Optimisation of Robust Active Flow Control Technologies for Motorsport Applications

by

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Declaration

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Abstract

Active flow control systems have the potential to allow future designs of ground vehicles and aircraft to realise increased operational efficiency through improved optimisation of the flow, leading to decreased fuel use and reduced environmental impact. To achieve this, dynamic actuators are required that can adapt to the changing conditions experienced over low-speed, high-lift aerofoils where separation control can be particularly advantageous. Synthetic Jet Actuators (SJAs) are a form of the technology that shows promise; they are small, low-mass and low-power devices, which means that they can potentially realise the system efficiency a vehicular application of active flow control requires.

The aims of this research were directed to achieving better understanding and robustness of the control authority from SJA systems at Reynolds Numbers close to real-world operations. The characteristics of the control authority from a round-orifice SJA array positioned near the leading-edge position of an NACA0015 aerofoil have been investigated at \( Re = O(10^6) \). Measurements demonstrated how the jet flow imparts a controlling mechanism over the separating boundary layer flow, and hence can be used to improve the overall efficiency of the wing. A series of parametric alterations to the test conditions was made in order to understand the robustness of the control effect. The forcing frequency was decoupled from the dynamic response of the actuators themselves by means of amplitude modulation. The results demonstrated that successful control could be achieved with significantly reduced input power requirements, improving net efficiency. The effectiveness was shown to be largely independent of the frequency when used in this way. Using a counterstreamwise jet orientation to control the same basic separated flow condition was not found to generate significant improvements in operational efficiency. The results suggest an in-depth understanding of the jet flow, and the excitation location in relation to the point where the flow separates is important when designing the actuators. Tests also considered a different flow condition with a stronger adverse pressure gradient, by generating a ground-effect flow over the suction surface. The control authority afforded was diminished in the more adverse flow states. The performance of the actuators was considered, and the system achieved a larger than unity Figure of Merit, indicating the overall benefit of control shows direct relevance for realising practical flow control systems. The results indicate arrays of SJA’s are capable of delivering energy savings when managing this type of flow.
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Publications Related to this Thesis

Conference Papers


These papers focus on the results discussed Chapters 5, 6, and 7.

Journal Papers


This paper used understanding gained from the contents of Chapter 4 on a concurrent investigation of flow control devices by the author(s).
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Nomenclature

a Speed of sound (m/s)
A Area (m²)
c Chord of wing (m)
$C_D$ Coefficient of drag
$C_D'$ Coefficient of force
$C_L$ Coefficient of lift
$C_{Dc}$ Corrected Coefficient of drag
$C_{Lc}$ Corrected Coefficient of lift
$C_P$ Coefficient of pressure
$C_P'$ Canonical Coefficient of pressure
$C_\mu$ Momentum coefficient
d₀ Diameter of SJA orifice (m)
d_c Diameter of cavity (m)
D Drag (N)
f Oscillation frequency (Hz)
F Force (N)
$F^+$ Reduced frequency
h Ride height of wing above moving ground (m)
I Electrical current (A)
K Temperature (Kelvin)
l Length (height) of SJA orifice (m)
L Lift (N)
$L_D$ Non dimensional stroke length
M Moment (Nm)
N Number of samples
$p_d$ Differential pressure, (local static pressure – freestream static pressure), (Pa)
P Power (W)
$R_u$ Velocity Ratio
$R Reynolds number \ (Re = U_\infty \cdot c/v)$
V Voltage (V)
w  Weighting
W  Width (height) of actuator cavity (m)
S  Stokes Number
$S_p$  Distance to pressure distribution suction peak on aerofoil suction surface (m)
St  Strouhal Number
$t$  Time (s)
T  Time period of jet cycle (s)
$u,v,w$  Streamwise, vertical, spanwise velocity (m/s)
$x_{sep}$  Streamwise separation length scale (m)
$x,y,z$  Streamwise, vertical, spanwise coordinate
$\alpha$  Angle of incidence (°)
$\delta$  Boundary layer height (mm)
$\Delta$  Difference between measured quantities
$\varepsilon$  Error
$\varepsilon_{sb,wb}$  Blockage correction
$\eta$  Efficiency
$\theta$  Angle of incidence of the jet orifice (°)
$\lambda$  Spacing (m)
$\mu$  Dynamic viscosity (N s/m$^2$)
$\nu$  Kinematic viscosity (m$^2$/s)
$\rho$  Fluid (air) density (kg/m$^3$)
$\sigma$  Standard deviation
$\sigma_{sc}$  Streamline curvature correction
$\tau$  Shear stress (Pa)
$\phi$  Power spectrum of transducer signal (W)
$\varphi$  Phase angle (°)
$\omega,\zeta$  Vorticity (s$^{-1}$)
$\psi$  Wavelet
Subscripts

\( a \) Array
\( \text{Ave} \) Average
\( \text{Axial} \) Axial direction
\( c \) corrected
\( C \) Carrier
\( d \) Drag
\( f \) fluidic
\( H \) Helmholtz
\( m \) Modulated
\( \text{Max} \) Maximum
\( \text{Min} \) Minimum
\( j \) Jet actuator
\( O \) Orifice
\( pp \) Peak-to-peak
\( p \) pressure
\( \text{Pin} \) Invested power
\( s \) Spatial average
\( \text{sep} \) separated flow region
\( \text{te} \) Trailing edge
\( u \) Uncontrolled
\( w \) at the wall
\( x,y,z \) Streamwise, vertical, spanwise coordinate
\( \varphi \) Phase locked
\( \infty \) Freestream
Superscripts

\begin{itemize}
\item \text{mean}
\item \text{instantaneous}
\item \text{periodic}
\end{itemize}

Abbreviations

\begin{tabular}{ll}
AFM & Aerodynamic Figure of Merit \\
CFD & Computational Fluid Dynamics \\
SJA & Synthetic Jet Actuator \\
TTL & Transistor-Transistor Logic \\
RMS & Root Mean Square \\
RP & Rapid Prototyping \\
ZNMF & Zero Net Mass Flux \\
\end{tabular}
Chapter 1

Introduction

1.1 Background

The generation of aerodynamic lift or downforce (negative lift) on a vehicle is an engineering challenge in the aerospace and motorsport sectors respectively. Both industries share a number of commonalities; they invest significant resource in aerodynamics development, and exchange technology and knowledge, (Zerihan & Zhang 2001). With highly dynamic vehicles, aerodynamic drag is an undesirable result of the motion. One source of this is from the aerofoils used to generate the majority of the lifting forces required to act on aircraft and racecars. Inefficient design or operation of the wing systems generates excessive drag and reduced lift potential. If the efficiency of the aerodynamic forces can be increased, the lift per unit drag, then it creates many benefits. Aircraft concepts can be redesigned in the future to make use of the performance improvements to reduce fuel use. Motorsport vehicles can reach higher performance levels by increasing the size of the friction ellipse (Katz 2006), (Milliken & Milliken 1995), cutting lap-times to achieve greater performance in the sporting context.

Increasing efficiency has become a significant consideration in the recent generation of machinery in both industries. Civil aircraft manufacturers and racing car constructors need to consider how to make their products perform to the same levels, or better, on less energy (fuel) in the future (IMechE 2011), (MIA 2013). Active flow control is one enabling technology to achieve this, and has the potential to revolutionise vehicle performance.

By control of the separated flows over the wings, and other large surfaces, the pressure drag created can be reduced, increasing lift and efficiency. ‘Active’ means that the control of the flow is available when required, thereby reducing the penalties associated with passive techniques. As active methods rely on energy expenditure, the on-demand nature of the
technology can allow systems to be used at only the point of greatest affect on the flow, and thereby potentially enhancing overall efficiency.

Active flow control systems for vehicles have to be small, robust, low-mass, work on minimal energy input, and in a range of flows. Unless they meet these requirements, it serves no benefit to the vehicle; it has to be efficient as a system, as opposed to just in terms of performance. This means the enabling technology to achieve the control authority is an important consideration. As such, active flow control is a multi-disciplinary area of aerodynamic research and represents a challenging and not fully understood topic.

1.2 Application of Flow Control

The pressure drag of a wing increases significantly when the boundary layer flow over the suction surface separates. Figure 1.1 depicts the flowfield around a typical aircraft multi-element aerofoil, showing the extent of the boundary layer flows.

As well as increasing drag, separation of the boundary layer is the limiting factor in attaining maximum lift (Houghton & Brock 1993) for the low-speed conditions of take-off and landing of aircraft, and the speeds reached by contemporary racing cars. Control of the boundary layer towards the leading and trailing edges is imperative in order to realise an ideal pressure distribution over the entire wing surface, and attain the maximal lift. By increasing efficiency of this lift for aircraft take-off and landing sequences, runway lengths, and the amount of fuel required for these parts of the flight can reduce (Van Dam 2002).

Figure 1.1 The boundary layer flow over a multi-element high lift aerofoil (Houghton & Brock 1993).

In the manner in which an aircraft requires a changeable level of performance for different stages of a flight, a race car also has differing aerodynamic requirements during operation.

The aerodynamic balance of the downforce acting on front and rear axles is highly important to the stability of a racing car. Figure 1.2 a) shows analysis by Dominy, Ryan, & Sims-Williams, (2000) to depict how the aerodynamic balance will alter the handling
characteristics, through modelling of a steady-state cornering condition. It shows that the balance changes the handling in a non-linear manner, depending on the layout of the corners on the circuit. Figure 1.2 b) shows experimental results of the downforce balance change of a modern, high downforce race car wind tunnel model.

Figure 1.2 a) From Dominy, Ryan, & Sims-Williams (2000), The effect of downforce balance on optimal steer angle for a range of corner radii on a racetrack; b), data from a race car wind tunnel model (Personal communications, Vehicle Performance Systems May 2014) showing contours of % downforce at the rear axle at various non-dimensionalised front ($h_f$) and rear ride heights ($h_r$), where $h$ = the vertical distance from vehicle underside, at the respective axle, to the ground.

Highlighted is the dynamic pitch range, from static vehicle attitude to the range typically encountered for the full scale vehicle at a speed of ~90m/s on a particular race circuit. The contour map is taken from wind tunnel testing of a 50% scale model. ~3% change is seen in
typical vehicle operating limits. Variation in the balance will impact on the confidence that the driver will have in being able to drive to the limits of adhesion around the race-circuit before losing control of the vehicle. A variation of 1% downforce balance away from what is optimal will be identified as undesirable by the driver of a high downforce, prototype sportscar (Personal communications, Vehicle Performance Systems, May 2014).

A loss in downforce due to flow separating from the wings therefore becomes a limitation to the vehicle performance, and potentially can impact on safety. It is clear therefore that if the aerodynamic balance of the vehicle could be altered actively to maintain neutrality of the handling characteristics, then performance can be increased over micro (corner by corner) and macro (over many laps of an endurance race for example) scales as the conditions of vehicle and circuit evolve.

Mechanical systems to alter the angle of attack of aerodynamic components are well established; but these require heavy, complex, and potentially unreliable actuators of limited response time. Active flow control systems represent a potentially more efficient means, and can therefore be considered a desirable system in both aeronautical and automotive engineering. Regulations in the world’s leading motorsport competitions, (those sanctioned by the Federation International de l’Automobile) for open wheel and closed wheel racing cars, currently do not allow active flow control systems to be used in competition. However, regulations are subject to change on a yearly basis, and with the potential benefits to performance, flow control systems could generate if their performance was unrestricted, gaining further understanding of flow control methods can be seen as a desirable research and development programme.

1.3 Synthetic Jets

The actuator of a flow control system is a primary consideration. This is simply the device that converts one form of energy, electrical for example, into the control authority over the mean flow. One of the more popular options is to use a synthetic jet actuator (SJA). The device operates by synthesising a jet of fluid via a zero net mass-flux (ZNMF) condition. In the far-field, the jet flow is similar in form to a steady jet (Cater & Soria 2002). These jet flows can also be generated from small, low mass devices, as seen in Figure 1.3.
The general layout of an SJA and the ZNMF condition is shown in Figure 1.4. The nature of the jet flow is unsteady; it is formed of periodic suction and blowing phases. However, this is of benefit when considering the manner in which the jet is used. As the collective understanding of fluid mechanics and separated flows has improved, small scale perturbations introduced to a mean flow structure have been understood to be able to leverage inherent instabilities in a separated flow, to impart large scale control authority. The small synthetic jet flow therefore represents a more efficient system when compared to the brute-forcing techniques that can impart the same manner of control authority in other active flow control systems.

Currently however, realising this flow control technique is still an elusive goal for the engineering community in practical environments; this is due to the many areas of the subject that need to be understood to effectively incorporate such systems into real-world situations.
1.4 Aims and Objectives

The aims of the present research are:

To achieve a better understanding of the behaviour of synthetic jets as an active flow control system, by investigating several parameters identified as contributing to the performance and efficiency of a system. These parameters will be considered with a flow control technology of industrial relevance, operating in an environment that shows similarities to real-world flow conditions. By achieving this, an improved understanding of the robustness of the control authority from synthetic jet flow control is expected to be produced.

To achieve this, a series of objectives have been produced.

— To create control authority of a separated flow condition using synthetic jets.
— To investigate how changes to the actuator jet flow affect control authority.
— To further understand control authority by making changes to the separated flow state.
1.5 Outline of the Thesis

The thesis aims to meet the project objectives by documenting the stages to the research;

Chapter 2 consists of a review of the literature covering the relevant topics to this project. The phenomenon of flow separation, separation control, active flow control technologies and recent research into SJA flow control is covered.

Chapter 3 covers the experimental facilities and techniques used to perform the research.

Chapter 4 details the pertinent findings from development of the active flow control actuator. It details the design of the synthetic jet, the manufacturing techniques, and the characterisation methodology for the jet performance. The results of the actuator performance are also discussed.

Chapter 5 presents the results for the characterisation of the flows over the wind tunnel model when the control system is not operated. The results characterise the initial flow condition for which control authority is required.

Chapter 6 presents the results of the actuated flows. The flow phenomena are investigated, and further experiments with passive techniques considered in attempt to further understand the manner of control authority.

Chapter 7 considers variation to the actuation strategy, and the effect on control authority.

Chapter 8 considers a change to the flow condition over the wing. Understanding of the respective unactuated and actuated flows is presented, and comparison to the previous flow conditions is considered.

Chapter 9 concludes the analysis from the previous chapters, highlights implications, and areas of further work.

Appendix A presents findings relating to the actuator performance during the tests conducted in the previous chapters. This highlights results that pertain to the system efficiency, and considers relevance for industry.
Chapter 2

Literature Review

The aim of this chapter is to provide a comprehensive review of the literature in order to identify areas where further research is required. As the topic of ‘active flow control’ is expansive in nature, this review concentrates mainly on recent developments. Where possible, studies that give detailed summaries on areas of preceding and related research that are of interest to the reader have been identified and referred to.

2.1 Flow Separation

Separated flows occur in a wide range of engineering applications, such as the flow over aerofoils at high angles of attack, and the other primary structures of air and ground vehicles. Separation generally has an adverse affect on performance; it reduces lift, increases drag, and also creates secondary effects such as undesirable structural loads and turbulent wake flows. Active flow control has the potential to address the problem, increasing vehicular performance and efficiency.

Flow separation is the process whereby the boundary layer that forms over a solid body moving through a fluid, reaches a condition where attachment cannot be sustained. This can be defined as when the shear stress ($\tau_w$) in the viscous layer at the solid surface disappears. For steady, two-dimensional separation on a stationary wall, this can be expressed as a function of the near wall velocity gradient ($du/dy$), such that at separation:

$$\tau_w = \mu \left( \frac{du}{dy} \right)_w = 0$$

(2.1)

Where $\mu$ is the dynamic viscosity of the fluid. The deceleration of the fluid, and hence the separation, is generally a consequence of an adverse pressure gradient being applied to the flow. This increase in pressure is due to changes of surface curvature, or the angle of the
surface to the mean flow direction, when considering the flow over an aerofoil profile or similar surface. Downstream of the point of separation, an inflection in the streamwise velocity profile occurs, and a bounded area of recirculating flow, a ‘separation bubble’ forms below the boundary layer fluid. When surface geometry changes occur downstream of the bubble, such as an increasing curvature, or a cessation in the form of a trailing edge, then the boundary layer never reattaches to a surface. It proceeds to mix with the bubble and form a turbulent wake. This is the flow behaviour which characterises a quasi-two dimensional flow over an aerofoil, where a characteristic trailing edge separation occurs. With more complex three-dimensional flows, the assumption of a vanishing shear stress and the point of separation are not so closely coupled (Houghton & Brock 1993).

Separation limits the performance envelope for an aerodynamic body; it increases pressure drag, resulting in decreased lift. Boundary layer control has therefore attracted considerable research interest. Increased maximum lift (or downforce) and efficient lift (maximum for minimum drag), can be enhanced by active flow control systems. Such industrial problems are a key driver for research activities.

### 2.2 Flow Separation Control

Historically, the problem of separating flows on aerofoils has been addressed with methods other than those that are considered active flow control. One situation where boundary layer control is imperative is on the high-lift systems of civil transport aircraft. In this configuration, deployable additional elements to the wing are used to enhance plan area, creating sufficient low-speed lift for takeoff and landing flight stages. Figure 2.1 depicts the role of the slat and flaps in lift enhancement at high angles of attack.

The performance envelope of these devices can be limited by flow separation from the suction surface. The multi-element design of the devices performs a well established, complex, boundary layer control mechanism (Smith 1975) that can maintain flow attachment over the highly cambered overall wing profile. However, such high-lift devices, although effective, have disadvantages of being heavy, complex and costly to develop for commercial civil transport aircraft. Other methods of flow control have therefore been looked to in recent times (Anders, Sellers. & Washburn 2004).
A means of boundary layer control that does not rely on multi-element designs is the vortex generator, which is a form of passive flow control. They are designed to generate coherent structures within the flow that are on the physical scale of the boundary layer (Brunet, Dandois, & Verbeke 2013). The streamwise vorticity acts to redistribute the momentum in the near-wall flows, such that the higher velocity fluid in the upper region of the boundary layer or freestream is mixed with that lower down, redistributing the momentum, increasing the shear-stress and delaying separation. With vortex generators being simple, discrete devices of small-scale, they are robust and lightweight, which is highly advantageous for certain applications. However, by always being in the flow, they create a parasitic drag even when not required. To remove this problem of a persistent inefficiency, active methods have been much sought after, to give an ‘on-demand’ effect.

Active methods of boundary layer control can be via steady removal of the low momentum flow (suction) (Gad-El-Hak 2000), or the addition of higher momentum fluid (blowing) (Greenblatt & Wygnanski 2000). Both methods are established, effective techniques to control the boundary layer. They are used regularly in automotive wind-tunnels to create consistent flow conditions when the boundary layer needs to be removed. Examples of use in military aircraft however have proved such system are problematic to integrate into vehicles; the energy and mass requirements to incorporate a suction or blowing system into the wings means overall efficiency improvements cannot easily be reached.
2.3 Periodic Excitation

Momentum redistribution in the boundary layer can be created by active control systems. They rely on the intermittent addition of coherent structures (as opposed to being a steady technique) and have been shown to be effective for separation control. By having a system that is available on-demand and is energy efficient, the limiting parameters of the traditional suction and blowing techniques and passive control systems can be removed. Such systems have the potential to revolutionise the transport industry where aerodynamic efficiency and performance are pertinent issues. A detailed description of the concept of periodic excitation is covered by Greenblatt & Wygnanski (2000).

The key concept to periodic excitation is the generation of large coherent structures in the flow. These interact with the periodic motions of the unperturbed flow that are a characteristic of a flow that is on the verge of, or in a separated state. A visual depiction of this is shown in Figure 2.2.

![Three dimensional Large Eddy Simulation of a separated flow. The coherent structures are identified by iso-surfaces of pressure, superimposed with contours of vorticity (ω) in one spanwise plane, from Franck (2009).](image)

This gives rise to a key descriptor of the periodic excitation; the dimensionless frequency,

\[ F^+ = f x_{sep} / U_\infty. \]  

(2.2)

Where \( f \) is the controlling excitation frequency, \( U_\infty \) the freestream velocity, and \( x_{sep} \) a length-scale to the separated flow. When the streamwise length-scale of the separated flow is used for \( x_{sep} \), it implies that for \( F^+ = 1 \), the frequency of the periodic excitation is inversely proportional to the convective velocity of a fluid particle moving across the separated flow region. The two flow time-scales are commensurate. Many studies have concentrated the study of flow control at, and close to this frequency. The small-scale perturbations elicit a larger scale response in the stability of the shear layer due to receptiveness of the flow. It is suggested the shear layer response is coupled to the forcing frequency (Kotapati 2010).
harnessing this behaviour it acts as effective momentum redistribution motion in the boundary layer (Greenblatt & Wygnanski 2000). $F^+$ is a parameter common to all periodic flow control techniques, such as non-fluidic devices using oscillating flaps or wires in the flow (Stanek 2005). However, another periodic control type is where the addition of momentum to the flow is not via a surface mounted device. As such, parasitic drag is minimised. These flow control devices, referred to as synthetic jets, add momentum through ingesting the cross-flow fluid, and reintroducing it with added momentum. This is the flow control method considered in the present research.

### 2.4 Synthetic Jets

Synthetic jets allow perturbations to be added to the flow at a zero net-mass-flux condition (Glezer 2011). This removes the requirement for the ‘plumbing’ architectures of steady blowing or suction systems that have so far proven a weakness in applications of such control methods (Greenblatt & Wygnanski 2000). Synthetic jets possess a characteristic, intentional instability in the jet flow, compared to those naturally occurring in any jet due to the interaction of the ambient medium and the jet flow. The instability modes of jet flows and the evolution of the coherent structures generated has been an area of significant research, (Cohen & Wygnanski 1987; Gutmark & Ho 1983).

Synthetic jets they have received considerable research interest over the last two decades (Crowther 2010). The formation of a synthetic jet is discussed in detail by Glezer & Amitay (2002), and a brief overview will be presented here. Synthetic jets generate behaviour in the mean field that is not dissimilar to a steady jet. In the near field, discrete vortical structures are generated that convect into the surrounding fluid. The structures form from the periodic blowing and suction of the working fluid through the orifice. This motion is generated by the change in cavity volume, which is created by the moving wall. This is typically a piston, or oscillating diaphragm. Flow separation occurring at the orifice during the blowing phase, and moving from the near orifice region before the next entrainment phase, forms the vortex structure that is able to modify the surrounding flow without a net mass addition. Many studies have looked at the flow physics of a synthetic jet, which have identified a number of important parameters which define the jet flow. The momentum coefficient ($C_\mu$) of a jet is an important parameter. A key point to periodic excitation is that the momentum addition in order to attain control authority over a separated flow, has been demonstrated to be over an
order of magnitude less than for steady blowing. \( C_\mu \) is defined as the ratio of momentum addition by the jet, to that in the freestream flow (Greenblatt & Wygnanski 2000)

\[
C_\mu = \frac{\rho_o \overline{U_o}^2 d_o}{\sqrt{2 \rho_x U_\infty^2 c}}
\]  

(2.3)

where \( d \) is the characteristic length scale of the jet, the orifice width (diameter), subscript \( O \) relates to the jet flow, \( c \) is characteristic length scale of the body being considered, i.e. the chord of a wing. \( \overline{U_o} \) is the characteristic velocity of the jet at the orifice, and \( U_\infty \) is the velocity of the crossflow. The momentum input is hence dependant on the characteristic velocity from the jet, which proposed by Smith & Glezer (1998) is defined by:

\[
\overline{U_o} = \frac{1}{T} \int_0^{T/2} u'_o(t) dt
\]

(2.4)

where \( T \) is the period of the oscillation cycle, \( u'_o(t) \) is the instantaneous, spatially averaged velocity at the orifice exit. It is noted that this cannot be purely the mean velocity, as it will inherently be a net-zero value at the orifice for a single cycle. The characteristic velocity is an indicator of the strength of the vortical structure. The ability of the vortex sheet to roll-up into a discrete ring will relate to the jet’s effectiveness in entraining ambient fluid and hence the mixing ability. A failure to roll-up will mean the vortex sheet will act more in the manner of a slug of fluid. It has been shown that roll-up can be characterised by the Stokes number (Zhou 2009):

\[
S = \sqrt{\frac{(2 \pi f d_o^3)}{\nu}}
\]

(2.5)

where \( \nu \) is the kinematic viscosity. The parameters to alter the jet are the frequency \( f \), and the amplitude of the moving wall oscillation. The ejection of fluid from the orifice can also therefore be described as a stroke length \( L_o \), when considering the output as a slug of fluid giving:

\[
L_o = \overline{U_o} T
\]

(2.6)

where \( T \) is the jet time period. In the case of successive vortical structures emanating from an orifice into the fluid, it is understood that the frequency of the addition of rings into the flow effects resulting control. Strouhal number \( St \) is defined as:
and is a non-dimensional measure of the vortex shedding frequency. Consideration also needs to be given to the local boundary layer thickness ($\delta$), and Reynolds number. As the jet is intended for manipulation of the boundary layer, the relative ‘strength’ and hence axial velocity needs to be tailored such that it will have most effective trajectory. The relative jet velocity can be defined as the velocity ratio to the freestream, $R_U$ (Zhong et al 2007):

$$R_U = \frac{U_o}{U_{\infty}}$$

These descriptors of the synthetic jet flow will be used in Chapter 4 to characterise the flow from the actuator.

### 2.5 Synthetic Jet Flow Control

A synthetic jet flow is dependent on the geometry of the orifice; generally it is formed by either a long-aspect ratio slot, or a round hole. The flows generated from each orifice type are understood to differ (Leschziner & Lardeau 2011); the flow from a round hole and the manner in which it interacts with the cross flow is more complex, due to the greater three-dimensionality to the flow. Within the literature, a number of studies have addressed application of slot jets to separated flows, and will briefly be covered here. The mechanisms of control of a separated flow have been investigated experimentally (Greenblatt et al 2006), and numerically using Large Eddy Simulation (Saric et al 2006; Avdis, Lardeau & Leschziner 2009; Franck 2009), and in related studies (Dandois, Garnier & Sagaut 2007).

Common to (Greenblatt et al 2006; Avdis, Lardeau & Leschziner 2009; Franck 2009; Dandois, Garnier & Sagaut 2007), was the positioning of the actuator orifice with respect to the separated flow; it was positioned just upstream of the mean separation point. It has been understood that control authority is dependent on this excitation location length-scale. Generally as the distance between the actuator and separation point increases, the control effect over the flow will diminish (Greenblatt & Wygnanski 2000). Hence many studies have considered a wall mounted hump geometry when investigating other parameters. The experimental results of Greenblatt et al (2006) demonstrate the reduction of the height of the separated region is dependent on both the intensity of the periodic excitation, $C_{\mu}$, and the
frequency $F^+$. Optimal control occurred at $F^+ = 1.2$, with increasing $C\mu$ levels, from a threshold level of $O(0.1\%)$.

The results from experimental investigations (Gilarranz, Traub, & Rediniotis 2005) conducted at $F^+ O(1)$, have then also been numerically simulated, with good agreement in the control authority result found (Avdis, Lardeau & Leschziner 2009; Franck & Colonius 2009), (You & Moin 2008).

Numerical studies have extended understanding of the parameters, by investigating a range of actuation frequencies, yet with similar parameters to the previous experiments. In the work of Franck & Colonius (2012), $F^+$ was considered with a $C\mu = O(0.1\%)$, which in a practical sense, is a performance level viable from current sensor technology. The results suggested that separation control effect diminishes with increasing $F^+$. At $F^+ = 11.8$, there was no advantageous control authority over the flow. The effect of variation of $F^+$ is depicted in Figure 2.3, showing vortical structures at the peak blowing phase of the jet. The vorticity concentration is dissipated at $F^+ = 5$, with less defined separations between successive structures, as is seen when $F^+$ is closer to unity.

Figure 2.3 Phase averaged spanwise vorticity for different $F^+$ actuations for a controlled flow over a wall mounted hump, corresponding to the blowing phase (phase = 90°) for the jet cycle. Contour levels from -30 to 30, from Frank (2009).

Dandois, Garnier & Sagaut (2007) looked at similar conditions, at an order of magnitude higher $C\mu$, although still at a low velocity ratio. The results of low frequency tests $F^+ = 0.5$
corroborate with the findings of others; separation is reduced by the addition of large coherent structures to the flow.

However, it is the tests at high frequency $F^+ = 4$ that are of more interest. Actuation at these parameters was found to be counterproductive, as opposed to just ineffective. The extent of the separated flow modification due to actuation is shown in Figure 2.4. The high frequency control was described as a different manner of perturbation to the low frequency, termed an acoustically dominated mode, as opposed to the vorticity dominated mechanism. The mean velocity profile was altered by the pressure waves in the flow generated by the actuation. A reduction in the turbulent kinetic energy production in the flow delayed flow reattachment.

![Figure 2.4](image)

From Dandois, Garnier & Sagaut (2007); Streamwise velocity contours $U/U_\infty$ from 1 (black) to -0.2 (white), with streamlines a) the unactuated flow, b) $F^+ = 0.5$, c) $F^+ = 4$.

These results are interesting as the counterproductive control occurs at operating frequencies that would be feasible to achieve with modern sensor technology in certain applications. This would clearly be an important issue to understand in real-world application of flow control systems; as such a change in performance from a system would not be acceptable. Understanding of the frequency dependency of control authority is therefore a key issue.

Flow control has also been applied to aerofoil geometries. Separated flows at high angles of attack over a NACA0015 have showed receptivity to control at $F^+ = 0.6 – 1.2$. Gilarranz, Traub, & Rediniotis (2005) found separation would be delayed, creating enhanced lift. The
results showed a weak dependency of frequency within the small range tested, and little change to the force characteristics for the attached flow states. As with the separated hump flow, simulations were shown to be in good agreement, and demonstrated how the large coherent structures in the spanwise direction acted to redistribute momentum in the suction surface boundary layer, enhancing aerodynamic performance at high angles of attack. Table 2.1 summarises the findings of these pertinent studies. The results show experiments conducted across a range of scales of $C_\mu$ and $F^+$. All the tests use a flow control system that is a slot that spans the width of the flow domain. Low-frequency, quasi-two dimensional separation control would appear to be beneficial for separation control from the studies here. However, it is clearly based on having a good understanding of the characteristic scales to the unperturbed flow in order to generate control. It is of interest to consider other examples of high-frequency control in order to further assess some of the conflicting results that arise from quasi-two dimensional flows. High-frequency control has also been used on more complex flows over aerofoils.

<table>
<thead>
<tr>
<th>Type</th>
<th>Geometry</th>
<th>$Re$</th>
<th>$F^+$ Range</th>
<th>Control Authority $F^+$</th>
<th>$C_\mu$</th>
<th>Reference</th>
</tr>
</thead>
<tbody>
<tr>
<td>Experimental</td>
<td>Hump</td>
<td>$5.8 \times 10^5$</td>
<td>0.1 – 2</td>
<td>$F^+ = 1.2$</td>
<td>0.11%</td>
<td>Greenblatt† (2006)</td>
</tr>
<tr>
<td>Numerical</td>
<td>Hump</td>
<td>$5 \times 10^5$</td>
<td>0.84 – 11.8</td>
<td>$F^+ = 0.84$</td>
<td>0.11%</td>
<td>Franck (2009)</td>
</tr>
<tr>
<td>Experimental</td>
<td>NACA0015</td>
<td>$8.96 \times 10^5$</td>
<td>0.57-1.23</td>
<td>$F^+ = 1.13$</td>
<td>1.23%</td>
<td>Gilarranz† (2005)</td>
</tr>
<tr>
<td>Numerical</td>
<td>NACA0015</td>
<td>$8.96 \times 10^5$</td>
<td>1.13</td>
<td>$F^+ = 1.13$</td>
<td>1.23%</td>
<td>You &amp; Moin (2008)</td>
</tr>
<tr>
<td>Numerical</td>
<td>CBFS*</td>
<td>$1.98 \times 10^5$</td>
<td>0.5, 4</td>
<td>$F^+ = 0.5$</td>
<td>1%</td>
<td>Dandois† (2007)</td>
</tr>
</tbody>
</table>

Table 2.1 Summary of relevant studies of SJA flow control (* CBFS = curved backwards facing step) († = et al).

A mild trailing edge separation over an aerofoil has been controlled in the work of Goodfellow, Yarusevych & Sullivan (2013). The experiment considered a finite, long-aspect ratio slot jet embedded across a central section of an NACA0025 aerofoil. The array was positioned a small distance ($x/c = 0.03$) downstream of the flow separation point. Tests were performed at an $F^+ = \sim 40$. Separation was controlled, resulting in a drag reduction. A threshold $C_\mu > 0.3\%$ was required to attain control authority.

A notable study is the work of Melton, Yao & Seifert (2005, 2006), where for tests over a range $F^+ >0.3 – 13$, efficiency improvements were found from the control of separated flows over a high-lift aerofoil with relatively low momentum coefficients $C_\mu < O(1)\%$. Actuator arrays were acting at positions across the deflected flap of an advanced high-lift airfoil configuration. Control resulted in lift enhancement, as demonstrated at a range of configurations of $F^+, C_\mu, \alpha$. Within the work, a practical, piezoelectric based actuator was used. In order to alter the control from $F^+ = O(1)$ to $F^+ = O(10)$, an amplitude modulation
strategy was used for the driving signal, to decouple the forcing frequency from the dynamic response of the actuator. Interestingly in tests, for an $F^+ = 4 - 5$ and $C_\mu = < 0.5\%$, at the extremes of the range of flap deflection $\alpha$, actuation was found to be counterproductive to lift enhancement. At a number of other configurations tested however, $F^+ = 13$ actuation allowed greater reductions in pressure drag to be created then could be with the order-of-magnitude lower $F^+$ tests. These results and further associated tests (Melton, Schaeffler. & Lin 2007) are interesting as they demonstrate the use of high-frequency control over a range of conditions to the separated flow. Flow visualisation of the suction surface flow is shown in Figure 2.5. However as the experiments mainly concentrated on the global results, the precise reasoning and flow phenomena behind the variability in the control effect is not well understood. The results also highlighted that for a flow control system positioned at both the leading and trailing edge locations of an aerofoil, the benefits from simultaneous actuation at both locations is challenging to understand. However, characteristically high, sine-wave excitation was found to be more beneficial then lower frequency strategies.

![Figure 2.5](From Pack et al (2002); Flow visualisation of a) uncontrolled flow b) $F^+ = 12.2$ $C_\mu = 0.013\%$ controlled flow over a high lift aerofoil wind tunnel model.]

The results demonstrate however that as well as $F^+$ and $C_\mu$, important variables to control authority are the orifice location relative to the separated flow; all of which inevitability differ for each specific separated flow case considered in each flow control research experiment. The understanding and parameterisation of control authority is therefore a significant difficulty due to the numerous, and coupled, characteristic time and length scales occurring in the flows. One parameter that is unclear from experiments with flow control that spans the entire flow domain is the importance of the excitation location in relation to the separated flow. This is in terms of the spanwise direction, and hence relates to the role of finite width jets. Recent experimental studies have looked at the flows generated by low-aspect ratio
finite slot jets. (Sahni et al 2011; Rathay et al 2014a). They considered the flow from slot jets positioned on an aerofoil profile, both without and with a leading edge sweep. Both tests operated at characteristically high frequencies of $F^+ = O(10)$.

Work by Sahni et al (2011) looked at complimentary experimental and numerical simulation of the flow from the slot jet. It concentrated on the behaviour of the jet flow, as opposed to the effect on a separated flow. Measurements and simulation demonstrated a volumetric characterisation of the jet flow. Although the tests used rather higher velocity ratios than potentially practical with real-world synthetic jets, it was shown that the finite nature of the low aspect ratio jet creates a complex flow structure. The semi span, mid region of the jet generates spanwise vorticity, in the same manner as a large coherent vortex as characterised in the studies of hump flow control (Avdis, Lardeau & Leschziner 2009). However, the flows at the edges of the slot act to generate secondary streamwise structures as the cross-flow outboard of the jet accelerates around the orifice. The effect is such that the perturbation width reduces further downstream, but a streamwise modification in the flow is generated.

In a similar manner, Rathay (2014a) conducted experiments with a high frequency slot jet array. This was positioned a short distance upstream of a deflected flap, and the flow over it modified with actuation. The effect of adjacent jets was shown to be important. Significant effects on the flow were able to be generated when alternate jets in the array were deactivated. Similar findings of the flow field effect were summarised by Vasile, & Amitay (2013) on a separated flow over the trailing edge of a swept NACA4421.

Such work shows the finite nature of the flow control jet is an important consideration to the manner in which control authority exists, as opposed to just the two-dimensional characterisation of how a certain parameter elicits a response in the flow. Excitation location in the streamwise and spanwise scales is important parameter to consider along with the $F^+$ and $C_\mu$. 
2.6 Round Synthetic Jets

The majority of studies from the literature have considered slot jets; however, round jets have also been investigated. A round orifice creates a more complex flow than the more quasi-two dimensional type at the semi-span of a slot (Leschziner & Lardeau 2011) (see Figure 2.6). Owing to the wide range of characteristic scales to the flow, the computational resources required to resolve such flow conditions can be prohibitive. This may explain the lack of studies to date concentrating on the simulation of round jets.

Figure 2.6 Numerical simulation images from Leschziner & Lardeau (2011) a) to d); Phase-averaged, equi-spaced points through the cycle for a round synthetic jet flow upstream of a turbulent separated flow. The coherent structures are visualised by iso-surface of pressure. The line represents the boundary to the reverse flow spatial area.

Round jets are of practical importance to investigate as arrays of round jets create less structural complications than long slots on highly loaded aerodynamic surfaces. They could therefore be a desirable form of practical flow control actuator assuming they can generate control authority.

In comparison of the two forms of jet, work by Kim, Kim & Jung is of interest, (2012). Experiments were conducted to understand the characteristics of the separation control due to the orifice configuration. A long slot was compared to an array of round holes of the same overall spanwise area coverage. With parametric changes of $C_\mu$, for a characteristically low range of $<0.1\%$, and tests at $F^* = 0.5 – 2$, round holes were shown to be a more effective configuration for controlling a separated flow. It is suggested this is owing to the greater streamwise persistent of the discrete, highly three-dimensional structures. They are able to generate a more substantial downstream effect on the flow when compared to the rectangular
jet flow. However the work only considered the quiescent form of each jet flow to surmise the different flow states. The round jet is a more complex flow structure to consider. Many studies have looked at the generation of a synthetic jet into quiescent surroundings, but given that the cross-flow condition will significantly alter the development of the jet flow, they are only mildly pertinent to the cross-flow condition of interest here.

A number of studies have considered the development of the vortex structure in a cross-flow boundary layer (Jabbal 2008; Zhou 2010). Jabbal identified that the flow in a zero pressure gradient laminar boundary layers develops downstream such that the vortex ring ejected by the actuator is manipulated by the boundary layer. It takes the form of a classical hairpin vortex structure, or a tilted vortex ring. The form is dependent on the shedding frequency and the freestream-to-jet velocity ratio. The coherent structure also gives rise to a secondary vortex pair that is embedded in the near wall flow. However, the coherent nature of the structures seen in these previous studies would be more difficult to understand in an adverse pressure gradient, turbulent flow. This is an area of research requiring further investigation.

Studies have recently been conducted however that move towards more challenging flow situations; by investigating the control authority of an array of round jets in a turbulent, separating boundary layer flow. Experimental and numerical simulations of an array of jets on a curved backwards facing step represent the state of the art in the understanding of round jet flow control (Zhang & Zhong 2011; Zhong & Zhang 2013; Lardeau & Leschziner 2011). The flow conditions were more similar to the previous cases of a curved backward facing step flow (Dandois, Garnier & Sagaut 2007) where a separated, reattaching flow was to be controlled, but in this case, the actuator array was finite, covering a small span at the centre of the array. The jets were positioned upstream of the mean separation point, approximately 7d in the length scale of the jet.

In the work of Zhong & Zhang (2013), detailed experimental measurements were made of the controlled flow around an array of three jets with spanwise spacing of 10d. A sensitivity to the velocity ratio was found, when tests were performed at low $F^+ (<1)$. Of two actuation frequencies tested, control authority from the higher frequency was more receptive to increased velocity ratio; the separated flow extent would be reduced as jet intensity increased. Increased jet velocity was not as beneficial at the lower frequency however. It suggested at a distinct difference in the control mechanism due to actuation frequency, and the control authority also being coupled to the jet intensity ($R_U$). Flowfield measurements of the
evolution of the coherent structure revealed with striking clarity the vortical structures (shown in Figure 2.7), and how their growth diminishes with increased frequency. The smaller, higher frequency structures generated less variation in the temporal behaviour of the cross-flow. Higher frequency control showed to be the more robust method of reducing the separated flow.

Interestingly however, within these tests there was no discernable interaction from the adjacent jets, suggesting spanwise authority of the control is weak outboard of the diameter of the jet.

Numerical simulation of the same flow has shown good agreement to experiments at the low actuation frequency, and further extended the understanding of the mechanism of control. One interesting parameter considered with the study of Lardeau. & Leschziner (2011) was the injection angle of the jet. A counterstreamwise orientation generated further complexities to the evolution of the jet flow. However, the strategy enhanced the spanwise control authority,
and therefore the overall control authority afforded from a single jet over the flow. The change in angle was seen to be more effective than an equivalent 50% increase in jet velocity ratio for the wall-normal jet. This was due the angled jet creating higher levels of turbulent activity in the flow, and a more vigorous mixing enhancement to generate further momentum transfer through the cross-flow. This mechanism is believed to be less dependent on the cross-flow being receptive to the jet flow, as the frequency dependency of slot jets has shown. With the finite nature of the circular jet however, this is a mechanism only very localised to the jet flow, so careful consideration of the excitation location was deemed to be important to enhancing overall control authority.

Circular jets flow control systems show potential to be a desirable active flow control strategy. To date however the coverage in the literature is particularly sparse. Given the complexities of a controlled flow and the number of parameters, preliminary studies to assess capabilities are still very much of interest, in order to better understand the potential of such systems. The important aspects where fundamental, highly detailed investigation is best directed in order to generate a predictable, robust active control technique could then be less challenging to understand. The effects on performance due to some of the primary parameters identified from the literature are still to be investigated and understood in detail; so efforts to achieve this are very much required.

2.7 Flow Control Application

The use of synthetic jet flow control in specific experiments that move away from purely attaining separation control has also been investigated. The on-demand nature of the benefit to aerofoil performance means such devices show potential as replacement for control surfaces such as ailerons on UAV’s (Farnsworth, Vaccaro & Amitay 2008), or as force enhancement of an aircraft tail fin (Rathay et al 2014b). In these applications, the on-demand benefit of the control becomes highly advantageous, as it means an aerodynamic body such as a tail fin could be reduced in size, or an aileron removed completely. This then has a net benefit as an overall mass saving for the vehicle. Such flow situations are more complex to understand however.

Recent lab based studies of swept wing flows have found that slot-type jet arrays with certain spanwise spacing show potential to robustly control the complex flow with high frequency
$(F^+ = O(10))$ actuation. Rathay et al (2014b) considered the area (the array acts over) based
momentum coefficient as opposed to the chord based, by looking at the spacing of jets and
how this affected control authority. An interesting result from this work was the authors
surmised that jet spacing could potentially be too dense in order to generate most effective
separation control of the flow state covered. This is interesting, as the placement of control in
terms of the minimum actuator density for control authority is an important point, yet has
only been briefly covered in the literature for active flow control (Jabbal et al 2013). The
important role the device would have in such applications means they would possess no fail-
safe operating method, so the reliability and performance of the active flow control is critical
to safety. Thorough understanding of the performance of such systems is therefore required if
such uses are to be realised. A notable demonstration of synthetic jet control technology was
made by Nagib et al (2004), where a performance efficiency improvement was generated on
a tilt-rotor aircraft in real flight conditions. Further such studies encompassing all of the
challenges associated with realising a practical flow control system are however not prevalent
in the literature, due to the significant effort required to achieve such results. The potential of
flow control systems however can be seen from such studies.

2.8 Flow Control Technology and Modelling

Common to all the studies discussed is the laboratory-based application of synthetic jet
technology. This owes to the inherent challenges of understanding the flow control
mechanisms, and realising how to scale these to $Re$ and jet $RU$ of real world conditions. This
also leads to discussion of the enabling technologies for real world flow control systems.
Detailed coverage of actuator technology is covered in Cattafesta & Sheplak (2011). In this
research only the fluidic zero net mass flux type has been considered, so the reader is directed
to the literature for an overview of the other periodic excitation technologies of interest in
active flow control, such as plasma actuators, and non-zero net mass flux systems, such as
fluidic oscillators (Martin, Bottomley & Packwood 2014), and sweeping jet actuators
(Woszlido et al 2010; Melton 2014). As demonstrated previously by others, an actuator can
be composed of a voice coil (Greeblatt et al 2006; Khodadoust & Washburn 2007), a
piston/rod arrangement (Gilarranz, Traub, & Rediniotis 2005; Zhang & Zhong 2011), or a
piezoelectric diaphragm (Martin, Bottomley & Packwood 2014; Chen & Beeler 2002), as the
moving wall to the actuator. The former methods have previously been viable options for
certain experiments; as they both have bandwidths that have been applicable to $F^+$ actuation at around unity. However when considering the requirements of real-world application, piezoelectric devices present far fewer design challenges when bandwidth and mechanical design (mass, thermal output) considerations are important.

The use of piezoelectric elements in an actuator design is challenging however. They are resonant, non-linear, coupled devices that are highly sensitive to boundary conditions (Gomes 2009). However, despite this challenge, many research programmes have developed effective actuators based on this the technology (Melton, Yao & Seifert 2007). To the authors knowledge, they represent the most practical method of producing a bandwidth of actuation that can provide a jet flow of $F^+ > = 1$ for relevant flow conditions. However, with lab based studies conducted at $F^+ = O(10)$ producing interesting results, realising this excitation level for higher $Re$ flow situations is unrealistic with contemporary commercial off the shelf piezoelectric elements. These devices have typical frequency ranges of several kHz (Cattafesta & Sheplak 2011), which may limit $F^+$ to $\leq ~10$ in real world low-speed high lift aeronautical and motorsport applications, (i.e. localised flowfield velocities of ~150 m/s over aerofoils for example) This is clearly a limitation to investigating very high $F^+$ with zero net mass flux devices. Studies that look to develop actuator technology are therefore a key requirement.

There has also been significant research in terms of investigating the performance prediction of actuators, through the use of low-order modelling techniques. These are referred to as Lumped Element Models (LEM’s). The aim of such models is to predict the output velocity from knowledge of just the input power or another quality that can be easily measured, and the physical parameters of the actuator (Gallas 2005; Tang & Zhong 2009; Persoons 2012). Predictive tools are an important future consideration. Successful models can enable efficient definition of the effect of changing the parameters of the design on the performance of the actuator. Understanding of a characteristic velocity output of a flow control device from a low order model is highly useful for defining input parameters for more complex numerical simulations. These continue to be a vital development tool for the conceptual overall aerodynamic design, and rapid optimisation process of aircraft and racecars (Ciobaca et al 2011, 2013). Although predictive tools clearly are of future interest however, in a research programme such as this, where the development of a practical SJA, and subsequent testing of it in cross-flow conditions are prerequisite to any optimisation, experimental measurements of the jet velocity are still the preferred method of characterisation in this study.
2.9 Real-World Flow Conditions

With regards to application of active flow control, as identified previously, the high-lift requirements of civil transport aircraft and race car aerofoils represent flow-states that are of interest to consider for future application of separated flows. Both applications deal with flows over the wings that are subject to a ground effect flow condition. However, from the literature, there are very few studies that consider the addition of this parameter to laboratory-based flow control system tests. To the best of the author’s knowledge, only passive flow control in ground effect has been sparsely investigated, looking at vortex generators operating on aerofoils (Kuya et al 2009a, 2009b).

![Image](image)

Figure 2.8 Wind tunnel testing of ground effect aerodynamics on a) aircraft (Van Dam 2002), and b) Le Mans prototype sportscar (TMG 2014) wind tunnel models in close proximity to a rolling road.

Ground effect flow has distinct differences in the aeronautical and automotive environments however. Wing in ground effect flow for air vehicles is an enhancement in lift due to the proximity of the wing to the ground. The presence of the ground alters the flow over the pressure surface (which faces the ground), creating an increase in pressure over the underside of the wing compared to the freestream condition, enhancing overall lift (Ahmed, Takasaki & Kohama 2007).

For automotive flows where downforce (negative lift) is required, a nose down, inverted wing configuration is employed, and the suction surface is facing the ground. As such, the ground effect mechanism is subtly different. An increase in downforce over that from a freestream condition is due to an acceleration of the suction surface flows. This is due to a convergent channel that is created by the leading edge of the wing and the ground, and the associated increase in flow velocity, and low pressure magnitude this produces.

The research here is orientated towards robust development of an active flow control system desirable for a motorsport application, in terms of the systems operating requirements. So as
such, only the automotive ground effect will be considered from the literature. A comprehensive review of the flows around a racing vehicle that are dominated by ground effect is detailed by Zhang, Toet, & Zerihan (2006).

In order to properly depict the ground effect flow over an aerofoil in experimental research programmes, a rolling road is required such as those shown in Figure 2.8. Such experimental facilities are costly, which could be a likely reasoning for the limited published literature on the topic of fundamental ground effect flow conditions.

A significant contribution to the understanding of ground effect flows was provided by the work of Zerihan (Zerihan 2001), where the global characteristics and behaviour of the flowfield around an aerofoil in ground effect was investigated. The generation of increased levels of downforce as a function of angle of incidence ($\alpha$) and ride height ($h$) was understood to be complex, and the reader is directed to the literature (Zerihan 2001) for a more comprehensive description; a brief portrayal will be covered here. For the cambered aerofoil profile tested, levels of downforce (and drag), increased over those generated in freestream conditions. This effect increased as ride-height reduced. This however would only occur until a critical ride is reached; as at a non-dimensional ride height of $h/c = <0.15$, a force reduction phase is encountered. As $h/c$ reduced further, downforce decreased. This is attributed to an increased trailing edge separation which develops in the adverse pressure gradient.
The downforce reduction is seen to increase as ride height decreases down to very low levels and/or angle of attack increase further. The effect of decreasing ride-height is analogous to the increase of incidence for the freestream condition. Figure 2.9 depicts the effect of ride height on the rate of change of downforce, and on trailing edge separation point for a fixed $\alpha$, taken from a) (Zhang 2006), and b) (Zerihan 2001) respectively. Ride-height is an important parameter that governs the flows around an aerofoil and the overall forces generated.

In real world conditions, $\alpha$ and $h$ vary constantly, so the potential for separated flow on aerofoils working in ground effect is clear. With the finite nature of the wings, endplate effects were also studied (Zerihan 2001). The coherent streamwise vorticity that forms at the endplate also plays a significant role in the downforce characteristics of a wing in ground effect. The breakdown of this vortex is associated with very low ride heights, and an associated force reduction.

The on-demand nature of active-flow control is therefore of interest for automotive applications due to the environment created by ground effect phenomena. As well at limiting downforce and increasing drag from the wing in isolation, the turbulent wake of separated flows can negatively interact with other aerodynamic devices further downstream on a vehicle, in the case of the wings or similar structures positioned towards the rear of the car. Active systems therefore could be beneficial for improving the performance of downforce generating structures on race-cars, across a range of speeds and attitudes. Fundamental understandings of such flows are of interest.
2.10 Further Contributions to Research

The review of the literature has highlighted many points that are valuable information to assess the current context of flow control research.

Periodic excitation of the flow represents a very promising method of separation control. The dimensionless frequency plays a significant role in the efficacy of flow control. However, $F^+$ is just one of many parameters that can define control authority. Due to this, a wide range of results, some of which with conflicting findings, are demonstrated in the literature. A general agreement on which frequency is preferential for a given separated flow is therefore not well understood. In certain instances however, a characteristically high frequency control strategy has shown to be an effective manner in which to control separation (Melton et al 2005, 2006).

As well as frequency, momentum coefficient and actuator orifice form are important to control authority. $C_\mu$ is relatively well coupled to control authority with a linear relationship, so any flow control system needs to maximise the frequency bandwidth of $C_\mu$ generation in order to provide a robust control technique. Application of slot jets to quasi-two dimensional flows has led to a range of results from control strategies that have been investigated to date by others (Greenblatt et al 2006; Dandois, Garnier & Sagaut 2007). When a round jet is considered with a high-frequency actuation, limited investigations to date are available to draw conclusions of the effectiveness of the control. Martin (2014) recently investigated such control, demonstrating encouraging results for the control of trailing edge separation with an array of high frequency round jets.

Experimental work is somewhat limited by the current development of actuators, which in its own right is a significant requirement. Numerical work is limited by the high cost of detailed simulations that have to be used to depict the spatial and temporal scales of the flow interactions in detail. This observation is not to discredit research; it however highlights the framework into which any parametric research of synthetic jet flow control can fit.

Within the practicalities of experimental work however, there are still a number of findings that have not been covered, which represents a gap in the literature. By considering the following parameters to a set of experiments, then further understanding of weakly understood flow control issues will be gained:
Experiments that investigate the use of $F^+ > 1$ round jets, operating in a finite array would be investigating a set of parameters that are not well understood. Control that considers different excitation locations to those investigated to date is of interest. Recent studies (Jabbal 2013; Zhang 2010), have not considered spanwise spacing ($\lambda$) of $< 10d$. Experiments using closely arranged orifices that could allow for interactions between multiple jet flows, has been identified as a being of interest to investigate (Lardeau. & Leschziner 2011).

Likewise, streamwise positioning needs to be considered. Distances of $>O(10d)$ between the mean flow separation point and the flow control array have not been widely covered in the literature. Considering these parameters would be relevant, as with the practical requirements of flow control system designs, the precise location of separation may not always be understood, it may change, or actuators may not be able to be integrated in the ideal position due to physical constraints. Understanding of how excitation could be applied from a location a significant distance away from its point of requirement is important.

Finally, the technology of an active flow control system used for research should be relevant to potential real-world use. This means consideration of the aspects such as availability, mass and power efficiency of an actuator should be considered. Piezoelectric systems are the most promising in the perspective of the research programme conducted here. The effectiveness of relevant, practical flow control systems needs to be considered. System efficiency will always be required for a flow control system to be accepted by industry. Siefert covered this topic in recent work (2014), and detailed some evaluation criteria that are relevant to consider when developing flow control systems.

Findings on the practical aspects of the development of actuators are also of interest to report (Ternoy et al 2013). As discussed by Cattafesta & Sheplak (2011), with active flow control being a multidisciplinary field, the difficulties in communication across the disparate research areas means that reporting findings of the less well understood limitations, or difficulties of actuator development and operation could reduce a potential ‘bottleneck’ to future progress in flow control research. This is therefore an area of research that new studies should consider.
Chapter 3

Experimental Techniques

In this chapter the methods and experimental facilities used in the research will be outlined.

3.1 Methodology

In order to investigate high frequency active flow control, it was decided to use an aerofoil in order to generate a separated flow, to which control strategies would be applied. The separation would therefore be due to an adverse pressure gradient, which is a flow condition encountered in many situations in the typical flows over aircraft and ground vehicles. The use of an aerofoil model, as opposed a curved backwards facing step, which has been used in other studies to generate a separated flow condition (Greenblatt 2006) also presented the challenge of embedding the active flow control system inside the aerodynamic body. The design would therefore need to address the associated physical constraints. This compromise would align the work with the practical considerations encountered in real-world industrial applications; for instance if similar flow control systems were to be integrated into the wing and bodywork components of a race car, the length scales of the aerodynamic components are of a similar scale to the wind tunnel model here.

From wind tunnel tests, forces and surface pressures acting on the wing were measured, at a range of incidences and for different flow states. For some tests the suction surface was positioned close to a moving ground plane, to generate a ground effect flow. The parameters of the flow control system were altered in order to understand the sensitivity of control authority, and methods of increasing system efficiency.

Off-surface flowfield results were acquired using non-intrusive measurement techniques of Laser Doppler Anemometry (LDA), and Particle Image Velocimetry (PIV), to understand the time-averaged and phase-locked flows when control authority was created over the suction surface flow. Using these result sets, in the following chapters an analysis of the
mechanisms of control authority are described and the associated performance of the wing in different flow configurations. The equipment and techniques used to achieve this will be described in this chapter.

3.2 Wind Tunnel

All tests were conducted in the University of Surrey low-speed wind tunnel. The tunnel is of a conventional closed working section, closed return ‘Gottingen’ design. The streamwise turbulence intensity is 0.15% ±0.05% in the working section. Figure 3.4 shows a diagram of the wind tunnel layout.

Downstream of the working section, is the return circuit consisting of vertical tower with two sets of 90° turning vanes. This connects to a diffuser, downstream of which is the axial fan with the motor in the flow, which is not shrouded. The return section then features a second tower of the same layout, connecting to a finned heat exchanger that maintains a consistent temperature of the flow.

The exchanger exits into a setting chamber with four fine mesh screens and a honeycomb straightening panel. Wind tunnel speed was referenced with a pitot static tube placed upstream of the model, at the front of the working section. Metadata for the operating parameters of the tunnel are recorded for all tests undertaken, ensuring flow consistency was monitored throughout the research programme.

3.2.1 Working Section

The wind tunnel features a 9.85m long, constant profile working section of 1.4 x 1.1m cross sectional area. Testing was conducted in a plane 3m along the length. The wind tunnel walls at this point in the working section are created by large, high quality glass windows. This
allows optical access for the non-interference LDA and PIV measurement systems that are located external to the working section on a three-axis traverse system.

The temperature of the flow in the working section was maintained using a Proportional-Integral-Derivative (PID) controlled valve for the cooling system, to ensure stability of the temperature throughout a test to ±0.1 K, at user defined setting between 292 – 300 K. A closed loop control system was used to select the fan motor speed, creating a consistent flow speed throughout the duration of a test. The flow speed variation was ±0.02 m/s for a typical test at $Re = 8.9 \times 10^5$, based on the wind tunnel model chord.

#### 3.2.2 Blockage

The wind tunnel blockage due to the wing model, (including the shroud and strut) varied from 11.2 % at $\alpha = 0^\circ$ in the freestream model configuration, to 18.8% at $\alpha = 16^\circ$ when in the ground effect configuration. The blockage within the wind tunnel was higher than a value typically used in other facilities for the performance testing of wings. No blockage correction was applied to the results presented other than those considered in Section 5.2.1, where the results are compared to other experiments. The main consideration of the results is the non-dimensional change, ($\Delta$) of actuated to the unactuated flow states. Tuft visualisation during commissioning had shown that attached flows were evident towards the trailing edge of the balance shroud. Calibration of the wind tunnel working section velocity was performed for each configuration of the working section with the balance shroud in position.

#### 3.2.3 Rolling Road System

For correct modelling of the ground plane flow in the ground effect tests, a moving ground system is positioned in the working section. The system was manufactured by ATE Aerotech Ltd and is shown in Figure 3.2. The rolling road is 0.6m wide and 1.4m long. The belt therefore does not span the entire width of the model. However, as will be discussed in the following chapters, the active flow control system is positioned across the central-quarter of the model, so the flows about the central region of model are the focus of the work. As large negative pressures are generated by the inverted wing, an under-belt suction platen is used to keep the belt flat under the wind tunnel model. A cooling system is integrated into the platen to counter the heat generated due to the belt friction. Closed loop control of the belt tension, steering, suction and platen temperature ensure a consistent ground plane condition is maintained during tests. The rolling road speed is controlled via closed loop control, in order
to be maintained to within ±0.2 % of the wind speed. As well as the belt, the near floor flow condition is maintained with an upstream boundary layer removal system.

![Image](image-url)

**Figure 3.2** The rolling road (blue belt), and boundary layer removal system (perforated plate).

### 3.2.4 Boundary Layer Removal

For the tests in ground effect, the working section configuration was altered from the setup used in the freestream tests. A streamwise slot upstream of the road had suction applied across it, in order for the boundary layer that grows along the tunnel floor to be partially removed. Downstream of the slot, but upstream of the belt, a perforated plate is positioned that has a suction plenum beneath to remove further low momentum fluid. Fillets were added to the corners of the tunnel roof in order to nullify the increase in area that the ramped boundary layer scoop creates to the working section upstream of the road. The fillets decrease in size over the length of the working section, acting to reduce horizontal buoyancy across the working section. In the configuration used for tests, the measured boundary layer profile of the working section flow without the model installed is shown in Figure 3.3. A velocity deficit in the boundary layer was seen to reach a maximum of 2.5%. The overall boundary layer height was 16mm at the tunnel centreline and midpoint along the belt, the same location of the model during tests.
3.2.5 Traverses

Another important aspect to the wind tunnel was the use of traverse systems for the measurement equipment. The wind tunnel had a three-axis traverse system located outside of the working section, onto which the two optical measurements systems were installed. They allow manoeuvring of the PIV camera system and LDA probe across the spatial range of the chord, and the majority of the span of the wing. All axes are operated on lead screw traverse rails moved by computer controlled stepper motors which incorporate backlash compensation. Positional repeatability of the traverses was therefore estimated to be ±0.1mm. The same basic control and traverse configuration was used for the PIV laser light sheet, which was able to be manoeuvred over a range of field of view of the suction surface of the wing.

For the wing inclination, the linear traverse movement was translated into angular displacement, for which repeatability was ±0.1°. Angles were calibrated with a digital inclinometer.

3.3 Wind Tunnel Model

Since the intention of the research was to investigate the control authority of an active flow control system for a range of parameters, it was decided that a simplified, separated flow condition should be investigated, compared to the more complex separated flows that are
generated by multi-element wings (main element and a flap) typically used on aeronautical high-lift systems and race cars (Bauer et al 2014). A single element, symmetric aerofoil that exhibits a trailing edge separation was therefore chosen. A NACA0015 profile has been shown in previous studies to generate a flow that would be suitable to investigate, (Gilarranz, Traub, & Rediniotis 2005), (Melton et al 2008).

3.3.1 NACA0015

A NACA0015 profile wind tunnel model was designed for the research programme. It had a chord of 0.43m, and aspect ratio of 2.465. The model spanned the working section of the wind tunnel, but with a 2mm ±0.5mm gap to the wind tunnel walls either side, in order to prevent wall contact. The chord of the wing was constant across the entire span, with no taper, sweep or twist. A finite trailing edge of 1.3mm ±0.1mm was chosen for manufacturing reasons. As the model spanned the width of the tunnel, endplates were not used.

3.3.2 Construction

The model was manufactured by the industrial partner in two halves, in CNC milled and polished aluminium. The design was such that the split lines were on the pressure surface at $x/c = 0.013$ and $x/c = 0.93$ respectively, and can be seen in Figure 3.4.

![NACA0015 wind tunnel model](image)

Figure 3.4 NACA0015 wind tunnel model below model support; A = Aluminium shells of pressure and suction surfaces. B = Polymer outboard sections C = Apertures for flow control actuator array (array shown installed) Actuators located at $z/c = 0.29, 0.23, 0.17, 0.11, 0.05$ in ± Z axis D = centreline pressure tappings.
The active flow control system and some of the pressure measurement instrumentation were installed in the void inside the model. A Selective Laser Sintering (SLS) Rapid Prototyping technique was used to manufacture the outboard sections to the model, forming the outermost 5% to the span on either side. The surface of the model was aerodynamically smooth and finished with a matt black painted finish. Apertures in the surface of the wing were positioned across the central quarter of the span, towards the leading and trailing edge, for flow control actuator arrays. Further details on the array are presented in Chapter 4. The model is shown in Figure 3.4

3.3.3 Pressure Tappings

All tests were conducted with pressure tappings on the model surfaces. These were installed on the centreline in a spanwise staggered layout, (see Figure 3.5). 0.8mm internal diameter stainless steel hypodermic tubing was bonded in place flush to the wing surface, and connected to flexible silicone tubing that routed through the model strut to a mechanical indexer and pressure transducer. 30 tappings were installed; 22 were on the suction surface with clustering toward the leading edge so the pressure distribution is well resolved, (shown in Figure 3.5). The remaining 8 were distributed across the pressure surface.

Figure 3.5 Detail view of central span of suction surface of model; = wind tunnel model external geometry; = chordwise planes of measurement z/c locations; A = leading edge B = pressure transducer location; C = trailing edge pressure transducer location.
As well as pressure distributions at the semi span, surface pressures were acquired at two further points on the model. These measurements were acquired from transducers installed very close (<5 mm tube length) to the wing surface, in order to investigate the unsteady response. These were positioned at \( x/c = 0.76, \ z/c = 0.17, \) and \( x/c = 1, \ z/c = 0 \) respectively (see Figure 3.5). All pressure tapping locations are detailed further in Appendix D.

3.4 Model installation in the Wind Tunnel

The installations of the wing in the wind tunnel can be seen in Figure 3.6, for the two different setups used. For both models positions, a Cartesian axis system is used (shown in Figure 3.1). The origin is positioned by the semispan, leading edge and chord (camber) line planes. The x-axis is the streamwise direction, the-y axis the vertical, and the z-axis the spanwise respectively. For the tests in a freestream condition, the wing was suspended at approximately the mid-height of the tunnel. The ground effect configuration suspended the model above a rolling road system via an extended strut. Both setups used of two struts that supported the wing from an overhead balance. The struts were positioned at \( z/c = \pm 0.895, \) and pivot about the quarter-chord position from brackets internal to the wing. At the trailing edge, a third strut is positioned at \( x/c = -0.09. \) This was connected to a computer controlled vertical traverse, allowing the incidence of the wing to be altered. For the tests in ground effect the lower section to the strut was telescopic, allowing manual adjustment of the ride height of the wing.
The strut was connected to a six-component force and moment balance. This was positioned in the working section and shrouded by a NACA0030 profile with rounded tip. The struts were faired with rapid prototyped shrouds to minimise interference with the flow. These acted as a conduit for the pressure tapping tubing, and data acquisition and flow control system electrical cabling between the model and the associated equipment external to the wind tunnel.

For all measurements, the wing was positioned in an ‘inverted’ configuration, such that the suction surface faces the floor of the wind tunnel. Positive angles of incidence are due to a nose-down pivoting of the model. This ensured minimal interference of the suction surface flow with the model supports. As such, all positive lifting forces quoted in the analysis were in fact a downforce, but are simply referred to as lift as a matter of convenience.
3.5 Data Acquisition

3.5.1 Software

A data acquisition software program was used to process and store the incoming digital signals from the various sources. It was an in-house developed suite of programmes written in LabVIEW programming language. The software is designed in a series of modular sub-programs, so that acquisition and analysis processes could be integrated into a single step. It allowed specific requirements for certain sets of measurements to be efficiently developed. Software was developed in order to control the active flow control system and perform signal processing tasks, synchronise data acquisition from multiple sources, and automate the data reduction processes for the datasets acquired.

3.5.2 Hardware

Analogue signals of the sensors used had maximum output ranges ±5V, and were converted to digital signals by a Data Translation DT9836 analogue to digital converter. The device allowed up to 12 single ended channels to be simultaneously acquired. Channels are digitised to 65536 levels, allowing the voltage outputs of the sensors to be well resolved.

3.6 Force Balance

The force balance was an ATE-Aerotech six component balance, measuring lift, drag, and side aerodynamic forces, and pitch, roll, and yaw moments. A proprietary calibration analysis program performs a multiple regression analysis of data from six strain gauge bridges to output engineering unit results for each axis. The signal conditioned, filtered data is output at 200Hz to the LabVIEW program. Data were acquired for 30 seconds for all balance measurements. All force results presented in the results chapters are corrected for tare readings of the model support strut for the lift and drag respectively. No further corrections are applied to the results. Coefficients of force are presented in the standard convention, being non-dimensionalised as \( C_F = \frac{F}{0.5 \rho U_x^2 A} \), where \( A \) is the plan area of the aerofoil.

3.7 Pressure Measurement

All pressure data were acquired using temperature compensated Honeywell SDX01G2 pressure transducers, having an operating range of 0-6.9KPa, and a bandwidth of 10KHz.
Sensor output is directly proportional to measured pressure, but as the unit output is ratiometric to input voltage, signal consistency was enhanced with a custom built high quality power supply to minimise voltage drift. The output signal was amplified via a custom amplifier circuit built using INA128P and LM6171B amplifier boards in order to provide very low voltage offset, and ensure impedance matching to the data acquisition system respectively. A calibration was then applied to the acquired data. This calibration was generated in a separate experiment, using an applied steady pressure, and a reference pressure transducer, a Furness Controls FC012, that had routine calibration certification. The transducer is employed in many similar applications with other research programmes within the department. The sampling rate for all pressure measurements was 20 kHz (mean pressures were significantly over-sampled as a matter of convenience). Data was acquired over 30 seconds for all pressure measurements. Time-averaged pressure measurements were referenced to the tunnel freestream static pressure, taken at the wall of the working section, and non-dimensionalised as the pressure coefficient $C_p = \frac{p_d}{0.5\rho U_\infty^2}$ (where $\rho$ is the fluid density). The spectra are presented as the power spectra $\phi$ of the voltage signal from transducer; because the transducer response is linear, this is equally representative of the pressure.

3.8 Laser Doppler Anemometry Technique

Laser Doppler Anemometry (LDA) was used to measure the velocity in the flow field over the wing, and also in the actuator jet flow. The well established LDA technique (Tropea 2007) was ideal for such measurements; it could be used to identify the blowing and suction phases of the jet flow, and data could be acquired at relatively high acquisition frequencies, of $O(10^3)$Hz. It also allowed a fully non-intrusive measurement of the flows to be conducted, as the probe was located entirely outside the working section.

The measurements were made with a Coherent Innova 70C argon laser in conjunction with a Dantec Dynamics 2-component Fibreflow backscatter system, producing 514.5nm and 488nm wavelength light beams from the Colour splitter and Bragg Cell. The burst spectrum analyser used for processing was a Dantec Dynamics F60 Flow processor which output to the data acquisition computer and LabVIEW program by Ethernet connection. A fibre optic cable transmitted the beams to the measurement probe, from the laser and system located at the side of the working section. The probe featured a beam expander (visible in Figure 3.7),
which increases the beam separation of the 60mm probe, and provided a 0.8m stand-off distance of the measurement volume. This created a beam half angle of 4°, and a measurement volume of 1.1mm long (span across the tunnel), and ~ 0.062mm diameter. The probe was aligned with the axis system of the wind tunnel model and tunnel, such that measurement of the streamwise velocity for wake profiles was conducted with one component, and axial measurements of the jet flow with the other (in the y-axis of the tunnel axis), when the wing was inclined at $\alpha = 10^\circ$.

In order to make measurements close to the surface of the wing, the probe was rotated about the x-axis by the half angle, in order to align the top beam to the wing surface. Without being able to measure the out of plane component to the flow, it was not possible to transform the vector into the in-plane direction. The assumption was made since the angle was small the difference was also. A wake profile was taken with the setup to show a negligible difference in the results of in-plane and out of plane orientations (See Appendix B). For wake profile measurements conducted in Chapter 8, the probe was rotated in the opposing direction, in order to align the lower beam parallel to the rolling road. The beam orientation in this instance can be seen in Figure 3.7.

In order to rotate the probe around the x-axis and (for certain measurements of actuator jet flows in Chapter 7) around the y-axis, the platform of the probe mount was fitted with two rotational degrees of freedom to pivot the probe accordingly. These were manually adjusted and calibrated with a digital inclinometer, giving ±0.1° repeatability to the orientation of the probe. As sampling of LDA data is inherently non-uniform in time, sampling rates would vary from $10^3$ to $10^4$ Hz depending on the flow state tested. For cross-flow measurements, more than $10^5$ samples were recorded in all measurements presented; tests were conducted.
during simultaneous acquisition of analogue data, and so longer sampling times were employed, and high data rates and large samples sizes were met as matter of convenience.

For quiescent conditions, average sampling frequencies were greater than $10^3$ Hz, and a total of more than $10^4$ samples were recorded. In order to achieve desired sample sizes however, a sample and hold technique was used; where measurements would be taken in ‘blocks’ and measurements continued until a desired number of samples was met.

3.8.1 Data Processing

With the nature of the measurements taken, further data processing techniques were used on the velocity data acquired from the BSA. This was through a number of in-the-loop data processing sub programs that were integrated into the LabVIEW measurement and analysis program. As phase-locked velocity data of the jet flow was of interest to acquire, this required the synchronisation of the actuator driving waveform with the velocity data, as the data was uniformly and non-uniformly sampled in time respectively. This required a burst pulse array to additionally be acquired from the BSA. This is a TTL trigger signal that is output when velocity data is acquired. This signal is only ‘high’ when the measurement volume contains the data generating particle. As this data may occur within the data acquisition period and hence not be identified correctly, an in-house developed pulse extender system was used, in order to allow the pulse to be sampled simultaneously with the analogue data rate. This system incorporated a burst inhibit channel, which would be generated for input to the BSA, such that the BSA would then only process, and hence output when the signal from the pulse extender was not active. This removed the potential for the BSA to activate more than once during each extended signal. A calculation of the time-lag between analogue signal and LDA could therefore then be achieved, and the velocity output be understood as a function of the time of the jet driving signal, and hence the precise point in the jet cycle. The technique had successfully been used by Nathan (2012), where it is discussed more in detail. A further process was then applied to the data in the process of acquisition and output to a results file. This was to consider correction of the biasing of a seeded flow that will not be homogenous. This is generally the case for the majority of wind tunnel studies using particle imaging techniques to derive the velocity. As the sampling frequency is dependent on the flow rate of the seeding, there will be an increase in the amount of data points occurring at high velocities. Taking mean quantities from the biased
data would result in a higher mean velocity, as there is more acquired data at the higher speed when considering a flow following an oscillating, sinusoidal behaviour about a non-zero midpoint for example. Therefore, a weighting correction is required to combat this, and is applied to the data, using transit time as an associated weight \( w \). For quantity \( u \):

\[
\bar{u} = \frac{\sum_{i=0}^{N-1} w_i u_i}{\sum_{i=0}^{N-1} w_i}
\]

(3.1)

Where \( \bar{u} \) is the statistical mean. For higher order statistics, they are given by:

\[
\bar{u}^a = \frac{\sum_{i=0}^{N-1} w_i (u_i - \bar{u})^a}{\sum_{i=0}^{N-1} w_i}
\]

(3.2)

Where \( N \) is the total number of samples, and \( a \) is the order of the statistic. Work by Nathan (2012) found the transit time to be the favourable value to use to provide suitable correction. These statistical means are the results presented in all the LDA measurements.

### 3.9 Flow Seeding

Flow seeding was required for the flow for both the LDA and the PIV measurements. A TSI 9307 Atomiser was used for all tests, with olive oil as the seeding fluid. A typical diameter of \(~1\mu m\) is created by the Laskin nozzle. The compressed air supply to the device was controlled by a computer program to provide a consistent seeding supply to the flow over long test durations. Seeding was introduced at a single point at the end of the working section, which meant good seeding homogeneity throughout the flow was achieved as it passed through the return circuit and contraction, before reaching the point where testing is undertaken in the working section. For the quiescent conditions tests, seeding was introduced much closer to the model in the working section.
3.10 Particle Image Velocimetry Technique

Instantaneous velocity flow-fields over the suction surface were obtained using a Particle Image Velocimetry (PIV) system. The two-dimensional PIV method is well-established (Tropea et al 2007), and has been shown to be a successful measurement technique for the understanding of coherent flow structures observed in turbulent flows. It therefore was a suitable means of understanding the flow phenomena of the controlled flows. Measurements acquired data of the streamwise and vertical velocity over the suction surface, and in quiescent conditions, of the jet axial flow (as a vertical flow in the tunnel). The PIV system had been integrated into the wind tunnel systems to allow fully non-intrusive measurements of the flows. The setup and measurement techniques will be described in the following sections.

3.10.1 Mechanical Installation

The flow over the wing suction surface was illuminated using dual 50mJ Nd:YAG 532nm wavelength lasers, expanded into a light sheet of ~1mm thickness and ~0.1m height. Two different setups for the light sheet orientation were used, and are shown in Figure 3.8. Both setups used a two-axis traverse to orientate the laser system. This was positioned downstream and underneath the model (and wind tunnel). Access to the working section was through an aperture in the floor fitted with modular slats to limit the open cavity size. The traverse allowed spanwise movement of the light sheet across the central region of the model, and a pitching movement altered the chordwise position of the light sheet. For the freestream tests the laser system and collimator were moved together on the traverse to illuminate the flow over the suction surface. For the ground effect setup, a further three-axis traverse system installed inside the tunnel was used; the collimator was installed on this traverse, and positioned the light sheet parallel to the flow, illuminating the flow between the wing and the rolling road. An adjustable Nd:YAG Laser Line mirror re-orientated the coherent laser beam from the laser system into the collimator. A profiled fairing was attached to the collimator to reduce flow separation and buffeting, and when positioning of the two traverses was synchronised, the entire assembly was mechanically fastened to the wind tunnel floor to further eliminate movement in the light sheet due to being in the flow. As with the other traverse systems, repeatability of position was ±0.1mm. The PIV camera system was mounted perpendicular to the light sheet on the three-axis traverse located external to the working section, with optical access through the large windows. The measurement plane was in the $xy$ plane in the flow.
Figure 3.8  PIV Setups for the a), freestream configuration b) ground effect configuration
A = collimated light sheet B = 2-axis traverse C = light sheet collimator D = Nd:YAG Laser system
E = adjustable mirror F = 3-axis overhead traverse.

3.10.2 Measurements

Image pairs were acquired at a frequency of 7.25Hz, with a TSI Powerview 12-bit CCD
camera having 2048 x 2048 pixels. Aperture settings were an f-stop of ~4 in order to
minimise the field of view, to focus imaging on in-plane particles. Image magnification was
via a telephoto lens, which gave a spatial resolution of ~40μm per pixel. PIV images were
taken at 5 different locations along the chord to cover the areas of interest.

Calibration of the length scale to the image was performed prior to testing, with a ruler
positioned normal to the axis of the image in the plane of the light sheet.

The timing of the image capture was performed with a TSI 610035 Laserpulse synchroniser.
Synchronisation of the images could be resolved to a time scale of ns. The synchroniser
allowed for acquisition triggering to be performed by a TTL signal which was synchronised
to the phase of the driving signal of the synthetic jet. This allowed the images to be phase-
locked to the jet cycle(s). For all tests, 1000 image pairs were collected at each phase angle
selected. This had been shown to be a suitable resolution of data, in order to resolve the
mean statistics of the flow, in work by Littlewood (Littlewood 2013). In a previous study
using the same PIV system used in this experiment, a 1000 image dataset also showed good
agreement with results acquired with the previously detailed LDA system with a larger
sample size (Martin 2014). This data sample size was therefore used throughout the research.

PIV is a measurement technique which is susceptible to uncertainty in the result due to a
number of potential sources. The validity of the raw images was therefore routinely checked
for consistency. As the velocity vector calculation relies on the reflected light from a particle
approximating its displacement, result processing can be susceptible to peak locking;
whereby the reflected light from the particle is only imaged as one or less pixels. The shift
between images cannot therefore be correctly resolved. This source of error can be
minimised selecting a suitable field of view. All experiments in the research used the same
approximate magnification levels, ensuring consistent image quality. When acquiring data,
the particle pixel shift between the two images needs to be considered along with the
interrogation frame size. In order to ensure consistent processing, the identified particles in a
frame on the first image are required to remain in the frame on the second image, for the first
pass in any multi-step processing technique. This occurrence is a function of the speed of the
flow, the time duration between the images ($\Delta t$), and the out-of-plane movement of the
particle. Laser pulse separations of 8$\mu$s were used, minimising out-of-plane particle
movement and ensuring particle movements were generally up to $\sim\frac{1}{4}$ of the frame size (TSI
2011). During tests additional problems that adversely affect image quality are the residual
light in the wind tunnel, and reflections of the laser light from surfaces. Residual light would
enhance the background brightness of the second image, and hence affect the image quality.
Care was taken to cover all windows to the working section, and cover all ‘panel-gaps’ in the
booth the working section is housed in. Reflections of the laser light are problematic; they
saturate the light intensity at points in the image, skewing the cross-correlation between
images. Reflections were minimised by applying matte finishes to components; an anti-
reflective matte tape was applied over the suction surface and pressure surface of the wing
across the centre half of the span. This ensured a smooth continuous surface was created over
the spanwise area of the array, and greatly reduced reflections of the laser light sheet. This
same setup was used for all tests conducted throughout the research. Where reflections were inevitable, for instance on the rolling road belt, processing ‘masks’ were applied to prevent vector calculation in that specific area. In order to minimise this issue affecting the results calculation, at the start of any tests a smaller sample set of data was collected, and each image pair inspected and processed for validity, in order to avoid generating quantities of unacceptable data.

3.10.3 Data Processing

TSI Insight 4G software was used for the image-to-vector processing. A multistep analysis processing method was applied; the process used the image deformation technique with a multipass calculation. This iteratively reduces the grid size for calculation of the vector field. The image deformation technique is suitable for robust calculation of boundary layer flows and regions of vorticity, indicating at its suitability for the flows encountered over the wing. The method is covered in more detail in Tropea et al. (2007). The process uses a Fast Fourier Transform to generate the correlation map for the image pairs. The method is iterative, in that the first passes are conducted on an interrogation window size of 48 × 48 pixels. The final pass is performed on a 24 × 24 window. The peak of the correlation map is identified with sub-pixel interpolation, where a Gaussian curve is fit to the data-points, to obtain an improved resolution to the velocity vector output to the final vector field.

The multistage analysis then applies post-processing methods to the flow field in order to identify spurious vectors within flow. These are generally caused when the highest correlation peak from particles is not due to the ones following the flow. As out-of plane motions and lost particle pairs are inevitable in turbulent, complex flows, the stage maintains the validity of the final vector field. The conditioning forms two further stages; the first identifies and removes spurious results based on meeting a sufficient similarity to the surrounding vectors. The holes are then filled with interpolated values based on the neighbour values, in descending validity. This is done recursively until the field is filled.

These additional processes are however not the best methods to produce a reliable dataset; the emphasis was therefore on acquiring good quality raw data, and as such, monitoring of the acquisition was regularity undertaken to ensure conditioning requirements were minimised.

With vector-fields processed for all image pairs, grids of the instantaneous velocity, \( \hat{\mathbf{u}} = (x, y, t) \), \( \hat{\mathbf{v}} = (x, y, t) \) were further processed in order to generate flow-field descriptors. With the two-dimensional acquisition of the flow field, no out-of-plane component to the flow can be
understood, which will exist to some order of magnitude within a turbulent flow. However, the assumption of the flow conditions over the wing suction surface is that $\bar{u} > \bar{v} > \bar{w}$, and so the in-plane motions still give a sufficient description of the dominant motions to the fluid.

The mean velocity field was obtained by taking a statistical average calculated from all image pairs. As discussed previously, the mean velocity for the phase-averaged data was collected as a function of the jet cycle, for which the mean was also calculated. From the averaged data, the spanwise vorticity ($\zeta_w$) magnitude was calculated from the phase-locked flowfields, where:

$$\zeta_w = \frac{\partial v}{\partial x} - \frac{\partial u}{\partial y} \quad (3.3)$$

With conventional time-averaged $u(x, y)$, $v(x, y)$, and phase-averaged $\langle u(x, y, t) \rangle$, $\langle v(x, y, t) \rangle$ datasets generated from the instantaneous velocity components $u'$, $v'$, the periodic velocity fluctuations were then obtained by:

$$\tilde{u}(x, y, t) = \langle u(x, y, t) \rangle - \bar{u}(x, y) \quad (3.4)$$
$$\tilde{v}(x, y, t) = \langle v(x, y, t) \rangle - \bar{v}(x, y) \quad (3.5)$$

These results were used for the analysis of the flowfield in the subsequent chapters.

### 3.11 Summary

The experimental equipment and the associated analysis techniques used to measure and understand the fluid flow have been detailed. By using a range of well-established measurement techniques to evaluate the flows generated in the experiments, the mechanisms of control for the active flow control were able to be defined and understood, and the control authority quantitatively assessed. Prior to demonstrating the results of separation control over the wind tunnel model however, the development and experimental testing of the synthetic jet actuator will be detailed in the next chapter.
Chapter 4

Development of the Synthetic Jet Actuator Flow Control System

4.1 Introduction to Chapter

In this chapter the development and testing of the synthetic jet actuator is detailed. Prior to conducting tests to assess the ability of the flow control system to impart control authority to the wing flow, a methodology was developed in order to create a set of actuators of similar performance.

4.1.1 Synthetic Jet Actuator

Synthetic jet actuators can be created from various mechanical designs, however all share a common layout; they have one or more moving walls that vary the volume of the cavity, and an orifice. In previous studies, piston/rod arrangement SJAs have been developed and used for similar laboratory based experiments (Zhang & Zhong 2011), (Gilarranz, Traub, & Rediniotis 2005).

With consideration of the actuator parameters to be investigated however, (jet of \( F^+ = >O(1) \), in a flow \( Re = ~O(10^6) \)) the moving wall in this design was created using a commercially available piezoelectric disc.

Piezoelectric discs have the benefits of high availability, low cost, low mass, and can reach oscillating frequencies of \( O(10^3\text{ Hz}) \). They are also of minimal form factor, in that no other connecting mechanical assembly is required, i.e. a motor/rod arrangement.

They are the most practical option for the actuator design here, when considering the design of the aerofoil wind tunnel model. Using a piezo element was the enabling technology in order to create a ten jet array used in the tests. Many examples could be created efficiently. This was a desirable attribute of the concept, as for potential future applications of the basic design, hundreds of examples of the devices maybe required for more in-depth investigations.
4.2 Synthetic Jet Actuator Design

4.2.1 Disc

The disc was a PUI Audio AB4113B (PUI 2014). The disc has a 41mm diameter and a 0.4mm overall thickness. The disc is formed by a round brass diaphragm of ~0.1mm thickness. On either side a piezoceramic patch is bonded. The bi-morph patch layout generates displacements of the diaphragm that are higher than those generated with only one active layer, as found on a unimorph disc arrangement (Martin 2014). The two-patches are electrically connected in parallel but in opposite polarisation, in order that they both deflect in the same direction. The diaphragm deflection creates a change in the actuator cavity volume.

4.2.2 Cavity Design

The design of the cavity, orifice and surrounding housing was constrained around the geometric parameters imposed by the disc and wind tunnel model. The idealised geometry for the cavity that had been identified and developed in previous studies (Martin 2014). This was an orifice diameter of 5mm, when a cavity volume of $7.35 \times 10^{-6}$ m$^3$ is used.

Ten SJAs were to be clustered with a spanwise spacing of $5d$, and positioned at $x/c = 0.12$. The jets either side of the semi span required a $9d$ spacing due to the pressure tappings. These specifications were kept constant throughout the research in order to enable use of the existing NACA0015 wind tunnel model. The spatial volume into which the actuators could be fitted is shown in the diagram in Figure 4.1.

![Diagram of the space in the wing, showing the disc and the surrounding space available in which to integrate the SJA. A = wing structure, B = actuator RP part, C = actuator metallic part, D = clamp E = disc. (Dimensions in mm).](image)
Piezo discs require a means of sealing them to the cavity; this typically is a clamping plate (Ohanian 2011). During the development of the actuator, the disc clamping condition was found to be critical to the repeatability of the performance when creating multiple actuators, which has been similarly reported by others (Jabbal 2013). The actuator design used a two-piece metallic annulus clamp to constrain the disc. This clamp was separate to the cavity and housing, and was CNC machined from brass.

The clamp assembly featured an axial fine-pitch thread, which was tightened to apply a clamping force to the outer ~1mm circumference of the disc. A precision machined flange and groove radially located the disc in the clamp. Using a torque wrench to apply a torque of 3Nm ± 0.2, the clamping force could be applied in a repeatable manner to all clamps. This assembly allowed the deflection of the constrained disc to be measured prior to assembling the disc with the cavity housing. The disc and clamp is shown in Figure 4.2.

![Figure 4.2](image)

**Figure 4.2** Cross-section isometric views of the two-part clamp and disc.

The rest of the actuator design necessitated using a combination of a machined metallic annulus to provide a locating diameter for the clamp, and a fused deposition modelling rapid prototyped (RP) polymer bracket. This part also closed the cavity volume to the actuator, and located it in the wind tunnel model. Using the rapid prototyping manufacturing technique meant the complex geometry of the cavity could be easily manufactured in a repeatable manner. The volume of the cavity has been understood to be the most significant parameter to the response of the actuator, compared to the internal shape (Jain, Puranik & Agrawal 2011).

The orifice was drilled through the metallic annulus wall, using a jig and pillar drill, allowing a sharp edged hole of 5mm ±0.05mm to be accurately added to each actuator. The actuator assembly is shown in Figure 4.3
For the tests that will be discussed in chapters 6 to 8, two configurations of actuator were used. Both were the same construction, but differed in the orientation and length of the orifice. The first configuration, (shown in Figure 4.3) had the orifice normal to the wing surface. This created an orifice height of $l/d = \sim 1.56$. The second configuration (Figure 4.4) was of a counterstreamwise orientation for the jet flow, so that the orifice was $\theta = -45^\circ$, which increased the length of the orifice to $l/d = \sim 2.85$.

Figure 4.3 Cross sectional diagram of the actuator, showing the disc and cavity assembly;
A = clamp male part B = clamp female part C = metallic annulus D = RP part, E = disc.

Figure 4.4 Cross section through the actuator at the orifice for the counterstreamwise version.
For the actuator design, the Helmholtz frequency of the cavity is defined as; (Jabbal 2013);

\[ f_H = \frac{a}{2\pi} \sqrt{\left(\frac{d_O}{d_C}\right)^2 \frac{1}{Wl}} \]  

\[(4.1)\]

Where \( a \) is the speed of sound, \( d_O \) the diameter of the orifice, \( d_C \) the diameter of the cavity, \( l \) the height of the orifice and \( W \) the height of the cavity volume. For the wall normal orifice, \( f_H = \sim988\text{Hz} \), and for the counterstreamwise orientation, \( f_H = \sim741\text{Hz} \). The Helmholtz frequency is an important consideration; the acoustic resonance characteristics of the cavity and orifice are coupled with the disc resonance in a complex manner. Pressure rise in the cavity imparts a boundary condition on the disc, as its deflection will hence be damped (Gomes 2009), so matching of disc and cavity resonant frequencies is a parameter that affects the performance. The dependence of the actuator response on Helmholtz frequency and disc resonant frequency shows the actuator behaves a two coupled, second order systems, as reported by a number of others (Gallas et al 2003), (Gomes 2009). As discussed, Martin (2014) considered the optimisation of the cavity. The SJA output was found to be sensitive to cavity height, as decreasing height would increase the output velocity magnitude, and increase the frequency response of this peak. However, the experiments did not investigate the effect of increasing the orifice length.

4.2.3 Actuator Assembly

Actuators were assembled with a rubber gasket compound applied to the polymer and metallic parts to create an airtight seal. Retaining screws were used to accurately locate the disc clamp in the actuator body, and sealing provided by a grease seal. This assembly allowed the clamped disc to be removed and replaced with ease. Each cavity module formed part of the pressure and suction surface of the wind tunnel module due to the geometrical constraints imposed. The installation of the actuators inside the wind tunnel model is shown in Figure 4.5.
4.2.4 Hardware/Software

The piezo discs were supplied with the electrical driving signal from the hardware architecture shown in Figure 4.6. A continuous sinusoidal waveform was used to generate the suction and blowing phases to the cycle. The waveform can be described with respect to time by:

\[ y(t) = V \sin (2\pi ft + \phi) \]  \hspace{1cm} (4.2)

Where \( y \) is the input voltage, \( V \) is the peak amplitude (voltage), \( f \) the desired actuation frequency and \( \phi \) the phase at \( t = 0 \).

The waveform was generated by a Tektronix AFG3022B arbitrary function generator, and was supplied to ten individual amplifier channels from five W-Audio DA800 two-channel power amplifiers. The function generator signal received a maximum signal gain from the amplifiers of \( \sim 70 \). Such power amplifiers are generally best served for driving inductive loads, whereas the piezoelectric disc is a capacitive load. The low impedance of the disc meant it did not have a discernible effect on the current supplied from the amplifier however. The use of an individual amplifier channel per disc allowed individual attenuation of each jet.

From the amplifier, the power was supplied to the disc via cables routed through the wind tunnel model and strut. The disc required three electrical connections, one positive to each piezoceramic substrate and one ground connection to the brass clamp.
Figure 4.6 The hardware setup for supplying the driving signal to the actuator array
A = function generator; B = Power amplifiers; C = Synchroniser; D = data acquisition system.

The connecting wires to the piezo patch from the electrical connectors (visible in Figure 4.5) proved to be the part to the system architecture where mechanical reliability was an issue. With the disc oscillation at $O(10^3)$ Hz, the wire is subject to movements of the same frequency or higher harmonics of this. Where the wire connects to the piezo patch a rigid joint is created, which forms a hinge point in the wire. Early in the research, in tests with the disc where a traditional soldered joint and multi-stranded filament wire was used, the connections would fail after a number of hours, in a manner indicative of structural failure due to typical fatigue limits for metallic materials, of $\sim 10^7$ cycles. A solution to this was found by using a less rigid joining method. Lightweight woven ‘litz’ wire was connected to the piezo-patch with electrically conductive adhesive woven tape, (3M™ 2191FR). This gave a degree of compliance to the connection, and allowed the wire to flex at the connection point, as opposed to bending. This reduced failure, and significantly increased the length of time for which any given jet actuator would successfully operate. The robustness of the jet operation will be discussed further in chapter 9. The wire and tape attached to a disc is shown in Figure 4.7.
4.3 Characterisation Methodology

4.3.1 Disc Displacement

The requirement to characterise discs became apparent during the early stages of the research. A significant variation in the responses from a random collection of discs was found. Stringent control of the clamping force reduced the problem; although from numerous tests it became apparent that further improvement was required. However, within a given number of discs, some would compare far more closely than others. As mentioned previously, the actuator was developed such that the disc could have the clamping condition applied prior to assembly with the cavity. This allowed the centre point displacement to be measured without the cavity in place, to characterise the frequency response of the disc. Disc displacement is highly dependent on the resonant mode of the disc, which is dependent on the geometric properties and material composition. As these factors could not be readily controlled with the ‘off-the-shelf’ nature of the component, understanding the frequency response of a number of discs, and only using those of the most common response behaviour was deemed a practical methodology to use to ensure close agreement in disc characteristics. The disc is specified by the manufacturer as having a typical resonant frequency of 1,300Hz ±500Hz, which represents a similar range whereby peak displacements for a given disc are found. With the element to the actuator being an easily available, low cost item, this methodology of testing many discs presented itself as an effective manner to overcome the deficiencies of the component. A means of efficiently conducting tests was created in order to reduce the time required for this process. Tests measured the disc displacement, input voltage and current. The input current was measured using an in-house manufactured current sensor. It was based around a LEM CKSR 6-NP transducer and custom power supply. The signal was calibrated
against a constant load supplied by a constant current mode laboratory-use switch mode power supply. The instantaneous position of the centre point of the patch was measured from underneath using a Micro Epsilon OPTO-NCIDT 2220 laser displacement sensor. The device had a bandwidth of 10 kHz and a measurement resolution of less than 1 μm. The sensor output was sampled with the data acquisition setup detailed in Chapter 3. The setup used for the tests is shown in Figure 4.8.

Discs were driven at a constant amplitude signal, and the position of the disc was taken as a displacement relative to the static position at the start of each test. Position could then be understood as a function of the input voltage to the jet, which was simultaneously sampled by the data acquisition system. The discrete samples were phase averaged, by ‘binning’ the data at 1 degree divisions of the cycle phase, creating a phase-averaged characterisation of the displacement. The data acquisition program repeated this process by sampling and processing data in blocks, until the prescribed number of samples in each bin had been acquired. The maximum displacement of the disc was output to a results file. This process could then be repeated for a number of actuation frequencies in an automated manner, as control of the function generator was integrated into the data acquisition software program. A high fidelity sweep of frequency response for a given disc could be acquired and analysed in a short timescale using this automated acquisition and analysis program (typically ≤60 seconds per disc test). This meant a significant number of discs could be tested in a time-efficient manner; and discarded or identified for use. Figure 4.9 shows the response of 20 discs randomly tested during the process, Figure 4.10 shows a selection of 10 discs used for the array assembly. With the assembled actuators, flowfield measurements were next used to
assess the performance. With the candidate discs in Figure 4.10 actuated at frequencies of >950Hz in the flow control tests, all discs experienced their peak displacements at a lower characteristic frequency. This meant they could be attenuated to give similar behaviours, as they were operating at post-resonant modes, where dynamic behaviour was less sensitive to input signal frequency.

![Graph showing frequency response of the disc.](image)

**Figure 4.9** A random sample of 20 tests of the frequency response of the disc.

![Graph showing normalized displacement of discs.](image)

**Figure 4.10** A sample of 10 discs of similar characteristics used for the actuator array.

### 4.3.2 Flowfield Measurements

Measurements of the flowfield were conducted in quiescent conditions in order to understand the nature of the flow. The jet operated at 980Hz. Data were collected as phase averaged
datasets of the axial velocity at the centreline. This was acquired with the PIV setup detailed in Chapter 3. Phase averaging was achieved by using a triggering TTL signal generated by the function generator at a user-defined point in the jet signal cycle to synchronise the image capture sequence. The TTL signal had a response time of <9ns, which was more than an order of magnitude less than the timescales of the jet flow. 20 equi-spaced points throughout the jet phase were measured with 1000 data samples (image pairs) acquired at each. The development of the jet axial velocity across the cycle can be seen in Figure 4.11. Results have been non-dimensionalised by the maximum velocity measured across the full jet cycle, and the diameter of the jet orifice as the length scale.
Figure 4.11  Isocontours of phase-locked axial velocity. Contour levels are shown in increments of $U_p/U_{max} = 0.1$. a) $\phi = \pi/10$, b) $\phi = 4\pi/10$, c) $\phi = 7\pi/10$, d) $\phi = 10\pi/10$, e) $\phi = 13\pi/10$, f) $\phi = 16\pi/10$, g) $\phi = 19\pi/10$.

The jet flow is axis symmetric in the quiescent surroundings. Throughout the phase the peak velocity occurs in the vicinity of the dipoles of the flow. These convect away from the orifice in the axial direction, without a significant spreading in the spanwise direction.

As the phase of the cycle progresses, the suction and blowing phases are evident; A negative axial velocity develops in the flow near the orifice at $\phi = 13\pi/10$ and diminishes by $\phi = 19\pi/10$, signifying the peak of the suction phase. Peak axial velocity is generated at $1 < y/d < 2$.

The velocity fields imply a coherent vortex ring is created at the orifice exit, and is sustained in the flow in the axial direction as it convects away.

The phase averaged vorticity magnitude $\zeta d/U_{max}$, is plotted in Figure 4.12. A coherent vortical structure is generated during the blowing phase of the cycle. This structure successfully escapes the near field region of the flow before the suction stroke entrains the surrounding fluid from close to the orifice. The roll up of the shear layer from the jet flow to a vortex ring corroborates the Stokes Number of the jet of $\sim100$ which is above the threshold value of $<10$ whereby roll up will occur (Zhou, Tang & Zhong 2009).
Figure 4.12 Isocontours of phase-locked vorticity. Contour levels are shown in positive increments of $\zeta x/u_{max} = 0.5$ a) $\phi = \pi/10$, b) $\phi = 4\pi/10$, c) $\phi = 7\pi/10$, d) $\phi = 10\pi/10$, e) $\phi = 13\pi/10$, f) $\phi = 16\pi/10$, g) $\phi = 19\pi/10$.

The axial velocity profile over the jet phase, at a height of $\sim 1d$ above the orifice is plotted in Figure 4.13. Peak velocity is generated at the centreline at all points in the phase, so identifying the peak velocity in the time-averaged flow will reflect the position of the jet axis.
The time averaged result is shown in Figure 4.14, which demonstrates a peak velocity occurring at the centreline across the cycle. A time-averaged suction flow can be seen at $0.7 < \omega/d < 2$. In Figure 4.15 the ratio of the spatially averaged velocity, $(\overline{u_s})$ to the centreline velocity ($U_{c\text{line}}$) over the jet cycle is shown. The averaged velocity is taken across the orifice width, at $\sim 1d$ above the orifice. Knowing this relationship therefore means a set of single point measurements of the centreline velocity can be acquired, and be used to estimate the velocity $\overline{u_o}$ (see Chapter 2, eq 2.4).

Figure 4.13 Phase averaged velocity profile $U_{\phi}/U_{\text{max}}$ at points across cycle: $\Box$, 0; $\square$, $3\pi/10$; $\triangle$, $6\pi/10$; $\ldots$, $9\pi/10$; $\ldots$, $12\pi/10$; $\ldots$, $15\pi/10$; $\ldots$, $18\pi/10$.

Figure 4.14 Mean velocity profile $U/U_{\text{max}}$.

Figure 4.15 Velocity profile $\overline{u_s}/U_{c\text{line}}$ across phase of jet.
4.3.3 Jet Axis Measurements

With the flowfield for the jet mapped, an efficient manner of testing the performance of the full array was developed. LDA measurements were used to measure the jet outputs. The orientation of the LDA measurement volume at the jet orifice is shown in Figure 4.16.

![Diagram of the LDA measurement volume positioning in relation to the SJA orifice.](image)

In order to ensure a repeatable measurement of the jet centreline velocity, an automated measurement process was used. A program would traverse the LDA measurement volume across the spatial area at a height of ~1d above the orifice, moving on a Cartesian grid of step size of ~0.1d. Time averaged measurements of the axial velocity across the area over and surrounding the orifice would be continually taken until the maximum was identified. The location of the maximum was defined as the point on the measured grid that was surrounded by at least three further measurements outboard in all directions, and the highest velocity recorded. The data was then interpolated with a cubic fit to identify the position of the orifice centreline. Figure 4.17 shows a result from one jet in the array of the non-dimensionalised mean velocity. The centre point was used as the measurement point during the iterative adjustments of the jet output.
A process of iteratively adjusting the input voltage to each jet was undertaken, in order to create a consistent output across the array. In order to do this, the LDA setup shown in Figure 4.16 was used in the same manner as previously, however, additional processing techniques to the data were used to characterise the output of the jet across the cycle. In the same manner as the measurement of the disc displacements, the velocity measurement was phase-averaged. Data was collected with a $2^\circ$ bin size, in order to create a high fidelity characterisation of the flow over the cycle, allowing the peak velocity of each jet to be defined. Each bin had the minimum number of data points, being $>500$ samples. As with the PIV characterisation of the jet, the blowing and suction phase of the jet cycle were identified, and these maxima and minima velocities were identified and output to a results file as part of the automated data acquisition process.

### 4.4 Array Results

#### 4.4.1 Baseline Configuration

Measurements were conducted with all jets operating simultaneously, as the performance of a jet operating independently of the others is different to that when operating as part of an array. A level of ‘cross-communication’ exists between the jets, in that their performances are highly non-linear and interlinked, due to the proximity of the flow fields. The process of tuning the array was established by applying the maximum operating voltage to all jets in the array initially, and attenuating the input voltages to match to the lowest output jet in the array.
However, with the complex flows, a number of iterations to the applied voltages were required in order to best match the outputs. Figure 4.18 shows the reduction in variation from a number of tuning iterations. Figure 4.19 shows the output velocity non-dimensionalised by the maximum attained across the array for the first and final iterations. Although variation is still apparent at the final settings, the significant outliers that occur when a constant voltage is applied are significantly minimised. Figure 4.20 shows the input voltages for each jet non-dimensionalised by the maximum. A range of input voltages is applied over the array, in this configuration the maximum input was 80 Volts peak-to-peak ($V_{pp}$). Creating an array of consistent performance for all jets is challenging, due to the flows associated with multiple jets and the sensitivity of the devices. In the configuration for tests performed in Chapter 6, the array created a median peak velocity across all the jets of 23.7 m/s with a range of ±8.4% (2 m/s). The mean velocity was 23.5 m/s.

![Figure 4.18](image1.png)  
**Figure 4.18** Standard deviation ($\sigma$) of the tuning iterations of the SJA array.  

![Figure 4.19](image2.png)  
**Figure 4.19** Velocity output across SJA array  
$\Delta$ = first iteration; $\square$ = last iteration.
Figure 4.20  Input voltage variation across SJA array

\[ \Delta = \text{first iteration}; \quad \square = \text{last iteration.} \]

With the array optimised, the momentum coefficient \( (C_\mu) \) of the jet flow could be characterised. \( C_\mu \) is based on the performance attained for the jet positioned at \( z/c = 0.17 \).

\( \bar{U}_o \) is defined by equation 2.4; where \( u_o \) is estimated from the centreline velocity measurement relationship shown in Figure 4.15 with the centreline velocity measurements acquired with the LDA system. \( C_\mu \) is calculated using equation 2.3 (see Chapter 2). For the tests that will be detailed in Chapter 6, the jet Reynolds number was 7900 based on the peak velocity. The velocity ratio was 0.22 based on the characteristic velocity. In a cross-flow velocity of 30 m/s (the freestream velocity used for wind tunnel tests), the jets have a momentum coefficient of \( C_\mu = 0.11\% \), at an actuation frequency of 980 Hz.

When the tests detailed in Chapter 8 were conducted, the jet measurement and optimisation process was repeated. The array created a median velocity across all the jets of 28.1 m/s with a range of \( \pm 21.3\% \) (6 m/s). The mean velocity was 29.4 m/s. The array achieved a less consistent performance than found previously, yet at higher peak outputs for more of the jets. This was achieved by using a maximum input voltage of 105V \(_{pp}\). However, at the jet at \( z/c = 0.17 \) where flow field measurements were to be conducted, the jet performance was similar to that of the previous setup for that specific jet. The jet Reynolds number was 7500 based on the peak velocity. The velocity ratio was 0.26 based on the characteristic velocity and the freestream velocity of 25 m/s (the velocity used for the wind tunnel tests in chapter 8). The jets have a momentum coefficient of \( C_\mu = 0.15\% \), at an actuation frequency of 1000 Hz.
Hz. Flowfield measurements therefore are conducted at a jet flow of similar performance to the tests conducted in chapter 6.

4.4.2 Signal Modulation Strategy

For tests that will be discussed in Chapter 7, a signal modulation strategy was used to alter the perturbing frequency of the jet flow. A signal transform is used to periodically reduce the amplitude of the carrier frequency $f_c$. The waveform can be considered a Morlet wavelet transform (Tropea et al 2007) and be described by:

$$\psi(t, f_c, f_m) = \frac{1}{\sqrt{\pi f_m}} e^{\frac{i t}{f_m}} e^{-i 2 \pi f_m t}$$  \hspace{1cm} (4.3)

where $f_c$ is the carrier frequency, $f_m$ the frequency of the zero-mean profile to the modulation envelope. This can typically be a Gaussian function centred about the time-period of the windowing. In the tests here a sine function was used for continuous modulation.

In order to understand the effect on the jet flow, phase-averaged data were taken at the frequency of the modulation. The carrier $F_c$ and modulation $F_m$ frequencies can clearly be identified in the input voltage trace shown in Figure 4.21.

![Amplitude modulated signal operating at $F_c = 980Hz$, $F_m = 61Hz$.](image)

Although the jet flow still oscillates at the carrier frequency, which remained at 980Hz as used in previous tests, the modulation frequency reduces the amplitude over a lower frequency. A pulsed jet is created on the time scales of the modulation frequency. Phase averaged data were taken with a bin width of ~0.5° of the modulation phase. Acquiring such
data therefore creates a time average of the carrier frequency within the modulation phase locked result. This creates a ‘noisy’ measurement of velocity, as bins contain random points across the carrier frequency. However, the jet flow on the modulation timescale is revealed. Figures 4.22 and 4.23 shows the phase averaged modulated jet flows at $F_m = 61$Hz and $F_m = 7$Hz respectively, representing low frequency perturbations to the flow, in comparison to the carrier frequency of $F_c = 980$Hz. Characterisation of the modulated jet flow is complex due to the two timescales involved in the flow structure. These both need to be resolved in the data to calculate the constantly varying peak velocity of the jet. The results collected here however show that the modulated signal generates a specific frequency to the flow on the modulation timescale. This pulsed jet flow is at a characteristically lower $C_\mu$ than the unmodulated signal.

![Graphs showing axial velocity over jet phase for different modulation frequencies.](image)

**Figure 4.22** Axial velocity over jet phase for 61Hz modulation frequency.  
**Figure 4.23** Axial velocity over phase for 7Hz modulation frequency.

### 4.4.3 Angled Orifice Configuration

Measurement of the array was repeated for the counterstreamwise orientation of the jet orifice, used in tests detailed in Chapter 7. A normal sine-wave driving signal was used for the jets in these tests. As seen in the previous tests, the behaviour of the array was non-linear. Measurements of the peak output were made for a range of frequencies. The most consistent performance across the array was found at 940Hz, which represented a slight reduction in frequency from those tests conducted previously. In Figure 4.24, results from some of the input voltage optimisation iterations are shown. The non-attenuated array of input voltage of $80V_{pp}$ was found to have a significant variation in the initial iteration, as shown by a standard deviation of 4.6m/s, with a mean peak velocity of 10.9 m/s. The optimised array however generated a similar mean peak velocity, (10.2m/s) yet with less variation. This relatively low
level of peak velocities however was not ideal for the flow control tests. Therefore a higher voltage input of $100V_{pp}$ was used, but for the same attenuation levels used across the array. This was able to generate higher peak velocities across the array, reflected in the higher mean of $\sim 16.3 \text{m/s}$ however, the standard deviation was $4.25 \text{ m/s}$, which was comparable to the variation from running all jets at the same input voltage of $80V_{pp}$. The counterstreamwise orientation was found to create challenges in producing a consistent array. This can be better understood when considering the cavity volume had been best suited for the orifice length used in the wall-normal configuration. The less consistent performance of the longer orifice design of the non-normal orientation demonstrates the sensitivity of the actuators to their boundary conditions. When these are not optimal, the array performance becomes subject to more significant variation.

Figure 4.24  Mean axial velocity for all jets in the array, for various input voltages, where:
- $\Delta$ = first iteration, $V_{pp} = 80$, $\sigma = 4.58 \text{ m/s}$;
- $\square$ = final iteration, $V_{pp} = 80$, $\sigma = 1.30 \text{ m/s}$;
- $\bigcirc$ = test configuration, $V_{pp} = 100$, $\sigma = 4.25 \text{ m/s}$.

In the final iteration of the angled orifice array, the median velocity across all the jets was $15.7 \text{m/s}$ with a range of $\pm 41\%$ (6.5 m/s). Taking the output of the jet positioned at $z/c = 0.17$, and using the calculation method as was used previously, the jet Reynolds number is 4115 based on the peak velocity. The velocity ratio is 0.095 based on the characteristic velocity. In a cross-stream velocity of 30 m/s (the velocity used for wind tunnel tests), the jets have a momentum coefficient of $C_{\mu} = 0.02\%$, at an actuation frequency of 940 Hz.
4.5 Power Requirements

For the momentum output generated from the actuator array, the power consumption is of interest to understand. In real-world applications, a flow control system would be required to reach a certain energy efficiency level, in order to provide a net benefit, so is an important consideration. A piezoelectric element has the electrical property of capacitance, whereby the input electrical power that generates the deformation of the disc is not completely expended by the dynamic movement, as piezoelectric material generates an electrical field with deformation. As well as this, the behaviour of the capacitive load on the electrical circuit (the disc is part of) will alter with temperature and as the disc operates at mechanical resonance. The power consumption of a disc can therefore be challenging to characterise. However, the averaged power consumption for a number of discs can be estimated. The parameters of the disc taken from the product datasheet (PUI 2014) for the disc are shown in Table 4.1.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Resonant Frequency, f (Hz)</td>
<td>1300 ± 500</td>
</tr>
<tr>
<td>Resonant Impedance, Z (Ohm)</td>
<td>200</td>
</tr>
<tr>
<td>Typical Input Voltage, V (V_{pp})</td>
<td>30</td>
</tr>
<tr>
<td>Capacitance at 1kHz, C (μF)</td>
<td>0.15 ± 0.045</td>
</tr>
</tbody>
</table>

Table 4.1 Summary of the piezoelectric disc properties (PUI 2014).

If the actuator is modelled as operating at non-resonance, in an unloaded condition, with a constant capacitance, then the maximum current draw can be simply estimated to be:

\[
P_{\text{AVE}} = \left( \frac{V_{pp}}{\sqrt{2}} \right) \cdot \left( \frac{I_{\text{MAX}}}{\sqrt{2}} \right) = V_{\text{RMS}} \cdot I_{\text{RMS}}
\]

The idealised and measured power inputs for a single disc are shown in Figure 4.25 as a function of frequency. The measured value shows non-linear behaviour compared to the idealised result. However, both are to the same order of magnitude over the range of frequencies tested. Over this range, both at and off mechanical resonance modes are encountered. At < 800Hz, which is below the manufacturer lower limit for the mechanical resonance for the disc, the power input to the disc is in closer agreement with the idealised value. However, this does not represent a frequency where a significant jet flow is generated. At higher frequencies of > 900Hz, power consumption of the disc is greatly reduced. The effect of operating the disc at close to the resonant frequencies is significant on the power
input requirement to the disc. The averaged power input for the 980Hz signal for an 80V_{pp} driving signal is 0.33W, which for the full array, would give a power consumption of 3.3 W. However, as seen in Figure 4.20, the same power input was not supplied to all discs. In addition to this, the response of the disc operating when assembled as an actuator, in terms of the load applied, will differ to the conditions of the quiescent air tests. However, the results here indicate the order of magnitude of power consumption of the array during the tests conducted.

For the tests conducted with a frequency modulation strategy, which will be discussed in detail in Chapter 7, power consumption was reduced. Non-dimensionalised voltages are plotted for three discrete frequencies in Figure 4.26.

As all the voltage traces used a sinusoidal waveform to the amplitude reduction envelope, this results in the same magnitude of reduction over the period, as can be seen in the figure when the differing frequency signals are overlaid in a non-dimensionalised form. The \( V_{\text{RMS}}(f_m) \) is equal to an \( O(30\%) \) reduction compared to the unmodulated signal \( V_{\text{RMS}}(f_c) \). This represents a significant power saving.

The overall efficiency improvement generated by use of the flow control system during the wind tunnel tests is discussed further in Appendix A.
4.6 Discussion

The methodology discussed allowed an array of SJAs to be created. Early on in the research an inconsistency of the output of the actuators was identified. One of the key reasons to this was due to the variation in the performance of the piezoelectric discs. This was a consequence of the device being a low-cost, high availability commercial component, being used in an application different to that in which such components are generally required to perform. The variability in the discs meant the frequency response needed to be measured before the assembly of an actuator. Understanding the characteristics of piezo discs was of interest, as it addressed some practical but relevant questions posed by the research. The process showed that such commercial piezoelectric discs are not of a technology readiness level whereby they would be viable for a mass-production flow control actuator. However, this issue does not discredit the idea of a piezoelectric based SJA. Given that developing a bespoke piezoelectric element of improved repeatability was not feasible within the research, characterisation of the disc allowed production of the numbers of discrete actuators required. Such testing of an actuator element could potentially scale to the requirements of even larger arrays than manufactured here, which would still render such an actuator of interest for the prototype nature of the technology still required in the research and development environment. Once the actuators were assembled, the methodology of individual signal attenuations generated consistent outputs for the array. As an in-situ characterisation of
discrete jets is required for a flow control system, identifying effective methods in order to achieve this is of interest. Using non intrusive optical measurements systems located external to the working section of the tunnel meant that the uncertainty of sensor calibration or drift was reduced, which is a problem associated with the use of hot-wire anemometry measurements or pressure based characterisations of the jet flow (Ternoy et al 2013).

However, in real-world type applications, where the environment of operation would be more adverse than the laboratory environment, such laser based measurements would be less practical. There would be value therefore in the investigation of similarly time-efficient, non-intrusive methods that could be employed in different environments. The results here highlight that understanding of the coupling of the input voltage to the jet velocity for piezoelectric SJAs requires flowfield measurements of each discrete jet flow (in an array), due to the variability in the outputs of multiple actuators. The input voltage to output velocity relationship is difficult to model. The problem of sub-optimal actuator parameters, such as achieving practical orifice lengths, will be a limitation in flow control systems designed for use in real-world applications. The results found here for the consistency of jet performance when the parameters of the actuator are altered from the ideal demonstrate that the detail design and measurement of actuator performance is therefore an important requirement for effective flow control systems. This highlights the need for detailed measurements of each actuator’s performance, in order to create a consistent flow control array.

4.7 Summary

The methods used to create an SJA design, produce multiple examples of it, and generate consistent jet outputs from an array have been detailed in the chapter. A simple SJA was designed and manufactured using modern and readily available manufacturing techniques. Non-intrusive, optical, (laser based) measurements have been used; firstly, to characterise the frequency response of the piezoelectric disc. Secondly, the flowfield of the jet was measured using PIV techniques. Vorticity concentration in a classical vortex ring structure is ejected during each cycle, and expelled to the far field flow. This is the mechanism whereby the time-averaged jet flow is generated, in the classical behaviour of a synthetic jet. From this data, the spatially averaged velocity of the flow was acquired. In order to configure the array of actuators, both time and phase-averaged point measurements of the flow above the orifice were made using LDA measurements for all jets in the array. The discrete jet flows in the
flow control array were adjusted through individual attention of the jet input power. Alternative orifice geometries and actuation strategies where then used and the performance mapped with the same methodologies. These differing approaches to actuation generated less consistent results. With an array of actuators created, an understanding of the flow condition into which the jet flows would be introduced was of importance to acquire. The results for tests of the flow over the wind tunnel model without the effect of control will be discussed in the next chapter.
Chapter 5

Understanding the Uncontrolled Flows Over the Aerofoil

5.1 Introduction to Chapter

In this chapter results are presented for the aerodynamic performance of the NACA0015 model in freestream conditions, without the flow control system being actuated. All tests were performed at a wind speed of 30m/s, giving a chord based Reynolds Number of $Re = 8.9 \times 10^5$. Tests were performed with free transition of the flow. Initial tests were conducted with the flow control system removed, whereby the pressure and suction surfaces had aerodynamically smooth infill plates installed on the model in place of the actuator modules. Identifying the characteristics of the unperturbed flow will allow the manner in which successful flow control is achieved to be better understood.

5.2 Force Balance Results

Results from the overhead force balance for the overall lift and drag values of the wing are shown in Figures 5.1 and 5.2 respectively. The angle of attack ($\alpha$) at which $C_{L_{\text{max}}}$ occurs is $\alpha = 14^\circ$. At angles beyond this, lift decreases, which coincides with significant increases in drag, as separated flows develop.
5.2.1 Comparison with Published Experimental Results

The data for corrected coefficients of lift $C_L$ and drag $C_D$ are compared to results from other NACA0015 models taken from the literature (See figures 5.3, 5.4). The experiments of Melton et al (2008) and Gilarraz, Traub & Rediniotis (2005), used wind tunnel models of different model setups, having lower blockage levels. The data from the Surrey wind tunnel model is corrected for solid blockage, wake blockage and streamline curvature using the correction methods outlined in Barlow, Rae & Pope (1999), and demonstrated by McAlister, & Takahashi (1991) and Selig, Deters & Williamson (2011).
Solid blockage ($\varepsilon_{sb}$) is;

\[ \varepsilon_{sb} = \kappa \frac{\text{object frontal area}}{\text{test section area}} \]  

(5.1)

Where $\kappa = 0.25$ (McAllister and Takahashi 1991). Wake blockage ($\varepsilon_{wb}$) is;

\[ \varepsilon_{wb} = \frac{c}{4 h_{ws}} C_D \]  

(5.2)

Where $c =$ chord length, $h_{ws} =$ height of working section, $C_D$ is uncorrected drag coefficient. Streamline curvature correction ($\sigma_{sc}$) is given by

\[ \sigma_{sc} = \frac{\pi^2}{48} \left( \frac{c}{h_{ws}} \right)^2 \]  

(5.3)

From which corrected lift ($C_{Lc}$) can then be given by:

\[ C_{Lc} = \frac{1 - \sigma_{sc}}{(1 + \varepsilon_{sb} + \varepsilon_{wb})^2} \]  

(5.4)

And corrected drag ($C_{Dc}$) is:

\[ C_{Dc} = \frac{1 - \varepsilon_{sb}}{(1 + \varepsilon_{sb} + \varepsilon_{wb})^2} \]  

(5.5)

In the examples from the literature, the Melton model was designed such that it spanned the working section (as with the Surrey model). No sting was attached to the wing pressure surface however, as the model was attached to a balance located below the tunnel floor, and the wing was positioned vertically in the tunnel.

The Gilarranz model employed a pressure surface sting attachment and used endplates to constrain the flow; the model did not span the working section and some of the flow control system architecture was mounted in the tunnel on the opposing side of the endplate to the wing flow. These differences of the Surrey model may suggest how the model orientation changes the fundamental flow characteristics of the NACA0015 profile. Both sources used force measurements taken from a force balance. All models have different aspect ratios: 0.95, (Gilarranz, Traub,& Rediniotis 2005); 2 (Melton et al 2008); 2.5 (present work).

The tests of Melton et al (2008) were at $Re \sim 2.5 \times 10^5$, giving a $Re$ below the transitional range. The tests of Gilarranz, Traub,& Rediniotis (2005) were at $8.9 \times 10^5$.  

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At low angles of attack, all the results show general agreement in the lift slope. At $\alpha = 14^\circ$, the literature predicts stall has occurred, however the Surrey NACA0015 creates maximum lift at this incidence. The drag results seen demonstrate differing trends to the Surrey model. The Gilarranz model showed a gradual increase in drag from $\alpha = 11^\circ$, with a higher $C_D$ then the Surrey NACA0015 within the attached flow range. The Surrey and Melton models produce more similar characteristics, with significant increases in drag occurring closer to the incidence of $C_{L_{\text{max}}}$. The different aspect ratios of the two cases from the literature compared to the present work is reflected in the lower efficiency performance. The higher drag levels in the Gilarranz model maybe due to the significantly different model layout in the tunnel, the smaller aspect ratio, and use of very large endplates, when compared the Surrey and Melton models (Chaudhary & Williamson 1992). The trends seen in the literature show that NACA0015 models can have differing drag characteristics at the critical angle range. The flows around each different model will likely have certain behaviours bespoke to each setup. Comparisons on the efficacy of the flow control system are therefore considered in comparison to the unactuated flows over the same model, as opposed to other flow control system performance results from the literature.

![Lift Coefficient ($C_L$) against incidence angle ($\alpha$)](image)

Figure 5.3   Lift Coefficient ($C_L$) against incidence angle ($\alpha$), □= Present data, 
Figure 5.4 Drag Coefficient ($C_D$) against incidence angle ($\alpha$), □ = Present data, \underline{---} = Melton et al (2008); \underline{-----} = Gilarranz, Traub, & Rediniotis (2005).

5.3 Static Pressure Distribution over the Model

5.3.1 Chordwise Pressure Distribution

Chordwise pressures coefficients ($C_p$, where $C_p = p_d / 0.5 \rho U_\infty^2$) at the semi span are presented in Figure 5.5. Increasing $\alpha$ increases the loading at the leading portion of the wing. The low pressure peak increases in magnitude at $\sim 0.004 \times/c$, up to the critical $\alpha$ of $14^\circ$. The $-C_p$ peak location is upstream of the orifice of the actuators, which when installed, are at $x/c = 0.12$.

At $\alpha = 13.5^\circ$ the pressure distribution plateaus near the trailing edge. This is as the pressure recovery to the trailing edge cannot be maintained in the adverse pressure gradient. As $\alpha$ increases further, the pressure recovery diminishes and the constant pressure plateau migrates forward. This incidence is the point of the onset of significantly separated flows.

After $\alpha = 14^\circ$, separation becomes apparent over the rearmost portion of the wing. $\alpha = 14^\circ$ is the threshold point whereby the trend of the leading edge suction peak increase with $\alpha$ increase, is sustained with pressure recovery to the trailing edge. With further $\alpha$ increase, pressure recovery diminishes.

This corroborates the overall global results of the point of $C_{L_{\text{max}}}$, and with the loss of loading measured after $\alpha = 14^\circ$. It hence represents a condition at which flow control methods could alter the flow. As $\alpha$ increases past $14.5^\circ$, low pressure peak reductions are seen at increased $\alpha$. 

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5.3.2 Suction Surface Canonical Pressure Distribution

Canonical pressures ($C_p'$) are useful in the analysis of separation of flow from the suction surface (Smith 1975). Replotting the surface pressures as $C_p' = 1 - (C_p - 1/C_{p_{max}} - 1)$ identifies the non-dimensional pressure across the suction surface, where 0 represents the peak and 1 the stagnation pressure. These are then plotted against the distance from the suction peak ($S_p$) non-dimensionalised by the distance from the peak to the trailing edge ($S_{pte}$). The form of the separated flow can therefore be identified. The plots for the higher angles of attack are shown in Figure 5.6.
Across the frontal region of the suction surface flow, $S/S_{te} < -0.4$, with increasing angle of attack, canonical pressures increase. The flows at $\alpha = < 14^\circ$ at the region $S/S_{te} > 0.4$, show a progressively less gradual pressure recovery. At higher angles of attack however, this is not the case, and the pressure recovery cannot be sustained. At the higher angles of attack the flow is prone to adverse pressure gradient induced trailing edge separation. This is opposed to the other characteristic flow separation mechanism for symmetric aerofoils, where an abrupt leading-edge separation occurs. The suppression of the trailing edge separation would enhance performance of the wing. It is therefore of interest to understand the nature of the trailing edge flow at the higher angles of attack where this separation occurs.

### 5.4 Effect of Flow Control Orifices

The flow control system was installed for all further tests conducted. This created discontinuities to the suction surface across the central quarter of the span. All other aspects to the model setup remained the same.

#### 5.4.1 Force Balance Results

The overall force coefficients are shown in Figure 5.7 and 5.8, compared to the previous configuration. Data was taken only at the higher $\alpha$ range. The two configurations have
differences in the lift and drag characteristics in the critical $\alpha$ range. The system installed configuration generates more drag, and a reduction in lift.

Figure 5.7  Lift Coefficient ($C_L$) against incidence angle ($\alpha$) at $Re = 8.9 \times 10^5$; □, = no SJA array; ○, = SJA array installed.

Figure 5.8  Drag Coefficient ($C_D$) against incidence angle ($\alpha$) at $Re = 8.9 \times 10^5$; □, = no SJA array; ○, = SJA array installed.
5.4.2 Chordwise Pressure Distribution

The static pressure coefficient distribution at $\alpha = 14^\circ$ is shown in Figure 5.9.

![Figure 5.9](image)

As the force balance results demonstrate differences at this incidence with the addition of the orifices, the pressure distributions too show differences. At $x/c = 0.4$, the characteristic plateau in the rearward pressures is still generated, but to a greater magnitude. It commences from a further forward tapping location. The separation of the flow with the control system installed therefore remains as a mild trailing edge separation, however to an increased extent compared to the clean wing configuration. This would be expected to be primarily the consequence the macro-scale alterations to the suction surface in the sensitive leading edge region.

5.4.3 Flow-field Measurements

At $\alpha = 14^\circ$, wake profiles were taken at $x/c = 1.25$, at both the semi span, $z/c = 0$ and $z/c = 0.17$, and are presented in Figure 5.10. The wake profile varies across the spanwise extent. The centreline wake has an increased wake height, and alteration to the overall profile compared to the flows further outboard, where the flow control actuators are positioned. This suggests the flow towards the semi span separates further towards the leading edge then that seen at the more outboard region of the jets. The fluctuating velocity $(\bar{u}'^2 / U_\infty^2)$ is shown in Figure 5.11, and highlights that the inboard flow is more unsteady than that at the more outboard point. The result from $\alpha = 13.5^\circ$ at $z/c = 0.17$ is plotted for comparison. The
magnitudes of fluctuation are seen to vary more significantly across the wing, than they do when compared to the slight decrease in $\alpha$. At the semi span fluctuations are significantly larger in magnitude. The difference in wake topology across this spanwise extent suggests there is a complex separated flow state created at these high incidences.

Figure 5.10 Streamwise Velocity ratio ($U/U_\infty$) against vertical position ($y/c$), downstream of the aerofoil at chordwise location ($x/c$) = 1.25; for different planes across the wingspan, ($z/c$), at incidence angle $\alpha = 14^\circ$; $\square$, $z/c = 0$; $\bigcirc$, $z/c = 0.17$

Figure 5.11 Variance of the Streamwise Velocity ratio ($\langle u'^2 \rangle / U_\infty^2$) against vertical position ($y/c$), downstream of the aerofoil at chordwise location ($x/c$) = 1.25; for different planes across the wingspan, ($z/c$) and at different incidence angles; $\bigtriangleup$, $z/c = 0$, $\alpha = -14^\circ$; $\bigcirc$, $z/c = 0.17$, $\alpha = -14^\circ$; $\Delta$, $z/c = 0.17$, $\alpha = 13.5^\circ$.

PIV measurements were taken along the chord at $\alpha = 14^\circ$; the results can be seen in Figure 5.12. Discontinuities in the velocity contours are the result of patching together the discrete
PIV datasets that were obtained separately, using differing camera and light-sheet positions. Upstream of the jets, the peak velocity is \( \sim 1.8 U_\infty \). At the position of the jets, this is \( \sim 1.7 U_\infty \). The jets are therefore introduced into a region of high momentum flow, giving a characteristically low velocity ratio. Towards the trailing edge, due to the adverse pressure gradient, the flow cannot remain attached and development of the separated region at the wall occurs. The flowfield is of an open-type separation, where the flow separates towards the trailing edge, and does not reattach before it has passed the trailing edge.

![Figure 5.12](image)

Figure 5.12: Isocontours of streamwise velocity across the suction surface, at incidence angle \( \alpha = 14^\circ \) at \( Re = 8.9 \times 10^5 \). Contour levels are shown in increments of \( U/U_\infty = 0.2 \).

### 5.5 Defining the Separated Length

The precise point of flow separation in the time-averaged flow can be difficult to characterise. A high fidelity measurement is required to acquire data down to the wing suction surface. Measurements were taken with the non-intrusive optical measurement systems in order to understand the separated flow close to the surface. With the large stand-off distance of the LDA probe requiring a finite offset between the measurement volume and the wing surface, and the flow creating fluctuations in the loading and subsequent dynamic movement of the model, the measurement volume was traversed down to a minimum height of \( \sim 5\text{mm} \) above the wing surface. PIV measurements also present difficulties in acquiring measurements down to the surface, as they are typically subject to light-sheet reflections close to surfaces. Additionally, with the nature of the vector calculation, results are averaged to the centroid of a finite cell size, offsetting results from a physical boundary. Identifying the zero streamwise velocity position in the flow was therefore used to characterise the separated flow region. This would signify the top boundary of a reverse flow region, being the dividing
line between the backwards and forwards flows to the separation bubble. LDA vertical profiles of the boundary layer were collected at points, $x/c = 0.85, 0.90$ and $0.95$ respectively to understand the extent of the separated flow over the wing close to the trailing edge. The results from PIV and LDA measurements, identifying the wall normal height of the zero contour are shown in Figure 5.13. Both sets of measurements are in close agreement. As the PIV data was affected by optical reflections of the light sheet close to the separation point, a second order polynomial is used to extrapolate the zero velocity line taken from point measurements spaced $-0.025 \, x/c$ along the chord. The curve fit gives good agreement to the data, of $R^2 = 0.998$. The separation commences at $x/c = 0.7$. By taking the length scale of the separated flow to be $-0.3 \, x/c$ this allows the reduced frequency $F^+$ of the flow to be quantified.

![Figure 5.13](image)

**Figure 5.13** Separated flow over the suction surface, at incidence angle $\alpha = 14^\circ$ at $Re = 8.9 \times 10^5$, $U/U_c = 0; \, \square = \text{LDA}; \, \bigcirc = \text{PIV}; \, \longrightarrow = P, \, f(\bigcirc)$.

The unsteady surface pressures recorded at the respective positions along the chord and span detailed in Chapter 3, are shown in Figure 5.14 at $\alpha = 14^\circ$. The power spectra $\phi$ of the signal from the pressure transducers are plotted against the non-dimensional frequency, or Strouhal number $St = f_t \, x_{sep} / U_c$, where $f_t$ is the transducer data frequency.

With the flow separating from $-0.7 \, x/c$, it creates a distinct response of a broadband range of $St = -0.15$-0.2. This is similar to the results found for other investigations of stalled wings and bluff-body type wake flows, where a similar shedding frequency of the wake is found (Kotapati et al 2010). This result is indicative of a dominant large scale instability, of Kármán vortex shedding type characteristic to the flow. An energetic mode is seen at a non-dimensional frequency of $\sim 3$-4. With this frequency being an order of magnitude higher than that of the characteristic timescales of the Kármán vortex street, and being a high $St$, i.e. $>1$, it
would be more characteristic of the smaller scale perturbations of the flow associated with the shear layer. Of interest also is that the frequency scales show no evidence of very low frequency shedding; of the order $St = 0.01$, as would be expected from the unsteady stall-cell flow type (Zaman, McKinzie & Rumsey 1989).

![Figure 5.14](image.png)

**Figure 5.14** Static pressure spectra of the flow, at incidence angle $\alpha = 14^\circ$, at $Re = 8.9 \times 10^5$; at different chordwise locations ($x/c$), of $---$, $x/c = 1$; $---$, $x/c = 0.76$.

### 5.6 Discussion

The flow over the wing has shown to demonstrate variation across the span, from measurements taken outboard of those at the semispan. It does not pertain to a quasi-two dimensional flow, which makes the understanding of a controlled instance of the flow potentially complex. There are multiple reasons for the differences in the flow.

The semi span point of the wing has a staggered arrangement of pressure tappings, creating discontinuities to the surface. This creates a disturbance to the flow at the sensitive leading-edge flow at this point, which advances the transition of the flow compared to the outboard and downstream location of the actuator orifices.

The aspect ratio of the wing is $< 3$; this is a further mechanism for creating spanwise flow variations. The centre region flow therefore may be subject to a stall-cell type separation bubble at the wing centreline, that is not as prevalent at the $z/c = 0.17$ outboard position. Results of the pressure spectra are not indicative of this as a dominant shedding mode however (Zaman, McKinzie & Rumsey 1989), but a low-frequency sub harmonic is a potential facet to the flow.
The side wall effects are also a mechanism for generating the change in flow condition in the spanwise direction. A coherent vortical structure will be generated at the wind tunnel wall and wing junction; this will be expected to have a trajectory that will generate an increased upwash effect in the vicinity of the vortex. This region of enhanced upwash may well extend significantly inboard of the wall, and hence create reduced flow separation at the points outboard of the semi span plane, due to the finite nature of the wing flow.

The flow state here can be assumed to be a more challenging situation in which to demonstrate and understand control authority, then that of a more quasi-two dimensional type such as a backward facing step type flow. However, as real world flows on aerofoils are equally or typically more complex then the flow-state here, and then it represents a suitable, repeatable condition to investigate, in order to understand the scope of the active flow control system performance.

5.7 Summary

The unperturbed flow has been characterised for the NACA0015 wind tunnel model. The onset of significant trailing edge flow separation is seen to commence at around the point where $C_{L_{\text{max}}}$ occurs, at $\alpha = 14^\circ$.

A trailing edge separation is the mechanism whereby drag increase and loss of lift occur over the wing. The most robust control of this flow therefore would be generated by placing of the flow control device upstream of the mean separation point. By placing it a significant distance upstream, then effective control over a wide range of incidence range can also be investigated, as the mean separation point alters with incidence.

The effect of the unactuated flow control system compared to a continuous surface configuration of the wing has been of interest to quantify. However in the following chapters all further results will be to compare the system operating to the unperturbed case with the system installed.

The flowfield at $\alpha = 14^\circ$ is seen to show differences in the characteristic of the wake across the width of the area where the flow can be controlled. This presents challenges in order to understand the nature of the control mechanism. The effects of the actuated flow control system will be looked at in the following chapters.
Chapter 6

The Effect of Applying Control

6.1 Introduction to Chapter

In this chapter results are presented when the flow control system is actuated. All tests were performed at a wind speed of 30m/s, giving a chord based Reynolds number of $Re = 8.9 \times 10^5$. Control was applied with the baseline array discussed previously in Chapter 4. This was with the wall-normal configuration of the orifice, and with a normal, sine wave input. It should be noted also that the ‘controlled flow’ is where the system is actuated, having previously been deactivated at the same incidence and wind-speed. ‘Control authority’ in these tests refers to the ability of the device to reattach the uncontrolled separated flow, as opposed to a separation delay mechanism.

6.1.1 Reduced Frequency

With the jet flow operating at 980Hz, a freestream velocity of 30m/s, and the unperturbed flow separated length scale ($x_{sep}$) of ~0.3 x/c, (equivalent to a dimensional value of ~0.13m), this gives a reduced frequency of the flow control of $F^+ = \sim 4.2$ (when $F^+ = f_{x_{sep}}/U_\infty$). Flow control is therefore acting at a characteristically high frequency, $F^+ > O(1)$, in that the timescales of the jet cycle are smaller than those of the separating flow.

6.2 Force Balance Results

Results from the overhead force balance for the overall lift and drag values of the wing are shown in Figure 6.1 and 6.2 respectively. With actuation, $C_{L_{\text{max}}}$ is increased, and occurs at a 1° higher incidence. Lift is increased by +0.07 $C_L$ along with a reduction in drag by -0.023 $C_D$. This generates an improvement in wing efficiency ($C_L/C_D$) from 15.3 to 24.5.
Figure 6.1  Lift Coefficient ($C_L$) against incidence angle ($\alpha$) at $Re = 8.9 \times 10^5$; for $\blacksquare$, = Flow Control on; $\bigcirc$, = Flow Control off.

Figure 6.2  Drag Coefficient ($C_D$) against incidence angle ($\alpha$) at $Re = 8.9 \times 10^5$ for $\blacksquare$, = Flow Control on; $\bigcirc$, = Flow Control off.

6.3  Static Pressure Distribution over the Model

6.3.1  Chordwise Pressure Distribution

Surface pressures are shown in Figure 6.3 for $\alpha = 14^\circ$. Actuation increases the low pressure peak towards the leading edge, and enhances the pressure recovery. The constant pressure plateau towards the trailing edge for the unactuated case is mainly suppressed.
6.3.2 Pressure Spectra at Chordwise Locations

The pressure spectra are shown in Figure 6.4. The broadband response in the unperturbed flow evident at \( x/c = 0.76 \), \( St = \sim 0.2 \) corresponds to the shedding frequency of the wake. This response does not exist in the controlled flow. Another difference can be seen with a distinct peak that appears in the controlled flow at \( St = \sim 4.3 \). This coincides with the frequency of the synthetic jet flow and appears at both chord locations. It demonstrates that a characteristic frequency is introduced to the flow by the SJAs as the flow convects downstream from the point of the control system.
Figure 6.4 Pressure spectra of the flow, at incidence angle $\alpha = 14^\circ$, at $Re = 8.9 \times 10^5$; at different chordwise locations ($x/c$) for:

- ———, Flow Control off, $x/c = 0.76$;
- ————, Flow Control on, $x/c = 0.76$;

- ————, Flow Control off, $x/c = 1$;
- ————, Flow Control on, $x/c = 1$;

6.4 Velocity Profiles

6.4.1 Wake Profiles

Wake profiles for the two flow states are shown in Figure 6.5 at $z/c = 0.17$, $\alpha = 14^\circ$. Control significantly reduces the width of the wake flow coming from the suction surface, and increases the velocity gradient across the region of the flow. The minimum velocity point is shifted upwards in the wake; however the magnitude is roughly the same. The pressure surface flow however has a greater velocity deficit compared to the unactuated case. Looking at the fluctuating velocity profile, shown in Figure 6.6 there is a significant difference between the two states for the flow downstream of the array. Actuation creates a large reduction in the fluctuations towards the edge of the wake. The region of the large peak in the unperturbed case is suppressed, this being in the flow downstream of the suction surface, to the extent that its fluctuation magnitude is no greater the fluctuating flow magnitude from the pressure surface.
Figure 6.5  Streamwise Velocity ratio \( (U/U_\infty) \) against vertical position \( (y/c) \), downstream of the aerofoil at chordwise location \( (x/c) = 1.25 \) at incidence angle \( \alpha = 14^\circ \); at \( Re = 8.9 \times 10^5 \); for □, = Flow Control on, \( z/c = 0.17 \); ○, = Flow Control off, \( z/c = 0.17 \).

Figure 6.6  Variance of the Streamwise Velocity ratio \( (\overline{u'^2} / U_\infty^2) \) against vertical position \( (y/c) \), downstream of the aerofoil at chordwise location \( (x/c) = 1.25 \), spanwise location \( (z/c) = 0.17 \); at incidence angle \( (\alpha) = 14^\circ \) for □, = Flow Control on.; ○, = Flow Control off.

Figure 6.7 shows the fluctuating velocity at the centreline of the wing and at \( z/c = 0.17 \), \( \alpha = 14^\circ \) for the actuated flow. Wake unsteadiness shows similar agreement in the magnitude and peak location, which is a significant reduction compared to the unperturbed flow at the \( z/c = 0.17 \) seen previously. Actuation suppresses fluctuations across the entire span of the control array. The jets either side of the semispan therefore have a spread in terms of control authority in the spanwise extent. They are able to impart an effect on the area of flow between adjacent jets, as the semi span flow is a region at \( \sim 4.5d \) between adjacent jets.
Figure 6.7  Variance of the Streamwise Velocity ratio \((\overline{u'^2}/\overline{U^2}_\infty)}\) against vertical position \((y/c)\), downstream of the aerofoil at chordwise location \((x/c) = 1.25\), at different spanwise locations \((z/c)\); at incidence angles \((\alpha) = 14^\circ\) for \(\square\), = Flow Control on, \(z/c = 0\); \(\bigcirc\), = Flow Control on, \(z/c = 0.17\)

6.4.2    Spanwise Profiles

Spanwise profiles of the non-dimensionalised streamwise velocity, were taken at a height of \(~1d\) above the jet, at \(x/c = 0.12\), and \(x/c = 0.122\), which are in line with, and just downstream of the orifice axis respectively. These are shown in Figures 6.8 and 6.9. All plots show a disturbance in the flow at the vicinity of the centreline tappings, creating a significant reduction in the streamwise flow. The flow between the two chord locations shows a slight reduction in speed over the streamwise length, which corroborates with the adverse pressure gradient. When the flow is actuated, low-speed spikes are generated at the jet locations. These are only significant at the point downstream of the orifice, where the efflux has developed in the time-averaged flow. Little response is evident from the jet positioned at \(z/c = -0.05\). The magnitudes of the velocity deficit downstream of the jets vary across the span. It could be expected that small variations in the cross-flow conditions across each jet would give rise to differing actuator responses, given the sensitivity of the device to boundary conditions. In such a cross flow state, the creation of a homogenous jet effect across the entire span of an array is a complex issue. The levels of flow unsteadiness are significantly enhanced in the region downstream of the jets, as seen in the fluctuating velocity, shown in Figures 6.10 and 6.11. Fluctuation magnitude is in good agreement with the velocity deficit downstream of the jet. The strength of the jet flow therefore directly affects the intensity of the mixing effect downstream of the jet.
Figure 6.8  Streamwise Velocity ratio ($U/U_\infty$) against spanwise position ($z/c$), at different chordwise locations ($x/c$), at incidence angle $\alpha = 14^\circ$; at $Re = 8.9 \times 10^5$; for \(\longrightarrow\), $x/c = 0.12$, Flow Control off; \(\cdot\cdot\cdot\longrightarrow\cdot\cdot\cdot\longrightarrow\cdot\cdot\cdot\), $x/c = 0.122$, Flow Control off.

Figure 6.9  Streamwise Velocity ratio ($U/U_\infty$) against spanwise position ($z/c$), at different chordwise locations ($x/c$), at incidence angle $\alpha = 14^\circ$; at $Re = 8.9 \times 10^5$; for \(\longrightarrow\), $x/c = 0.12$, Flow Control on; \(\cdot\cdot\cdot\longrightarrow\cdot\cdot\cdot\longrightarrow\cdot\cdot\cdot\), $x/c = 0.122$, Flow Control on.
Figure 6.10  Variance of the Streamwise Velocity ratio \( (\overline{u'^2} / U_{\infty}^2) \) against spanwise position \((z/c)\), at chordwise location \(x/c = 0.12\), at incidence angle \(\alpha = 14^\circ\); at \(Re = 8.9 \times 10^5\); for;

\[\begin{align*}
- & \quad \square, x/c = 0.12, \text{Flow Control off}; \\
\cdots & \quad \circ, x/c = 0.12, \text{Flow Control on}.
\end{align*}\]

Figure 6.11  Variance of the Streamwise Velocity ratio \( (\overline{u'^2} / U_{\infty}^2) \) against spanwise position \((z/c)\), at chordwise location \(x/c = 0.122\), at incidence angle \(\alpha = 14^\circ\); at \(Re = 8.9 \times 10^5\); for;

\[\begin{align*}
- & \quad \square, x/c = 0.122, \text{Flow Control off}; \\
\cdots & \quad \circ, x/c = 0.122, \text{Flow Control on}.
\end{align*}\]

6.5 Flow-field Measurements

PIV measurements were made along the chord at \(\alpha = 14^\circ\), \(z/c = 0.17\). Discontinuities in the velocity contours are the result of patching together the discrete PIV datasets that were obtained separately, using differing camera and light sheet positions. The mean streamwise velocity is shown in Figure 6.12. At the chord location of the jets the streamwise velocity is
\( \sim 1.7U_\infty \), which is very similar to the result seen for the unactuated flow (see figure 5.10). From the position of \( x/c > \sim 0.2 \) however, differences in the flows become apparent. The wall-normal gradient becomes far greater with actuation as the boundary layer growth is significantly reduced. The onset of separated flow that was seen in the unactuated case at \( x/c = >0.7 \) is therefore avoided when actuation is applied. The zero velocity contour is not measured at a point \( x/c = < 0.95 \). With the flow over the chord understood in the time-averaged form, further measurements in the vicinity of the jet flow as a function of the jet cycle were considered.

![Isocontours of streamwise velocity across the suction surface, at a spanwise position of \( z/c = 0.17 \), incidence angle \( \alpha = 14^\circ \) and \( Re = 8.9 \times 10^5 \) for control applied. Contour levels are shown in increments of \( U/U_\infty = 0.2 \).](image)

Measurements of the flow were taken along the chord between \( 0.12 < x/c < 0.42, z/c = 0.17 \) in the area of flow directly downstream of the jet. Phase averaged PIV was taken in the same manner of acquisition triggering as described in Chapter 4. Measurement points throughout the phase allow coherent structures in the flow to be identified, in terms of their movement throughout the field of view, and how they dissipate. The streamwise velocity at 4 equi-spaced points throughout the phase is shown in Figure 6.13. It is clear that a low speed bulge is generated across the cycle of the jet actuation. This is seen to move downstream, following the curvature of the wing. The area of flow investigated shows successive structures convecting downstream. The spacing, \( \lambda \), of these is \( \sim 8d \) and since \( f\lambda/U_\infty \sim 1 \), the convective velocity of the structure is closely approximated by the freestream velocity.
For the same region of the flow, phase-locked spanwise vorticity $\zeta c/u_\infty$ is shown in Figure 6.14. Coherent regions of vorticity are introduced by the jet in to the cross-flow. The centre of each successive structure is seen to coincide with the centre of the low speed bulges identified in the phase locked flow. As the structure develops downstream from $x/c = \sim 0.24$, a discrete region of vorticity develops above a larger concentration close to the wall. With the measurements being on the centreline of the jet, this could indicate a stretched hairpin vortex structure developing directly downstream of the orifice. Towards the far end of the field of view, the head of the vortex structure starts to diminish in strength. This could due to dissipation, or be the result of a spanwise migration out of the plane of measurement. The vorticity field reveals that the streamwise velocity fluctuations are a result of this addition of a coherent vortical structure to the flow. With the local velocity at the orifice point seen to be $\sim 1.7U_\infty$, this creates a characteristically low velocity ratio for the jet. The structures that are generated remain close to the wall during their development. The height of the structure is $\sim 2d$ at their highest point. The jet strength effects the development of the vortical structures. However, this jet flow has been of sufficient strength to generate a structure that persists $>20d$ along the flow.
The periodic streamwise velocity components $\tilde{u} / U_\infty$ and $\tilde{v} / U_\infty$ are shown in Figures 6.15 and 6.16. Low speed bulges in the flow are seen in the phase averaged velocity field, but the jet flow generates alternate regions of positive and negative fluctuation about the mean flow. The low speed streamwise regions are in agreement in spatial position with those in the phase averaged flow. The convective velocity of the perturbations is closely linked to the freestream velocity. The fluctuations therefore appear to be a direct result of the vortical structures generated in the flow, as opposed to fluctuations linked to other mechanisms of the jet, such as acoustic disturbances due to the jet cavity pressure fluctuations.

Looking at the periodic vertical velocity component in Figure 6.16, there are distinct fluctuations seen to the flow here also. These components seem to be associated with the opposite sign in the streamwise fluctuations. This alternating motion in the fluid has the mechanism of bringing fluid at greater wall normal distance, with a higher streamwise velocity, towards the wall. This mechanism of enhanced mixing would be expected to be the manner in which the adverse pressure gradient seen in Figure 6.3 is altered, where the pressure recovery up to $x/c = \sim 0.4$ is improved, and hence the separation seen further downstream in the unperturbed flow from $x/c = 0.7$ is avoided.
Figure 6.15  Contours of $\tilde{u} / U$ across the suction surface, at spanwise position $z/c = 0.17$, for flow control on, iso-contour lines at ±0.1 at phases of a) 0, b) $\pi/2$, c) $\pi$, and d) $3\pi/2$.

Figure 6.16  Contours of $\tilde{v} / U_\infty$ across the suction surface, at spanwise position $z/c = 0.17$, for flow control on, iso-contour lines at ±0.1 at phases of a) 0, b) $\pi/2$, c) $\pi$, and d) $3\pi/2$. 
The downstream behaviour of the fluctuation in the flow towards the point of separation was not tracked with further measurements due to the significant amounts experimental data capture required to interrogate the flowfield to that point on the chord. However, the measurements of the flowfield along with the pressure distribution results imply this enhanced mixing is the dominant mechanism that generates flow reattachment when the control is applied.

6.6 Discussion of Actuated Flow

Actuation has been shown to generate improvements to the wing efficiency at angles of attack where trailing edge separation occurs. The actuators generate periodic fluctuations that delay the trailing edge separation, as enhanced mixing in the immediate region of adverse pressure gradient is created, generating increased momentum redistribution levels in the flow. These are of sufficient magnitude to keep the flow attached. It is suggested this mechanism is subtly different to the lower frequency modes investigated by Zhong & Zhang (2013), in that it depends less on the natural flow frequencies resulting from the separation, in order to impart control. Being a characteristically high frequency, the mechanism is time invariant to the separated flow.

Of note is that the excitation created at a significant distance upstream from the unactuated flow separation point. The separated region for the unactuated flow is established by \( \sim 50d \) downstream from the location of the jets. This length scale is greater than that investigated previously with round jets (Zhong & Zhang 2013). However a similar jet efflux and perturbation mechanism was identified for the control imparted in those experiments. Previous work looked at characteristically lower actuation frequencies, where the interaction of the separated flow with the jet flow was identified. With a low frequency jet flow, the shear layer flow was noted to have an oscillation of its wall normal height. This was due to jet flow frequency coinciding with the shedding frequency of the flow field. With the actuation strategy here acting at \( O(>1) \) compared to the timescales of the separated flow, the jet flow here was operating on different timescales to the large scale instabilities of the naturally separated flow.

Indeed, with increased jet frequency, the shear layer effect diminished in the experiment of Zhong & Zhang (Zhong & Zhang 2013). However, the higher frequency actuation of that work was still lower than the non-dimensional frequency used here. However the results
comparison suggests the actuation mechanism still proves effective as $F^+$ increases. The idea of frequency sensitivity of the flow however will be examined further in the following chapter, where an amplitude modulation strategy is investigated.

Returning to the results of the actuation strategy used here, with the spanwise measurements of the streamwise velocity downstream of the jet in the time-averaged flow, the results show a velocity deficit in the flow behind each discrete jet. It therefore would be feasible that the jet flow possesses the same time-averaged effect of that of a physical boundary in the flow. If the high frequency, time invariant nature of the jet flow causes an effective bluff-body ‘wake-like’ flow, from the successive low speed bulges that are part of the mixing enhancement mechanism, then similar structures maybe generated by an obstruction.

Hairpin vortices have been understood to be coherent structures that develop from the synthetic jet flow. The work of Rostamy et al (2012), has previously studied the flowfield downstream of a wall mounted finite cylinder. A complex flow structure is created, comprising of a tip vortex structure and a Kármán vortex formation. In addition however, the flow can be characterised by the ground plane flow that generates a horseshoe vortex system at the wall junction. This base vortex system therefore may act in a similar manner to the synthetic jet flow on a downstream separating flow. Based on this hypothesis, an array of finite vertical cylinders was installed and tested in the same manner as the active flow control system.

### 6.7 Passive Control

For the tests, an array of finite length cylinders of height $h = 4d \pm 0.25d$ were used initially, and are shown in Figure 6.17. The cylinders were of diameter $d = 5\text{mm}$, and located in the same chordwise and spanwise locations as the actuator orifice. A configuration of height $h = 2d \pm 0.25d$ was also tested. It had been found by Adaramola et al (2006) that two distinctly different flow regimes occur at aspect ratios of $h = <3d$ and $h = 3d$, where the base vortex structure will not develop at lower aspect ratios. The two heights therefore should develop different flows to assess the effect of any near-wall coherent structures in the flow.
Tests were conducted at a chord Reynolds number of $Re = 8.9 \times 10^5$ as previously and at $\alpha = 14^\circ$. Based on the diameter of the cylinder, the Reynolds number of the flow was $\sim 1 \times 10^4$. With the chordwise location of the array being in the area of high flow speed, the cylinders would be partially submerged in the boundary layer for both configurations.

6.7.1 Force Balance Results

Force-balance results were recorded at $\alpha = 14^\circ$, and are shown in Table 6.1. With cylinders, compared to the jets, an increase in drag is generated, and a decrease in lift. The two results together indicate that no enhancement to the suction surface flow is generated with the physical boundary. The cylinders are counterproductive. The addition of the physical boundary will inevitably generate additional drag; however, any reduction to the trailing edge separation may offset this to some extent, if favourable flows are generated from the cylinders. The significant increase in drag and decrease in lift for both cylinder heights suggests that any momentum addition generated by the cylinders is not sufficient to influence the separated flows far downstream.

<table>
<thead>
<tr>
<th>Cylinder Height</th>
<th>$\Delta C_L$</th>
<th>$\Delta C_D$</th>
<th>$\Delta C_L/C_D$</th>
</tr>
</thead>
<tbody>
<tr>
<td>$2d$</td>
<td>-0.089</td>
<td>+0.039</td>
<td>-9.40</td>
</tr>
<tr>
<td>$4d$</td>
<td>-0.184</td>
<td>+0.046</td>
<td>-10.91</td>
</tr>
</tbody>
</table>

Table 6.1 Summary of forces for the cylinder array controlled flow.
6.7.2 Static Pressure Distribution over the Model

The pressure distributions around the model configurations were measured. The result of the controlled flow with the SJA system is also shown in Figure 6.18 for comparison. The cylinders are seen to have the inverse effect of the jet fluidic mechanism. The pressure recovery is significantly reduced downstream of the array, which results in a constant pressure plateau becoming apparent from \( x/c > 0.25 \). This suggests the separated length of the flow is increased compared to the non-perturbed case. The suction peak is reduced in magnitude. The cylinder flow acts in the manner of a disturbance to the flow that creates adverse effects, as opposed to the mechanism seen with the jets whereby the upstream and downstream flowfield have improved characteristics.

![Graph showing coefficient of static pressure against chordwise location](image)

Figure 6.18 Coefficient of Static Pressure \((C_p)\) against chordwise location \((x/c)\) over the pressure and suction surfaces of the aerofoil, at incidence angles \((\alpha) = 14^\circ\); for \(\square\), Flow Control on; \(\bigcirc\), \(h = 2d\); \(\Delta\), \(h = 4d\).

6.7.3 Wake Profiles

Wake profiles were collected at \(x/c = 1.25\). Figure 6.19 shows the wake profiles at the two cylinder heights, at both the semi span and the \(z/c = 0.17\) position. The cylinders act to significantly enhance the height of the wake when compared to the results seen in Figure 6.5 for the jet controlled flow. The separation bubble over the suction surface is increased in length, in so much that flow for the taller cylinder configuration even reaches a reversed flow condition at the wake profile plane. The shorter cylinder height generates the lower velocity deficit; however the overall wake heights of both configurations are approximately the same. The two configurations produce close agreement in the fluctuating velocity profiles for the taller cylinder cases shown in Figure 6.20, indicating consistent fluctuating behaviours to the
flow across the entire span of the array. The similar results for each height however suggest both forms of the cylinder wake structure generate similar, ineffective perturbations in the flow in the vicinity of the cylinder.

Figure 6.19  Streamwise Velocity ratio \((U/U_\infty)\) against vertical position \((y/c)\), downstream of the aerofoil at chordwise location \((x/c) = 1.25\); for spanwise position \((z/c) = 0\), at incidence angle \(\alpha = 14^\circ\), for \(\Box, h = 4d\), \(\bigcirc, h = 2d\).

Figure 6.20  Variance of the Streamwise Velocity ratio \((\overline{u'^2}/U_\infty^2)\) against vertical position \((y/c)\), downstream of the aerofoil at chordwise location \((x/c) = 1.25\); for spanwise position \((z/c) = 0\), at incidence angle \(\alpha = 14^\circ\), for \(\Box, h = 4d\), \(\bigcirc, h = 2d\).
6.7.4 Flow-field Measurements at the Array

To understand the flow in the vicinity of the cylinder further, PIV data was collected in the centreline plane of the cylinder at $z/c = 0.17$. Only data for the time-averaged flow was collected. For the $4d$ height cylinder, iso-contours of $U/U_{\infty}$ and $V/U_{\infty}$ are shown in Figure 6.21. It is apparent that the flow immediately downstream of the cylinder forms a region of separated flow. Within the spatial region measured, the flow is not seen to reattach, and a stagnated flow region extends across the chordwise length to $x/c = 0.21$. Looking at the vertical velocity $V/U_{\infty}$ there is a small increase in the flow speed towards the wing surface downstream of the cylinder tip face, generated by the flow accelerating around the tip. There are not seen to be any significant disturbances to the flow at a point closer to the wing surface however, indicating the wake near to the wing surface on the centreline is not perturbed to a significant extent. Rostamy et al (2012) found a distinct change in the downstream flow with a cylinder of $3d$ height, due to the lack of a base vortex structure. The effect of this was a reduction in the upwash in the lower region of the flowfield.
Figure 6.21 Isocontours of streamwise velocity downstream of the cylinder on the suction surface, for spanwise position $(z/c) = 0.17$, at incidence angle $\alpha = 14^\circ$. Contour levels are shown in increments of $U/U_\infty = 0.25$, for $h = 4d$ cylinder at $z/c = 0.17$.

Isocontours of vertical velocity downstream of the cylinder on the suction surface, for spanwise position $(z/c) = 0.17$, at incidence angle $\alpha = 14^\circ$. Contour levels are shown in increments of $V/U_\infty = 0.25$, for $h = 4d$ cylinder at $z/c = 0.17$.

Figure 6.22 Isocontours of streamwise velocity downstream and upstream of the cylinder on the suction surface, for spanwise position $(z/c) = 0.17$, at incidence angle $\alpha = 14^\circ$. Contour levels are shown in increments of $U/U_\infty = 0.25$, for $h = 2d$ cylinder at $z/c = 0.17$.

Isocontours of vertical velocity downstream and upstream of the cylinder on the suction surface, for spanwise position $(z/c) = 0.17$, at incidence angle $\alpha = 14^\circ$. Contour levels are shown in increments of $V/U_\infty = 0.25$, for $h = 2d$ cylinder at $z/c = 0.17$.

In the $2d$ cylinder tests the reattachment of the flow is seen to be further upstream than for the taller cylinder. The size of the separated region is reduced, with the flow reattaching at the point $x/c = \sim 0.2$. A greater downwash of the shorter cylinder case is seen in the vertical velocity plot in the flow downstream of the cylinder. This may be indicative of the lack of the base vortex system and the associated upwash. Considering the overall force results, the $2d$ case has a lower lift alleviation to the flow compared to the $4d$ case, therefore the lower aspect ratio form of cylinder is the less counterproductive form of passive disturbances tested.
However, there is no indication with the time averaged flow results here that the passive disturbance to the suction surface flow via a cylinder, can have the same manner of control authority as that demonstrated with the fluidic mechanisms of the synthetic jets. The jet flow introduced to the cross-flow enhances wing efficiency, while the passive disturbance introduced from the cylinder, is counterproductive to wing performance.

6.8 Discussion of Passive Control

Active flow control is shown to generate improvements in the efficiency of the wing. This is achieved by a periodic mechanism that redistributes momentum in the boundary layer early in the development of the adverse pressure gradient. The manner in which this is achieved is understood to be relatively insensitive to the natural frequencies of the flow, due to the fact that the jet flow operates on a cycle timescale shorter than that of the dominant frequency to the unperturbed flow. A passive means of flow perturbation (via finite cylinders) in the same chord location however imparts significant disturbance to the flow in the near flow field, which is globally counterproductive.

The cylinder dimension tested, being the same diameter as the jet orifice, equates to a Strouhal number $St = f d/U_\infty$ of ~0.16 when the frequency $f$ is taken to be the frequency of the jet. In the work by Adaramola et al (2006), the finite cylinder configuration was seen to have $St = 0.16$ so a similar wake shedding timescale across the experiments may be expected. However, the active control mechanism of disturbance to the boundary layer in the zero-net-mass-flux manner of the jet, elicits a different response to that of the addition of vortex shedding of a passive, cylinder flow. It suggests the coherent structure of the synthetic jet flow is a key facet of the successful control authority; it does not just act as a time-averaged, velocity deficit in the cross-flow. An optimal, discrete mixing effect is generated from each jet cycle, which enhances momentum transfer and generates the flow reattachment. This is not achieved with a cylinder wake. Although the investigation into passive physical boundaries has not considered further the effect of different geometric configurations, and the optimal form of these, the results here do identify that the active control system is able to impart robust control authority, in that it can generate control in a flow-state where an in-depth understanding of the mechanism generated by passive means would be required to reach the same desired result. This can be thought of as a desirable result, as the optimisation
of placement of developed passive vortex generator systems, and accurate modelling of this, is still a challenging engineering task (Godard. & Stanislas 2006; Zastawny 2014).

6.9 Summary

In this chapter the effect of the flow control system has been investigated. Actuation has been demonstrated to provide effective separation control, enhancing overall efficiency.

The momentum transfer mechanism of the active control, although relatively time invariant, is not able to be reproduced by a passive, physical obstruction on the flow, attempting to work on a similar spatial scale. The two configurations are seen to elicit opposite results. The unsteady jet flow generates a flow disturbance that is beneficial to the cross-flow and can impart momentum transfer a significant region downstream of the excitation location. This is due to the coherent nature of the vortical structure that is introduced to the flow. A cylinder flow however is seen to be an opposing type of mechanism, in that a non-optimal set of discrete perturbations are generated in the flow by the cylinders. Although these are a significant disturbance to the near-field flow, this is counterproductive downstream, where the momentum transfer to the boundary layer is required.

As effective separation control has been demonstrated here, to further develop practical control strategies, two actuation methods to produce system efficiency improvements will be investigated in the next chapter.
Chapter 7

The Performance of Efficiency Improvement Strategies

7.1 Introduction to Chapter

In this chapter results are presented of differing actuation strategies with the flow control system. Two distinctly different methods of altering the form of the perturbation generated have been considered. However, both had the same concept, to attain the same, or improved control authority, when compared to the actuation strategy looked at in the previous chapter. This actuation will be referred to as the baseline strategy throughout the proceeding results.

The two concepts possess the potential to offer increased efficiency of the overall system for certain flow states. The first method was to use an amplitude modulation strategy for the driving signal. The second was to look at altering the orientation of the jet efflux to the cross flow. All-tests were conducted in freestream conditions with the same model location in the wind tunnel as used in all previous tests in Chapter 6.

7.1.1 Amplitude Modulation

Amplitude modulation creates a periodic reduction of the amplitude of the driving signal, at a frequency \( f_m \) that differs to that of the sine-wave waveform used throughout, referred to as the carrier frequency \( f_c \). The carrier frequency of 980Hz was the same as was used with the unmodulated strategy used in the tests in Chapter 6. Modulation creates two distinct perturbation frequencies in the flow. By the very nature of reducing the input voltage to the disc over a time period, when compared to the baseline strategy, the power input to the disc is reduced overall. A number of previous studies have considered amplitude modulation, giving credence to its potential (Melton et al 2005, 2006). However, the response in the flow due to such an actuation strategy is still poorly understood, so further investigative experiments are required. By using a modulation strategy, a perturbing frequency can be effectively
decoupled from the dynamic response of the actuator. As a piezo-disc based SJA relies on a relatively narrow-band frequency response for maximum attainable velocities, signal modulation allows a much more wide-band use. It is therefore of interest to understand the sensitivity of the control authority to this manner of perturbing the flow. Understanding of the effectiveness of modulated signals is of benefit when considering the development of large scale arrays of actuators for use in dynamic applications, such as systems on vehicles. Power efficiency is a key requirement where a ‘break-even’ performance criteria would be required to justify their use, so a minimal power input is highly desirable.

7.1.2 Injection Angle

The angle at which the orifice is orientated to the cross-flow is a key parameter in the design of an SJA. The development of the jet structure is defined in part due to the cross-flow velocity. In Chapter 6, with a wall-normal orientation of the orifice, the jet structure develops in a manner that creates a successful control mechanism. By making the injection angle non-wall-normal, then it is likely the flow will behave in a different manner. Counterstreamwise injection has been investigated previously in a numerical study of a circular orifice (Lardeau & Leschziner 2011). The method was shown to be more effective when compared to a streamwise angled orifice or a wall normal-orientation, in that the jet generated more vigorous mixing in the near field creating greater spanwise control authority. By having a jet flow that is more effective in the spanwise extent, an array of jets could be formed from fewer jets, enhancing the power efficiency of the overall array. The wall-normal configuration investigated in the previous chapter used a spacing of 5d; the effect of the injection angle will be investigated for the same jet spacing.

7.2 Amplitude Modulation

7.2.1 Force Balance Results

All tests were performed at a wind speed of 30m/s, giving a chord based Reynolds Number of $Re = 8.9 \times 10^5$. A range of frequencies was tested by reducing the modulating frequency by $x/2^n$ down from the 980Hz ‘baseline’ value to a lowest value of a frequency of $O(1)$Hz, equating to an $F^+_{m}$ of $O(10^{-2})$. $F^+_{m}$ is calculated taking $x_{sep}$ as the unperturbed separated length, however $f$ is now taken as the modulation frequency, $f_m$.

In the same manner as the tests in the previous chapter, force-balance data is presented at a range of angles of attack, shown in Figures 7.1 and 7.2. The results of the unperturbed and
baseline perturbed flows are also shown for comparison. Certain frequencies are shown to be more efficient than others. However, no frequency is counter-productive, in that the lift is increased, and the drag reduced compared to the unperturbed case for all $F_m$ states.

Figure 7.1 Lift Coefficient ($C_L$) against incidence angle ($\alpha$) at $Re = 8.9 \times 10^5$ for $\ldots$, $F^* = 4.2$; $\ldots$, $F_m^* = 2.1$; $\ldots$, $F_m^* = 1.05$; $\ldots$, $F_m^* = 0.53$; $\ldots$, $F_m^* = 0.26$; $\ldots$, $F_m^* = 0.13$; $\ldots$, $F_m^* = 0.06$; $\ldots$, $F_m^* = 0.03$.

Figure 7.2 Drag Coefficient ($C_D$) against incidence angle ($\alpha$) at $Re = 8.9 \times 10^5$ for $\ldots$ = Flow control off; $\ldots$, $F^* = 4.2$; $\ldots$, $F_m^* = 2.1$; $\ldots$, $F_m^* = 1.05$; $\ldots$, $F_m^* = 0.53$; $\ldots$, $F_m^* = 0.26$; $\ldots$, $F_m^* = 0.13$; $\ldots$, $F_m^* = 0.06$; $\ldots$, $F_m^* = 0.03$.

Frequencies of $\sim 0.1 < F_m^* < \sim 0.5$ generate a slight increase in lift (compared to pure sine-wave excitation) at $\alpha = 14^\circ$. This also occurs at the higher angles of attack. Frequencies of $\sim 1 < F_m^* < \sim 2$ however produce a reduction in lift enhancement.
With regards to the effect on drag, all modes of actuation are beneficial compared to the unactuated condition. However, when results are considered against the baseline actuation, \(1 < F^+ m < 2\) is least effective. At the higher angles of attack, the differences are more significant. At \(\alpha = 15^\circ\), two distinct modes are created. One is apparent where \(1 < F^+ m < 2\) and \(F^+ m = 0.03\), where frequencies are least effective. The other frequencies all compare closely, and generate similar performance to the baseline actuation. At \(\alpha = 16^\circ\), the baseline actuation is more effective than all \(f_m\) modes in reducing drag. These time-averaged force results show that there is a frequency dependency to the efficacy of control authority. However, there is no indication that counter-productive modes can be generated with a frequency modulation strategy for this flow state. Variation in the efficacy of certain modulation frequencies is apparent with change in angle of attack however. This change demonstrates that the frequency dependency is non-linear, complex to understand, and depends on the unperturbed flow-state.

By considering the variance in the force results, further understanding of the results becomes clear. The results of the variance of the force are shown in Figures 7.3 and 7.4 respectively. At \(\alpha = 14^\circ\), without control applied, the largest variations of lift are seen. All modes of actuation create decreases in the magnitude of the fluctuation of lift found for the unperturbed, separated flow. \(\alpha = 14^\circ\), the results show more complex trends, particularly as the frequency of modulation is reduced. Actuation frequencies \(0.03 < F^+ m < 0.06\) are seen to be counter-productive, in that they create greater unsteadiness to the lifting force then experienced for the unactuated flow. This would not be a desirable characteristic for a practical flow control system in a dynamic environment, i.e. on an air or ground vehicle. At \(\alpha = 16^\circ\), where the separated flows are generated for the unperturbed flow, the baseline condition of unmodulated actuation is significantly less unsteady than the other frequencies tested however.
Figure 7.3 Lift force variance $\sqrt{(L')^2/L}$ against incidence angle ($\alpha$) at $Re = 8.9 \times 10^5$ for: $-\cdots- = \text{Flow control off;}$ $\rightarrow F^+ = 4.2; \rightarrow F'_m = 2.1; \cdots \cdot \cdot \cdot \cdot \cdot \cdot \cdot, F'_m = 1.05; \triangle \cdots \cdot \cdot \cdot \cdot, F'_m = 0.53; \cdots \cdot \cdot \cdot \cdot \cdot, F'_m = 0.26; \square \cdots \cdot \cdot \cdot \cdot, F'_m = 0.13; \bigcirc \cdots \cdot \cdot \cdot \cdot, F'_m = 0.06; \star \cdots \cdot \cdot \cdot \cdot, F'_m = 0.03.$

Figure 7.4 Drag force variance $\sqrt{(D')^2/D}$ against incidence angle ($\alpha$) at $Re = 8.9 \times 10^5$ for: $-\cdots- = \text{Flow control off;}$ $\rightarrow F^+ = 4.2; \rightarrow F'_m = 2.1; \cdots \cdot \cdot \cdot \cdot \cdot \cdot \cdot, F'_m = 1.05; \triangle \cdots \cdot \cdot \cdot \cdot, F'_m = 0.53; \cdots \cdot \cdot \cdot \cdot \cdot, F'_m = 0.26; \square \cdots \cdot \cdot \cdot \cdot, F'_m = 0.13; \bigcirc \cdots \cdot \cdot \cdot \cdot, F'_m = 0.06; \star \cdots \cdot \cdot \cdot \cdot, F'_m = 0.03.$
Summarising the force results, there are potential benefits from actuation at lower frequencies. However, these come with the potential disadvantage of greater unsteadiness compared to the baseline actuation with only a small change in the unperturbed flow state. ~1 < \( F^+ \) < ~2 is generally shown to be the least efficient frequency range. Actuation at \( F^+ \) < ~2 demonstrates more complex behaviour to summarise for differing flow states, as Figure 7.5 demonstrates when the efficiency, L/D is shown over the different \( \alpha \) range investigated. The unmodulated baseline condition could be considered the most consistent actuation strategy, considering efficiency and unsteady performance across the different \( \alpha \) range investigated. However, the energy input reduction from amplitude modulation means it could prove to be a desirable concept to develop in certain flow control contexts. It is therefore of interest to conduct investigations of the flowfield, in order to understand the control authority mechanisms. However, given the large range of configurations of \( F^+_m \) with \( \alpha \), only a single configuration was considered in further tests, of \( \alpha = 14^\circ \) with \( F^+_m = ~0.3 \). With the modulated perturbation being greater than one order of magnitude different to the carrier frequency, then a significant alteration to the control mechanism maybe identifiable.
7.2.2 Static Pressure Distribution over the Model

7.2.2.1 Chordwise Pressure Distribution

The flow at the semi span plane for the modulated actuation generates a pressure distribution similar to the baseline case, as shown in Figure 7.6. This corroborates the force balance results for the two actuation strategies. The trailing edge separation occurring in the unperturbed flow is suppressed to a similar extent for both actuated flows.

![Graph](image)

Figure 7.6 Coefficient of Static Pressure ($C_p$) against chordwise location ($x/c$) over the pressure and suction surfaces of the aerofoil, at incidence angle ($\alpha$) = 14°; for: □, $F^+$ Control; ○, Flow Control off; Δ, $F^+_{m=0.3}$ Control.

7.2.2.2 Pressure Spectra at Chordwise Locations

The unsteady pressure spectra at $x/c = 0.76$ and the trailing edge for the differing strategies actuation are shown in Figure 7.7. For the modulated result, a substantial peak is generated at $St = \sim 0.3$, evident at both chord locations. This signifies that the jet flow modulation does manifest as a characteristic perturbing mode in the flow towards the trailing edge of the wing. A smaller peak is seen at $F^+_{m} = \sim 4$ which demonstrates the carrier frequency perturbation is still evident within the flow. The baseline and modulated actuation spectra both show similar energy contents when compared to the unactuated flow. Both act to reduce the broadband response apparent in the unperturbed flow at $St = 0.2$, which is a result of suppression of the trailing edge separation.
7.2.3 Flow-field Measurements

PIV measurements were made along the chord at $\alpha = 14^\circ$. Discontinuities in the velocity contours are the result of patching together the discrete PIV datasets that were obtained separately, using differing camera and light sheet positions. For the amplitude modulated actuation, the time-averaged flowfield was collected in the manner previously detailed for the suction surface of the wing. The results show that in a similar manner to the baseline actuation, the trailing edge separation occurring with the unperturbed flow is mainly suppressed. The mean streamwise velocity can be seen in Figure 7.8.
Figure 7.8  Isocontours of streamwise velocity across the suction surface, at a spanwise position of $z/c = 0.17$, incidence angle $\alpha = 14^\circ$ and $Re = 8.9 \times 10^5$, for control applied at $F_m = 0.3$. Contour levels are shown in increments of $U/U_{\infty} = 0.2$.

The wall normal velocity gradient is similar to the baseline actuation, shown in Figure 6.12 in Chapter 6. The result is interesting. It implies that in the vicinity of the separated flow (for the unactuated case), sufficient momentum has been introduced by the control system to invoke flow reattachment. This is despite the differing control strategies. Modulation generates a jet flow that imparts a reduced momentum coefficient in a time-averaged form (over many cycles of the carrier frequency), compared to the baseline actuation mode.

The periodic velocity components were calculated as detailed in Chapter 3. Unlike the baseline signal however, phase locking was synchronised to the phase of the modulation frequency. As discussed in Chapter 4, the flowfield of the jet efflux is not phase averaged by the suction and blowing cycle of the jet when data are captured in this manner. The carrier frequency cycle to the jet flow is therefore not captured in the data, although any cyclic motion or large coherent structures in the flowfield occurring at the modulation frequency will be captured.

To investigate fluctuations in the flow over the modulation phase, boundary layer profiles are a useful descriptor. Phase-averaged boundary layer profiles were investigated at positions of $x/c = 0.75$ and 0.85 and are shown in Figures 7.9 and 7.10 respectively. Each profile at an instance in the phase is plotted alongside the $\phi = 0$ profile for comparison. For the unperturbed flow, separation was evident at both chord locations. Due to limitations of the measurement area at $x/c = 0.85$, the entire height of the boundary layer profile could not be
collected. For both chord locations, data are normalised by $\delta$ at $x/c = 0.75$. The boundary layer is of an increased height at $x/c = 0.85$, compared to the upstream location.

At $x/c = 0.75$, only small fluctuations in the flow are generated throughout the phase. However, at $y/\delta > 1$, oscillations about the $\varphi = 0$ profile can also be seen. Understanding if these motions are directly related to the perturbation is complex. However, they could be indicative of a large scale motion occurring in the flow over the modulation time duration. The flow at $x/c = 0.85$ experiences greater fluctuations than seen upstream, particularly in the upper part of the boundary layer. However, as with the upstream location, a distinct, cyclic motion to the flow is not recognisable.
\[ \phi = 0 \]
\[ \phi = 2\pi/8 \]
\[ \phi = 4\pi/8 \]
\[ \phi = 6\pi/8 \]
\[ \phi = 8\pi/8 \]
\[ \phi = 10\pi/8 \]
\[ \phi = 12\pi/4 \]
\[ \phi = 14\pi/8 \]

Figure 7.10  Boundary layer profiles at chordwise location \( x/c = 0.85 \), at incidence angle \( \alpha = 14^\circ \) and \( Re = 8.9 \times 10^5 \), for various points in the jet phase, where \( \cdots \), \( U_\phi/U_\infty \) for condition \( F_{m=0.3}^* \), \( \phi = 0 \);
\( \cdots \cdots \cdots \), \( U_\phi/U_\infty \) for condition \( F_{m=0.3}^* \), \( \phi = \) respective point in phase.

The two boundary layer profile measurement points are separated by a distance corresponding to \( \sim 9d \) for the jet length scale. In the measurements of the actuated flow in Chapter 6, (Figure 6.15, 6.16), a similar separation length between successive fluctuations in the flow was identified. The results here however, do not show evidence of distinct fluctuations moving downstream in a cyclic manner, between the two points in the flow.

The variation of the periodic streamwise velocity for each point in the modulation phase can identify the wall-normal extent, and ‘strength’ of any fluctuation to the flow on the timescales of the modulation. \( \tilde{u}/U_\infty \) at \( x/c = 0.75 \) and \( x/c = 0.85 \) are plotted in Figures 7.11 and 7.12 respectively.

At \( x/c = 0.75 \), in comparison to the zero point in the phase, the negative (low speed) peak is at the midpoint of the phase, \( 6\pi/8 \). As seen with the boundary layer profile, the fluctuation in the flow over the phase does not suggest that a cyclic mechanism of defined low and high speed streamwise variations, at the same frequency as the modulation is generated. The
nature of flow perturbation would therefore not appear to be acting in the same manner as the baseline phase averaged flow that was investigated closer towards the jets, as identified in Figure 6.15.

Figure 7.11 Periodic streamwise velocity $\tilde{u}/U_\infty$ at chordwise location $x/c = 0.75$, at incidence angle $\alpha = 14^\circ$ and $Re = 8.9 \times 10^5$, for various points in the jet phase ———, $\tilde{u}/U_\infty$ for condition $F^+_m = 0.3, \varphi = 0$;

———, $\tilde{u}/U_\infty$ for condition $F^+_m = 0.1, \varphi = \text{respective point in phase}$.

The shape of the profiles are skewed towards having the peak magnitude of fluctuation at a height of >0.6, and therefore being in the upper portion of the boundary layer. In the baseline flows looked at previously, the fluctuations in the flow were concentrated close to the wall, when they were a direct result of the jet flow, further upstream on the chord. Any perturbations in the flow will convect downstream at close to the freestream velocity, given the proximity to the boundary layer edge. With the low $F^+$ number of the modulation, these are likely to be larger structures when compared to the spatial scales of the baseline perturbation. At $x/c = 0.85$ the greatest magnitude to the fluctuation is also concentrated in the upper portion of the boundary layer. This suggests that the trajectory of any coherent perturbation across these two chordwise points remains similar.
The nature of the fluctuations measured over the period of the modulation however does not follow convention in suggesting that a distinct perturbation at the frequency of the modulation is harnessed. Throughout the equi-spaced points in the phase a sequential increase and decrease to the periodic flow would be expected if a flow perturbation and the modulation frequency were ‘locked-in’ and were related. Such behaviour is not clear from the results however.

### 7.2.4 Wake Profiles

Wake profiles at $x/c = 1.25$, $\alpha = 14^\circ$ were conducted in order to understand the effect on the downstream flow. Plots of the streamwise velocity and the fluctuating component, at the semi span and $z/c = 0.17$, are plotted in Figure 7.13. The results for the baseline actuated flow are also shown for comparison.
The topology of the wake for the modulated strategy is similar to the baseline case. A significant reduction in the height of the velocity deficit downstream of the trailing edge, compared to the unactuated flow is created. However, subtle differences in the suction and pressure surface sides to the velocity profile are apparent. Higher levels of fluctuation to the flow are measured for the modulated flow. The inverse behaviour is seen on the pressure surface flow, in that the modulated result has higher mean velocities than the baseline case. The fluctuation magnitude is still higher for the modulated case.

![Streamwise Velocity ratio (U/U∞) against vertical position (y/c), downstream of the aerofoil at chordwise location (x/c) = 1.25, and spanwise location (z/c) = 0 at incidence angle α = 14°; Re = 8.9 × 10^5; for □, F^s_m Control; ○, Flow Control off; Δ, F^s_m = 0.3 Control.](image1)

![Variance of the Streamwise Velocity ratio ((ui²)/U∞²) against vertical position (y/c), downstream of the aerofoil at chordwise location (x/c) = 1.25, and spanwise location (z/c) = 0 at incidence angles (α) = 14°, Re = 8.9 × 10^5; for □, F^s_m Control; ○, Flow Control off; Δ, F^s_m = 0.3 Control.](image2)

Streamwise Velocity ratio (U/U∞) against vertical position (y/c), downstream of the aerofoil at chordwise location (x/c) = 1.25, and spanwise location (z/c) = 0.17 at incidence angle α = 14°; Re = 8.9 × 10^5; for □, F^s_m Control; ○, Flow Control off; Δ, F^s_m = 0.3 Control.

Variance of the Streamwise Velocity ratio ((ui²)/U∞²) against vertical position (y/c), downstream of the aerofoil at chordwise location (x/c) = 1.25, and spanwise location (z/c) = 0.17 at incidence angles (α) = 14°, Re = 8.9 × 10^5; for □, F^s_m Control; ○, Flow Control off; Δ, F^s_m = 0.3 Control.
The differences between the two actuation strategies however are relatively small compared to the difference to the unactuated flows. As found with the overall forces in Figures 7.3 and 7.4, fluctuations in the wake for the modulated flow though are still significantly lower than that experienced with the unactuated flows.

The flowfield data shows that the amplitude modulated flow generates a slightly more unsteady flow control mechanism then the baseline actuation. This is primarily due to the low frequency mode of the modulation being less time-invariant to the natural frequencies in the separated flow.

7.3 Discussion of Amplitude Modulation

Tests have demonstrated that the use of amplitude modulation of the jet flow can generate control authority, but at a reduced power input to the flow control system. Analysis of the overall forces and flowfield have shown that modulation with a frequency of an order of magnitude lower than that of the carrier frequency creates a different manner of control. The mechanism is however challenging to understand. If there is a relationship, or coupling of the carrier frequency with the modulation frequency for this successful control mechanism, is not fully understood from these results. However, the \(0.1 < F_m^+ < 0.5\) low frequency range was generally most effective.

Perturbations were seen to be evident at the modulated frequency time scale in the flow. It suggests that this could still be the more dominant timescale to the control compared to the carrier frequency when modulation is used. The interaction of the two scales however would require fully time-resolved investigations in order to be further understood. As \(F_m^+\) modulation was most effective at values less than unity, it is possible that mixing enhancement mechanisms of the low frequency actuation elicit a response that operates at a sub harmonic of the natural frequencies in the flow, as is discussed by Ho & Huang (1982) and in the context of flow control by Kotaptai (2010). This suggests that the shear layer stability could be a dominant mechanism in how the controlling effect is created. Such mechanisms were investigated by looking at the phase averaged flowfield in the otherwise separated flow (i.e. when the flow is not controlled). The modulation frequency was 0.3, so the perturbation was more closely aligned with the natural frequency of the flow then that of the carrier frequency. The results suggested irregular fluctuations are generated in the flow on this timescale however. Perturbations identified in the far-field flow did not appear to ‘lock-
in’ to the modulation frequency, and have a cyclic behaviour commensurate with the modulation timescale. The flow behaviour is seemingly on a different timescale to the modulation therefore. An alternative modal response could be generated in the flow, at a different harmonic to the modulation frequency; however this was not clear from the results. With the results seen from the modulation sensitivity study on overall forces at different angles of attack, the optimum low frequency would also appear to be dependent on the parameters of the unperturbed flow also, so the performance of a particular frequency is non-linear, and requires understanding of other parameters in the flow, (such as the separated length) to better understand.

The benefit of frequency modulation is reduced power consumption due to the reduced duty cycle of the jet. This is potentially desirable for applications of flow control. However, from the analysis of force data and the wake surveys, the low frequency mechanism is seen to generate a more unsteady control compared to the higher frequency actuation for certain flow states. For dynamic environments of flow control, such as air and ground vehicles, then this would potentially be more undesirable then the power efficiency improvements are desirable.

Jet efflux orientation was considered in a number of tests, as an alternative actuation strategy to signal modulation. These results will now be discussed in the following sections. Unlike the tests of the signal modulated flows, these tests only considered the normal-sine wave actuation strategy.

### 7.4 Counterstreamwise Actuation

The counterstreamwise orientation of jet at $\theta = -45^\circ$ ($F^+_{\theta = -45}$) was tested at a wind speed of 30 m/s, giving a chord based Reynolds number of $Re = 8.9 \times 10^5$. In order to relate findings to the wall-normal jet configuration ($F^+_{\theta = 0}$), measurements were conducted at the reference angle of attack of $\alpha = 14^\circ$. A slight change in actuation frequency was required, such that $F^+ = 4$ was used for the tests, however it was still very similar to the $F^+_{\theta = 0}$ tests where the normalised frequency was 4.2.

#### 7.4.1 Force Balance Results

Force-balance results are presented in Table 7.1, as a change to the unactuated flow i.e. $\Delta = (actuated \ − \ unactuated)$. The results from the wall-normal setup $F^+_{\theta = 0}$ presented for comparison.
$F^+_{\theta = -45}$ actuation was unable to impart global control authority. As discussed in Chapter 4, the synthetic jet design was not optimised for the orifice dimensions resulting from the counterstreamwise orientation. Lower levels of $C_{\mu}$ than those used in the wall-normal configuration were achieved. Control authority was therefore not achieved to the same magnitude as in previous tests, reflected in the lack of efficiency improvement. This was an immediate limitation to understanding the flow mechanisms of successful control authority with purely a different jet orientation. However, as seen with the previous analysis, characterising the flowfield is desirable in order to understand the limitations of the flow control mechanism. A short series of further tests was conducted to investigate this.

### 7.4.2 Static Pressure Distribution over the Model

#### 7.4.2.1 Chordwise Pressure Distribution

In Figure 7.14 the pressure distribution is compared to that of the unactuated flow. A small enhancement in the leading edge low pressure peak is created with actuation and an increase in the pressure recovery across the central third of the chord. However, despite this improvement in the pressure distribution, which was a characteristic seen in the $F^+_{\theta = 0}$ tests, an improvement is not seen in the overall forces. When comparing the results here to the results presented in Figure 6.3, the unperturbed flow for the counterstreamwise orifice setup generates an altered pressure gradient. The control therefore is acting on a characteristically different flow state than the previous tests, where a less severe trailing edge separation will exist, so the effect of the control is less. The orifice therefore has an effect on the unperturbed flow, so direct comparison across experiments is less applicable.

<table>
<thead>
<tr>
<th>$\theta$</th>
<th>$\Delta C_L$</th>
<th>$\Delta C_D$</th>
<th>$\Delta C_L/C_D$</th>
<th>$C_{\mu}$</th>
<th>$R_U$</th>
</tr>
</thead>
<tbody>
<tr>
<td>-45</td>
<td>+0.019</td>
<td>+0.0005</td>
<td>-0.15</td>
<td>0.02%</td>
<td>0.095</td>
</tr>
<tr>
<td>0</td>
<td>+0.07</td>
<td>-0.023</td>
<td>+6.34</td>
<td>0.11%</td>
<td>0.22</td>
</tr>
</tbody>
</table>

Table 7.1 Summary of effect of control on overall forces.
Figure 7.14  Coefficient of Static Pressure ($C_p$) against chordwise location ($x/c$) over the pressure and suction surfaces of the aerofoil, at incidence angles ($\alpha$) = 14°; for $\square$, $F^\phi_{\theta=0}$ Control; $\bigcirc$, Flow Control off; $\Delta$, $F^\phi_{\theta=-45}$ Control.

7.4.2.2 Pressure Spectra at Chordwise Locations

Unsteady pressures were recorded for the flows, and are shown in Figures 7.15. As was seen for the baseline actuation, a response is clear at the actuation frequency at both points along the chord.

The broadband response of the unperturbed flow at $St = -0.2$ is significantly suppressed. If the results are compared to the Figure 6.4 for the flows with the wall-normal configuration,
the response of the jets is in similar agreement. However, the forms to the unperturbed flow response show differences. This corroborates that a small difference in the naturally separated flows between the two configurations exists. Actuation still suppresses the energetic low frequency mode in the unperturbed trailing edge flow in the same manner. This does not however translate into control authority being afforded to the overall forces.

7.4.3 Velocity Profiles

Figure 7.16 shows a lateral profile of the streamwise velocity of the counterstreamwise jets, and also the wall-normal orientation for comparison. The traverse is at a location of \( x/c = 0.122 \), \( \sim 1d \) downstream of the jet orifice. The velocity is plotted as the difference to the mean unactuated flow \( (\Delta U = \text{actuated} - \text{unactuated}) \), and non-dimensionalised by the freestream velocity. It is apparent there is variation across the array of the output of the jets, due to the complex nature of the flows discussed previously in Chapter 4. However, the magnitude of velocity deficit for both jet orientations is roughly similar. This is despite the momentum of the counterstreamwise orientation jet being lower than that measured for the wall-normal orientation in quiescent conditions. For the majority of the jets in the array, the spanwise spread of the velocity deficit generated by each jet is equal or greater than that for the wall-normal orientation.

![Figure 7.16](image)

Figure 7.16 The change in Streamwise Velocity ratio \((\Delta U/U_\infty)\) with actuation against spanwise position \((z/c)\), at chordwise location \((x/c) = 0.122\), at incidence angle \(\alpha = 14^\circ\); \(Re = 8.9 \times 10^5\), for; \(-\cdot-\cdot-\cdot\), \(F^*_{\theta} = 0\) Control; \(-\cdot-\cdot-\cdot\), \(F^*_{\theta} = -45\) Control.
This is interesting as it suggests that the near jet flowfield of the counterstreamwise orientation is altered. It increases the spanwise range of influence of each jet on the flow, yet at a lower $C_{\mu}$.

Figure 7.17 Streamwise Velocity ratio ($U/U_\infty$) against vertical position ($y/c$), downstream of the aerofoil at chordwise location ($x/c$) = 1.25, and spanwise location ($z/c$) = 0 at incidence angle $\alpha = 14^\circ$, $Re = 8.9 \times 10^5$; for $\square$, $F^*_{\theta = -45}$ Control; $\bigcirc$, Flow Control off.

Variance of the Streamwise Velocity ratio ($\frac{\langle u'^2 \rangle}{U_\infty^2}$) against vertical position ($y/c$), downstream of the aerofoil at chordwise location ($x/c$) = 1.25, spanwise location ($z/c$) = 0; at incidence angles ($\alpha$) = $14^\circ$, $Re = 8.9 \times 10^5$ for; $\square$, $F^*_{\theta = -45}$ Control; $\bigcirc$, Flow Control off.

Figures 7.17 shows wake profiles conducted at the semi span and $z/c = 0.17$, $x/c = 1.25$. Measurements were conducted in the same manner as previous tests. Control generates a
modest decrease in the height of the wake, and a reduction in the intensity of the fluctuations of the streamwise velocity.

The difference between unperturbed and perturbed states however is far less significant than that observed in the tests at the wall-normal orientation. This corroborates with the lack of a significant effect on the drag (Table 7.1), as an overall drag force reduction is coupled with a significant reduction in the wake height. This suggests the counterstreamwise jet does not create significant boundary layer reattachment across the span, as was achieved with the wall-normal jet.

### 7.4.4 Flow-field Measurement of the Jet

In order to investigate the jet flow, PIV data of the surrounding flow-field was taken at $0.145 < z/c < 0.17$. A difference in spanwise disturbance was seen compared to the wall-normal jet in the lateral time-averaged velocity profiles. Data were collected in the same manner as previously, but at seven planes spaced $0.4d$ in the spanwise direction, to acquire flow-field data outboard of the jet centreline. The planes investigated the region from the centreline of the jet to the midpoint between the next jet in the array. The development of the periodic flow over the jet phase is shown in Figure 7.18. The data has been mirrored about the centreline to visualise the flow across the full orifice width.
Figure 7.18  Iso-surfaces of $\frac{\bar{u}}{U_\infty}$ across jet at $z/c = 0.17$ at $Re = 8.9 \times 10^5$, of +0.1 (blue), and -0.1(red) at phases of a) 0, b) $2\pi/6$, c) $4\pi/6$, d) $6\pi/6$, e) $8\pi/6$, f) $10\pi/6$. Spanwise axis = $6d$ depth, with divisions of $1d$.

The measurement region did not extend as far downstream as that used in the wall-normal jet measurements of the periodic streamwise velocity shown in Figure 6.15, as approximately only $9d$ downstream of the jet is measured. However, the development of the perturbation is revealed in both the chordwise and spanwise extent.

Throughout the jet cycle, the convective velocity of the high-speed bulge is lower than that seen in the wall-normal case. This would be expected in the flow immediately downstream of the jet; where the jet efflux was initially against the streamwise flow direction. The injection angle of the jet to the cross-flow has a significant bearing on how the jet structure develops downstream. The results show however that a structured flow perturbation is created across the jet cycle. However, a reason for the lack of control authority in this instance could be due to the reduction in $C_\mu$ (compared to the wall-normal case) creating a reduction in wall-normal height of the structures; these may not be sufficient to entrain the higher momentum fluid toward the wall, and enhance mixing downstream. Considering the spanwise
characterisation, the magnitude of the perturbations are not so significant in the spanwise extent investigated to suggest the adjacent jet flows will interact. Only the domain $<0.4d$ outboard of the jet centreline shows the perturbation extending to the wall normal heights of $\sim 1d$, which is similar to that seen with the wall-normal jets.

Another limitation could be in that the structures generated could not be of the coherent form that is able to persist in the flow far downstream to the separated flow in this instance.

7.4.5 Reduced $Re$ Tests

In order to understand if $C_\mu$ of the jet is a significant parameter to the control authority, tests were performed at a lower freestream velocity, of $U_\infty = 15$ m/s and chord Reynolds number $Re = 4.5 \times 10^5$, in order to create a higher $Re$ jet. This also had the effect of increasing the $F^+$ value of the jet to $F^+ = 8$. As the higher $Re$ tests previously were also of a characteristic actuation frequency $F^+$ of $O(>1)$ however, the same high frequency mechanism of control would be expected to be realised, yet with a test condition now developing a jet of a momentum magnitude closer to that of the wall-normal jet test. The jet and parameters and force results for the actuated flow are shown in Table 7.2.

7.4.5.1 Force Balance Results

The effect of this higher $C_\mu$ jet was that the overall lift and drag characteristics were improved by $+5\%$ and $-22\%$ respectively compared to the unactuated condition. This suggests control authority was being realised in a similar manner to the wall-normal case at the higher Reynolds number, lower $F^+$ flow.

<table>
<thead>
<tr>
<th>$\theta$</th>
<th>$\Delta C_L$</th>
<th>$\Delta C_D$</th>
<th>$\Delta C_L/C_D$</th>
<th>$C_\mu$</th>
<th>$U_\theta/U_\infty$</th>
</tr>
</thead>
<tbody>
<tr>
<td>-45</td>
<td>+0.054</td>
<td>-0.018</td>
<td>+4.23</td>
<td>0.08%</td>
<td>0.19</td>
</tr>
</tbody>
</table>

Table 7.2 Summary of force changes for controlled flow at reduced $Re$.

7.4.5.2 Flow field Measurement of the Jet

Further flowfield measurements were conducted with six spanwise planes acquired on a spacing of $0.1d$, representing a spatial area of $\sim 1d$, (assuming the flow to be symmetric about the jet centreline when the results are mirrored). The results can be seen in Figure 7.19.
Looking at the characteristics of the flowfield, the successive velocity perturbations generated by the jet persist in a coherent form across the span of the measured planes. The wall normal height is of \( \sim 1d \) for the respective ‘low’ and ‘high’ speed bulges to the streamwise flow. The higher frequency of the jet efflux creates shorter timescales for the development of the structures compared to the higher \( Re \) case. These observations of the flowfield suggest a structure of sufficient coherence to generate a more significant fluctuation in the streamwise flow is realised, indicating \( C\mu \) is still a primary parameter in order to generate control authority, even with a change to the efflux orientation.
7.4.5.3 Static Pressure Distribution over the Model

Figure 7.20: Coefficient of Static Pressure ($C_p$) against chordwise location ($x/c$) over the pressure and suction surfaces of the aerofoil, at spanwise position ($z/c$) = 0, incidence angle ($\alpha$) = 14° $Re = 4.5 \times 10^5$; for: $\square$, $F^*_{\theta = -45}$ Control; $\bigcirc$, Flow Control off.

Figure 7.20 shows the pressure distribution for the perturbed and unperturbed flows for $Re = 4.5 \times 10^5$. The unperturbed flow is different to that of the higher $Re$ tests; a more adverse pressure gradient, with the associated trailing edge pressure plateau exists. The flow appears to separate at a point further upstream then it does for the higher Reynolds number tests, as the constant pressure plateau starts at $\sim 0.5 \times c$. However, as the force balance results corroborate, actuation results in a greater low suction peak being generated, and together with suppression in the trailing edge separation, this creates the increases in lift and decrease in drag.
7.5 Discussion of Counterstreamwise Actuation

The investigation of the flowfield has revealed that the development of the counterstreamwise jet efflux is sufficiently different to that of the wall normal configuration. However, how this can be best optimised for a separated flow state is challenging to understand. The tests conducted indicate the characteristics of the separated flow and the $C_{\mu}$ of the jet are still the most significant parameters that will be responsible for realising control authority in a complex flow. Orientation of the jet flow does not appear to overcome the requirement for a threshold level of momentum being imparted to the flow, for achieving control authority. One difference which may exist but has not been investigated is the robustness of each jet flow type. When considering the benefit of counterstreamwise actuation compared to a wall-normal orientation, previous work in the literature can be of interest to consider. Lardeau. & Leschziner (2011) investigated the effect of counterstreamwise actuation, and found advantageous characteristics for the specific flow state investigated. Numerical simulation of the development of the jet flow shows similarities to the measurements of the high Re tests here. Although it remains to be investigated, wall normal actuation would look to generate a more persistent structure in the streamwise extent, while the counterstreamwise generates more persistence in the spanwise extent. These differing characteristics therefore could be advantageous when the practicalities of the application of a flow control system need to be considered, in terms of the spanwise clustering of jets, or the streamwise location where they could be installed. Further investigations are required however to better understand this control strategy.

7.6 Summary

Two methods of alteration to the actuation strategy have been developed and tested. Both of which are potential methods that can reduce the power requirements of an actuator array.

Amplitude modulated waveforms were used to apply a range of forcing perturbations to the flow, from the carrier frequency $F^+ = 4$, over a range down to $F^+_m = 0.03$. All frequencies were shown to create efficiency improvements in the flow. Actuation at $\sim 1 < F^+_m < \sim 2$ however proves to be the least beneficial frequency of perturbation. Lower frequency actuation therefore can result in lower power requirements from an array, but this will generally be at a reduced efficiency and/or increased unsteadiness of the flow, when expected
to operate at a range of flow states. Modulation is a less robust strategy than a high frequency, pure sine-wave strategy.

Counterstreamwise jets with a characteristically high frequency actuation have been shown to create an effect on the flowfield comparable to the wall-normal orientation, for jets that generate similar momentum levels. However, when control was applied to a similar flow state, global control authority could not generated with a $C_{\mu}$ magnitude of ~18% of that which was effective with the wall-normal jet. To define the configurations in which such a strategy can create system efficiency improvements therefore requires further investigation.

The experiments looking at two alternative actuation strategies have demonstrated that for a complex flow, attaining a threshold $C_{\mu}$, and the manner of the jet flow’s interaction with the cross flow, in terms of the excitation position on the lifting surface, are the more important parameters to the array for achieving control authority over the flow, more than defining the ideal perturbing frequency. Both methods studied have the potential to increase the efficiency of a flow control system, however, further investigation is needed to build on these initial findings in order to better quantify the improvements possible.

With the parameters of the actuation considered, the research will consider the ability of the control system to operate in a different, more complex flow state in the next chapter.
Chapter  8

Applying Control in a Ground Effect Flow Condition

8.1    Introduction to Chapter

In this chapter results are presented for tests conducted with the wind tunnel model in a different flow condition to that used previously. Gaining a preliminary understanding of the control system when applied to different flow-states, and hence the robustness of the control authority, was the main point of interest.

The effect of an increased adverse pressure gradient flow over the wing suction surface was the main change. This would create a flow-field more closely aligned to those of potential real-world applications, such as cambered aerofoils, or aerodynamic surfaces operating in ground effect. This change was achieved by generating a ground effect flow regime across the central ~ 0.6 of the span, using the moving ground system in the wind tunnel, which was discussed in Chapter 3. This would understandably create highly complex flows across the suction surface, from the juxtaposition of the flows situated outboard and over of the rolling road, where the respective boundary layer growth rates differ. In the same manner as the previous tests, creating a significant global force enhancement for the actuated flow would reflect the flow control system generating control authority. Analysis of the flow-field in the vicinity of the actuator array would then allow further understanding of the controlled flows.

8.1.1.    Ground Effect Testing

The addition of ride height to the parameter space for experiments means that a significant number of tests would be required to assess the optimal condition for the flow control system configuration. Both inclination (\(\alpha\)) and ride height (\(h\)) will affect the adverse pressure gradient created over the suction surface, and the length scale of the separated flow.
Investigating such a wide parameter space was not practical within the limitations of this research. Ground effect testing was thus only considered at two non-dimensional ride heights. These were h/c of 0.05, and 0.2, where h is the length scale between the lowest point on the suction surface at $\alpha=0^\circ$, and the ground plane. All tests were conducted at a chord Reynolds Number of $Re = 7 \times 10^5$, (equating to a freestream velocity of 25m/s). Tests were conducted at this flow speed due to limitations of the experimental setup, specifically the boundary layer reduction system configuration, and balance loading limits.

8.2 \( h/c = 0.05 \) Ride Height

The lower ride height was relevant for investigation, as the suction surface of the NACA0015 geometry positioned at such a non-dimensional height and at $\alpha = 7^\circ$ then shows similarities in curvature and position above the ground plane to the underside of a contemporary Le-Mans prototype sportscar frontal aerodynamic structure (see Figure 2.8). Fundamental understanding of actively controlled flows in such context is of interest, when understanding the requirements of future real world application.

8.2.1. Force Balance Results

![Figure 8.1](image.png)

Figure 8.1 Lift Coefficient ($C_L$) against incidence angle ($\alpha$) at $Re = 7 \times 10^5$; for $\square$, $h/c = 0.05$.
Flow control on; $\bigcirc$, $h/c = \infty$, Flow control on; $\triangle$, $h/c = 0.05$, Flow control off; $\Diamond$, $h/c = \infty$, Flow control off.
Figures 8.1 and 8.2 show the overall lift and drag forces respectively. Both unactuated and actuated flows are plotted, and the results of the freestream flow ($h/c = \infty$) tests from Chapter 6 are shown for comparison.

The overall lift forces are reduced compared to the freestream flow in both the actuated and unactuated flow states. The increase in downforce as $\alpha$ increases to the maximum force is more gradual in the ground effect flow. In contrast to the freestream results, however, at the higher angles of attack, a downforce reduction phase is not experienced within the incidence range tested. For the actuated flow, no lift enhancement effect is seen for the incidence range investigated. The drag force is significantly increased compared to the freestream condition. There is a negligible change in the drag levels for the actuated flow across all incidences. A gradual increase occurs across the entire incidence range, as opposed to an abrupt increase with $C_{L_{\text{max}}}$, as is seen in freestream conditions. This would suggest significant separated flows are generated from even the very low angles of attack in ground effect.

The unperturbed flow state is unable to be significantly altered by the jets. Creating significant force enhancement may not be as straightforward as for the freestream condition investigated previously. It is however still of interest to investigate the flow field, to assess the effects of actuation.
8.2.2. Flow-field Measurements

The flowfield across the chord was measured at $\alpha = 7^\circ$. To understand spanwise variation of the flow, data were collected at both the centreline and also at the $z/c = 0.17$ position. The unperturbed flows are shown in Figure 8.3. For both spanwise positions, the suction peak is located at $x/c \approx 0.18$. This is a small distance downstream of the actuator array. The jet flow is therefore introduced into a favourable pressure gradient flow, and hence is a different flow-state to what was encountered in the flowfield measurements in freestream conditions at $\alpha = 14^\circ$ in Chapter 6. Due to the ground effect accelerating the flow at the leading edge, the peak velocities are $>2U_\infty$. However, with the lower freestream velocity then the previous tests, the jet flow is therefore introduced into a crossflow that is of a similar $R_U$ to that of the tests conducted at the higher Reynolds number, freestream condition used in Chapter 6.

![Figure 8.3](image)

Figure 8.3 Isocontours of streamwise velocity across the suction surface, at various spanwise positions, incidence angle $\alpha = 7^\circ$ and $Re = 7 \times 10^5$. Contour levels are shown in increments of $U/U_\infty = 0.5$. a) $z/c = 0$, Flow control off. b) $z/c = 0.17$ Flow control off.
The unperturbed flow separates at $x/c = 0.5$ for the centreline flow. At the outboard plane ($z/c = 0.17$), the zero velocity contour develops further downstream at $x/c > 0.6$. Separation occurring further upstream at the semi span than at the more outboard position was also revealed with the freestream results for the high angle of attack conditions from the investigation of the wake profiles in Chapter 6. Taking the length scale of the separation at the $z/c = 0.17$ position in the span, $F^+$ of the jet is $\sim 6.9$ (when $F^+ = f x_{sep}/U_\infty$).

Results of the actuated flow at the same spanwise positions can be seen in Figure 8.4. The flow speed is increased at the suction surface leading edge, increasing the near wall jet effect to the flow. This occurs at both points across the span. However, downstream of the peak velocity, the flows at the semi span and $z/c = 0.17$ differ somewhat. At the semi span, the separation point has moved upstream compared to the unactuated flow. However, the wall normal height of the zero-velocity contour is highly similar across the rest of the chord. There is no reduction to the height of the separated flow region due to actuation. These results suggest that in this significant adverse pressure gradient flow, a low level of spanwise control authority from the jets is generated, as little effect is seen at the centreline. The distance to the jets either side of the semi span is $\sim 4.5d$, so their spanwise authority is relatively weak (Lardeau. & Leschziner 2011). At $z/c = 0.17$ however, the separated region is reduced in height. The zero streamwise velocity contour has a reduced wall-normal distance compared to the unactuated flow. It now occurs much closer to the wall, such that it is not fully captured in the area of measurement. The wall-normal velocity gradient is very similar to the unactuated flow however, suggesting a thickening of the boundary layer still occurs towards the trailing edge in the adverse pressure gradient. The flow in the vicinity of the jets has a modest enhancement due to actuation. With the change in flow condition and actuation frequency, multiple parameters that affect the control have varied compared to the previous freestream tests, so there are a number of reasons to attaining a less significant control authority in comparison. As previous tests in Chapter 7 identified, $F^+ = > 4$ actuation generates control authority (for the counterstreamwise jet setup), so this would not be expected to be the primary reasoning for the results here; Frequency is therefore thought to be the least sensitive, with regards to control authority, of the primary parameters discussed in this research.
Figure 8.4  Isocontours of streamwise velocity across the suction surface, at various spanwise positions, incidence angle $\alpha = 7^\circ$ and $Re = 7 \times 10^5$. Contour levels are shown in increments of $U/U_\infty = 0.5$, a) $z/c = 0$, Flow control on, b) $z/c = 0.17$ Flow control on.

Phase-averaged data of the streamwise and vertical periodic fluctuations was taken for the flow at $z/c = 0.17$, at $x/c = 0.12$ downstream to $x/c = \sim 0.4$. The results are shown in Figure 8.5 and 8.6. Throughout the jet cycle, each streamwise fluctuation is stretched across a longer extent of the chord in the favourable pressure gradient flow at $0.15 < x/c < 0.25$, with a lower wall normal height, when compared to the freestream condition investigated previously in Chapter 6. In those tests a more significant control authority was attained. The development of the jet flow therefore may not be sufficient in this crossflow, in order to generate a coherent vortical structure that can persist downstream in the adverse pressure gradient.
Figure 8.5 Contours of $\tilde{u} / U_\infty$, across the suction surface, at spanwise position $z/c = 0.17$, for flow control on, $h/c = 0.05$; at phases of a) 0, b) $2\pi/6$, c) $4\pi/6$, d) $6\pi/6$, e) $8\pi/6$, and f) $10\pi/6$. 

Figure 8.6 Contours of $\tilde{v} / U_\infty$, across the suction surface, at spanwise position $z/c = 0.17$, for flow control on, $h/c = 0.05$; at phases a) 0, b) $2\pi/6$, c) $4\pi/6$, d) $6\pi/6$, e) $8\pi/6$, and f) $10\pi/6$. 

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As with the previous tests, in the adverse pressure gradient flow further down the chord, the effect of the jet flow starts to dissipate. This reduction in the intensity of the jet flow appears to mainly be due to how it develops in the cross-flow from $0.15 < x/c < 0.25$. This manifests as a decrease in overall control authority.

Based on these results, it was deemed of interest to investigate a less adverse flow-state to further understand the effect of differing flow states. This was achieved by reducing the severity of the ground-effect mechanism. An $h/c = 0.2$ ride height was therefore investigated. This ride height would be expected to be above the critical ride height range identified by Zerihan (2001), where significant force reduction occurs with a ground effect regime.

8.3 $h/c = 0.2$ Ride Height

8.3.1. Force Balance Results

Force balance data for the wing at $h/c = 0.2$ is shown in Figures 8.7 and 8.8. The results from the tests conducted in freestream conditions in Chapter 6 are shown for comparison. In the unactuated state, the ground effect flow generates significant increases in downforce compared to the freestream condition. Throughout the incidence range tested, a downforce reduction phase is not experienced. However from $\alpha = 7^\circ$, the downforce increase is asymptotic to the maximum angle tested. At higher angles of attack, the downforce compared to the freestream condition is increased by up to 35%. The ground effect flow however significantly increases drag. When the flow is actuated, from $\alpha \Rightarrow 7^\circ$, additional downforce is generated. A significant increase in downforce is generated at the higher incidence range. At $\alpha = 14^\circ$, a ~7% increase in downforce is created for a ~5% decrease in drag. Actuation therefore generates a small increase in the efficiency at $\alpha \Rightarrow 7^\circ$. Control authority is therefore generated to a greater extent than encountered at the lower ride height.
Figure 8.7  Lift Coefficient ($C_L$) against incidence angle ($\alpha$) at $Re = 7 \times 10^5$; for □, $h/c = 0.2$, Flow control on; ○, $h/c = \infty$, Flow control on; △, $h/c = 0.2$, Flow control off; ◊, $h/c = \infty$, Flow control off.

Figure 8.8  Drag Coefficient ($C_D$) against incidence angle ($\alpha$) at $Re = 7 \times 10^5$ for □, $h/c = 0.2$, Flow control on; ○, $h/c = \infty$, Flow control on; △, $h/c = 0.2$, Flow control off; ◊, $h/c = \infty$, Flow control off.

8.3.2. Static Pressure Distribution over the Model

8.3.2.1 Chordwise Pressure Distribution

The pressure distribution for the unactuated flow is shown in Figure 8.9, for $\alpha = 14^\circ$. The results for the freestream flow condition at the same angle of attack are also plotted for comparison. The chordwise length of the low pressure region is enhanced in the ground effect flow compared to the freestream. This is due to the constriction on the flow enforced by the moving ground plane. The increased low pressure region, and increase in pressure over the pressure surface would appear to be the reasoning for the greater lift generated in the ground
effect flow. Along the chord a greater adverse pressure gradient is experienced in ground effect. The pressure recovery cannot be maintained to the same extent seen in the freestream flow, and a constant pressure plateau of higher pressure is experienced across from ~x/c = 0.4. The actuated flow is shown in Figure 8.10.

![Graph](image)

Figure 8.9 Coefficient of Static Pressure ($C_p$) against chordwise location ($x/c$) over the pressure and suction surfaces of the aerofoil, at incidence angle ($\alpha$) = 14°; for □, h/c = 0.2, Flow control off; ○, h/c = $\infty$, Flow control off.

![Graph](image)

Figure 8.10 Coefficient of Static Pressure ($C_p$) against chordwise location ($x/c$) over the pressure and suction surfaces of the aerofoil, at incidence angle ($\alpha$) = 14°; for □, h/c = 0.2, Flow control off; ○; h/c = 0.2, Flow control on; △; h/c = $\infty$, Flow control on;

Actuation increases the low pressure peak magnitude towards the leading edge. Downstream of the jet array, the pressure recovery is enhanced to $x/c = \sim 0.6$. Towards the trailing edge a slight increase in pressure is seen. No change to the flow over the pressure surface is created.
Actuation therefore generates the downforce enhancement and drag reduction in a similar manner to the freestream tests.

8.3.2.2 Pressure Spectra at Chordwise Locations

The pressure spectra at $x/c = 0.76$ and trailing edge chordwise locations are shown in Figure 8.11, for the unperturbed and perturbed cases. Frequencies are non-dimensionalised by the separated length and freestream velocity. The actuation has a small effect on the frequency response of the flow at both chordwise positions.

At both positions, a distinct peak is seen, at $St = ~8.6$, which is directly coupled to the reduced frequency of the jet flow. At $x/c = 0.76$, a response of $St = ~0.3-0.4$ that is evident in the uncontrolled case, is suppressed in the actuated state. This is a similar response to what was found in the freestream controlled flow, but control appears to create a less significant change in the ground effect flows. The results suggest the natural large scale shedding mode of the unperturbed flow at this point is slightly modified with actuation.

8.3.3 Flow-field Measurements

The flow field was captured for the suction surface at $z/c = 0.17$. Due to limitations on the range of the light-sheet in the vertical plane, the flows close to the wing surface were captured from the leading edge to $x/c = ~0.6$. The streamwise velocity of the unactuated flow is shown in Figure 8.12. The peak velocity is experienced towards the leading edge, upstream of the actuator array. Downstream of this, the flow quickly decelerates in the adverse pressure
gradient. By $x/c = 0.45$, the flow separates. With this length stale, $F^+$ of the jet is $\sim 9.3$ (when $F^+ = f \frac{x_{sep}}{U_\infty}$), which is in the close agreement to the previously identified Strouhal number to the trailing edge flow. The flow control therefore is operating at a characteristically high $F^+$. 

![Figure 8.12](image)

Figure 8.12 Isocontours of streamwise velocity across the suction surface, at spanwise position $z/c = 0.17$, incidence angle $\alpha = 7^\circ$ and $Re = 7 \times 10^5$, for Flow control off., $h/c = 0.2$ Isocontours of streamwise velocity. Contour levels are shown in increments of $U/U_\infty = 0.2$.

For the actuated flow, the separation point moves down the chord. Figure 8.13 shows that the boundary layer height is significantly reduced at $x/c = \sim 0.5$. This implies that in the vicinity of the jet, the efflux is able to impart significant control authority to reattach the flow a significant distance further downstream, when compared to the separation of the unperturbed flow.
Figure 8.13  Isocontours of streamwise velocity across the suction surface, at spanwise position $z/c = 0.17$, incidence angle $\alpha = 7^\circ$ and $Re = 7 \times 10^5$. For; Flow control on, $h/c = 0.2$  Isocontours of streamwise velocity. Contour levels are shown in increments of $U/U_\infty = 0.2$.

For this flow condition, the length scale from the excitation location to the mean separation point is shorter, and a greater adverse pressure gradient is generated, in comparison to the freestream tests in Chapter 6; yet significant control authority has still been generated.

Phase averaged datasets were acquired at the same location in the flow for six equi-spaced points throughout the jet cycle. The phase averaged data in Figure 8.14 reveals low streamwise velocity bulges to the flow are generated downstream of the array. They convect downstream at an average speed of $U_\sigma$. Looking at the flow behaviour throughout the phase, the streamwise development of the jet flow is highly similar to that seen in the freestream condition. However, the point of mean flow separation in the unperturbed flow here is far closer to the point where the jet flow develops. Throughout the cycle, as bulges convect downstream, the streamwise velocity profile at the area where the flow separates is affected as each subsequent jet structure interacts with the flow in this area. At $\phi = 4\pi/6$, the velocity iso-contour of 1.4 can be seen to shift further upstream as the jet bulge increases the wall normal velocity gradient. The coherent structures associated with the low speed bulges therefore appear to be important in generating the reattachment of the flow.
Figure 8.14  Isocontours of phase-locked streamwise velocity ($U_\phi$) across the suction surface, at spanwise position $z/c = 0.17$, for Flow control on, $h/c = 0.2$ velocity. Contour levels are shown in increments of $U_\phi/U_{\infty} = 0.2$. a) $\phi = 0$, b) $\phi = 2\pi/6$, c) $\phi = 4\pi/6$ d) $\phi = 6\pi/6$ e) $\phi = 8\pi/6$ f) $\phi = 10\pi/6$.

In Figure 8.15, phase averaged plots of the spanwise vorticity magnitude are plotted. The jet structure develops in wall normal height as it conveys downstream. The streamwise low speed bulges in Figure 8.14 corroborate the spatial location of the vorticity concentrations.

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The characteristics of the perturbations in the streamwise flow are highly similar to those identified in the lower $F^+$ tests detailed in chapter 6 at $F^+ = \sim 4$. This further corroborates the previous results, implying the control behaviour when $F^+ = > 2$, is time invariant compared to the separated flow, and relatively insensitive to frequency across the range $F^+ = 2 - 10$. As successive structures move downstream, at $\phi = 4\pi/6$ in the phase, a concentration of vorticity is positioned at $x/c = \sim 0.4$. At this point the jet structure starts to interact with the depleted momentum flow identified in the time averaged streamwise velocity flowfield.
Figure 8.15   Isocontours of phase-locked spanwise vorticity $\zeta/c_{u_\infty}$ across the suction surface, at spanwise position $z/c = 0.17$, for; Flow control on, $h/c = 0.2$ Contour levels are shown in increments of $\zeta$ $c/u_\infty = 20$ at phase a) $\phi = 0$, b) $\phi = 2\pi/6$, c) $\phi = 4\pi/6$ d) $\phi = 6\pi/6$ e) $\phi = 8\pi/6$ f) $\phi = 10\pi/6$.

This corroborates the point at which the wall-normal velocity profile alters most significantly throughout the jet phase, shown in Figure 8.13. In the unperturbed flow, $x/c = \sim0.45$ is seen to be the time averaged separation point.

In the spanwise vorticity plots in this area, an irregular region exists when compared to the concentrations that are formed from the jet, which are of a more consistent shape throughout the phase. This will be a point of a complex three-dimensionally separated flow. The flow
behaviour here will not be fully or best characterised by the planar measurements acquired. It is apparent however that the jet flow acts to alter the flow field in this vicinity. This mechanism of enhanced mixing and additional momentum transfer by the coherent jet flow is that which promotes the reattachment of the flow.

The periodic fluctuations to the streamwise and vertical velocity throughout the phase are plotted in Figures 8.16 and 8.17. Fluctuations in the flow are generated in the same manner as measured in freestream conditions, further corroborating that the nature of perturbation is very similar across the flow-states, despite the differences to the unperturbed flow condition length and time scales. Positive (high speed, blue) and negative (low speed, red) streamwise flows, and vertical (towards the wall, blue) and (away, red) fluctuations of the highest intensity are concentrated at the near-wall region. The low streamwise velocity bulge occurs in front of each coherent vortical structure seen in Figure 8.15. Again, their trajectory and speed are a function of the convective velocity of the jet flow. The successive alternating motions in the flow persist to the point where the enhanced mixing is required at \( x/c \approx 0.5 \), and act to maintain the boundary layer attachment. The highly similar behaviour in this \( h/c = 0.2 \) ground effect flow to the freestream condition demonstrate that the jet structure is a robust mixing enhancement mechanism. The results here have shown more similarities to the freestream case then the lower ride height case, demonstrating a limiting parameter for control authority can be the adverse pressure gradient of the flow.

This supports the hypothesis that control authority is strongly dependant on a jet structure being created of sufficient strength to generate intense fluctuations in the flow. The strength depends of the vorticity concentrations being of a coherent form that can persist in the flow. The more adverse pressure gradient experienced in the lower ride height tests would appear to be the primary mechanism for less effective control, as it causes vortex breakdown (Zerihan 2001; Pegrum 2006).
Figure 8.16 Contours of $u / U_\infty$ across the suction surface, at spanwise position $z/c = 0.17$, for flow control on, $h/c = 0.2$; at phases of a) 0, b) $2\pi/6$, c) $4\pi/6$, d) $6\pi/6$, e) $8\pi/6$, and f) $10\pi/6$.

Figure 8.17 Contours of $v / U_\infty$ across the suction surface, at spanwise position $z/c = 0.17$, for flow control on, $h/c = 0.2$; at phases of a) 0, b) $2\pi/6$, c) $4\pi/6$, d) $6\pi/6$, e) $8\pi/6$, and f) $10\pi/6$. 
8.3.4. Wake Profiles

Wake profiles at $x/c = 1.25$ were taken for both the actuated and unactuated flows. Measurements were taken down to approximately 2mm above the rolling road. In order to achieve this LDA probe was orientated at $+4^\circ$ around the x-axis, as opposed to the $-4^\circ$ angle that had been used previously in the similar tests conducted in the previous chapters. Figure 8.18 shows the flows at $z/c = 0.17$ and $z/c = 0$ respectively. As found in the freestream case, the flows at the centreline are subject to an increased velocity deficit in the wake, due to an earlier separation of the flow. For the actuated flows, a decrease in the height of the wake is seen at both spanwise locations. The general effect of the controlled flow on the wake is therefore consistent for both freestream and ground effect regimes. This suggests that in ground effect, although the global results of drag forces show little effect from actuation, the effect on the wake is still fairly significant.

Figure 8.18 Streamwise Velocity ratio ($U/U_\infty$) against vertical position ($y/c$), downstream of the aerofoil at chordwise location ($x/c$) = 1.25, spanwise location ($z/c$) = 0; at incidence angle ($\alpha$) = 14°; at $Re = 7 \times 10^5$, for □, = Flow Control on, $h/c = 0.2$ ⊙, = Flow Control off, $h/c = 0.2$. Variance of the Streamwise Velocity ratio ($\overline{(u')^2}/U_\infty^2$) against vertical position ($y/c$), downstream of the aerofoil at chordwise location ($x/c$) = 1.25, spanwise location ($z/c$) = 0; at incidence angle ($\alpha$) = 14°; at $Re = 7 \times 10^5$, for □, = Flow Control on, $z/c = 0$; ⊙, = Flow Control off, $z/c = 0$. 
Streamwise Velocity ratio \( (U/U_\infty) \) against vertical position \((y/c)\), downstream of the aerofoil at chordwise location \((x/c) = 1.25\), spanwise location \((z/c) = 0.17\); at incidence angle \(^\alpha = 14^\circ\); at \(Re = 7 \times 10^5\); for
\[ \square, \text{ Flow Control on, } h/c = 0.2 \]
\[ \bigcirc, \text{ Flow Control off, } h/c = 0.2. \]

Variance of the Streamwise Velocity ratio \((\langle \overline{u'}^2 / U_\infty^2 \rangle)\) against vertical position \((y/c)\), downstream of the aerofoil at chordwise location \((x/c) = 1.25\), spanwise location \((z/c) = 0.17\); at incidence angle \(\alpha = 14^\circ\); at \(Re = 7 \times 10^5\); for
\[ \square, \text{ Flow Control on, } h/c = 0.2 \]
\[ \bigcirc, \text{ Flow Control off, } h/c = 0.2. \]

Actuation reduces the magnitude of fluctuations in the wake downstream of the jets at \(z/c = 0.17\), but not at \(z/c = 0\). Comparing the results to the tests conducted in freestream however, Figure 6.6 in Chapter 6, the magnitude of the fluctuation of the controlled flow in ground effect is far greater than in the freestream condition.

These results suggest that although the controlled flow represents an enhancement compared to the unperturbed flow in ground effect, the unsteady behaviour of the unactuated ground-effect flow is sufficiently different to the freestream condition. Considering the nature of the ground effect flow however, the unsteadiness increase experienced will be due to many parameters that have changed stability of the flow; so the relationship to one specific parameter such as the actuation frequency, is challenging to understand.

### 8.4 Discussion

The tests conducted in a ground-effect flow condition have allowed the flow control system to be tested in a different flow state compared to the previous tests, but still where a trailing edge separation exists over the suction surface. This flow differed to the freestream condition, with a larger length scale to the separated flow bubble, and hence a shorter length scale between the actuators and the mean separation point. A greater adverse pressure gradient was created, and also a greater degree of three-dimensionality to the flow, due to the non constant
moving ground condition across the span. These characteristics are relevant to real-world application flows; the results were therefore of interest to understand.

The results highlight that generating a robust control system, in terms of operating over a range of flow conditions when multiple, real world parameters are considered, is challenging. In the tests here, a reduction in ride height altered the characteristics of the unperturbed flow sufficiently that a significant control authority could not be created in the lower ride height tested.

Understanding the reasoning for this however is not straightforward, due to the variables that exist. In this flow where the separation point is not fixed, four important parameters of the flow condition alter concurrently between two tests of differing ride heights; the length-scale of the jet array to mean flow separation point, the $F^+$, cross flow velocity ratio, and adverse pressure gradient have all been different in each test considered.

For the results here, the most significant changes between tests are understood to be the adverse pressure gradient and mean separation point to excitation location length scale ratio. In conditions where control authority is achieved, as the adverse pressure gradient increased and the mean separation point moved forward on the chord, the overall drag reduction afforded by the controlled flow diminished. Tests in varying pressure gradients may allow a parametric understanding of these two parameters to be better understood. Predicting the parameters for an unperturbed flow where control may be most, or inversely least successfully created, is a significant task, when considering the nature of real-world flows, such as the complex regimes around air and ground vehicles. However, improving such understanding of the parameters is important in order to develop successful flow control.

8.5 Summary

The results from the tests conducted in a ground effect flow have demonstrated that the mechanism whereby the jet flow generates control authority has no significant differences when applied to a flow condition of an increased adverse pressure gradient, compared to that investigated previously in freestream conditions.

The behaviour of the controlled flow follows the same general trend; persistent, periodic, three dimensional fluctuations in the flow are responsible for the momentum transfer in the
boundary layer. The general manner in which the jet flow develops, and interacts with differing cross-flows remains fairly consistent.

Tests conducted at a ride height of $h/c = 0.05$ and 0.2 demonstrate that attaining significant control authority is not only based on the parameters of the jet flow, namely the $C_\mu$ and $F^+$, but also the nature of the unperturbed flow. The adverse pressure gradient and length scales of the separated flow were seen to be significantly different in ground effect compared to the freestream condition investigated previously, and these effected control authority. A limit to the overall control authority possible is apparent when jets are introduced to an adverse flow condition, due to these parameters.

The development of the jet flow differed most in the lower ride height investigated compared to the others conditions tested. Earlier breakdown of the perturbation pattern occurred, meaning the enhanced mixing upstream of the separation point (for the unperturbed flow) is diminished. At increased ride height, control authority was more evident; implying that a limitation to the flow control system is the adverse pressure gradient range it can operate in. Further study of the control authority of an actuator array in a range of flow conditions relevant to real world applications is therefore an interesting and important requirement for future investigations.
Chapter 9

Conclusions and Future work

9.1. Concluding Remarks

The aims of the work as presented in Chapter 1 were:

*To achieve a better understanding of the behaviour of synthetic jets as an active flow control system, by investigating several parameters identified as contributing to the performance and efficiency of a system.*

The study of synthetic jet flow control over a characteristic trailing edge separated flow was investigated. In broad terms, the parametric tests conducted have furthered understanding of this type of active flow control system. The tests were conducted at Reynolds numbers of $O(10)^6$ and considered an unperturbed flow that was complex in the variation of the mean separation point across the span of the wing. The synthetic jet flow used was of $F^+ > 1$, emitted from a round orifice, and with a specific excitation location. The experiments considered a combined parameter space for which very few comparable findings were available, to the best of the author’s knowledge. A greater awareness of the control authority of high frequency flow control techniques has been formed.

The global results of aerofoil efficiency improvement are in general agreement with those of Melton et al (2005, 2006) which had previously considered high frequency control; $F^+ = \sim 4$ control is effective for enhancing lift by controlling separation. Despite the difference in the conditions used here to those previous experiments, results have demonstrated the key flow control parameters remain in agreement across different flow-states.

A key finding from this work was that the coherent structure of the round jet flow develops a periodic mixing enhancement in the same manner as understood by Zhong & Zhang (2013). In research performed here however, control was effective, but with a significant increase in the streamwise length scale ratio of the natural separation point to the array location,
compared to previous study. To expand on this finding, in the cross-flow condition examined in Chapter 8, the results showed that the same basic convective behaviour of the jet flow remained, along with the associated overall lift enhancement. This demonstrates the robustness of the high-frequency actuation across changing cross-flow states. There is a limitation to control authority however, due to momentum coefficient magnitude, and when sufficiently adverse flow conditions exist. The high frequency actuation strategy does not overcome the inherent limit of the control mechanism common to all other synthetic jet flows. These results demonstrating successful lift enhancement by separation control have been collected at higher actuation frequencies than the previous round orifice studies. The results suggest an insensitivity to the actuation frequency at $F^+O(1) – O(10)$ for the round jet flow exists. This is a desirable characteristic for creating a consistent control effect.

To further expand on these results, the effect of amplitude modulation strategies was looked at, which allowed the effect of frequency to be investigated. Results demonstrated that although still effective compared to no actuation, very low actuation frequencies i.e $F^+ = < ~0.1$, can create less consistent efficiency improvements, due to the control not being time invariant compared to the separated flow. The results demonstrate that characteristically higher frequency actuation, $F^+ > 2$, represents a more robust strategy; separation is constantly suppressed by the jet flow and hence the variation in force enhancement could be less likely to occur with small variation in the mean flow. It is the author’s belief therefore that high $F^+$ control presents a more desirable control authority with all aspects considered, so represents the more likely route for realising flow control systems on vehicles. With such a strategy, amplitude modulation could be introduced to allow the use of more energy efficient, lower frequencies when required. This would therefore avoid potentially undesirable results from lower $F^+$ actuation found here, which could occur when a flow control system is reliant on the actuators fundamental frequency limit to attain the desired response range.

The results showed jet orientation is a parameter that effects control authority also. The limited results in this study would require further investigation to understand if the strategy investigated could realise an overall enhancement in system efficiency, when compared to wall-normal configurations. The location of counterstreamwise jets in relation to the mean separation line however is an important parameter to consider.

The findings that consider the operation of the piezoelectric actuator have enhanced understanding of the potential robustness of this method of active flow control. The system
here shows potential relevant to the requirements of industry; the system design can operate for significant periods of time and generate aerodynamic figure of merit performance better than unity. This suggests a fully developed, practical system could be of overall benefit to a vehicle. The actuator is inherently lightweight and simple, so net benefits with a larger array of actuators than tested here look promising within the context of automotive applications. It is the author’s opinion that these findings suggest potential suitability for UAV application in the same manner. To understand the potential for piezoelectric actuator use in more advanced aeronautical applications, such as commercial or military aircraft design, however requires further consideration. The actuator technology has been shown to be practical to integrate into an aerodynamic structure, which is a key consideration for industrial flow control systems. This may make it more preferable than other technologies currently in development (Cattafesta & Sheplak 2011) providing authority over the flow can be realised. In general terms, the findings by Seifert (2014) are in general agreement with those found here, in that piezoelectric actuators show potential. There were difficulties in the development of the system however, so significant work is still required to further develop more bespoke and effective actuators.

9.2. Future Work

As discussed, there are limited demonstrations of effective high-frequency round jet piezoelectric synthetic jet flow control in the open literature, so further research is clearly required in this area. Using the findings here as a basis for assessing the potential to realise efficient systems with large actuator arrays would be valuable. Actuator density, in terms of area-based momentum coefficient levels, and excitation location then become further parameters to investigate when practical flow conditions are identified. Achieving net efficiency gains with larger arrays of actuators using piezoelectric technology still requires further assessment.

In the author’s opinion, the motorsport environment becomes an ideal test-bed for such further study. The wings and bodywork of race cars represent practical locations to consider using such separation control. The integration of the technology into a contemporary race car development program would represent a way to realise the requirements of real-world conditions if tested on full-scale vehicles. This would create a greater level of confidence in the findings for all industries, than the current laboratory experiments do. Such an environment represents a practical progression route for such technology.
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Appendix A  Assessment of the Performance of the Flow Control System

A.1  Introduction

In this section, results of the performance of the synthetic jet actuator array will be covered. These findings on the practical aspects of the operation of the flow control system are of interest to discuss. The potential of synthetic jet actuators to be a viable flow control system for use in many real world applications depends on the successful operation of an actuator array. Although many studies have investigated the flows generated by a single jet, (Gallas 2005), (Gomes 2009) fewer have covered the performance of a number of piezoelectric actuators (Ternoy 2013) (Chen 2002).

The success of a flow control system depends on the ability of the actuator(s) to impart control authority to the cross flow. However this needs to be achieved in a specific environment, in a reliable manner, and for practical timescales. Defining such aspects is important for understanding the technology readiness level of flow control systems, and the general robustness they could have for extended and varied uses.

A.1.1  Synthetic Jet Actuator

The SJA design uses commercial ‘off-the shelf’ piezoelectric discs for the moving wall. The use of these piezoelectric discs in the majority of other consumer-type products is as a compact source of low complexity audible information, i.e. emitting noise for an alarm system. They are understandably therefore not optimised for precise displacement characteristics, or longevity of operation. Due to the advantageous properties of piezoelectric elements however, they are a technology that should not be discredited for certain flow control applications. General findings that relate to the operation of them in a flow-control system environment are of interest to detail in order to better understand their potential in relation to other candidate technologies (Cattafesta 2011).

A.2  Synthetic Jet Actuator Performance

A.2.1  Operating Life Limitations

From initial tests in the early stages of the research, it was understood using a range of input voltages would allow the output of the SJA to be varied. Operating voltages of up to $120V_{pp}$ were applied to the disc. However, a piezoelectric element is susceptible to polarisation saturation, whereby a fracture to the element endured by a sufficiently high electric field
occurs (Gomes 2009). A limit to the input voltage for a disc therefore exists. However, disc displacement magnitude increases with input voltage, so generally the higher input voltages within a usable range are more desirable. The brass shim that the piezo elements are bonded to is a relatively ductile material. It can withstand deformation with elastic behaviour. The ceramic composition of the patch however is brittle, and strain induced cracking can occur with large displacements. Two modes of failure of the piezo patch are therefore possible, from electrical breakdown, or excessive deformation. Both however render the disc unable to oscillate and the SJA unable to create a jet flow. With the nature of the tests undertaken, the longevity of a consistent performance level from disc needed to be understood. Flowfield measurements take a relatively long time to complete, compared to the timescales of the jet cycle. Within a research environment, having a disc with a consistent output over a long timescale allows a number of parametric tests to be conducted. It is also desirable to understand the behaviour of the piezo-element and SJA assembly over extended periods that would be pertinent to a typical real-world use. As mentioned, initial tests showed $120V_{pp}$ represented an upper limit of the disc operating voltage. Failure of the disc was likely to occur at levels above this voltage after a relatively short period of time, in a short number of tests undertaken. This was primarily due to excessive deformation induced fracture of the piezo patch. With wind tunnel tests performed initially with the entire array, a $100V_{pp}$ input voltage was used. Tests showed that such an input voltage allowed the disc to operate for durations typical of discrete tests, without affecting the displacement performance. However when a more extensive series of wind tunnel tests was conducted, two actuators were seen to fail within quick succession, within the time duration of a series of flowfield measurements. On disassembly of the wind tunnel model the cause was shown to be fracture of the piezo patch, which can be seen in Figure A.1. The failure modes are highly similar, where fracture occurs towards the centre of the patch, indicative of crack propagation. The precise reasoning for cracking occurring is unclear, however, with failure occurring after a significant number of cycles, $O(10)^6$, it suggests it is not purely due to excessive displacement of the diaphragm, (where failure occurs far quicker). This result therefore necessitated conducting the longer duration tests at lower input voltages, for which quantitative data of the performance was recorded.
A.3 Results

Presented in Figure A.2 are the results of measurements of the mean axial velocity output of all the SJAs in the array, at ~1d above the orifice. Significant reductions from the mean would be indicative of a degrading performance, or failure of a piezoelectric patch in an actuator. Tests were conducted after periods of significant operation of the disc, over an entire timescale of approximately 60 days. All the tests were conducted without disassembly or replacement of any of the SJA modules in the wind tunnel model. The results are therefore representative of the operating life characteristics for the flow control system. The velocity is shown as a function of the number of oscillations, and hence the operating time. This was estimated from the time duration of each discrete set of measurements, when the actuated flow was measured. The 980Hz driving frequency was used throughout at the same input voltage of 80Vpp. The result shows that each actuator creates a consistent performance. The majority of results are within one standard deviation (σ) of the mean (μ), when the disc has been operating for $O(10^6)$ oscillations. When the operating duration reaches $\sim 2 \times 10^7$ however, more significant variations are seen. With the measurements at $\sim 2.7 \times 10^7$ cycles, which is equal to ~7 hours of cumulative running, a reduced velocity is seen for the actuator positioned at $z/c = -0.17$. During inspection of the array during disassembly however, cracking of the patch was not shown to have occurred in the same manner as previously. The lack of jet output was attributed to failure in the connecting wires, and not in the structural breakdown of the patch.

However, the operation of the entire array has been shown to become less consistent over these extended time durations.
A guideline to the number of cycles of $\sim 10^7$, before the risk of failure becomes significant, can be summarised. This is based on the level of polarisation saturation that is generated for the $80V_{pp}$ input. The results suggest that in the consistent environment of wind tunnel testing, the actuator has a lifespan of consistent operation of $\sim$6 hours for a high frequency control strategy, before a significant uncertainty in continued performance is reached.

A.3.1 Summary

In terms of a real-world application, this timescale is adequate the requirements of flow-control technologies for ground vehicle aerodynamics; specifically the motorsport environment. Time durations where a system on a vehicle could be required to operate without any interference/replacement could range from 10 minutes to 6 hours depending on the competition discipline. An upper limit of 6 hours therefore for a typical piezo-element therefore would be suitable. However, there is a clear interest in increasing this operating envelope in order to make such an actuator-type desirable for many other uses, such as in aviation flow control, where operating timescales would be far greater.

It has been understood that the discs have an operating lifespan that depends on the power input. As the dynamic requirement of a disc increases, in terms of a larger deflection to increase jet velocity, this affects the longevity. There is therefore an optimal compromise for a piezoelectric actuator, dependant on performance level, and lifespan.
A.4 Performance Criterion

With the results of overall forces in Chapter 6, Figure 6.2, where a drag reduction was generated with the flow control actuated, the invested energy to perform the control can be considered as a ratio to the saved fluidic power to give $\eta_{Pin}$:

$$\eta_{Pin} = \frac{P_d}{P_a} = \frac{(D_u \cdot U_u) - (D_p \cdot U_u)}{\sum_{j=0}^{n} P_j} = \frac{1/2 \rho U_u A \Delta C_D}{\sum_{j=0}^{n} P_j}$$ (A.1)

Where $D$ is the drag, subscript $u$ is the uncontrolled condition, subscript $p$ the perturbed, controlled conditions, $P_a$ the power provided to the array $P_j$ the power to the each actuator, and $P_d$ the fluidic power. Here when $\eta$ is being considered as a function of $\alpha$, the trends of $\eta_{Vin}$ is equal to those that are given by $\eta_{min}$ as defined by Siefert (Seifert 2014) of mechanical conversion efficiency. The reader is directed to the literature for further information. However as performance considered against measured electrical power is more of interest in this instance $\eta_{Pin}$ is a more suitable descriptor, as will be seen with the Aerodynamic figure of Merit which will be discussed. In Figure A.3 $\eta_{Vin}$ is plotted against $\alpha$. For $\eta_{Pin}$, at $\alpha = >13^\circ$ the invested energy is $< O(0.1)$ of the energy saved. At angles below this $\alpha$ however, it is clear that there is little overall benefit to the system as the input to the flow control system is comparable to the power savings generated. This demonstrates that the low-power nature of the piezo actuators may mean a ‘break-even’ performance threshold for an overall benefit from flow control system has potential to be met, in certain drag reduction applications. This is in addition to the accompanying increase in lift.

Figure A.3. $\eta_{Pin}$ against angle of incidence ($\alpha$).
Further consideration of the efficiency of the flow control can be made by looking at the Aerodynamic Figure of Merit criterion as proposed by Seifert (Seifert 2014);

\[
AFM1 = \frac{U_a L_p}{(U_a D_p + P_a)} (L/D)_a
\]  

\[
AFM 4 = \frac{U_a D_p + P_a}{U_a D_p}
\]  

Where \( L \) is the Lift. AFM1 is pertinent to the efficiency enhancement due to the active flow control system, while AFM4 is in consideration of drag reduction purposes. Results above unity equate to conditions where activation of the flow control system is energy efficient. The results are shown in Figure A.4

The results for AFM1 show that at \( \alpha = <14^\circ \), the flow control has no real benefit to overall efficiency (AFM1) or drag reduction (AFM4). At \( \alpha = 14^\circ \) however, there is a significant increase and the flow control system is beneficial to efficiency. Of interest is that this occurs within a relevant flow condition for the aerofoil performance envelope, i.e. the flow is on the verge of separation, at a \( C_{L_{\text{max}}} \) condition, so improvements in efficiency at this flow-state extend the overall performance envelope. The results corroborate with the previous findings of Siefert, in that piezo element based systems can attain energy efficiency above unity for
separation control in low speed flows (Seifert 2014). These findings are a relevant result for overall system efficiency, in terms of providing a useful, practical separation control effect, from a lightweight, compact actuator, which is a key requirement for such systems being accepted by industry.

A.5 Discussion

The results of the measurements conducted specifically on the jet flow have been of interest to consider. As a flow control system for use in real-world applications requires a reliable and efficient operation, these results have been of interest to quantify. The longevity of the system, and specifically the piezo elements, was seen to be ~ 6 hours of continual operation. Such a timescale would be adequate to use in the identified real-world applications of motorsport events, in terms of being integrated into a race car. In real-world application however, a number of further limitations to lifespan for the piezo elements would be encountered; such as modes of vibration applied to the disc other than its natural modes, significant temperature variations, contaminant ingress, and further factors which may limit the lifespan of a SJA design. These are further considerations for systems that move further towards real-world use.

Analysis of the power consumption of the actuators identified that for a lift enhancement mechanism, desirable energy efficiency levels could be achieved. From the tests conducted, certain flow states can be improved with a fractional amount of invested energy required to achieve it. These results were found for the operating parameters of the disc which created sufficient jet output to impart control authority, and with an adequate operating lifespan. However, these preliminary results would need to be looked at in further detail in order to assess if this result could scale within the requirements of a real-world application, as such parameters are likely to differ to those used in the study here.

As has been seen, increased jet velocities are closely linked to the input voltage magnitude. The scaling of jet velocities and frequencies for differing flow states, yet at efficient energy input levels, and with a desirable actuator lifespan, is complex multi-parameter consideration. This is challenging to understand and optimise. The viability of a flow control system also needs to consider additional factors such as the mass of the system and the energy conversion efficiency in a real-world application, as outlined by Crowther (Crowther 2008). Further development of actuators in order to attain longer operating life, higher momentum output, higher operating frequencies, and less variability would be of significant interest, in order to
make the flow control system practical for potential applications, such as on a large air or ground vehicle.

A.6 Summary

Measurements of the actuator were conducted that concentrated on the practicalities that need to be considered in order to generate a robust, and relevant flow control system. These were namely consideration of the repeatability and longevity of the jet flows across the entire array. The results have demonstrated that the array produces an output that is sufficiently robust, such that it could operate for timescales that could be practical in other applications. Power requirements of the system have also shown that there is potential to generate a net benefit when used in similar separation control applications. There would therefore be interest in further assessment and development of such systems, in order to better understand if they could adequately scale to the requirements of real world applications based on the results seen here.
Appendix B  Experimental Uncertainty

This chapter discusses the quality of the experimental results for the various measurement methods employed during the research.

B.1 Force Measurements

For a number of tests early in the research shown in Appendix C, data from the force balance was recorded at 200Hz for 30 seconds. A statistical analysis was performed for the results acquired from the force balance over the entire period. At $\alpha = 14^\circ$, for a $C_L$ of 1.044, a standard deviation of 0.002 was estimated. The 95% level for confidence gives an uncertainty based on this of $C_L \pm 0.004$. This result is taken at an $\alpha$ where the flow has shown to be separated towards the trailing edge. Given, this, it is assumed to be a conservative estimate for the more attached flows. The uncertainty is also seen to be an order of magnitude less than the alteration to forces which were deemed to be a condition where control authority was generated.

The standard deviation of the freestream velocity during the measurements at the constant $\alpha$ was 0.01 m/s, corresponding to an uncertainty of the $C_L \pm 0.001$. The uncertainty in $\alpha$ was estimated at $\pm 0.1^\circ$, based on the calibration method. Using the lift slope for the polar, this contribution is estimated to be $C_L \pm 0.001$. In the tests of the ground effect flow, the alteration of ride height was performed using machined holes in the telescopic struts. This uncertainty in position was estimated at $\pm 0.1$mm. Due to the limited range of ride heights tested, the contribution to $C_L$ of this uncertainty was not deemed significant to characterise.

B.2 Pressure Measurements

Pressure measurements were taken using the instrumentation discussed in Chapter 3. With pressure acquired via the use of a mechanical indexer ‘scanivalve’ system located externally to the wind tunnel, lengths of flexible tubing used to connect the system to the model. A settling period after each movement of the indexer for the pressures to equalise along the tubes was used for each test. This was approximately 30 seconds. When tests incorporated a change of wing incidence, the settling time at a new incidence was taken after a period of typically 2 minutes where the freestream velocity would settle due to the change in blockage.
Data was acquired at 20kHz at each tapping for approximately 30 seconds. Taking the standard deviation of the transducer output at each location, the maximum was seen at \(x/c = 0.004\) on the suction surface, corresponding to the suction peak. This corresponds to an uncertainty of \(C_p \pm 0.027\). For the transducer used for all time-averaged and frequency spectrum results, a Honeywell SDX01G2, the repeatability of the sensor gives an uncertainty of \(C_p \pm 0.026\).

B.3 LDA Measurements

The LDA measurements were conducted using the same traverse system for the probe throughout the research. The repeatability of the traverse position is estimated to be ±0.1mm. This is mainly due to backlash compensation in the movement of the traverse motor gearing. In order to reduce positioning variability on repeated runs, the spanwise and lateral traverses, were always conducted in the same direction from first to last measurement points. Variability in result due to this measurement position uncertainty for the axial flow about the SJA orifice axis gives an uncertainty of ±0.01 \(U/U_{\text{max}}\). This is estimated from the measurements of the mean velocity distribution over the area of the orifice for an arbitrary test number from the numerous instances conducted throughout the research.

To consider an estimate of the streamwise velocity uncertainty, a wake profile for the unactuated flow was conducted at \(z/c = 0, x/c = 1.25\), for \(\alpha = 13.5^\circ\). The greatest uncertainty in the wake is at the point of lowest velocity, where turbulence levels are most significant. For the wake profile, the uncertainty of the statistical mean streamwise velocity with a 95% confidence is of ±0.0005 \(U/U_{\infty}\), using an established method of estimate (Robins & Hayden 2008); The error can be defined by \(\varepsilon = \sigma/\sqrt{N}\) where \(N\) is the number of independent samples, and \(N\) is equal to \(T_{\text{total}}/T_0\). For each measurement, the LDA acquisition program estimates \(T_0\) by integrating the area under autocorrelation down to 25% and multiplying this value by a factor of 4. The number of independent samples can then be calculated from the total time trace length \(T_{\text{total}}\) divided by \(T_0\). Figure B.1 shows the wake profile for which uncertainty is discussed. Figure B.2 shows a plot of the estimated uncertainty for the variance with error bars plotted for a 95% confidence level. Error is defined as

\[
\varepsilon(s^2) = \frac{\sigma^2}{\sqrt{N}} \left( \frac{m_4}{\sigma^4} - \frac{N-3}{N-1} \right)^{1/2}
\]

Where \(m_4\) is the central fourth moment of the underlying distribution.
The error is greatest at the middle parts to the wake. However, uncertainty is far lower than the magnitudes of fluctuation in the wake. In Figure B.1 a profile taken with the LDA probe aligned at 90° to the xy measurement plane, as opposed to the 4° out-of-plane orientation used throughout all measurements (which was required for measurements close to the wing surface), is plotted. It can be seen that the translation had little effect on the time-average flow, as both profiles are in good agreement.

Figure B.1  □, LDA probe orientation = 0°; ○, LDA probe orientation = -4° at z/c = 0, x/c = 1.25, for α = 13.5°

Figure B.2  95% confidence error bars for estimated uncertainty in $(\overline{u'}^2)/U_{\infty}^2$ at z/c = 0, x/c = 1.25, for α = 13.5°.
B.4 PIV Measurements

PIV data was collected in a number of tests over the duration of the research. The same general test procedures were maintained in order to minimise variations from test to test. The seeding in the tunnel was an important consideration for maintaining image quality throughout the duration of a test. 1000 image pairs were collected during each measurement at 7.25Hz, so consistent seeding levels needed to be maintained for a period ~140 seconds during each test. The closed-return configuration of the wind tunnel allowed the seeding level to be kept consistent by applying an automated duty-cycle to the running of the seeding system. Images were checked for seeding levels before the start of each test through the capture and processing of a smaller dataset; the full dataset was acquired shortly after with the same seeding strategy.

Uncertainty of the result is estimated by considering average particle displacements for the maximum flow speeds in the streamwise direction. This propagated error will vary with flow speed. Based on this however, the uncertainty in the PIV measurements was estimated to be to a maximum of 3%, due to an average displacement of 7 ±0.2 pixels in the image pairs.

When considering the uncertainty of the positioning of the flow field, the traverse system used throughout the tests had an overall positional uncertainty for repeated positioning of the equipment of ±0.1mm, which is equivalent to 0.0002c.

With the processing of images, all statistics of the flow field were post-processed in an automated manner, using a LabVIEW program to compute the statistical results. Given the significant size of the datasets, this program was also used in order to check the validity of the flow field in terms of the percentage of the field with uncorrected (‘true’) vectors, and interpolated vectors. This was done for all tests. The program ensured that calculations of the mean results were not based on any spurious vectors that can commonly be generated when problems arise with the cross-correlation algorithm, yet an unrealistic vector is still computed. Random inclusions into the mean results of spurious vectors were therefore able to be minimised. Typically >90% of a vector field would be made up of true vectors for a typical acquisition setup.
Appendix C  Numerical Simulation of the Wind Tunnel Model

C.1 Introduction

An active flow control (AFC) research programme at Surrey Advanced Vehicle Analysis Group was undertaken with experimental testing of an aerofoil wing model, in the ‘Aero Tunnel’ at the Enflo Laboratory.

With the model having not been tested in this or another facility prior to the research programme, preliminary measurements of the model performance were compared to numerical simulations. The potential to predict the detailed characteristics of the separated flows with numerical simulation were also assessed.

C.2 Wind Tunnel

The tunnel is a 1.065 x 1.37m cross section closed working section, closed return design. The maximum freestream velocity is 40m/s. The wind tunnel model was a 0.43m chord by 1.06m span NACA0015 profile. It mounts to a six component overhead balance, via a two arm main strut and angle adjustment single arm strut. The balance is located in the working section, and shrouded with a NACA0030 profile fairing with rounded base tip. The balance fairing with model at a zero degree angle of attack corresponds to a tunnel solid body blockage ratio of ~11%. The model is orientated so that the suction surface is facing the floor of the working section, creating downforce when the wing angle to the flow is >~0°. The model is able to move through an angular range between -5 and 25° ±0.1°.

C.3 Experimental Setup

Data were taken from the range $\alpha = 0^\circ - 20^\circ$ in $1^\circ$ increments with a wind speed of 30m/s, giving a test Re of ~ 8.9 x 10^5, based on the wing chord. The model had a painted metallic outer surface that was aerodynamically smooth, with free transition. The model had openings on both surfaces for a spanwise array of AFC actuators to be installed for later tests. When these were blanked off with infill plates to produce a surface without orifices, small discontinuities and a slight change in surface roughness was introduced at this point on the surface. Measurements of static pressure at 30 pressure ports normal to the wing surface were taken. 20 ports were on the suction surface, 8 on the pressure and one at the chord line.
leading and trailing edges respectively. The mean readings for the forces were taken from data collected at 200Hz for 15 minutes at each angle, as pressures were acquired for 30 seconds at each port sequentially. Data were taken during both positive change in the angle and negative, to assess the effects of flow hysteresis. The results from increasing angles of attack only are shown in the results below, representing the flow transitioning from attached to separated states.

C.4 Numerical Setup

Simulations were run for wing inclination angles of 0° to 20° in 4° increments. Simulations were performed using STAR-CD V4 Computational Fluid Dynamics (CFD) code, to run Reynolds Averaged Navier Stokes (RANS) solutions for the wind tunnel working section flows. The wind tunnel setup was three-dimensionally modelled. The prominent features to the model that could alter a characteristic NACA0015 flow structure were included in the model. These were namely the strut, balance fairing and wind tunnel walls. The geometry was modelled with the flow inlet being positioned 6.49c upstream of the leading edge, corresponding to the start of the working section. The outlet was positioned 7.23c downstream of the wing leading edge. The physical model is designed so it spans the working section, though with a 2±1mm gap to the tunnel walls to prevent bridging of the balance. The CFD model was simplified to merge the model surface and wall. This merging of the surfaces along with several other simplifications was added in order to allow efficient discretization of the fluid domain. This kept mesh sizing and therefore solution times, to sensible levels. These simplifications were namely the removal of the sting attachment points to the wing. The fluid cell mesh was produced using the structured trimmed hexahedral cell technique incorporated into CD-Adapco PRO-STAR pre-processor. The number of cells would range from ~8 x 10^6 to ~10 x 10^6, depending on the wing inclination angle being modelled. The Z direction dimension of the fine mesh domain at the wing surface, (shown in Figure C.1) increased with wing inclination. Five levels of mesh refinement were used around the model, that resulted in y+<~150 at the model surfaces at all wing angles tested, and freestream velocities. The inlet was a turbulent flow, and specified by the turbulent kinetic energy (K), and its dissipation (ε). The simulation used values for these typical for a low turbulence automotive wind tunnel, which corresponds to a turbulence intensity of 0.27%. The outlet was specified with a constant pressure condition. The inlet flow was a wind speed of 35m/s, giving a test Re of ~1 x 10^6, based on the wing chord.
The wind tunnel walls, floor, and wind tunnel model were modelled as aerodynamically smooth, and set as no-slip conditions. The Semi Implicit Method for Pressure Linked Equations (SIMPLE) algorithm was used for the pressure velocity coupling of the solution. The differencing scheme to evaluate convective and diffusive fluxes at the control faces was the Upwind Differencing scheme for the turbulence model and momentum equations. For the mass equations, the Central Differencing scheme was used. The turbulence model used was the High Reynolds K-ε. Simulations were run on a high performance computing cluster, to allow parallel computation of the model. The cluster has 8 nodes, each with two, hex-core 3 GHz processors and an infiniband interconnect architecture, which reduced solution times to practical levels, of ~2.5 hours when using 48 CPU’s.

C.5 Results

The experimental and numerical static pressure coefficients ($C_p$), as a function of $x/c$ position are presented below. At $\alpha =0^\circ$, across the centreline the profile is producing lift. This is due to
the increased low pressure magnitude on the upper surface, with respect to that of the lower. This is deemed due to the larger blockage ratio in the upper half of the wind tunnel, and hence higher fluid velocities in the freestream around the upper surface. Similar trends are seen in the results. Both compare well in the \( x/c \) positioning of the low pressure peak.

Figure C.2. Coefficient of Static Pressure (\( C_p \)) against chordwise location (\( x/c \)) over the pressure and suction surfaces of the aerofoil, at incidence angle (\( \alpha \)) = 0°, where; □ = Experimental, pressure surface; ○ = Experimental, suction surface; — — — — = Numerical, suction surface; — — — — = Experimental, pressure surface.

Lift and drag forces on the model were calculated for = 0°. This was by summation of the forces at each wall boundary cell face on the model. Force is calculated from the static pressure acting on the normal component of the cell area in the horizontal and vertical planes. The non-dimensionalised results are summarised in table C.1.
The results were not seen to be in close agreement to the experimental case. The drag force is significantly under predicted. A sensitivity study was therefore undertaken to look at the effects of alteration of the boundary conditions and the model geometry to better correlate the results.

Improvement was found from the addition of further detail to the geometry, by adding the wall gap and a simplified surface for the tail strut. These details acted to increase drag and reduce lift. The walls of the wind tunnel were re-modelled as a slip wall. The result can be seen in table C.2. Additional detail to the model was seen to improve the results agreement in drag.

<table>
<thead>
<tr>
<th>$C_D$ Num</th>
<th>$C_D$ Exp</th>
<th>$C_D$ Num/$C_D$ Exp</th>
<th>$C_L$ Num</th>
<th>$C_L$ Exp</th>
<th>$C_L$ Num/$C_L$ Exp</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.0255</td>
<td>0.0477±0.00008</td>
<td>0.535</td>
<td>0.0461</td>
<td>0.0510±0.0002</td>
<td>0.903</td>
</tr>
</tbody>
</table>

Table C.2: Coefficient of force results for $\alpha =0^\circ$ with refined numerical model.

A wake survey was carried out at $\alpha = 0^\circ$. The numerical model results can be compared to the experimental result. This was with the higher fidelity model discussed previously, with a simplified tail sting and a wall gap added. The simulations were those carried out at $Re \approx 1 \times 10^6$, while the experiments were performed at $Re \approx 8.9 \times 10^5$. The velocity components are non-dimensionalised by the freestream velocity of each test. Data were taken with a Dantec Fibreflow 2D Laser Doppler Anemometry (LDA) system, measuring the streamwise flow. Measurements were taken through windows into the working section, at a position $0.25c$ downstream of the wing. The measured region, outboard of the semi span plane can be seen in figure C.3.
Figure C.3 Wake flow $U/U_\infty$ at $x/c = 1.25 \alpha = 0^\circ$; a) measurement point isometric view; b) experimental; c) numerical.
The width of the wake from the tail sting is under predicted by the simulation. This bluff body separation would lead to the production of drag. This lack of agreement could be part reasoning to the force results differences seen previously. The experimental result shows an inboard movement of the centre point of the velocity deficit around the strut base. With the numerical simulation, no noticeable spanwise movement of the velocity deficit around the strut is predicted. This suggests that the flows have a significant three dimensionality close to the strut which the simulation is failing to accurately predict.

Further simulations at $\alpha \neq 0^\circ$ using the simplified model used initially were performed, in order to assess the key aspect of large scale changes in the drag and downforce levels, with the change in $\alpha$. The force coefficients across the angle range are shown in figure C.4 and C.5 with comparison to forces measured experimentally. Error bars display ±1 standard error of the mean. Pressure data at each tapping location is used to derive a force result based on integrated pressures (see Notes). For the attached flows, up to $\alpha = 12^\circ$, the balance data and the numerical result for lift show good agreement. At larger $\alpha$, the agreement of the results is less. These angles generate separated flows. The numerical prediction is for a significantly higher lift force than that seen with the experimental data at these angles. Comparison of balance data and numerical results are shown in Table C.3.

![Figure C.4: Lift Coefficient ($C_L$) against incidence angle ($\alpha$). □ = $C_L$, Integrated Pressures; ○ = $C_L$, Balance; — = $C_L$, Numerical.](image-url)
The integrated pressures force data will be a prediction of the forces generated by a ‘clean’ wing geometry, where no wall or strut interference effects are experienced. Good agreement in trends is seen between the numerical predictions and the balance data, at angles of attack corresponding to the region of the polar when the flow is still attached before $C_L_{\text{max}}$. The pressure derived predictions of lift and drag are lower than the balance data. At the highest angles of attack numerical model prediction of drag is closer to that extrapolated from the quasi two dimensional pressure distributions. This would suggest the complexity to the partially and fully separated flow are not particularity well predicted by the numerical result.

Understanding a prediction of trailing edge separation was of interest. Particle Image Velocimetry (PIV) measurements were taken to understand the flows at $\alpha = 13.5^\circ$. A TSI Powerview 2D PIV system was used. Velocity vectors were calculated over a region close to the trailing edge. 200 image pairs were recorded. The instantaneous results were then averaged to give a time-averaged flow field. The numerical analysis was the same simulation setup as previously discussed, and performed at $\alpha = 13.5^\circ$. $Re$ for both tests was $\sim 1.1 \times 10^6$, based on the characteristic length being the model chord, and a freestream velocity of 40m/s.

---

**Table C.3: Summary of Coefficient of force results.**

<table>
<thead>
<tr>
<th>$\alpha$</th>
<th>$C_D_{\text{Num}}$</th>
<th>$C_D_{\text{Exp}}$</th>
<th>$C_D_{\text{Num}}/C_D_{\text{Exp}}$</th>
<th>$C_L_{\text{Num}}$</th>
<th>$C_L_{\text{Exp}}$</th>
<th>$C_L_{\text{Num}}/C_L_{\text{Exp}}$</th>
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</tr>
<tr>
<td>12</td>
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<td>0.056 ±5.8e-4</td>
<td>0.956</td>
<td>1.055</td>
<td>1.059 ±1.1e-3</td>
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</tr>
<tr>
<td>20</td>
<td>0.207</td>
<td>0.333 ±2.1e-4</td>
<td>0.621</td>
<td>0.915</td>
<td>0.581 ±1.7e-3</td>
<td>1.574</td>
</tr>
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</table>
The simulation predicts less velocity deficit in the near wall region flow at the trailing edge. The simulation predicts continuing boundary layer growth in the adverse pressure gradient that develops towards the trailing edge. However, the experimental data shows that the flow is prone to separation at a point further upstream on the chord. A vertical profile of the streamwise velocity (shown in figure C.6) at the trailing edge shows flow reversal in the experimental data reaches \( U_{\infty} \), while the numerical prediction is for no reversed flow. The height of the velocity deficit region it also significantly under estimated by the numerical result in comparison to the experiments.

![Figure C.6 Wake flow \( U/U_{\infty} \) at \( x/c = 1 \), \( \alpha = 13.5^\circ \) □ = Experimental; ○ = Numerical.](image)

C.6 Conclusions

The study of the NACA0015 wind tunnel model has demonstrated the performance of the model over a large range of inclination angles, both numerically and experimentally. Numerical results and experimental results show overall force generation and surface pressures to be in good agreement for the attached flows. The wind tunnel model generated a characteristic NACA0015 performance at low angles of attack, and a trailing edge separation as the inclination approaches the critical angle of attack. With the flows encountered at higher angles of attack where the boundary layer is prone to separation, however, the agreement seen with the numerical result is far less. Such numerical simulations are therefore not applicable in order to predict the characteristics of the separated flows over the wing at high angles of attack. The specific characteristics of the separated flow of the Surrey NACA0015
model therefore need to be understood experimentally, as opposed to considering numerical predictions. Prior to conducting investigations of active flow control system performance, such results need to be collected.

Notes:

Integrated Pressure Method;

Knowledge of the discrete pressure measurements \(p\) around the aerofoil can allow an overall force to be calculated, based on the assumption of a constant pressure distribution in the spanwise direction; and using trapezoidal rule integration; The forces can be calculated by;

\[
F_y = w \int_{o}^{c} (p_l - p_u) \, dx
\]

\[
F_x = w \int_{y_{min}}^{y_{max}} (p_f - p_r) \, dy
\]

Such that the forces based on the inclination of the aerofoil \(\alpha\) can be resolved by;

\[L = F_y \cos(\alpha) - F_x \sin(\alpha)\]

\[D = F_x \sin(\alpha) + F_y \cos(\alpha)\]

And the respective coefficients can be calculated;

\[C_L = \frac{L}{\frac{1}{2} \rho V_{\infty}^2 \, A}\]

\[C_D = \frac{D}{\frac{1}{2} \rho V_{\infty}^2 \, A}\]
# Appendix D  Pressure Tapping Locations

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</table>

Unsteady Pressure Transducer          0.76  -0.038196  0.17  
Unsteady Pressure Transducer          1     0       0      

200