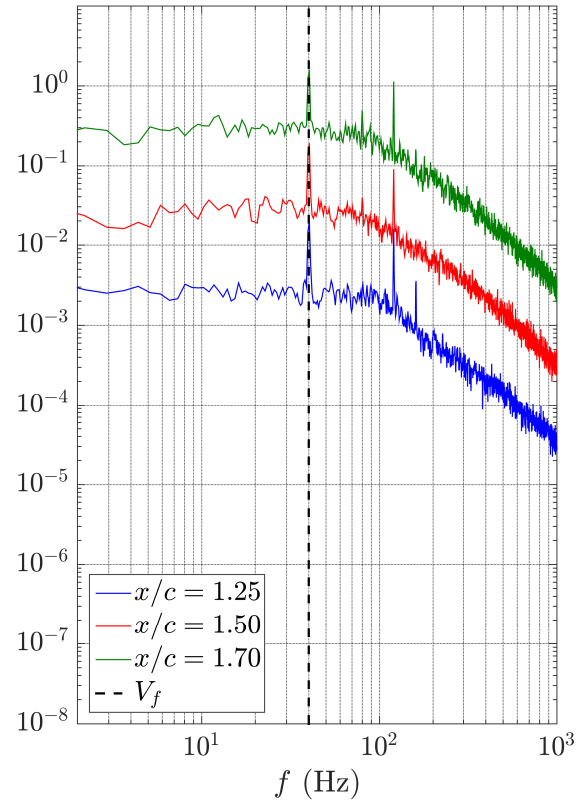


(b)

Figure 5.12: Frequency spectra at (a) the trailing edge of the aerofoil and (b) in the near wake when $\alpha = 0^\circ$ and $V_f = 10$ Hz. Dashed lines represent the actuation frequency.



(a)



(b)

Figure 5.13: Frequency spectra at (a) the trailing edge of the aerofoil and (b) in the near wake when $\alpha = 0^\circ$ and $V_f = 40$ Hz. Dashed lines represent the actuation frequency.

Actuation at 70 Hz (Figure 5.14) causes the spectra at all locations to be entirely dominated by V_f and its harmonics. This therefore means the surface motion is generating LCS at the V_f and these are dominating the flow development. Such structures are known to enhance momentum entrainment to the near-wall region and therefore suppress separation [74, 12]. This is the reason for the significant reductions in separation and velocity deficit when $V_f = 70$ Hz.

It is worth highlighting the strong presence of a sub-harmonic towards the trailing edge. It was mentioned previously that Yarusevych *et al.* attributed such harmonics to vortex merging [26]. Embacher and Fasel [101] also observed such behaviour when investigating periodic forcing of a separated laminar shear layer on a flat plate and showed that vortex merging was indeed responsible. This was not investigated during the experiments.

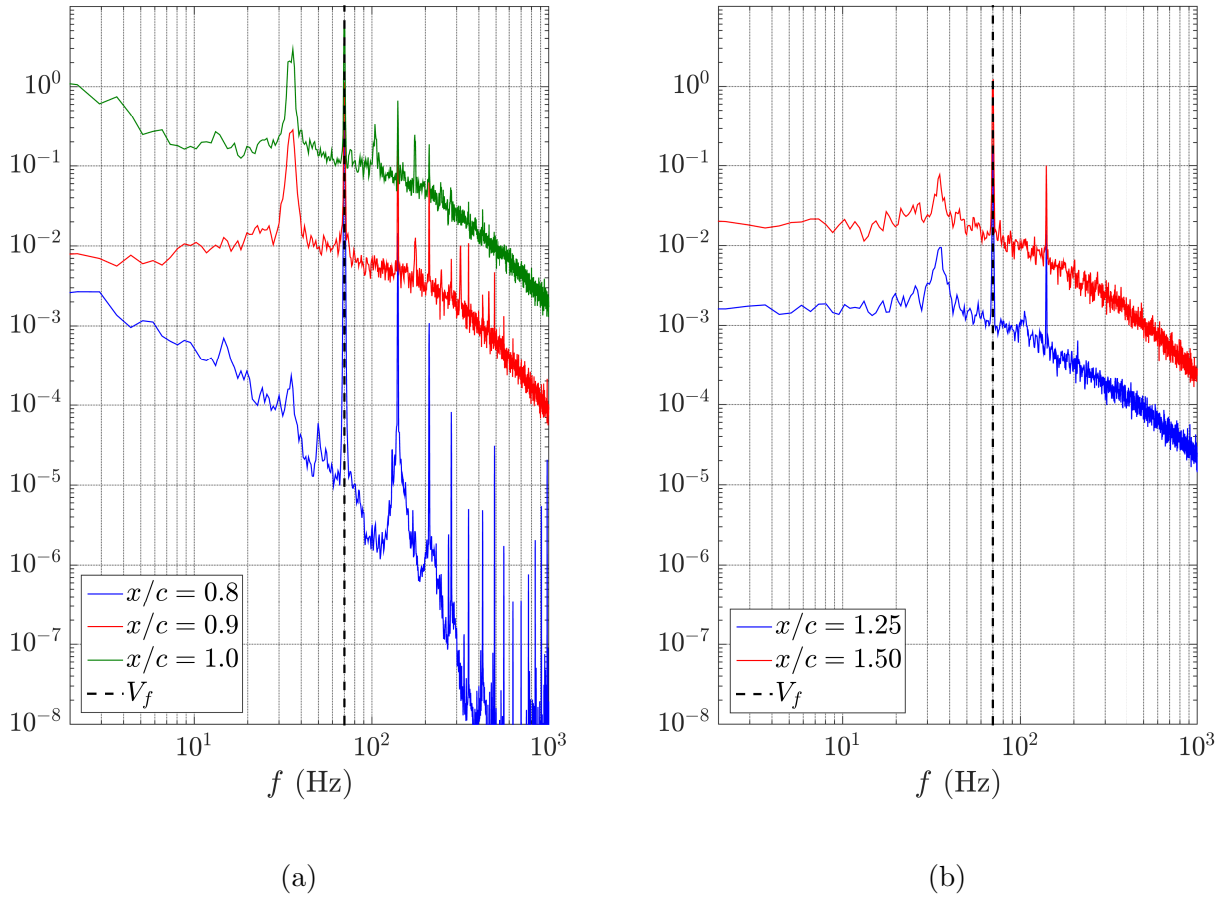


Figure 5.14: Frequency Spectra at (a) the trailing edge of the aerofoil and (b) in the near wake when $\alpha = 0^\circ$ and $V_f = 70$ Hz. Dashed lines represent the actuation frequency.

5.2.2 Flow Fields at 5° , 10° and 15°

Flow fields for the actuated aerofoil when $\alpha = 5^\circ$, 10° and 15° are shown in Figures 5.15-5.17. When $\alpha = 5^\circ$ (Figure 5.15) broadly the same observations can be made as those when $\alpha = 0^\circ$ (Figures 5.9). There is no significant reduction in the size of the separated region when $V_f = 10$ Hz when compared with the baseline case while the shear layer seems to reattach when $V_f = 70$ Hz. The case of $V_f = 40$ Hz exhibits a modest reduction in separation, more significant than it was when $\alpha = 0^\circ$, but not nearly as large as when $V_f = 70$ Hz. Again, it is expected that the aerofoil performance when $V_f = 10$ Hz will be similar to that of the non-actuated case at this angle of attack but improvements would be expected for $V_f = 40$ Hz and, more significantly, for 70 Hz.

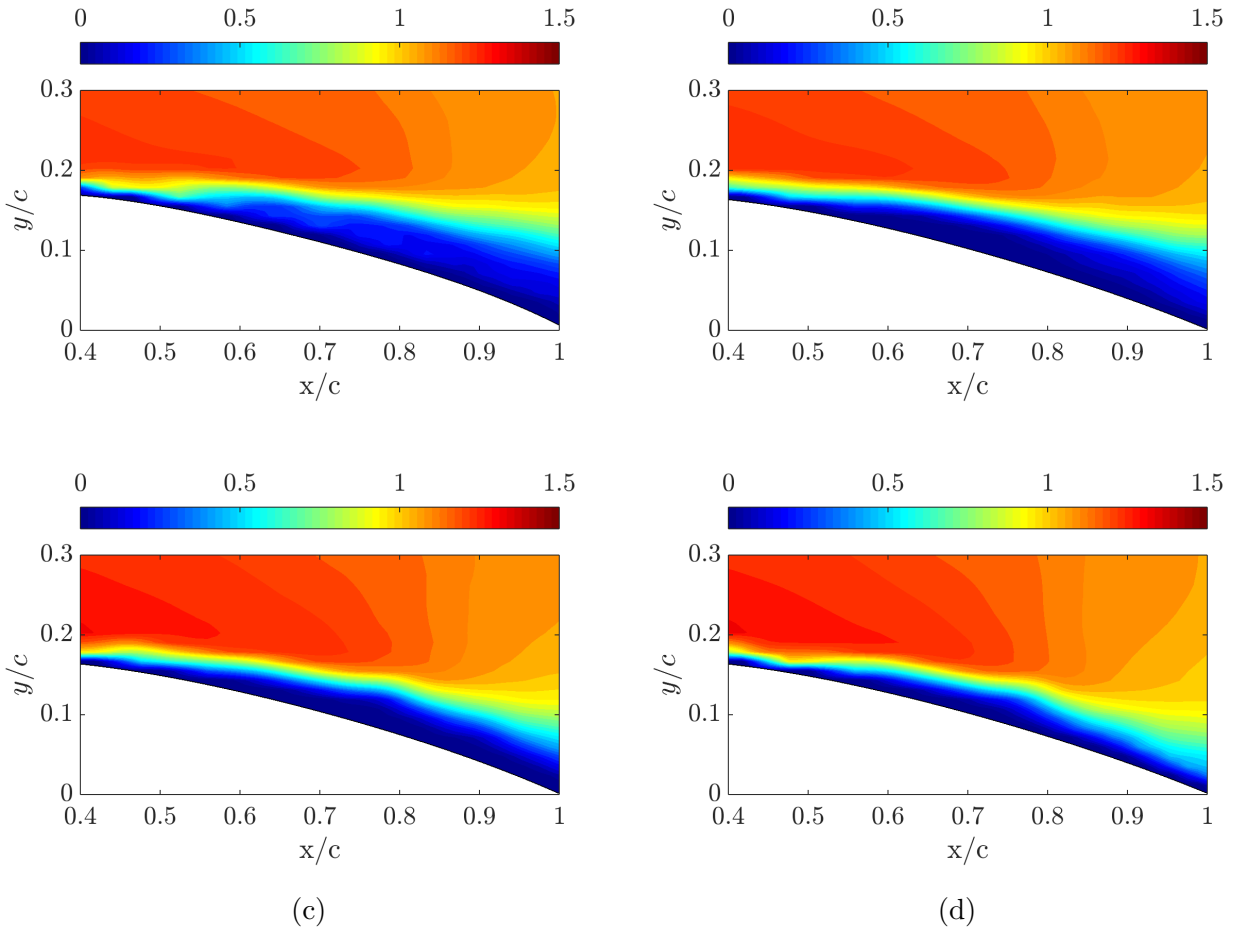


Figure 5.15: Iso-contours of $|\bar{U}|/U_\infty$ around the aerofoil when $\alpha = 5^\circ$: a) $V_f = 0$ Hz; b) $V_f = 10$ Hz; c) $V_f = 40$ Hz; d) $V_f = 70$ Hz.

When $\alpha = 10^\circ$ (Figure 5.16), $V_f = 70$ Hz again causes a large reduction in the size of the separated region. However, at this angle of attack, $V_f = 40$ Hz is also observed to exhibit significant control effectiveness. There is even a small reduction in the size of separation at $V_f = 10$ Hz at this angle.

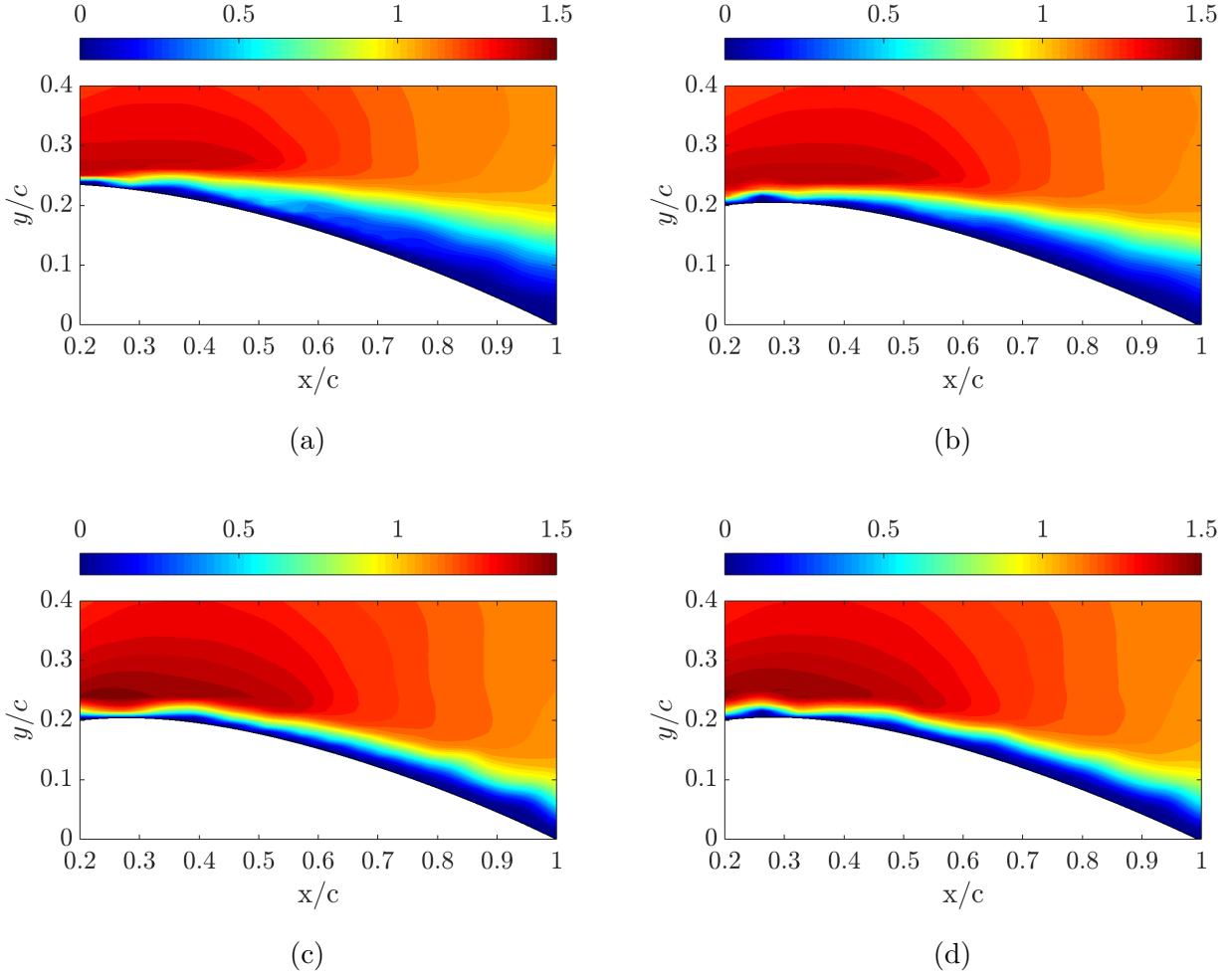


Figure 5.16: Iso-contours of $|\bar{U}|/U_\infty$ around the aerofoil when $\alpha = 10^\circ$: a) $V_f = 0$ Hz; b) $V_f = 10$ Hz; c) $V_f = 40$ Hz; d) $V_f = 70$ Hz.

At the post-stall angle of 15° (Figure 5.17), the massively separated flow has been suppressed to a large extent when $V_f = 70$ Hz i.e. the stall has been delayed. Actuation with $V_f = 40$ Hz does not entirely remove the separated region but it is noticeably reduced when compared to the baseline flow field. Again, a modest reduction in the size of the separated region occurs when $V_f = 10$ Hz. Actuation can therefore be expected to greatly improve aerofoil performance at high angles.

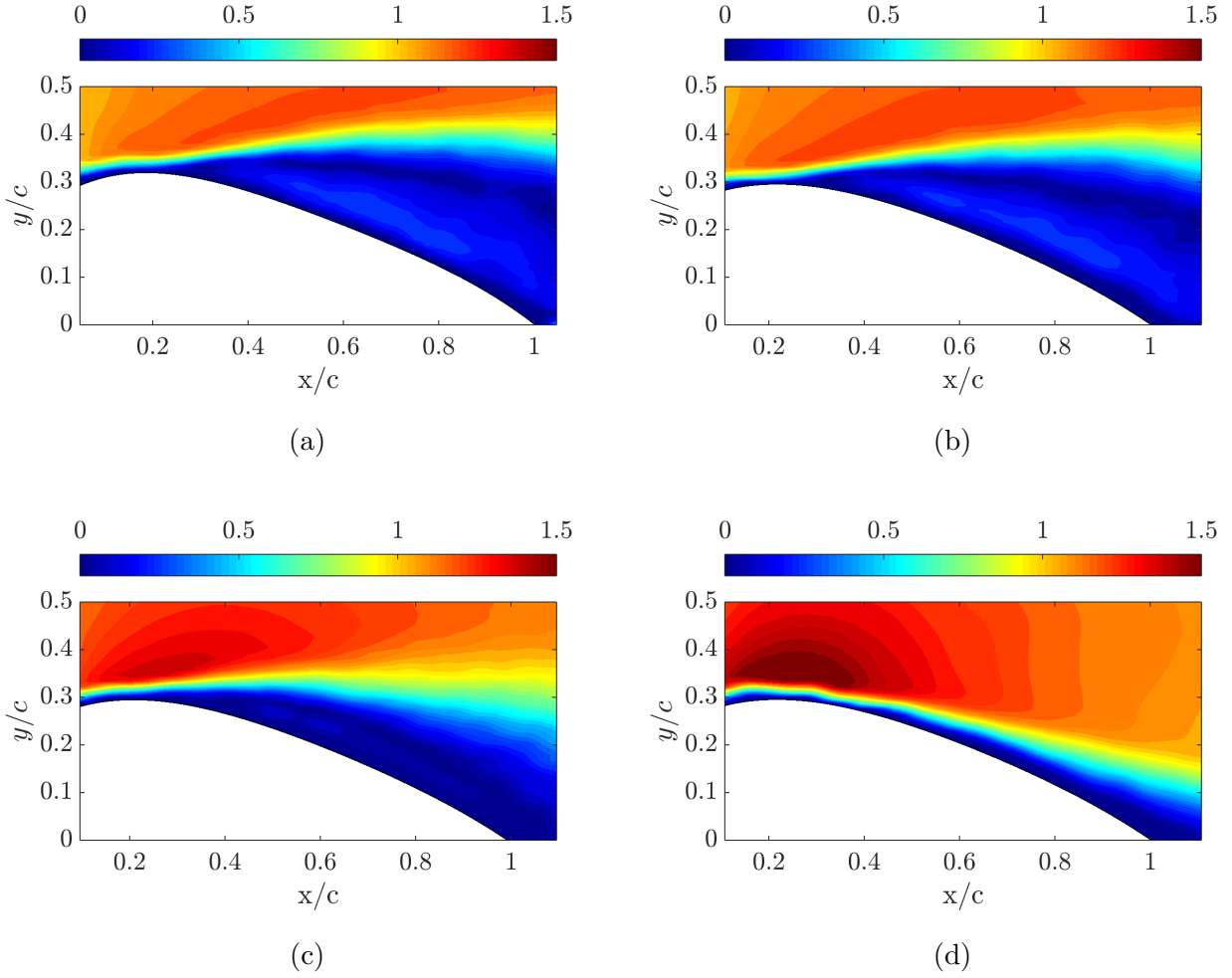


Figure 5.17: Iso-contours of $|\bar{U}|/U_\infty$ around the aerofoil when $\alpha = 15^\circ$: a) $V_f = 0$ Hz; b) $V_f = 10$ Hz; c) $V_f = 40$ Hz; d) $V_f = 70$ Hz.

5.2.3 Actuated Aerofoil Performance

The above discussion regarding the flow development around the aerofoil highlighted reductions in the size of the separated region and velocity deficit in the wake when $V_f = 40$ Hz and 70 Hz. In contrast, when $V_f = 10$ Hz there is very little changed compared with the baseline case, albeit there is some evidence of separation reduction at higher angles. It is therefore expected that improvements in aerofoil performance would result from actuation at 40 Hz and, more significantly, at 70 Hz. Whereas $V_f = 10$ Hz would have little effect. Figure 5.18 shows the force coefficients as a function of α and corroborate these findings. The C_L curves for the non-actuated and $V_f = 10$ Hz cases are very similar and result

from the actuation's inability to affect the natural flow development, which was shown in Figure 5.12 (note: there is a slight performance improvement when $V_f = 10$ Hz and this will be touched on with the aid of the computational results in Chapter 8). At higher values of V_f , the previously reported reduction in the size of the separated regions and velocity deficit have translated into significant improvements in performance. For example, when $V_f = 70$ Hz the regions of separation were reduced significantly at all angles of attack and correspondingly a significant increase in C_L and decrease in C_D is observed for all values of α . Furthermore, Figure 5.18 shows that when $V_f = 40$ Hz the effect on C_L and C_D increases with α which would support the greater reduction in separation at $\alpha = 10^\circ$ and 15° when compared with the lower angles. In addition, $V_f = 70$ Hz is capable of delaying the onset of stall to beyond 16° . As a consequence $C_{L_{max}}$ is increased by roughly 30%, from 0.9 to 1.2.

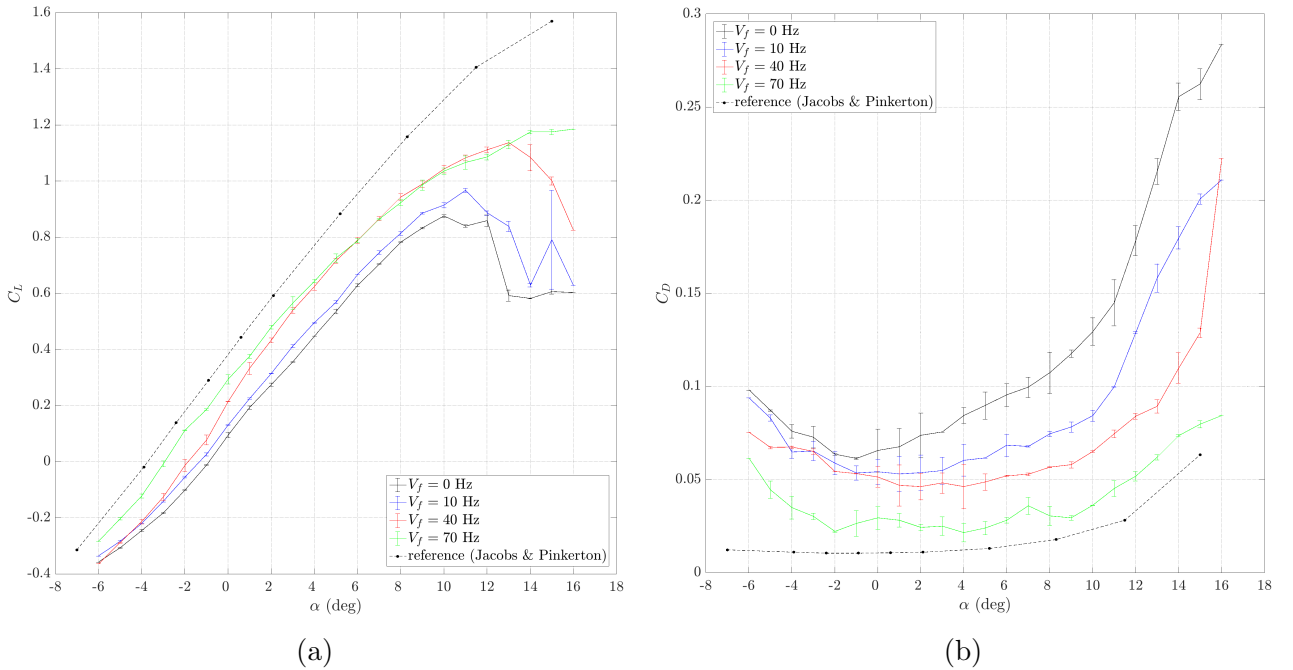


Figure 5.18: Aerodynamic force coefficients obtained from the actuated aerofoil compared to the baseline case and the high Re_c reference case from Jacobs and Pinkerton [5] : a) C_L ; b) C_D .

The values at each α in Figure 5.18 are the mean of 4 realisations of the experiment. The error bars represent the standard deviation of the values and are plotted to show the repeatability. There are a few ‘spikes’ present in the C_D curves, most noticeably at $\alpha = -2^\circ$ and 4° when $V_f = 70$ Hz but also at $\alpha = 4^\circ$ when $V_f = 40$ Hz. Since the L/D ratio is

calculated by dividing with such small numbers, these spikes cause a discontinuity in the L/D curve. For this reason they were removed and a 4th-order polynomial was fitted through the remaining points. This polynomial was then used for each angle to plot the L/D curve seen in Figure 5.19. A four-fold increase in L/D at 0° and L/D_{max} is achieved when $V_f = 70$ Hz although this is still some way short of the high Re_c performance. This is expected since the improvements in both C_L and C_D shown in Figure 5.18, while significant, did not completely recover the high Re_c coefficients.

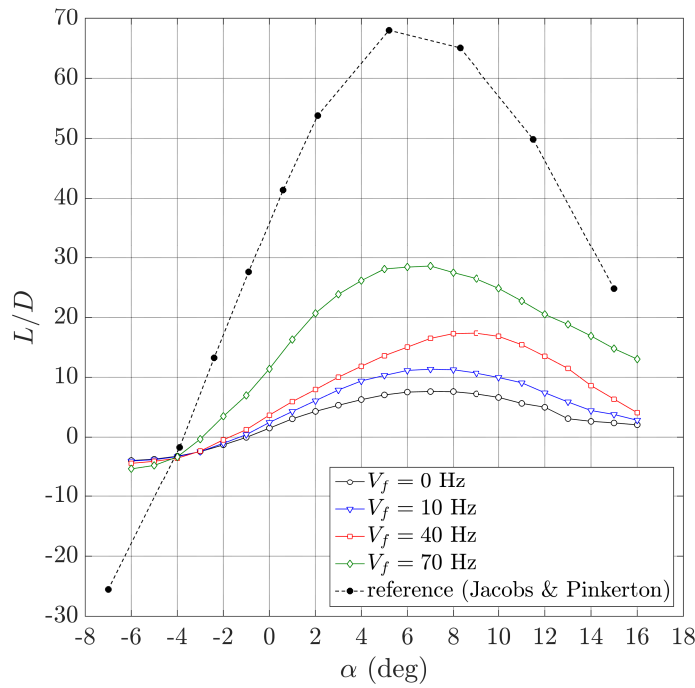


Figure 5.19: L/D ratio obtained from the actuated aerofoil compared with the baseline case.

Energy Consumption

Throughout the course of this chapter it has been established that actuation at 70 Hz is capable of causing a change in the development of the flow field which, results in a reduction in the drag. It is therefore shown that the control method being investigated is capable of saving energy. The efficiency of the control method however is determined by the amount of energy saved relative to the amount consumed. Since this is an electronic system the energy required can be determined from the electrical input signal. An example of 10 cycles of the

pre-amplified voltage and current signals are shown below in Figure 5.20.

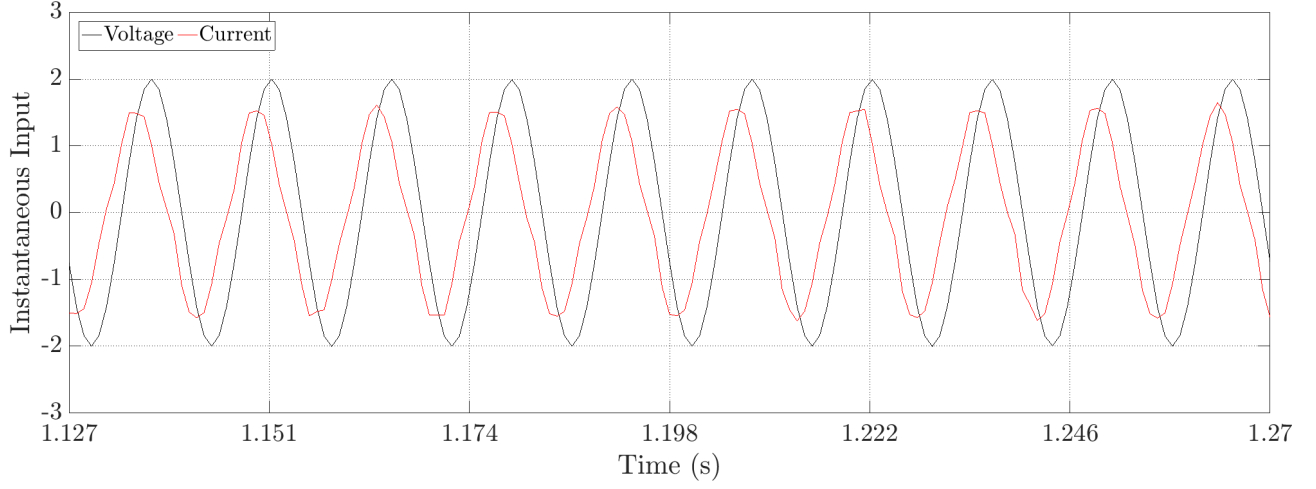


Figure 5.20: Signals of instantaneous electrical output when $V_f = 70$ Hz.

The average power consumption of the actuators over the 10 cycles is given by Equation 5.1 [102] and was calculated numerically using the trapezoidal rule to be 1.08 Watts.

$$P_{avg} = \frac{1}{T} \int_0^T V(t) \cdot I(t) dt \quad (5.1)$$

By multiplying by the freestream velocity (m/s), the reduction in drag by actuation can be expressed in terms of power saving. Figure 5.21 shows the power saving at each condition tested.

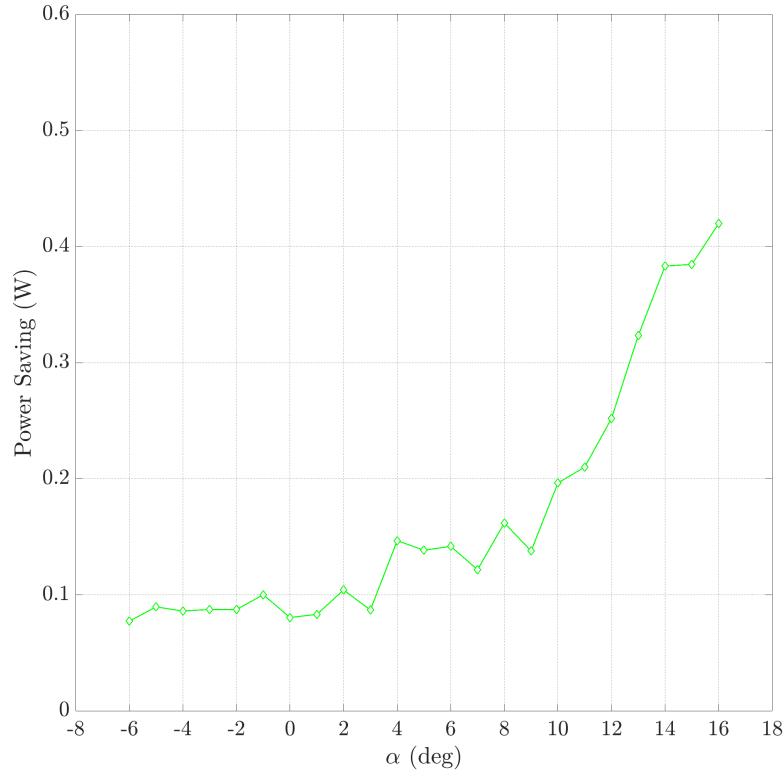


Figure 5.21: Power saving against Angle of Attack.

It can be seen in Figure 5.21 that the power saving increases with α . This is expected since the effect of separation, and consequently its control, is greater at higher angles. The maximum power savings occur at the conditions at which the actuation was able to suppress stall. Figure 5.22 shows how efficient these energy savings are when the energy consumed is considered. This is expressed as in Equation 5.2, at each condition.

$$\text{Efficiency} = \frac{\text{Power Saving}}{\text{Power Consumption}} \quad (5.2)$$

With an efficiency rating of 0.1 the actuators are consuming 10 times the energy they are saving for $\alpha < 4^\circ$, which is clearly very inefficient. Since the energy consumption is a function of the V_f and V_A , it remains constant as α increases, however the energy saving increases. This leads to the system becoming more efficient at large values of α , reaching 40% efficiency when $\alpha = 16^\circ$.

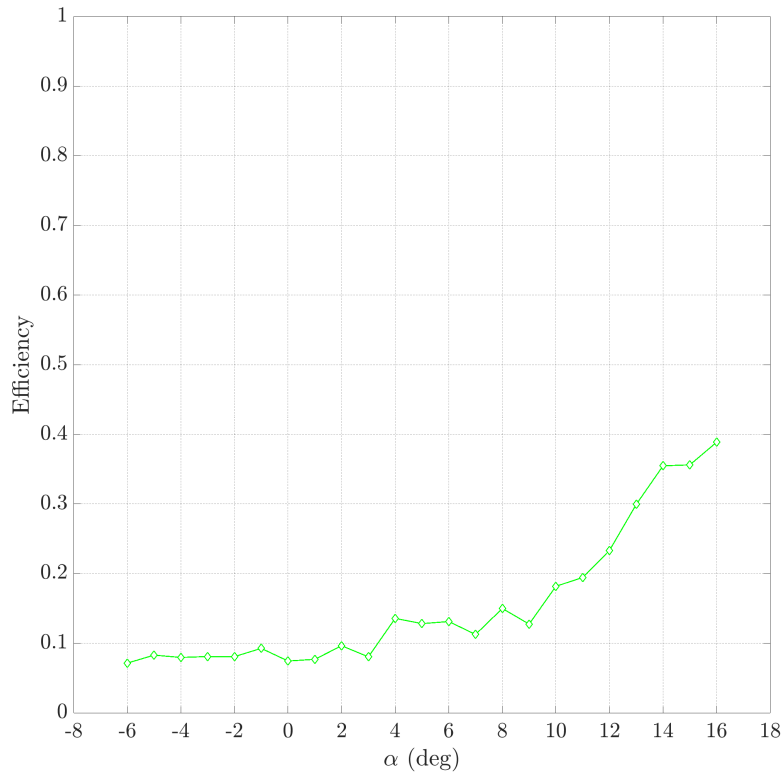


Figure 5.22: Efficiency of the Actuation method for each condition tested.

Considering how un-optimised the model is, these results are not discouraging. It is likely that the efficiency of this technique could be improved with different materials/thickness for the skin and a better distribution of actuators along the span. In addition, this discounts the increase in C_L and $C_{L_{max}}$ achieved when $V_f = 70$ Hz.

5.3 Summary

The experiments revealed that the flow development around the non-actuated aerofoil is similar to the separation without reattachment flow mode described in the literature. Large separated regions and a wide wake were observed which are responsible for the significant performance losses when compared with the same aerofoil geometry at higher Re_c . Analysis of the spanwise vorticity and frequency spectra suggested that the initially laminar shear layer experienced a ‘roll-up’ and subsequent vortex shedding before transition to turbulence occurred towards the trailing edge. Vortex shedding was then set up in the wake.

When the surface was actuated, the frequency spectra revealed peaks at the actuation frequency. However, when $V_f = 10$ Hz the effect was not strong enough to dominate the flow development and the naturally occurring frequencies remained. This resulted in little change in the separated region and wake and consequently only a modest effect on the aerofoil performance was observed. In contrast, when $V_f = 70$ Hz the spectra were dominated by the actuation frequency, with the naturally occurring frequencies absent. The strong, defined peaks in the spectra are the result of the periodic surface motion producing large coherent structures. These structures add momentum entrainment and cause significant reduction in separation and velocity deficit in the wake. This was found to translate into a large improvement in performance.

Owing to the time-averaged nature of the flow field data and the limitations of the spatial resolution, the presence of shear layer ‘roll-up,’ and coherent structures could not be proven beyond doubt. Therefore, while the experiments have proved that periodic surface morphing is capable of significant performance improvements, the control mechanism thus far remains not fully explained. Direct Numerical Simulations will be presented to fill in some of the gaps in the experimental data and support the hypotheses made in this chapter. This is the subject of the next two chapters.

Chapter 6

Direct Numerical Simulations of a Static Aerofoil

In Direct Numerical Simulations (DNS), the equations governing fluid flow are solved in a very fine mesh so that all length and time scales are resolved. The smallest length and time scales dictate the size of computational grid and the time-step respectively. The number of cells increases at a rate of $Re_c^{9/4}$ [103] and therefore only relatively low Re_c conditions can be simulated. High Re , complex flows are modelled using turbulence models, based on the Reynolds-Averaged Navier-Stokes (RANS) equations. The Large Eddy Simulation (LES) is an intermediate approach that solves for the large scales directly and models the scales smaller than a characteristic filter length, which are closer to being isotropic. This imposes a less debilitating constraint on grid size and time-step.

Generally LES has proved very successful in simulating turbulent flows, however it is still difficult to model transitional flows correctly [42]. Given the importance of transition of the separated shear layer, DNS is the most suitable option for the present study. Increases in computational power have recently made DNS on aerofoils at low Reynolds numbers possible and this approach will therefore be used in this thesis. This chapter will first introduce the development of the computational model and then go on to present and discuss the results.

6.1 The Governing Equations

When applied to a fluid flow, conservation of mass and Newton's second law become the continuity, Equation 6.1, and Navier-Stokes equations, Equation 6.2, respectively. Using Einstein's summation notation of repeated indices to concisely express the system in three dimensions gives

$$\frac{\partial \rho}{\partial t} + \frac{\partial (\rho u_i)}{\partial x_i} = 0 \quad (6.1)$$

$$\frac{\partial (\rho u_i)}{\partial t} + \frac{\partial}{\partial x_j} (\rho u_j u_i) = - \frac{\partial}{\partial x_i} \left(P + \frac{2}{3} \mu \frac{\partial u_i}{\partial x_i} \right) + \frac{\partial}{\partial x_j} \left[\mu \left(\frac{\partial u_i}{\partial x_j} + \frac{\partial u_j}{\partial x_i} \right) \right] \quad (6.2)$$

where u_i are the 3 velocity components in the 3 Cartesian directions while P , ρ and μ are the pressure, density and dynamic viscosity respectively. At low Mach numbers the flow can be considered incompressible i.e. its density, ρ , can be considered constant and can be factored out of the derivatives. This reduces the continuity equation to its incompressible form.

$$\frac{\partial u_i}{\partial x_i} = 0 \quad (6.3)$$

Equation 6.2 can now be simplified by the incorporation of Equation 6.3 because $\partial u_i / \partial x_i$, which appears within brackets in the first term on the right hand side, is equal to zero resulting in

$$\frac{\partial u_i}{\partial t} + \frac{\partial}{\partial x_j} (u_j u_i) = - \frac{\partial p}{\partial x_i} + \frac{\partial}{\partial x_j} \left[\nu \left(\frac{\partial u_i}{\partial x_j} + \frac{\partial u_j}{\partial x_i} \right) \right] \quad (6.4)$$

In Equation 6.4 all terms have been divided by the constant density so that $\nu = \mu / \rho$ and $p = P / \rho$, where ν is the kinematic viscosity. Furthermore, if the temperature remains constant (as in an isothermal flow) viscosity is constant and can also be factored out of the derivatives to give

$$\frac{\partial u_i}{\partial t} + \frac{\partial}{\partial x_j} (u_j u_i) = - \frac{\partial p}{\partial x_i} + \nu \frac{\partial}{\partial x_j} \left(\frac{\partial u_i}{\partial x_j} \right) + \nu \frac{\partial}{\partial x_j} \left(\frac{\partial u_j}{\partial x_i} \right) \quad (6.5)$$

The final term of the right hand side of Equation 6.5 is zero due to the continuity equation and simplifies to the incompressible, isothermal Navier-Stokes equations

$$\underbrace{\frac{\partial u_i}{\partial t}}_{\text{transient term}} + \underbrace{\frac{\partial (u_j u_i)}{\partial x_j}}_{\text{convective term}} = - \underbrace{\frac{\partial p}{\partial x_i}}_{\text{pressure term}} + \underbrace{\nu \frac{\partial}{\partial x_j} \left(\frac{\partial u_i}{\partial x_j} \right)}_{\text{diffusive term}} \quad (6.6)$$

This system of partial differential equations (PDE) is second-order in space and non-linear and, except in the simplest of cases, has no analytical solution. Their solution is instead sought using a numerical approach.

6.2 Discretisation of the Equations

The finite volume method (FVM) is the discretisation method used in this thesis. The method requires integration of the governing equations over discrete control volumes

$$\int_V \frac{\partial u_i}{\partial t} dV + \int_V \frac{\partial}{\partial x_j} (u_j u_i) dV = - \int_V \frac{\partial p}{\partial x_i} dV + \int_V \nu \frac{\partial}{\partial x_j} \left(\frac{\partial u_i}{\partial x_j} \right) dV \quad (6.7)$$

During discretisation the PDE are transformed to a set of non-linear algebraic equations, which approximate the PDE at discrete points in space and time. The solution domain must also be discretised into a set of discrete control volumes (CV), known as cells. The details of the FVM are provided in standard textbooks such as Ferziger and Peric [104] and Versteeg and Malalasekera [105].

6.2.1 Spatial Discretisation

The discretisation of the spatial derivatives after volume integration will be presented. Their integration consists of two parts: the transformation of volume integrals into surface integrals,

followed by the approximation of the surface integrals with discrete expressions. The first part is based on Gauss' Divergence Theorem: the volume integral of the divergence of a vector is equal to surface integral of the component of the vector normal to the surface. Using this statement, Equation 6.7 can be written as

$$\int_V \frac{\partial u_i}{\partial t} dV + \oint_S n_j (u_j u_i) dS = - \oint_S n_i p dS + \oint_S \nu n_j \frac{\partial u_i}{\partial x_j} dS \quad (6.8)$$

where n_j are the components of the unit normal vector to the surface, pointing outwards from the CV. The CV faces provide the bounding surface over which the integration is applied and consequently Equation 6.8 can be expressed as the sum of the integrals over the CV faces

$$\int_V \frac{\partial u_i}{\partial t} dV + \sum_f \left(\int_{S_f} n_j (u_j u_i) dS \right) = - \sum_f \left(\int_{S_f} n_i p dS \right) + \sum_f \left(\int_{S_f} \nu n_j \frac{\partial u_i}{\partial x_j} dS \right) \quad (6.9)$$

where S_f is the surface of face f .

Precise evaluation of the integrals would require knowledge of the integrand everywhere on the surface but such information is not available. Instead, the integrand is approximated as the product of the face-centre value and the area of that face. The face-centre values are computed from nodal (i.e. cell-centre) values using second-order approximations as explained below.

The discretisation of each term in Equation 6.9 will now be introduced.

Diffusive Term

The diffusive term can be written as

$$\sum_f \left(\int_{S_f} \nu n_j \frac{\partial u_i}{\partial x_j} dS \right) = \sum_f \nu S_f \left(\frac{\partial u_i}{\partial \vec{n}} \right)_f \quad (6.10)$$

where $\frac{\partial u_i}{\partial \vec{n}}$ denotes the derivative of u_i in the face-normal, $\vec{n}$, direction and S_f is the area of face f . For the evaluation of the derivative at the face, a finite difference approximation is applied based on nodal values. The central differencing scheme was chosen for this study in which

$$\sum_f \nu S_f \left(\frac{\partial u_i}{\partial \vec{n}} \right)_f \approx \sum_f D (u_{i_A} - u_{i_P})_f \quad (6.11)$$

where $D = \nu S_f / \Delta x_f$ and represents the diffusive conductance at the cell face. Expression 6.11 is correct for orthogonal grids. If the grid is non-orthogonal, an additional correction term must be included [104]. The schematic representation of adjacent control volumes in Figure 6.1 shows the relationship between points P and A and the cell face normal vector $\vec{n}$. The above expression is the contribution of face f of the CV surrounding P to the diffusive term. Assuming grid uniformity and orthogonality, Equation 6.11 represents a second-order accurate approximation.

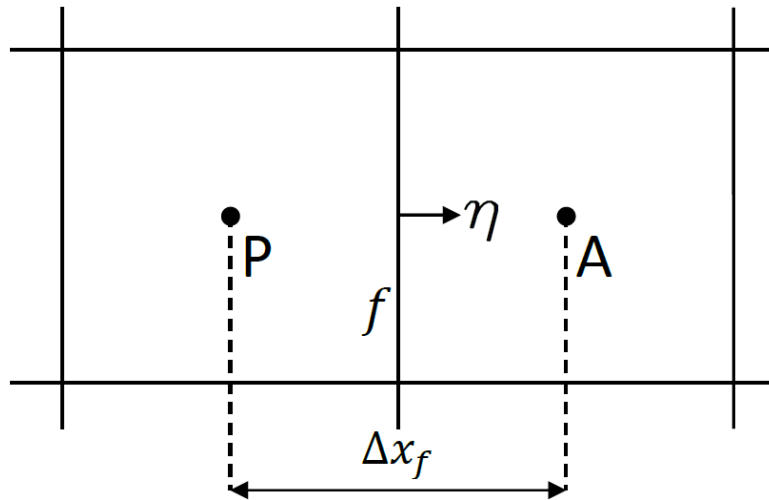


Figure 6.1: Schematic of adjacent cells, cell faces and centroids.

Convective Term

Similarly, the integral form of the convective term is expressed as the sum of face-centre values in the following manner

$$\oint_S n_j (u_j u_i) dS = \sum_f (n_j u_j S_f)_f u_{i_f} \quad (6.12)$$

The unknown u_{i_f} is approximated using linear interpolation, which for uniform grids gives $u_{i_f} = (u_{i_P} + u_{i_A})/2$, where u_{i_P} and u_{i_A} are the values of u_i at node P and its adjacent node A respectively. Therefore one gets

$$\sum_f (n_j u_j S_f)_f u_{i_f} \approx \sum_f \frac{F}{2} (u_{i_P} + u_{i_A})_f \quad (6.13)$$

where $F = (n_j u_j S_f)_f$ represents the mass flux through cell face f . Equation 6.13 represents a second-order accurate approximation of the convective term.

Pressure Term

The pressure term is handled in the same way as the convective term. The surface integral is expressed as the sum of integrals at the cell-faces. Pressures are approximated using linear interpolation, which for uniform grids gives

$$\oint_S n_i p dS = \sum_f (n_i p S_f)_f \approx \sum_f n_i S_f \left(\frac{p_P + p_A}{2} \right)_f \quad (6.14)$$

Discretised Navier-Stokes Equations

Combining all the above discretised terms reveals the discretised, Navier-Stokes Equations

$$\int_V \frac{\partial u_i}{\partial t} dV + \sum_f \left(\frac{F}{2} - D \right)_f (u_{i_P} + u_{i_A})_f = - \sum_f n_i S_f \left(\frac{p_P + p_A}{2} \right)_f \quad (6.15)$$

This can be expressed more concisely using the coefficients a_P and a_A which include both the diffusive and convective flux

$$\int_V \frac{\partial u_i}{\partial t} dV + a_P u_{i_P} + \sum_{all A} (a_A u_{i_A}) = - \sum_f n_i \left(S_f \frac{p_P + p_A}{2} \right)_f \quad (6.16)$$

6.2.2 Temporal Discretisation

The first term in Equation 6.7 represents the integral of the transient term and is non-zero for unsteady problems. This term can be discretised as

$$\int_{\Delta t} \int_V \frac{\partial u_{i_P}}{\partial t} dV dt \approx \int_{\Delta t} \frac{u_{i_P}^{t+\Delta t} - u_{i_P}^t}{\Delta t} V dt = (u_{i_P}^{t+\Delta t} - u_{i_P}^t) V \quad (6.17)$$

where V is the volume of the cell. In Equation 6.17 it was assumed that the integral of the transient term in the whole volume can be approximated by the value at the centroid multiplied by V . This is a second-order accurate approximation.

Integrating Equation 6.16 over the time-step, Δt , and using Equation 6.17 one gets

$$(u_{i_P}^{t+\Delta t} - u_{i_P}^t) V + \int_{\Delta t} \left[a_P u_{i_P} + \sum_f (a_A u_{i_A})_f \right] dt = - \int_{\Delta t} \sum_f n_i \left(S_f \frac{p_P + p_A}{2} \right)_f dt \quad (6.18)$$

For the approximation of the time integral on the right-hand side, the second-order Crank-Nicolson time scheme is used

$$\int_{\Delta t} \phi(t) dt = \frac{1}{2}(\phi^t + \phi^{t+\Delta t})\Delta t \quad (6.19)$$

Applying Equation 6.19 to Equation 6.18 provides a discrete representation of the Navier-Stokes equations in space and time

$$\begin{aligned} \frac{u_{i_P}^{t+\Delta t} - u_{i_P}^t}{\Delta t} V + \frac{1}{2} \left[\left(a_p u_{i_P} + \sum_f (a_A u_{i_A})_f \right)^t + \left(a_p u_{i_P} + \sum_f (a_A u_{i_A})_f \right)^{t+\Delta t} \right] \\ = -\frac{1}{2} \left[\sum_f n_i \left(S_f \frac{p_P + p_A}{2} \right)_f^t + \sum_f n_i \left(S_f \frac{p_P + p_A}{2} \right)_f^{t+\Delta t} \right] \end{aligned} \quad (6.20)$$

This results in a system of equations that must be solved at each time-step. In order to increase the diagonal dominance of the resulting system, the coefficient $V/\Delta t$ of $u_{i_P}^{t+\Delta t}$ is included in the a_p coefficient.

6.2.3 Pressure-Velocity Coupling

The correct velocity field should satisfy the continuity equation and such a velocity field would be produced by the correct pressure field. In the absence of an explicit equation for pressure, the continuity equation is used to extract pressure. An initial pressure field is estimated, which is then corrected by solving a pressure-correction equation derived from the continuity equation. The velocity field is corrected and another pressure-correction is solved again using the corrected velocity. This process is repeated until the resulting velocity fields satisfy the continuity and momentum equations to within a defined tolerance. This is known as the PISO algorithm [104] and usually 3 iterations are needed for convergence. The approach is implemented using `pimpleFoam`, a PISO algorithm (when run as an unsteady simulation; it is `SIMPLE` when run as a steady simulation) in OpenFOAM (version 2.2.2).

6.3 Computational Details

A grid was generated with an in-house MATLAB code using a curvilinear coordinate system (ζ, η) , shown schematically in Figure 6.2. Zhang and Samtaney [73] fixed all boundaries $10c$ away from the aerofoil surface. In the DNS study by Jones *et al.* [47], for flow past an aerofoil at the same $Re_c = 5 \times 10^4$, a boundary located at $5.3c$ was found to be sufficient to capture the outer potential flow. The boundaries in this thesis are set to $10c$ away from the aerofoil.

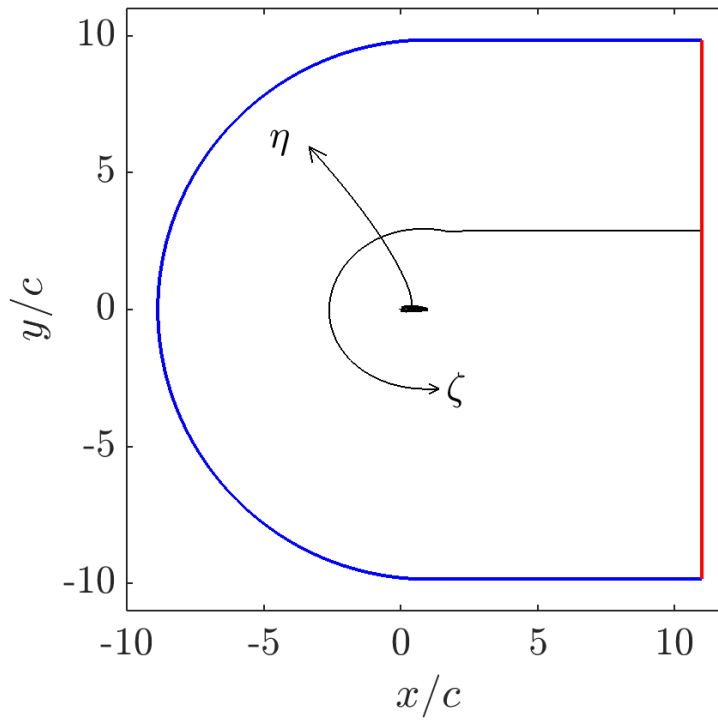


Figure 6.2: C-type computational domain based on curvilinear coordinates η and ζ .

The grid boundary node distribution is specified on the aerofoil surface and the outer boundary. In total 1024 points were specified in the ζ direction around the aerofoil surface and 220 in the η direction. The domain was extruded by $0.2c$ in the z direction to form a 3D grid. This has been found to be sufficient for capturing the 3D flow behaviour in previous studies [47, 10].

Two grids were generated for the computations in this thesis. A grid in which 16 planes were specified in the z direction is used in simulations that will henceforth be referred to as

coarse3D. A finer, fully resolved grid with 64 planes in the z direction will be used for better resolved simulations, referred to as fine3D. The number of cells in the aerofoil and wake regions of the grid are displayed in Table 6.1. It should be noted that there are two wake regions in the domain: the upper and lower portion of the grid.

Grid	Aerofoil			Wake			Total
	ζ	η	z	ζ	η	z	
Coarse3D	1024	220	16	1084	220	16	11,235,840
Fine3D	1024	220	64	1084	220	64	44,943,360

Table 6.1: Number of cells in the coarse and fine grids.

The internal mesh is generated using linear transfinite interpolation between corresponding boundary coordinates on the aerofoil and the outer boundary [106]. Since the flow velocity varies rapidly close to a solid wall, it is essential that this region is resolved well. For improved accuracy in the near-wall region it is also essential that the grid lines are normal to the boundary and therefore an elliptic refinement with a Neumann boundary condition is implemented to improve grid orthogonality at the aerofoil surface. The resulting near-wall grid lines in the vicinity of the leading and trailing edges can be seen in Figure 6.3. Table 6.2 shows the effect of the elliptic grid refinement procedure on grid orthogonality.

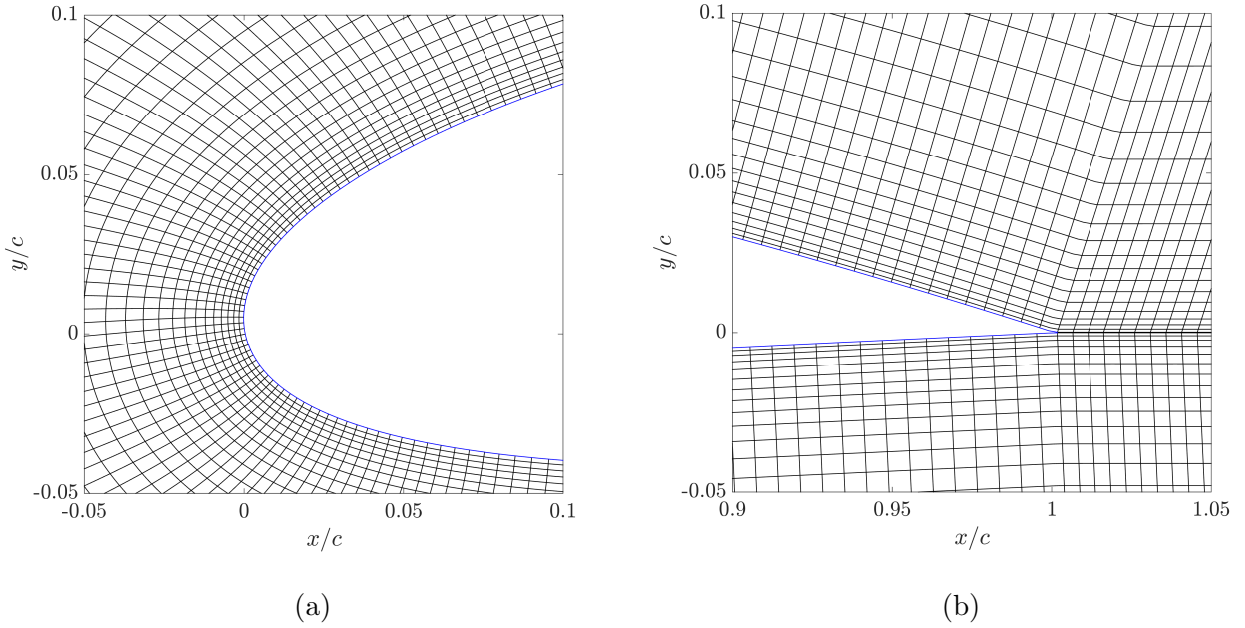


Figure 6.3: Grid lines in the near wall region after orthogonality has been enforced by elliptic refinement and Neumann boundary condition: a) leading edge; b) trailing edge. Every 4th line is plotted for figure clarity.

Table 6.2: Grid Quality

Attribute	Mesh Generation Procedure	
	Linear-Transfinite	Elliptic-Neumann
Max non-orthogonality ($^{\circ}$)	73	26
Mean non-orthogonality ($^{\circ}$)	19	6

The improvement in orthogonality using the elliptic refinement approach is clearly seen.

The simulation was started from uniform initial conditions (freestream velocity). A pressure outlet was used at the outflow exit (shown as a red line in Figure 6.2) and a uniform freestream velocity was used on the rest of the external boundary (shown as a blue line in Figure 6.2). The no-slip condition was enforced on the aerofoil surface. A time-step, Δt , of 1×10^{-4} was used to keep the CFL number below 1.

6.4 Coarse3D Simulation

To enable the flow to develop at a reduced computational cost, the simulation was marched from the aforementioned initial conditions for 200,000 time-steps ($t^* = tU_\infty/c = 20$) on the Coarse3D grid. The instantaneous flow field at this time instant then provided a more appropriate initial condition for the fine3D simulations. The evolution of mean force coefficients, C_L and C_D , for the coarse3D grid can be seen in Figure 6.4.

Although the main purpose of the Coarse3D simulations is to provide the starting field for the Fine3D simulation, it is still possible to make use of the results. By investigating the transient flow behaviour from the starting flow field to the fully developed state, qualitative comparison with characteristic flow modes in the literature can be made.

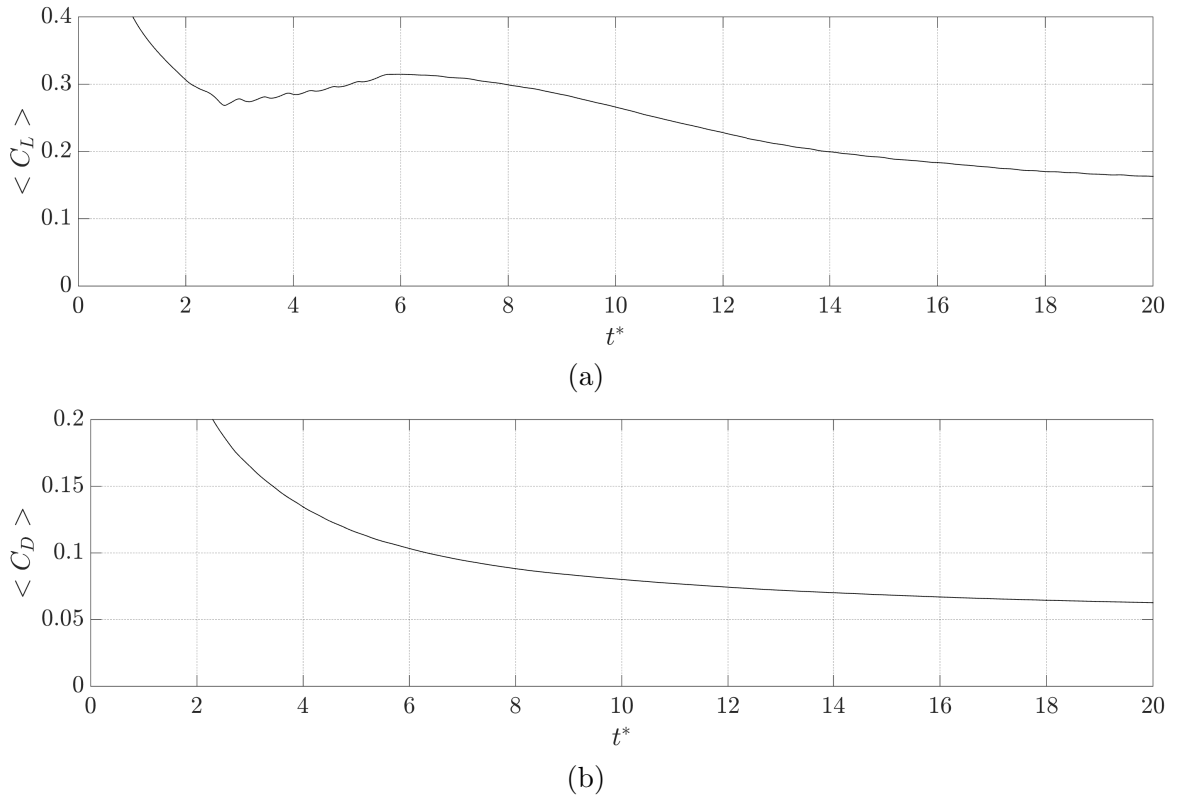


Figure 6.4: Running mean of a) C_L and b) C_D

The initial values of C_L and C_D are initially very large but decay rapidly. At $t^* = 2.7$, C_L begins to increase again before it peaks at $t^* = 5.8$. For $t^* > 6$, C_L drops continuously until roughly $t^* = 18$ where it begins to level off. C_D does not exhibit this rise-and-fall behaviour.

Insight into the reasons for this evolution of C_L is provided by looking at the evolution of skin-friction on the aerofoil's suction surface. This is depicted by the space-time diagram of instantaneous spanwise-averaged skin-friction coefficient, C_f , in Figure 6.5. This time-evolution is also compared with the impulsively started experiments of Huang *et al.* [27] so a qualitative appreciation of the characteristic flow mode can be established by comparison with Section 2.1.2. This can then be compared to that which was found in the experimental discussion in Section 5.1.

When $t^* < 0.19$, C_f is positive across the entire suction surface, meaning the flow is fully attached. This is confirmed by the iso-contours of instantaneous spanwise-averaged vorticity at $t^* = 0.13$ shown in Figure 6.6a; the vorticity sheet is attached everywhere to the aerofoil surface. Negative surface shear first occurs when $t^* = 0.19$, indicating separation, and is located very close to the trailing edge. Between $t^* = 0.19$ and 2.6 the location of the point of separation, x_{sep} , moves upstream with increasing time as can be seen in Figure 6.5. From Figure 6.6b at $t^* = 0.88$, it can be confirmed that the boundary layer does indeed separate close to the trailing edge. The rate of upstream movement of x_{sep} initially increases with time before an inflection point at roughly $t^* = 1.4$. Figure 6.6c, which corresponds to $t^* = 2$, shows that the separated shear layer has now begun to roll up over the aerofoil surface. The upstream movement of x_{sep} continues with time until a drastic change in flow behaviour occurs at $t^* = 2.6$.

At that time the upstream movement of x_{sep} ceases and instead begins to oscillate around $x/c = 0.47$ with a small peak-to-peak amplitudes. Downstream of x_{sep} , strong positive and negative surface shear stresses are present. From Figure 6.6d it can be seen that the negative shear is the result of downstream shedding of large clockwise rotating vortices. The positive shear is located in the region between two adjacent vortices. This behaviour continues until $t^* = 5.6$, at which point the strong surface shear stress disappears. This period of time between $t^* = 2.6$ and 5.6 also corresponds to the region of increasing C_L in Figure 6.4a. This is a result of the low pressure cores in the large vortices having a positive effect on lift. When these large vortices break down ($t^* > 5.6$), C_L decreases again.

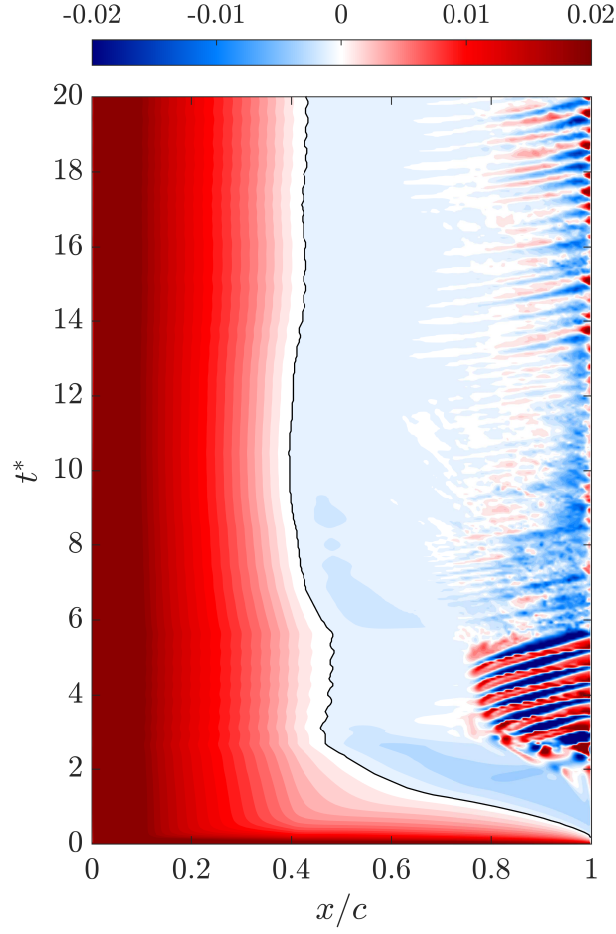


Figure 6.5: Space-time diagram of wall shear stress depicting vortex evolution over the aerofoil's suction surface. The thick black line indicates the location of the separation point ($C_f = 0$).

To summarise the flow development for $0 < t^* < 5.6$, the boundary layer is initially fully attached before separation starts at the trailing edge. The separation point moves upstream with increasing time. Eventually, the separated shear layers roll up over the aerofoil suction surface and large-scale clockwise-rotating vortices are shed downstream. This is very reminiscent of the early flow development described by Huang *et al.* [27] for the separation vortex mode. In the literature review it was concluded that this particular ultra-low Re_c mode was observed in the low Re_c regime by Wang *et al.* [22] and corresponded to their mode (C). Importantly, it was also found that the same flow characteristics were observed by Yarusevych *et al.* [26] for their laminar separated flow without reattachment and this agrees qualitatively with the discussion in Section 5.1 where separation without reattachment was also observed.

For $t^* > 16$, x_{sep} is approximately constant and Figure 6.6e reveals that the large vortices have broken down and a flow field reminiscent of that presented in Chapter 5 can be seen. The negative shear at the trailing edge at these time instances in Figure 6.5 is a result of the reverse flow vortex mentioned in Section 5.1. A comparison with the experimental flow fields in Section 5.1 will be presented in greater detail in the following section.

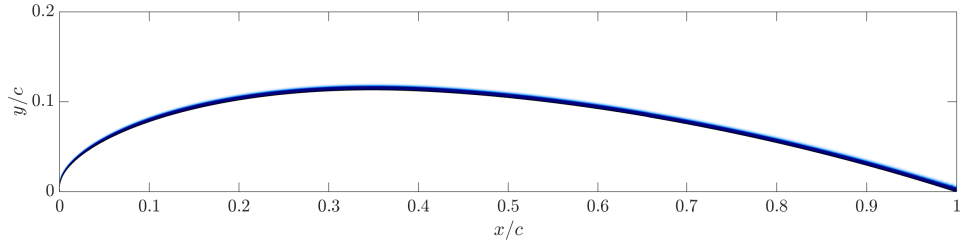
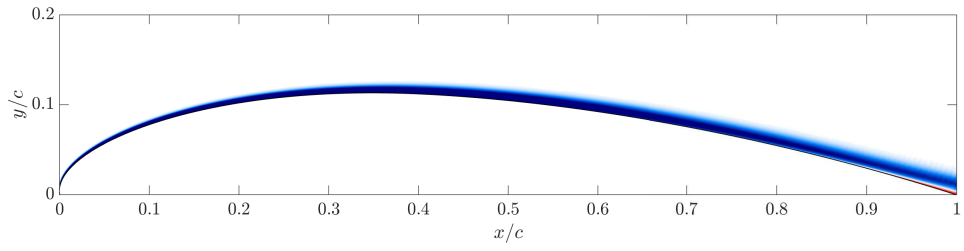
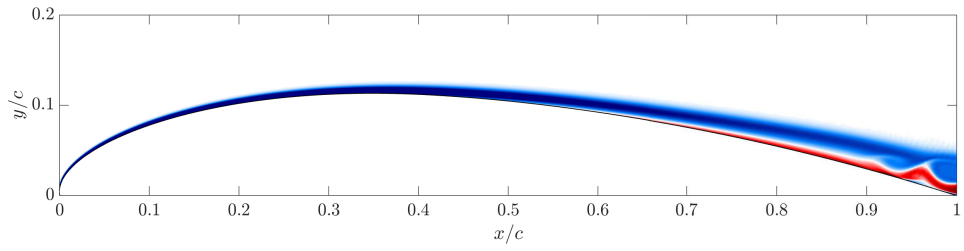
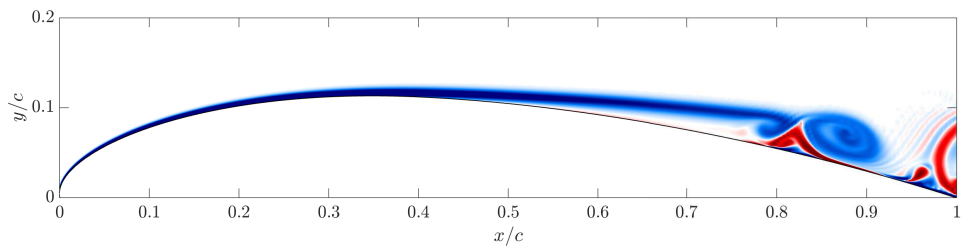
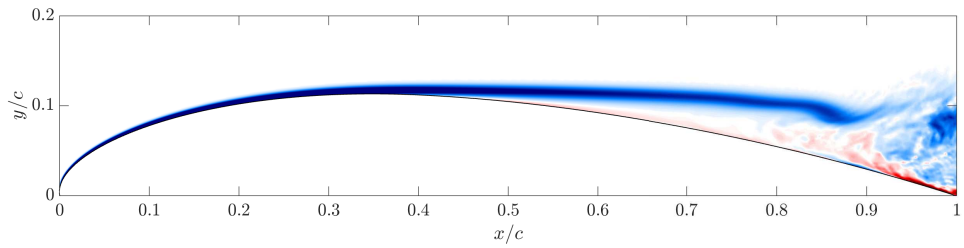
(a) $t^* = 0.13$ (b) $t^* = 0.9$ (c) $t^* = 2$ (d) $t^* = 4$ (e) $t^* = 19$

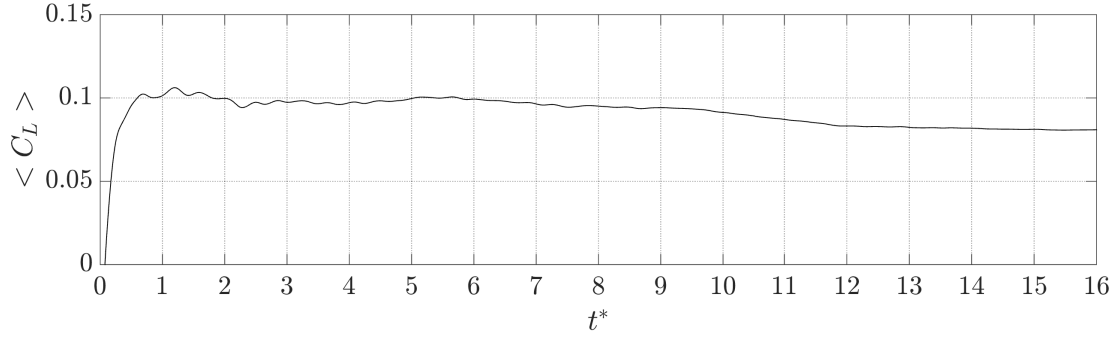
Figure 6.6: Iso-contours of instantaneous, spanwise-averaged Ω_z during the initial flow development. The different instances depict different breakdown regimes: (a) fully attached; (b) trailing-edge separation; (c) shear layer roll-up, (d) 2D shedding; (e) 3D shedding.

6.5 Fine3D Simulation

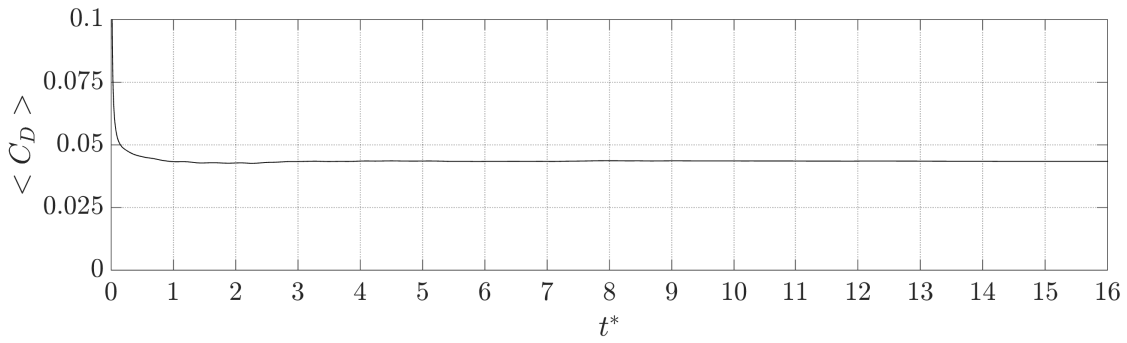
At $t^* = 20$ a 3D flow has fully developed. The instantaneous velocity and pressure fields were then interpolated to the fine grid and used as initial conditions for the fine3D simulation. This simulation is then run until a statistical steady-state is achieved. Once this is reached, the simulation was restarted and statistics were collected. The following section will verify that the statistical steady-state has been reached and that the grid resolution was sufficient for accurate results to be achieved.

6.5.1 Near-wall Grid Resolution and Statistical Convergence

Figure 6.7 shows the evolution of C_L and C_D with time. Statistical steady-state for C_D is reached when $t^* > 8$ but it takes C_L until $t^* = 13$ to statistically converge. With this in mind, the solution at $t^* = 16$ is considered to be fully developed. The simulation was restarted from $t^* = 16$ and run until $t^* = 20$. Snapshots of the instantaneous flow fields are captured every 125 time-steps, meaning 321 snapshots were captured over the course of the simulation.



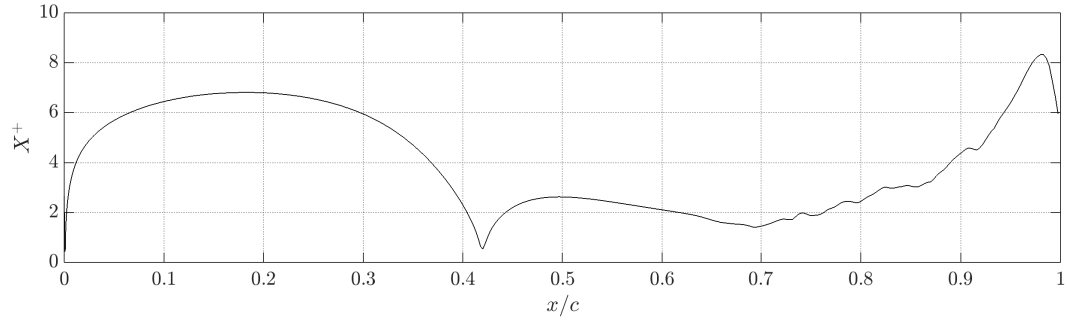
(a)



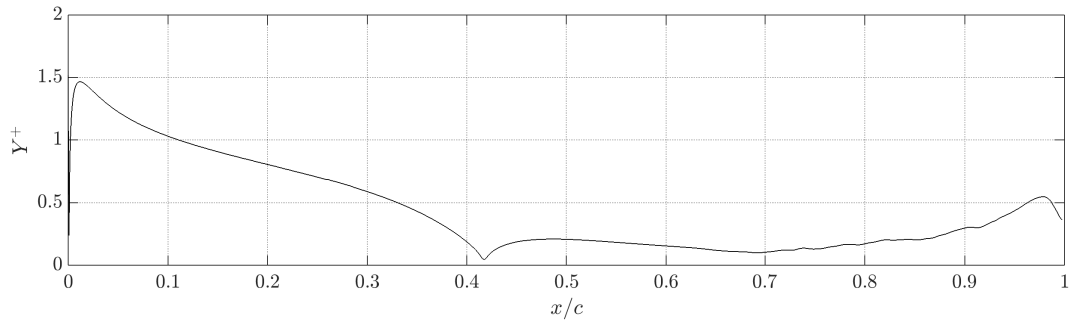
(b)

Figure 6.7: Running mean of force coefficients during the fine3D simulation: a) C_L and b) C_D .

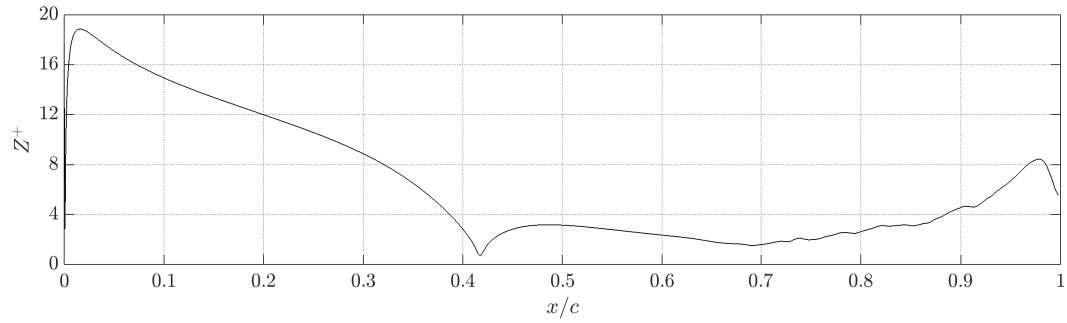
The time-averaged friction velocity, u^* , was calculated from 321 snapshots of the wall shear stress, τ_w , and used to obtain the x^+ , y^+ and z^+ values at the centroid of the cells closest to the wall. Their surface distribution is provided in Figure 6.8. The maximum y^+ and z^+ values are 1.5 and 18.9 respectively and occur close to the leading edge. The flow at this location is still laminar and the friction velocity, u^* , is maximum owing to the large acceleration of the flow around the nose of the aerofoil [73]. Further downstream, the maximum y^+ drops to 0.55 and maximum z^+ is 8.5. The maximum x^+ value is 8.3. These values are compared with other recent DNS studies involving laminar separation on aerofoils in Table 6.3. The values are comparable with those in literature and indicate that the near-wall resolution in the region of turbulent flow is adequate to resolve the turbulent length scales.



(a)



(b)



(c)

Figure 6.8: Grid resolution in wall units: a) X^+ ; b) Y^+ ; c) Z^+ .

	x^+	y^+	z^+	$N_y : y^+ < 10$
fine3D	8.3	0.55	8.5	10
Zhang and Samtaney [73]	5	0.6	5	-
Jones <i>et al.</i> [47]	3.4	1	6.5	9

Table 6.3: Near-wall grid resolution in wall units compared with recent other studies in the literature.

In addition to the comparison with the literature given above, the boundary layer profiles were checked to ensure the near-wall flow was being correctly captured. Time-averaged wall-normal profiles of $\bar{U}$ and $\bar{\Omega}_z$ in the vicinity of separation ($x/c = 0.4$) are presented in Figure 6.9.

Locating the edge of the boundary layer is difficult owing to the acceleration of the potential flow by the aerofoil's suction surface (the velocity profile will not return to U_∞ until a long way from the wall). However, boundary layers are associated with large wall-normal velocity gradients and high vorticity. The region of flow with a more-or-less constant velocity profile and $\Omega_z = 0$ can therefore be considered as outside the boundary layer. From Figure 6.9 it can be seen that this occurs at $\eta/c > 0.02$. From the number of cell-centres below $\eta/c = 0.02$ it can be determined that 33 cells lie within the region of high velocity gradients. This is more than enough to correctly capture the boundary layer profile – which appears very smooth. This gives further confidence that the computational grid is appropriate for the simulations.

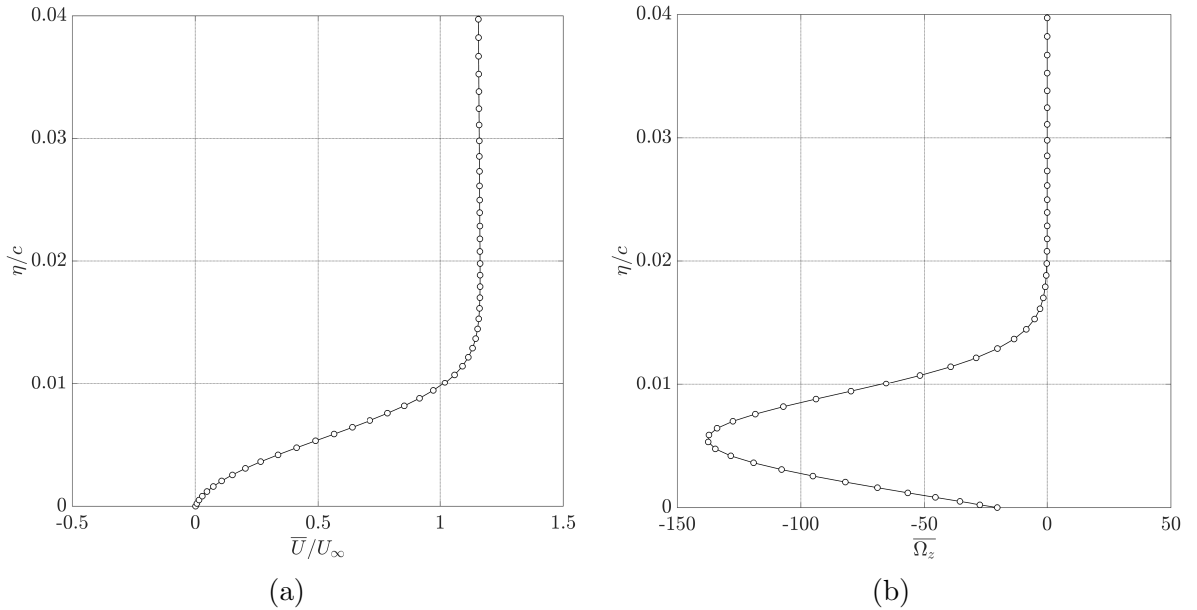


Figure 6.9: Boundary layer profiles at $x/c = 0.4$ (near the separation point): a) profile of $\bar{U}/U_\infty$; b) profile of $\bar{\Omega}_z$. The circles represent cell centres.

6.5.2 Comparison with XFOIL

Further verification was sought by means of XFOIL simulations. XFOIL uses a e^n transition model, which is critical in this case as the location of transition is of paramount importance

to the development of the flow over the aerofoil. By testing a few values of n , it was found that $n = 10$ agreed with the location of transition in the *fine3D* simulations fairly well. This is shown in Figure 6.10, where the termination of the pressure plateau and onset of pressure recovery for $n = 10$ and *fine3D* are very similar. The default value of n is 9 and, according to the software, $n = 10$ corresponds to a very low turbulence intensity level of 0.05%. This agrees with the fact that no background disturbances were input in the current computations.

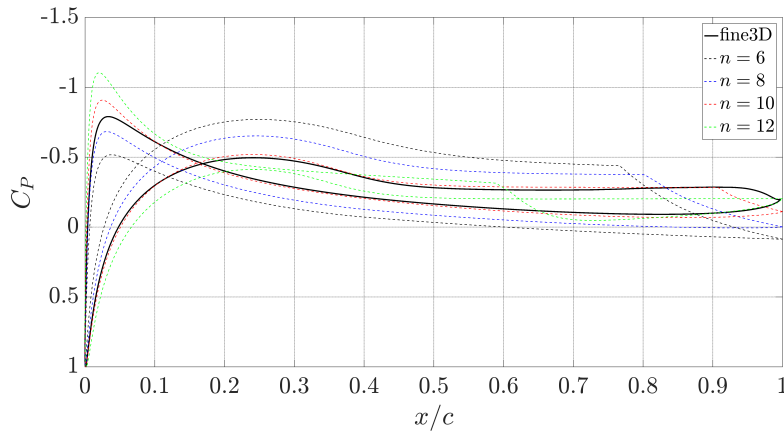


Figure 6.10: XFOIL C_P distributions with different values of exponent n in the e^n transition model, compared with *fine3D*

The C_P and C_f distributions from the *coarse3D*, *fine3D* and XFOIL simulations are shown in Figure 6.11. It can be seen that, with the transition location roughly matching that of the *fine3D* simulation, XFOIL gives a very similar pressure distribution over the rest of the aerofoil – particularly the suction peak and the location of the pressure plateau. In addition, the C_f distributions reveals that XFOIL gives a time-averaged separation location that is slightly further upstream than that in the *fine3D* simulation, but on the whole they are both in very good agreement. This gives further confidence that the *fine3D* simulations are correctly capturing the flow fields.

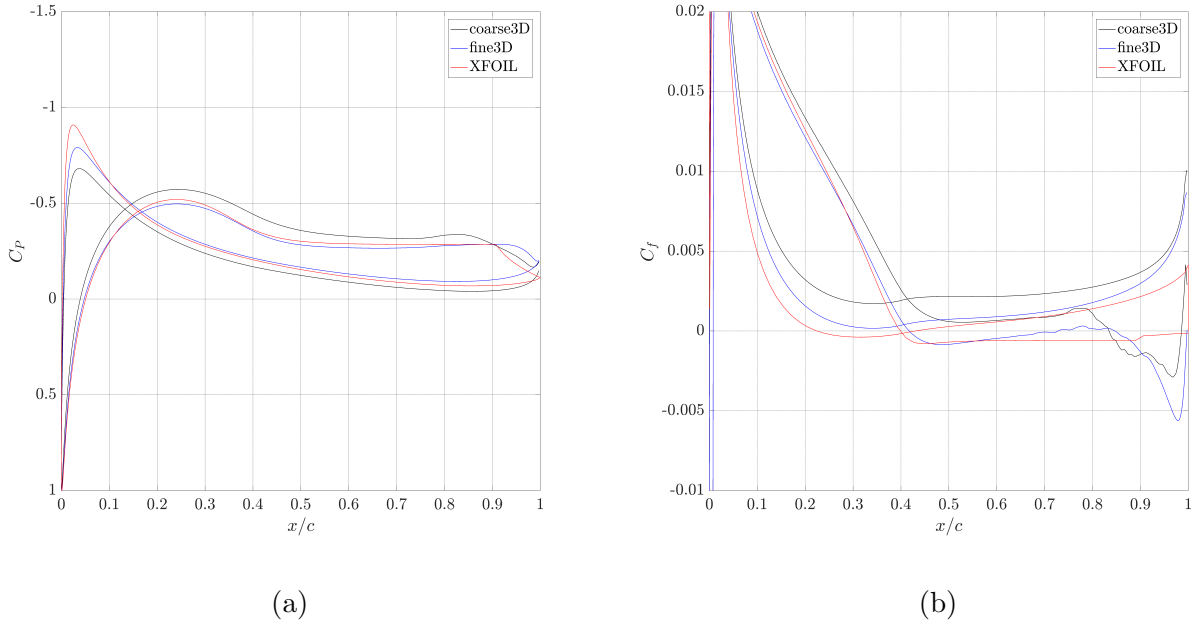


Figure 6.11: Distributions of a) C_P and b) C_F plotted from coarse3D, fine3D and XFOIL data.

6.5.3 Comparison with Experiments

In this section, time-averaged computational data will be compared with the corresponding experimental data. Figure 6.12 shows the time-averaged velocity magnitude obtained by (a) CFD and (b) PIV. The comparison is restricted to qualitative because of a couple of factors. The spatial resolution of the PIV is much lower than that of the CFD which affects the accuracy with which the curved surface is captured. Furthermore, reflections from the surface cause errors in the near-wall flow field. Nevertheless, the location and magnitude of the maximum accelerated flow and the location and extent of the separated region and wake are qualitatively very similar.

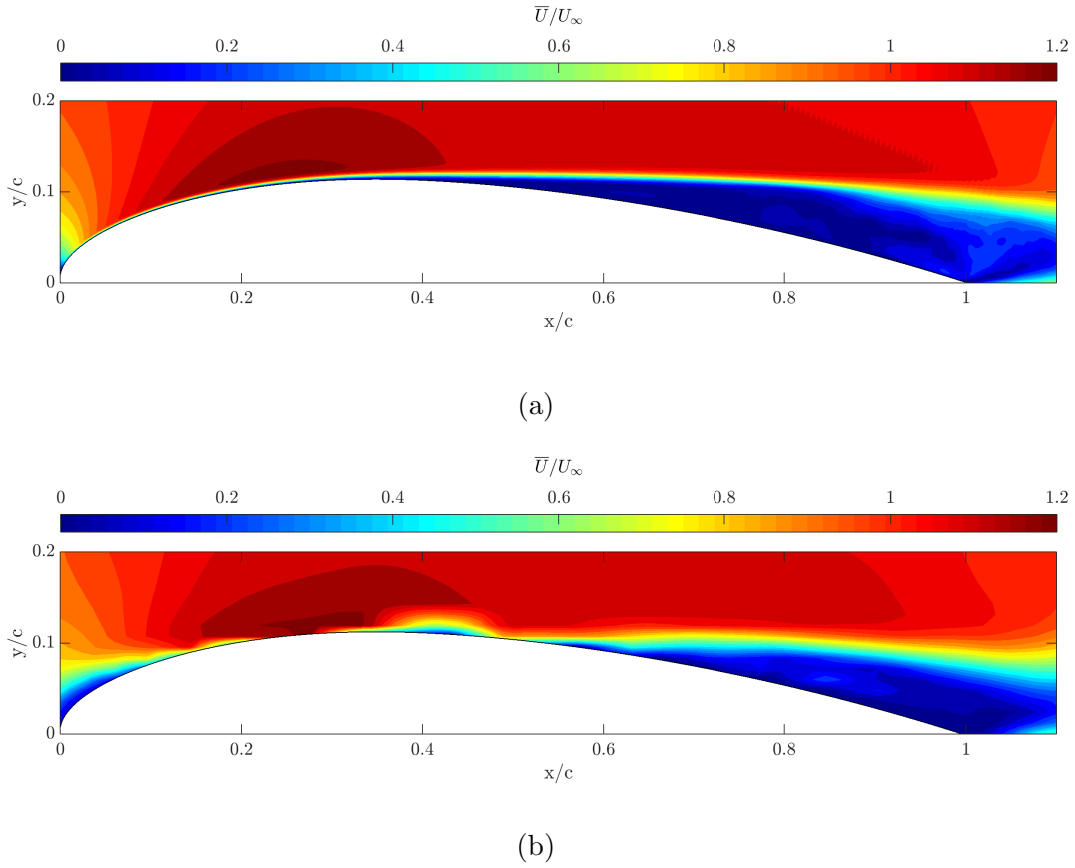


Figure 6.12: Comparison of mean velocity field: a) CFD; b) PIV

Velocity profiles at three locations in the aerofoil wake, obtained from both PIV and hotwire data, are shown in Figure 6.13. The results show that the velocity fields also compare well quantitatively. The presence of the splitter plate made it impossible to obtain velocity profiles upstream of $x/c = 1.25$. The data presented in Section 5.1.3, which are at locations upstream of $x/c = 1.25$, were obtained using a smaller probe mounting which was unsuitable for taking velocity profiles owing to some ambiguity in measuring the probes' cross-stream location. However, it was possible to obtain a profile in the near-wake (at $x/c = 1.10$ in Figure 6.13a) using the PIV data. The difference in the velocity profiles at $y/c = 0.024$ for $x/c = 1.1$ appears to be the result of an outlier in the PIV curve since this point does not follow the trend of the rest of the profile. Other than this, both profiles are in reasonably good agreement. At $x/c = 1.25$, there is a 8% difference between the maximum velocity deficit.

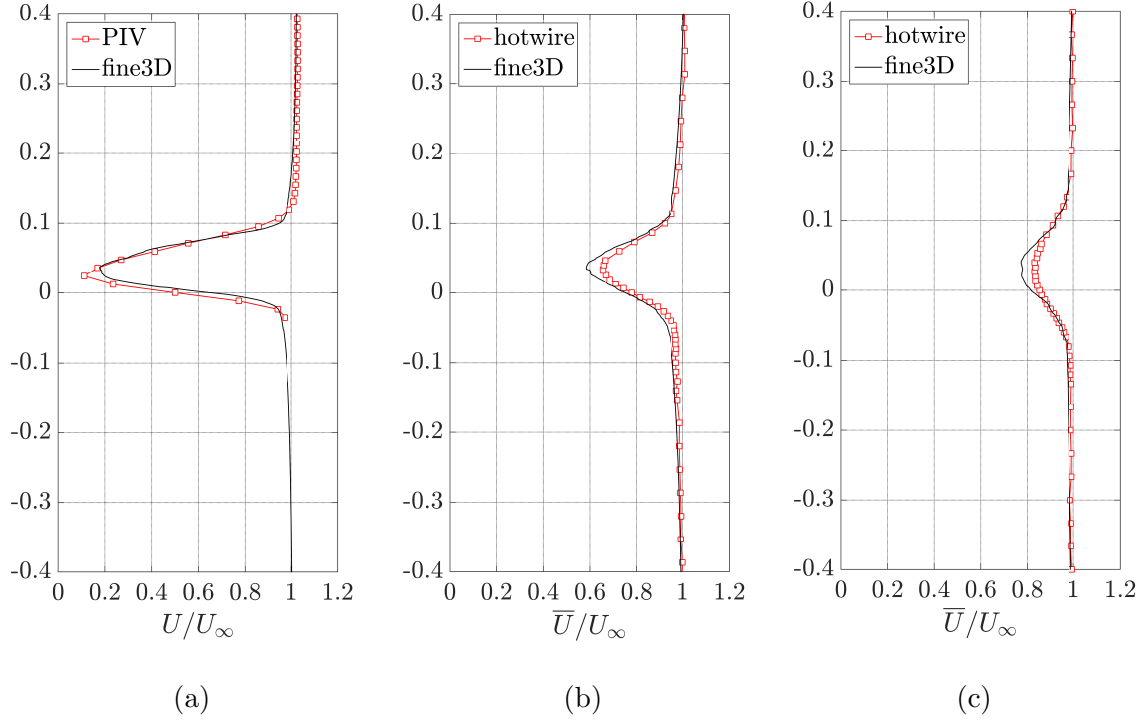


Figure 6.13: Comparison of wake profiles at: a) $x/c = 1.10$; b) $x/c = 1.25$; c) $x/c = 1.50$.

Differences can be attributed to the slightly different aerofoil geometry and uncertainties in freestream velocity and turbulence intensity. In addition, the boundary conditions at the walls in the tunnel are different from those in the simulations. The tunnel had fixed walls at $\pm 2c$ in the cross-stream direction (U_∞ at $10c$ in the computations) and at the ends of the aerofoil in the spanwise direction (modelled as periodic in the computations). Nevertheless, the good quantitative and qualitative agreement means that the computations are capable of providing additional insight into the flow physics that is not possible from the experimental data alone.

Time-Averaged Results

Section 5.1 of the experimental chapter concluded that the unactuated aerofoil at $\alpha = 0^\circ$ was experiencing laminar separation without reattachment and this was the cause of the serious performance losses observed relative to the same aerofoil geometry at high Re_c . It was also hypothesised that a transition of the laminar separated shear layer takes place. The computations can be used in conjunction with the experimental results to provide more

insight into the flow physics.

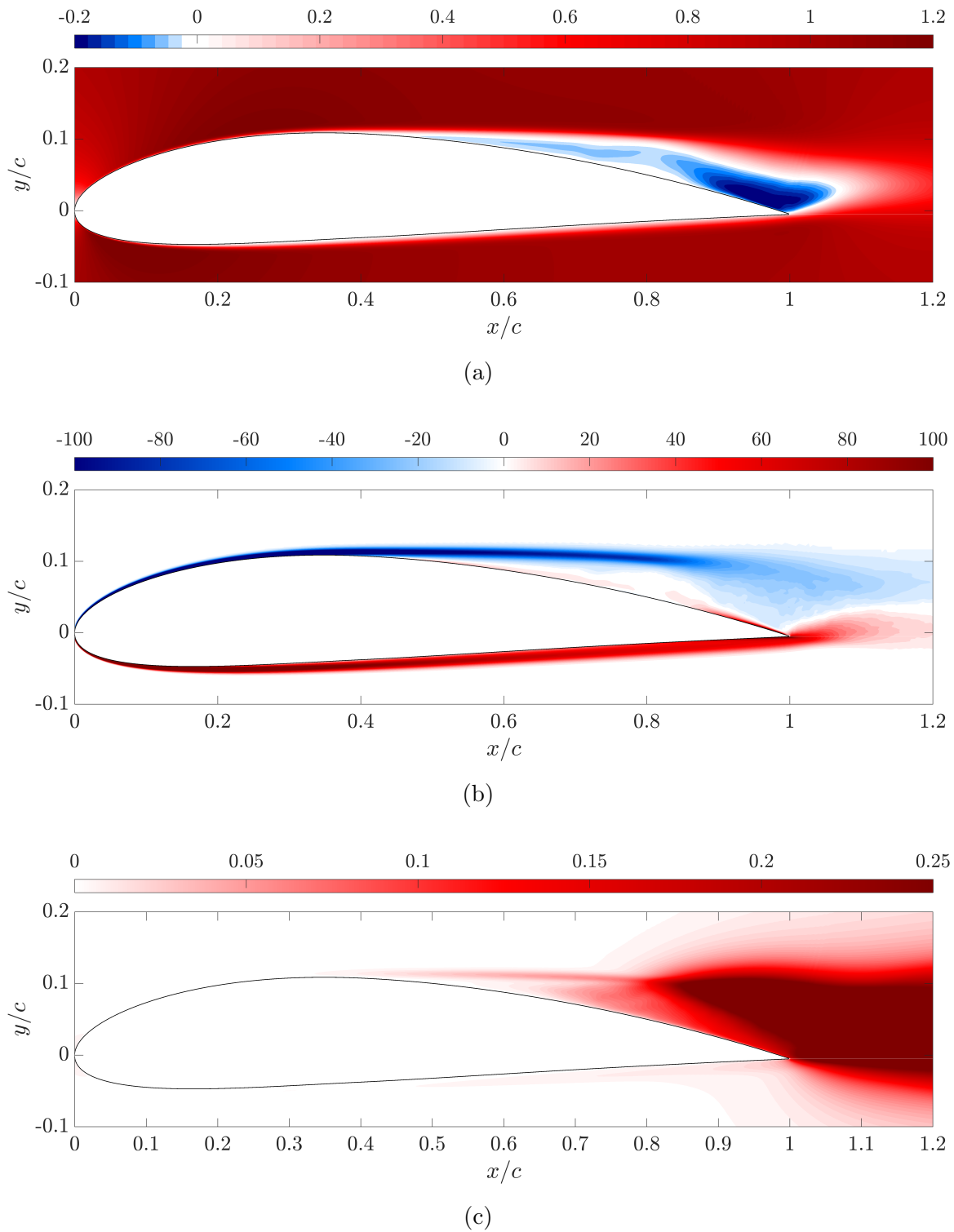


Figure 6.14: Iso-contours of spanwise and time-averaged flow fields obtained from 321 snapshots acquired for $16 \leq t^* \leq 20$. a) $\bar{u}$; b) $\bar{\Omega}_z$; c) $1/2\sqrt{u'^2 + v'^2 + w'^2}$.

Figure 6.14b shows the time- and spanwise-averaged spanwise vorticity. This can be compared with Figure 5.5 in Section 5.1.2. It is seen that the attached sheet of strong vorticity, $\overline{\Omega_z}$, separates from the aerofoil surface at $x/c \approx 0.4$ and remains concentrated exclusively in a thin separated layer until $x/c \approx 0.85$. This was also observed in Figure 5.5 and, since it occurred in a time-averaged sense, it was interpreted as the steady, laminar portion of the shear layer. It was stated that a quiescent flow should develop beneath the laminar portion of the shear layer [31], resulting in a flattened region in the surface pressure distribution, known as a pressure plateau [26]. From Figure 6.14a it can be seen that the time-averaged streamwise velocity component in the region located underneath the strong concentrated shear layer is very close to zero. Furthermore, in Figure 6.15 a pressure plateau is indeed found to extend from a location shortly downstream of separation to close to the trailing edge. As can be seen in Figure 6.14c, the region between the separation point and $x/c = 0.8$ not only exhibits a pressure plateau but also very low values of turbulent kinetic energy. This confirms the presence of the laminar shear layer hypothesised in Section 5.1.2.

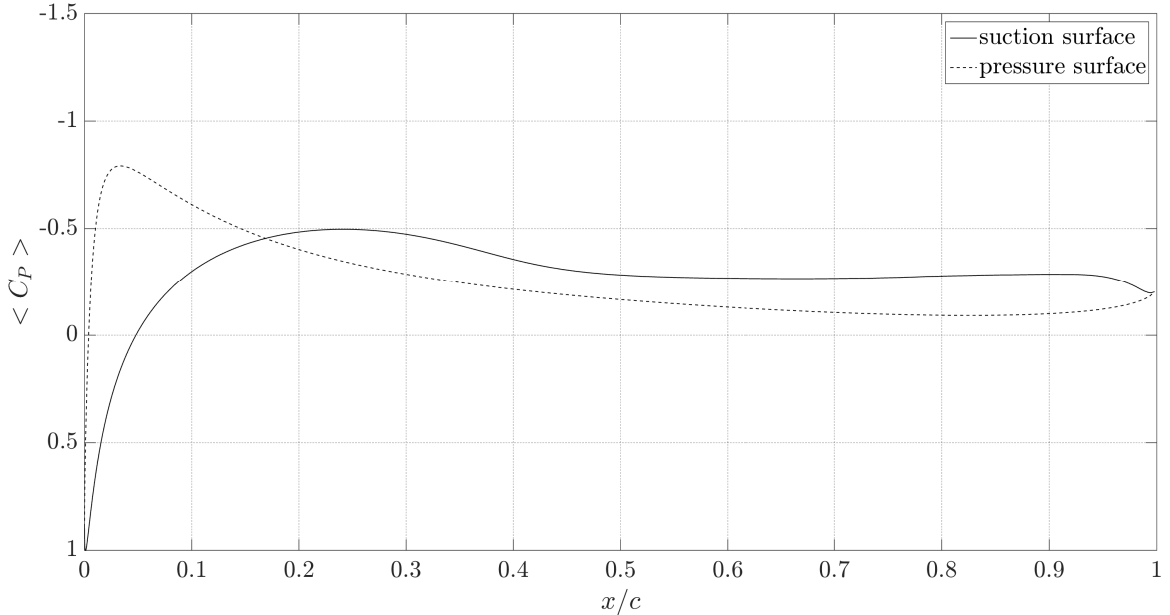


Figure 6.15: Time-averaged distribution of C_P .

Further downstream (i.e. $x/c \geq 0.85$), the magnitude of $\overline{\Omega_z}$ decreases and the region of high vorticity becomes more dispersed. This is exactly what was observed in Figure 5.5 and

interpreted as the onset of unsteadiness and breakdown of the laminar separated shear layer. It was suggested that the dispersion of high vorticity in the shear layer is the result of turbulent transition. This can be confirmed from inspection of Figure 6.14c in which the turbulent kinetic energy, (TKE), upstream of $x/c = 0.8$ is very small. However, downstream of $x/c = 0.8$, TKE increases rapidly indicating turbulent transition. Horton [3] found that a reverse-flow vortex and pressure recovery should accompany the turbulent portion of an LSB. Figure 6.16 shows that a reverse flow vortex, also observed in Figure 5.5, is observed and confined to a region of high TKE. In addition, Figure 6.15 reveals a pressure recovery does indeed occur, however since transition to turbulence does not happen until close to the trailing edge, the pressure cannot recover sufficiently for reattachment to occur before the trailing edge is reached. Hence, the characteristic flow development can be summarised as a laminar separation, with transition but without reattachment. The fact that both the experiments and computations agree with these characteristics gives further confidence that the computations are a suitable supplement to the experiments.

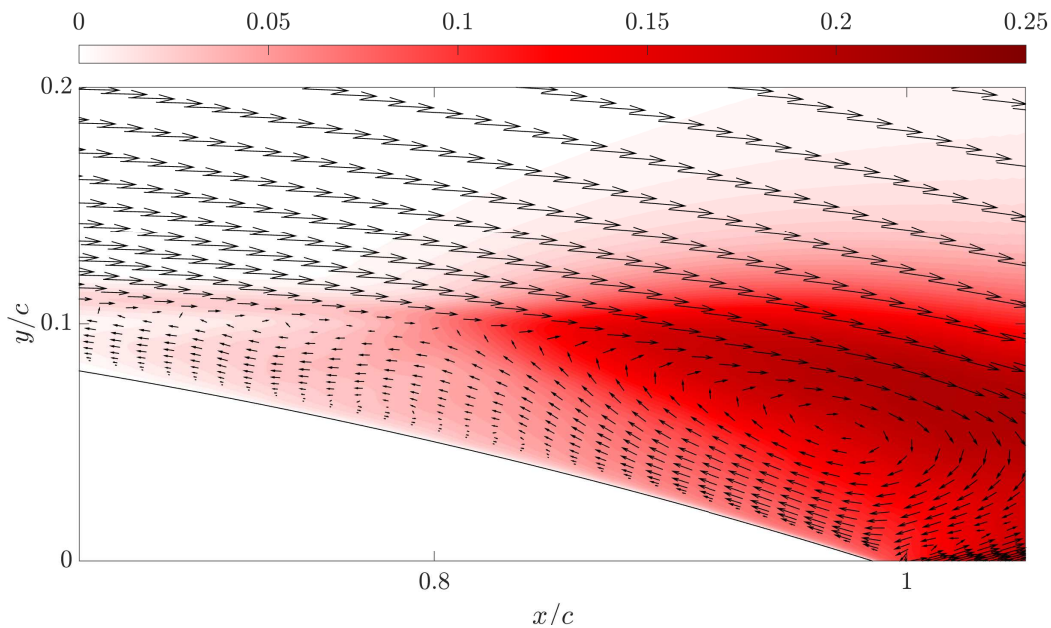


Figure 6.16: Local velocity vectors showing reverse flow vortex superimposed on the TKE contours.

6.6 Dynamic Mode Decomposition of the Flow Field

It is widely accepted that large vortical, spanwise structures are the essential ‘building blocks’ of the mixing layer and are responsible for virtually all of the momentum transfer across it [67]. Separation control schemes that harness these large vortical structures and their ability to entrain momentum from the outer flow towards the aerofoil surface are based on the assumption that the dynamics of a separated flow over aerofoil is dominated by a characteristic frequency of the separated region [71].

However, as seen in Section 5.2.1, there is more than one naturally occurring frequency playing a role in the dynamics of the flow around this aerofoil. Given the different flow features present in the flow around the aerofoil, including laminar shear layer, transition to turbulence, vortex shedding etc., it is perhaps not surprising that it is not all governed by the same characteristic frequency. Kotapati *et al.* [71] summarised that the shear layer frequency was one of at least 3 different characteristic frequencies that can occur in a separated flow over an aerofoil, along with the shedding frequency of the wake and the characteristic frequency of the separated region. In general, when two or more basic frequencies are present (i.e. frequencies that are not harmonically related) sum-difference modes due to non-linear interactions occur, and result in a broad spectrum [71]. A lack of appreciation for the various frequencies and their resonant interactions could explain the wide spread of definitions of optimal frequencies in the literature [71].

To understand the nature of the structures that govern the dynamics of the uncontrolled flow field, and to ascertain those that are most important to its development, the reduced-order modelling technique Dynamic Mode Decomposition (DMD) [107, 108], is employed. From this, the most dynamically important spatial structures, and their corresponding temporal behaviour can be analysed. This information can then be used to help explain the control effectiveness at different frequencies found in Section 5.2.3. A brief description of the method is given below and a more detailed overview of the algorithm is provided in Appendix A.4. For a thorough description of the method the reader can refer to Schmid [107], Jovanovic and Schmid [108] and Tu *et al.* [109].

6.6.1 DMD Background

Snapshots of the cross-stream velocity component, v , were arranged into an $M \times N$ snapshot matrix, Ψ , as in Equation 6.21 where M is the number of grid points and N is the number of snapshots. The superscript denotes the grid point number and the subscript denotes the snapshot number (i.e. the time instance).

$$\Psi = \begin{Bmatrix} \psi_1^1 & \psi_2^1 & \cdots & \psi_N^1 \\ \psi_1^2 & \psi_2^2 & \cdots & \psi_N^2 \\ \vdots & \vdots & \ddots & \vdots \\ \psi_1^M & \psi_2^M & \cdots & \psi_N^M \end{Bmatrix} = \{\psi_1, \psi_2, \dots, \psi_N\} \quad (6.21)$$

The v -component was used because it is most sensitive to frequency-centred activity [26].

It is assumed that a dynamic linear propagator, $\mathbf{A}$, maps one realization of the flow, $N - 1$, to the subsequent realization, N . This is represented by

$$\psi_N = \mathbf{A}\psi_{N-1} \quad (6.22)$$

Since the governing equations of the flow are nonlinear, this assumption amounts to a linear tangent approximation [107]. The dynamics of the fluid process described by sequence Ψ are therefore given by the eigen-decomposition of $\mathbf{A}$. However, since $\mathbf{A}$ is not known an approximation, $\mathbf{S}$, is sought whose dynamics are an approximation of the dynamics of $\mathbf{A}$

$$\mathbf{A}\Psi_0 = \mathbf{S}\Psi_0 + \mathbf{r} \quad (6.23)$$

in which $\mathbf{r}$ is the residual error in the approximation and Ψ_0 is equal to Ψ but from time instances $0 \mapsto N-1$ and defined in Equation A.12.

The objective of the DMD is the formation of $\mathbf{S}$ such that $\mathbf{r}$ is minimised. The eigenvalues, μ_i , of $\mathbf{S}$ provide the temporal information and the corresponding eigenvectors provide the spatial structures (the DMD modes). A linear combination of all DMD modes, $\hat{\phi}_i$, scaled by their

amplitude coefficients, a_i , and advanced in time according to their temporal information, μ_i , approximates the time evolution of the entire data sequence. The amplitude coefficients of every mode are computed using the optimisation algorithm of Jovanovic *et al.* [108].

6.6.2 DMD Results

A portion of the flow encompassing the trailing edge and near wake was extracted from the computational flow field to restrict the analysis to the most relevant region for the dynamics of the flow. This region spanned from $0.6 \leq x/c \leq 1.3$ in the streamwise direction and $-0.1 \leq y/c \leq 0.2$ in the cross-stream direction. The full 3D data were included in the DMD algorithm, however the modes were spanwise-averaged for presentation. The spectral results of the DMD are shown in Figure 6.17.

The frequency spectra are plotted by scaling each mode's norm, $\|\phi_i\|$, by μ_i^{N-1} (the final column in the Vandermonde matrix in Equation A.16) which represents the temporal information about the spatial structure at the final time-instance. In this way, modes with very large initial values (and therefore large norms) but quickly decaying eigenvalues (and therefore little contribution to the long-term evolution of the system) are discounted. The presence of such modes can result in misleading spectral peaks if the mode norms are not scaled [109]. Note that ϕ , without a hat, signifies that these norms are calculated from the scaled DMD modes (i.e. $\hat{\phi}_i \cdot a_i$).

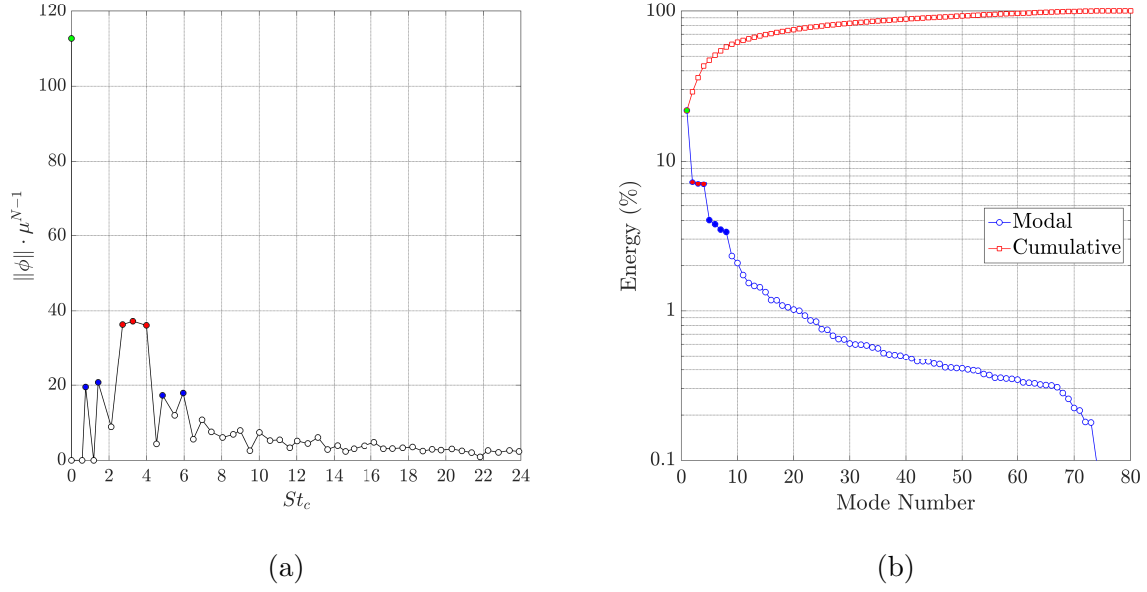


Figure 6.17: Spectral Analysis of the trailing edge and near-wake using DMD: a) Frequency Spectrum; b) DMD modal percentage and cumulative energies.

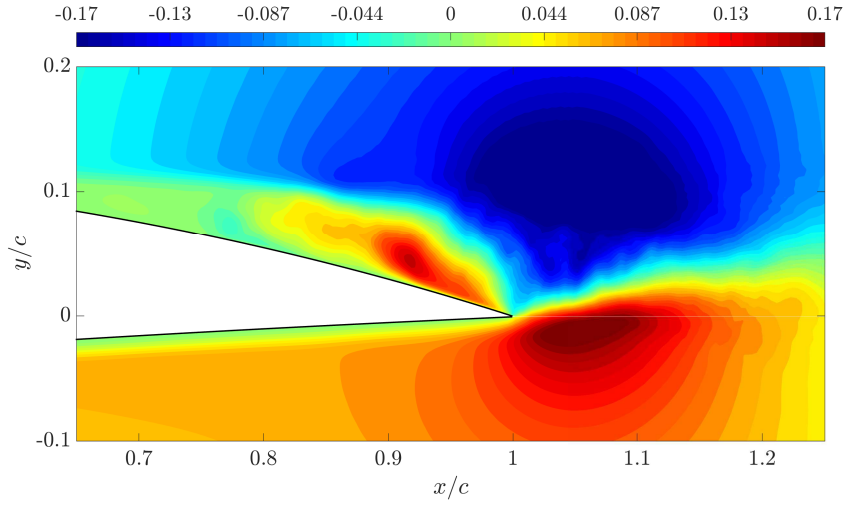


Figure 6.18: The spatial structure of DMD mode 1, equivalent to the time-averaged v velocity component.

As seen in Figure 6.17b, the time-averaged mode, mode 1 (shown as a green circle) has the single highest contribution to the total energy content. This mode contains 22% of the total energy and its spatial shape can be seen in Figure 6.18.

There is then a cluster of frequencies, shown by red dots, that each contribute roughly 7% of

the energy. From the frequency spectrum in Figure 6.17a, this cluster of frequencies occurs between $St_c = 2.7$ and 4. From their spatial structures in Figure 6.19, it can be concluded that these modes are related to the vortex shedding in the wake. The structure corresponding to $St_c = 2.7$ is the only one of the three that is concentrated entirely in the wake and is therefore considered to be the pure wake shedding frequency. This corresponds well with the experimentally found value of $St_c = 2.4$, or 85 Hz in Figure 5.8a in Section 5.1.3.

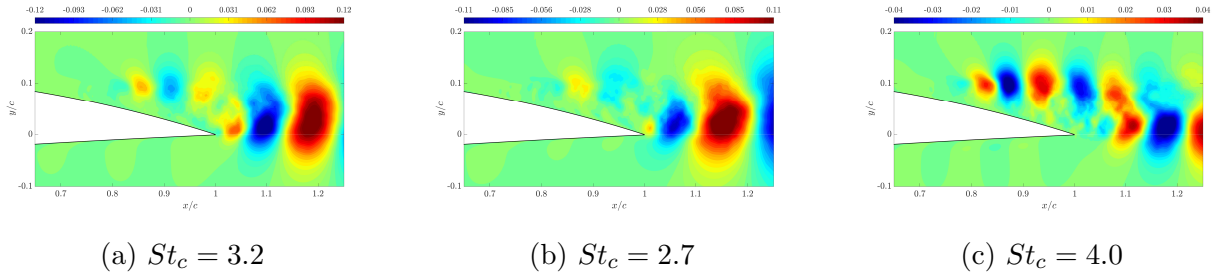


Figure 6.19: The spatial structures of the 3 most dominant DMD modes.

The other two structures also have some presence upstream of the trailing edge and may be a result of sum/difference modes from the interaction of different frequencies in the trailing edge and near wake region. The presence of additional modes in the region very close to the trailing edge is confirmed by the cluster of four modes shown in blue in Figures 6.17a-b. The mode at $St_c = 5.9$, shown in Figure 6.20c, appears to be most pronounced in the wake, although it has some presence in the shear layer. This is likely to be a result of harmonic interaction of the modes at $St_c = 2.7$ and 3.2 (Figures 6.19a and 6.19b) which also occur in the same region.

The spatial structure of the mode corresponding to $St_c = 4.9$ in Figure 6.20d clearly suggests it is related to the shear layer frequency and agrees well with the experimental value of $St_c = 4.8$, or 170 Hz (Figure 5.8b in Section 5.1.3). This mode does not appear as one of the most prominent when looking at the whole flow field in this region since, as mentioned by Kotapati *et al.* [71], the presence of many different frequencies relating the different flow features clouds the analysis.

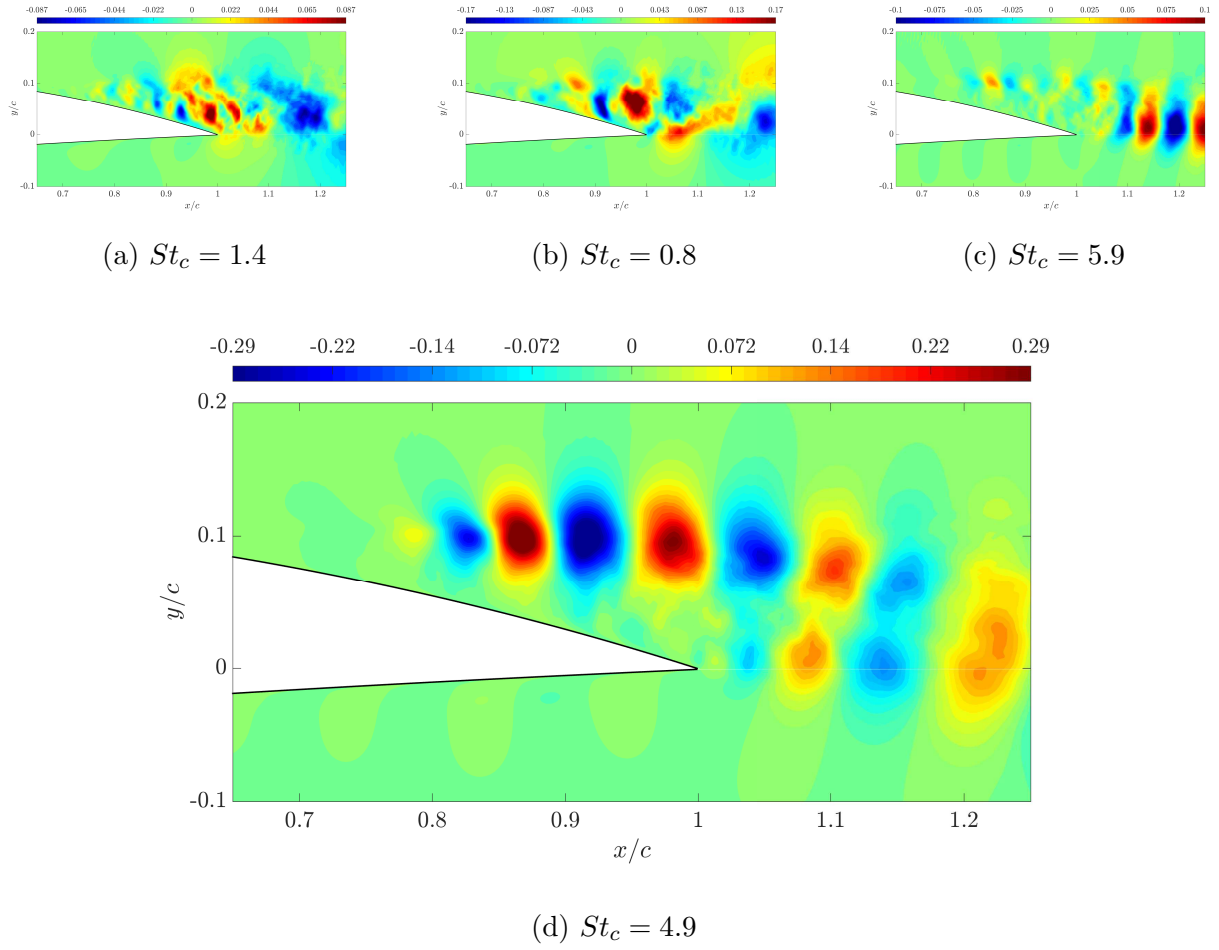


Figure 6.20: The spatial structures of the next 4 most energetic DMD modes.

In an effort to remove some of the confusing sum/difference modes, the DMD algorithm was re-run on a spatial section of the flow field containing only to the reverse flow region and shear layer, thus removing and contributions of the wake. The results of this are shown below.

Shear Layer

The spectrum in Figure 6.21a now exhibits a clear peak at $St_c = 4.9$. Furthermore, Figure 6.22b shows that this has the same spatial structure as that found in Figure 6.20d and confirms that it is the shear layer frequency. The energy content in this region of the flow field in Figure 6.21b reveals that, again, the time-average makes the single largest contribution. In contrast to the discussion above, the shear layer frequency is now very prominent, rather than being within a cluster of frequencies, and contributes 7% of the total energy.

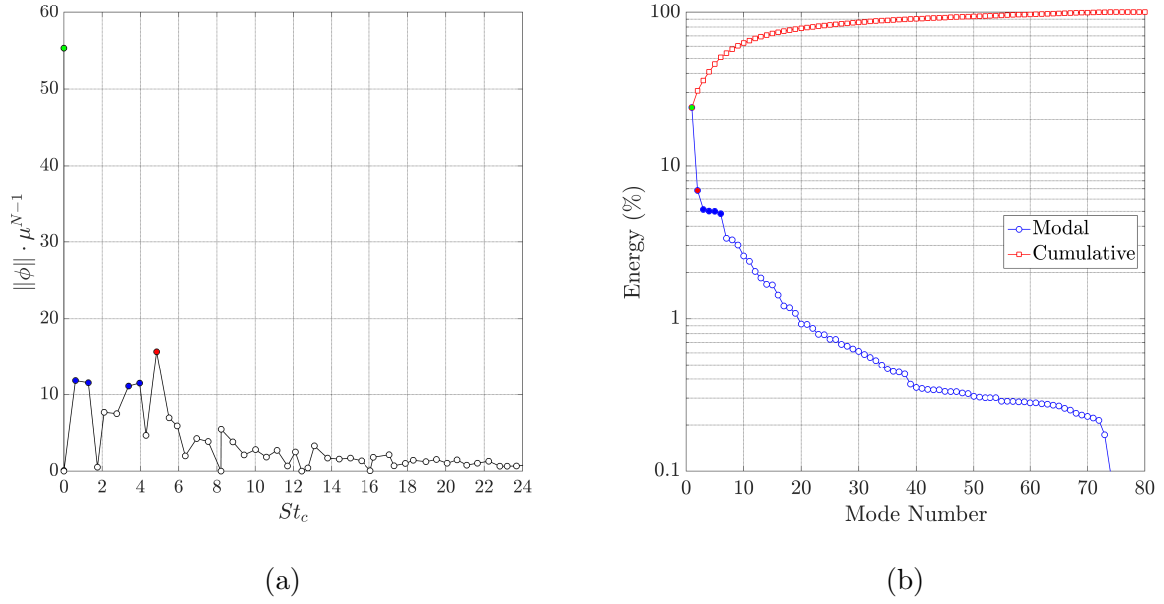


Figure 6.21: Spectral Analysis of the trailing edge using DMD: a) Frequency Spectrum; b) DMD modal percentage and cumulative energies.

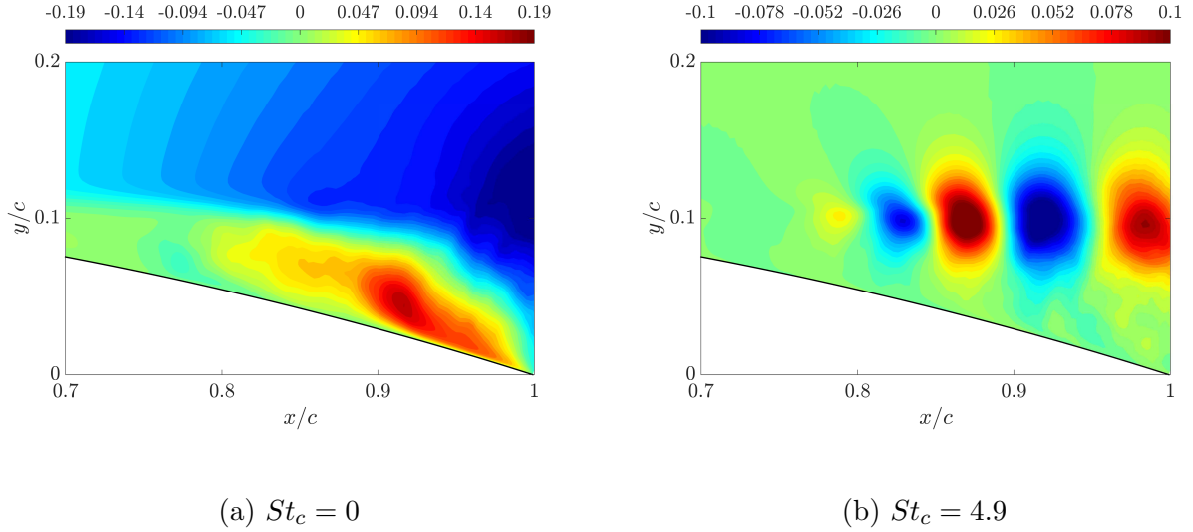


Figure 6.22: The two most energetic DMD modes in the region around the aerofoil trailing edge: a) the time-average; b) the shear layer frequency.

6.7 Summary

In this chapter it has been shown that the Direct Numerical Simulations of the flow around a NACA 4415 at $\alpha = 0^\circ$ agree reasonably well with the experiments presented in Section 5.1. The time-averaged flow fields reveal that a laminar separated shear layer exists until $x/c \approx 0.85$. Downstream of that location a rapid increase in turbulent kinetic energy and break down of the shear layer confirms transition to turbulence. In the turbulent portion of the separated flow field a reverse-flow vortex is present and a small pressure recovery occurs. Since this takes place very close to the trailing edge, insufficient pressure recovery occurs to enable reattachment. As a result, the flow remains separated. This agrees qualitatively with the flow development described in Section 5.1.

By performing a reduced-order modelling technique, the individual ‘building blocks’ of the flow field have been extracted. Dynamic modes with spatial structures related to the shear layer and wake shedding were found with frequencies that agree well with those found experimentally. However, owing to the complex nature of the flow over an aerofoil at low Re , several other dynamic modes appear when the whole flow field is analysed. This agrees with the statement made by Kotapati *et al.* [71] regarding the presence of sum-difference modes.

Chapter 7

Direct Numerical Simulations of the Morphing Aerofoil

In this chapter the numerical implementation of the deforming surface into the solution algorithm as well as the results from the numerical simulation of the periodically morphing aerofoil will be presented. The phase-averaged, time-averaged and instantaneous flow fields will be analysed to provide insight into the flow control mechanism responsible for the performance improvements in Section 5.2.

7.1 Introduction

The implementation of the mesh motion will be introduced first. For the purpose of comparison with the experiments, the actuation frequencies, V_f , are non-dimensionalised as

$$V_{f+} = \frac{V_f c}{U_\infty} \quad (7.1)$$

to give the reduced frequencies, V_{f+} .

Three frequencies were chosen for in-depth investigation to help explain the experimental results in Section 5.2: $V_{f+} = 0.4$, 2 and 5, which correspond to $V_f = 14$, 71 and 177 Hz

respectively. The two lower frequencies were chosen to closely match the actuation frequencies that have a negligible effect ($10 \text{ Hz} \approx V_{f+} = 0.4$) and a significant effect ($70 \text{ Hz} \approx V_{f+} = 2$) as explained in Section 5.2. The higher frequency ($V_{f+} = 5$) was chosen to closely match the frequency of the shear layer in Figure 5.8. Owing to structural constraints this frequency could not be tested experimentally with the current set up and can be thought of as an extension of the experiments.

7.2 Implementation of Periodic Morphing

The motion of the dynamic grid due to the aerofoil morphing consists of two parts: the temporal variation of the aerofoil boundary and the temporal variation of the internal grid points. In Section 6.1, the governing equations were integrated over control volumes to provide discrete equations for the static grid. The same approach can be used for the dynamic grid but the motion of grid must be accounted for. The equations of motion for the dynamic grid case can be derived as explained in [104]. The difference compared with the static grid is that the velocity u_i in the convective term is replaced with by the relative velocity, $u_i - u_{p_i}$ [104], where u_{p_i} is the grid velocity. When the motion of the grid is not uniform (i.e. the control volumes change size with time), the different velocities of the cell faces create an artificial mass source (see Ferziger and Peric [104] for details). A further conservation law must therefore be introduced — the so-called Space Conservation Law [110] which may be expressed as

$$\frac{d}{dt} \int_V dV - \int_S u_{bj} \cdot n_j dS = 0 \quad (7.2)$$

7.2.1 Boundary Motion

Since the flow was found to have little effect on the surface motion in the experiments, the structural behaviour of the morphing surface can be considered independent of the flow solution. It is not necessary therefore to solve a coupled fluid-structure interaction problem.

Instead, an amplitude and angular frequency are prescribed at the moving boundaries. The definition of the angular frequency is trivial and is found from the reduced frequency that is being investigated. The prescription of the amplitude is more involved. In the physical model the leading and trailing edges of the deforming portion of the surface — herein referred to as the control surface — are attached to the aerofoil body by a rigid bond and a slide joint respectively. As a result, the wall-normal displacement of the control surface is zero at these chordwise locations and varies smoothly with x/c in-between. To ensure the surface motion in the computational model was physically realistic, a Bézier curve was used to define a C^1 continuous function from which a distribution of discrete amplitudes could be obtained. A Bézier curve is a parametric curve which is defined in this case by the locations of 4 control points, of which 2 lie at each end of the curve. This can be seen in Figure 7.1.

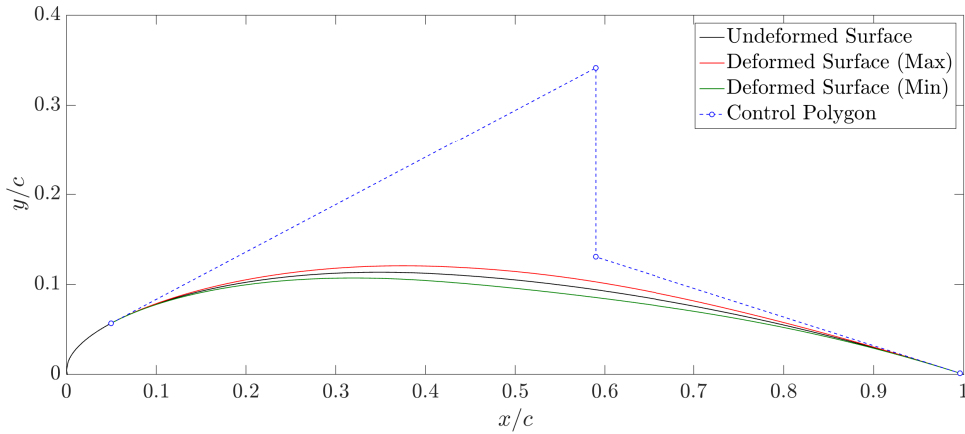


Figure 7.1: A Bézier control polygon defining deformed surface coordinates with C^1 continuity. The peak-to-peak displacement of the deformed surface has been multiplied 5 for clarity.

The Bézier curve was discretised to provide the coordinates at the intersection of the curve and the face-centre normal vector (see Figure 7.2a), for every cell on the control surface. The difference between the undeformed wall-face centre coordinate and the corresponding coordinate on the Bézier curve (given in Cartesian coordinates) is used as the amplitude of that particular wall-face. The distribution of amplitudes along the chord can be seen in Figure 7.2b. The value is zero at $x/c = 0.07$ and 0.93 and varies smoothly in-between along the surface, with a peak-to-peak displacement of 0.2% at $x/c = 0.4$ — matching what was

specified in the experiments.

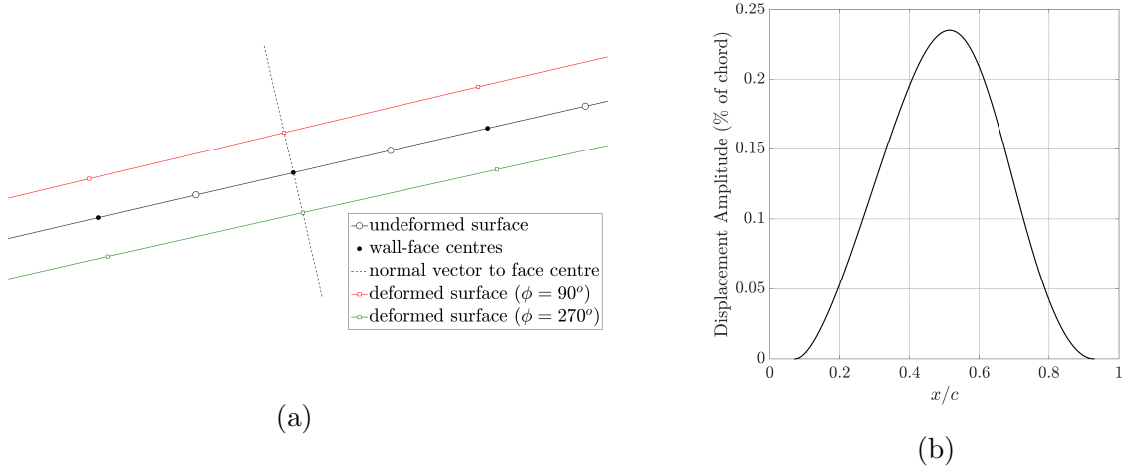


Figure 7.2: Definition of the external boundary motion: a) Definition of amplitude coordinates; b) Distribution of peak-to-peak amplitude along chord.

7.2.2 Internal Point Motion

The purpose of the internal grid point motion is to accommodate the prescribed boundary motion (defined above) whilst preserving grid quality. This enters the solution procedure at each time-step as the velocity of the grid points, u_{p_i} , at that time instant. On the moving boundary u_{p_i} will be equal to the prescribed velocity of that boundary and at the external boundaries u_{p_i} is zero. A spatial distribution of u_{p_i} is therefore sought that provides acceptable grid distortion at each time instant and is found by solving a mesh motion equation. A smooth distribution can be achieved if this equation takes the form of a Laplacian, with the diffusion coefficient set to the square of the inverse of the distance from the moving surface, l , according to

$$\frac{1}{l^2} \frac{\partial}{\partial x_j} \left(\frac{\partial x_{b_i}}{\partial x_j} \right) = 0 \quad (7.3)$$

where x_{b_i} is the point displacement and u_{b_i} can be calculated from its derivative in time.

The solution is found iteratively prior to the PISO loop before each time-step. Solving Equation 7.3 ensures that the displacement of the internal grid points is diffused inside the

domain, as can be seen in Figure 7.3a. Given that close to the wall the smallest cells are located and the highest accuracy is sought, as little cell distortion as possible is desirable in this region. As can be seen from Figure 7.3b, the grid maintains its highly orthogonal attributes despite the moving boundaries. The maximum non-orthogonality is unaffected by the grid motion and the mean non-orthogonality varies by less than 0.2% throughout the whole actuation cycle.

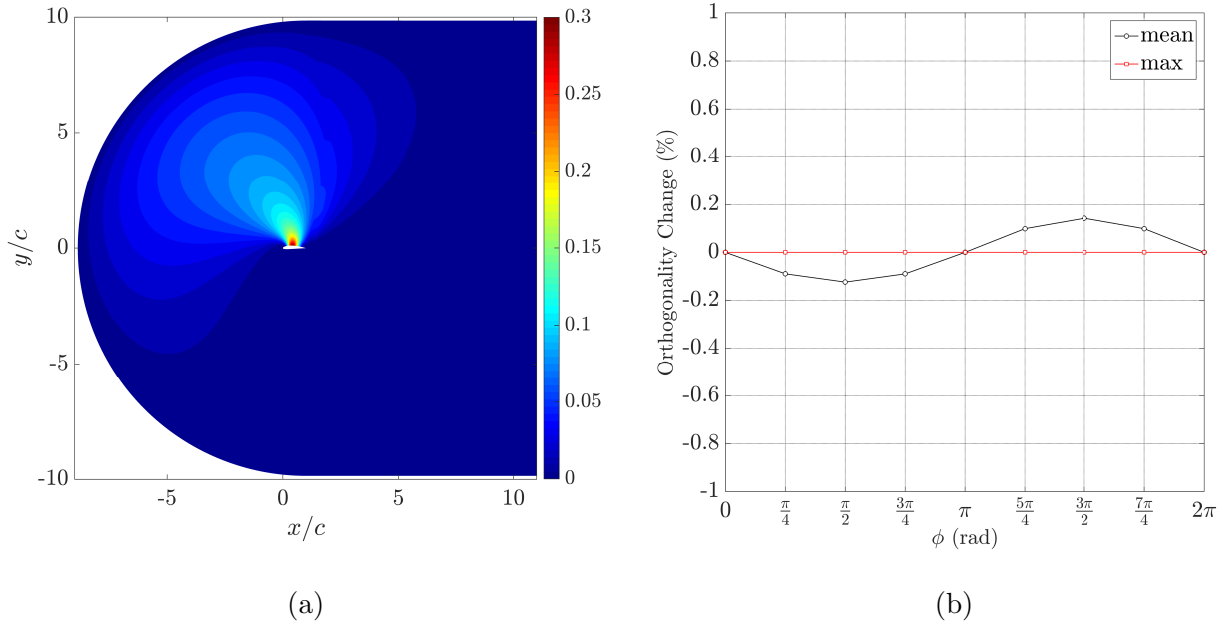


Figure 7.3: (a) The diffusion of the internal grid point motion in which the contours show the cell displacement as a percentage of the chord and (b) its effect on grid orthogonality during an actuation cycle.

7.3 Results and Discussion

For each V_{f+} tested, the solution fields at the final time instance from the static simulations (i.e. $t^* = 20$) were used as initial conditions. The flow was allowed to develop at each V_{f+} from the same starting conditions. In Section 5.2 it was found that $V_f = 10$ Hz has little effect on the flow development around the aerofoil, while 70 Hz significantly reduces the size of the separated region and improves performance as a consequence. These findings are further examined using numerical simulations. Furthermore, forcing at $V_{f+} = 5$, corresponding to the

natural frequency of the shear layer will be tested. The separated shear layer was found to have a characteristic frequency of 4.9 in Section 6.6, in agreement with the experiments.

7.3.1 Instantaneous Response of the Flow Field to Surface Motion

Figure 7.4 compares the time development of C_f in each case for the time duration $20 < t^* < 25$. This corresponds to 3, 10 and 25 actuation cycles for $V_{f+} = 0.4, 2$ and 5 respectively. The strong positive C_f from $0 < x/c < 0.4$ is a consequence of the attached flow accelerating around the aerofoil's leading edge. Further downstream, C_f becomes zero at the point of separation and is followed by a region with weak negative C_f . This is a result of the slowly recirculating flow in the dead-air region below the laminar portion of the separated shear layer (described in Section 6.5.3). In the case of $V_{f+} = 0.4$, the slowly recirculating region extends all the way to $x/c = 0.9$ before a small region of stronger negative C_f occurs. This is confined to $0.9 < x/c < 1.0$.

In contrast, substantial changes in the distribution of C_f downstream of separation take place when $V_{f+} = 2$ and 5. The slowly recirculating region is still present downstream of separation but terminates at around $x/c = 0.7$, at which point an alternating positive and negative distribution is set up. The appearance of regions of strong negative C_f is indicative of shear stress resulting from clockwise rotating vortices convecting downstream. The regions of strong positive C_f are located between two adjacent vortices, known as a ‘braid’ region [111].

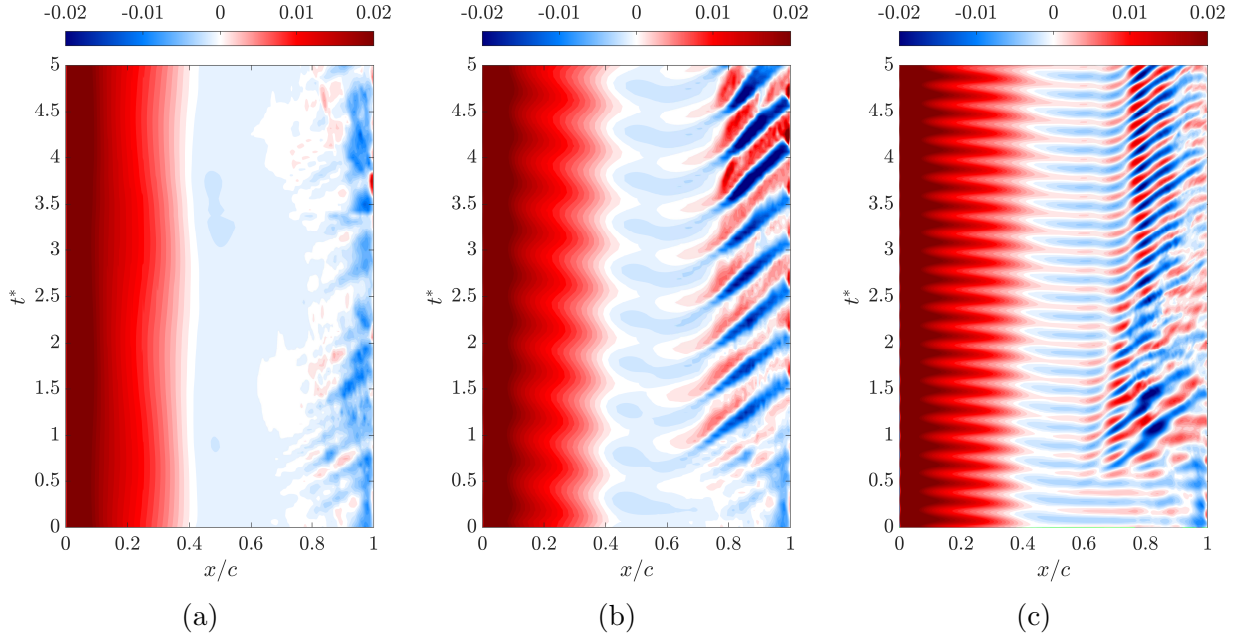


Figure 7.4: Space-time diagram of instantaneous spanwise-averaged C_f : a) $V_{f+} = 0.4$; b) $V_{f+} = 2$; c) $V_{f+} = 5$.

Despite the different number of actuation periods having taken place, the strong negative C_f first occurs at $t^* = 0.8$ in both Figure 7.4b ($V_{f+} = 2$) and Figure 7.4c ($V_{f+} = 5$). This suggests the flow needs to have developed almost entirely over the actuated surface before it responds to the actuation. When $V_{f+} = 2$, the regions are thicker and fewer than when $V_{f+} = 5$. This is expected because the actuation period is longer.

Further insight into the cause of the C_f distributions is given by the instantaneous, spanwise-averaged Ω_z contours in Figure 7.5. Figure 7.5a shows the instant of actuation start (i.e. $\phi = 0^\circ$ of the first actuation cycle where ϕ is the phase angle of the actuation cycle) and is therefore the same for all values of V_{f+} . The figures beneath correspond to the same phase of actuation ($\phi = 0^\circ$) for the first 4 cycles at each actuation frequency.

When $V_{f+} = 0.4$, the shear layer can be seen to roll up and form a clockwise-rotating vortex at $x/c = 0.9$. This vortex remains at the same chordwise location and the same distance away from the aerofoil surface. This agrees with the region of negative C_f described above for the $V_{f+} = 0.4$ case. The fact that the negative C_f is confined to $0.9 < x/c < 1.0$ is explained by the location of the shear layer roll up. The relatively weak magnitude of C_f is a result of the vortex remaining far from the aerofoil surface.

The roll-up of the separated shear layer is also observed at the beginning of the 3rd and 5th cycles when $V_{f+} = 2$ and 5 respectively. However, compared with the $V_{f+} = 0.4$ case, this occurs much further upstream and causes a significant change in the flow development. This explains why the region of slowly recirculating fluid only extends up to $x/c = 0.7$ at these frequencies as shown in Figure 7.4b and 7.4c. The vortices at the two higher frequencies also appear more distinct and are located closer to the surface than when $V_{f+} = 0.4$, causing the stronger negative C_f .

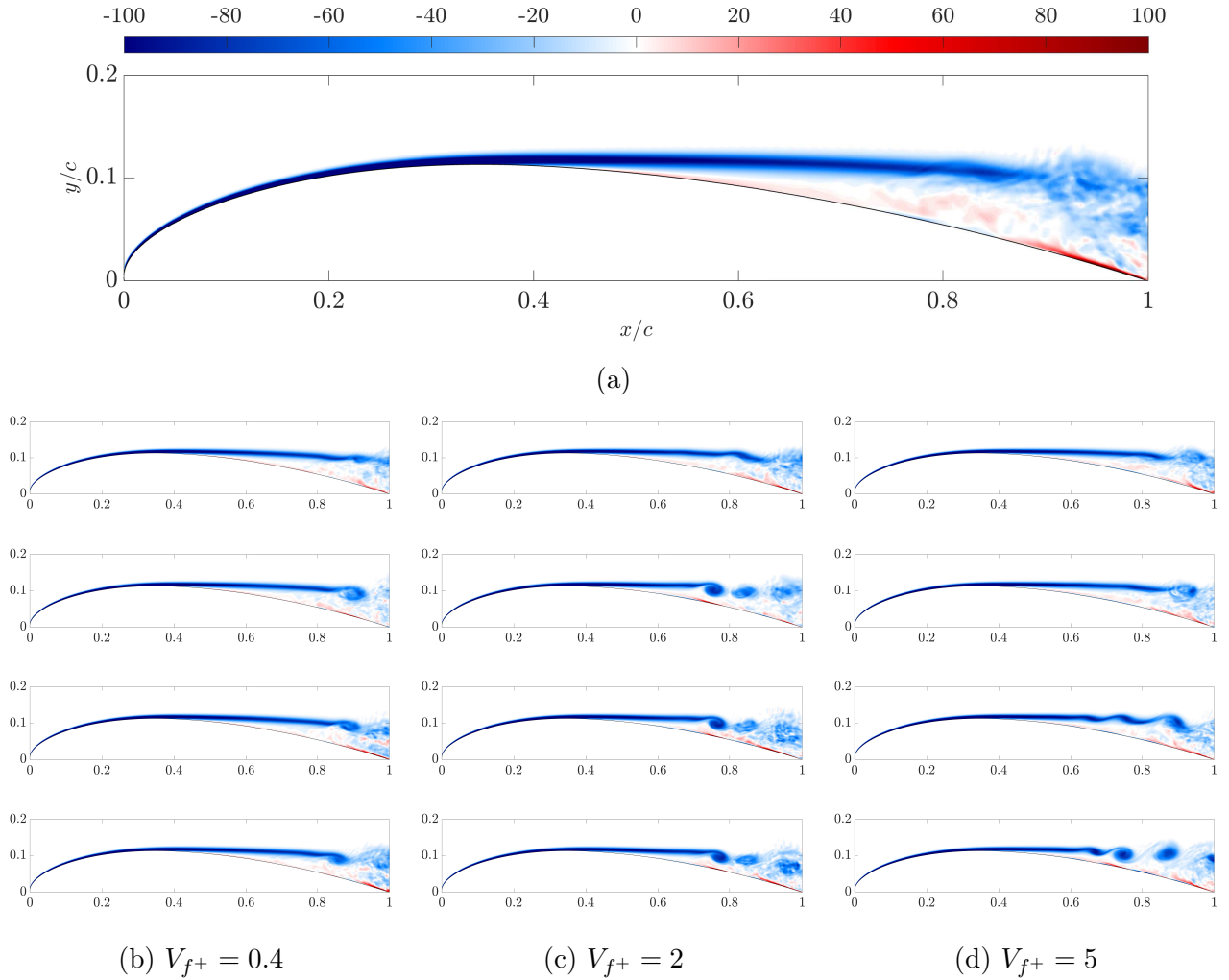


Figure 7.5: Iso-contours of instantaneous spanwise-averaged Ω_z at $\phi = 0^\circ$ for the first 5 cycles. Cycle number increases from top to bottom: a) $\phi = 0^\circ$ of first cycle (instance of actuation start is the same for all V_{f+}).

It is worth reiterating at this point that $V_{f+} = 0.4$ and 2 correspond to 10 Hz and 70 Hz in experimental dimensions respectively. In Section 5.2 it was found that $V_f = 10$ Hz has only a

small effect on aerofoil performance at $\alpha = 0^\circ$, while $V_f = 70$ Hz significantly reduces separation, increases C_L and decreases C_D . Evidently then, the early roll-up of the separated shear layer (Figure 7.5) and resulting strong vortex shedding (Figure 7.4) play an important role in the control mechanism. The slight shortening of the shear layer, and apparent stabilisation of the vortex shedding from it, when $V_{f+} = 0.4$ is the cause of the small performance improvement at $V_f = 10$ Hz observed in Section 5.2.

To investigate the onset of the roll-up process, 7 instantaneous snapshots of the spanwise-averaged Ω_z during the cycle in which it took place are shown in Figures 7.6 and 7.7 for $V_{f+} = 2$ and 5 respectively. These snapshots begin (top) from $\phi = 45^\circ$. The subsequent figures (from top to bottom) are each separated by 45° such that, when the corresponding $\phi = 0^\circ$ snapshot from Figure 7.5 is included, the entire actuation cycle in which shear layer roll-up occurs is presented.

At the beginning of cycle 4, the shear layer is similar to that which is seen in Figure 7.5a, except that a waviness is present at $x/c = 0.8$. As the actuation cycle progresses, this waviness is amplified until it develops into a distinct vortex at $x/c = 0.82$ when $\phi = 225^\circ$. This vortex then sheds into the wake and another vortex develops behind it, at roughly $x/c = 0.75$. This also sheds before a further vortex forms, even further upstream, at $x/c = 0.7$.

A similar pattern is observed when $V_{f+} = 5$ but the initial waviness contains two vortices. As the cycles progresses these vortices become more distinct. A similar roll-up of the separated shear layer is frequently referred to in the literature of periodic flow control and mixing layers. It is attributed to the K-H instability mechanism [112, 111, 71].

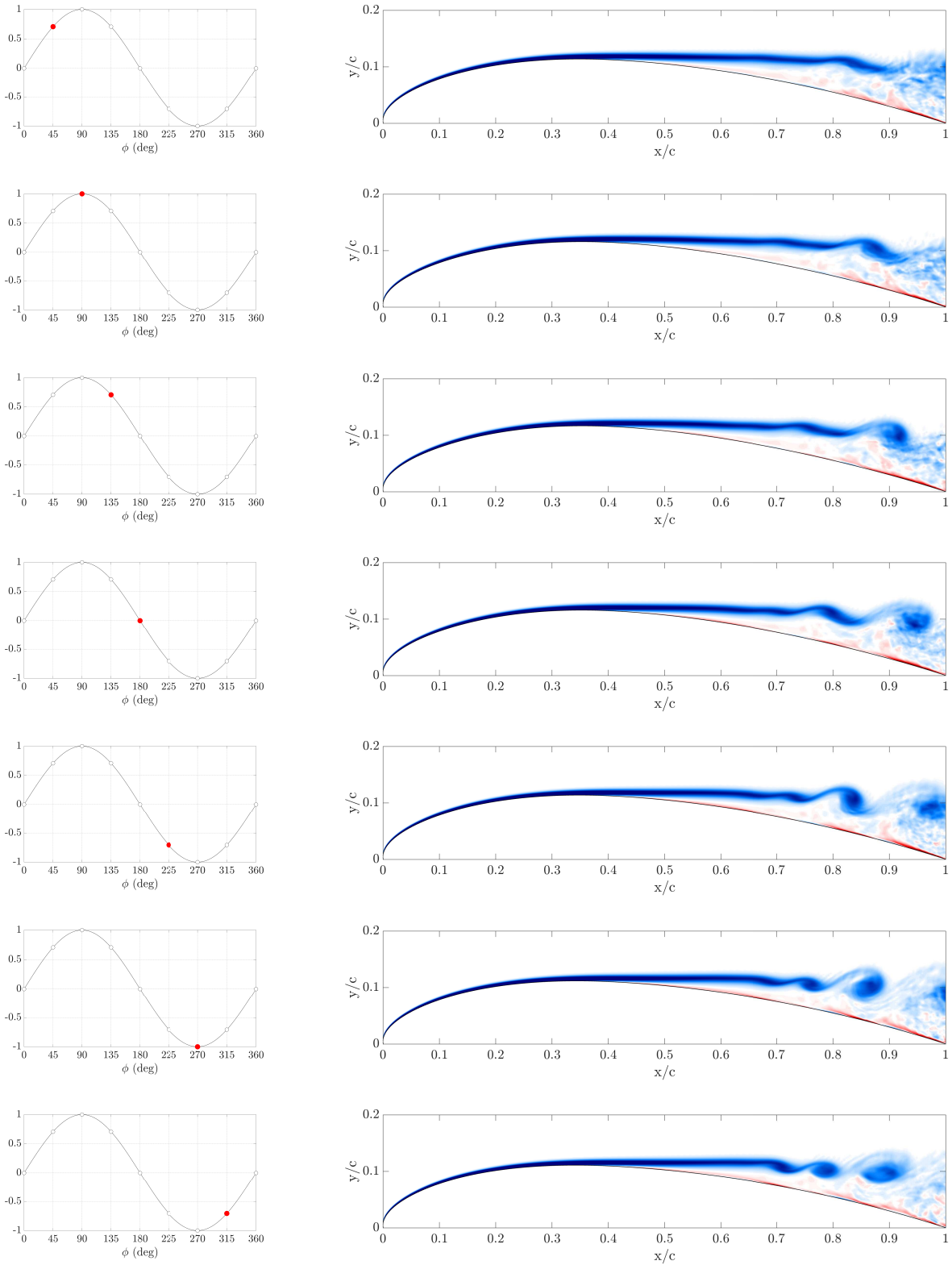


Figure 7.6: Contours of instantaneous spanwise-averaged Ω_z depicting the onset of shear layer roll up during the 2nd actuation cycle when $V_{f+} = 2$. Phase angle ϕ increases by 45° from 45° to 315° radians from top to bottom.

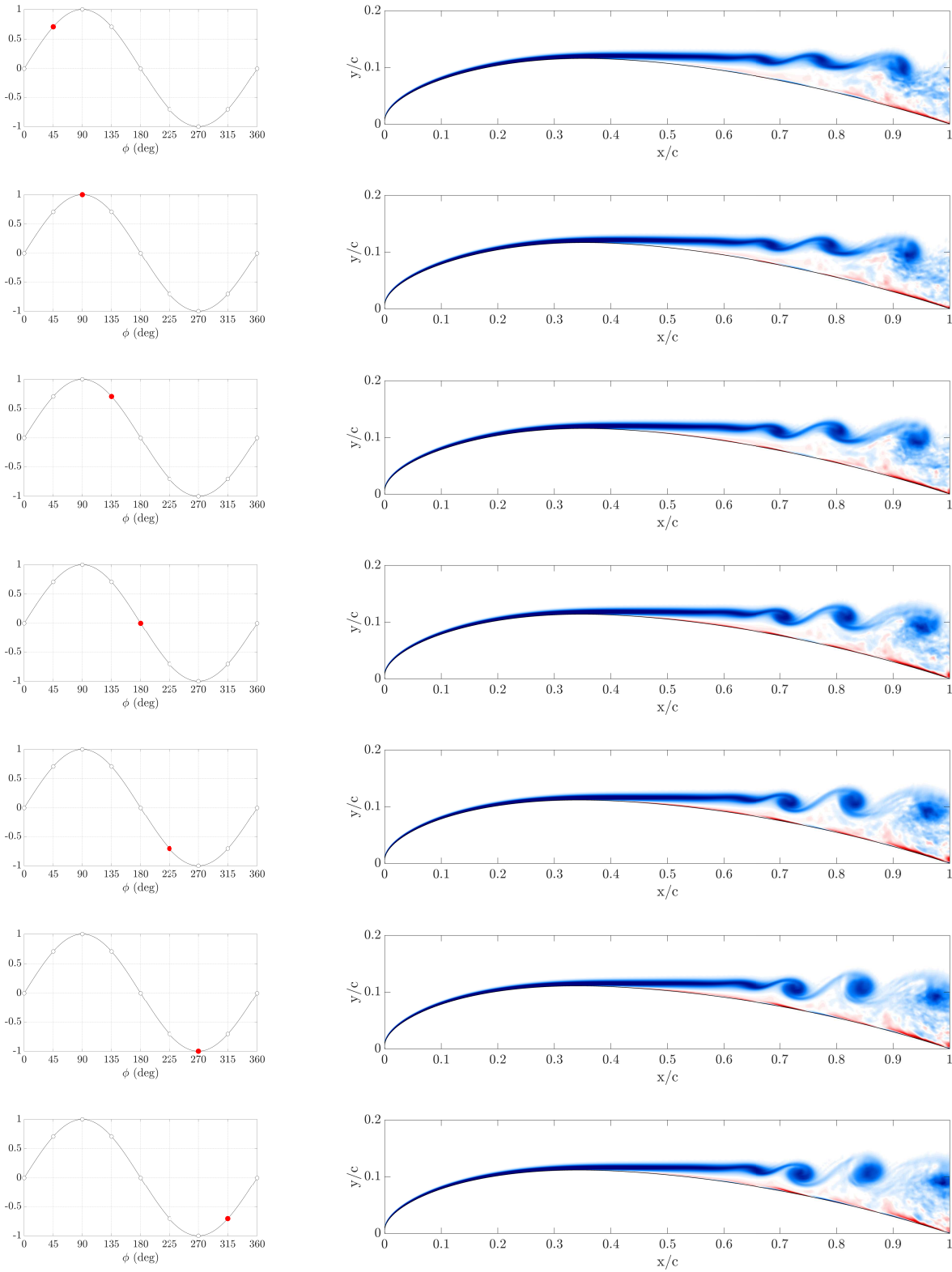


Figure 7.7: Contours of instantaneous spanwise-averaged Ω_z depicting the onset of shear layer roll up during the 4th actuation cycle when $V_{f+} = 5$. Phase angle ϕ increases by 45° from 45° to 315° radians from top to bottom.

It is clear from the difference between the $V_{f+} = 0.4$ case and the $V_{f+} = 2$ and 5 cases in Figure 7.4 that the amplification of the K-H instability has a profound impact on the downstream development of the flow field. Furthermore, as already mentioned above, this change in flow development (i.e. early roll-up of the shear layer) plays a vital role in the performance improvements observed experimentally in Section 5.2.3.

Figure 7.8 displays instantaneous snapshots of the streamwise velocity, u/U_∞ , and Q-criterion at $\phi = 360^\circ$ for the cycle 2nd and 4th cycle for $V_{f+} = 2$ and 5 respectively, i.e. the end of the actuation cycle in which shear layer roll-up occurred (equivalent to $\phi = 0^\circ$ for 3rd and 5th cycle in Figure 7.5). The Q-criterion reveals the appearance of large, spanwise-coherent vortical structures (LCS). These LCS are a characteristic feature of the K-H type instability [111] and have been identified in the literature [71] as the fundamental mechanism in periodic control. As they propagate downstream they entrain high momentum fluid from the outer flow towards the aerofoil surface, thus re-energising the near-wall flow.

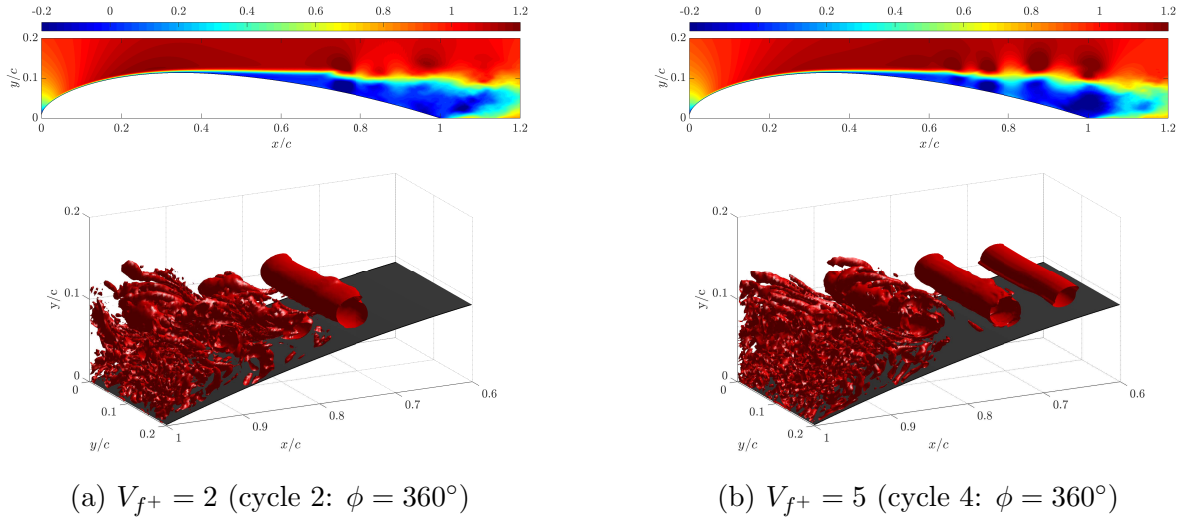


Figure 7.8: Instantaneous spanwise-averaged u/U_∞ (top) and Q-criterion (bottom) for $\phi = 360^\circ$ of the cycle in which the shear layer roll-up occurred.

From what is presented above, and knowledge of the available literature, the K-H instability certainly plays a pivotal role in the flow control mechanism since it initiates a clear change in the state of the flow field. However, as seen in Figure 7.8, the flow fields upon the triggering of this instability still exhibit large regions of separation and reverse flow. This remains true even 3 cycles after the shear layer roll-up occurred, when the LCS have had the chance to

propagate beyond the trailing edge and therefore impact upon the entire separated region (Figure 7.9).

It was stated in the experimental results of Section 5.2 that the control objective being pursued was separation control so, in spite of the triggering of the K-H instability, the control objective has not yet been achieved. That said, the separated regions are more suppressed in Figure 7.9 than they are in Figure 7.8 and so it does seem that the actuation is having an increasing positive effect with an increasing number of actuation cycles.

To investigate the continued development of the flow field under the influence of control, the $V_{f+} = 2$ simulation was continued until $t^* = 32.5$ so it had also performed 25 cycles – equal to that of $V_{f+} = 5$ case. The iso-contours of Ω_z , u/U_∞ and the iso-surfaces of $Q = 250$ for $\phi = 360^\circ$ of the 25th cycle are shown in Figure 7.10.

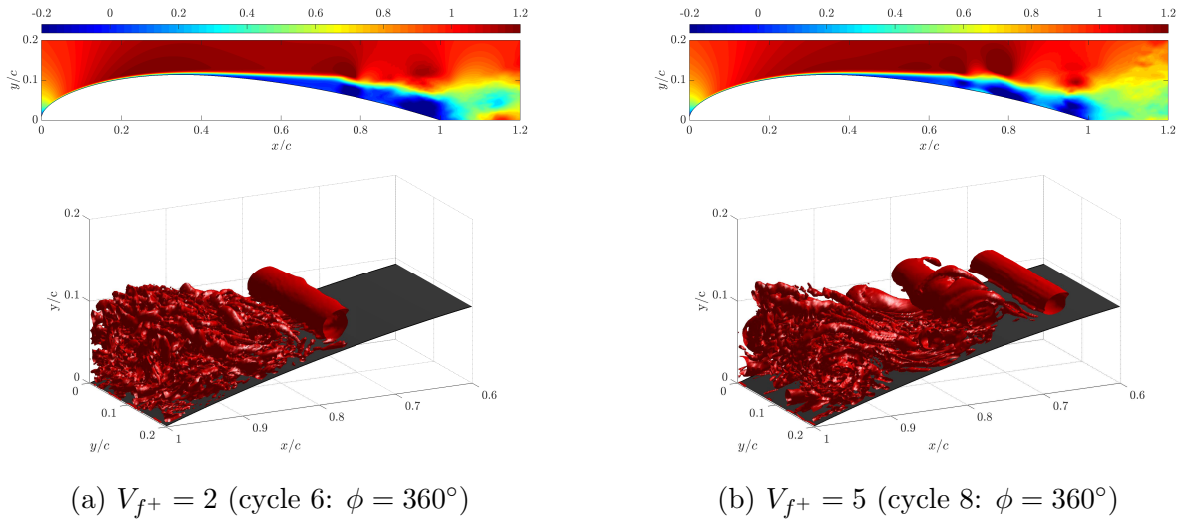


Figure 7.9: Instantaneous spanwise-averaged u/U_∞ (top) and Q -criterion (bottom) for $\phi = 360^\circ$ 3 cycles after the shear layer roll-up occurred.

Comparing the size of the separated regions in Figure 7.10 with those in Figure 7.8 it is clear the actuation at $V_{f+} = 2$ and 5 have significantly reduced the size of the separated region. This is consistent with what was found in the experiments of Section 5.2. When $V_{f+} = 5$ the flow appears more spanwise coherent than when $V_{f+} = 2$ and two LCS are present of the aerofoil surface (compare Figure 7.10a and Figure 7.10b). This agrees with the spatial DMD structure found at the characteristic shear layer frequency of 4.9 in Section 6.6. The enhanced

mixing that these produce has almost entirely eliminated the separated region at the trailing edge. This method of separation control can therefore be considered a success.

The control mechanism began with the early roll-up of the separated shear layer owing to the amplification of the K-H instability. This first occurred during the 2nd and 4th cycle for the $V_{f+} = 2$ and 5 cases respectively and results in the formation of LCS, which are effective at transferring momentum. The presence of these structures further upstream means more momentum transfer can occur over the aerofoil surface. Over time, this has enabled a significant reduction in separation by re-energising the near-wall flow, helping it to overcome the APG. However, from analysis of the flow fields in Figures 7.8-7.10, it has been found that this is a gradual process. As the number of actuation cycles increases, the shear layer gradually gets closer to the aerofoil surface and the size of the separated region decreases. For the reader's reference, the flow fields at $\phi = 0^\circ$ for all actuation cycles from 1-25 can be found in appendix A.5 and A.6.

Figure 7.11 shows an $x-t^*$ diagram of C_f for the final 10 cycles (i.e. cycles 15-25) and shows the flow has developed into a repeatable, periodic state after 15 cycles. The remainder of this chapter will concentrate on the developed flow and add weight to the arguments made in the above paragraphs. The statistics presented in the sections that follow were taken during the last 10 cycles.

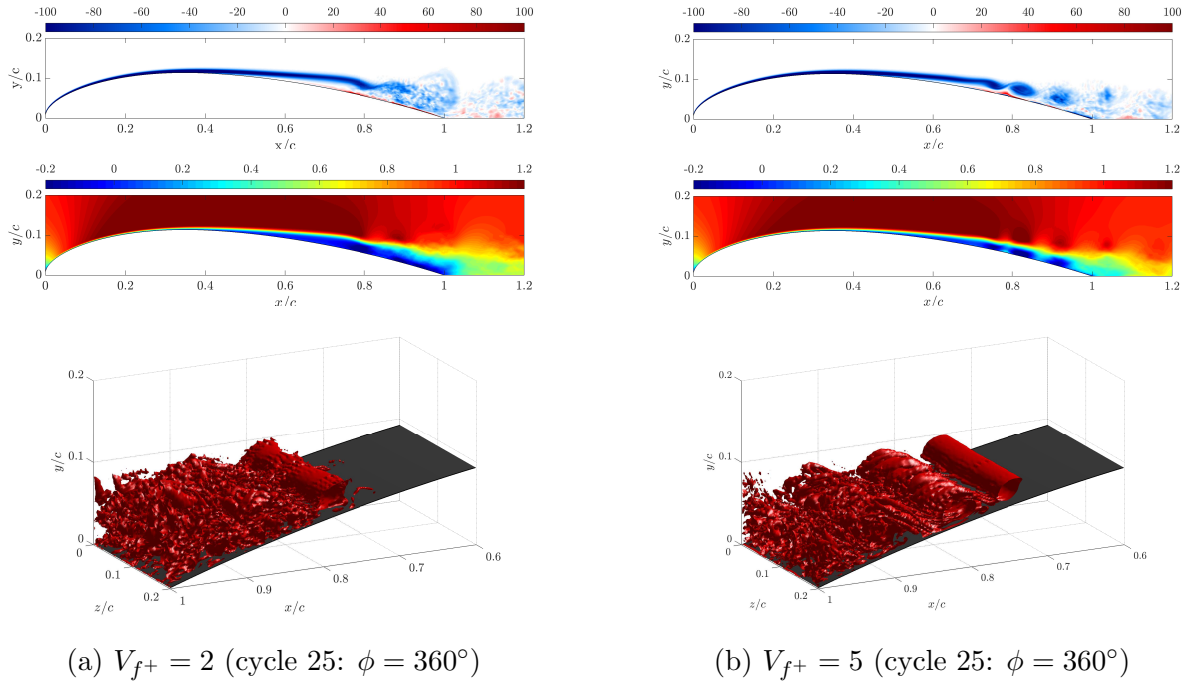


Figure 7.10: Instantaneous spanwise-averaged Ω_z (top), spanwise-averaged u/U_∞ (middle) and Q-criterion (bottom) for $\phi = 360^\circ$ 25 cycles after actuation start.

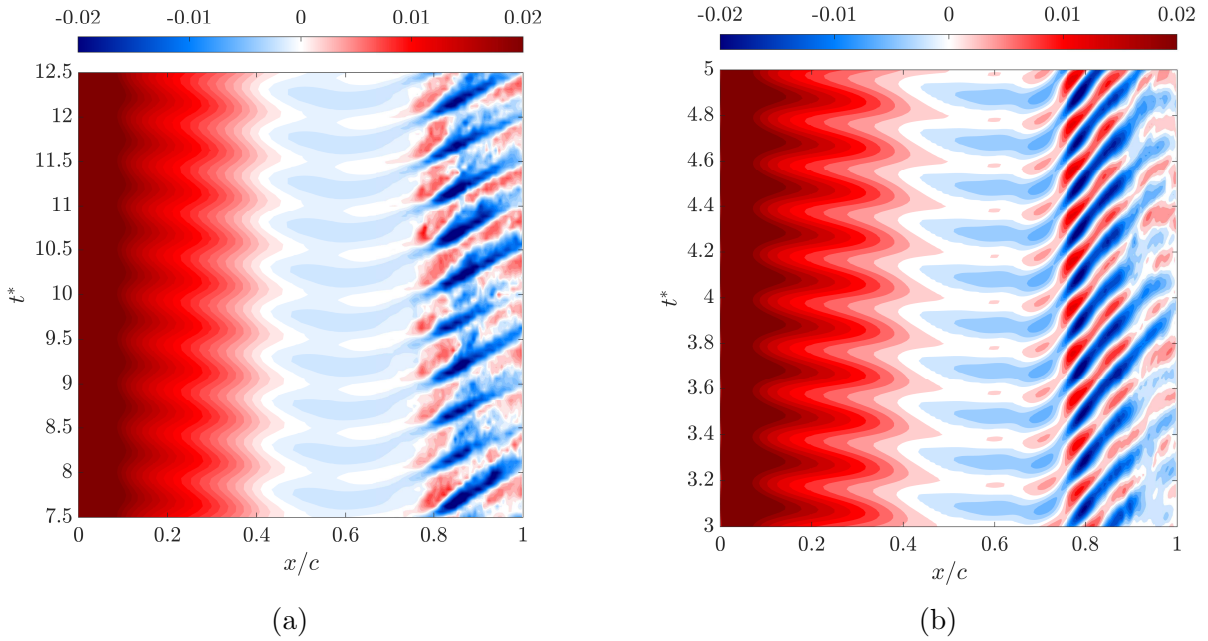


Figure 7.11: Space-time diagram of instantaneous spanwise-averaged C_f : a) $V_{f+} = 2$; b) $V_{f+} = 5$.

7.3.2 Phase-Averaged Results

For a better understanding of the control mechanism, the spanwise-averaged Ω_z and Q-criterion were phase-averaged every 45° for the last 10 cycles. The temporal evolution of the flow fields is shown in Figure 7.12 and Figure 7.13 for one complete cycle.

In the case of $V_{f+} = 2$ (Figure 7.12), the shear layer at the beginning of the cycle ($\phi = 0^\circ$) is located much closer to the surface than it was in Figure 7.5a, which is consistent with the large reduction in the size of the separation region observed in Figure 7.10. The Q-criterion reveals an LCS located at $x/c = 0.8$. This structure is coincident with the location of shear layer roll-up, seen clearly at 90° – the peak of the surface motion. As the cycle continues to evolve, the LCS is ejected from the shear layer and propagates towards the trailing edge, losing its spanwise coherence as it does so. By $\phi = 270^\circ$ – the trough in surface motion – only a modest spanwise coherence remains. At $\phi = 315^\circ$, when the surface is moving in an upward direction, a new LCS has been produced at $x/c = 0.8$.

The above description supports the idea that periodic actuation promotes and regulates the production of LCS [69]. Dynamic surface actuation at $V_{f+} = 2$ generates an LCS during the up-stroke of an actuation cycles before ejecting this structure from the shear layer on its downstroke.

When $V_{f+} = 5$ the shear layer can again be seen to roll up to form an LCS. The LCS are produced further upstream than they are in the $V_{f+} = 2$ case and appear capable of maintaining their spanwise coherence for the duration of the cycle. It can also be seen that two such structures are present on the aerofoil surface which is in contrast with the $V_{f+} = 2$ case where only one was observed during a cycle. The smaller size of these vortices, relative to the $V_{f+} = 2$ case, means that there is a smaller region of the aerofoil surface under the influence of their low pressure core. The fact that the spatial structure that forms at this frequency is, as already mentioned, similar to that of the characteristic shear layer frequency (seen in Section 6.6) suggests that the periodic actuation is reinforcing a naturally occurring structure. This was a concept that was postulated by Benard and Moreau who suggested that the actuation acts as a catalyser to amplify flow structures that already exist in the natural flow [75].

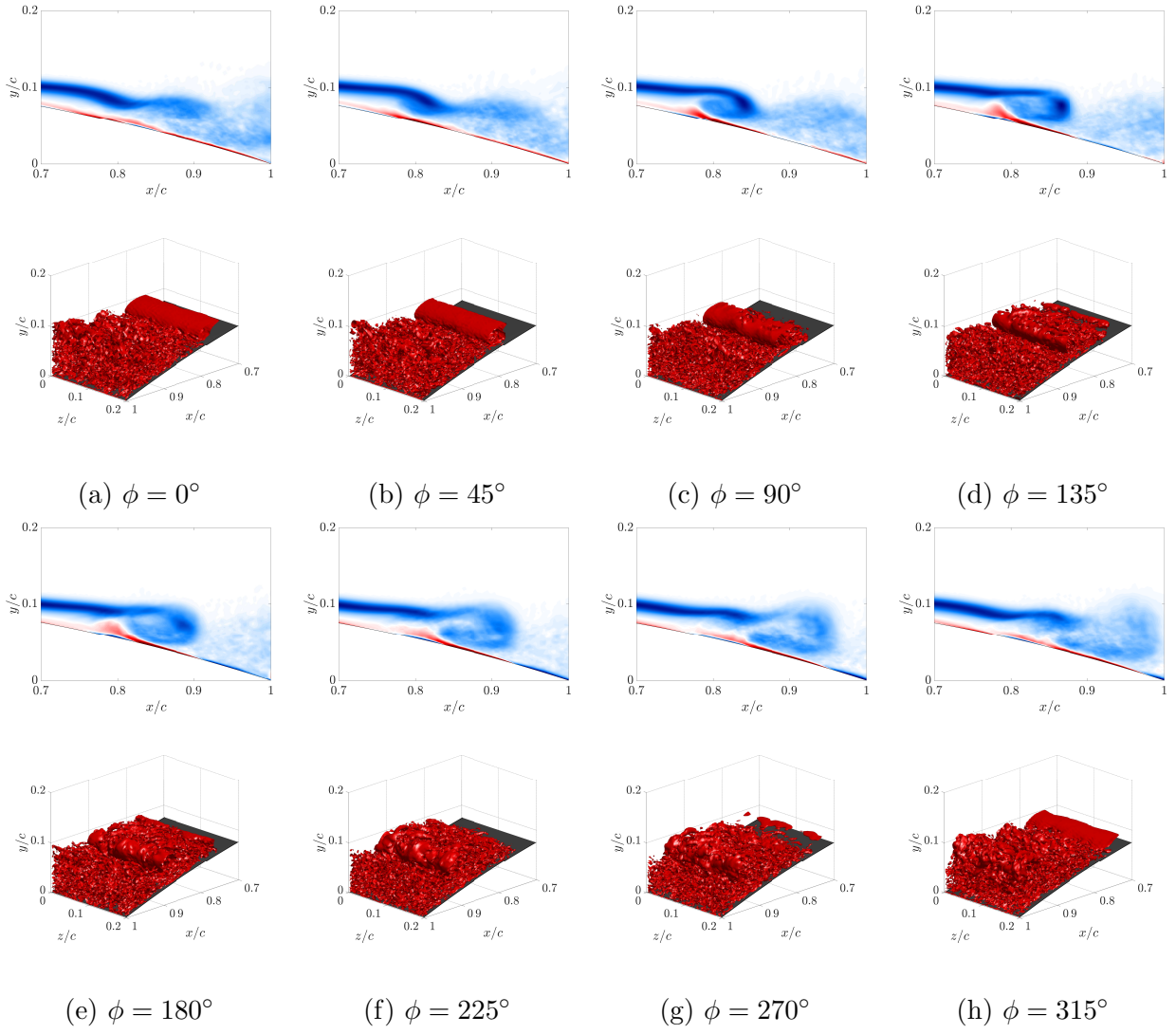


Figure 7.12: Spanwise-averaged Ω_z (Top) and Q-criterion (Bottom) phase-averaged over 10 actuation cycles when $V_{f+} = 2$.

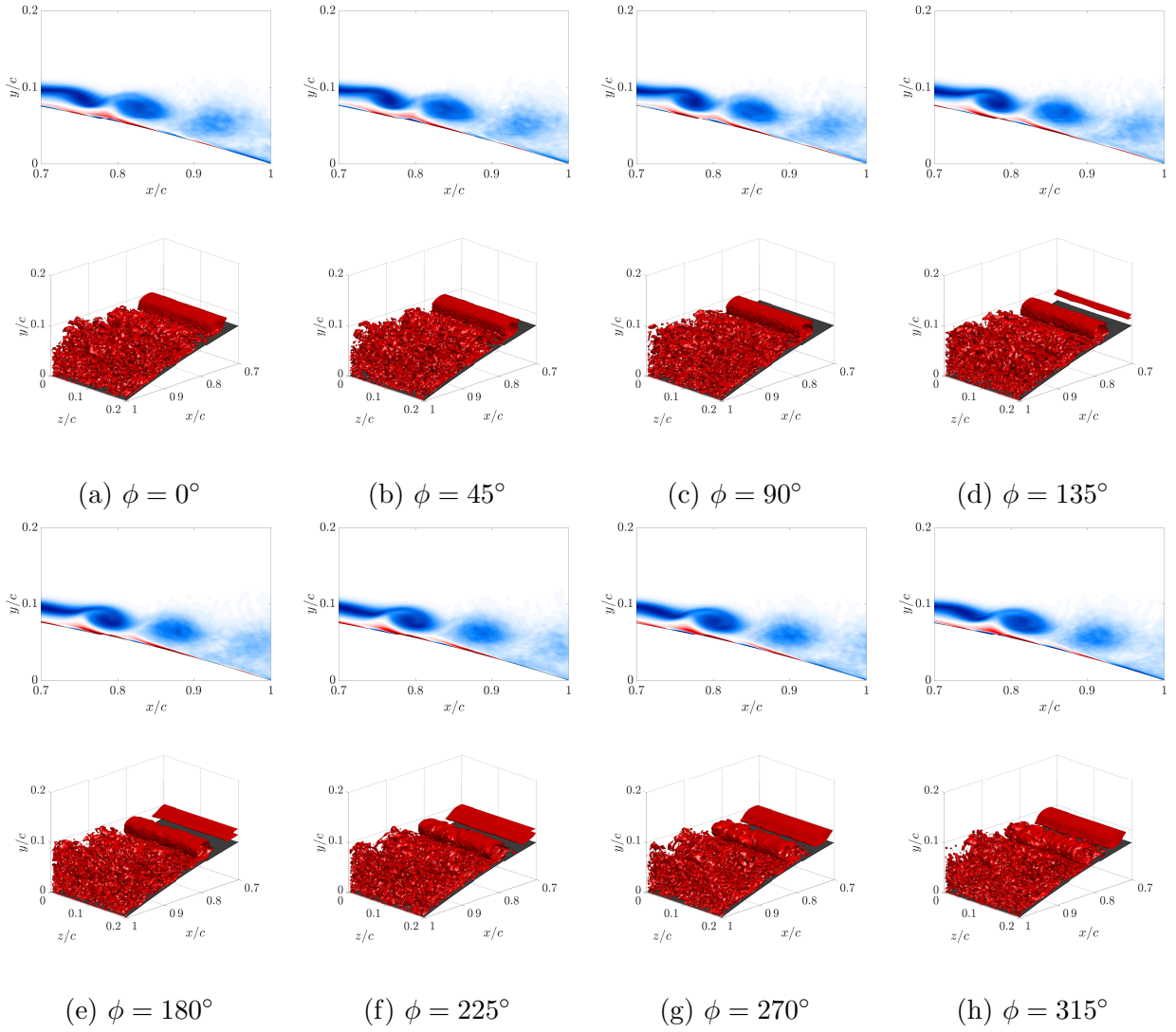


Figure 7.13: Spanwise-averaged Ω_z (Top) and Q-criterion (Bottom) phase-averaged over 10 actuation cycles when $V_{f+} = 5$.

The phase-averaged plots reveal that once the flow has developed over the actuated aerofoil, LCS are produced during every actuation cycle. This is consistent with what was seen in Figure 7.4 and suggests that the shear layer is ‘locked-on’ to the surface motion. The locking-on of the shear layer to the forcing frequency was observed by Kotapati *et al.* [71] in their investigation of synthetic-jet based periodic control. In agreement with what has already been described, they found an amplification of the K-H type instabilities and subsequent roll-up of the separated shear layer into discrete vortices with the presence of shear layer lock-on. The well organised vortical flow seen in Figure 7.13 is a consequence of only one characteristic frequency and therefore, the absence of sum/difference modes which occurred in

the uncontrolled case. This has also been observed by Wu *et al.* [113] and Kotapati *et al.* [71]. Moreover, since the vortices are being created and shed along the aerofoil surface, the associated entrainment of far-field flow is concentrated in the near-wall region, thus energising the boundary layer. Note that this was not the case when these LCS first appeared in the flow at the onset of shear layer roll-up. This supports the idea that there is more to the control mechanism than just the K-H instability.

It is proposed that the triggering of K-H instability is the first in a chain of events that eventually leads to separation control. Amplification of the K-H instability causes the early onset of shear layer roll-up and the creation of LCS. By entraining fluid from the outer flow and re-energising the separated region, these structures can form closer and closer to the aerofoil surface with each passing actuation cycle until such a point (around the 15th cycle) when they form on the aerofoil surface. Once this happens they are directly re-energising the near-wall flow. As a result, the near-wall flow recovers enough energy to overcome the APG. Consequently the separation zone is reduced. This hypothesis will be further explored in the following section.

7.3.3 Time-Averaged Results

The process of time-averaging a velocity field involves averaging the velocity components at fixed points in space (the cell centres). Since the dynamic simulations involve time-varying cell centre positions, care must be taken in the time-averaging process. Quantities on the aerofoil surface (such as C_P and C_f) can be time-averaged as normal since they are always stored at the same locations on the aerofoil boundary but flow fields can only be averaged in a region in which the cell centres are stationary – or moving by only a very small value. The maximum cell centre displacement from the unactuated position was shown in Figure 7.3a and is reproduced in Figure 7.14a in the region around the aerofoil. The cell centres move a maximum of less than 0.3% of the chord, which corresponds to the maximum surface displacement in Figure 7.2b. While this is a small variation, the velocity gradients are high in this region so a small change in spatial location can produce a relatively large change in velocity. It is therefore not appropriate to perform a time-averaging over much of the aerofoil

surface. However, owing to the definition of the boundary motion, as seen in Figure 7.2b, the surface does not move for $x/c > 0.9$. Furthermore, as shown in Figure 7.14b, the diffusion of the grid motion does not affect the cells in this region, or downstream of it.

The maximum cell displacement in the region enclosed by the white square in Figure 7.14a is less than 0.03% of the aerofoil chord and this occurs at a location ($x/c = 0.9$, $y/c = 0.2$) where the velocities gradients are small. It is therefore possible to perform time-averaging in this small window. It is also clear from Figures 7.14c and 7.14d that this is a region of particular interest since it encompasses the majority of the reverse flow and the region of high TKE (Figures 7.14c-e refer to the uncontrolled flow).

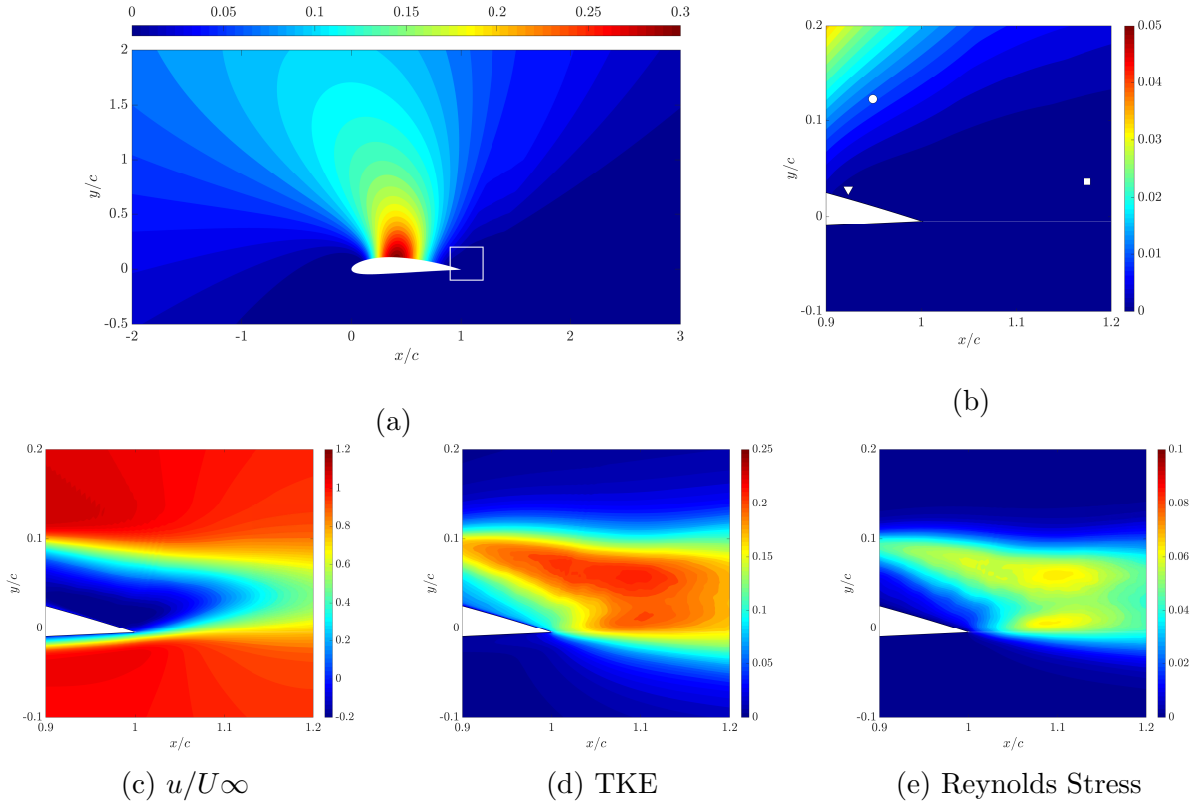


Figure 7.14: Time-averaged, spanwise-averaged contours of the uncontrolled flow fields in a region close to the trailing edge, shown in (a) as a white square. This region has negligible grid motion (b) and also contains the majority of the reverse flow and TKE (c-e).

For the cases of $V_{f+} = 2$ and 5, the time-averaging was performed using 80 snapshots from the final 10 cycles (i.e. 8 snapshots per cycle). Figure 7.15 shows that this number of samples is sufficient to provide a statistically converged time average for both actuation frequencies. The

time-averaged streamwise velocity, TKE, and Reynolds stress, $\overline{u'v'}$, are shown in Figure 7.16 and 7.17.

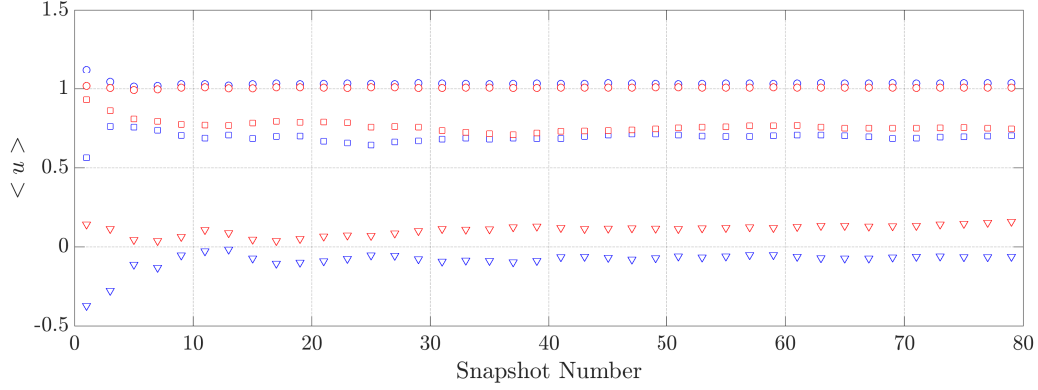


Figure 7.15: Value of the time-average streamwise velocity with increasing number of snapshots when $V_{f+} = 2$ (blue) and $V_{f+} = 5$ (red). The symbols correspond to points in Figure 7.14b.

In Section 6.5.3 it was found that transition to turbulence occurred as a result of the natural breakdown of the separated shear layer. This means that the regions of high TKE and Reynolds stress occur in the shear layer, away from the aerofoil surface. This can be seen in Figure 7.14d where a clear gap between the region of high TKE and the aerofoil surface is present. The mixing of momentum in regions of high TKE is well promoted but, since it occurs far from the aerofoil surface, this mixing does not naturally cause reattachment [74]. However, in the actuated cases in Figures 7.16 and 7.17, the region of high TKE (high mixing of momentum) has been brought closer to the aerofoil surface, enabling it to re-energise the boundary layer. This is consistent with the hypothesis made in Section 7.3.2. Comparison of the streamwise velocity fields in Figures 7.16a and 7.17a with the unactuated time-averaged fields in Figure 7.14c shows that this has caused a large reduction of the size of separated region at both frequencies and the reverse flow region, located at the trailing edge of the unactuated aerofoil, has been eliminated.

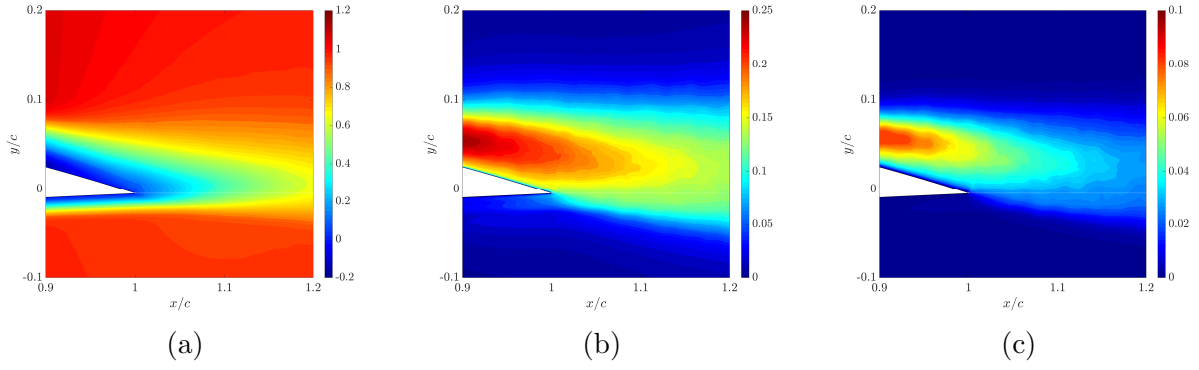


Figure 7.16: Iso-contours of time-averaged flow fields close to the trailing edge when $V_{f+} = 2$: a) $\bar{u}/U_\infty$; b) TKE; c) Reynolds Stress.

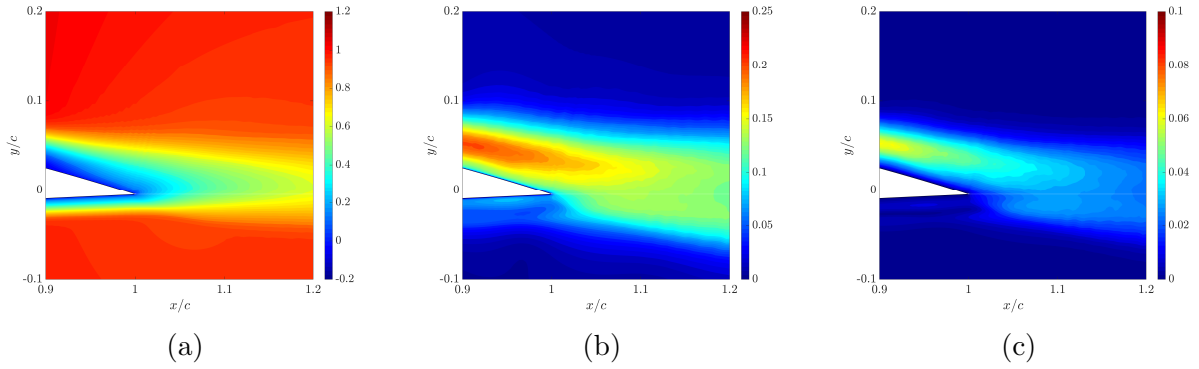


Figure 7.17: Iso-contours of time-averaged flow fields close to the trailing edge when $V_{f+} = 5$: a) $\bar{u}/U_\infty$; b) TKE; c) Reynolds Stress.

It can therefore be confirmed that dynamic surface motion at $V_{f+} = 2$ and 5 is able to gradually move the region of high momentum transfer from a region far away from the aerofoil surface to region close to the aerofoil surface. This then results in significant reduction in separation. The effect this has on aerodynamic performance is presented below.

7.3.4 Effect of Actuation on Aerodynamic Performance

It was found from the experimental results in Section 5.2.3 that when the surface was actuated at $V_f = 70$ Hz the resulting suppression of separation led to a significant performance recovery by increasing C_L and decreasing C_D . In agreement with this, the time-averaged results in the previous section showed that actuation at $V_{f+} = 2$ almost eliminated flow separation. The effect this has had on the aerodynamic performance of the aerofoil can be seen from the C_P

distribution in Figure 7.18 and the force coefficients in Table 7.1. The time-averaging was performed over the same time periods as in the previous section (Section 7.3.3).

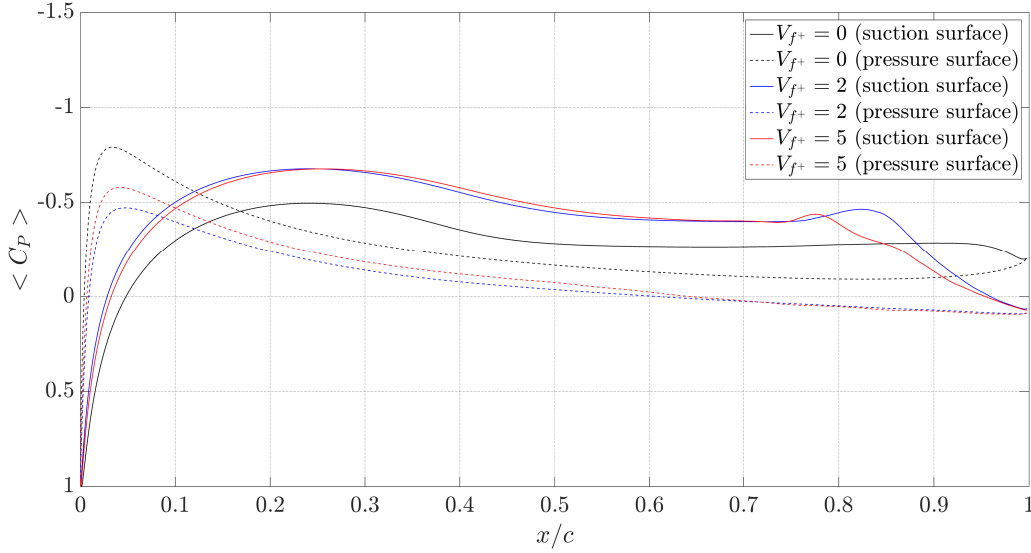


Figure 7.18: Time-averaged, spanwise-averaged C_P distributions over the actuated aerofoil surface compared with the unactuated aerofoil surface.

It has already been described in Section 6.5.3 that separation without reattachment in the unactuated case caused a flat C_P distribution, with a low pressure peak. From Figure 7.18, dynamic surface actuation at $V_{f+} = 2$ and 5 causes the pressure plateaus to be much shorter, terminating at $x/c = 0.76$ when $V_{f+} = 2$ and 0.72 when $V_{f+} = 5$ compared with $x/c = 0.93$ in the unactuated case. The improvement in pressure recovery over the aft portion of the aerofoil allows for a much larger negative pressure at the suction peak.

As can be seen in Table 7.1, this results in improvements in the aerodynamic force coefficients – which compare reasonably well with the experiments. When $V_{f+} = 2$, C_L increases by 183% and C_D decreases by 32%. $V_{f+} = 5$ has a greater decrease in C_D with a 37% reduction but the lift recovery is slightly less at 141%. This maybe a consequence of the larger vortex (and therefore larger low pressure core) in the $V_{f+} = 2$ case. This equates to a threefold L/D increase from 2.3 to 9.4 and 8.8 for $V_{f+} = 2$ and 5 respectively.

V_{f+}	Experimental		Computational			
	C_L	C_D	C_L	C_D	C_{D_p}/C_D	C_{D_v}/C_D
0	0.10	0.065	0.12	0.043	0.80	0.20
2	0.28	0.029	0.34	0.036	0.76	0.24
5	—	—	0.29	0.033	0.72	0.28

Table 7.1: Aerodynamic force coefficients from the dynamic simulations. Experimental values are included for comparison.

Table 7.1 also shows a key advantage of this control method. It is clear from the contribution of pressure drag, C_{D_p} , and viscous drag, C_{D_v} , that the vast majority of the total drag comes from the pressure. This is expected since there is a large separated region. Passive methods mentioned in the literature review attempt to reduce this by forcing the boundary layer to become turbulent upstream of the natural transition point. This keeps the boundary layer attached and hence reduces pressure drag. However, since the boundary layer is turbulent, the viscous drag is substantially increased. Often the decrease in the C_{D_p} is greater than the associated increase in C_{D_v} and therefore a net reduction in C_D does occur. However, this particular control method does not rely on tripping the boundary layer and as a result, a 20% and 31% reduction in C_{D_p} is achieved when $V_{f+} = 2$ and 5 respectively, with negligible change in C_{D_v} . The slight tendency for the portion of C_D coming from C_{D_v} to increase with increasing V_{f+} is a result of the strong C_f seen in Figure 7.11 associated with the vortices propagating along the suction surface.

7.4 Summary

In this chapter the effect of dynamic surface motion on the flow development around the aerofoil was explored computationally. Frequencies were chosen with reference to the experimental results. In the cases where performance was improved, the control is based on amplifying naturally occurring frequencies, resulting in earlier vortex formation. It was found that after a few actuation cycles a K-H instability was amplified and the shear layer proceeded to roll-up much earlier upstream than it did naturally. LCS were formed in the shear layer, which enhanced momentum transfer from the outer flow. As these were formed and propagated downstream, gradually this momentum transfer impacts more and more

directly to the near-wall fluid. At such a time as these vortices were in direct contact with the near-wall region the flow became reasonably repeatable. From the phase-averaged vorticity it was found that the generation and shedding of the LCS was locked-on to the actuation frequency and the vortices were shed along the aerofoil surface. Time-averaged plots of TKE and Reynolds stresses revealed that this resulted in the region of high momentum exchange occurring in the near-wall region, thus re-energising the boundary layer. The overall result was an almost complete elimination of the separated regions and a significant improvement in aerodynamic performance. The surface motion can therefore be thought of as a catalyser that reinforces the naturally occurring flow structures, found in Section 6.6.

Chapter 8

Conclusions and Recommendations for Future Work

This chapter will summarise the work carried out throughout this thesis and highlight the main findings chapter by chapter. Recommendations for furthering the research follow.

8.1 Summary

This thesis began by introducing the low Re flight regime and its relevance to practical aeronautical engineering applications, notably small UAV and MAV (**Chapter 1**). It was stated that a laminar boundary layer on the aerofoil's suction surface can easily separate and, as a result, conventional aerofoils suffer significant performance losses at low Re_c compared with higher Re_c . The literature review (**Chapter 2**) revealed that the onset of separation was dependent on both Re_c and α . Also, downstream of the separation point, the separated shear layer can undergo transition to turbulence. Depending on the flow conditions, this can be followed by reattachment or the shear layer could remain separated at the trailing edge. This complex relationship between Re_c , α and separation and transition points leads to various characteristic flow modes developing in the low Re flight regime – each of which is associated with the creation of quite different aerodynamic forces. However, research into active flow control methods has proved that those based on periodic forcing are an efficient way of

accomplishing high levels of control authority and achieving significant performance improvements.

With this in mind, the objective of this thesis was to investigate a novel means of active flow control for aerofoils at low Re – namely, the periodic morphing of the suction surface. This was achieved practically by incorporating MFC actuator patches into an experimental aerofoil model, based on NACA 4415. The extent to which the experimental model could achieve periodic deformations was investigated by laser and photogrammetric measurements (**Chapter 3**). Having ascertained what the model was capable of, a number of frequencies were investigated experimentally at a range of angles, α . These experiments were carried out in the closed-loop wind tunnel at the Temasek Laboratories, National University of Singapore, with the use of PIV, hotwire and a force balance (**Chapters 4-5**).

Although the experiments enabled a wide range of parameters to be investigated, the relatively low spatial and temporal resolution of the measurement devices meant in-depth insight into the flow physics was not possible. Therefore, fully resolved, three-dimensional Direct Numerical Simulations were carried out to complement the experiments. Static simulations were validated against the corresponding experimental data and help explain the flow development observed in the experiments (**Chapter 6**). Then three frequencies of interest, established from the experimental results, were simulated to shed light on the control mechanism involved (**Chapter 7**).

8.2 Main Findings

The research comprised two main areas: (1) the performance of a NACA 4415 aerofoil at low Re_c and (2) the investigation of periodic surface morphing to control the flow around the aerofoil and improve performance. In the following, the major findings from this research regarding those two areas will be highlighted.

8.2.1 Baseline Flow

It was found that the aerofoil model experienced flow separation without reattachment throughout the range of α tested ($-6^\circ \leq \alpha \leq 16^\circ$). This resulted in low C_L and high C_D at all angles and therefore a poor L/D ratio. It was found that at 0° the boundary layer separated to form a laminar separated shear layer. Further downstream a $f^{-5/3}$ gradient in the frequency spectra was found. At the same location a reverse-flow region in the velocity contours was also observed which, with reference to classical LSB studies [31, 3], confirms that the shear layer experienced transition at around $x/c = 0.8$.

The computations, which were shown to be well resolved when compared with previous studies, agreed well with this. It was found that pressure recovery does indeed take place, as expected, in the turbulent portion of the separated region. However, this occurred so close to the trailing edge that it was insufficient to enable reattachment.

The spectral data obtained from the hotwire shows a baseline flow development for $\alpha = 0^\circ$ that includes a Strouhal number based on shear layer frequency of 4.8, and its harmonics, at $x/c = 0.8$ and 0.9 . The wake shedding was found to have a Strouhal number of 2.4 which is a subharmonic of the shear layer frequency. Again, the computations agreed well with this.

8.2.2 Control Mechanism

Actuation frequencies of 10 Hz, 40 Hz and 70 Hz were tested experimentally. At 10 Hz, little change was observed in the flow fields and not much improvement in the aerodynamic force coefficients. On the other hand, both $V_f = 40$ Hz and 70 Hz were capable of reducing separation and consequently, of significantly improving performance. The ability to improve the performance was rooted in the ‘lock-on’ of the natural shear layer and shedding frequency to the surface motion. When $V_f = 10$ Hz, the naturally occurring characteristic frequencies were still present in the flow. In contrast, at 40 Hz and 70 Hz both the wake and shear layer development were dominated by the actuation frequency, suggesting that the flow had locked-in the surface motion.

Through simulation of the actuation frequency that did not work (10 Hz) and a frequency

that caused a large change in the flow field (70 Hz), insight into the fundamental control mechanism was achieved. These were non-dimensionalised to $V_{f+} = 0.4$ and 2 respectively. It was found that at $V_{f+} = 2$ an early roll-up of the shear layer occurred as a result of the Kelvin-Helmholtz instability and spanwise coherent structures were produced. Phase-averaged plots revealed that, once the flow had developed, these were locked to the aerofoil motion and shed along the aerofoil surface during an actuation cycle. The presence of such structures near the aerofoil surface increased mixing in the near-wall flow and therefore re-energised the boundary layer. As a result, flow separation was suppressed and performance improved. Notably, the pressure drag was improved by more than 20% without substantially increasing the contribution of friction drag — which would have occurred if the boundary layer had been tripped through a passive mechanism.

An actuation frequency equal to the shear layer frequency (beyond the capability of the physical model) was investigated to see if further performance improvements were theoretically possible. The results after 25 cycles were more-or-less the same with the $V_{f+} = 2$ case. It should be remembered however that 25 cycles would correspond to a fraction of a second in the experimental dimensions and further flow development may occur if the simulation was continued.

8.3 Recommendations for Future Work

This thesis has proved that periodic surface morphing is an effective method to improve aerofoil performance at low Re_c . During the process of performing the research, the evolving understanding of what was required brought some possible improvements to light. Owing the broad, multidisciplinary (and international) nature of this project, it was not possible to explore all of these improvements during the time available. However, they are listed here as recommendations for future work on this project:

- An experimental model which spanned the full height of the wind tunnel could improve experimental results and, owing the 3-dimensional tip effects, improve the comparison with the computations. Removing the necessity for the splitter plate would also give

greater access to the region of flow around the aerofoil. Furthermore, it would allow for the camera to be mounted closer to the light sheet in the PIV experiments, enabling much greater spatial resolution. The benefit of having more detailed experiments is the physics at many more parameters, such as α , Re_c and V_f , can be examined much quicker than with a DNS. This could enhance the understanding of the relationship between the natural frequencies and effective actuation frequencies.

- By modelling the structural performance of the morphing aerofoil using a Finite Element solver, the choice of suction surface material and position/number of actuators could be optimised. This could result in improved efficiency of the control technique.
- By sensing the shedding frequency, a feedback loop could be employed and the surface's actuation frequency adjusted to match it. This way, the current open-loop control strategy could be turned into a closed-loop strategy. This would be useful when the aerofoil performs complex motion such as pitching or plunging.

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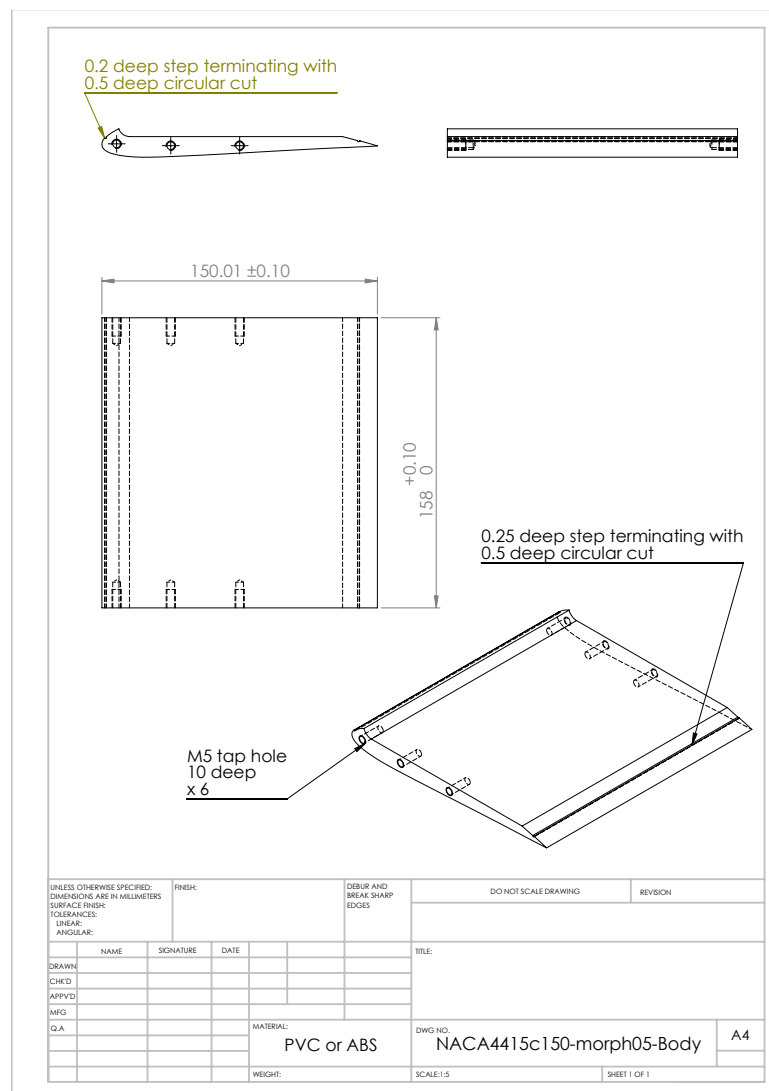
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Appendix A

Appendix

A.1 Technical drawings of the aerofoil



A.2 Photogrammetry Procedure

To convert data from the CCD sensor to meaningful 3D information, the data must be mapped from the sensors pixel coordinates to the coordinate system of the real world. There are two real world coordinate systems involved; one is with reference to the camera, $\mathbf{X}_c$, and the other is with reference to the physical aerofoil model, $\mathbf{X}$. It is the latter of the two that is of interest. As such, it is $\mathbf{w}(u, v) \mapsto \mathbf{X}(X, Y, Z)$ that is ultimately desired, where $\mathbf{w}$ is the sensors coordinate system in pixel numbers u and v , and $\mathbf{X}$ is the Cartesian coordinate system with reference to the physical aerofoil model. The coordinate system is defined in Figure 3.14.

The 2D sensor data must first be mapped onto the 3D Cartesian camera coordinate system, $\mathbf{X}_c$, via an intermediate mapping to $\mathbf{x}$, the 2D Cartesian system of the image plane:

$$\mathbf{w}(u, v) \mapsto \mathbf{x}(x, y) \mapsto \mathbf{X}_c(X_c, Y_c, Z_c)$$

Converting from sensor coordinates to image plane coordinates involves accounting for an offset between the origin of the image plane and pixel (0,0) and the distortion of the image caused by the lens. These are accounted for using the equation below where (u_0, v_0) is the offset value, known as the principle point, and k_u and k_v are lens distortion parameters.

$$\begin{aligned} u &= u_0 + k_u x \\ v &= v_0 + k_v y \end{aligned} \tag{A.1}$$

Then mapping from a 2D image plane to a 3D system, referenced to the camera, can be achieved by applying the concept of the pin-hole camera [82]. Using the similarity of triangles, it is found that the image plane and camera-centered coordinates are related via the focal length, f , as in Eq. A.2.

$$\begin{aligned}\frac{x}{f} &= \frac{X_c}{Z_c} \\ \frac{y}{f} &= \frac{Y_c}{Z_c}\end{aligned}\tag{A.2}$$

Rearranging Eq. A.2 and substituting into Eq. A.1 reveals $\mathbf{w}(u, v) \mapsto \mathbf{X}_c(X_c, Y_c, Z_c)$ using the following equation:

$$\begin{aligned}u &= u_0 + k_u f \frac{X_c}{Z_c} \\ v &= v_0 + k_v f \frac{Y_c}{Z_c}\end{aligned}\tag{A.3}$$

At this point it is convenient to convert from Cartesian to homogeneous coordinates, which involves introducing an extra dimension and an arbitrary scaling. This enables the mapping to be implemented as successive matrix operations. The homogeneous coordinates are given below:

Camera-Centered Coordinates: $\mathbf{X}_c = (X_c, Y_c, Z_c) \longrightarrow \tilde{\mathbf{X}}_c = (X_c, Y_c, Z_c, 1)$

Image Plane Coordinates: $\mathbf{x} = (x, y) \longrightarrow \tilde{\mathbf{x}} = (sx, sy, s)$

Pixel Coordinates: $\mathbf{w} = (u, v) \longrightarrow \tilde{\mathbf{w}} = (su, sv, s)$

Expressing the mapping in Eq. A.3 in matrix form, and homogeneous coordinates provides the *Perspective Projection matrix*, P_p , and the *CCD Calibration matrix*, P_c :

$$\tilde{\mathbf{w}} = \begin{bmatrix} su \\ sv \\ s \end{bmatrix} = \underbrace{\begin{bmatrix} k_u & 0 & u_0 \\ 0 & k_v & v_0 \\ 0 & 0 & 1 \end{bmatrix}}_{\mathbf{P}_c} \underbrace{\begin{bmatrix} f & 0 & 0 & 0 \\ 0 & f & 0 & 0 \\ 0 & 0 & 1 & 0 \end{bmatrix}}_{\mathbf{P}_p} \begin{bmatrix} X_c \\ Y_c \\ Z_c \\ 1 \end{bmatrix} \quad (\text{A.4})$$

$$= \underbrace{\begin{bmatrix} \alpha_u & 0 & u_0 \\ 0 & \alpha_v & v_0 \\ 0 & 0 & 1 \end{bmatrix}}_{\mathbf{K}} \begin{bmatrix} X_c \\ Y_c \\ Z_c \\ 1 \end{bmatrix} \quad (\text{A.5})$$

Multiplying P_c and P_p provides an upper triangular matrix known as the *Camera Calibration matrix*, $\mathbf{K}$. This matrix contains all the internal camera parameters, or the internal orientation of the camera [82].

Going from camera-centered coordinates to the desired coordinates of the actual model is achieved by a rigid-body rotation, $\mathbf{R}$, and translation, $\mathbf{T}$:

$$\begin{bmatrix} X_c \\ Y_c \\ Z_c \end{bmatrix} = \begin{bmatrix} r_{11} & r_{12} & r_{13} \\ r_{21} & r_{22} & r_{23} \\ r_{31} & r_{32} & r_{33} \end{bmatrix} \begin{bmatrix} X \\ Y \\ Z \end{bmatrix} + \begin{bmatrix} T_x \\ T_y \\ T_z \end{bmatrix} \quad (\text{A.6})$$

Therefore, the entire process of mapping CCD sensor coordinates to the aerofoil's coordinates is given by the following expression:

$$\tilde{\mathbf{w}} = \mathbf{K}[\mathbf{R}|\mathbf{T}]\tilde{\mathbf{X}} \quad (\text{A.7})$$

$$= \mathbf{P}\mathbf{X} \quad (\text{A.8})$$

P is the 3×4 *Camera Projection matrix* and contains the internal orientation of the camera via $\mathbf{K}$ and the exterior orientation via $[\mathbf{R}|\mathbf{T}]$. Determining the elements of the Camera

Projection matrix is the goal of camera calibration. Prior to calibration, $\tilde{\mathbf{w}}$ can be considered to relate to $\tilde{\mathbf{X}}$ via Eq. A.8 where P is a matrix of 12 unknown elements. This provides the system of two linear equations with 12 unknowns in Eq. A.9.

$$\begin{aligned} u &= \frac{su}{s} = \frac{P_{21}X + P_{22}Y + P_{23}Z + P_{24}}{P_{31}X + P_{32}Y + P_{33}Z + P_{34}} \\ v &= \frac{sv}{s} = \frac{P_{11}X + P_{12}Y + P_{13}Z + P_{14}}{P_{31}X + P_{32}Y + P_{33}Z + P_{34}} \end{aligned} \quad (\text{A.9})$$

The process of calibration uses a grid of points, shown in figure A.1, where X_i , Y_i and Z_i , the real world coordinates of the points, are known (subscript i denotes a particular point). Each point captured in an image provides a system of equations of the form of Eq. A.9, expect that u and v are measured and X , Y and Z are known. The elements in P can be solved for and the camera can be calibrated.

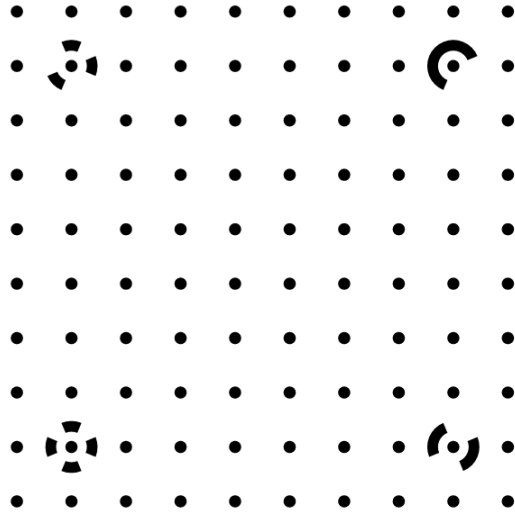


Figure A.1: Camera calibration grid

Considering the fact that $P = \mathbf{K}[R|\mathbf{T}]$, where $\mathbf{K}$ is an upper triangular matrix and R is an orthogonal 3×3 rotation matrix, the QR decomposition of the right hand 3×3 submatrix of P provides $\mathbf{K}$ and R . Hence, this enables the separation of the internal and external orientation of the camera, which is important since the camera will be moved during the experiments.

For more details see Multiple View Geometry in Computer Vision by Hartley & Zisserman[82].

A.3 Further insight on the C_L curve

The lift curve shown in Figure A.2a reveals C_L to increase linearly with α at a gradient of 0.09 (1.6π) for $\alpha < 6^\circ$ and then a more gradual 0.06 (1.1π) as stall is approached. Figure A.2b shows data points acquired from 6 different realizations of the experiment at pre-stall angles – some of which were taken over a year apart – to show the repeatability of the curve in this α region. The results are found to be extremely repeatable, despite the time between data capture and multiple re-mountings of the model. This gives confidence in these results.

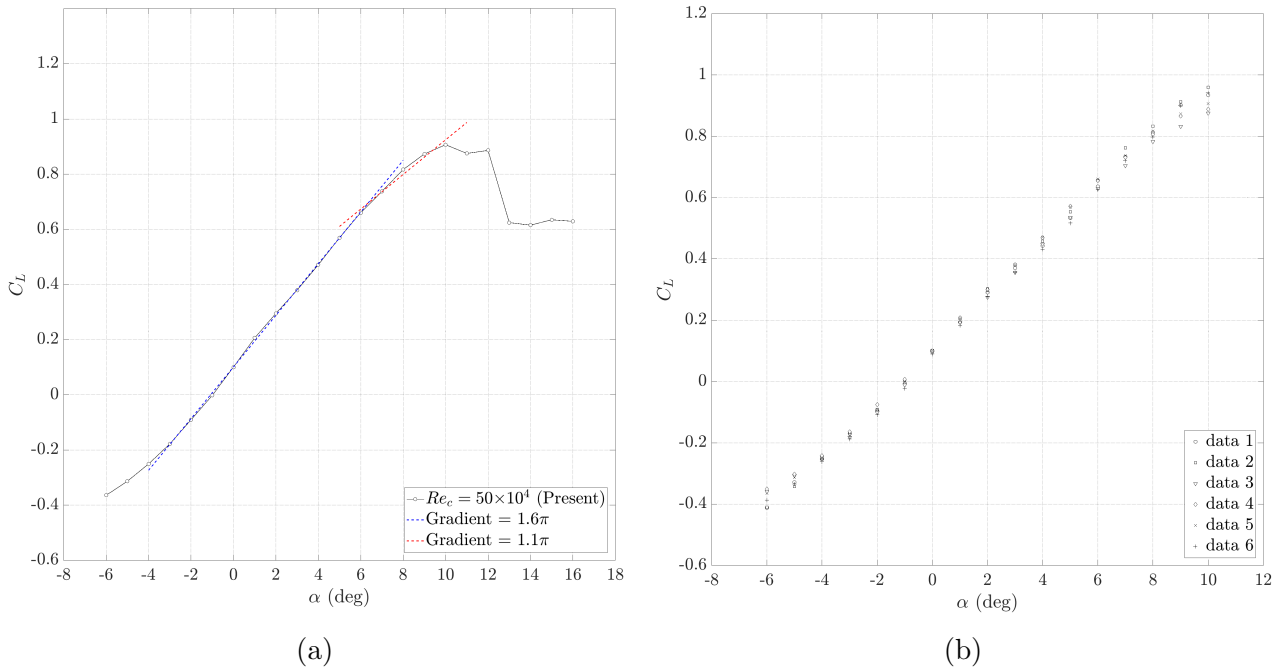


Figure A.2: Lift Coefficients against α obtained from non-actuated aerofoil: a) Lift-curve slopes; b) Repeatability of C_L for $\alpha \leq 10^\circ$.

As well as being extremely repeatable, the curve agrees well with the work of Gerakopulos *et al.* [91] who studied the effects of separation at low Re_c (80×10^3 - 200×10^3) on the performance of a NACA 0018 with an aspect ratio of 3. They also found a conventional linear C_L growth with α for $\alpha = 0^\circ$ to 6° despite the presence of separation. This is followed by a more gradual growth up to the stall angle. The appearance of these two distinct regions in the pre-stall lift curve are attributed to the rate of upstream movement of the separation, transition and reattachment locations – a more rapid upstream movement at lower angles resulting in a larger lift curve slope. The slope in the low α region of the current study is

considerably lower than Gerakopulos *et al.* [91] achieved at 80×10^3 . The results here give a lift curve slope of 0.09, while Gerakopulos *et al.* achieved 0.11. Their value was achieved at a different Re_c and with a different aerofoil but this difference can also be attributed to the lower aspect ratio in the current study (3 in [91] and 1 here), which will lower the lift curve slope.

A.4 Description of Dynamic Mode Decomposition

The cross-stream velocity component, v , from the snapshots used to obtain the time-average in section 6.5.3 were arranged into an $M \times N$ snapshot matrix, Ψ , where M is the number of grid points and N is the number of snapshots. The superscript denotes the grid point number and the subscript denotes the snapshot number (i.e. the time instance).

$$\Psi = \begin{Bmatrix} \psi_1^1 & \psi_2^1 & \cdots & \psi_N^1 \\ \psi_1^2 & \psi_2^2 & \cdots & \psi_N^2 \\ \vdots & \vdots & \ddots & \vdots \\ \psi_1^M & \psi_2^M & \cdots & \psi_N^M \end{Bmatrix} = \{\psi_1, \psi_2, \cdots, \psi_N\} \quad (\text{A.10})$$

It is assumed that a dynamic linear propagator, $\mathbf{A}$, maps one realization of the flow, $N - 1$, to the subsequent realization, N . This is represented by

$$\psi_N = \mathbf{A}\psi_{N-1} \quad (\text{A.11})$$

The dynamics of the fluid process described by sequence Ψ are therefore given by the eigen-decomposition of $\mathbf{A}$. However, since $\mathbf{A}$ is not known an approximation is sought, $\mathbf{S}$, whose dynamics are an approximation of the dynamics of $\mathbf{A}$. The objective of DMD is the formation of $\mathbf{S}$ such that the residual, $\mathbf{r}$, is minimised.

The process of computing $\mathbf{S}$ begins by defining two lagged matrices, Ψ_0, Ψ_1 as in Equation A.12.

$$\begin{aligned} \Psi_0 &= \{\psi_1, \psi_2, \cdots, \psi_{N-1}\} \\ \Psi_1 &= \{\psi_2, \psi_3, \cdots, \psi_N\} \end{aligned} \quad (\text{A.12})$$

Following this, Equation A.11 can be expressed in terms of these lagged matrices as

$$\begin{aligned}
\Psi_1 &= \{\psi_2, \psi_3, \dots, \psi_N\} \\
&= \{\mathbf{A}\psi_1, \mathbf{A}\psi_2, \dots, \mathbf{A}\psi_{N-1}\} \\
\Psi_1 &= \mathbf{A}\Psi_0
\end{aligned} \tag{A.13}$$

The Singular Valued Decomposition (SVD) of Ψ_0 (i.e. $\Psi_0 = \mathbf{U}\Sigma\mathbf{V}^*$) is substituted into Equation A.13 so that $\Psi_1 = \mathbf{A}\mathbf{U}\Sigma\mathbf{V}^*$ where $*$ denotes the complex conjugate transpose. Since $\mathbf{V}$ is a unitary matrix (i.e. $\mathbf{V}^* \cdot \mathbf{V} = \mathbf{I}$), pre-multiplying Ψ_1 by $\mathbf{U}^*$ and post-multiplying by $\mathbf{V}$ provides an approximation of $\mathbf{S}$.

$$\begin{aligned}
\mathbf{U}^*\Psi_1\mathbf{V} &= \mathbf{U}^*\mathbf{A}\mathbf{U}\Sigma\mathbf{V}^*\mathbf{V} \\
\mathbf{U}^*\Psi_1\mathbf{V} &= \mathbf{U}^*\mathbf{A}\mathbf{U}\Sigma \\
\mathbf{U}^*\Psi_1\mathbf{V}\Sigma^{-1} &= \mathbf{U}^*\mathbf{A}\mathbf{U} = \tilde{\mathbf{S}}
\end{aligned} \tag{A.14}$$

Matrix $\mathbf{U}$ contains the proper orthogonal (POD) modes of Ψ_0 and therefore $\tilde{\mathbf{S}}$ amounts to an optimal, low-dimensional projection of linear dynamical propagator $\mathbf{A}$ onto those POD modes [108].

A low-order dynamics of $\mathbf{A}$ is obtained by performing an eigen-decomposition of $\tilde{\mathbf{S}} : \tilde{\mathbf{S}}\mathbf{Y} = \mu\mathbf{Y}$, where μ is a square matrix with the eigenvalues of $\tilde{\mathbf{S}}$ on its diagonal and the columns of $\mathbf{Y}$ are the corresponding eigenvectors. Since $\tilde{\mathbf{S}} = \mathbf{U}^*\mathbf{A}\mathbf{U}$, the POD modes can be used to map the dynamics into a higher dimensional space in the following manner

$$\begin{aligned}
\mathbf{U}^*\mathbf{A}\mathbf{U}\mathbf{Y} &= \mu\mathbf{Y} \\
\mathbf{U}\mathbf{U}^*\mathbf{A}\mathbf{U}\mathbf{Y} &= \mathbf{U}\mu\mathbf{Y} \\
\mathbf{A}\mathbf{U}\mathbf{Y} &= \mathbf{U}\mathbf{Y}\mu
\end{aligned} \tag{A.15}$$

where the fact that U is also a unitary matrix was used.

So, finally, the spatial DMD modes are given by $\hat{\phi}_{\mathbf{i}} \approx \mathbf{U}y_i$ and the temporal behaviour is given by the eigenvalues, μ . The hat notation is used to signify that these mode shapes are unscaled. For correct scaling, amplitude coefficients, a_i , of each DMD mode is computed using the optimal amplitudes algorithm of Jovanovic *et al.* [108].

A superposition of all DMD modes, $\hat{\phi}_{\mathbf{i}}$, properly weighted by their associated amplitudes, $\mathbf{a}_{\mathbf{i}}$ and advanced in time according to their temporal growth rate, as in Equation A.16, optimally approximates the time evolution of the entire data sequence given by Ψ_0 .

$$\{\psi_0, \psi_2, \dots, \psi_{N-1}\} = \underbrace{\{\hat{\phi}_1, \hat{\phi}_2, \dots, \hat{\phi}_{N-1}\}}_{\text{spatial}} \underbrace{\begin{pmatrix} a_1 & & & \\ & a_2 & & \\ & & \ddots & \\ & & & a_{N-1} \end{pmatrix}}_{\text{temporal}} \underbrace{\begin{pmatrix} 1 & \mu_1 & \cdots & \mu_1^{N-1} \\ 1 & \mu_2 & \cdots & \mu_2^{N-1} \\ \vdots & \vdots & \ddots & \vdots \\ 1 & \mu_r & \cdots & \mu_{N-1}^{N-1} \end{pmatrix}}_{\text{temporal}} \quad (\text{A.16})$$

Furthermore, as can be seen in Equation A.16, DMD has enabled the temporal and spatial information to be separated.

A.5 Instantaneous Flow fields for $V_{f+} = 2$

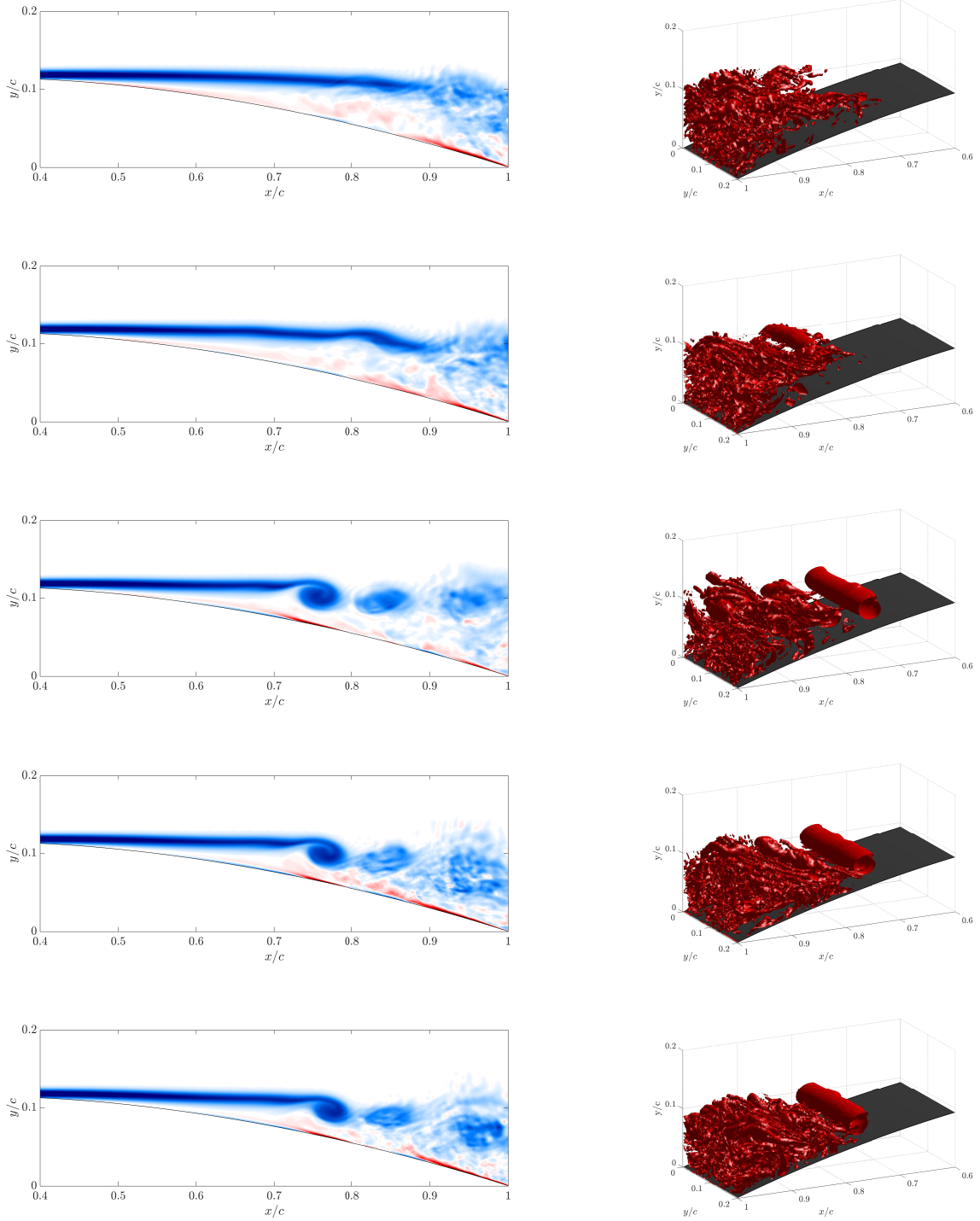


Figure A.3: Iso-contours of instantaneous, spanwise-averaged Ω_z (left) and instantaneous Q-Criterion (right) at $\phi = 0^\circ$ for cycles 1 to 5 (top to bottom) when $V_{f+} = 2$.

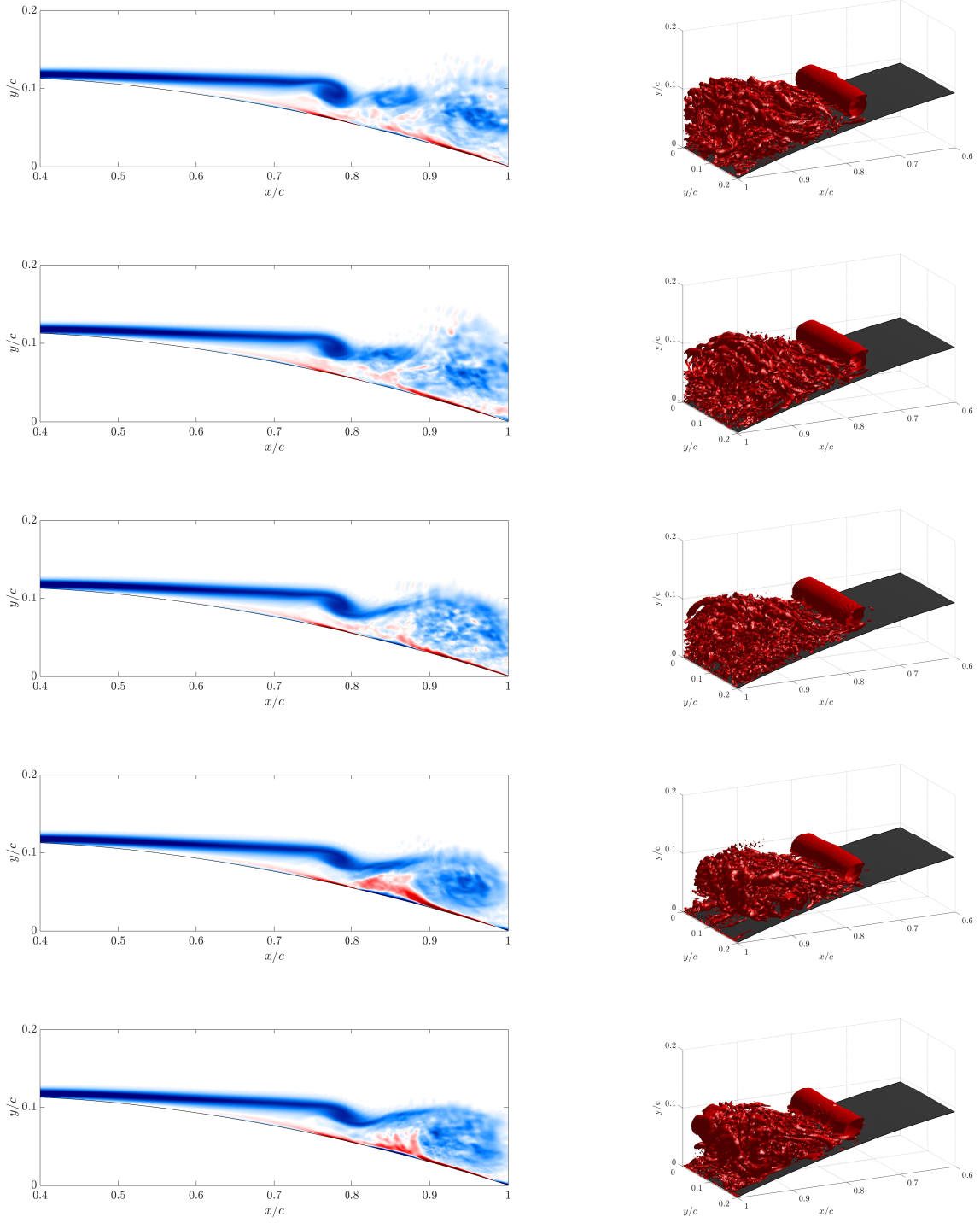


Figure A.4: Iso-contours of instantaneous, spanwise-averaged Ω_z (left) and instantaneous Q-Criterion (right) at $\phi = 0^\circ$ for cycles 6 to 10 (top to bottom) when $V_{f+} = 2$.

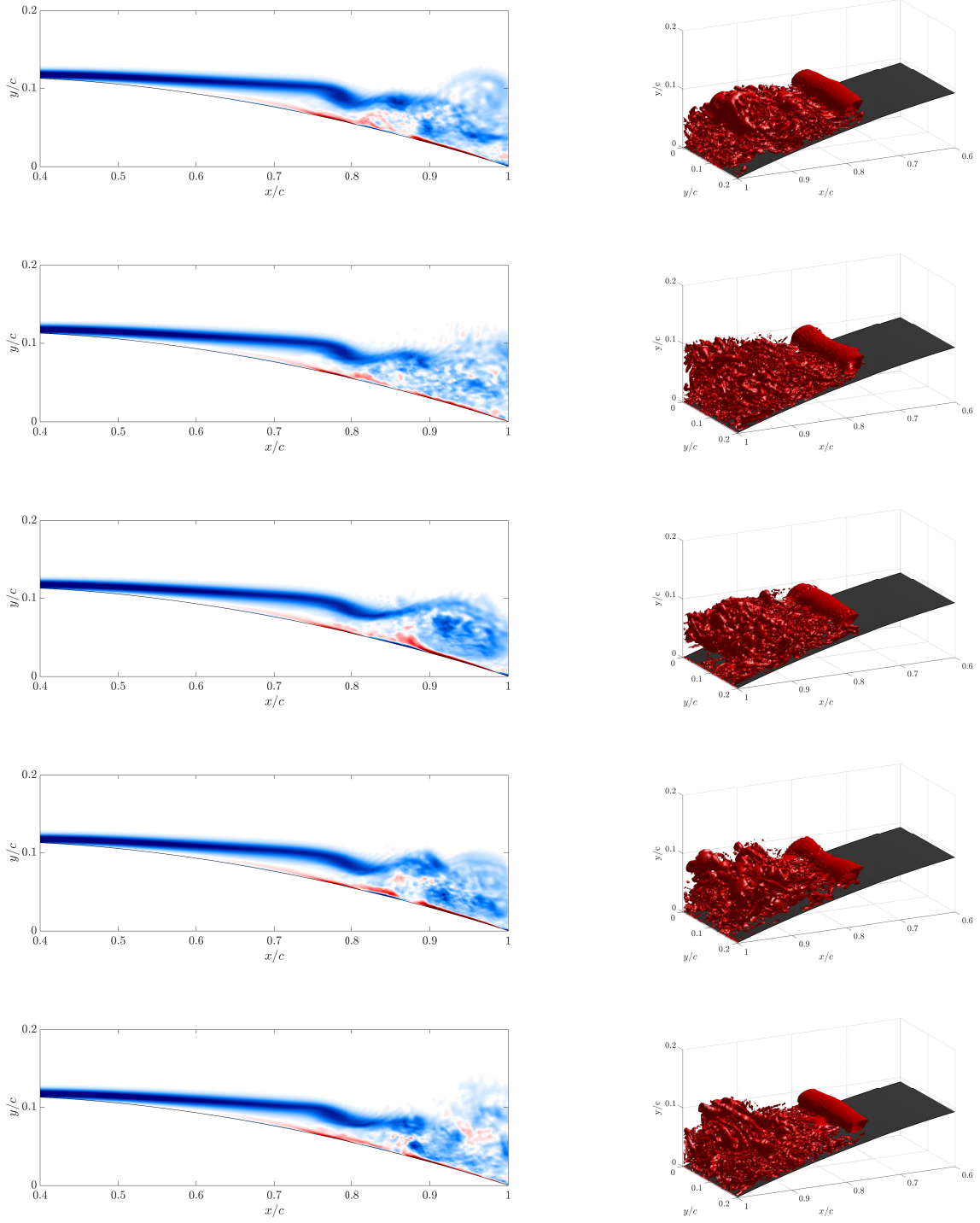


Figure A.5: Iso-contours of instantaneous, spanwise-averaged Ω_z (left) and instantaneous Q-Criterion (right) at $\phi = 0^\circ$ for cycles 11 to 15 (top to bottom) when $V_{f+} = 2$.

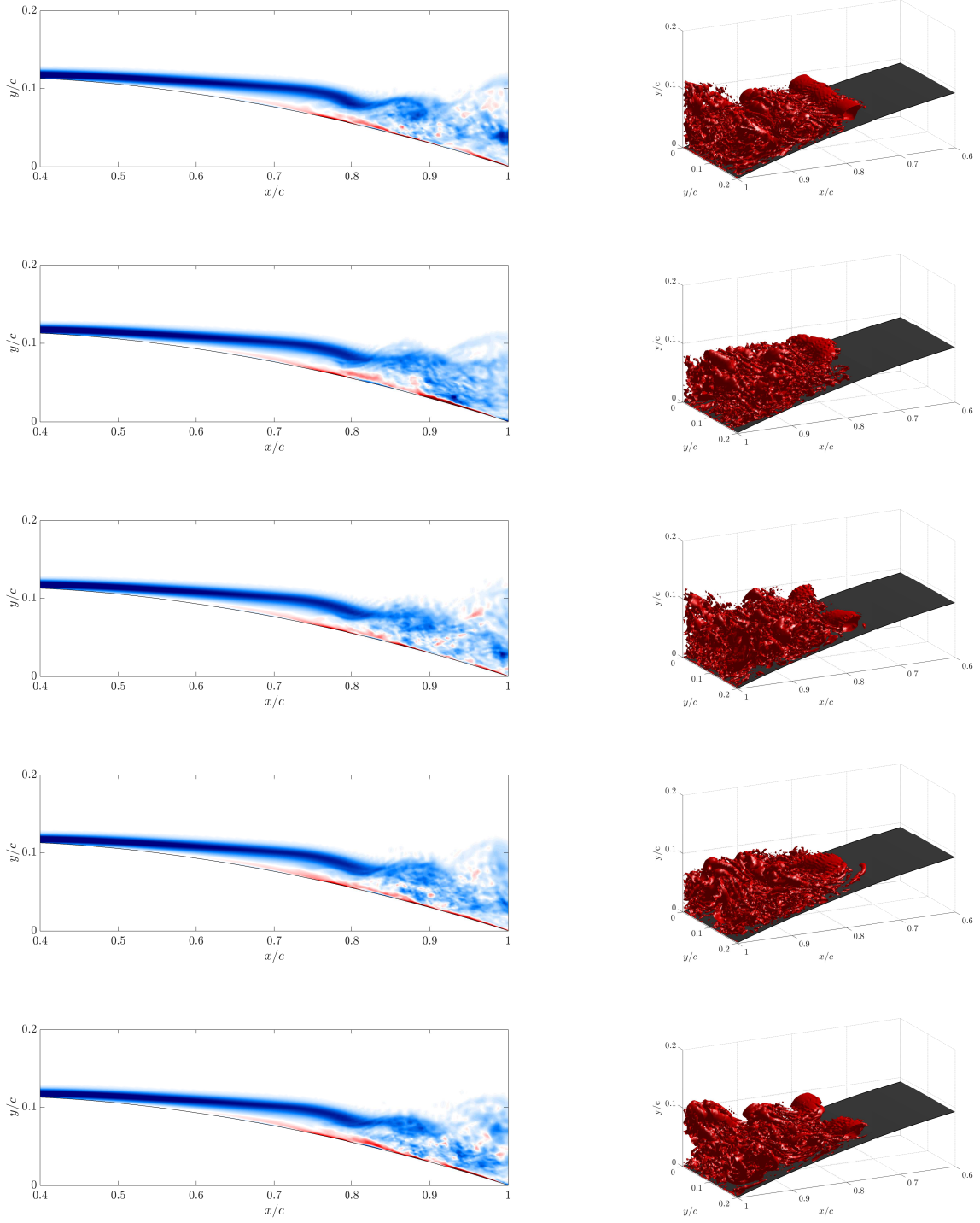


Figure A.6: Iso-contours of instantaneous, spanwise-averaged Ω_z (left) and instantaneous Q-Criterion (right) at $\phi = 0^\circ$ for cycles 16 to 20 (top to bottom) when $V_{f+} = 2$.

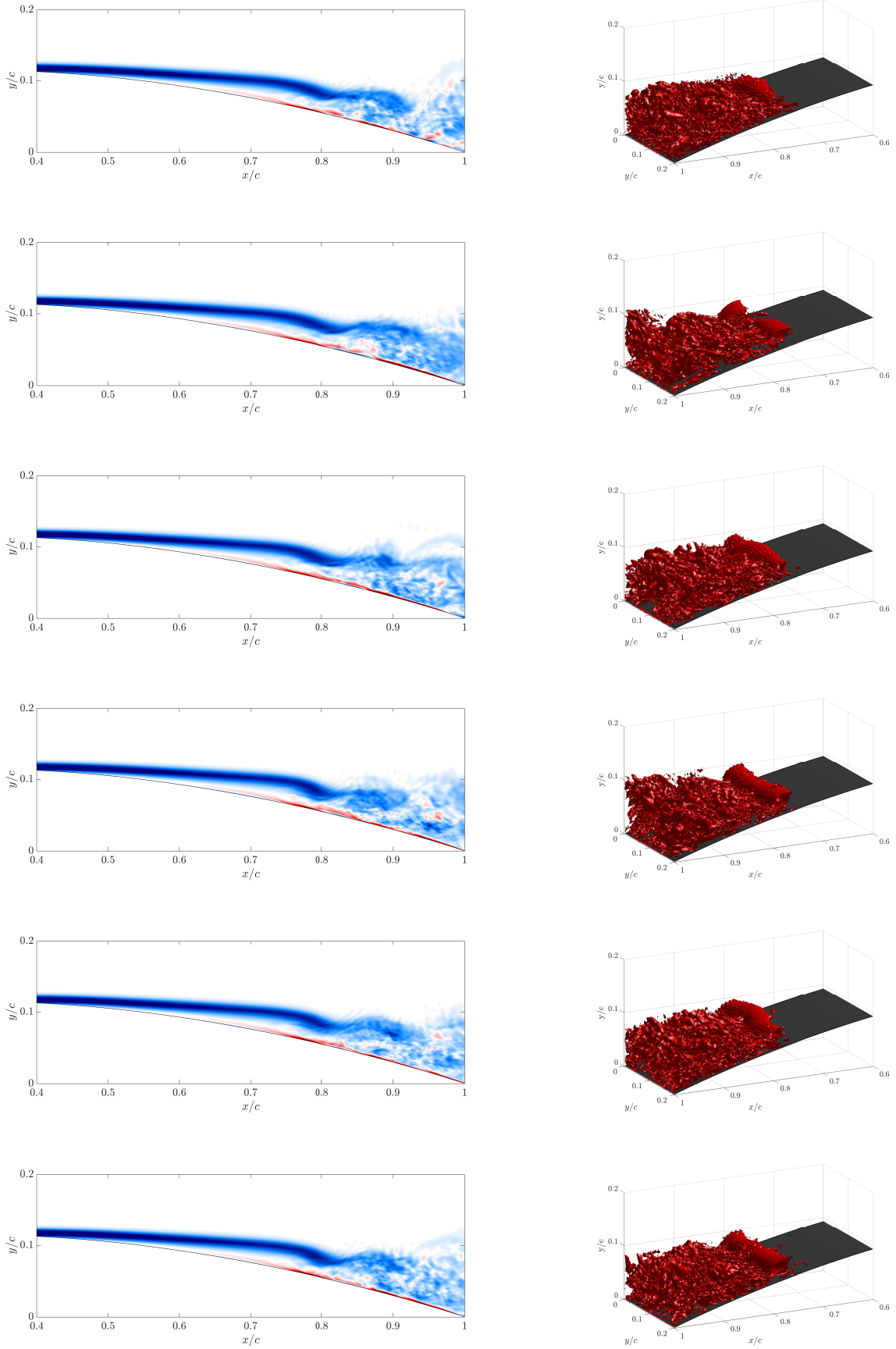


Figure A.7: Iso-contours of instantaneous, spanwise-averaged Ω_z (left) and instantaneous Q-Criterion (right) at $\phi = 0^\circ$ for cycles 21 to 26 (top to bottom) when $V_{F+} = 2$.

A.6 Instantaneous Flow fields for $V_{f+} = 5$

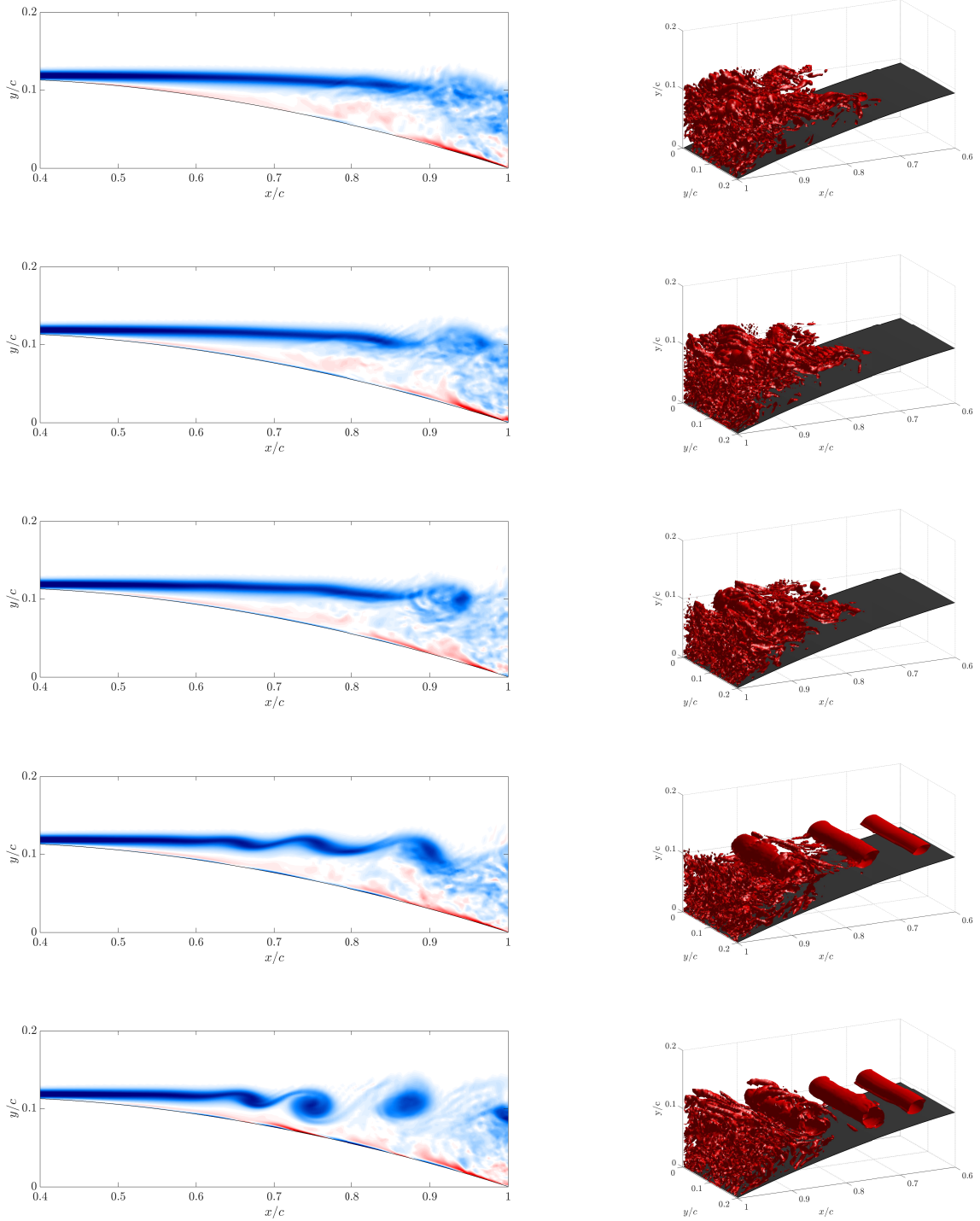


Figure A.8: Iso-contours of instantaneous, spanwise-averaged Ω_z (left) and instantaneous Q-Criterion (right) at $\phi = 0^\circ$ for cycles 1 to 5 (top to bottom) when $V_{f+} = 5$.

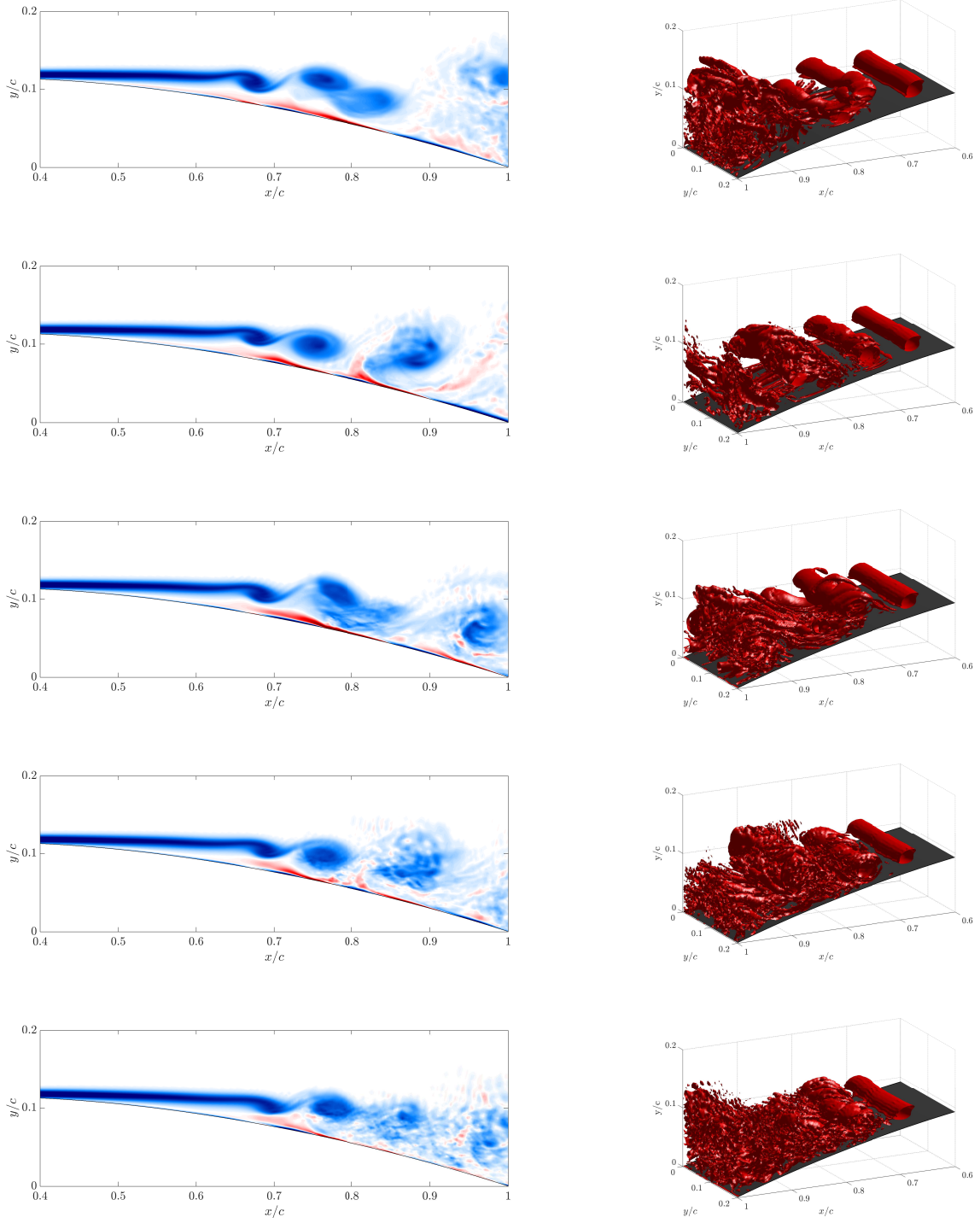


Figure A.9: Iso-contours of instantaneous, spanwise-averaged Ω_z (left) and instantaneous Q-Criterion (right) at $\phi = 0^\circ$ for cycles 6 to 10 (top to bottom) when $V_{f+} = 5$.

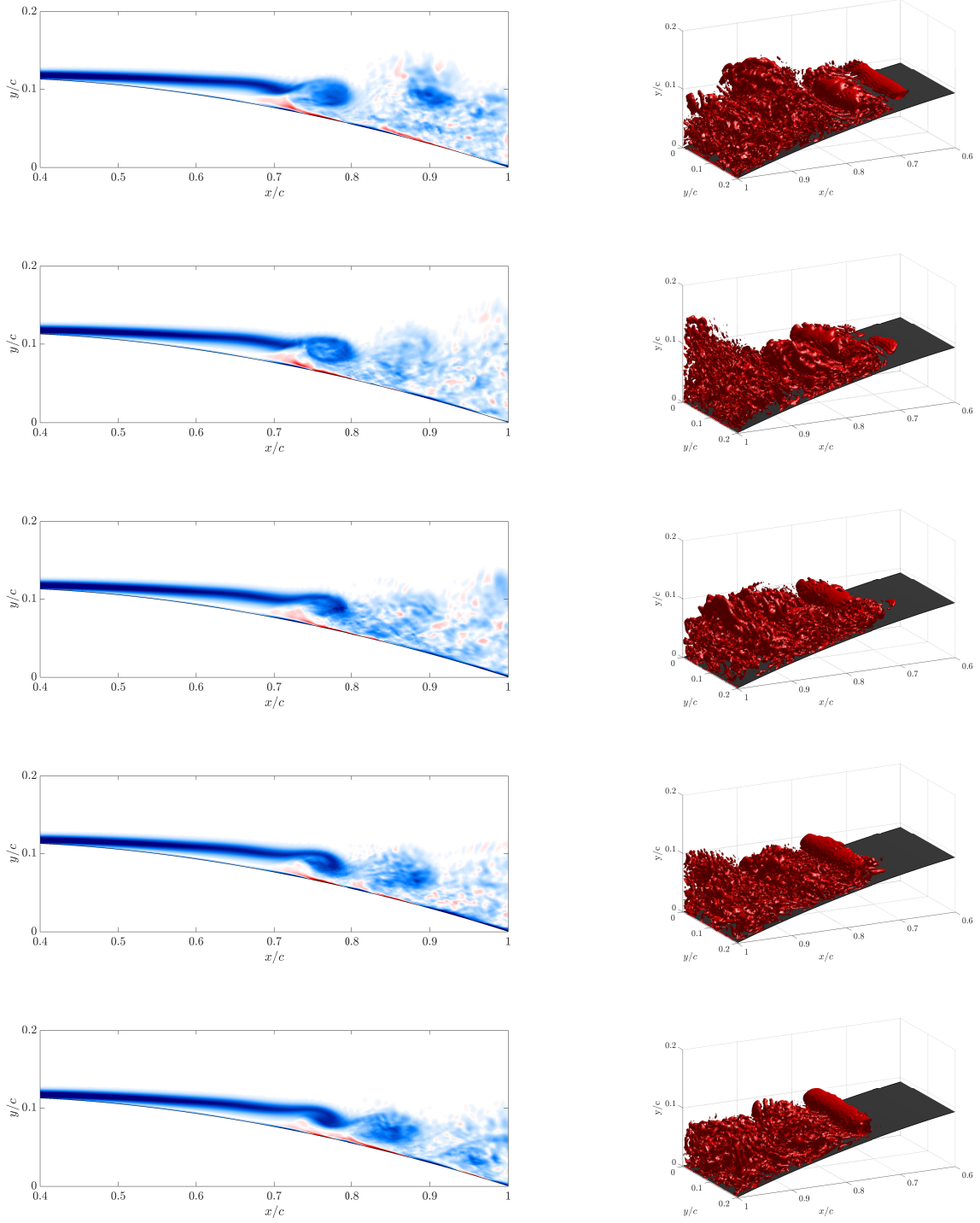


Figure A.10: Iso-contours of instantaneous, spanwise-averaged Ω_z (left) and instantaneous Q-Criterion (right) at $\phi = 0^\circ$ for cycles 11 to 15 (top to bottom) when $V_{f+} = 5$.

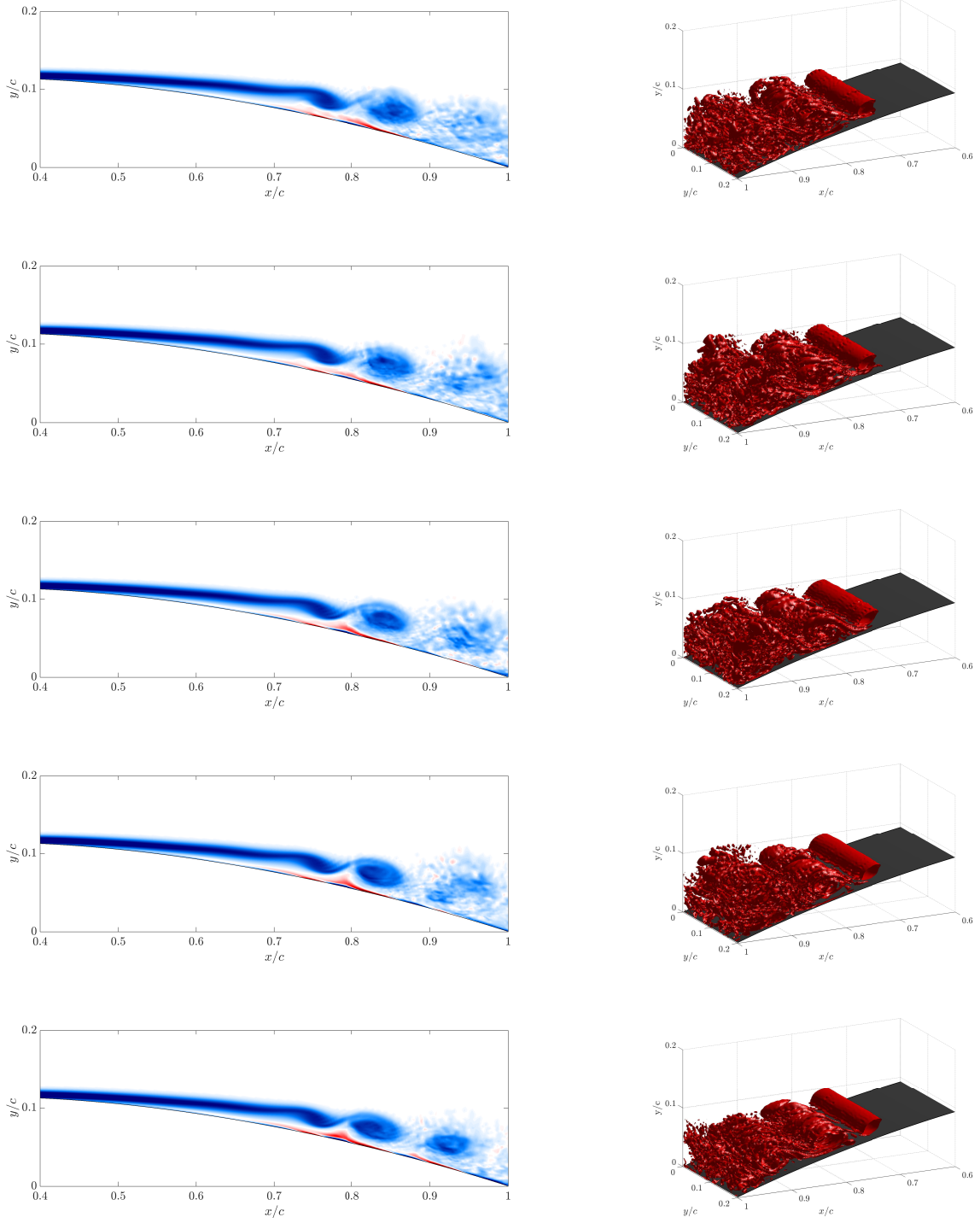


Figure A.11: Iso-contours of instantaneous, spanwise-averaged Ω_z (left) and instantaneous Q-Criterion (right) at $\phi = 0^\circ$ for cycles 16 to 20 (top to bottom) when $V_{f+} = 5$.

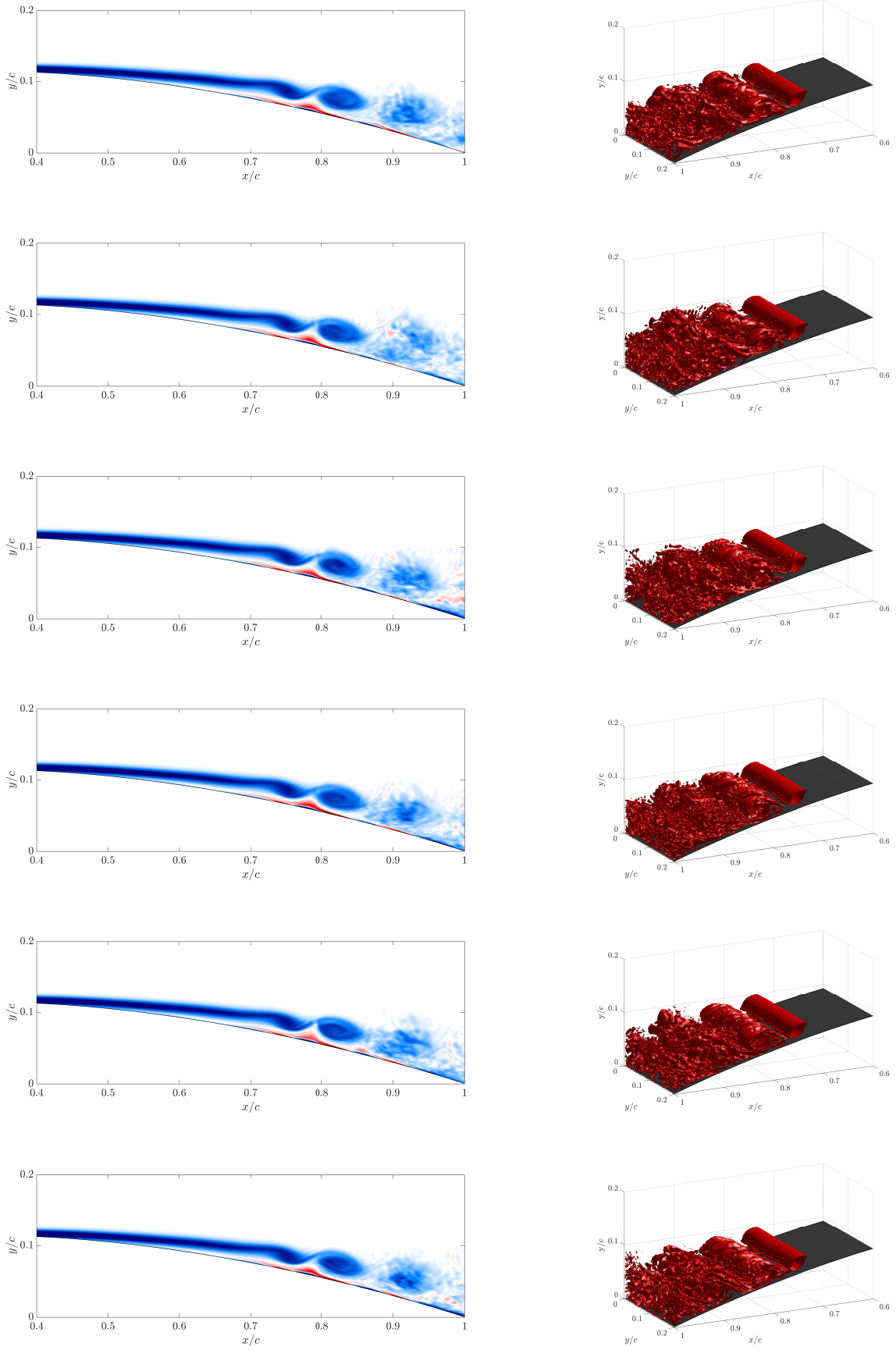


Figure A.12: Iso-contours of instantaneous, spanwise-averaged Ω_z (left) and instantaneous Q-Criterion (right) at $\phi = 0^\circ$ for cycles 21 to 26 (top to bottom) when $V_{f+} = 5$.