

**LEADING-EDGE FLOW SEPARATION CONTROL
OVER A NACA 0012 AEROFOIL WITH DBD
PLASMA ACTUATORS**

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Abstract

An experimental investigation has been conducted in a low-speed wind tunnel at the University of Nottingham to study the flow separation control capability of a wall-normal plasma jet by DBD plasma actuator over a NACA 0012 aerofoil. As an active flow separation control technique, DBD plasma actuators could be applied when required to manipulate a flow. They are surface-mounted and require no moving parts, ducts, holes or cavities, so no profile drag penalty will be caused. Moreover, they are fast responding since they are purely electrical devices and could be operated at a higher frequency relative to other flow control techniques. DBD plasma actuators are easy to manufacture, low in weight, low energy consuming and can be easily fitted to aerofoils. Therefore, they are ideal tools to control the flow separation around aerofoil. Up to date, wall plasma jet was used to add momentum to flow directly so that flow becomes more energetic and capable of withstanding adverse pressure gradient. In this study, a wall-normal plasma jet by steady actuation of plasma actuator was investigated and PIV results show that it has the capability of controlling the separation around aerofoil at post-stall angles of attack. The wall-normal jet is bent towards freestream direction and some small-scale vortical structures are created due to the interaction between the wall-normal plasma jet and freestream. These vortical structures could promote mixing and transport high-momentum fluids into the boundary layer, which affects the flow above the suction surface significantly. Moreover, unsteady actuation of plasma actuator was also utilised to control the flow separation around aerofoil. It was found that it has a stronger ability to control flow separation even at a much lower energy consumption than steady actuation of plasma actuator. PIV measurements demonstrate that separated flow could be reattached at post-stall angle of attack of 14° with only 10% of the energy consumption by steady actuation. Flow is well organized and a series of large-scale vortices are created with periodic activation of plasma actuator, these vortices enhance entrainment and the outwards transport of fluids from aerofoil surface leads to a favourable pressure gradient, resulting in a control of flow separation.

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List of Symbols

α	angle of attack
c	aerofoil chord length
C_μ	momentum coefficient
d	length of normal projection of aerofoil on a cross-stream plane
E_{p-p}	voltage
f	frequency
f_0	estimated vortex shedding frequency
f_e	excitation frequency
$f^+ = f_e / f_0$	normalized excitation frequency
ρ	air density
St	Strouhal number
Re	Reynolds number
t	time instant
$t^+ = tU_\infty/c$	normalized time instant
τ	duty cycle
u'	root mean square of streamwise velocity
v'	root mean square of vertical velocity
U_∞	freestream velocity
U	streamwise velocity
V	vertical velocity
ω_z	spanwise vorticity

x	streamwise distance
y	wall-normal distance
γ	separation angle
$\delta_{1/2}$	half width of plasma jet
$\int \omega_z U dy$	vorticity flux
$\int \rho U^2 dy$	momentum flux
φ	phase angle

1 Introduction

1.1 Nature of the problem

Flow separation, which normally occurs due to an adverse pressure gradient, is of great significance to engineering applications because it degrades the efficiency of aerodynamic vehicles. Specifically, it results in an abrupt decrease in lift and a sharp increase in drag, which might cause a flight accident. Therefore, a control of flow separation is important for the aeronautical industry. This will enable the aeroplanes to fly at higher angles of attack with a higher lift coefficient so that more payloads could be carried to achieve a better economic benefit.

Strides have been made in flow separation control over the past century. Passive flow separation control techniques were proved to be quite effective in controlling flow separation and they are now widely used in aeronautical applications. However, these passive separation control devices might introduce some negative effects into the flow at off-design conditions, for example, drag is normally introduced by these passive devices.

In the recent decades, a growing number of attentions are paid to active flow control such as blowing and suction, synthetic jet, DBD plasma actuator and so on. Active flow control consists of manipulating a flow to affect a desired change (Gad-el-Hak, 2000). In other words, it utilizes the interaction between the actuated flow and the local flow to achieve flow control. The advantage of active flow control is that it can occur at the proper time and location by a local energy input to obtain desired local or global flow changes.

As one of the active flow control techniques, the dielectric-barrier-discharge (DBD) plasma actuator is usually composed of an exposed electrode and an embedded electrode, separated by a dielectric sheet (Feng et al., 2012). The electrodes are energized at a high voltage, leading to the air over the embedded electrode to ionize which induces a wall-jet flow (Jukes et al., 2006). These DBD plasma actuators are purely electric devices so that they respond quickly and can be rapidly turned on and off as required (Jukes and Choi, 2011). Moreover, they could be mounted onto

the surface, therefore no profile drag penalty will be generated when they are not being used.

The plasma actuator was proved to be able to improve the aerodynamics of an aerofoil by controlling the flow separation. Examples include separation control on a high-angle-of-attack aerofoil using plasma actuators by M. L. Post and Corke (2004a), plasma flaps and slats: an application of weakly ionized plasma actuators by He et al. (2009a), autonomous sensing and control of wing stall using a smart plasma slat by Patel et al. (2007) and so on. The results of these studies demonstrate that separation could be effectively controlled via the use of DBD plasma actuator and an improvement in aerodynamic performance such as lift-to-drag ratio could be achieved. The previous work showed a promising future of DBD plasma actuator in flow separation control.

However, to the author's knowledge, most of these researches utilize downstream-directed wall jets by plasma actuator to control flow separation through the direct addition of plasma momentum. This research aims to control the flow separation around a NACA 0012 aerofoil with a wall-normal plasma jet at the leading edge. This is a totally different way of flow separation control compared with wall plasma jet, specifically, the interaction between the freestream and wall-normal plasma jet produces spanwise vorticity which promotes mixing and introduces the high-momentum fluid into boundary layer, thus enabling the boundary layer to withstand adverse pressure gradient. Moreover, periodic activations of the wall-normal plasma jet would also be utilized to control the flow separation.

1.2 Thesis outline

Chapter 2 is the literature review part of this thesis, which gives a discussion on flow separation, passive leading-edge separation control techniques, active leading-edge separation control techniques and flow separation control with DBD plasma actuators. This helps develop an understanding of background of this research. Chapter 3 describes the setup of wind tunnel, aerofoil, plasma actuator generation system and PIV for the measurements conducted in this PhD project. The characteristics of the baseline flow around a NACA 0012 aerofoil at different angles of attack and freestream velocity of 3m/s are summarized in Chapter 4. Chapter 5 provides a discussion on wall-normal plasma jet characteristics and the

effect of excitation parameters on the wall-normal plasma jet in initially quiescent air. Chapters 6 and 7 are concerned with flow separation control over a NACA 0012 aerofoil with steady and unsteady actuation of DBD plasma actuators respectively. Parametric effects of actuation signals and mechanism exploration were conducted through PIV measurements. Conclusion of this thesis and recommendation for future work are given in Chapter 8.

2 Literature review

2.1 Introduction

The aerodynamic performance of aircrafts is significantly influenced by the leading-edge separation, for example, lift decreases sharply and drag increases dramatically when stall occurs, which might result in a flight accident. Therefore, it's really of great significance to control the flow separation over an aerofoil. This chapter gives a review of flow separation, passive separation control, active separation control and separation control with DBD plasma actuators.

2.2 Flow Separation

Flow separation always refers to a breakaway or departure of fluid particles from a solid surface, which is usually caused by a severe adverse pressure gradient (Chang 1976, Tel 1979, Mohamed & Bushnell 1991, Greenblatt & Wygnanski 2000). In 1904, Prandtl (1904) first introduced boundary layer theory and explained the separation of a steady, two-dimensional boundary layer. From his point of view, flow will be retarded when it undergoes adverse pressure gradient. In other words, if the adverse pressure gradient is quite severe, flow will not be able to continue its motion and a reverse flow region will appear. This is quite widely accepted as the mechanism for flow separation. The point between the forward and backward flow is defined as separation point, which coincides with the point where the velocity gradient $\left[\frac{\partial U}{\partial y}\right]_{y=0}$ and shear stress are zero for steady, two-dimensional flow separation. In the flow separation region, there always appear a significant thickening of rotational flow and great increase in the wall-normal velocity value (Greenblatt and Wygnanski, 2000).

Flow separation is always accompanied by a sharp increase in pressure drag, an abrupt decrease in lift and even buffeting. Therefore, a physical understanding of flow separation is of essential significance in order to improve the performance of the related practical devices such as aerofoil wings, wind turbine blades and so on. Actually a number of experimental and numerical efforts have been conducted on flow separation over the past century to improve our understanding of flow

separation (Boutilier and Yarusevych, 2012; Huang and Lin, 1995; L. E. Jones et al., 2008; Numbers, 2010; Russell, 1979; Tel, 1979; Yarusevych et al., 2009, 2006).

For the two-dimensional, low Reynolds number laminar flow separation over an aerofoil at moderate angles of attack, disturbance amplifies dramatically downstream of the separation point, resulting in a transition to turbulence that is more capable of withstanding adverse pressure gradient, and the separated flow reattaches to the aerofoil surface, forming a separation bubble. The structure of a separation bubble is demonstrated in Figure 2-1 (O'Meara, M.M. and Mueller, 1987), which consists of two regions. One is formed between the aerofoil surface and the mean dividing streamline, where recirculation occurs. The other is the region between the dividing streamline and the outer edge of the boundary layer. In this region, the laminar detached shear layer undergoes transition to turbulent detached shear layer, which is more energetic and has a greater capability of tolerating adverse pressure gradient, thus flow reattaches to the surface downstream (Horton 1968; Meara & Muellert 1987; Sandham 2008; Jahanmiri, 2011).

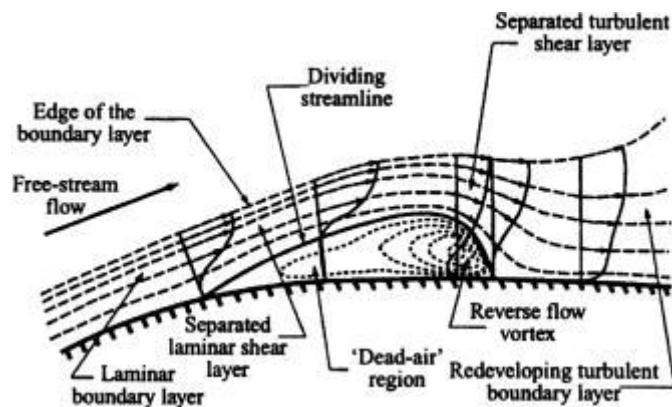


Figure 2-1 Schematic of laminar separation bubble (O'Meara, M.M. and Mueller, 1987).

The factors that affect flow separation were also investigated. Figure 2-2 shows the effect of Reynolds number on separation bubble length at different disturbance levels. As chord Reynolds number increases, the total length of separation bubble decreases. The reason might be that transition occurs earlier at higher Reynolds numbers, leading to an earlier reattachment of the separated flow. Also can be seen is that separation bubble length is reduced with an increase in disturbance.

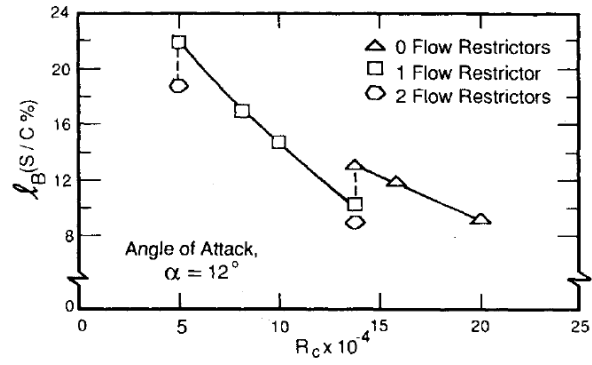


Figure 2-2 Laminar separation bubble length versus chord Reynolds number (O'Meara, M.M. and Mueller, 1987).

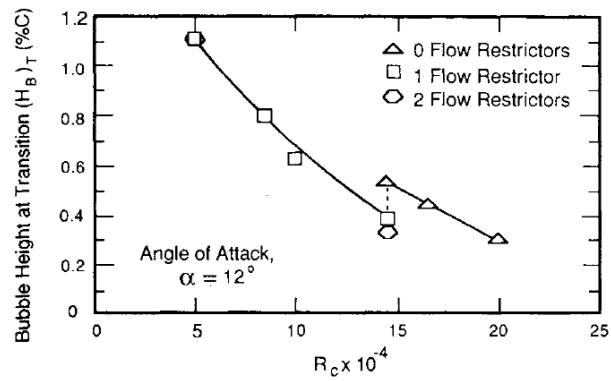


Figure 2-3 Laminar separation bubble height versus Reynolds number (O'Meara, M.M. and Mueller, 1987).

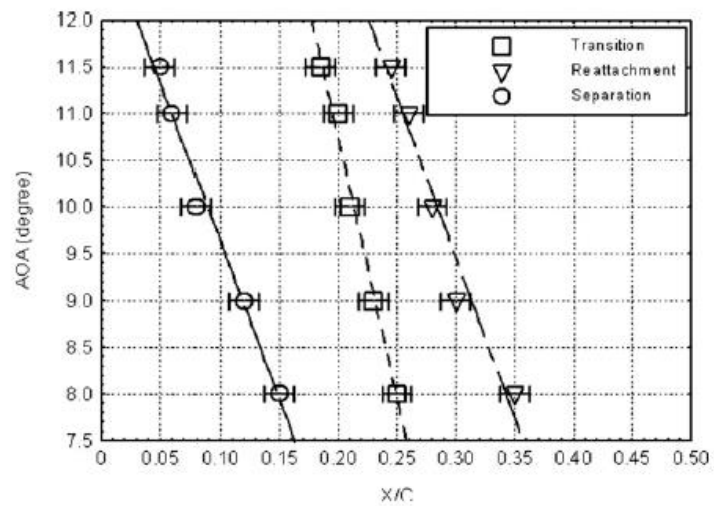


Figure 2-4 The location of separation point, transition point and reattachment point at different angles of attack (Hu and Yang, 2008).

Reynolds number and external disturbance also have an impact on the height of separation bubble, as illustrated in Figure 2-3. The thickness of the bubble reduces sharply when chord Reynolds number and disturbance become larger (O'Meara, M.M. and Mueller, 1987).

Angle of attack could also affect flow separation. The length of the laminar portion of separated shear layer slightly enlarges with an increase in angle of attack while that of the turbulent portion decreases (see Figure 2-4). What's also found is that the length of the separation bubble remains almost constant before the bubble bursts at stall and post-stall angles of attack (Hu and Yang, 2008).

The impact of flow separation on pressure distribution over an aerofoil is demonstrated in Figure 2-5. According to the theoretical model by Russell (1979), two portions consist of the detached shear layer: laminar portion and turbulent portion. The plateau of the pressure represents the laminar part of the separated shear layer, and the separation point, where flow starts to separate, can be indicated by the starting of the plateau. As turbulence will entrain fluid, the pressure would increase when transition begins, therefore, the end tip of the pressure plateau is the transition point. The pressure distributions of separation bubble and inviscid flow are quite different, so the point of equity between the actual and inviscid flow can be used to indicate the reattachment point, where flow reattaches to the upper surface of the aerofoil.

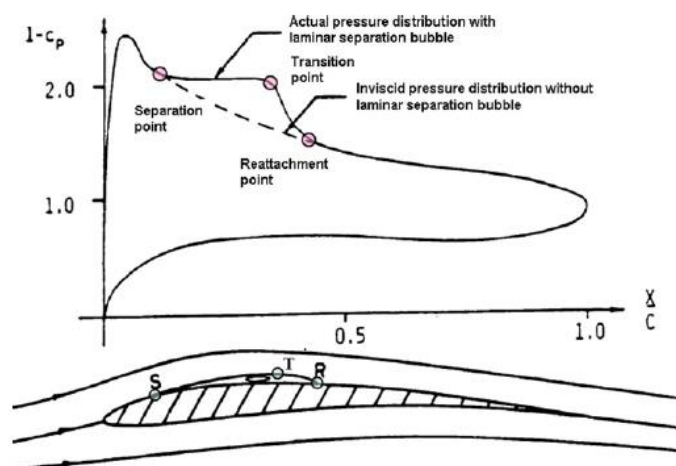


Figure 2-5 Pressure distribution over an aerofoil with flow separated on part of the upper surface (Russell, 1979).

Flow separation has a significant effect on lift and drag coefficients of an aerofoil, as shown in Figure 2-6, where the lift and drag curves of a GA (W)-1 aerofoil at Reynolds number of around 70,000 are presented. When the adverse pressure gradient is mild, for example, $\text{AOA} < 6^\circ$, there is no flow separation around the aerofoil, in this case, the lift coefficient increases linearly with angle of attack and the drag coefficient remains quite small. When the angle of attack is larger, the adverse pressure gradient increases and the flow separates from the upper surface of the aerofoil. After the transition to turbulence, flow reattaches to the surface because turbulence is more capable of withstanding adverse pressure gradient. Therefore, a separation bubble forms. Lift coefficient keeps increasing with angle of attack although the increase rate is found to decrease dramatically because of the appearance of separation bubble. On the other hand, drag coefficient increases faster as the angle of attack increases. If the aerofoil is operated at stall or post-stall angles of attack, at which the adverse pressure gradient is really big, separation bubble will burst and the flow will never reattach to the surface of the aerofoil, an example of this is illustrated in Figure 2-7 which shows the streamline of the mean flow field around an aerofoil at post-stall angles of attack. As a consequence, a sharp increase in drag coefficient and a significant decrease in lift coefficient are caused (Hu and Yang, 2008).

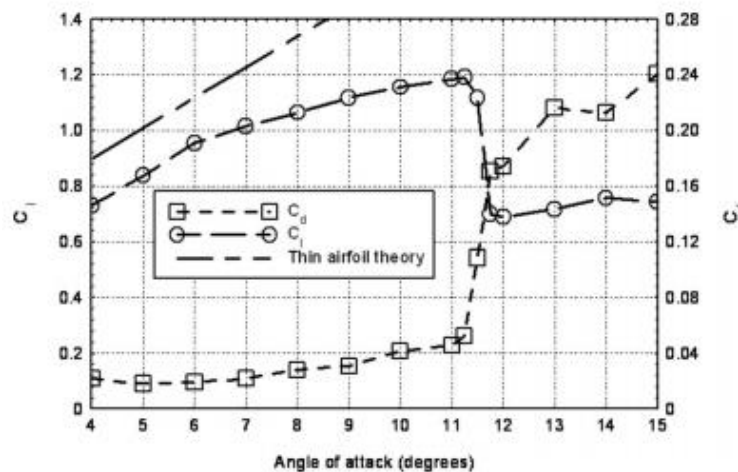


Figure 2-6 Lift and drag coefficient of an aerofoil against angle of attack (Hu and Yang, 2008).

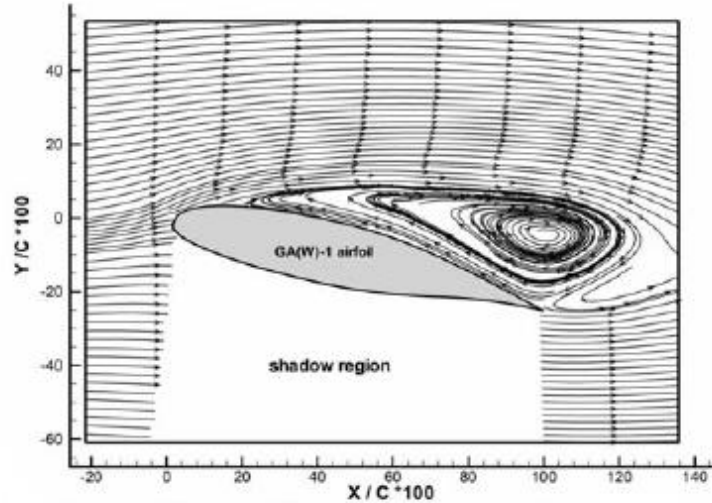


Figure 2-7 Time-averaged flow field with $\text{AOA} = 12^\circ$ (Hu and Yang, 2008).

2.3 Passive Leading-edge Separation Control Techniques

2.3.1 Fixed Leading-edge Slots

Leading-edge slots are fixed devices to delay flow separation, reduce the stall speed and improve the low-speed handling qualities. Usually, leading-edge slots are placed behind the leading-edge of the wing (see Figure 2-8). When air flows through the narrow slot from the lower surface, it will be accelerated. This accelerated flow could re-energize the boundary layer on the upper surface, enabling the air to flow further downstream and delaying separation.



Figure 2-8 Leading-edge slot (Roskam and Lan, 2003)

Leading-edge slots are very simple and effective means to delay separation. However, slots also cause significant drag penalty in high-speed flights, this increases fuel consumption per unit distance flown.

2.3.2 Leading-edge Slats

Slats are aerodynamic surfaces placed on the leading edge of wings (see Figure 2-9), which could delay the flow separation, thus enabling the aircrafts to operate at higher angles of attack. When deployed, the wing could produce higher lift

coefficient and the aircraft would be able to fly at lower speeds and take off or land in a shorter distance.

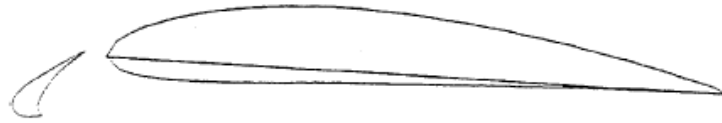


Figure 2-9 Leading-edge slats (Maddah et al., 2002)

Leading-edge slats were first developed almost one hundred years ago. In the past years, a lot of researches have conducted on leading-edge slats and a number of slats have been developed. Usually, they can be categorized as three types:

- a) Fixed — the slats are fixed in front of the nose of the main wing element and permanently extended. The advantage of these slats is simplicity, however, drag will increase rapidly when the flight speed grows, therefore, this kind of slats are only sometimes used on low-speed aircrafts.
- b) Automatic — the slats could be extended or retracted automatically by aerodynamic forces when required. Specifically, the slats will be retracted at low angles of attack and the reduced aerodynamic forces will make the slats extend at high angles of attack.
- c) Powered — the extension of this kind of slats can be controlled by pilots.

The principle of leading-edge slats has been examined and their beneficial aerodynamic action is due to five primary effects (Smith, 1975):

- a) Slat effect — the slat causes circulation around its trailing edge, the velocities due to this circulation run counter to the gap flow. Thus the velocity along the leading edge of the main element will be reduced. As a result, the severity of the adverse pressure gradient decreases and separation delays.
- b) Circulation effect — the main element enhances the flow circulation around the trailing edge of the slat, causing a higher lift.

- c) Dumping effect — because of the circulation effect, the velocity of the boundary layer along the slat's trailing edge is high, relieving the pressure rise.
- d) Off-the-surface pressure recovery — the boundary layers dumped from the slat's trailing edge decelerates to meet the pressure-rise requirements in an efficient way, out of a wall.
- e) Fresh-boundary-layer effect — the boundary layers on each element of the wing are independent with each other, this allows a fresh thin boundary layer. It is well known that thinner boundary layers are able to undertake severer adverse pressure gradients than thicker ones. So this effect delays separation.

The leading-edge slats are widely used on aircrafts and their advantages are obvious: delaying separation, improvements in maneuverability, higher maximum lift coefficient, shorter takeoff / landing distance and so on. Nevertheless, they also have some inherent disadvantages such as complexity, greater drag, masses, radar cross section, noise and vibration, etc.

2.3.3 Leading-edge Cuff

Leading-edge cuff is also called drooped leading edge. It is an outboard leading-edge modification (see Figure 2-10). It is used to produce a gradual pressure gradient over the leading edge and make the stall gentle, not sharp. With the leading-edge droop, the lift coefficient reduces smoothly instead of a normally abrupt drop at stall and obtains a relatively high lift coefficient at post-stall angles of attack (Anderson et al., 1980). Chandrasekhara et al. (2004) demonstrate that the leading-edge droop could decrease the strength of the dynamic stall vortex dramatically and delay stall. As for the mechanism of this outboard leading-edge modification, it decreases the angle between the freestream and the drooped section of the wing, thus reducing the tendency of the airflow to separate and enter the stall state. Moreover, the vortex generated by the cuff discontinuity prevents the separated flow from propagating outboard at high angles of attack (Murri et al., 1987), this makes the stall gentle and also ensures that aileron maintains its effectiveness at high AoA angles even when stall occurs on the inboard root of the wing.



Figure 2-10 Leading-Edge Cuff (Chandrasekhara et al., 2004).

Leading-edge cuff could increase the stall / spins resistance with a relatively small drag penalty (Newsom et al., 1982). However, this modification is a passive method that leads to leading-edge separation control and can't manipulate the flow as required according to the variations in the flight conditions. Moreover, the cuff splits the leading edge into several discrete sections, which may generate noise and vibration.

2.3.4 Wing Strake

Wing strake is an aerodynamic device that is mounted on the airframe from the leading edge of wing root to somewhere forward on the fuselage, Figure 2-11 shows a wing strake on F – 16. The airstream separates when it flows over the sharp strake leading edge at high angles of attack. The vortices induced by the separation over the strake travels downstream on the wing, the mixture between the strake vortices and the flow field above the wing re-energizes the boundary layer, making the boundary layer able to undertake severer adverse pressure gradient.

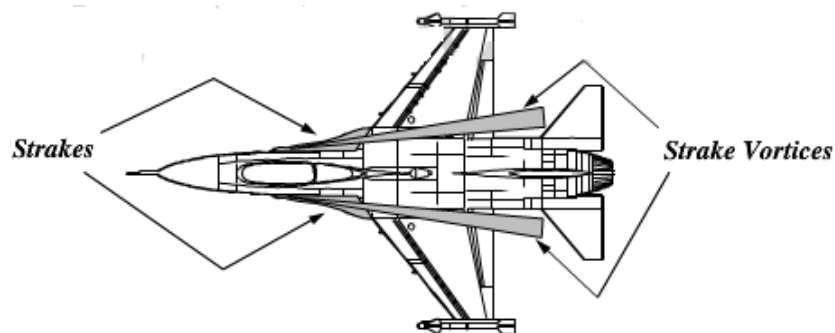


Figure 2-11 Wing strake of F-16 (Brandt et al., 2004).

The strake is widely used because it is light in weight, small in size and really simple. On the other hand, the vortices produced by strake may cause buffet problems on the tails of an aeroplane (Zuppari and Valenza, 1999). Another limitation is vortex breakdown (Frink, 1980), which could lead to serious flight problems.

2.3.5 Leading-edge Flap

Leading-edge flaps are also attractive tools to delay flow separation on the leading edge of airplane wings. These devices might be put on the upper surface or hinged at the leading edge or on the lower surface of aerofoil, as shown in Figure 2-12. Leading-edge flaps achieve separation control through the following possible mechanisms: 1) the angle between the freestream and the leading edge of the wing decreases when leading edge flaps deflected, thus the air could flow over the upper surface smoothly without separation; 2) the flaps' deflection makes the peak pressure point move downstream at high angles of attack, this extends the favourable pressure gradient region, making the separation delayed (Brandt et al., 2004). The effect of leading-edge flap on lift curve of a wing is illustrated in Figure 2-13, which demonstrates that the maximum lift coefficient is increased and the stall is postponed to a larger angle of attack with leading-edge flap deployed. Therefore, leading-edge flap could effectively control flow separation and has been widely used for aeronautical applications.

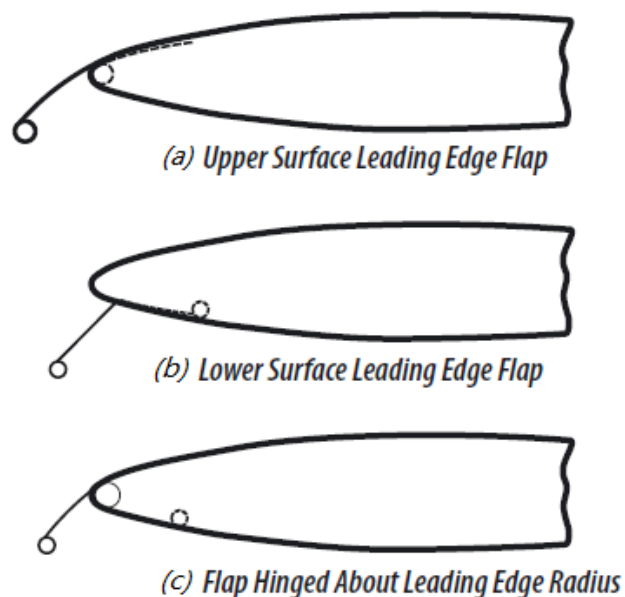


Figure 2-12 Leading-edge flaps (Nicolai and Carichner, 2010).

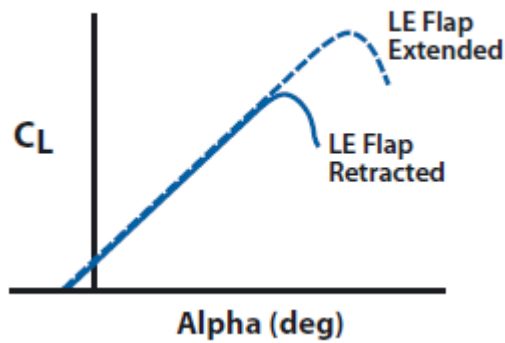


Figure 2-13 Lift curve of an aerofoil with leading-edge flaps retracted and extended (Nicolai and Carichner, 2010).

However, the effect of leading edge flaps is significantly influenced by leading edge radius, specifically, the effect of leading-edge flaps on the aerodynamic characteristics of the wing increases with the leading edge radius reduces (Abbott and Von Doenhoff, 1959).

One of the most widely used leading-edge flaps is Krüger flap, which was invented by Werner Krüger in 1943. Krüger flap is hinged at the lower surface of aerofoil and rotates out in the front of the main section of the wing when deployed, increasing the wing camber and area. This type of leading-edge flap is widely used on Boeing aircrafts, for example, Boeing 707, Boeing 727, Boeing 747 and so on.

The advantage of this leading-edge separation control technique is that it could delay separation and significantly increase the lift-to-drag ratio when deflected downward at high angles of attack (A. R. Jones et al., 2008). The penalty is that it adds additional drag, weight, complexity and so on.

2.3.6 Vortex Generator

Passive vortex generator is an aerodynamic device that was first introduced by Taylor in 1940s, consisting of a row of small plates or aerofoils mounted normal to the surface at a lateral angle of incidence to the local flow to induce an array of streamwise vortices (Taylor, 1947). These longitudinal vortices transport the outer-region high-momentum fluid downwards to the near-wall area to mix with and to a certain extent replace the low-momentum flow within the boundary layer, thus energizing the boundary layer so that it could withstand the adverse pressure

gradient and keep attached over the solid surface. Due to its simplicity and effectiveness, it's quite widely used to enhance aerofoil performance (Pearcey 1961; Nickerson J 1986), control the flow separation in diffusers (Brown et al., 1968; Feir, 1965) and so on.

Numerous researches have been conducted on conventional passive vortex generators, and general guidelines are provided by Pearcey (1961) and Taylor et al. (1948), which suggest that conventional vortex generator is normally in triangular or rectangular planform with a height of the order of boundary layer over lifting surfaces. The conventional vortex generators are effective in delaying flow separation, however, they produce a significant parasitic drag that adversely affects the aeroplane cruise performances.

A solution to reduce the parasitic drag by conventional vortex generator is submerged vortex generator, which is also called micro VGs (Lin, 1999; Lin et al., 1994; Tai, 2002), sub boundary-layer VGs (Ashill et al., 2002, 2001; Holmes et al., 1987), low-profile vortex generators (Lin, 2002; McCormick, 1993) and microvanes (Hamstra et al., 2000). This type of vortex generator was originally developed by Kuethe (1972) who examined several wavy-type vortex generators with h/δ of 0.27 and 0.42 that successfully reduced the intensity of separation-induced acoustic disturbances in the wake through the suppression of Karman vortex street and reduction of velocity-deficit area in the wake region. Here, h is the height of vortex generator and δ is boundary layer thickness. Experimental study on control of boundary layer separation over a flat plate by Rao and Kariya (1988) suggest that VGs with $h/\delta \leq 0.625$ have the potential of exceeding the performance of vortex generators with a height of the order of boundary layer thickness. The subsequent studies (Kerho et al., 1993) demonstrate that submerged vortex generators are able to suppress the flow separation with a lower drag penalty.

Another way to optimize the passive vortex generator performance is vortex “reinforcers”. Basically, the vortex “reinforcers” are vortex generators located further along the trajectory of the upstream-generated vortices to augment the vortex strength in the downstream flow field (Kuethe, 1973; Lin et al., 1989; Rao and Kariya, 1988; Wheeler, 1984). The vortex reinforcers enhance the vortex-

induced mixing within a larger field without increasing the height of vortex generators.

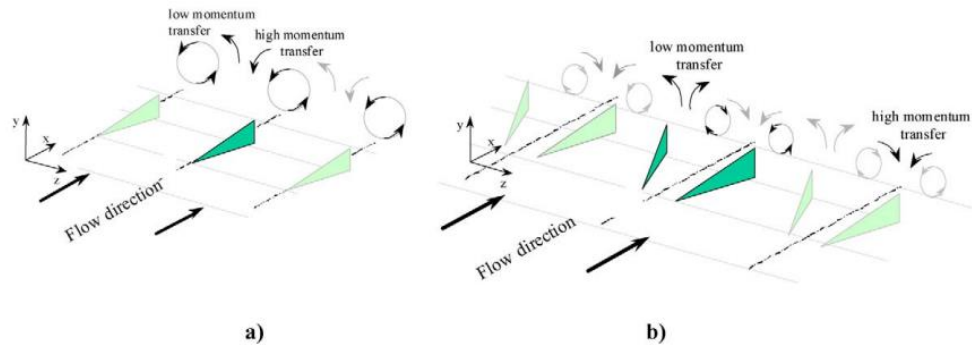


Figure 2-14 Configurations of vane-type vortex generators: a) corotating; b) counter-rotating (Godard and Stanislas, 2006).

Different spatial configurations of vortex generators have been utilized in flow control. Figure 2-14 shows two configurations of co-rotating and counter-rotating vortex generator arrays that produce co-rotating and counter-rotating streamwise vortex pairs respectively. For the co-rotating vortex generator arrays (shown in Figure 2-14 (a)), low momentum fluid is transported upward and high momentum fluid is transferred downward to the near-wall region between the two adjacent vortex generators, therefore, a proper spanwise spacing is required, otherwise this opposite motion results in mutual vorticity cancellation and reducing the vortex generators' effectiveness. The advantage of this vortex generator configuration compared with counter-rotating vortex generator array is that the co-rotating longitudinal vortices stay close to the wall even when they develop downstream (Godard and Stanislas, 2006). As can be seen in Figure 2-14 (b), the high momentum and low momentum fluid transfers are dissociated in counter-rotating vortex generator configuration, specifically, high momentum fluid is transported downward between the VG planes of symmetry while low momentum fluid transport occurs between different VGs. It was observed that the counter-rotating vortices might move away from the wall as they travel downstream in zero pressure gradient boundary layer (Betterton et al., 2000). There still remains a quite disputable issue that which of the co-rotating vortices and counter-rotating vortices could produce a better control effect on flow separation. Gad-el-Hak (2007) claims that co-rotating vortices are more efficient in suppression of flow separation

compared with counter-rotating vortices, however, Lin (2002) states that generally counter-rotating VGs perform better in 2D-type flow separation control while co-rotating VGs can exceed the performance of counter-rotating VGs in controlling 3D separation such as separation that occurs around highly-swept wings and compact-duct inlets.

2.4 Active Leading-edge Separation Control Techniques

2.4.1 Blowing

Steady blowing is an effective tool to delay flow separation. The idea was to allow pressurized air to pass through a slot and push the retarded fluid near the wall, thus accelerating it. Additional power is required for the pressurized air sources, which was a barrier against blowing flow control over an aircraft until jet engine was introduced (Radespiel et al., 2016). Jet engine could provide an efficient on-board source of pressurized air, and this stimulated intense studies on active flow control using blowing technique.

Figure 2-15 illustrates a tangential blowing at around the trailing edge of an aerodynamic geometry. Apart from the direct momentum injection, tangential blowing at trailing edge could also exploit the potential of Coanda effect (Radespiel et al., 2016), suppressing the flow separation. This circulation control concept through blowing was proposed by Cheeseman and Seed (1967). Since then, a number of researches have been done on blowing circulation control due to its promising potential in flow separation control (Kind, 1967; Novak et al., 1987; Wood and Nielsen, 1985). A drastic increase in lift coefficient was observed when tangential blowing was used at the trailing edge to influence the rear separation point of a circulation control aerofoil (McLachlan, 1989). Steady tangential blowing at leading edge and midchord was also utilised to control flow separation. Müller-Vahl et al. (2014) found that high-momentum blowing at leading edge could fully suppress the flow separation around a NACA 0018 aerofoil at deep stall angles of attack. However, blowing at midchord is only effective for trailing-edge separation and ineffective in controlling leading-edge separation.

Oscillatory blowing was used in controlling flow separation, which was found to be more effective than steady blowing (Seifert et al., 1996, 1993). It is capable of

achieving similar gains with one or two orders of magnitude smaller amount of momentum addition as steady blowing (Kral, 2000). This is due to that oscillatory blowing could exploit natural instabilities of the flow field to enhance shear layer mixing. The effectiveness of oscillatory blowing depends on many parameters, i.e. oscillation frequency, momentum coefficient of oscillatory blowing, blowing slot location, angle of attack of the aerofoil and so on (Seifert et al., 1996). Hites et al. (1997) conducted experimental measurements over a NACA 0015 aerofoil with oscillatory blowing and observed an increase of 80% in lift coefficient at high angles of attack. Numerical investigations of flow separation control through pulsed blowing were also undertaken, which showed great beneficial effects of oscillatory blowing on lift coefficients (Donovan et al., 1998; Deng et al., 2007). Closed-loop control of flow separation around a Glauert Glas II aerofoil has also been performed and achieved to improve the transient flow behaviours of open-loop response (Allan et al., 2000).



Figure 2-15 Illustration of tangential blowing (Radespiel et al., 2016).

2.4.2 Suction

Suction was first proposed by Prandtl (1904) over one hundred years ago, who introduced the concept of boundary layer and conducted several experiments on boundary layer control through suction. It was shown that flow separation is suppressed by removing the decelerated fluids in the near-wall region. Basically, the principle of this method is to eliminate the retarded fluids close to the wall and shift the high-momentum flow downwards to the surface (Greenblatt and Wygnanski, 2000), thus a boundary layer with fuller velocity profile is developed, avoiding a separation of the boundary layer.

Intense researches have been implemented to exploit the remarkable beneficial effects of steady suction on flow control. For example, Poppleton (1955) carried out wind tunnel experiments of boundary layer control over a swept wing by suction and the results show that distributed suction over the first 15% of aerofoil

chord could increase the lift coefficient by 40% at angle of attack of 15° . Karim and Acharya (1994) experimentally investigated dynamic stall control over a NACA 0012 aerofoil with leading-edge suction. It was demonstrated that suction from a slot over the first 2% to 5% of chord could effectively remove the reverse flow and suppress dynamic stall at various Reynolds numbers ($3.0 \times 10^4 < Re_c < 1.18 \times 10^5$). Atik et al. (2005) performed numerical simulations of separation control on a thin aerofoil with slot suction by extending Lagrangian approach. They concluded that even a weak suction at around the leading-edge could lead to substantial delays in flow separation.

Similar to blowing, periodic suction was also applied to flow separation control by exploiting the instabilities of the baseline flow. Sheikh and M. (2014) did numerical study of flow separation control over an aerofoil using both steady and unsteady suction at high angles of attack, and they found that unsteady suction is more effective in controlling flow separation than steady suction since separation control could be achieved by unsteady suction with a much less energy consumption. Zhang et al. (2017, 2018) studied the control effect of unsteady pulsed suction on the separation over a highly loaded compressor cascade. The results show that unsteady suction has a higher capability of controlling flow separation than steady suction with the same time-averaged suction flow rate, and a reduction of 16.8% in pressure loss is obtained due to the suppression of flow separation around the compressor cascade.

2.4.3 Sweeping jet

Sweeping jet actuators were introduced as an active flow separation control technique in recent years. This type of actuators could produce both temporally and spatially oscillating jets without any moving components required (Koklu and Pack Melton, 2013). The working principle of sweeping jet actuators is illustrated in Figure 2-16. When the pressurized air passes through the main channel of the actuator, it attaches on one of the walls forming the main channel due to Coanda effect, as shown in Figure 2-16 (a), it attaches on the right side wall. This jet is bent towards the other side at the exit of the main channel. The curving of the jet results in a pressure rise at the inlet of the feedback channel, leading to a reversed flow, which pushes the entering fluid to the left surface, see Figure 2-16 (b). In the end, the fluid attaches on the left surface (Figure 2-16 (c)), and the jet inclines towards

the right side. The backflow developed in the left feedback cavity will push the entering fluid to the right wall, in other words, the jet switches back to its initial state. So this is a self-sustaining process and the cycle will be repeated.

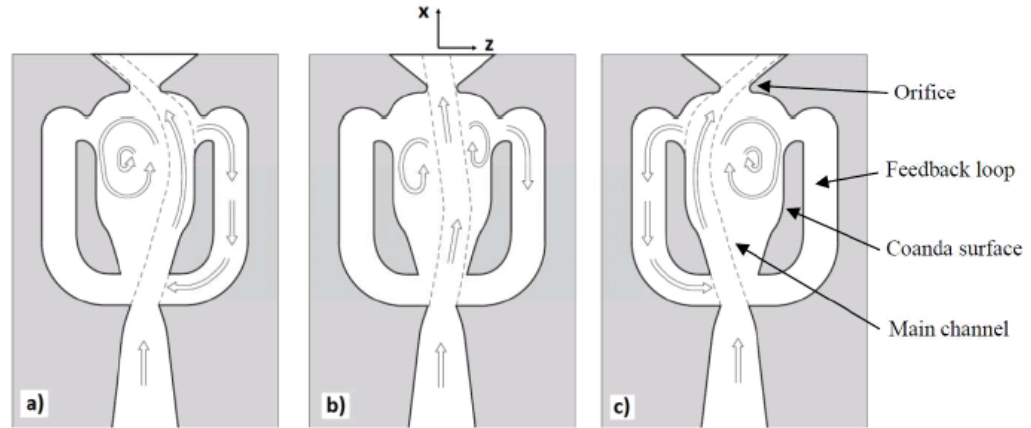


Figure 2-16 Schematic of a sweeping jet actuator (Koklu and Pack Melton, 2013).

Sweeping jet actuators were first developed at Harry Diamond Research Laboratories over fifty years ago, their early applications include windshield-washer fluid nozzles, shower heads, hot tubs, sprinklers and so on. Recently, sweeping jet actuators were introduced in flow separation control applications (Tewes et al., 2010; Wosizdlo et al., 2010). Since then this technique becomes a popular research subject of flow separation control field due to its unique features such as no-moving parts, maintenance free, high frequency and high amplitude bandwidths (Koklu and Owens, 2014). For the moment, there is still a lot remained to be studied regarding sweeping jet actuators.

Sweeping jet actuators are now widely used in various flow separation control applications. Cerretelli et al. (2010) performed a wind tunnel study of flow separation control over a DU 96 aerofoil with sweeping jet actuators, which are embedded in the aerofoil. Wind tunnel tests were conducted at full-scale Reynold numbers, $2 \times 10^6 \leq Re \leq 4.8 \times 10^6$. It was found that the effectiveness of sweeping jet actuators depends on Re , actuation amplitude and state of aerofoil surface. Moreover, a substantial stall margin extension and an increase of minimum 10% to over 60% in lift were reported with the application of sweeping jets. Koklu and Owens (2017) were able to show that sweeping jet are more efficient in flow separation control around an adverse-pressure-gradient ramp than steady blowing

with straight and angled jets. Sweeping jet actuators could also be used to influence the flow separation around a rudder and improve its control authority, for example, up to 50~70% increase in side force was observed by Seele et al. (2012).

2.4.4 Synthetic jet

Synthetic jet is also called zero-net-mass-flux jet (ZNMF) since it could be produced without any external net mass injection. The synthetic jet actuator normally consists of a membrane or diaphragm and an orifice that are located in a cavity, as shown in Figure 2-17. The membrane or diaphragm are always driven by electrical, mechanical, acoustic, magnetic or other means so that they could oscillate while in action (Mallinson et al., 1999). During the blowing stroke, the upward movement of membrane or diaphragm will expel the fluid through the orifice. The ejected fluid separates at the sharp edge of the orifice and forms a vortex pair which propagates away from the orifice. When the suction stroke starts, the membrane or diaphragm moves downwards, drawing the ambient air into the cavity. Normally, the vortex pair is sufficiently distant from the orifice so that it will not be influenced and entrained back into the cavity during the suction stroke. Instead, the vortex pair will coalesce to synthesize a jet in the far field. The net mass flux is zero over the entire cycle although there is a non-zero-velocity jet.

The uniqueness of synthetic jet is that no external air supply and complex piping are required. This allows a smaller size and a minor weight penalty compared with blowing or suction flow control techniques. So synthetic jet actuator becomes a promising and popular flow control device. For example, a number of researches have been conducted on flow separation control with synthetic jet actuator. Amitay and Glezer (2006, 2002) performed experimental investigations of separation suppression using synthetic jet. The results show that stall is dramatically delayed with the use of synthetic jet, as a consequence, an increase in lift and a decrease in pressure drag is resulted in. The authors also found that actuation at around the shedding frequency leads to a Coanda-like attachment of the separated shear layer, formation of large-scale vortex structures and time-periodic variation in circulation while the circulation is almost invariant at an actuation frequency that is an order higher of the baseline vortex shedding frequency. Stalnov et al. (2010) studied the effectiveness of synthetic jet on separation control over an aerofoil and compared with the results obtained using mechanical vortex generators. According to the

experimental investigations, a conclusion was made that synthetic jet could achieve separation control without inherent drag penalty that mechanical VGs impose. Moreover, synthetic jet works well on suppression of flow separation over a wide range of Reynolds numbers while VGs are only effective at the designed Reynolds number.

The mechanisms responsible for the realization of separation control were summarized by Tang et al. (2014). First, momentum is added to the boundary layer by ZNMF jet. During the suction stroke, the low-momentum fluids are entrained into the orifice, resulting in a thinner boundary layer. On the other hand, the expelled fluids could transfer momentum into the ambient flow over the blowing stroke. Second, the vortices generated by synthetic jet actuator enhances momentum exchanges between the inner boundary layer and the outer high-momentum flow, enhancing the boundary layer to overcome the adverse pressure gradient. Third, actuation at certain range of frequencies could interact with the instabilities of the separated shear layer or separation bubbles.

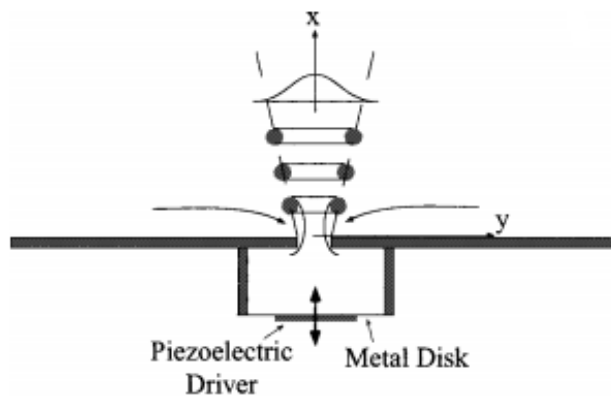


Figure 2-17 Schematic of a ZNMF jet actuator (Smith and Glezer, 1998).

2.4.5 Acoustic excitations

Over the decades before 1970s, flow separation control approaches usually relied on direct addition of momentum to the boundary layer or suction of low-momentum fluids. In the mid of 1970s, Collins and Zelenevitz (1975) demonstrated that flow separation around an aerofoil could be suppressed by sound waves without any direct momentum addition. This attracted large amount of attentions and extensive studies were followed (Ahuja and Burrin, 1984; Nishioka et al., 1990; KBMQ

Zaman et al., 1987; Zaman, 1992; Zhong et al., 1996). It was suggested that the excitation of flow instability is the key to understand flow separation control with acoustic excitations (Nishioka et al., 1990). This provided an innovative way to control flow separation without any direct momentum injection.

Acoustic excitations are not only very effective in controlling the flow separation around aerofoils but also flat plates, circular cylinder and so on. Moreover, a significant increase in lift can be produced with acoustic excitations, normally on the order of 50% (Greenblatt and Wygnanski, 2000). However, the drawbacks of this technique are also obvious. First, the acoustic excitations could excite the wind tunnel's resonance modes as well, which in return excites the separated shear layer. Therefore, the wind tunnel experimental results may not be applicable to the real aircraft wings. Second, a high-amplitude excitation up to 156 db is required to obtain significant separation control effects. This high-level sound could cause permanent damages to human ears even during a short time exposure. So this puts another barrier against this technique's practical applications.

2.4.6 Moving Surfaces

Moving surface boundary layer control is an effective means to prevent or delay separation, which has been investigated intensively. For leading-edge separation control, it is usually consisted of at least one rotating cylinder placed at the leading edge of the aerofoil (see Figure 2-18). It contributes to the delay of the flow separation in several ways (Modi, 1997): 1) by way of minimising the relative motion between the surface and the freestream, the moving cylinder retards the initial growth of boundary layer; 2) the moving cylinder re-energizes the boundary layer due to the injection of momentum.

Enormous researches on separation control over aerofoil by moving surfaces have been conducted over the past century. Modi and his colleagues (Modi et al., 1990, 1991, 1981; Modi and Mokhtarian, 1990) conducted systematic investigations of flow separation control around aerofoil with this technique, exploiting one or more moving surfaces at the leading edge, trailing edge or even the upper surface of aerofoil to delay stall and the wind tunnel experimental results are quite promising. Apart from this, flight tests of YOY-10A STOL-type aircraft with rotating cylinders were also conducted by NASA (Cichy et al., 1972; Cook et al., 1974; Weiberg et

al., 1973). With cylinder rotating at high speed and flap in lowered position, a lift coefficient of 4.3 was recorded while the aircraft flew at around 30m/s.

The moving wall could reduce the drag significantly. Moreover, the power consumption of this means is negligible. However, the cylinder needs to be actuated by a motor-driven system, which adds complexity and weight to the airplane. The aerofoil profile needs to be specifically modified to apply the rotating surfaces.

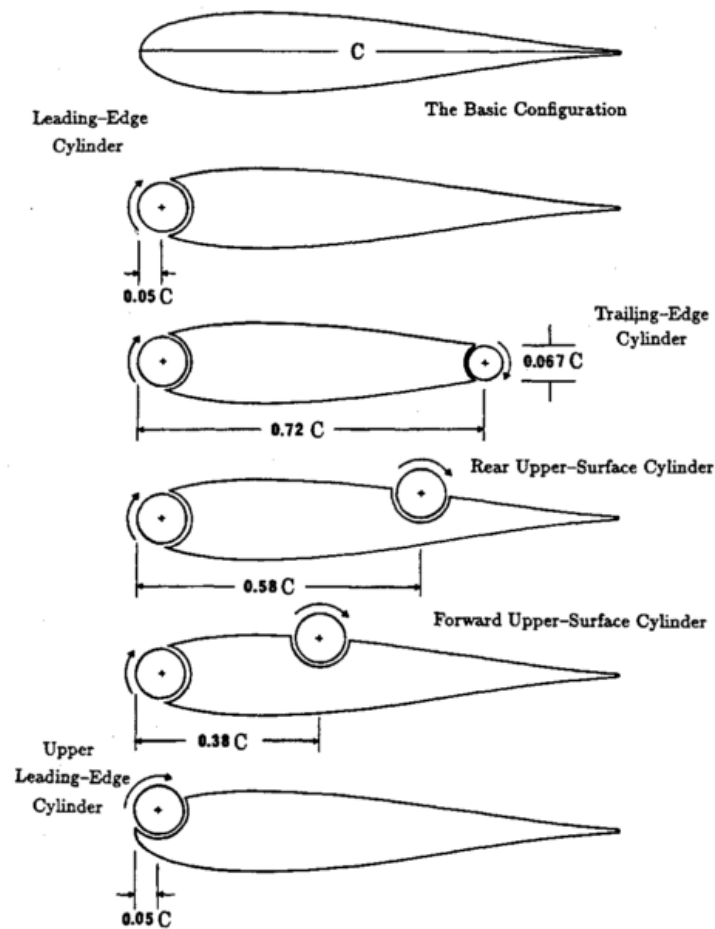


Figure 2-18 Leading-edge cylinder (Modi, 1997).

2.5 Flow Separation Control With DBD Plasma Actuators

2.5.1 DBD Plasma Actuators

DBD plasma actuators have received a number of attentions because of its uniqueness in flow control. The body force generated by them could be used to

manipulate a flow to obtain a desired flow state. Compared with other active flow separation control techniques, they are electric flow control devices with a high responding speed which could easily reach tens of thousands Hz, enabling a real-time flow control. They could be applied to surfaces without the need of moving parts or cavities. The DBD plasma actuators are low in mass, which is normally on the order of tens of grams. The energy consumption is typically very low. No pressurized air source or pipelining are needed, and this makes DBD plasma actuators much easier to be put into practice. Moreover, the DBD plasma actuators could be operated when required to achieve active flow control with no profile drag penalty because they are surface-mounted devices (Whalley, 2011). Currently the main limitation is that the maximum velocity induced by DBD plasma actuator is $8 \sim 10$ m/s, which is not strong enough to affect the flow around a commercial airliner.

The DBD plasma actuators are usually made of two electrodes separated by a dielectric sheet, with the upper electrode exposed to the air and the lower electrode covered by the dielectric material (see Figure 2-19). When the voltage supplied to the electrodes is high enough, the air over the lower electrode will be ionized (Corke et al., 2010). The ionized air is known as plasma, which appears a little purple. This plasma could only be seen by the naked eye in a dark space because the intensity is quite low. A body force is caused by ionized air thus active flow control could be achieved.

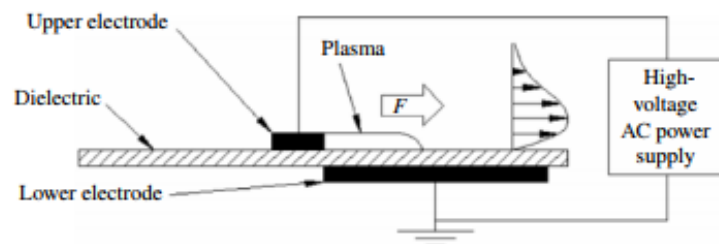


Figure 2-19 Schematic of a typical DBD plasma actuator (Jukes and Choi, 2013).

A momentum is always coupled with the body force. Demonstration of the body force and momentum exerted by plasma actuators on the surrounding air has been achieved. Figure 2-20 shows the plasma forcing of an asymmetric DBD plasma actuators on still air (Roth et al., 1998). This photograph was taken by J. R. Roth,

D. M. Sherman, and S. P. Wilkinson, who were the first to develop a DBD plasma actuator capable of working at atmospheric pressure. As can be seen, the jet flows horizontally if the actuators are switched off. When the actuators are switched on, the jet will be deflected down. This proves the existence of the forcing generated by plasma actuators.

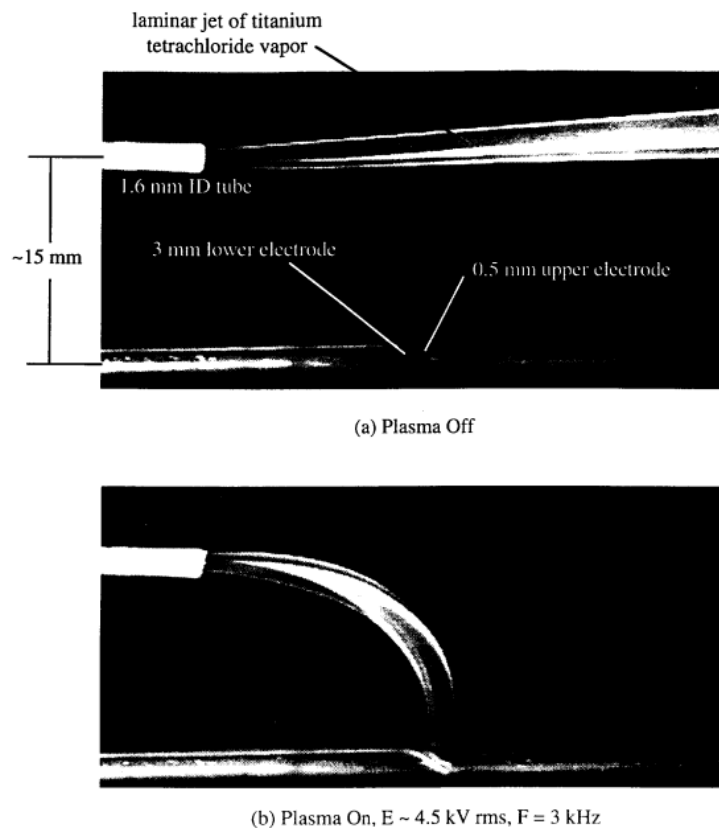


Figure 2-20 Demonstration of plasma forcing by an asymmetric plasma actuators on quiescent air: (a) plasma actuators off; (b) plasma actuators on (Roth et al., 1998).

As for plasma, there is still a lot unclear to us and our understanding of plasma mainly comes from Faraday and Maxwell who conducted undergrounding researches on plasma generated inside an electric discharge tube over one hundred years ago. In 1995, Roth implemented his experiments on the plasma that maintained by a high voltage between parallel plates. He studied the formation of plasma between electrodes and the Voltage-Current characteristics of plasma generated by this configuration (Roth, 1995). The current increases nonlinearly when the voltage initially increases (A-B), this is due to the movement of the

naturally occurring electrons and ions. During the stage B-C, the current stands still since the electrons and ions have already been removed from the volume. If the voltage increases further, the energy will enable the electrons to be emitted from the cathode and these electrons then ionize the surrounding air via collisions, causing the current to increase exponentially with voltage (C-E). At the breakdown voltage, transition occurs and plasma is transited from corona regime to glow regime (E-F), in which the plasma is visible to the eyes. After the transition, the plasma is in normal regime where the current is almost independent of the driving voltage. When plasma occupies the entire cathode surface, the discharge enters abnormal regime (G-H). In this discharge regime, the current increases with the voltages. At the point of H, the cathode is heated to incandescence because of the high current, triggering a transition of the discharge from glow to arc (H-I). From I to J, the voltage decreases with current. The temperature of electrons, ions and gas is not equal. In this stage, the discharge is called non-thermal arcs. At point J, the discharge begins to go into thermal arc regime, where the plasma is in a thermodynamic equilibrium.

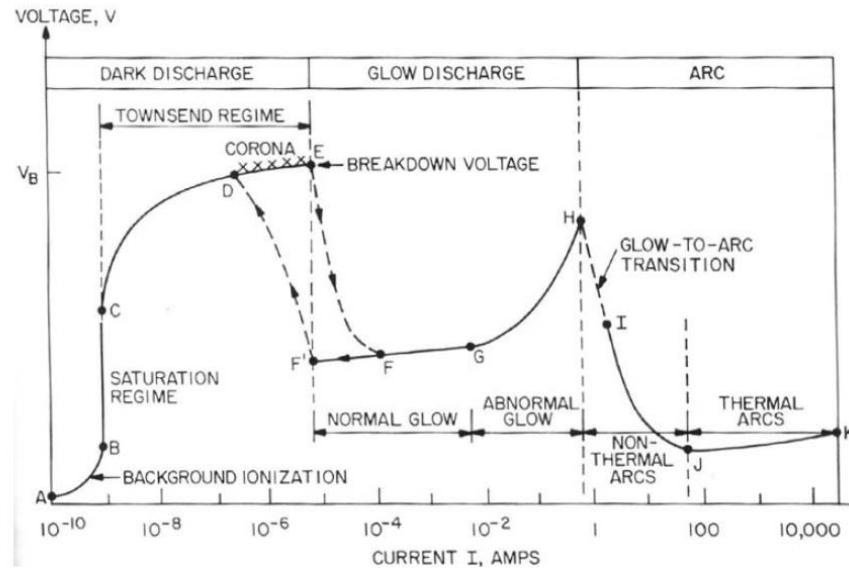
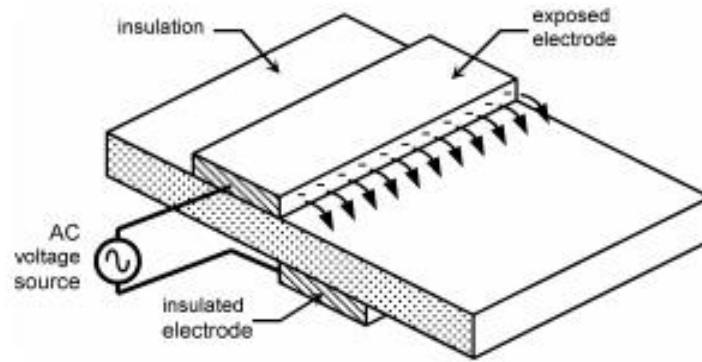
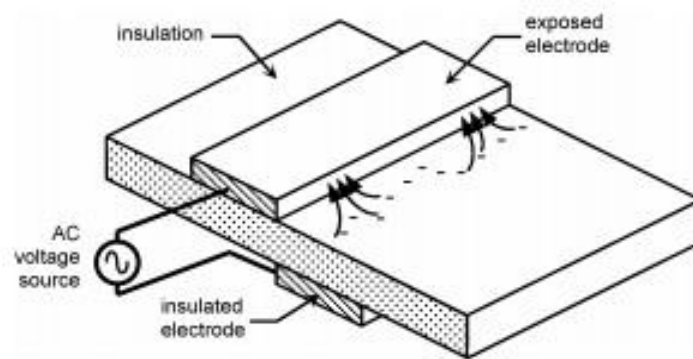


Figure 2-21 Curve of a DC low pressure electrical Discharge tube's characteristic (Roth, 1995).



(a)



(b)

Figure 2-22 Illustration of electrons' movement during ionization for a DBD plasma actuator (Enloe et al., 2004b).

Several researches have been done to investigate the DBD plasma actuator discharge. According to (Orlov and Corke, 2005) and (Orlov et al., 2006), the DBD plasma actuator discharge is thought to be neutral containing equal number of positive ions and negative electrons and ions. Moreover, the operation and self-limiting nature of DBD plasma actuators were also studied (Enloe et al., 2004b). The illustration of this is shown in Figure 2-22. In this case, a high AC voltage is applied to the electrodes. During the negative part, when the voltage is above the breakdown voltage of the ambient air, electrons will be emitted from the exposed electrode and travel to the dielectric surface. The build-up of the electrons on the dielectric surface will cause charge that opposes the applied voltage, quenching the plasma discharge and preventing the plasma from collapsing into an arc. Therefore, the plasma discharge will terminate unless the magnitude of the voltage continually

increases. As for the positive side of the AC voltage, the electrons deposited on the dielectric surface would move back to the exposed electrode and the charge is limited to that deposited on the dielectric surface during the previous half-cycle.

The nature of the discharge between the electrodes was studied by Gibalov and Pietsch (2000). In their experiment, a positive voltage pulse, a negative voltage pulse and an AC voltage were applied to a DBD plasma actuator respectively. They found that the discharge which looks homogeneous to our eyes is actually highly filamentary, shown in Figure 2-23. These filaments may contain microdischarges. They also note that increasing voltage results in larger discharge area on the surface, and the length of the discharge patterns decreases with increasing the relative permittivity while the number of discharge carriers increases with increasing the relative permittivity.

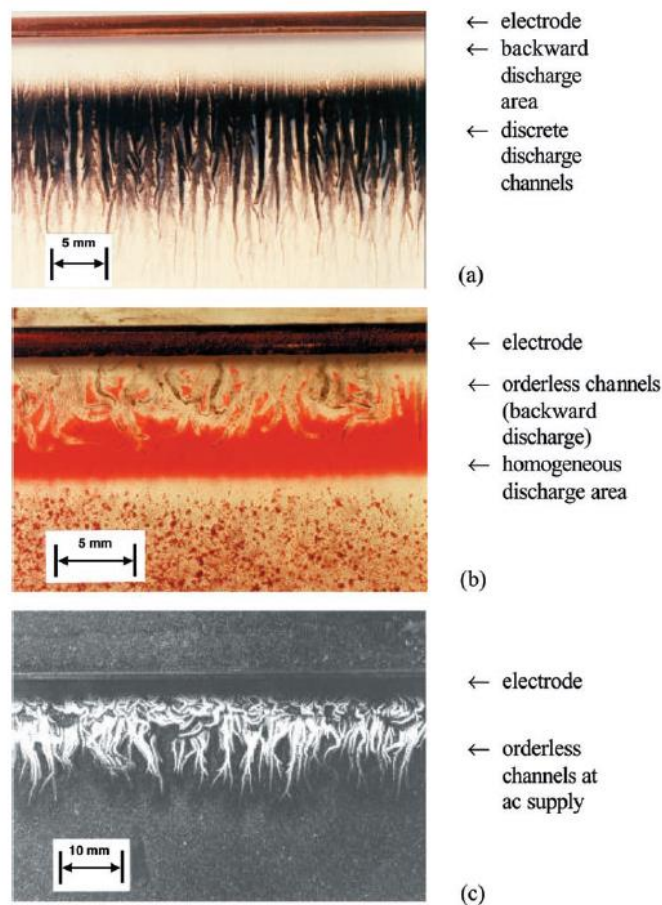


Figure 2-23 Discharge patterns on the dielectric surface: (a) voltage pulse of 20kV; (b) voltage pulse of -20 kV; (c) AC voltage of 10kV (Gibalov and Pietsch, 2000).

The filamentary nature of the surface discharge was also observed by Wilkinson (2003) and Enloe et al.(2004a, 2004b). As can be seen in Figure 2-24, the plasma is non-uniform along the electrode edge. Specifically, the plasma structure appears as discrete “fountains” in the left image, where the upper electrode is positive instantaneously. When the upper electrode turns negative, the plasma structure is in closed loops and there are bridges across the peaks of the plasma structure. Obviously, the plasma distribution is both spatially and temporally non-uniform. Enloe et al. (2004a, 2004b) also observed a series of irregular microdischarges by optical measurements. Moreover, they demonstrated that more irregular microdischarges are produced over the positive half-cycle compared with those generated during the negative-going portion.

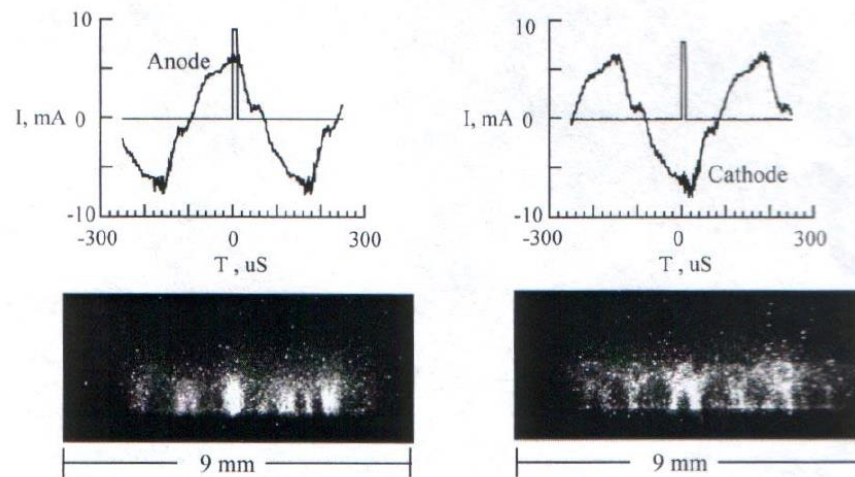


Figure 2-24 CCD phase averaged photographs of plasma on the edge of an electrode with a voltage of 8.5kVpp at 3kHz (Wilkinson, 2003).

The role of each portion of AC cycle on the body force generated by DBD plasma actuators is a very hot research topic. People found that the charged particles move in opposite directions for the positive and negative parts and there is a plasma forcing acts away from the upper electrode towards the lower one. However, the contribution of each half AC cycle to the body force was quite disputed in the beginning. Several numerical studies on DBD plasma actuators show that the positive half cycle plays a much larger role in plasma forcing than the negative half cycle (Boeuf et al., 2007b; Boeuf and Pitchford, 2005; Jayaraman et al., 2008). Font and Morgan (2005) simulated the discharges of a DBD plasma actuator and found

that the forces generated in the negative and positive half cycles are opposite although the net force is away from the exposed electrode. According to their results, the plasma forcing during the positive portion of the AC cycle acts downstream and is ten times larger than that during the negative half cycle. However, several experimental studies demonstrate that the negative half cycle plays a predominant role in the plasma forcing (Enloe et al., 2008; Kim et al., 2007). Enloe et al. (2008) demonstrated that 97% of momentum coupling occurs during negative portion by measuring the oscillatory motion of a mechanical system driven by a DBD plasma actuator with a Michelson interferometer. Kim et al. (2007) measured the velocity field in the vicinity of a DBD plasma actuator on a cylinder using phase-locked PIV, which demonstrates that the velocity near the actuator increases a lot during the first negative portion while the velocity growing diminishes but not eliminates in the positive half-cycle. Forte et al. (2007) synchronized the horizontal velocity, vertical velocity, current and voltage of a surface DBD plasma actuator, as shown in Figure 2-25. A higher horizontal velocity (3.6 m/s) is induced in the negative-going portion than that over the positive portion (2.4 m/s). The horizontal velocity is always positive over the entire cycle, confirming that plasma force acts from the exposed electrode towards the grounded electrode during both the positive and negative portions. It is believed that the omission of the negative ions in the inert gas and nitrogen modelling accounts for the difference between the numerical studies and the experimental results (Enloe et al., 2008; Kim et al., 2007). This was confirmed by Boeuf et al. (2007a) and Likhanskii et al. (2008), moreover, they also found that the plasma forcing is away from the exposed electrode and towards the encapsulated electrode during both strokes.

Up to now, a lot of attentions have been paid to the maximum velocity induced by DBD plasma actuators. Enloe et al. (2004a) used PIV to study the velocity induced by DBD plasma actuators. They found that the maximum induced velocity scaled with $E^{3.5}$, where E is the driving voltage, as shown in Figure 2-26(a), moreover, two DBD plasma actuators could generate double the maximum velocity although this might not be true for more actuators. M. Forte, J. Jolibois and E. Moreau (2006) also investigated the maximum velocity utilizing LDA. The results of their study are illustrated in Figure 2-26(b). The maximum velocity increases asymptotically

with the applied voltage, and reaches a limit of 7m/s with a voltage of 26kV. Jukes et al. (2006) did further investigations into this issue. In their study, the velocities induced by asymmetric and symmetric actuator are measured via hot-wire anemometry. Figure 2-26(c) shows that the velocity scaled with $E^{1.97}$ for the symmetric actuator while the velocity induced by asymmetric actuator scaled with $E^{3.63}$. The velocity generated by the symmetric actuator is higher than that by asymmetric actuator although it increases slowly. The difference between these three studies may lie in the actuator geometry, materials, measurement techniques or the power supplies (Whalley, 2011).

Starting vortex induced by DBD plasma actuators was first observed by Post (2004). Figure 2-27 shows the starting vortex observed by Whalley and Choi (2013) from smoke-flow visualization and PIV measurement experiment. As can be seen in the picture, the smoke circles around the centre of the vortex core. The mechanism of the formation of starting vortex was studied as well, which is illustrated in Figure 2-28. When the flow ejects laterally, the air over the actuators is entrained to replenish it, inducing the generation of vorticity and circulation of a starting vortex. Meanwhile, secondary vorticity is created around the starting vortex due to the no-slip boundary condition. These secondary vorticity is observed to modify the trajectory of the starting vortex.

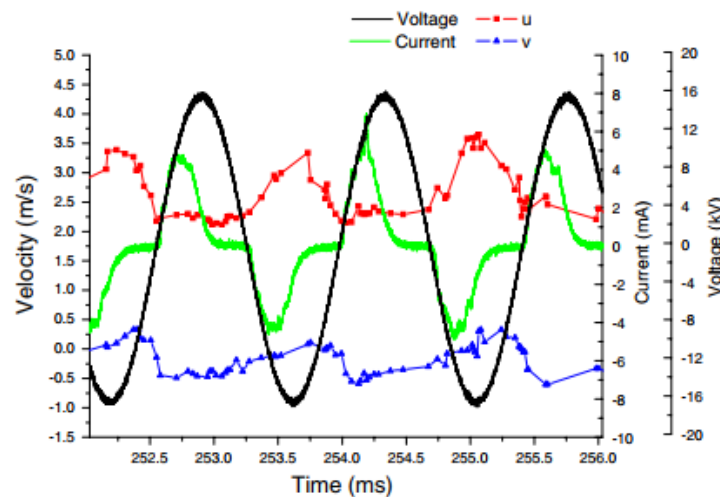
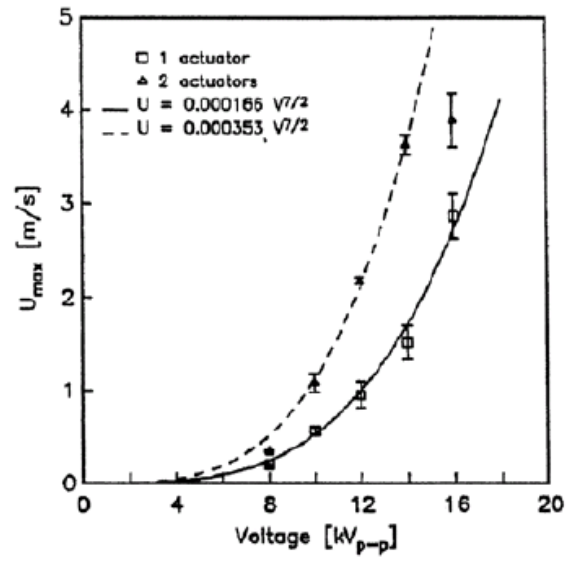
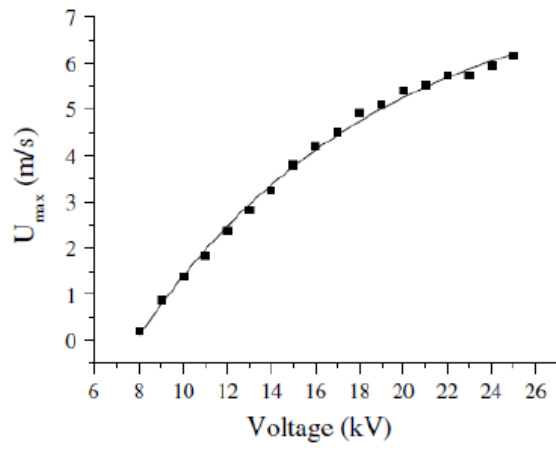


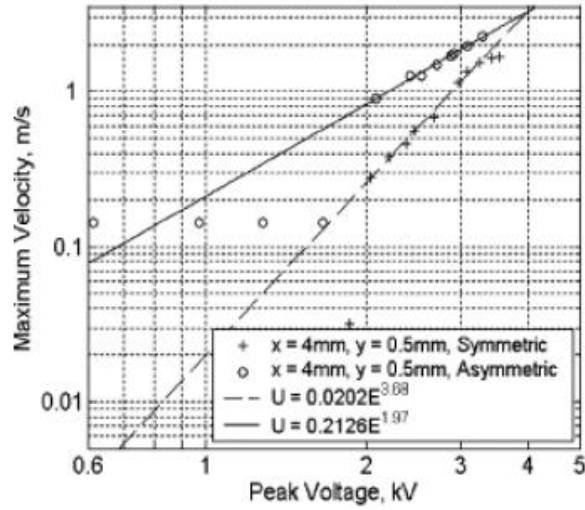
Figure 2-25 Horizontal and vertical velocity signals of the induced flow by a DBD plasma actuator in synchronization with voltage and current (Forte et al., 2007).



(a)



(b)



(c)

Figure 2-26 The relationship between the induced velocity and driving voltage, taken from (a) C L Enloe et al. (2004b); (b) M. Forte et al. (2006); (c) Jukes et al. (2006).

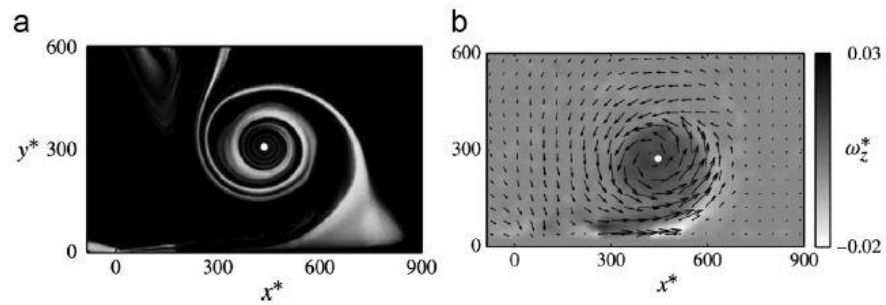


Figure 2-27 (a) Smoke-flow visualization and (b) PIV vorticity field of the starting vortex induced by DBD plasma actuators (Whalley and Choi, 2013)

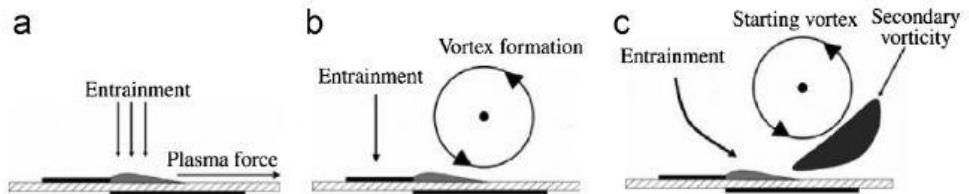
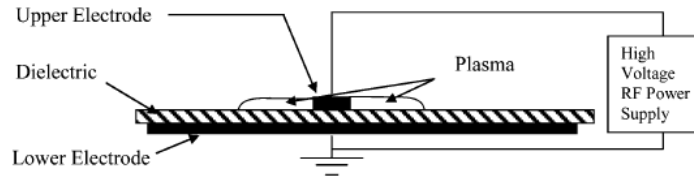
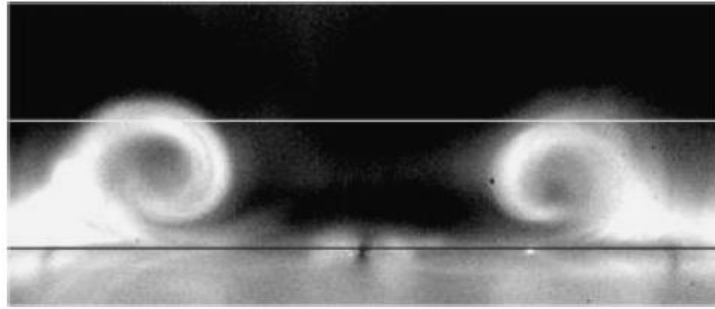


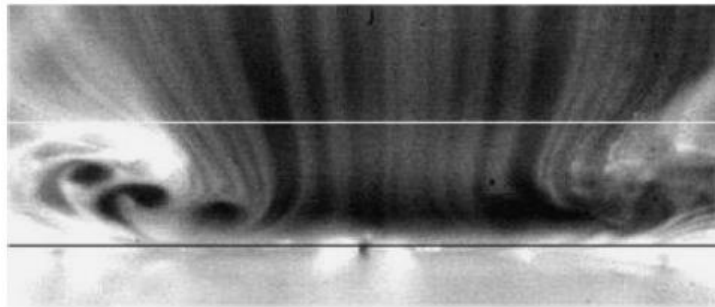
Figure 2-28 Mechanism of the formation of the starting vortex (Whalley and Choi, 2013).



(a)



(b)



(c)

Figure 2-29 Flow visualization of airflow induced by unsteady actuation of DBD plasma actuator in initially quiescent air. (a) geometry and cross section of electrode sheet; (b) $t = 0.04s$; (c) $t = 0.95s$ (Jukes et al., 2006).

Characteristics of the flow induced by unsteady actuation of DBD plasma actuators were measured by Jukes et al. (2006, 2008). Figure 2-29 (a) shows a symmetric DBD plasma actuator with an upper electrode situated on the centre of the lower electrode so that flow would be induced on both sides of the exposed actuator. In their studies, DBD plasma actuators were operated in pulsed mode at a frequency of 50 Hz. As can be seen in Figure 2-29 (b), a pair of starting vortices are generated on both sides upon the initiation of plasma actuator. After the initial stage, smaller vortices are generated by each pulsed actuation of plasma and they amalgamate into

a large starting vortex, shown in Figure 2-29 (c). Jukes et al. (2008) found that a laminar wall jet is developed with self-similar characteristics between pulses 0.3s after the initiation of plasma actuator. Thermal investigations of the flow around the symmetric plasma actuator show that the temperature is heated up 2° at 1mm away from the plasma actuator sheet surface with a driving voltage of 4.1 kV at 50 Hz. As the amount of heat produced by the actuators is pretty low, it is believed that the induced flow is not driven by heat (Jukes et al., 2006).

The DBD plasma actuators could be of different geometries. Figure 2-30 shows a doughnut-shaped type actuator used in the study by Segawa et al. (2008). This type of plasma actuator is also called plasma synthetic jet actuator (Jacob, Arvind Santhanakrishnan, 2006). It consists of two annular electrodes, separated by a dielectric layer, as shown in the figure. When the actuator is switched on, plasma appears (Figure 2-31 (a)) and the ambient air is driven to form a tangential jet which moves inwards the centre of the top electrode in Figure 2-31 (b). The tangential jets from various points interact and move up when they arrive at around the centre, thus a wall-normal jet is produced in Figure 2-31 (c). An annular starting vortex ring is formed after the initiation of the doughnut-shaped plasma actuator, which is similar to the observation of mushroom type vortex by Jacob, Arvind and Santhanakrishnan (2006).

Figure 2-32 (a) shows a DBD plasma actuator which has two exposed electrodes (electrode (1) and (3)) and an embedded electrode (electrode (2)) (Moreau et al., 2008). When a high AC voltage is applied on electrode (1) and no voltage on electrode (3), the discharge occurs between electrode (1) and (2), as shown in Figure 2-32 (b). The area of discharge is greatly increased with an additional DC high voltage applied to electrode (3), which is illustrated in Figure 2-32 (c). This is termed as sliding discharge by the authors.

Roy and his co-workers investigated the three-dimensional effect of curved plasma actuators (Wang et al., 2011). They developed several new types of plasma actuators: triangular, serpentine, and square plasma actuators, as shown in Figure 2-33. Vectors of forces induced by these three-dimensional plasma actuators and the traditional linear plasma actuator have been plotted in Figure 2-34. Plasma forcing usually act from the exposed electrode to the grounded electrode, and the

magnitudes of plasma forces for the four different type actuators are different. Figure 2-34 (b) demonstrates that the force produced by linear plasma actuator is predominately parallel to the surface while the forces by triangular, serpentine and square plasma actuators are purely three-dimensional.

Figure 2-35 illustrates the instantaneous vortical structures 16 ms after the plasma actuators are switched on. Linear plasma actuator produces the least vorticity and the vorticity is dominated by y-direction vortices. As for the three-dimensional plasma actuators, vorticity could be created in all the three directions downstream of the actuators. In addition, much more vorticity is generated by these three-dimensional plasma actuators especially by the serpentine and square actuators (Wang et al., 2011).

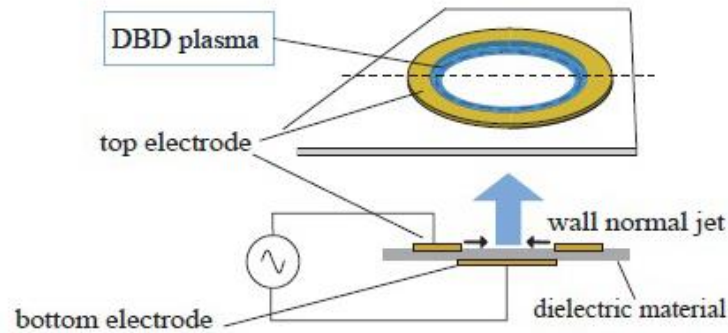


Figure 2-30 Schematic of a doughnut-shaped DBD plasma actuator (Segawa et al., 2008).

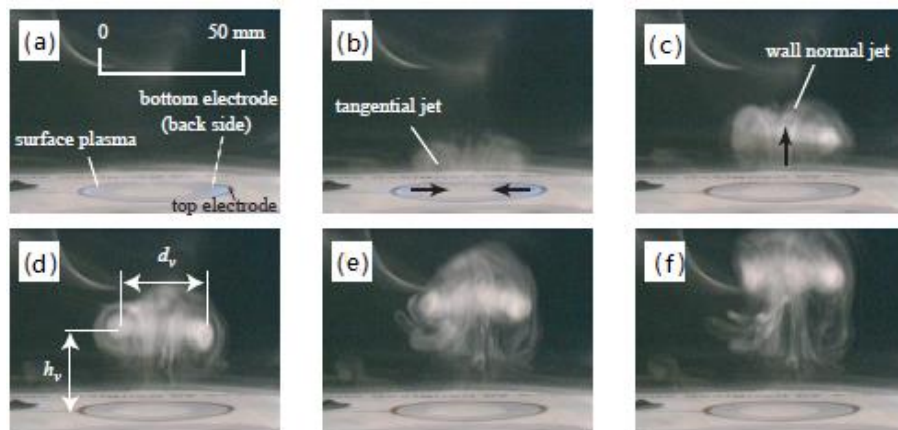
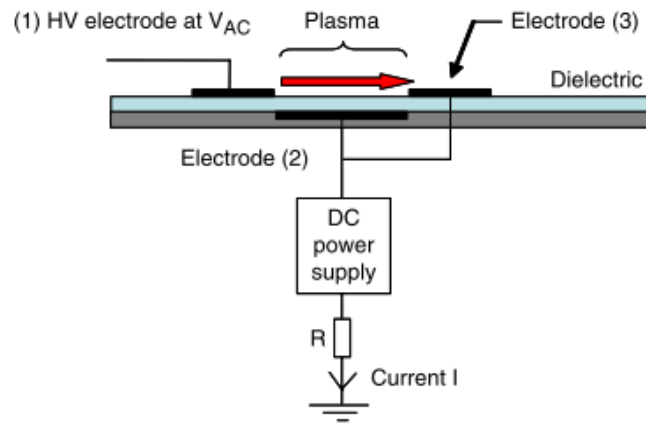
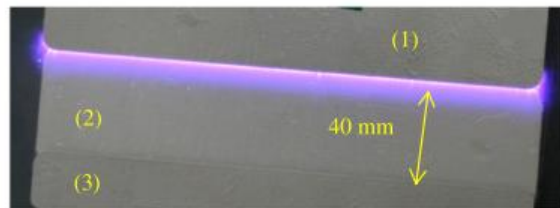


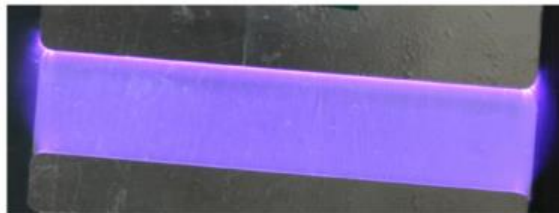
Figure 2-31 Flow visualization of the wall-normal jet induced by a doughnut-shaped DBD plasma actuator (Segawa et al., 2008).



(a)



(b)



(c)

Figure 2-32 Three-electrode based discharge:(a) schematic of three-electrode DBD plasma actuator; (b) surface discharge produced by electrodes (1) and (2); (c) surface discharge by all the three electrodes (Moreau et al., 2008).

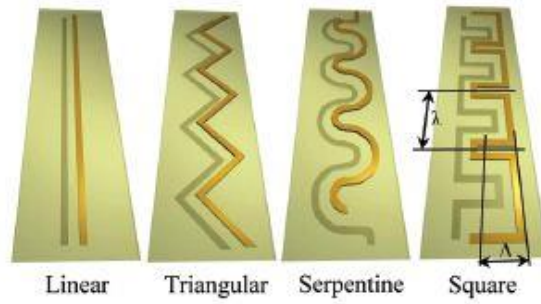
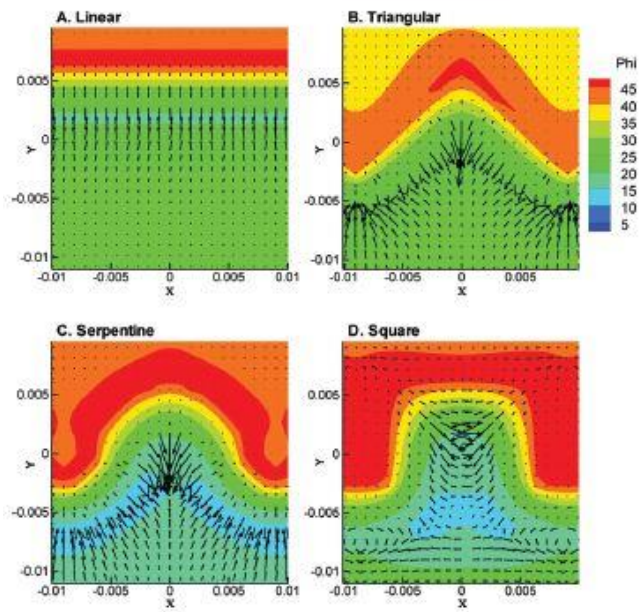
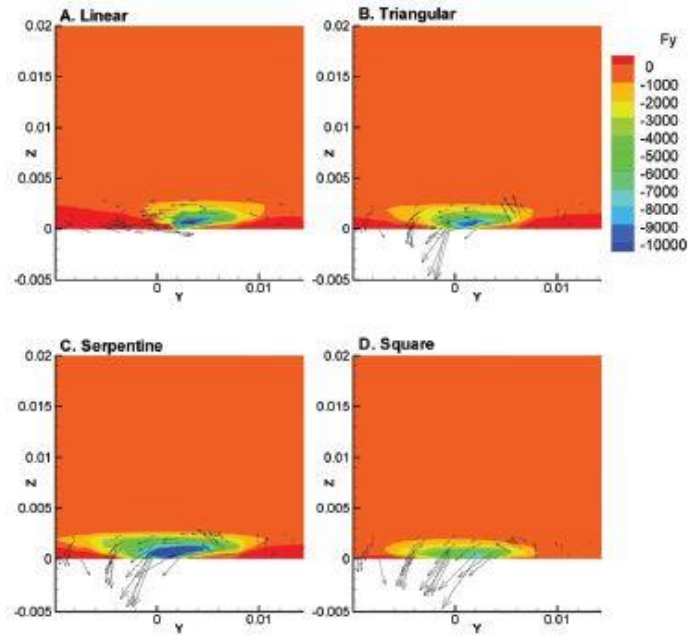


Figure 2-33 Schematics of several different types of three-dimensional plasma actuators (Wang et al., 2011).



(a)



(b)

Figure 2-34 Force vectors induced by linear, triangular, serpentine and square plasma actuators: (a) overlaid on potential contour; (b) overlaid on F_y map in yz plane (Wang et al., 2011).

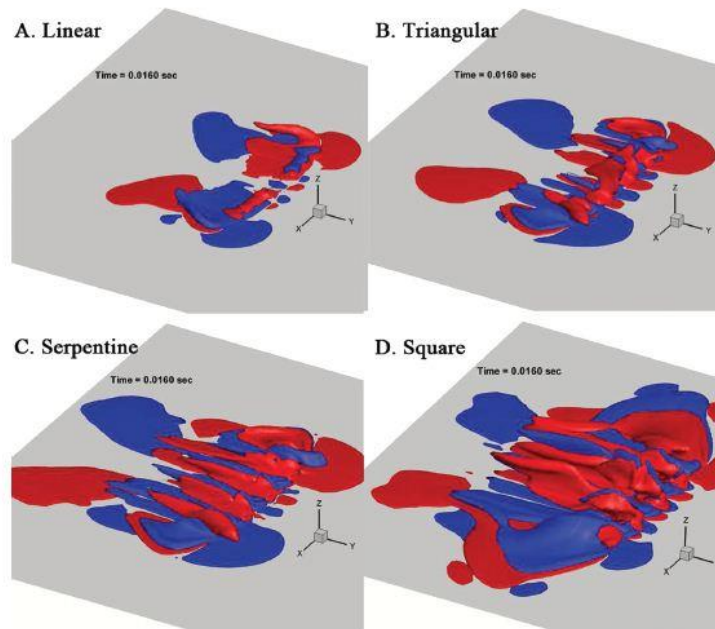


Figure 2-35 Vortices generated by different type plasma actuators at $t = 16\text{ms}$. Red color represents positive vortex while blue denotes the negative (Wang et al., 2011).

For practical applications with metallic surfaces or surfaces exposed to rain, moisture and conductive particles, it's not safe to use high voltages on the exposed electrode. One of the solutions is string-type DBD plasma actuators developed by Segawa et al. (2012, 2013). As illustrated in Figure 2-36, this type of DBD plasma actuator consists of a metallic wire coated with insulating materials, rather than a grounded sheet electrode. When the string-type DBD plasma actuators are applied to metallic surfaces, additional exposed electrode is unnecessary because the metallic surface itself can be chosen as exposed electrode and connected to ground. So they could be easily mounted to complex surfaces. Additional conductive exposed electrode is only needed when string-type DBD plasma actuators are applied to surfaces made from insulating materials. Compared with conventional sheet-type DBD plasma actuator, another advantage of string-type DBD plasma actuator is that it can be easily replaced when it gets broken.

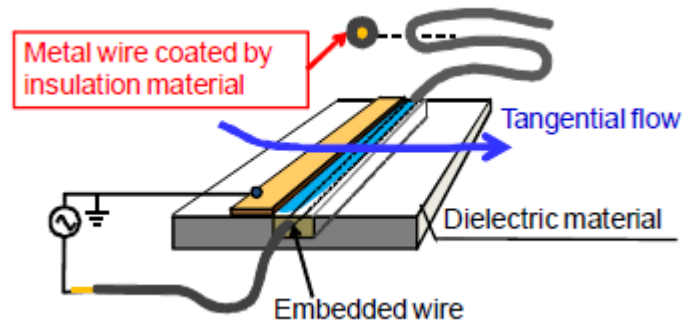


Figure 2-36 Schematic of a string-type plasma actuator (Segawa and Matsunuma, 2014).

2.5.2 Flow Control with DBD Plasma Actuators

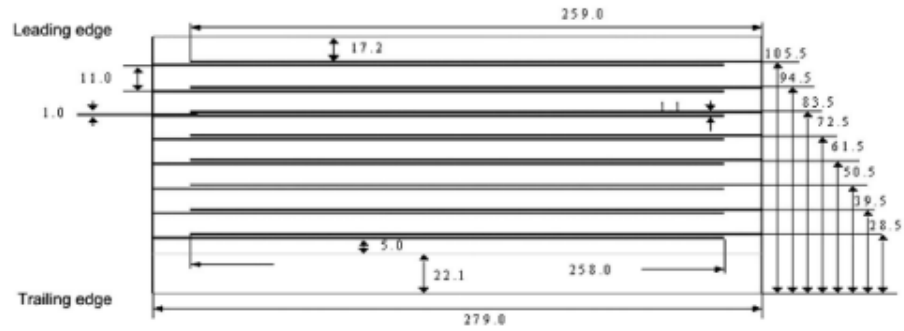
DBD plasma actuator has been widely used as an effective active flow control method, ranging from separation control over aerofoils (Corke and Post, 2005; He et al., 2009; Patel et al., 2008; M. L. Post and Corke, 2004) and cylinders (Jukes and Choi, 2009a; McLaughlin et al., 2004; Munska and McLaughlin, 2003), skin friction reduction (Jukes et al., 2006; Whalley and Choi, 2014), noise mitigation (Huang and Zhang, 2010; Li et al., 2010; Thomas et al., 2008) and tip clearance flow control (Matsunuma and Segawa, 2014; Saddoughi et al., 2015; Van Ness et al., 2006). It has also been demonstrated that DBD plasma actuator can work as

virtual Gurney flap to improve the aerodynamic performances of aerofoils (Feng et al., 2015).

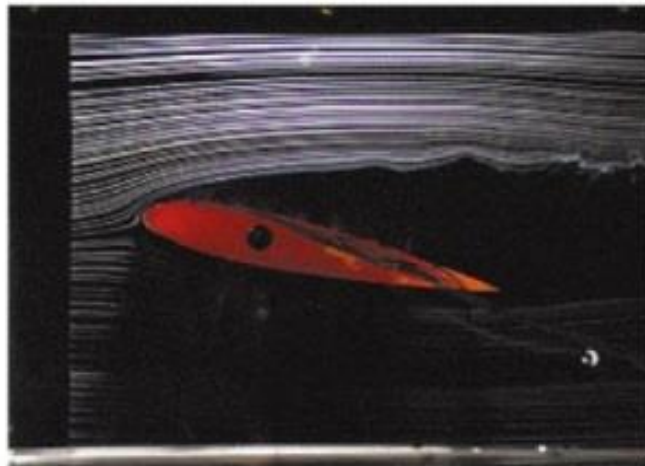
An example of flow separation control over aerofoil with DBD plasma actuator was conducted by Roth (2003). Figure 2-37 shows the flow visualization results of the natural flow and controlled flow around a NACA 0015 aerofoil at angle of attack of 12° and freestream velocity of 2.6 m/s. The corresponding chord-based Reynolds number is 22,000. As can be clearly seen in Figure 2-37 (b), flow separates from the whole upper surface of aerofoil for the uncontrolled case. With the application of eight surface asymmetric DBD plasma actuators mounted on the aerofoil surface at different chordwise positions (shown in Figure 2-37 (a)), momentum is added to the boundary layer, resulting in a flow reattachment to aerofoil surface in Figure 2-37 (c).

Experimental investigation of flow separation control over an aerofoil with DBD plasma actuator was conducted by He et al. (2009b). An asymmetric DBD plasma actuator which consists of two electrodes separated by a dielectric layer was used to control the flow separation around a NACA 0015 aerofoil with the junction between the exposed and grounded electrodes accurately at leading edge ($x/c = 0$). The exposed electrode is mounted on the pressure side and the embedded electrode is on the suction side, so a wall jet in freestream direction could be produced with application of high AC voltages. In their study, DBD plasma actuator was operated with steady actuation and unsteady actuation. Here, steady actuation refers to continuous actuation of plasma actuator at a quasi-steady frequency of several kilo Hertz or tens of kilo Hertz while unsteady actuation denotes an actuation of plasma actuator on and off alternately with a frequency usually an order lower than steady actuation frequency. Aerodynamic lift measurements were made via a force balance for the baseline cases and controlled cases with steady and unsteady actuation of DBD plasma actuators. For the unsteady actuation, Strouhal number $f_e L_{sep}/U_\infty = 1$ and duty cycle of 10% was used, where L_{sep} is the length of the separation region. The lift coefficients versus angle of attack at Reynolds number of 180,000 are plotted in Figure 2-38. As can be seen, the maximum lift coefficient increases with actuation of DBD plasma actuator compared with that for the baseline case, in addition, lift coefficient increases by a larger margin with unsteady

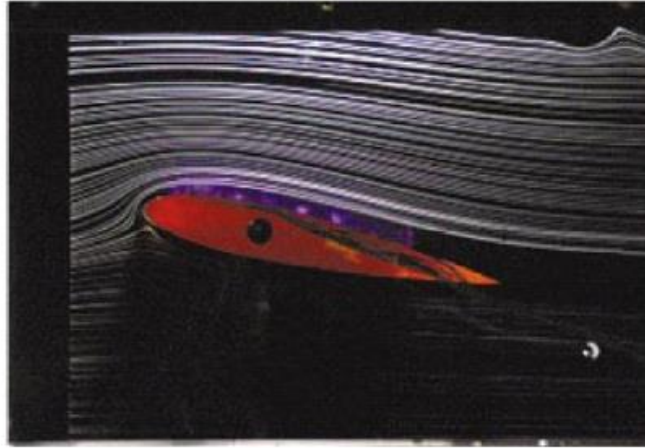
actuation of plasma actuator than that with steady actuation. The stall angle of attack is 14° without plasma while it delays to 18° and 22° respectively when plasma momentum is added continuously and intermittently, which demonstrates that flow separation is delayed with plasma and it seems that unsteady actuation of plasma actuator has a better effect on flow separation control with a lower power consumption than steady actuation.



(a)



(b)



(c)

Figure 2-37 Flow separation control over a NACA 0015 aerofoil at angle of attack of 12°: a) plasma actuators used in the study with dimensions in mm; b) baseline case; c) controlled case with steady actuation of DBD plasma actuators (Roth, 2003).

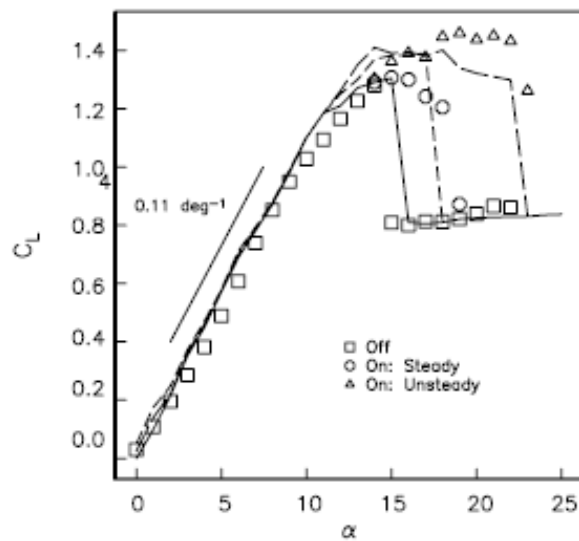


Figure 2-38 Lift coefficient versus angle of attack for a NACA 0015 aerofoil with and without plasma (He et al., 2009). Curves in the figure are from simulation works by (Voikov et al., 2004).

DBD plasma actuator was also applied to the separation control of wake around a circular cylinder. Jukes and Choi (2009a) found that lift of a circular cylinder could be increased by more than 300% at a price of less than 25% increase in drag with

application of DBD plasma actuator at the optimum location of 75° from the streamwise axis, 7° upstream of the natural separation point and plasma timing coincides with the retreat of the separation point. PIV and flow visualizations measurements were conducted to study the dynamics of flow around the circular cylinder before and after plasma is on, the corresponding results are shown in Figure 2-39. Before plasma is on, shear layer transition vortices appear downstream of separation point in frame (i). Momentum is added on the application of plasma, enabling the boundary layer to keep attached over a larger portion of the circular cylinder surface. This causes a break between the attached boundary layer and the old separated shear layer. The tail of the separated shear layer rolls up and lifts away in frame (ii) and (iii) while the re-energized boundary layer keeps attached up to 120° . Moreover, the authors also suggested that the flow separation delay involves not only a momentum addition but also a plasma-induced local flow separation, which increases the turbulence level.

Plasma actuator was used to reduce skin-friction drag as well by creating spanwise travelling waves. Jukes et al. (2006) applied spanwise oscillatory motions into the turbulent boundary layer using two sets of opposing asymmetric plasma actuators. Streamwise vortices are created by oscillatory actuation of plasma actuators, which may disrupt the naturally occurring quasi-streamwise vortices within the boundary layer, thus disrupting the turbulence generation cycle. Consequently, a maximum skin-friction drag reduction of 45% was observed downstream of plasma actuators. The DNS numerical simulation work of this experiment was also conducted and a skin-friction drag reduction of 47% was obtained (Elam, 2012).

Whalley and Choi (2014) did PIV measurements of turbulent skin-friction drag reduction with DBD plasma actuators at a freestream velocity of 1.8 m/s. Basically, unidirectional and bidirectional spanwise travelling waves were produced with configurations of plasma actuator arrays in Figure 2-40 (a) and (b) respectively. For each forcing configuration, plasma actuators were operated over a four-phase cycle, in sequence from phase (i) to (iv) for a duration of T^+ shown in Figure 2-40 (c, d). Here, T^+ is defined as $T^+ = Tu_\tau^2/\nu$, where T is the forcing period, u_τ is the friction velocity and ν is kinetic viscosity. The unidirectional plasma travelling waves are created by actuating a single plasma actuator during each phase of the forcing cycle,

similarly, bidirectional plasma travelling waves are generated by two plasma actuators operated in opposing directions simultaneously over each phase. Figure 2-41 shows the flow structures of a turbulent boundary layer taken at $3/4 T^+$ in y - z plane at $x^+ = xu_\tau/\nu = 200$ and x - z plane at $y^+ = yu_\tau/\nu = 5$ with spanwise plasma travelling waves. Streamwise vortices are created when plasma travelling waves are applied, as identified by λ_2 contours. A streamwise vortex with negative vorticity appears at $z^+ = zu_\tau/\nu \approx 375$ when unidirectional spanwise travelling waves are applied, however, both positive and negative vortices are created as a consequence of two plasma actuators operating in opposing directions during each phase of plasma forcing cycle, specifically, a large elongated negative vortex is centred at $(y^+, z^+) \approx (50, 350)$ and a positive vortex locates at $(y^+, z^+) \approx (100, 400)$ which is created during the second phase of plasma forcing, and they are interacting with each other. These streamwise vortices entrain low-speed streamwise velocity fluids from near-wall region into their cores and around the peripheries, causing a reduction in streamwise velocity inside the boundary layer. Moreover, wide ribbons of low-speed streamwise velocity are observed upstream of the streamwise vortices with a traveling direction in positive spanwise direction. The low streamwise velocity ribbons within the viscous sublayer are thought to be of great significance to turbulent skin-friction drag reduction (Du et al., 2002; Du and Karniadakis, 2000).

Plasma actuator could be used to control broadband noise as well. Li et al. (2010) conducted an investigation of noise control over a bluff body with plasma actuators, the motivation behind which is to reduce the landing gear noise during approach phase of an aircraft. The test model is shown in Figure 2-42 (a), which consists of a circular cylinder and an oblique strut. Plasma actuator is located at ± 90 deg of the cylinder surface relative to the oncoming flow and spans from the top end of the cylinder to the area that covers the strut. Measurements taken in an anechoic chamber at freestream velocity of $U_\infty = 30$ m/s show that noise level could be significantly reduced with plasma actuator on (see Figure 2-42 (b)). This well demonstrates plasma actuators' promising applications in noise attenuation. The authors suggested that the noise attenuation might be related to a reduction of flow impinging velocity on the strut, which is caused by the upstream plasma forcing and confirmed by particle image velocimetry measurements.

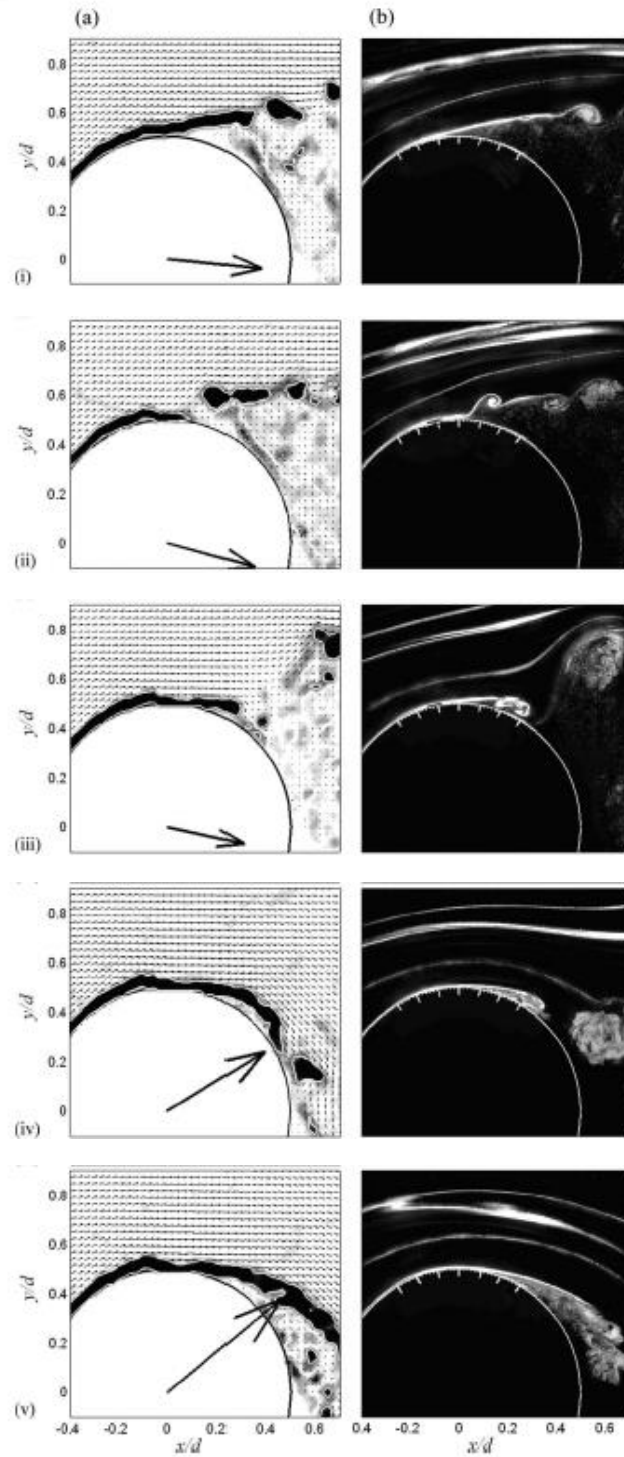


Figure 2-39 Instantaneous flow field above the upper surface of a circular cylinder during a plasma pulse: (a) instantaneous vorticity and velocity vectors; (b) flow visualization. Plasma is on in (ii) and (iii). Time increments from top to bottom are $t/T_K = -0.01, 0.07, 0.18, 0.43, \text{ and } 0.62$ (Jukes and Choi, 2009a).

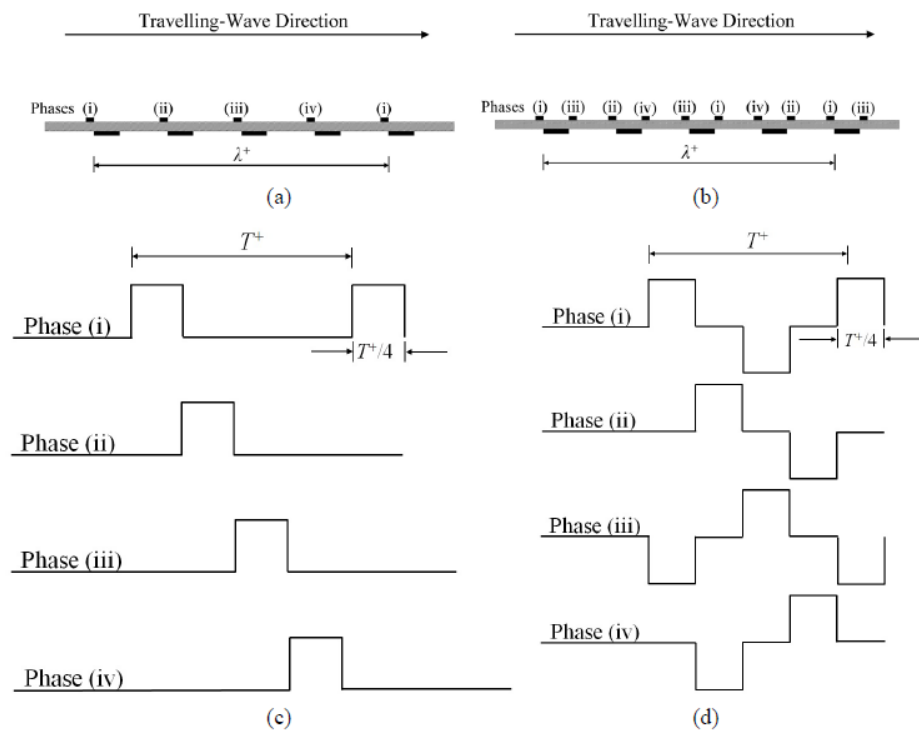


Figure 2-40 Schematic of plasma excitation for: (a) unidirectional spanwise travelling waves and (b) bidirectional spanwise travelling waves; (c) and (d) shows plasma forcing with unidirectional and bidirectional traveling waves respectively. $\lambda^+ = 500$ (100 mm) and $T^+ = 82$ (208 ms) (Whalley and Choi, 2014).

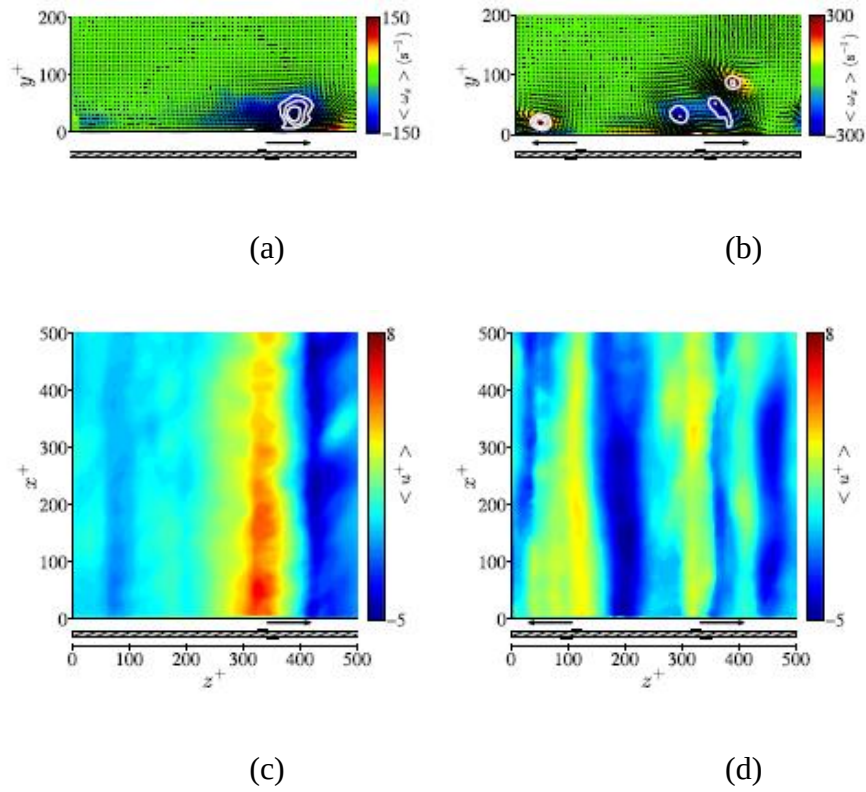
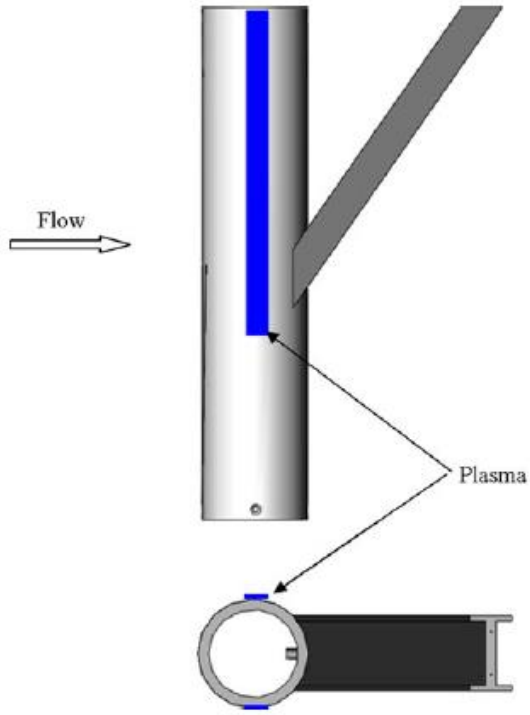
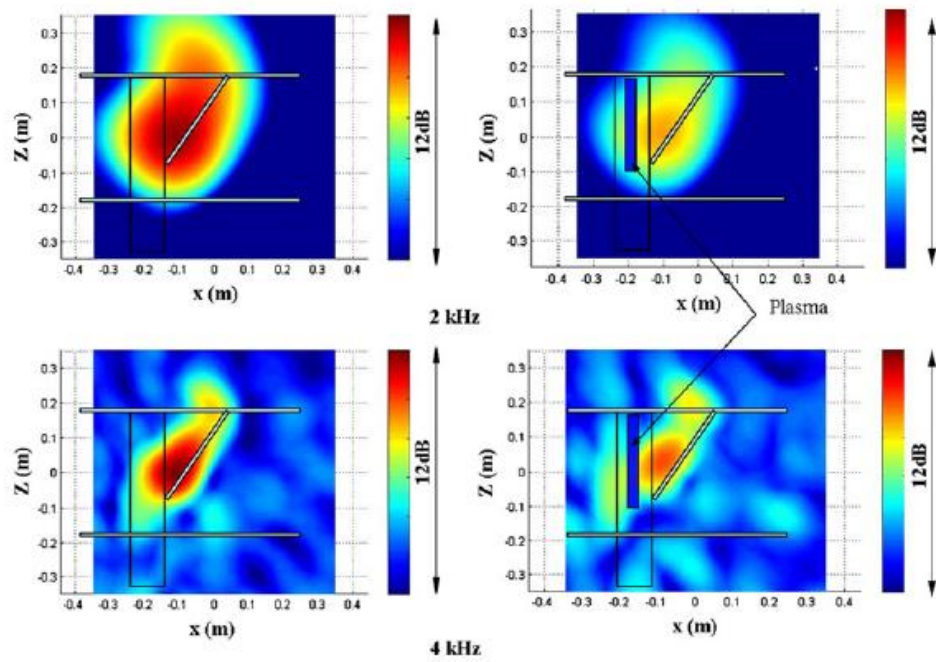


Figure 2-41 A comparison of data in y - z plane at $x^+ = 200$ and x - z plane at $y^+ = 5$ of turbulent boundary layer with unidirectional (a ,c) and bidirectional (b,d) spanwise travelling waves showing (a, b) streamwise vorticity (y - z plane (c, d) fluctuating streamwise velocity (x - z plane) at $3/4 T^+$. Vortical structures are indicated by white contours of negative λ_2 (Whalley and Choi, 2014).



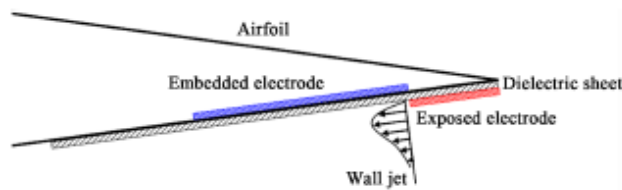
(a)



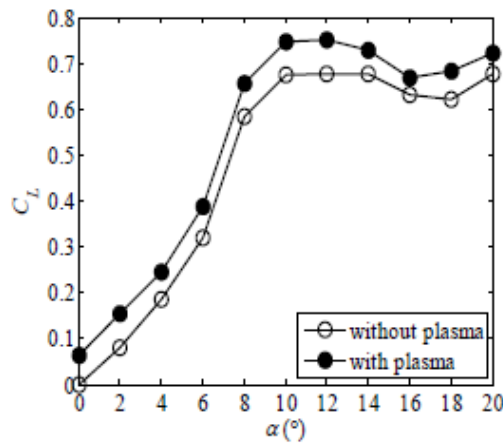
(b)

Figure 2-42 Noise control over a bluff body with DBD plasma actuators: (a) schematic of the test model; (b) noise source maps at 2 kHz (top row) and 4 kHz (bottom row) with (left) and without plasma (right) (Li et al., 2010).

Feng et al. (2014) proposed a concept of plasma Gurney flap by configuring a plasma actuator on the pressure surface of aerofoil trailing edge. The sketch is illustrated in Figure 2-43 (a). When plasma actuator is activated, a wall jet directed upstream will be induced. This plasma jet draws air from the suction surface to the pressure surface, as a consequence, velocity around the suction surface is increased while velocity around pressure surface is decreased. In addition, the interaction between this plasma jet and freestream results in a recirculation region around the plasma actuator over the pressure surface. The recirculation region downstream of aerofoil is decreased in size and the wake is shifted downwards (Figure 2-44 (a) and (b)), suggesting an increase in equivalent aerofoil camber. Such behaviours are similar to those by a mechanical Gurney flap. This plasma Gurney flap could effectively improve aerodynamic performance of an aerofoil, as can be seen in the lift curves with and without plasma Gurney flap in Figure 2-43 (b).

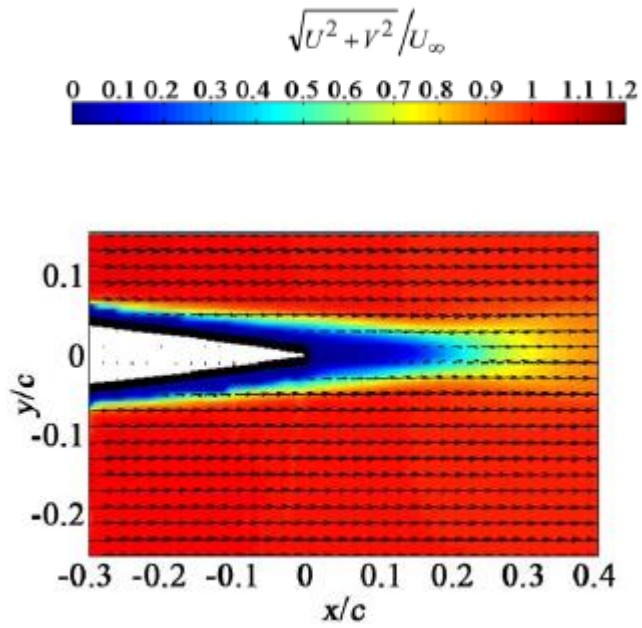


(a)

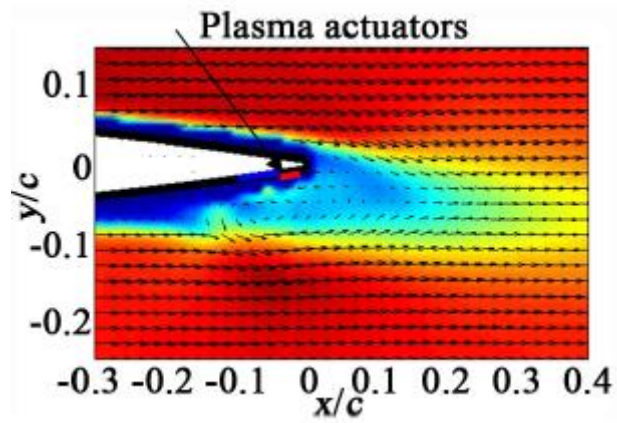


(b)

Figure 2-43 (a) Schematic of plasma virtual Gurney flap; (b) lift curves versus angle of attack with and without plasma virtual Gurney flap (Feng et al., 2014).



(a)



(b)

Figure 2-44 Time-averaged flow field around aerofoil trailing edge : (a) base flow; (b) controlled with plasma actuators (Feng et al., 2014).

3 Experimental setup

3.1 Wind tunnel

The experiments were conducted in an open-return, low-speed wind tunnel at University of Nottingham. This is a suction wind tunnel and the exhausted flow is ducted to the outside. It is driven by a motor with a maximum power output of 37 kW and a maximum rotating speed of 1750 RPM. The maximum achievable flow velocity is around 5.5 m/s.

The test section of this wind tunnel is 1.5m long and the cross section is $0.3\text{m} \times 0.3\text{m}$ in width and height respectively. A 7:1 area ratio contraction section is placed upstream of the test section. And a series of screens and honeycomb are utilized in front of the contraction section to improve the flow uniformity.

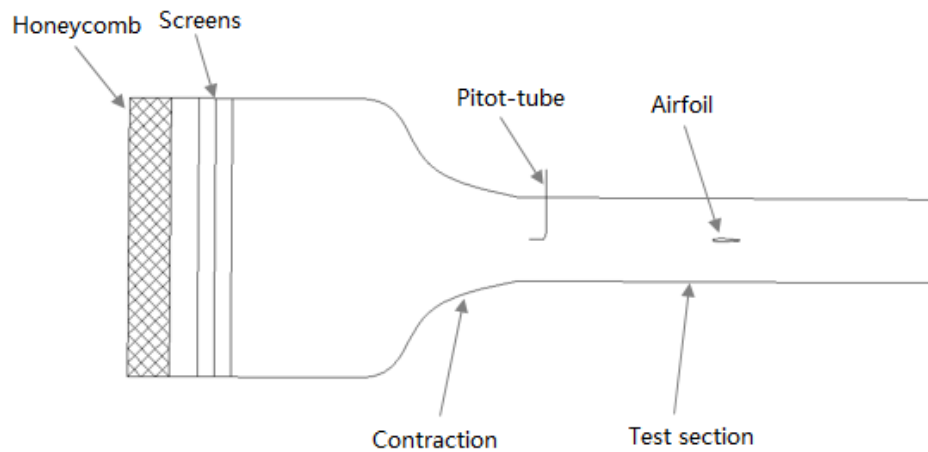


Figure 3-1 Schematic of the open-loop wind tunnel.

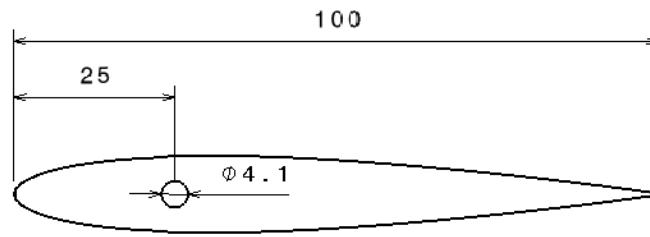
For the PIV measurements, a NACA 0012 aerofoil with chord length $c = 100\text{mm}$ and span $b = 250\text{mm}$ was set up in the wind tunnel test section and located at 0.7m downstream of the contraction section. The freestream velocity U_∞ was set at 3m/s, corresponding to Reynolds number $Re = 20,000$ based on the chord length of the aerofoil. The turbulence intensity is less than 0.3% at freestream velocity $U_\infty = 3\text{m/s}$. Two end plates with a size of $300\text{mm} \times 200\text{mm}$ in streamwise and spanwise directions respectively were used at both sides of the aerofoil span to enhance the two dimensionality of the flow around the aerofoil.

To make sure freestream velocity is set precisely, a Pitot-tube was always used to monitor the freestream speed. The Pitot-tube was placed at the centre of the test section in height and 0.1m downstream of the contraction section. The difference between the velocities measured by Pitot-tube and PIV was less than 1.5%. So the PIV measurement error can be considered to be within 1.5%. The temperature of the wind tunnel room was controlled by a Denco air conditioner and the ambient temperature was always kept at 20 ± 0.5 °C for the measurements.

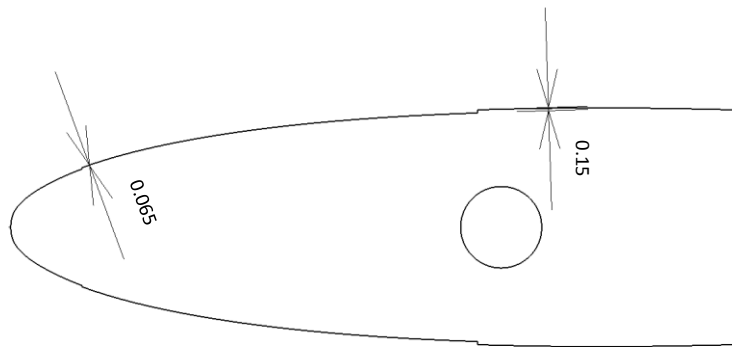
3.2 Aerofoil

A NACA 0012 aerofoil was used in the current study, and the aerofoil has a chord length of 100mm and a span of 250mm. It was set up in the wind tunnel via a rod, which was located at the aerodynamic centre, 25%c from the leading edge. A DBD plasma actuator consisted of two exposed electrodes, an embedded electrode and a Kapton dielectric sheet is mounted on the surface of the aerofoil and centred at the leading edge of the aerofoil.

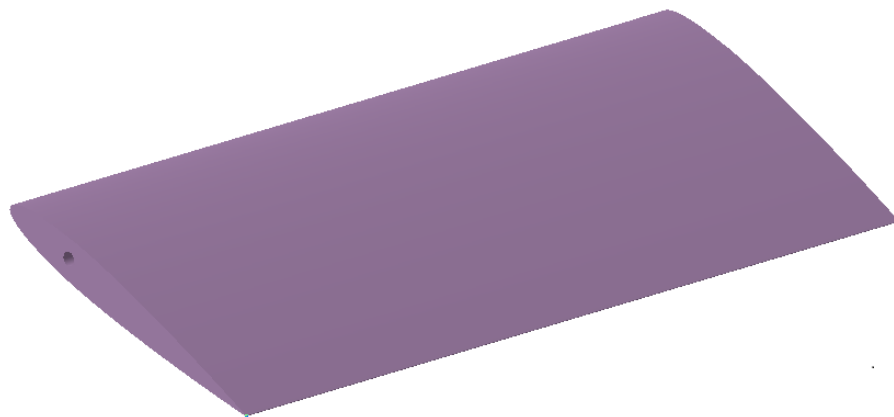
The aerofoil was made by 3D printing with ABS material which was chosen because of its high strength and melting point, otherwise the heat from the powerful laser light might damage the aerofoil. The schematic of this 3D printing aerofoil can be seen in Figure 3-2. As can be seen in Figure 3-2 (a), a circular hole with a diameter of 4.1mm and a length of 250mm was printed at 25%c which was used to locate the rod. In order to make the surface smooth, a recess with a depth of 150µm and length of 250mm was made to mount the Kapton dielectric sheet. Another recess, which is 65µm deep and covers full span length, was made as well to put the embedded electrode. These two recesses can be seen in the enlarged window of left view in Figure 3-2 (b). The centres of these two recesses locate at $x/c = 0$, namely, the leading edge of the aerofoil. A three-dimensional view of the aerofoil is shown in Figure 3-2 (c).



(a) Left view



(b) Enlarged window



(c) Three-dimensional view

Figure 3-2 3D printing aerofoil.

3.3 Plasma Generation System

DBD Plasma actuator was used in this study to control the flow separation around a NACA 0012 aerofoil. Design of plasma actuator is illustrated in Figure 3-3. It consists of two 10-mm-wide exposed copper electrodes and a 10-mm-wide embedded electrode separated by a Kapton dielectric sheet. The exposed electrodes are located symmetrically on both sides of the embedded electrode. The thickness of all the electrodes is $65\mu\text{m}$ and the dielectric layer is $150\mu\text{m}$ thick. Exposed electrode is exposed to the air while the embedded electrode is totally encapsulated by the dielectric sheet and the test model. When both of the exposed electrodes are driven by identical high AC voltages, a wall-normal jet will be generated. Moreover, this jet could be vectored via adjusting the input voltage signals applied to the two exposed electrodes. Plasma actuator is mounted on the aerofoil and centred at the aerofoil leading edge, namely, $x/c = 0$, as shown in Figure 3-3 (b).



(a)



(b)

Figure 3-3 Schematic of DBD plasma actuator: (a) plasma actuator configuration; (b) illustration of plasma actuator mounted on aerofoil with electrodes represented by red lines and Kapton dielectric sheet indicated by an orange line.

The plasma actuator was powered by PSI-MCPG2503C power amplifier devices purchased from KI Tech Ltd. This power supply could typically produce a sinusoidal voltage output with a maximum frequency of 25 kHz and a maximum voltage of 8 kV. However, the maximum voltage and frequency depends on many factors, for example, width of the electrode, length of plasma actuator, thickness and material of the dielectric barrier and so on. In this study, a sinusoidal voltage with a frequency of 34 kHz was used. The maximum power output of the power supply is 150 W.

3.4 PIV

A time-resolved Dantec PIV system was used to study the flow induced by a DBD plasma actuator and how the plasma jet helps to control the flow separation around the leading-edge of a NACA 0012 aerofoil. This is beneficial to understand how the plasma actuator affects the adjacent flow, thus we can make better use of plasma actuator to control the leading-edge flow separation.

This PIV system consists of a TSI 9307-6 oil droplet generator, a Litron LDY 302 PIV 100W Nd: YLF laser, a SpeedSense 9060 high-speed camera and a computer. The experimental arrangement can be found in Figure 3-4. The seeding particles should be small and light enough so that the particles could fully trace the flow without introducing any disturbances. In our experiment, Olive oil particles with a diameter of $1\mu\text{m}$ were generated through the droplet generator and uniformly seeded into the wind tunnel to trace the flow. The laser has a maximum output of 100W, the maximum responding frequency is 10000Hz and the wave length is 527nm. This powerful laser was used to illuminate the seeding particles so that the particles could be captured by the high-speed camera. The laser light plane was aligned along the centreline of the NACA 0012 aerofoil and measurements were conducted at the midspan. The camera was set to take 2000 images per second and the field of view was $-0.2 \leq x/c \leq 0.45$ and $-0.15 \leq y/c \leq 0.25$ with resolutions of 1280×800 pixels in image width and height directions respectively. Dual frame mode was used and the time between frames was set at $200\mu\text{s}$. Each image was exposed to a $10\mu\text{s}$ laser pulse.

Analysis was conducted to calculate the velocity field and vorticity field using Dantec DynamicStudio 4.10. A recursive cross-correlation technique was used to compute the velocity vectors of the flow field around NACA 0012 aerofoil. The interrogation window was set at 32×32 pixels with 50% overlap in both horizontal and vertical directions. These velocity vectors were validated through the local median filter, which usually results in less than 3% erroneous vectors by calculating the deviation from the surrounding vectors. These erroneous velocity vectors were then replaced by interpolation of the ambient velocity vectors.

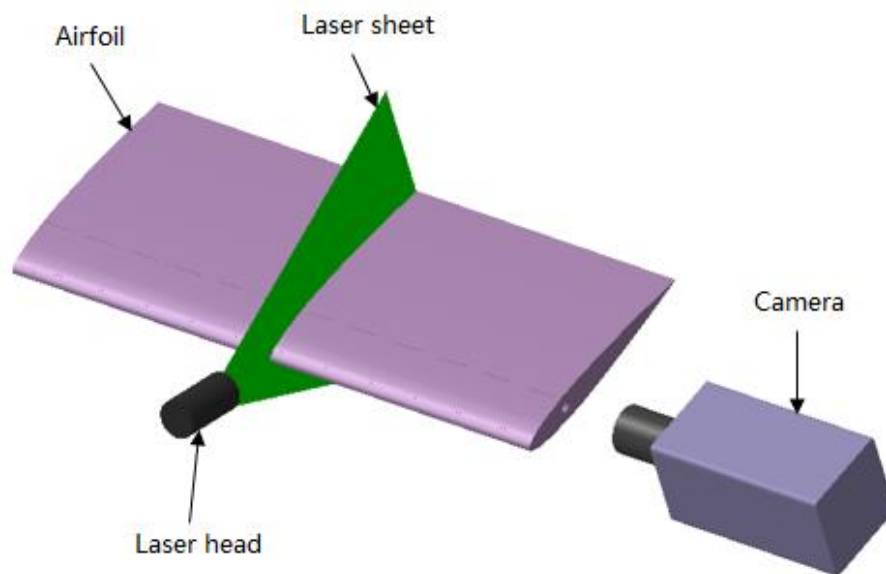


Figure 3-4 Setup of PIV system.

3.5 Experimental procedure

The experimental configuration is illustrated in Figure 3-5 and the corresponding experimental procedure is as follows:

- a) Mount DBD plasma actuator to aerofoil carefully to make sure the centre of plasma actuator is located at the aerofoil leading edge.
- b) Set up NACA 0012 aerofoil with end plates in the wind tunnel test section.
- c) Connect the DBD plasma actuator to power supplies.
- d) Set plasma power supplies to “remote control” mode and controlled by a function generator which is triggered by the camera signal.

- e) PIV setup: put the head of laser guide on top of the test section to illuminate the particles within the flow field above the aerofoil upper surface; adjust the laser head to make sure it gives a vertical plane laser light along the aerofoil midspan; set up the high-speed camera at a proper distance to capture the interested field of view; adjust the focus of the camera to assure camera is focused on the laser plane; adjust the aperture of the camera to get bright enough images so that the particles could be distinguished from the images; Measure the scale factor.
- f) Switch on the extractor fan so that the exhausted air with Olive oil particles could be extracted outside from the lab.
- g) Reset the interlock system: if somebody breaks in, laser system will shut down.
- h) Set the aerofoil at proper angle of attack.
- i) Switch on the wind tunnel and adjust the freestream velocity which is monitored by Pitot-tube.
- j) Switch on the laser and camera.
- k) Set the time between dual laser pulses, exposure time, sampling frequency and number of images to acquire in DynamicStudio software.
- l) Calibrate the camera.
- m) Seed Olive oil particles uniformly into the wind tunnel.
- n) Click “Acquire” button in DynamicStudio software to start measurement and save the collected data.

In addition, laser goggles, experimental boots and coat should be always put on while doing experiments to assure the experimental safety.

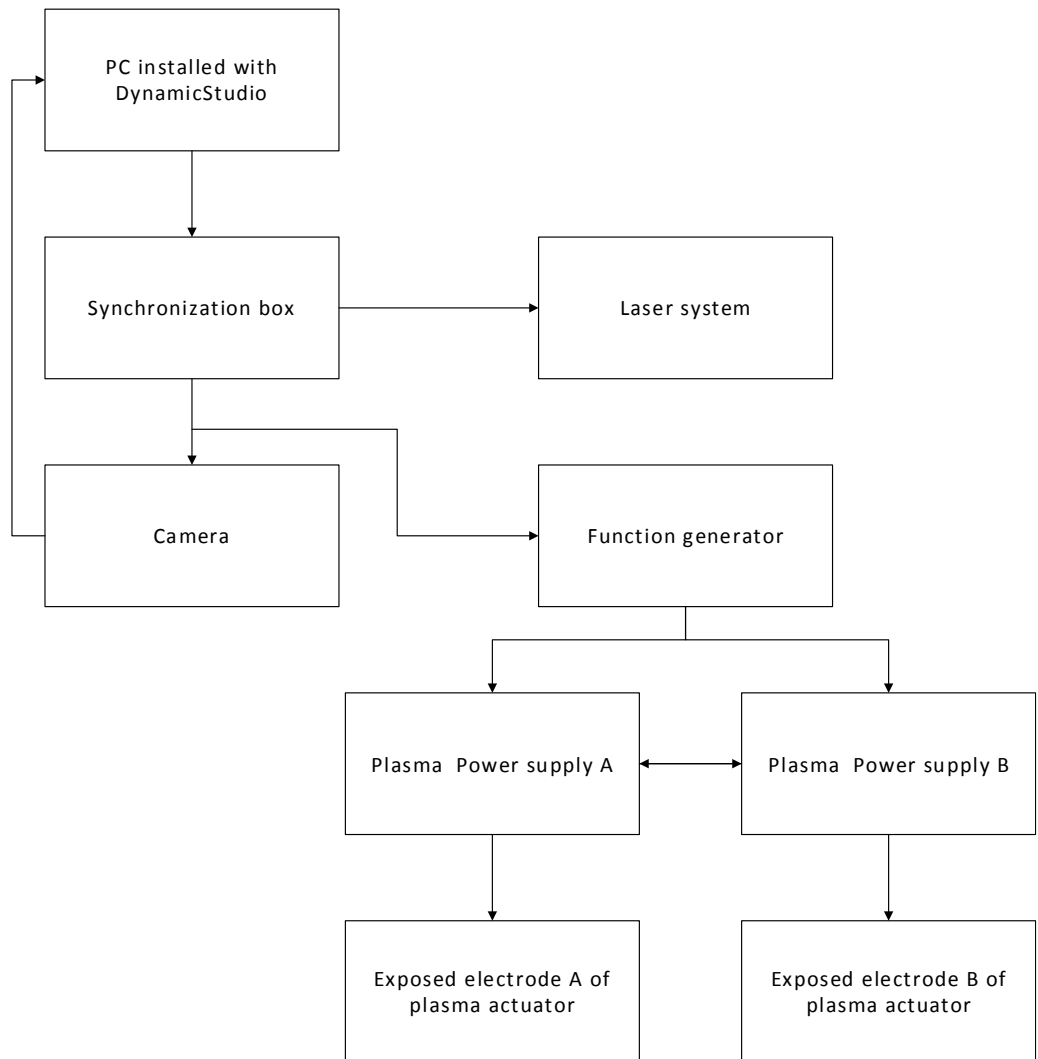


Figure 3-5 Schematic of experimental configuration including PIV system and plasma generation system.

4 Baseline Flow Characteristics of Flow Field Around NACA 0012 Aerofoil

In this chapter, characteristics of baseline flow around NACA 0012 aerofoil at different angles of attack was studied to characterise the baseline flow. Angles of attack of $\alpha = 0^\circ, 8^\circ, 10^\circ, 12^\circ, 14^\circ$ and 16° were chosen in this study, they represent no separation, pre-stall, stall, light post-stall, moderate post-stall and deep post-stall flow conditions at chord-based Reynolds number of 20,000 respectively. PIV technique is a laser-based measurement technique, which is nonintrusive and will not introduce any disturbances into the measured flow. In the meanwhile, this technique could give velocity vectors within an area rather than a point by some other techniques. Therefore, PIV technique was utilised in the current study. Since the whole project focuses on leading-edge flow separation control around an aerofoil, the flow field around the aerofoil leading edge is where we are most interested in. Accordingly, a measurement window of $-0.2 \leq x/c \leq 0.45$ and $-0.15 \leq y/c \leq 0.25$ was chosen for the results presented in this chapter and even the flow control chapters. Sampling frequency of high-speed camera is set at 2,000 Hz to capture flow structures above the upper surface of aerofoil at different angles of attack. Dual frame mode was used to improve the measurement accuracy and time between each frame is $200\mu s$. Freestream velocity is set at 3m/s, at which the chord-based Reynolds number is 20,000 and turbulence intensity is less than 0.3%.

Streamwise velocity contours superimposed by streamlines of the time-averaged flow fields around NACA 0012 aerofoil at angles of attack of $\alpha = 0^\circ, 8^\circ, 10^\circ, 12^\circ, 14^\circ$ and 16° are presented in Figure 4-1. At angle of attack of 0° , flow goes along the aerofoil surface and no negative velocity appears, indicating an attached flow moving in streamwise direction. A recirculation region appears above the upper aerofoil surface at angle of attack of 8° , which could be suggested by both the streamline and streamwise velocity contour. As angle of attack increases, the recirculation area becomes bigger and the magnitude of negative velocity increases as well, indicating a severer flow separation above aerofoil surface.

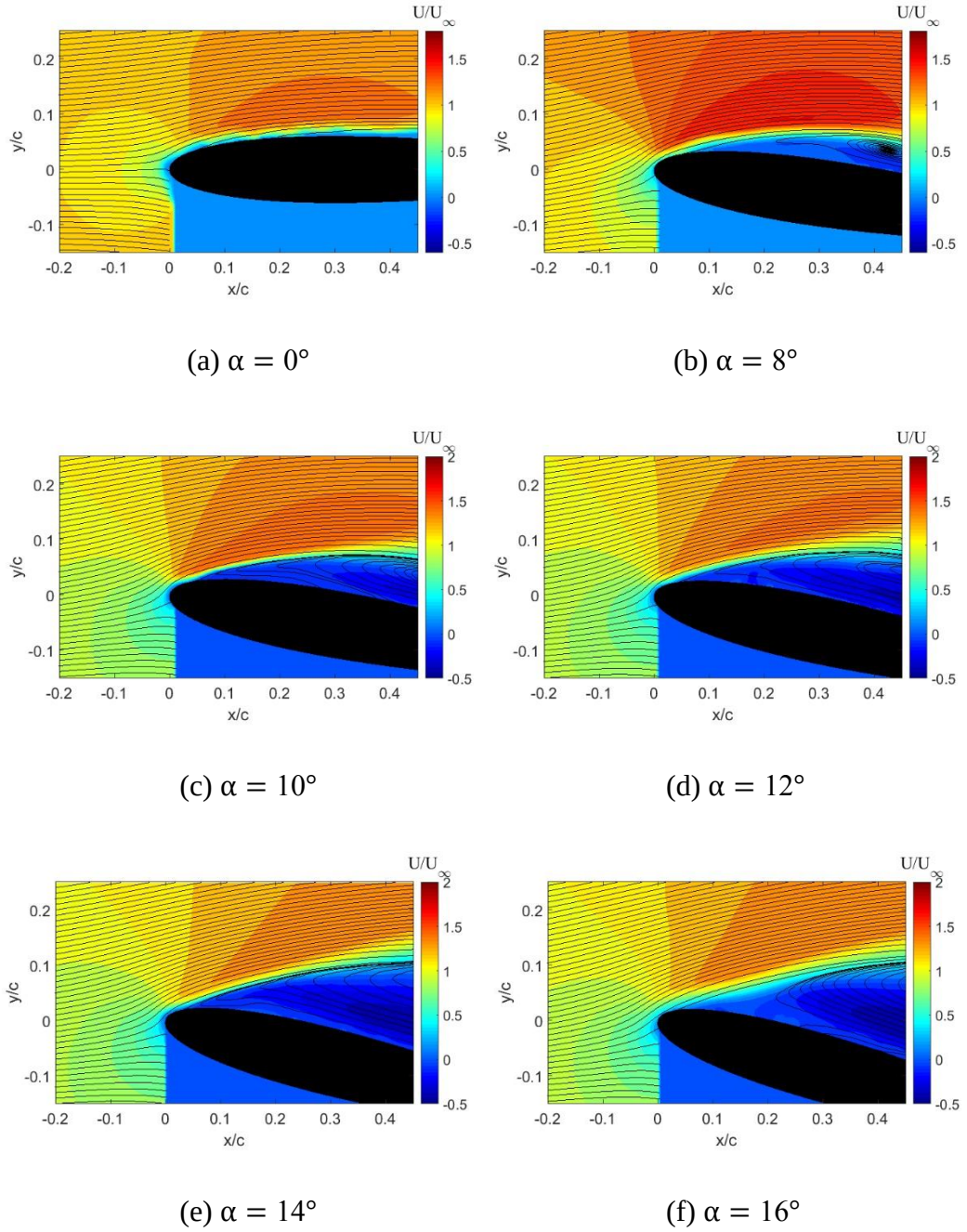
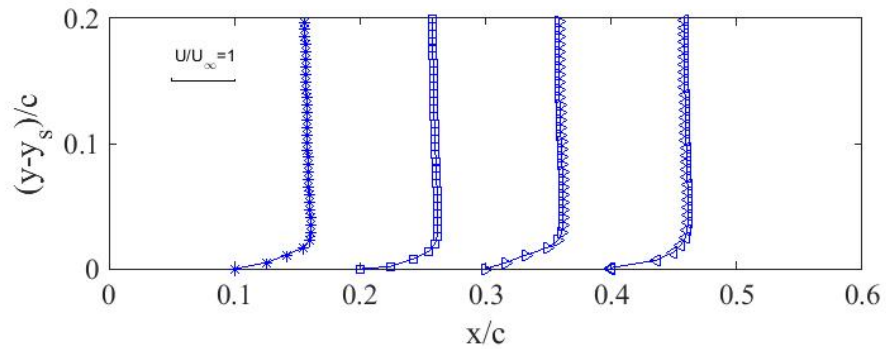


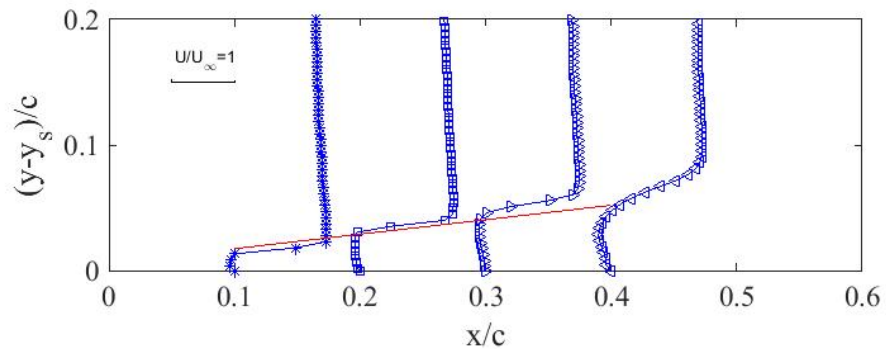
Figure 4-1 Normalized streamwise velocity contour superimposed by streamline of the time-averaged flow fields around a NACA 0012 aerofoil at varying angles of attack. Freestream velocity is 3m/s and the corresponding chord Reynolds number is 20,000.

Time-averaged streamwise velocity profiles are plotted in Figure 4-2. The linear fit lines of inflection points are plotted as well in red solid lines. PIV measurements only give a number of velocity vectors at a grid of points, to get an accurate linear fit of inflection points, interpolation of grid points and velocity vectors was used,

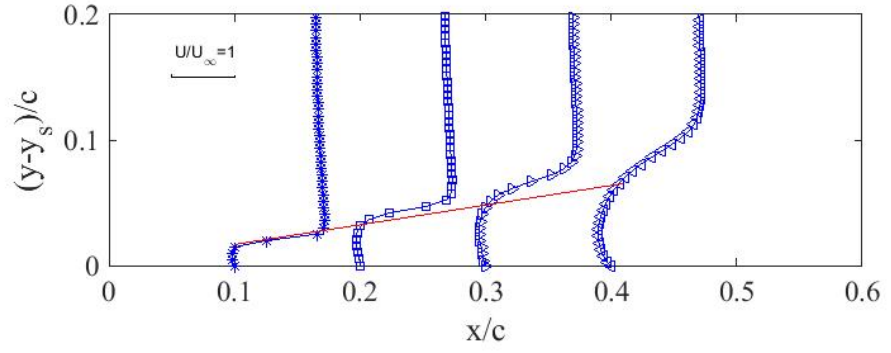
specifically, two more data points were interpolated between the original successive data points. As for the velocity profiles, the interpolated data points are not plotted in the figure. It can be obviously seen that inflection points locate further away from aerofoil surface at $x/c = 0.3$ and 0.4 as angle of attack increases. In the meanwhile, the height of inflection point increases with chordwise positions within the measured ranges. The observed phenomenon is consistent with previous findings by Brendel and Mueller (1988) over a Wortmann FX63-137 aerofoil. O'Meara, M.M. and Mueller (1987) proposed an angular parameter of separation angle γ to dictate flow separation, in the current study, it is used as a quantitative method to indicate the severity of flow separation, in other words, a larger γ always corresponds to a severer separation. Here, separation angle γ is defined as the angle between linear fit of recirculation region height and surface arc length, which is demonstrated in Figure 4-3. For simplicity, chordwise position is used in this study rather than surface arc length. The separation angles calculated on the basis of this technique are listed in Table 1.



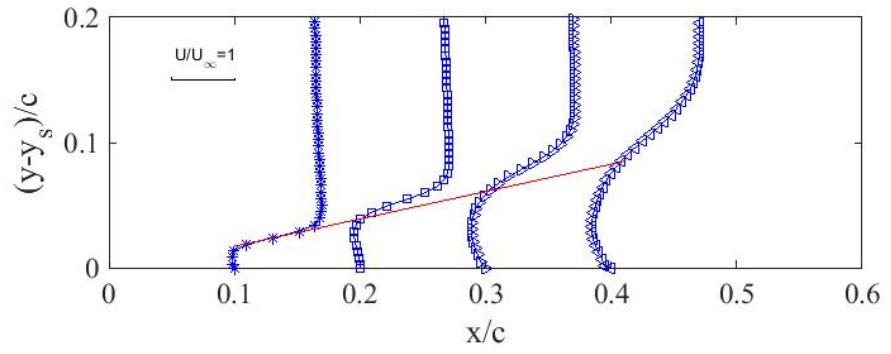
(a) $\alpha = 0^\circ$



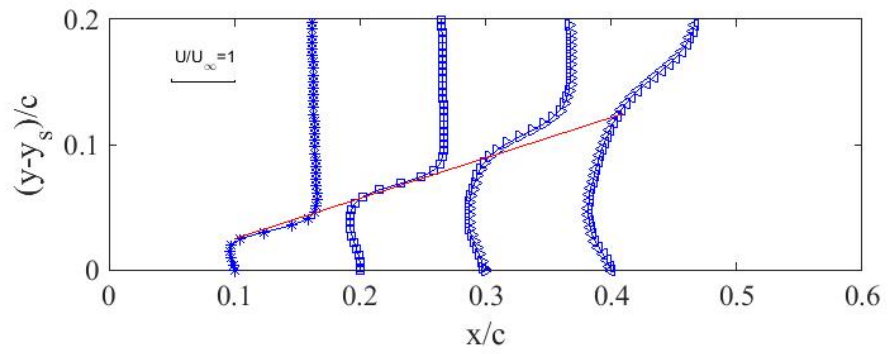
(b) $\alpha = 8^\circ$



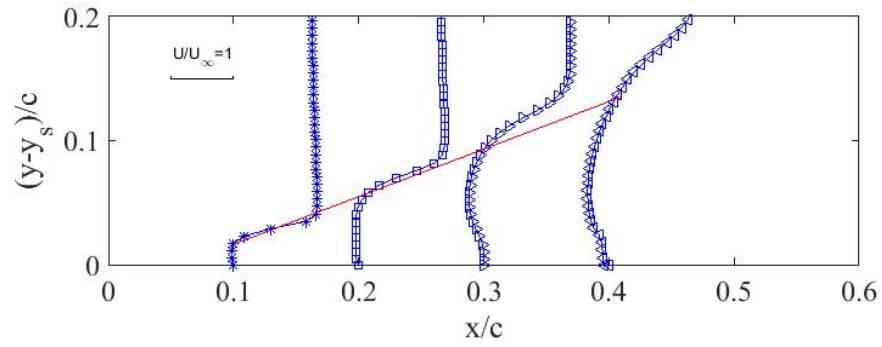
(c) $\alpha = 10^\circ$



(d) $\alpha = 12^\circ$



(e) $\alpha = 14^\circ$



(f) $\alpha = 16^\circ$

Figure 4-2 Mean streamwise velocity profiles at streamwise positions of $x/c = 0.1$, 0.2 , 0.3 and 0.4 . The solid red line indicates the linear fit of inflection points.

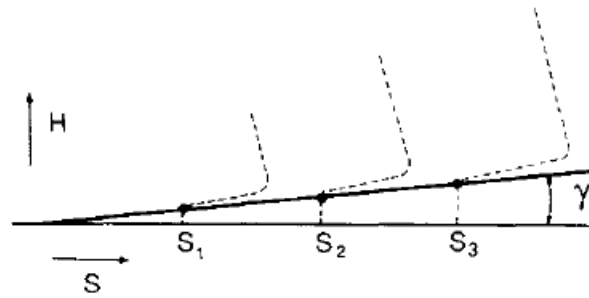


Figure 4-3 Method for separation angle calculation (O'Meara, M.M. and Mueller, 1987).

Table 1 Separation angles for the base flow around NACA 0012 aerofoil at Reynolds number of 20,000.

α	0°	8°	10°	12°	14°	16°
γ	N/A	5.7°	8.9°	12.2°	17.9°	19.8°

Contours of normalized Reynolds stress for the flow fields around NACA aerofoil 0012 at different angles of attack are computed and plotted in Figure 4-4. At angle of attack of 0° , Reynolds stress exists in the region just above the aerofoil surface and the magnitude is very small, which is in very good agreement with an attached flow around NACA 0012 aerofoil. As angle of attack increases to 8° , Reynolds stress of positive sign appears in a region away from the aerofoil surface, which

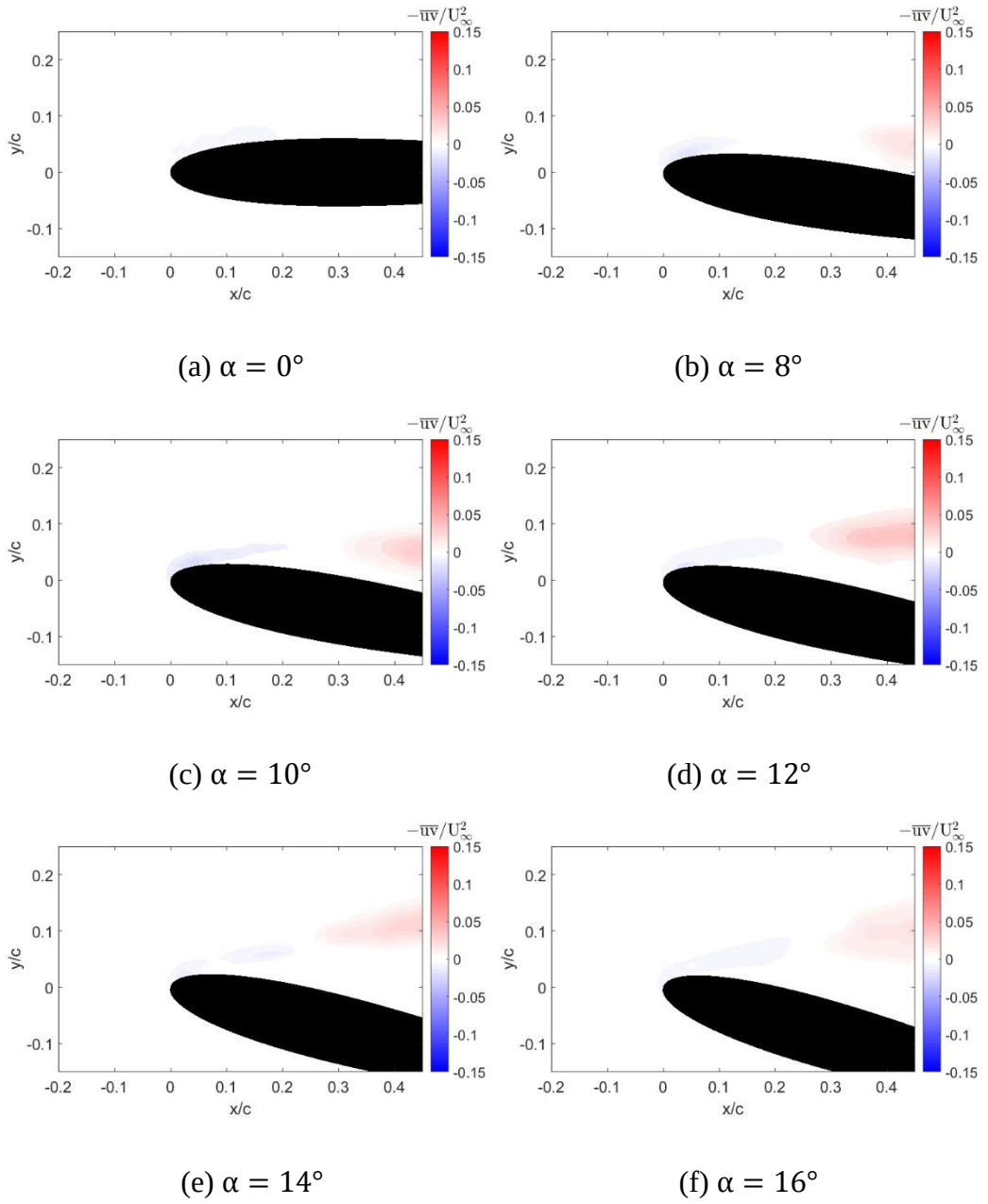
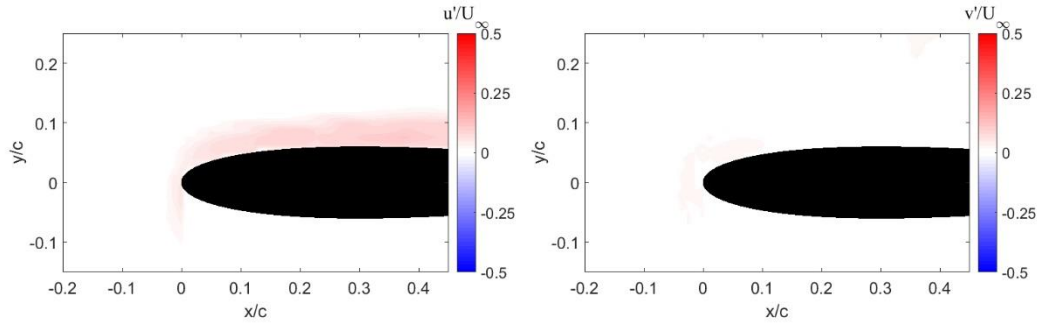


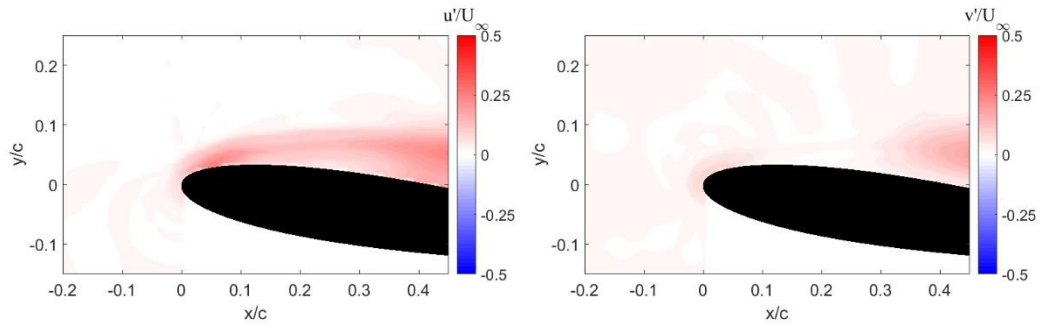
Figure 4-4 Normalized Reynolds stress contours with aerofoil at different angles of attack. Freestream velocity is 3m/s.

might be caused by the separated shear layer and shedding vortices. The vertical position of the positive Reynolds stress region moves further away from the suction surface with increasing angles of attack, suggesting the separated shear layer locates at a larger vertical distance from the aerofoil surface at larger angles of attack. It's interesting to mention that the Reynolds stress magnitude increases when the angle of attack increases from 8° to 12° while it decreases with angle of attack over the range of $12^\circ \leq \alpha \leq 16^\circ$. The streamwise and vertical r.m.s. velocity fluctuations are plotted in Figure 4-5. Generally, the r.m.s. velocity fluctuation amplitude increases in streamwise direction, which is similar to the findings over a stalled aerofoil by Yarusevych (2006). The location of the maximum r.m.s. velocity approximately corresponds to the maximum Reynolds stress location, moving away from aerofoil surface when it travels downstream. It is worth to note that the r.m.s. velocity fluctuations indicate a thick boundary layer near the aerofoil surface at angle of attack of 0° , which is probably not the physical case and this might be due to the measurement error of PIV technique in the near-wall region.

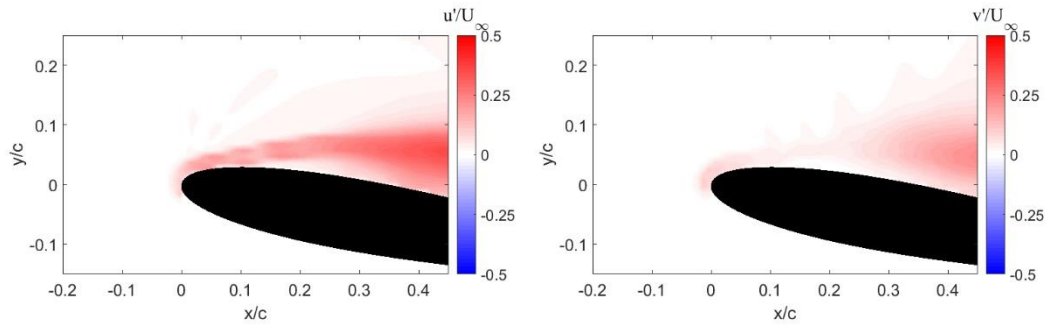
a) $\alpha = 0^\circ$



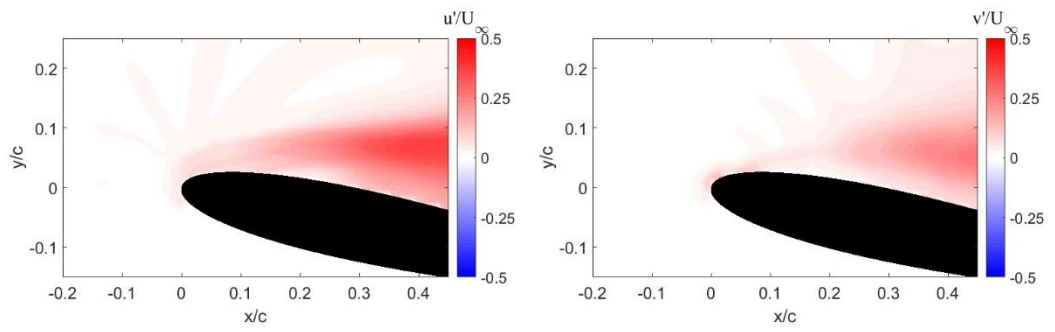
b) $\alpha = 8^\circ$



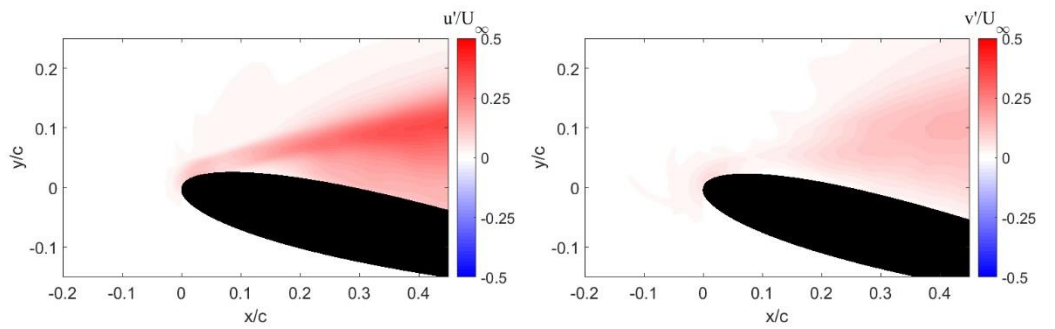
c) $\alpha = 10^\circ$



d) $\alpha = 12^\circ$



e) $\alpha = 14^\circ$



f) $\alpha = 16^\circ$

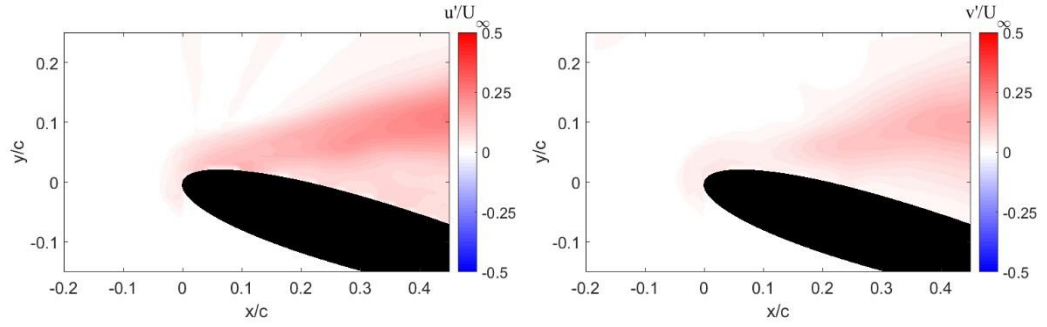


Figure 4-5 Contours of streamwise (left column) and vertical (right column) root-mean-square velocities for baseline flows at different angles of attack.

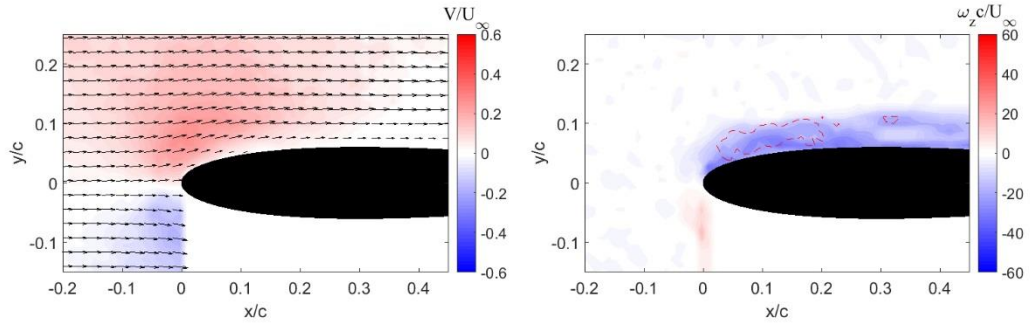
Dynamics of flow around a NACA 0012 aerofoil was also investigated at different angles of attack. Figure 4-6 shows instantaneous vertical velocity contours, velocity vectors, spanwise vorticity and swirling strength isolines at freestream velocity of 3 m/s and angle of attack of 0° . Here, swirling strength is a method that utilizes the imaginary part of the complex eigenvalue of velocity gradient tensor to visualize vortices (Zhou et al., 1999). Flow always goes along the aerofoil surface within the field of view at all the time instants plotted in Figure 4-6, specifically, $t^+ = 0, 0.09, 0.18$ and 0.27 , here, non-dimensional parameter t^+ is defined as $t^+ = tU_\infty/c$. This demonstrates that flow always attaches to aerofoil surface even in a transient sense at angle of attack of 0° . Moreover, the flow field looks similar at different time instants.

Instantaneous flow fields around NACA 0012 aerofoil at angle of attack of 8° are plotted in Figure 4-7. Due to the existence of adverse pressure gradient, flow separates and the spanwise vorticity away from aerofoil surface indicates a separated shear layer at this angle of attack. Compared with that at angle of attack of 0° , flow is much more unsteady and vortex shedding is accompanied when separation occurs. For example, at $t^+ = 0$, two vortices appear in the field of view with the first one at $x/c \approx 0.42$ and the second one locating at $x/c \approx 0.32$, shown in Figure 4-7 (a). These vortices travel downstream with time, as can be seen in Figure 4-7 (b), the first vortex travels out of the field of view and the second one travels to $x/c \approx 0.36$, in the meanwhile, the third one sheds from the separated shear layer and forms at $x/c \approx 0.27$. The second vortex moves further downstream in Figure 4-7 (c)

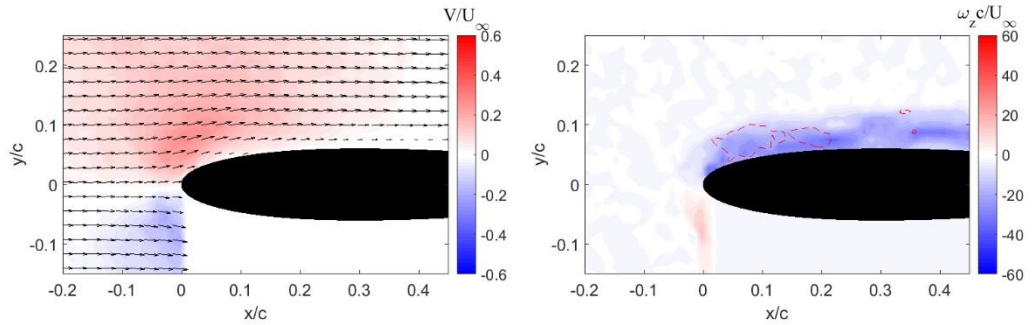
and out of the field of view in Figure 4-7 (d). The track of the third vortex at $t^+ = 0.09, 0.21$ and 0.29 clearly illustrates its movement from $x/c \approx 0.27$ to $x/c \approx 0.4$. This clearly shows the vortex formation and shedding process when flow separation happens around NACA 0012 aerofoil.

Dynamics of flow field around NACA 0012 aerofoil at larger angles of attack of $10^\circ, 12^\circ, 14^\circ$ and 16° were studied as well, the corresponding vertical velocity, streamline, spanwise vorticity and swirling strength contours are presented in Figure 4-8, Figure 4-9, Figure 4-10 and Figure 4-11 respectively. As shown in the figures, a similar process of vortex formation and shedding is observed at larger angles of attack. However, the reverse flow occupies a larger area and the shed vortices are shifted further away from aerofoil surface as angle of attack increases, confirming a severer flow separation at larger angles of attack. Generally, vortex starts to shed from more upstream positions with angle of attack, similar trend was also observed by Feng et al. (2015) over a NACA 0012 aerofoil at angles of attack of 6° and 10° .

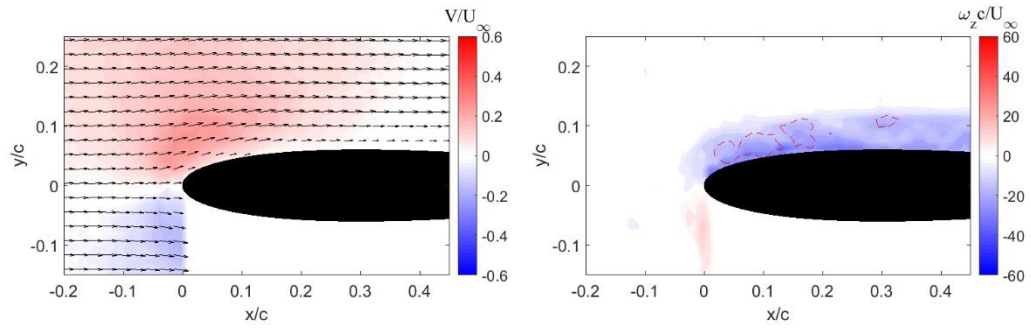
a) $t^+ = 0$



b) $t^+ = 0.09$



c) $t^+ = 0.18$



d) $t^+ = 0.27$

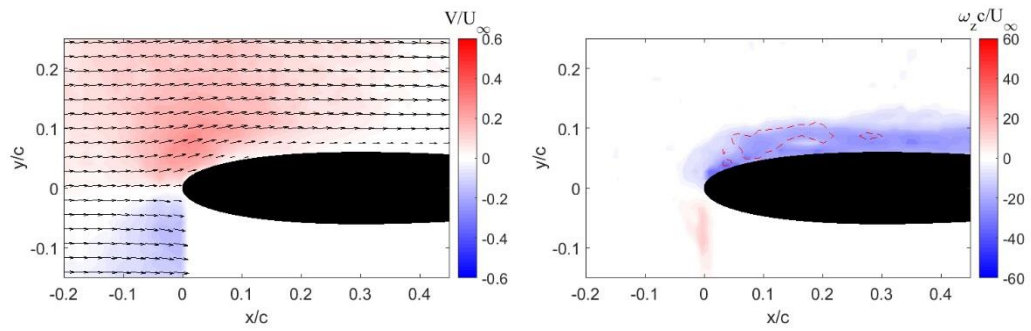
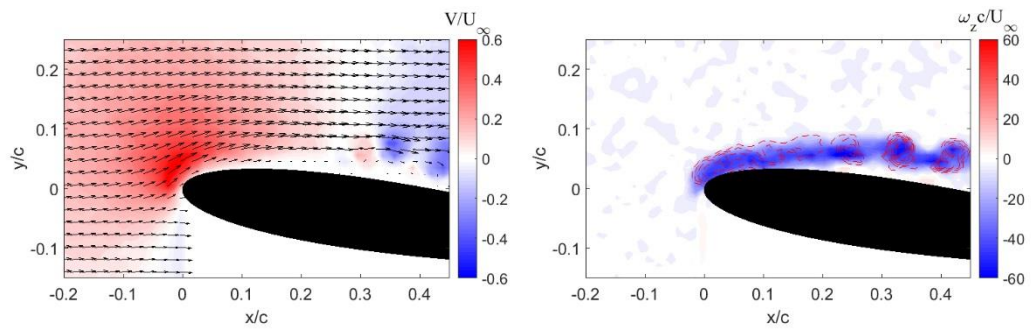
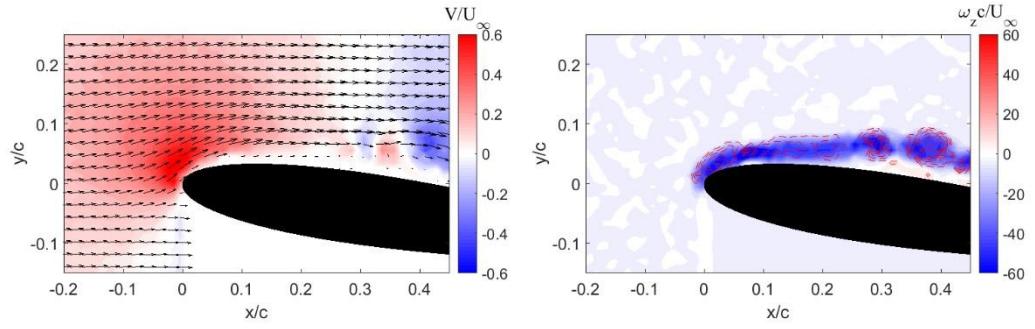


Figure 4-6 Instantaneous flow field at angle of attack of $\alpha = 0^\circ$. $t^+ = 0$ refers to the moment PIV measurement just starts. The red dashed lines indicate swirling strength isolines.

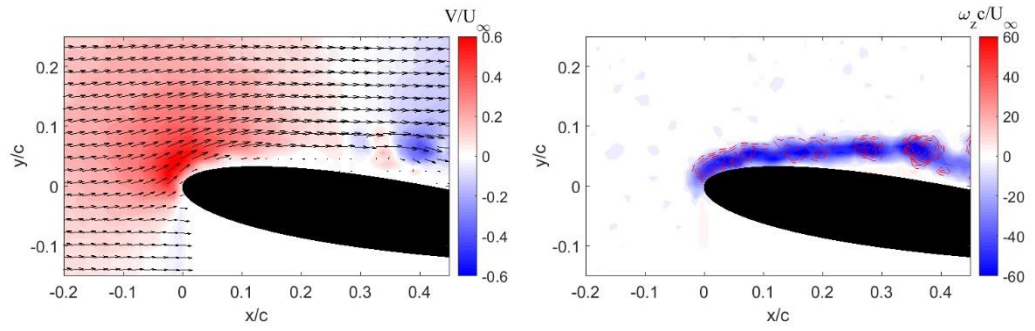
a) $t^+ = 0$



b) $t^+ = 0.09$



c) $t^+ = 0.18$



d) $t^+ = 0.27$

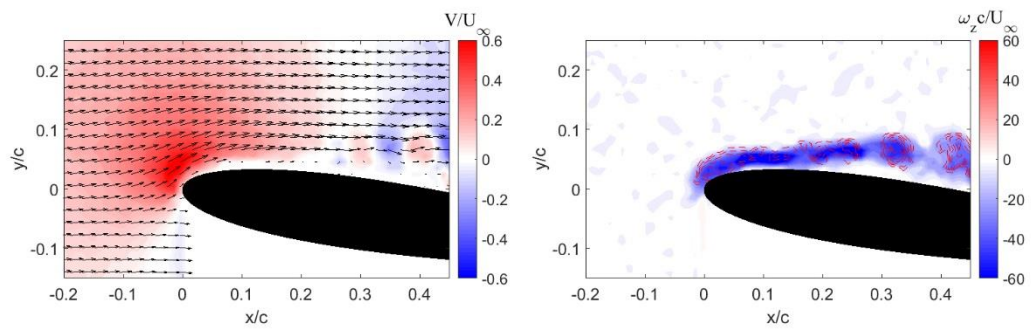
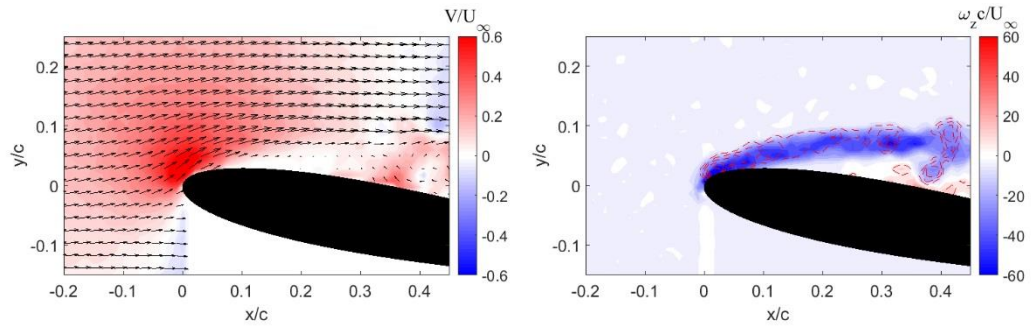
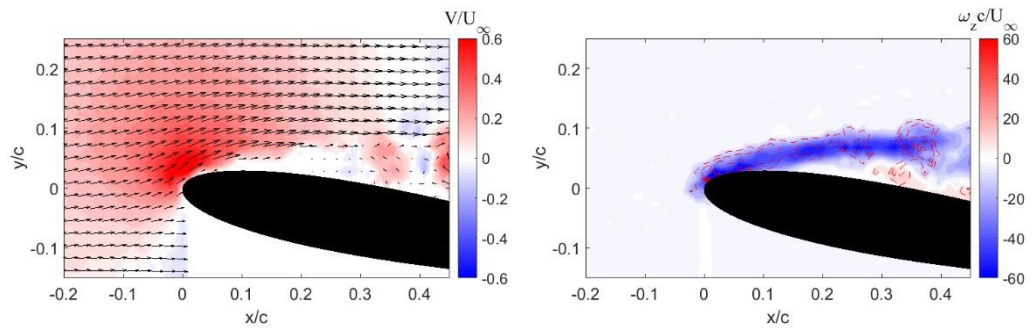


Figure 4-7 Instantaneous flow field at angle of attack of $\alpha = 8^\circ$.

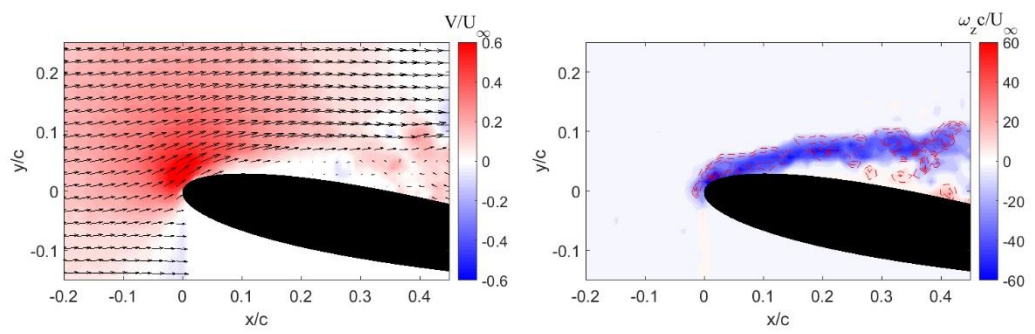
a) $t^+ = 0$



b) $t^+ = 0.09$



c) $t^+ = 0.18$



d) $t^+ = 0.27$

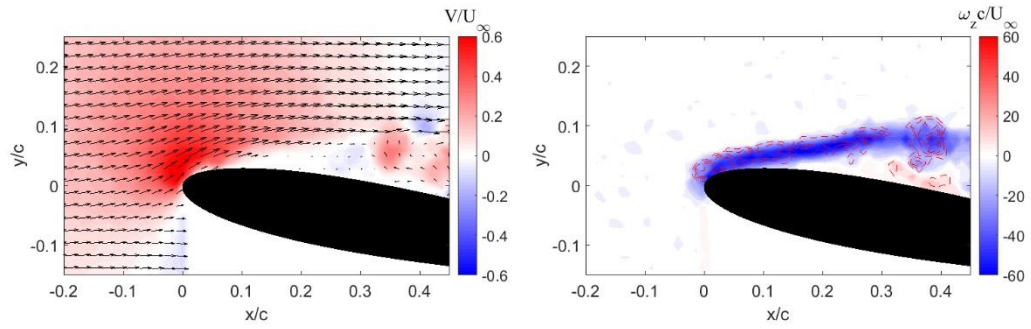
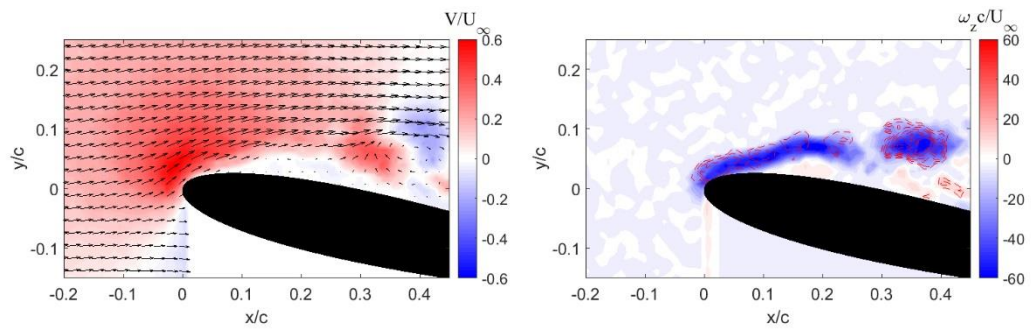
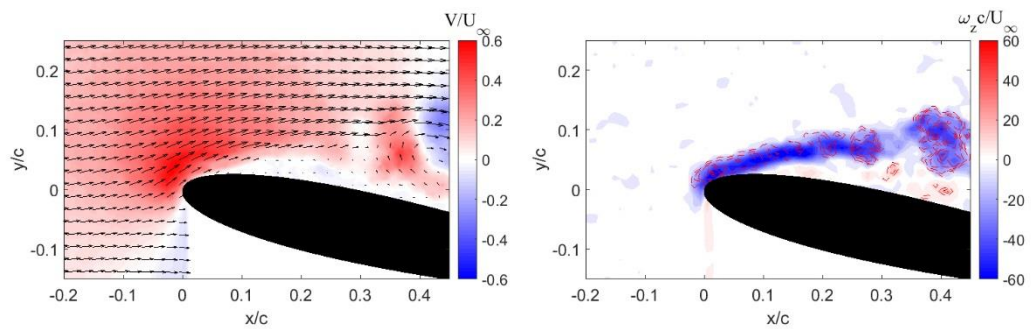


Figure 4-8 Instantaneous flow field at angle of attack of $\alpha = 10^\circ$.

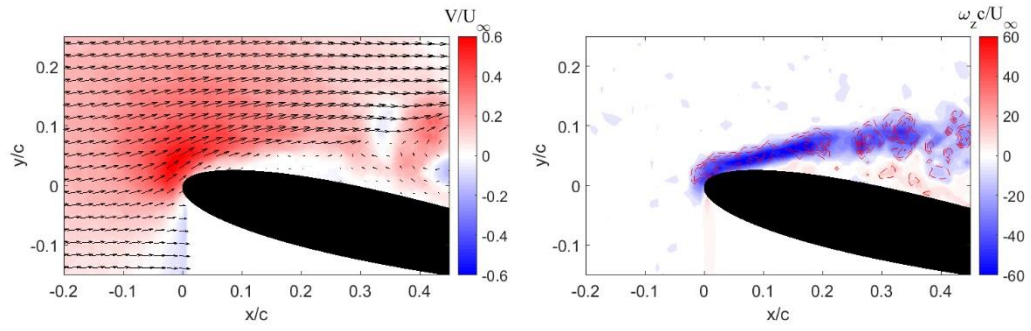
a) $t^+ = 0$



b) $t^+ = 0.09$



c) $t^+ = 0.18$



d) $t^+ = 0.27$

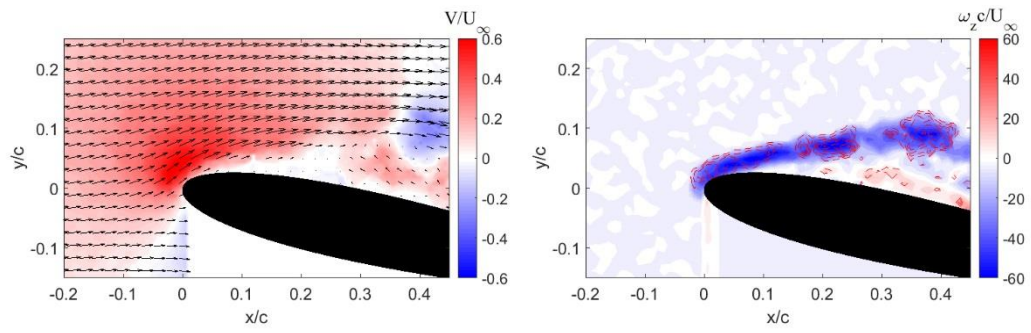
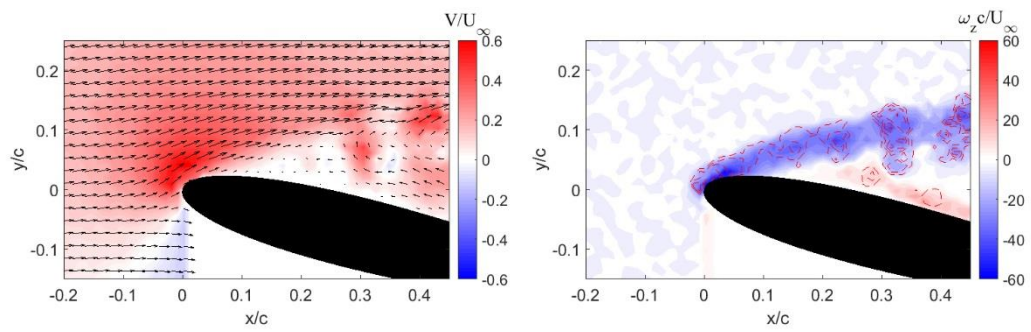
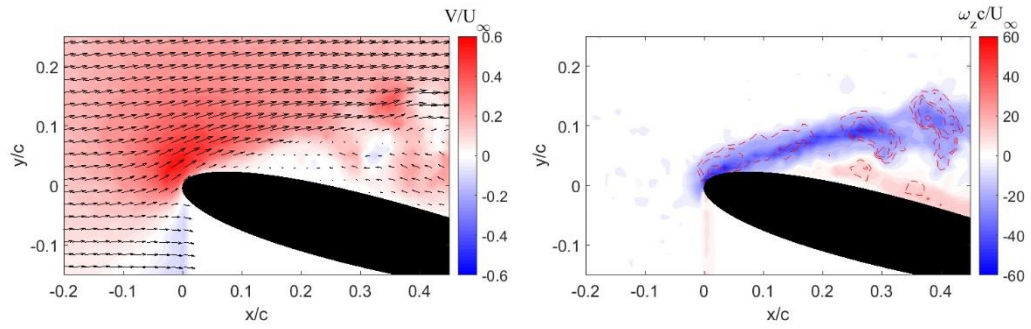


Figure 4-9 Instantaneous flow field at angle of attack of $\alpha = 12^\circ$.

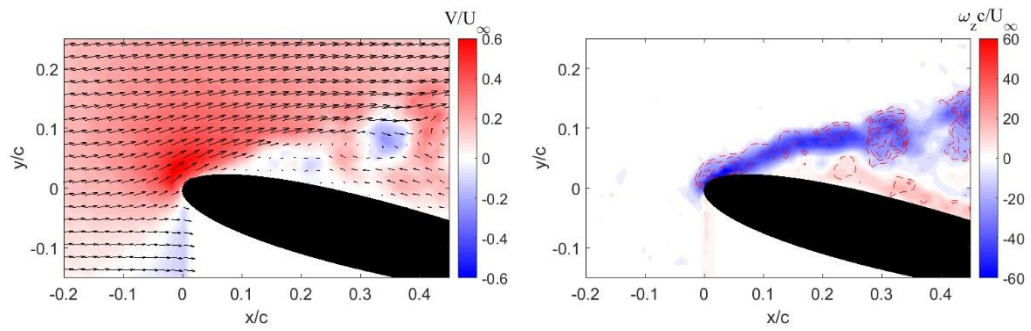
a) $t^+ = 0$



b) $t^+ = 0.09$



c) $t^+ = 0.18$



d) $t^+ = 0.27$

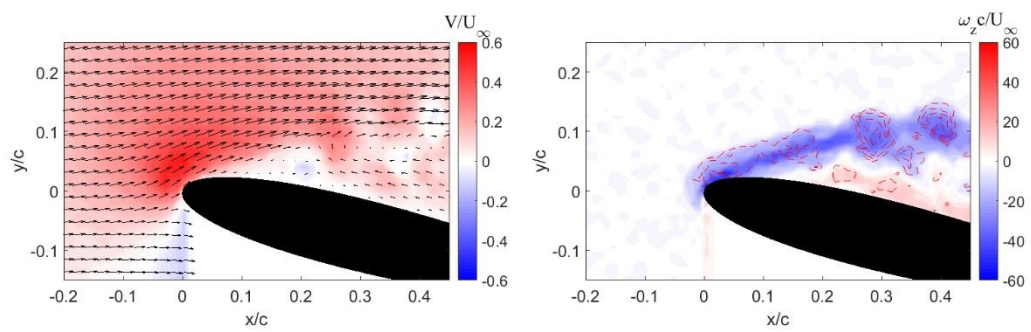
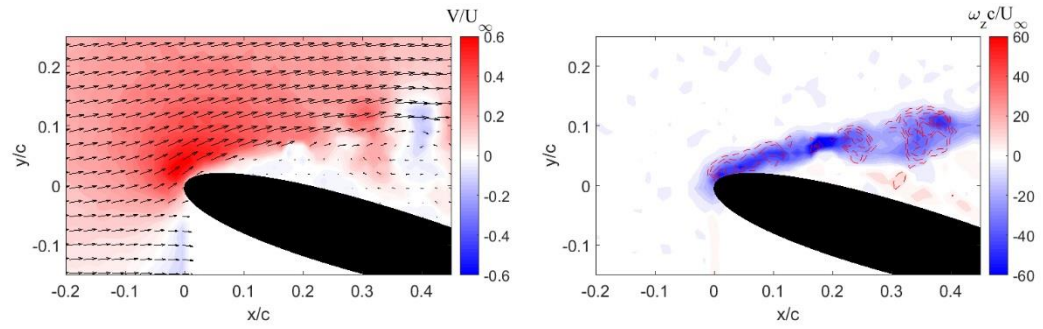
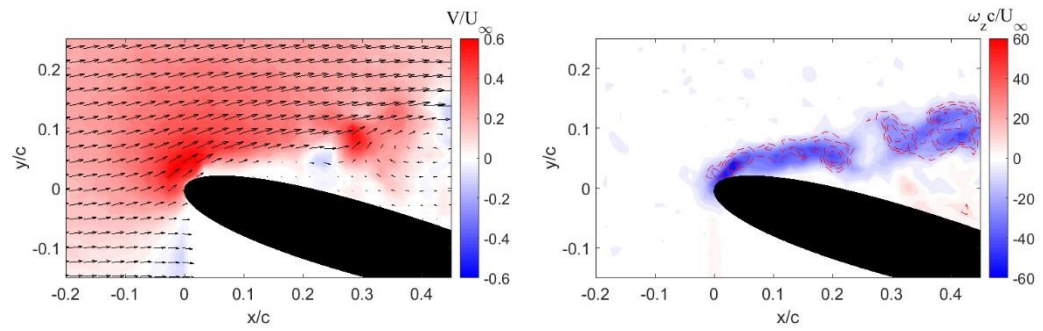


Figure 4-10 Instantaneous flow field at angle of attack of $\alpha = 14^\circ$.

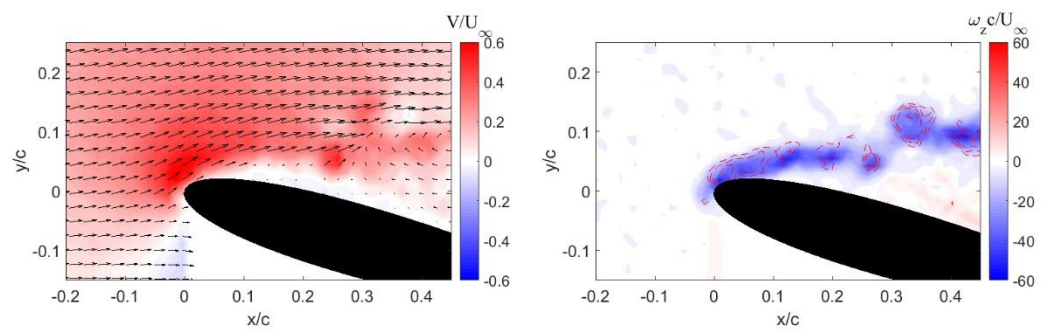
a) $t^+ = 0$



b) $t^+ = 0.09$



c) $t^+ = 0.18$



d) $t^+ = 0.27$

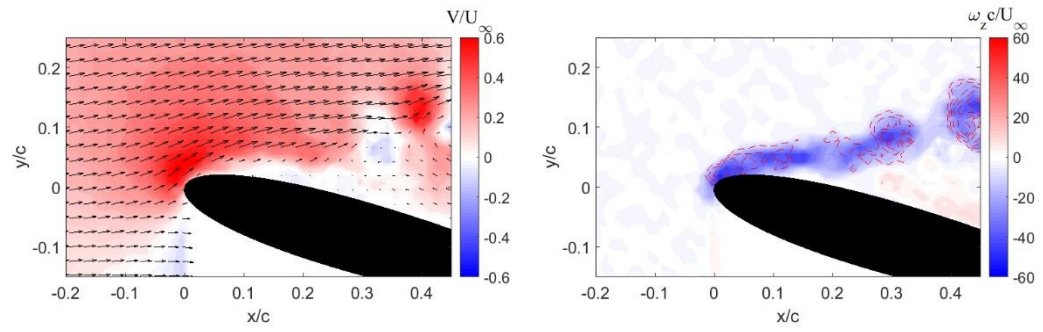


Figure 4-11 Instantaneous flow field at angle of attack of $\alpha = 16^\circ$.

5 Plasma Jet Measurements In Still Air

This chapter presents the experimental results of PIV measurements on the induced airflow by a DBD plasma actuator that consists of two exposed electrodes and one encapsulated electrode, separated by a Kapton dielectric sheet layer. The two exposed electrodes are located symmetrically on both sides of the encapsulated electrode (see Figure 3-3), therefore, a wall-normal jet could be produced when high AC voltages are applied to the plasma actuator electrodes. As the plasma wall-normal jet is similar to a synthetic jet, this type of DBD plasma actuator is also called linear plasma synthetic jet actuator (LPSJA). This wall-normal jet is of great significance to flow control and will be used to control the flow separation around a NACA 0012 aerofoil in Chapters 6 and 7. An understanding of the behaviour of this wall-normal jet mounted at the aerofoil leading-edge needs to be developed before conducting an investigation of flow separation control over an aerofoil with it. The wall-normal plasma jet characteristics, plasma jet vectoring capability, plasma jet strength modulation and pulsed modulation of the wall-normal plasma jet will be discussed in this chapter.

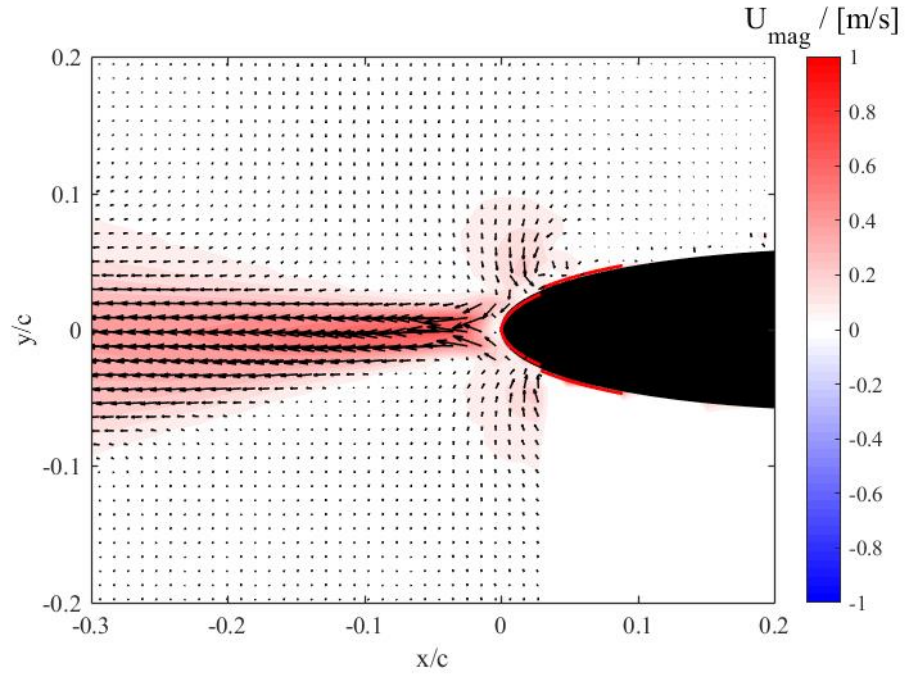
5.1 Characteristics of Wall-normal Plasma Jet

The investigation of wall-normal plasma jet characteristics was conducted via PIV measurements in quiescent air. This three-electrode DBD plasma actuator is mounted at the leading-edge of aerofoil, as shown in Figure 3-3. For simplicity, the electrodes are indicated by the three red lines in Figure 5-1. Two PSI-MCPG2503C power supplies were utilized to provide high sinusoidal voltages to exposed electrodes while the encapsulated electrode was connected to ground. Sinusoidal voltages of $E_{up\ p-p} = 5.5\text{ kV}$, $f_{up} = 34\text{kHz}$ and $E_{low\ p-p} = 5.5\text{ kV}$, $f_{low} = 34\text{kHz}$ were applied to the electrodes mounted on the upper and lower surface of aerofoil respectively for this test. PIV measurements were conducted continuously for 2.4 s which includes 0.2s without plasma, followed by 2s with plasma and another 0.2s without plasma.

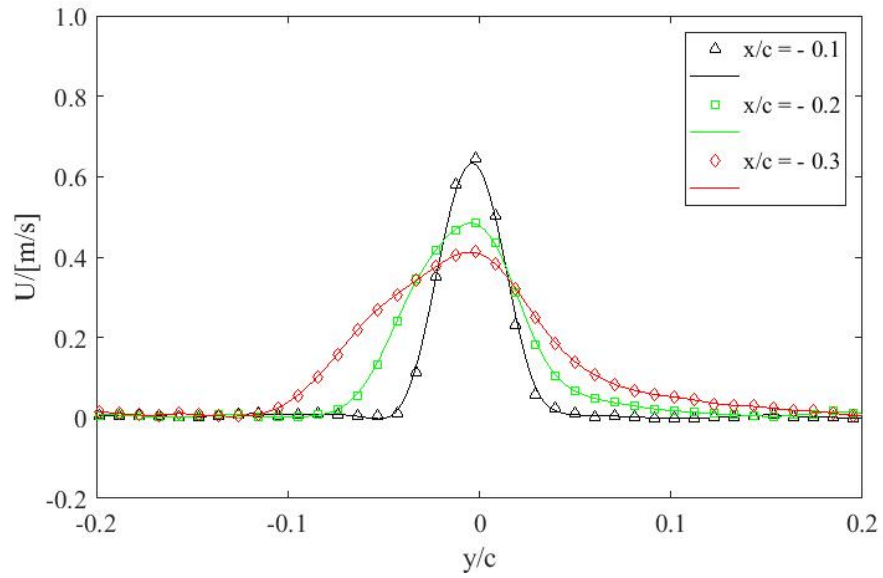
Velocity vectors of the time-averaged flow field with plasma are plotted in Figure 5-1. It can be clearly seen that a wall-normal plasma jet is produced with identical high sinusoidal voltages applied to the two exposed electrodes. This plasma jet goes in negative x direction and against freestream direction in flow separation control

studies in Chaps. 6 and 7. Both the velocity vectors and velocity magnitude contour are approximately symmetric about the jet centreline, indicating a symmetric plasma jet. This could be further confirmed by velocity profiles at varying positions in Figure 5-1 (b). This wall-normal jet is actually produced by the collision of the two wall jets induced by the three-electrode DBD plasma actuator. According to (N Benard et al., 2008; Nicolas Benard et al., 2008; Santhanakrishnan et al., 2009), the wall-normal jet is very sensitive to the differences between the two colliding wall jets and asymmetry was found in their studies. However, in this measurement, special care was paid when making the DBD plasma actuator and adjusting the high-voltage transformers, therefore, a jet that is more or less symmetric is obtained. Due to the curvature of aerofoil profile, the two wall jets don't meet at the aerofoil leading edge, instead, they meet and collide at $x/c \approx -0.035$, as shown in Figure 5-1 (a). Then a wall-normal jet forms and develops downstream. The vertical velocities around the aerofoil leading edge indicate an entrainment of the ambient fluids into the jet.

The centreline velocity development along the jet direction is plotted in Figure 5-2. As demonstrated in the velocity profiles, the maximum streamwise velocities with each cross-plane are located in the centreline of the jet, therefore, the centreline is determined by the linear fitting of the maximum streamwise velocity positions in this study. It can be clearly seen in Figure 5-2 that the centreline velocity increases to around 0.74 m/s at $x/c \approx -0.066$, after which it starts to decay, this trend is consistent with the findings of a wall-normal plasma jet produced on a flat plate (N Benard et al., 2008; Nicolas Benard et al., 2008; Santhanakrishnan et al., 2009, 2006). The half width of this plasma jet was calculated and presented in Figure 5-3. Wall-normal jet spreads linearly with distance from $x/c = -0.3$ to $x/c = -0.08$, which is referred to as fully developed jet zone. The spreading rate is $|d\delta_{1/2}/dx| \approx 0.31$, much larger than that of wall-normal plasma jet generated due to a collision of two wall plasma jets on a flat plate, which is around 0.1 according to Santhanakrishnan et al. (2009) and 0.13 found by N Benard et al. (2008). In present study, the two wall plasma jets are inclined when they collide rather than opposing to each other, which might account for a wider spreading wall-normal jet



(a)



(b)

Figure 5-1 Airflow induced by a symmetric three-electrode DBD plasma actuator attached onto a NACA 0012 aerofoil: (a) velocity magnitude contour superimposed by velocity vectors; (b) mean velocity profile. $E_{up\ p-p} = 5.5\text{ kV}$, $f_{up} = 34\text{ kHz}$ and $E_{low\ p-p} = 5.5\text{ kV}$, $f_{low} = 34\text{ kHz}$.

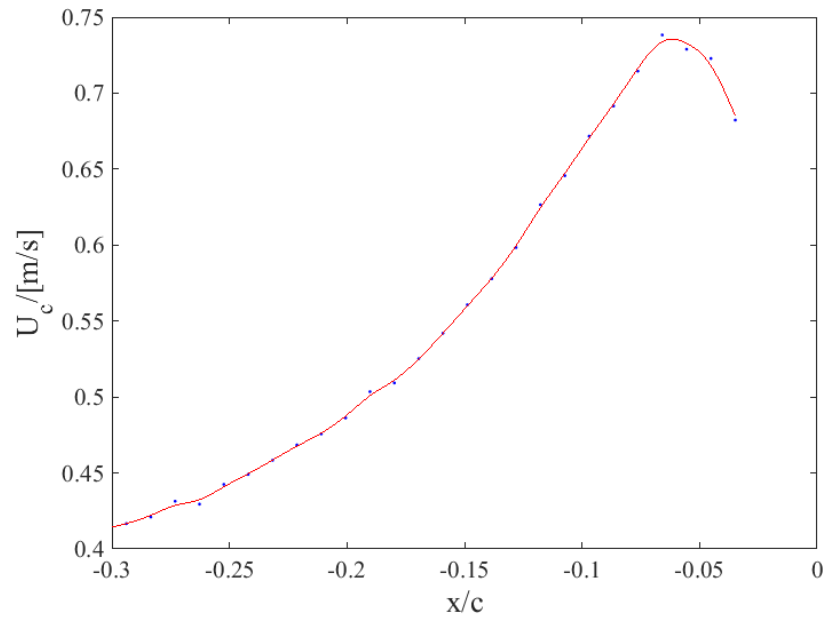


Figure 5-2 Centreline velocity of plasma jet. $E_{\text{up p-p}} = 5.5 \text{ kV}$, $f_{\text{up}} = 34 \text{ kHz}$ and $E_{\text{low p-p}} = 5.5 \text{ kV}$, $f_{\text{low}} = 34 \text{ kHz}$.

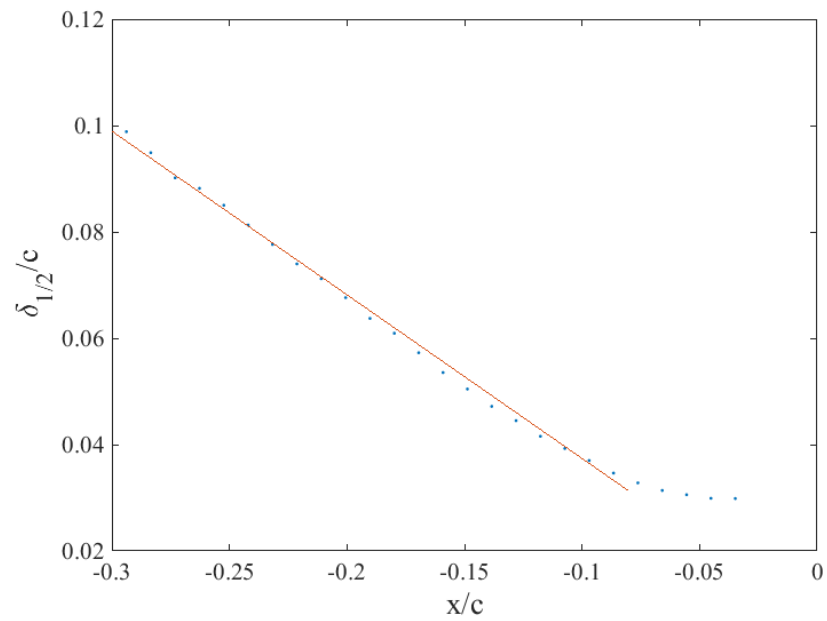


Figure 5-3 Half width variation in streamwise direction.

compared with that produced above a flat plate. However, from $x/c = -0.08$ to $x/c = -0.035$ where the two wall plasma jets meet, the half width seems not to increase linearly with distance, which was not reported in the studies by N Benard et al. (2008) and Santhanakrishnan et al. (2009). Probably this is due to that their plots start from a larger distance from plasma actuator and do not cover this region. This region is called jet establishment zone, where a jet spreads slowly compared with the fully developed jet zone, consistent with previous findings of a free jet (Fellouah and Pollard, 2013; Iqbal and Thomas, 2007; Quinn and Militzer, 1989).

Figure 5-4 shows the distribution of r.m.s. velocity normalized by the maximum jet velocity. Large velocity fluctuation was caused by this wall-normal plasma jet and a peak value of around 25% is found. This is much smaller than that by a wall-normal plasma jet above a flat plate, for example, N Benard et al. (2008) reported a value of 55% while Santhanakrishnan et al. (2006) found that the peak RMS could reach 85% of the maximum mean jet velocity. The reason accounting for this might be that the collision of two inclined jets is less severe than that of two opposing jets since the velocity gradient of two inclined jets is smaller. Accordingly, smaller velocity fluctuations are resulted in. In the meanwhile, electrode and dielectric materials, geometry of plasma actuator and AC voltages might also affect the test results.

To quantify the strength of the wall-normal plasma jet, momentum coefficient of plasma jet was calculated via $C_\mu = 2M/(\rho U_\infty^2 c)$, here, ρ is air density, U_∞ is freestream velocity in Chaps 4, 6 and 7, c is aerofoil chord length and M is plasma jet momentum flux which is obtained by integrating the jet velocity from PIV measurements, according to (Jukes and Choi, 2009), $M = \int \rho U^2 dy$, where U is velocity in jet direction. The dynamic variation of jet momentum coefficient at $x/c = -0.15$ was computed and plotted in Figure 5-5. A delay in jet momentum coefficient is observed, in other words, the momentum coefficient keeps almost constant at $C_\mu = 0$ and does not start to increase until the instant of $t \approx 0.05s$, this is because the airflow induced by plasma actuator does not reach this position and the air here still remains quiescent. Afterwards, an overshoot is observed and the momentum coefficient increases rapidly to 0.8% at $t \approx 0.094s$, which is in compliance with previous findings of maximum velocity of a wall plasma jet

(Whalley and Choi, 2013). Then momentum coefficient value reaches a steady state with a value of 0.25%, indicating a steady jet. When plasma is off, the momentum coefficient decays to zero within 0.12s.

The dynamics of airflow induced by DBD plasma actuator was also studied and shown in Figure 5-6. At $t = 0.012s$, two symmetric starting vortices appear in front of plasma actuator that is mounted on aerofoil leading edge, and the vortex cores locate around $x/c = -0.004$, as shown in Figure 5-6 (a). According to Whalley (2011), in quiescent air, the maximum vorticity location can be used as an indicator of vortex core although special care should be taken while applying this method since vorticity can't distinguish between vortices and shear layers. As time goes on, the starting vortices travel away and vorticity increases, shown in Figure 5-6 (b), (c) and (d). In the meanwhile, the vortex size increases as well. At $t = 0.05s$, the air at $x/c = -0.15$ starts to be affected by the starting vortices although the cores of starting vortices locate at around $x/c = -0.055$. This explains why the momentum coefficient keeps almost zero from $t = 0s$ to $t = 0.05s$ and starts to increase after $t = 0.05s$ in Figure 5-5. From Figure 5-6 (d) to (g), the vorticity decreases while the starting vortices grow in size and travel further away from plasma actuator, which is very similar to the findings of a single starting vortex by an asymmetric conventional DBD plasma actuator on a flat plate (Whalley and Choi, 2013). The vortex core is located at $x/c = 0.118$ at the moment of $t = 0.08s$ (Figure 5-6 (e)), and when t is $0.10s$, the vortex core travels to $x/c = -0.16$ (Figure 5-6 (f)), so the vortex core goes through $x/c = -0.15$ at the moment of $0.08s \leq t \leq 0.1s$ and closer to $t = 0.1s$, corresponding to the moment of $t = 0.094s$ when the peak jet momentum coefficient occurs in Figure 5-5. After the starting vortices pass through $x/c = -0.15$, the momentum coefficient at $x/c = -0.15$ starts to drop and reach a steady state, which might indicate that starting vortices contains more momentum than the steady jet. In Figure 5-6 (g), the starting vortices move upwards along y direction probably due to disturbances or buoyancy of the air heated by plasma. A steady jet is formed in Figure 5-6 (h), (i) and (j) after the starting vortices travel away and disappear in the field of view.

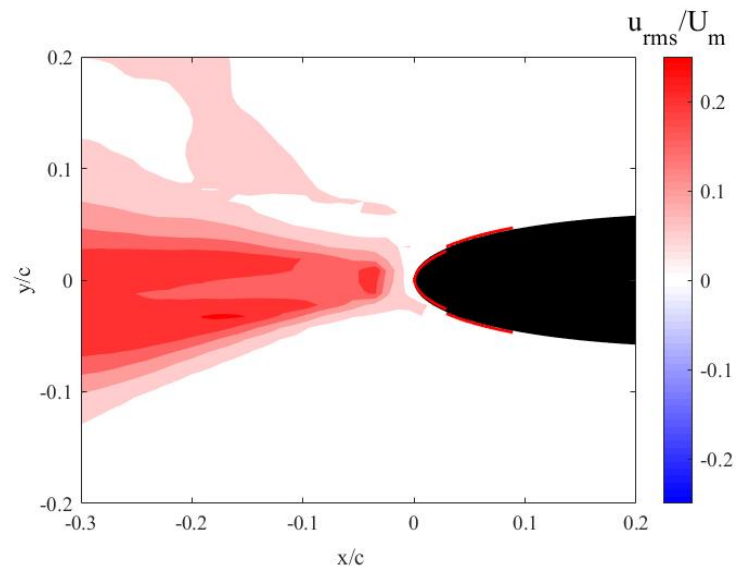


Figure 5-4 Normalized RMS velocity distribution.

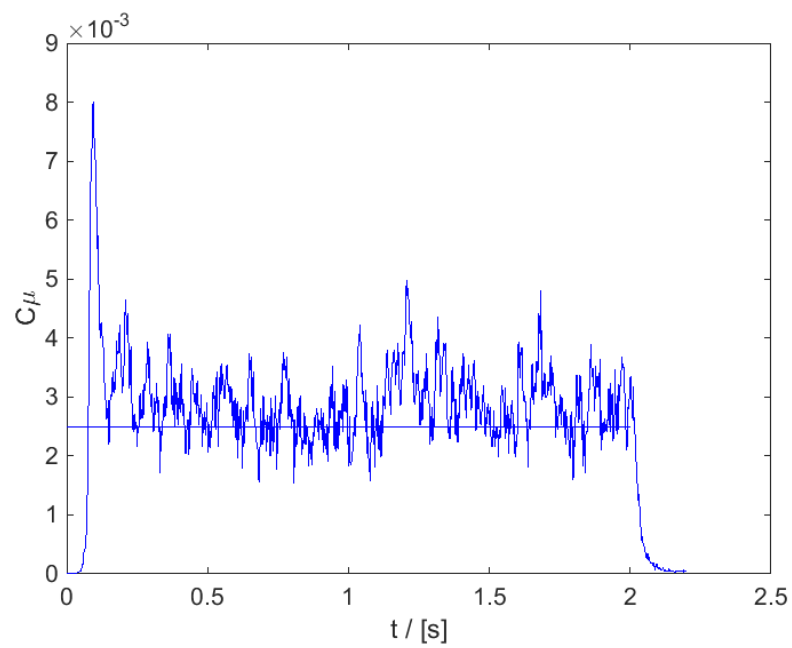
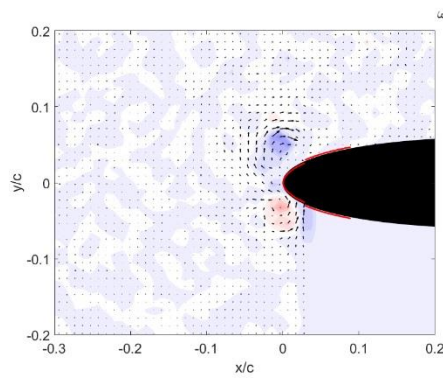
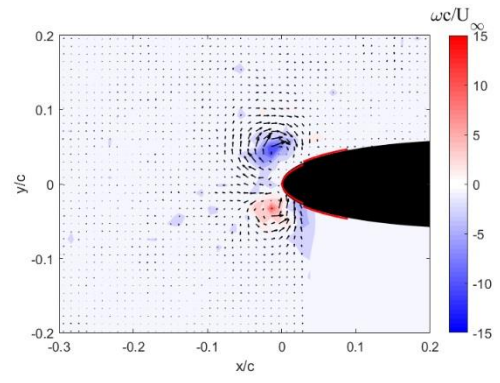


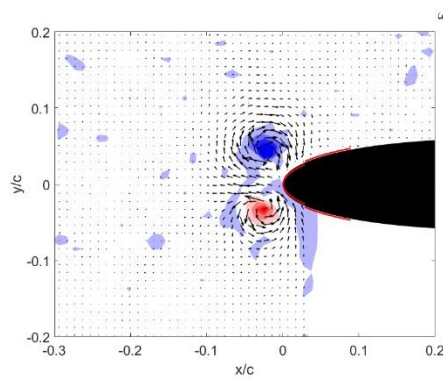
Figure 5-5 Instantaneous plasma jet momentum coefficient versus time. Jet momentum was calculated at $x/c = -0.15$.



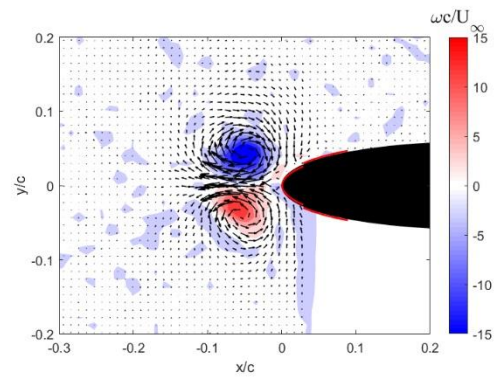
(a) $t = 0.012s$



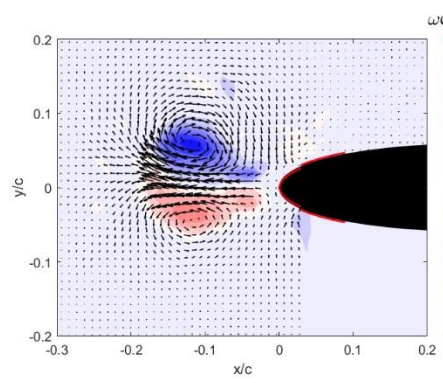
(b) $t = 0.02s$



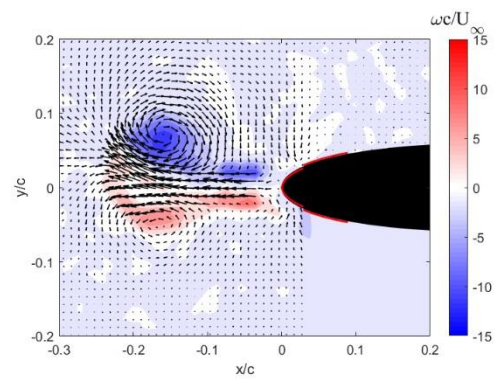
(c) $t = 0.03s$



(d) $t = 0.05s$



(e) $t = 0.08s$



(f) $t = 0.10s$

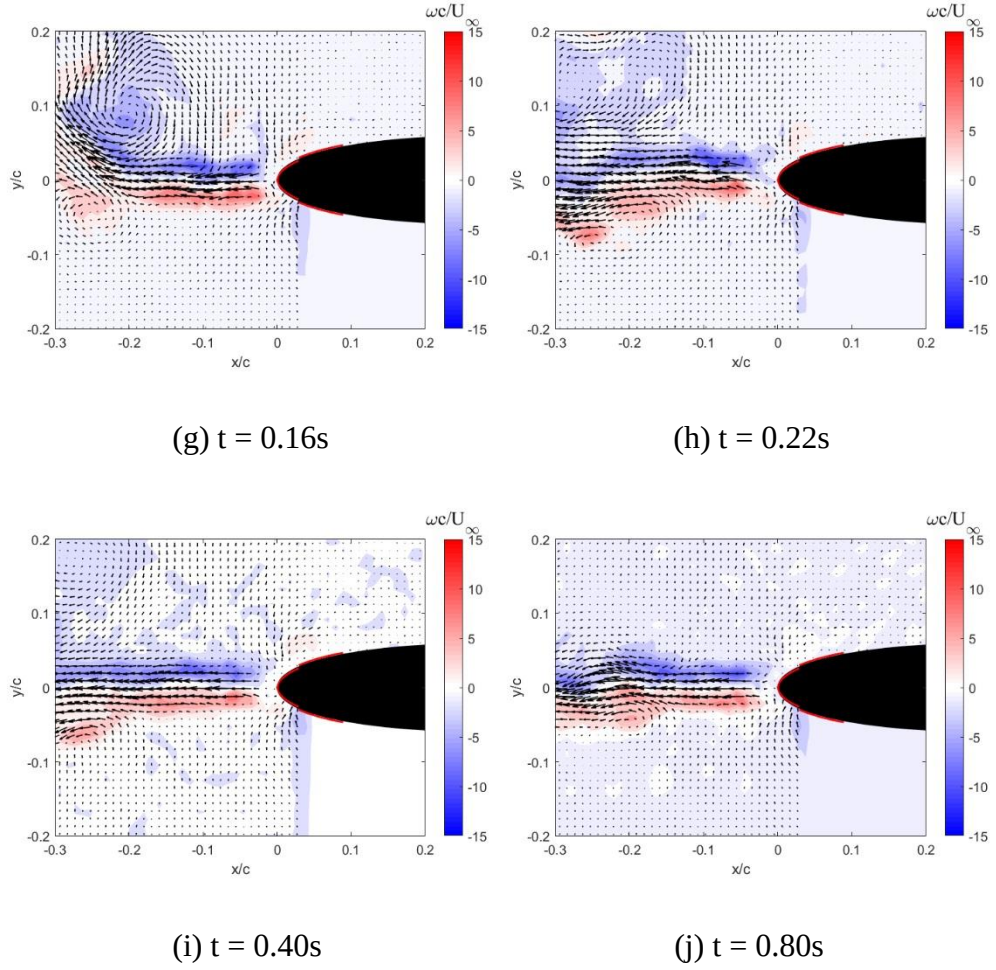


Figure 5-6 Instantaneous flow field induced by the three-electrode DBD plasma actuator. $t = 0$ refers to the moment plasma is about to be on.

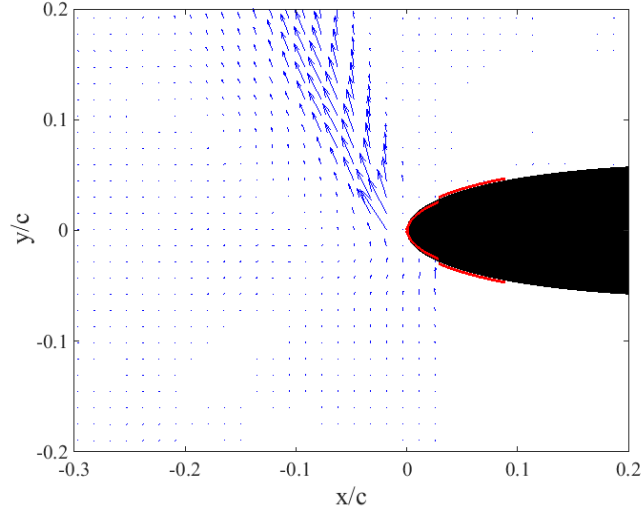
5.2 Vectoring Plasma Jets

The direction of plasma jet can be vectored via adjusting the voltage signals applied to the exposed electrodes, however, this study is mainly focused on leading-edge flow separation control with a wall-normal plasma jet, so only a wall-normal plasma jet was measured in the quiescent air and the characteristics of it is presented in 5.1. Investigations of plasma jet produced by conventional asymmetric single-exposed-electrode DBD plasma actuator were also conducted for comparison and the corresponding results will be included in this section. With the configuration of DBD plasma actuator in current study, the wall plasma jet by either of the exposed

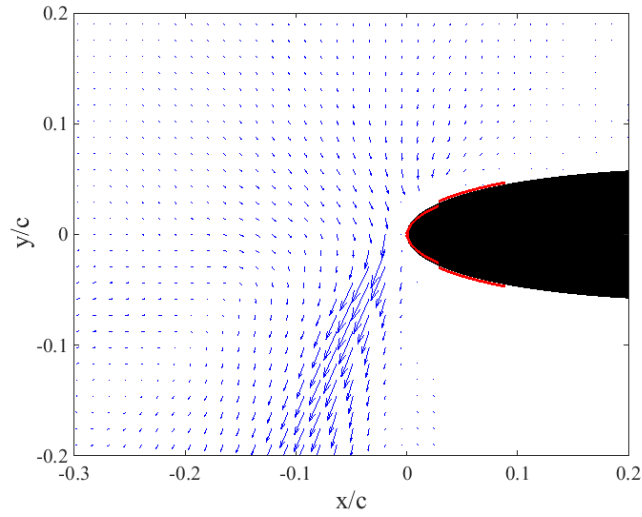
electrodes orient upstream. So only upstream-directed wall plasma jets are studied in this research.

The airflows induced by DBD plasma actuator with high voltage applied to lower-surface and upper-surface single exposed electrode are illustrated in Figure 5-7 (a) and (b) respectively. For the conventional asymmetric DBD plasma actuator mounted on a flat plate, shown in Figure 2-19 (a), plasma jet usually starts at the edge of the exposed electrode (Moreau et al., 2013; Thomas et al., 2009; Jukes et al., 2006), however, the virtual origin of the plasma jet locates slightly in front of aerofoil leading edge in this study although entrainment can be clearly seen at the edge between the exposed and encapsulated electrodes. In the meanwhile, the jet seems not tangential to the wall at the exposed electrode edge. As for the plasma jet on a flat plate, it's confined by the wall, so it has to go along the wall. But in this case, the aerofoil profile is quite curved, the entrained fluid probably will go along the curved aerofoil surface before reaching aerofoil leading edge, and then it is not confined within the region in front of aerofoil leading edge, it will develop like a free jet after the entrained fluid leaves the aerofoil surface. As can be expected, an upward jet is produced with high AC voltages applied to the exposed electrode on the lower aerofoil surface while a downward jet would be generated with exposed electrode on the upper aerofoil surface driven by high AC voltages, which can be seen in Figure 5-7 (a) and (b) respectively. Moreover, the angles between the plasma jet centreline and aerofoil chord are $\alpha_j = -67^\circ$ and $\alpha_j = 63^\circ$ for the jets in Figure 5-7 (a) and (b) respectively. Here, the negative sign represents an upward jet, conversely, the positive sign indicates the jet goes downwards.

Dynamics of plasma jet by single-exposed-electrode plasma actuator mounted on aerofoil lower and upper surface were investigated and shown in Figure 5-8 and Figure 5-9 respectively. The dynamics of these two cases are similar, and more or less symmetric about $x = 0$. Now the dynamics of the induced flow by single-exposed-electrode plasma actuator on the lower surface will be taken as an example to illustrate the development of a plasma jet by single-exposed-electrode DBD plasma actuator mounted on the aerofoil. When the single surface DBD plasma actuator is driven by a high AC voltage, a starting vortex is generated, as can be seen in Figure 5-8 (a). This starting vortex travels along its trajectory and grows in



(a)

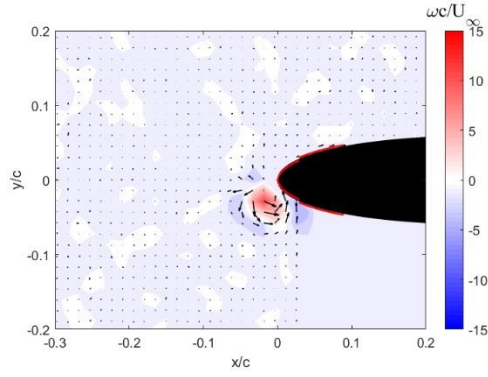


(b)

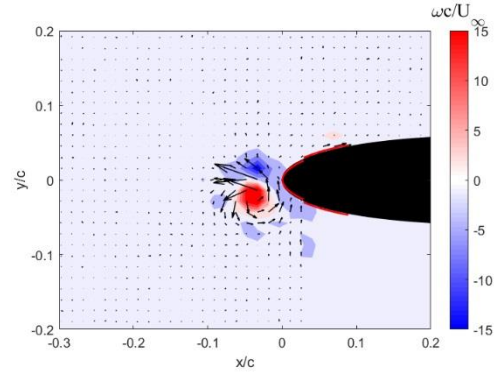
Figure 5-7 Airflow induced by DBD plasma actuator attached onto a NACA 0012 aerofoil: (a) $E_{up\ p-p} = 0\text{ kV}$, $f_{up} = 0\text{ kHz}$ and $E_{low\ p-p} = 6.3\text{ kV}$, $f_{low} = 34\text{ kHz}$; (b) $E_{up\ p-p} = 6.3\text{ kV}$, $f_{up} = 34\text{ kHz}$ and $E_{low\ p-p} = 0\text{ kV}$, $f_{low} = 0\text{ kHz}$.

size, this is quite similar to the development process of starting vortex pairs generated in the initial stage of a wall-normal plasma jet and that by a single surface DBD plasma actuator over a flat plate (Whalley and Choi, 2013). The vortex core can be determined by the primary vorticity region in each image (Whalley, 2011), and secondary vorticity can also be seen around the starting vortex. In the very beginning of the starting vortex development, both primary and secondary vorticity

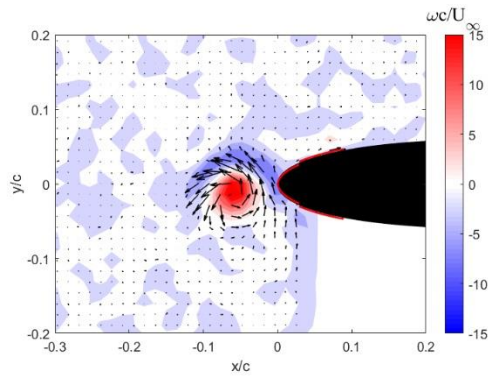
increases with the size of starting vortex because it is fed by the plasma momentum, shown in Figure 5-8 (a) and (b). After that, vorticity decreases as starting vortex size increases since the vortex can not be nourished by plasma momentum when it travels away, which is demonstrated in Figure 5-8 (c), (d), (e), (f) and (g). However, it was found in (Whalley, 2011) that vorticity decreases with starting vortex size, probably this is because the comparison was made at $t = 0.12s$ and afterwards, in other words, the stage when vorticity increases with starting vortex size was missed out in their study as vorticity increases at $t = 0.02s$ and starts to decrease with vortex size at $t = 0.03s$ in this study. As time goes on, the starting vortex goes away from the field of view and a steady jet is eventually formed in Figure 5-8 (i) and (j).



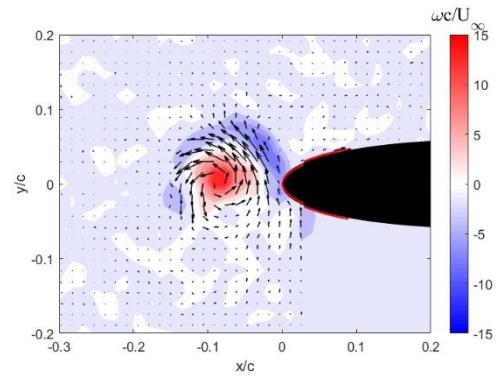
(a) $t = 0.012s$



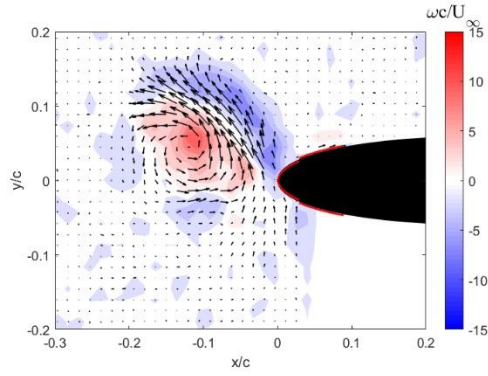
(b) $t = 0.02s$



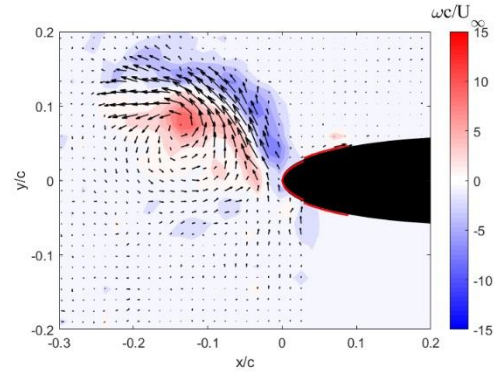
(c) $t = 0.03s$



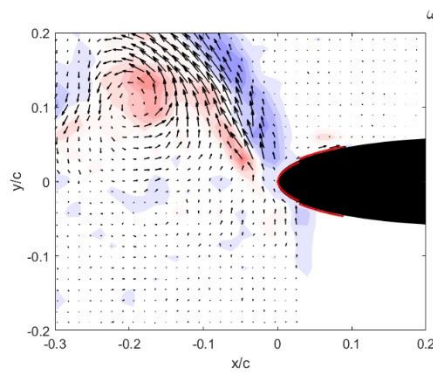
(d) $t = 0.05s$



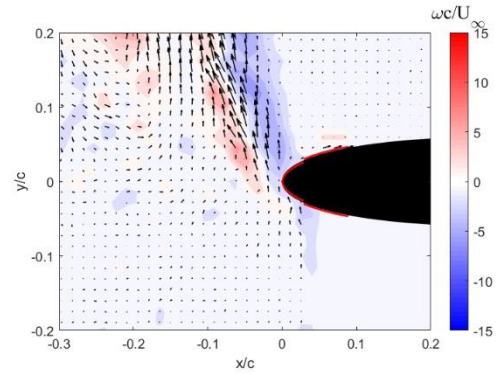
(e) $t = 0.08s$



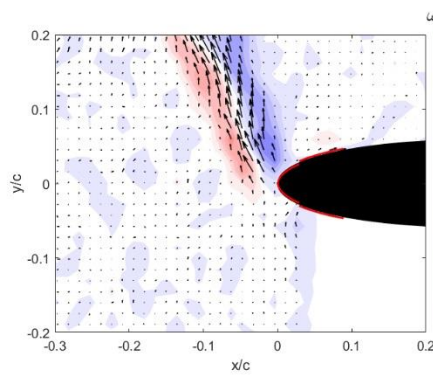
(f) $t = 0.10s$



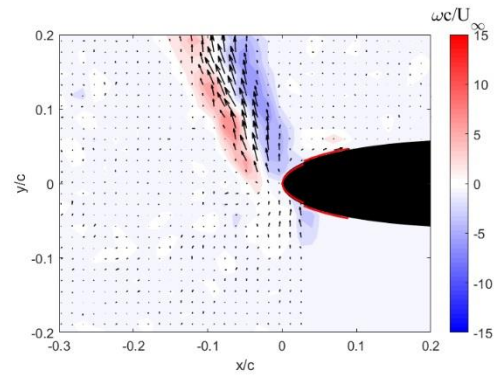
(g) $t = 0.16s$



(h) $t = 0.22s$

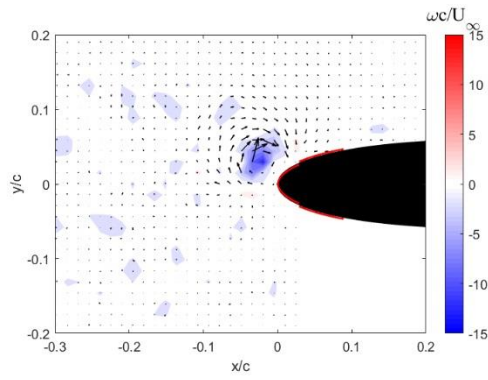


(i) $t = 0.40s$

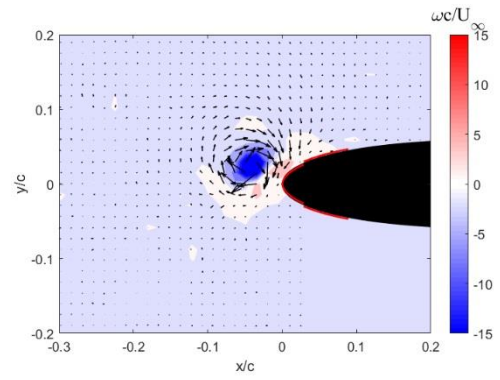


(j) $t = 0.80s$

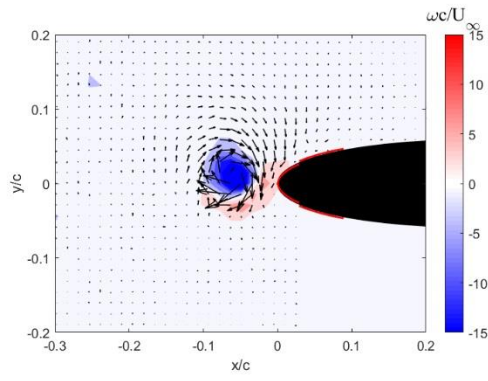
Figure 5-8 Instantaneous flow field induced by the three-electrode DBD plasma actuator. $t = 0$ refers to the moment plasma is about to be on. Sinusoidal voltage of $E_{low\ p-p} = 6.3\text{ kV}$, $f_{low} = 34\text{ kHz}$ is applied to the exposed electrode on the lower surface and encapsulated electrode.



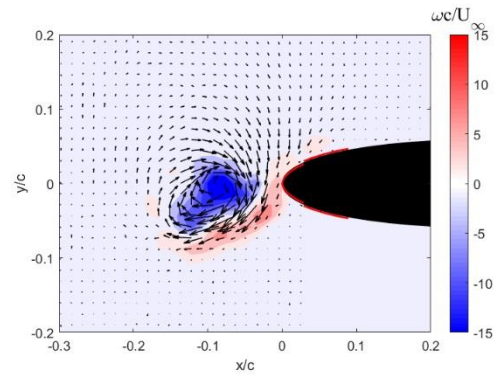
(a) $t = 0.012s$



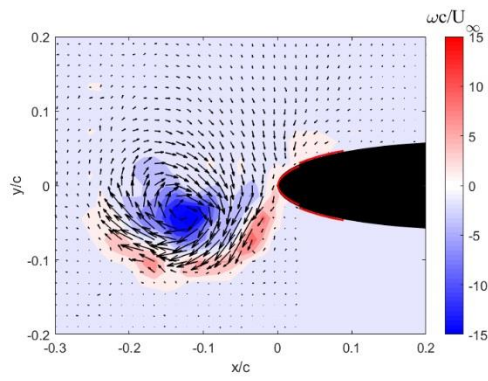
(b) $t = 0.02s$



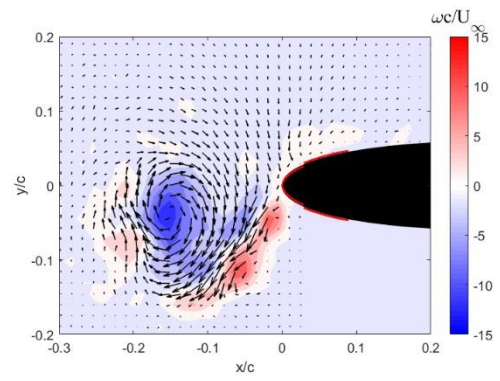
(c) $t = 0.03s$



(d) $t = 0.05s$



(e) $t = 0.08s$



(f) $t = 0.10s$

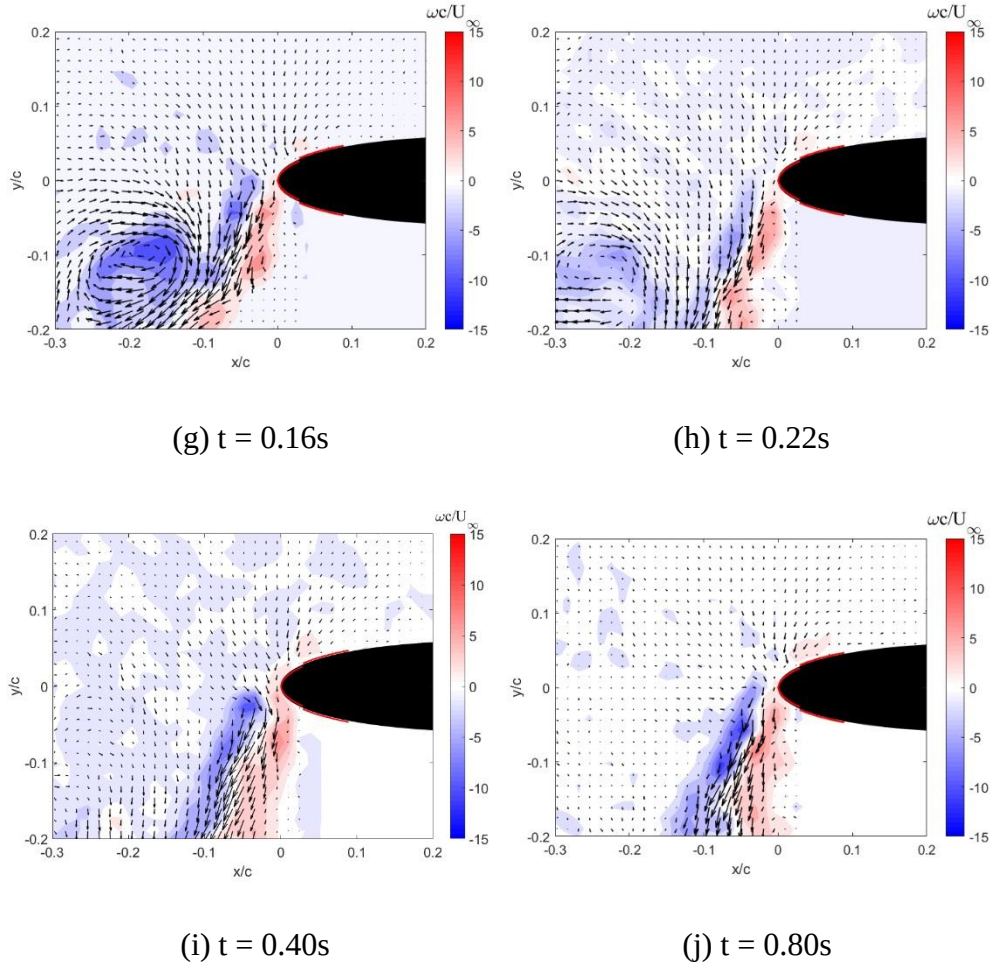


Figure 5-9 Instantaneous flow field induced by the three-electrode DBD plasma actuator. $t = 0$ refers to the moment plasma is about to be on. Sinusoidal voltage of $E_{up\ p-p} = 6.3$ kV, $f_{up} = 34$ kHz is applied to the exposed electrode on the upper surface and encapsulated electrode.

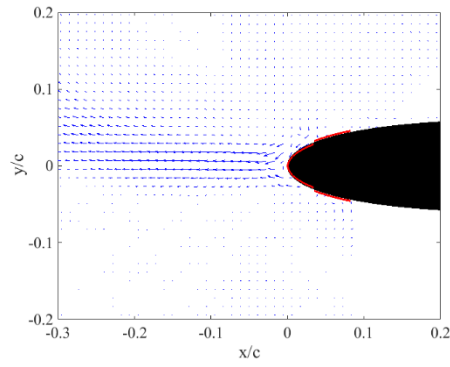
5.3 Plasma Jet Strength Modulation

The strength of plasma jet could be manipulated by adjusting the high AC voltage applied to exposed electrodes. Basically, both amplitude and frequency could affect the plasma jet strength, however, in this study, only voltage amplitude was used to adjust plasma jet strength while the frequency of carrier voltage is fixed at 34 kHz. Table 2 presents a few cases tested in current measurements and the corresponding plasma force coefficients. It should be also mentioned that the voltages applied to both the exposed electrodes on the upper and lower surfaces are identical, in other words, sinusoidal voltages of the identical amplitude and frequency are applied to

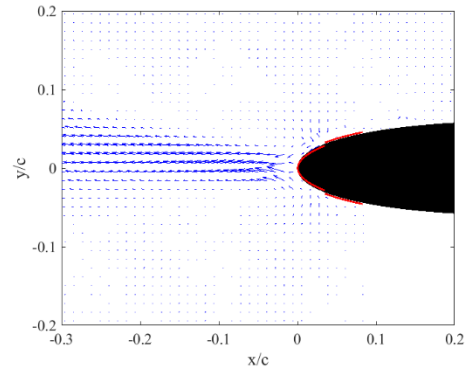
both exposed electrodes for each case. As can be seen in the table, plasma force coefficient increases with the amplitude of voltage within the tested range.

Table 2 Plasma momentum coefficients with plasma actuator driven by different voltages.

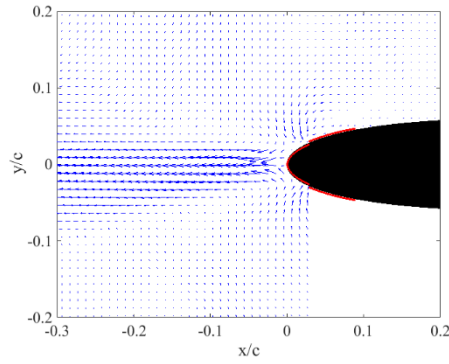
$E_{p-p}/(\text{kV})$	3.36	5.00	5.25	5.50	5.75	6.00
$C_{\mu}/(\%)$	0.076	0.14	0.20	0.25	0.38	0.48



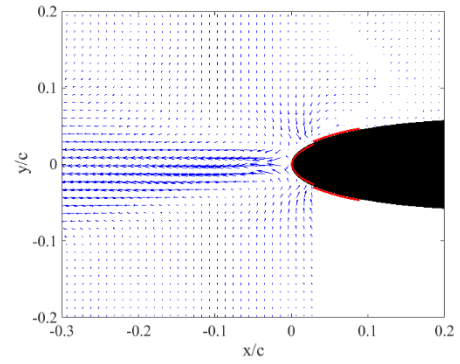
(a)



(b)



(c)



(d)

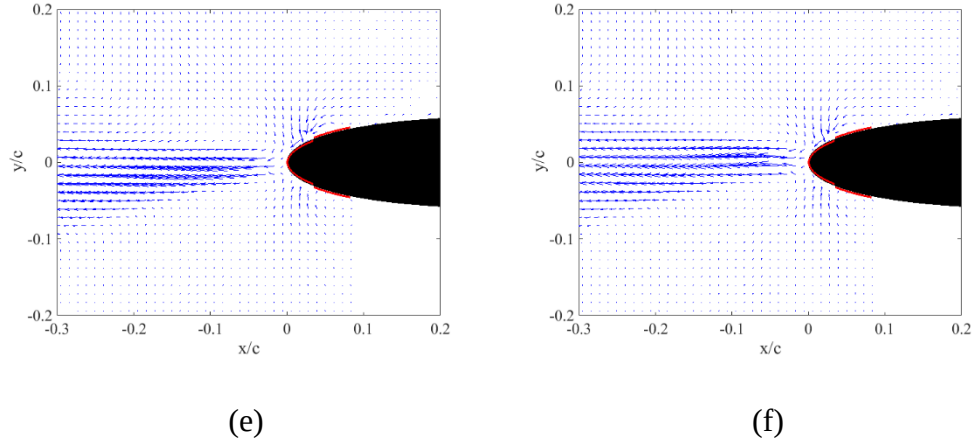


Figure 5-10 Time-averaged flow field induced by DBD plasma actuator with voltages applied to both exposed electrodes at carrier frequency of 34 kHz and amplitude of : (a) 3.36 kV; (b) 5.00 kV; (c) 5.25 kV; (d) 5.5 kV; (e) 5.75 kV; (f) 6.00 kV.

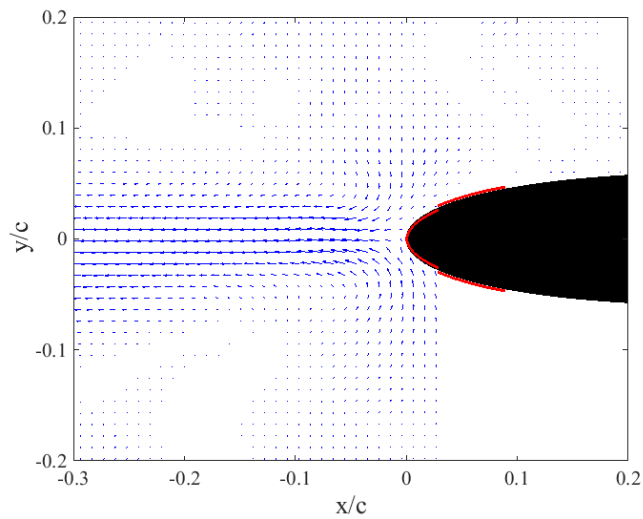
Figure 5-10 presents the time-averaged velocity fields induced by DBD plasma actuator with different driving voltage amplitudes. It seems that plasma jets induced by small voltages are more sensitive to disturbances such as the difference between the two exposed electrodes (geometry, size, material, voltage and so on) and buoyancy, which is demonstrated in Figure 5-10 (a) and (b) that the induced flow drifts upwards after it leaves plasma actuator, in other words, asymmetry of jet appears, similar phenomenon was also observed by N Benard et al. (2008) and Santhanakrishnan et al. (2009). As for the induced flow by a higher voltage, it's less susceptible to the disturbances or buoyancy and goes more or less normal to the wall, this is because it's more difficult for disturbances to affect a stronger jet.

5.4 Pulsed Modulation

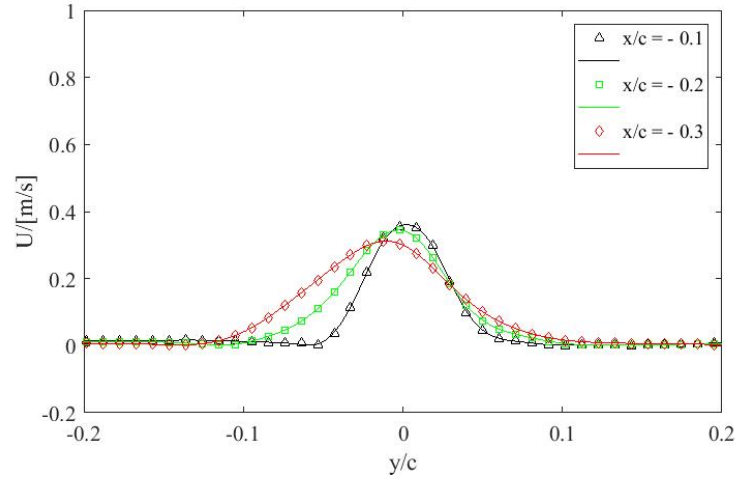
Pulsed modulation of a wall-normal plasma jet was also investigated in the initially quiescent air to study the effect of pulsed modulation on the characteristics of the airflow by a linear plasma synthetic jet actuator. A BNC 575 pulse generator was used to generate a series of pulses which remotely control the power supplies that provide high AC voltages to plasma actuators so that plasma actuators could be switched on and off alternately. PIV measurements of the flow caused by periodic actuation of plasma actuator with a modulation pulse frequency of 29Hz and duty

cycle of 25% were conducted to investigate the characteristics of the induced flow with pulsed modulation of carrier voltage signal that applied to plasma actuator.

Time-averaged flow field by unsteady actuation of DBD plasma actuator in initially still air was calculated and presented in Figure 5-11. As can be clearly seen from the velocity vectors in Figure 5-11 (a), a wall-normal jet is formed in front of aerofoil leading edge in a time-averaged sense. However, it's much weaker compared with the jet (shown in Figure 5-1 (a)) by continuous actuation of plasma actuator, this could be confirmed by the velocity profiles in Figure 5-1 (b) and Figure 5-11 (b). This jet by pulsed activation of plasma is more or less symmetric although there is some small drift in the velocity profiles at $x/c = 0.3$. Jukes et al. (2006) investigated the airflow by on and off alternately actuating a single asymmetric DBD plasma actuator attached on 2D surfaces, and they found that the time-averaged induced wall flows seems to be a laminar wall jet. All these findings seem to indicate that the flow by pulsed modulation of a jet, no matter whether it's a wall jet or a wall-normal jet, is still a jet in a time-averaged sense.



(a)



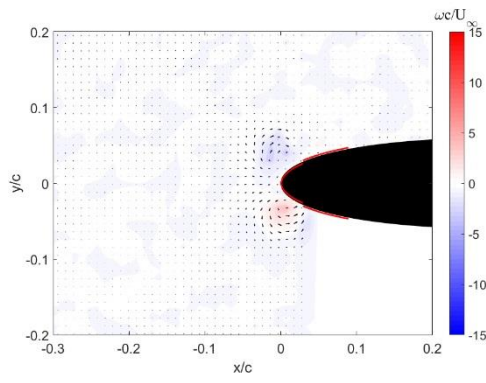
(b)

Figure 5-11 Airflow induced by unsteady actuation of a symmetric three-electrode DBD plasma actuator attached onto a NACA 0012 aerofoil: (a) velocity vectors; (b) mean velocity profile. The carrier voltage signal is $E_{up\ p-p} = 5.5\text{ kV}$, $f_{up} = 34\text{ kHz}$ and $E_{low\ p-p} = 5.5\text{ kV}$, $f_{low} = 34\text{ kHz}$. Modulation pulse frequency is 29Hz and duty cycle is 25%.

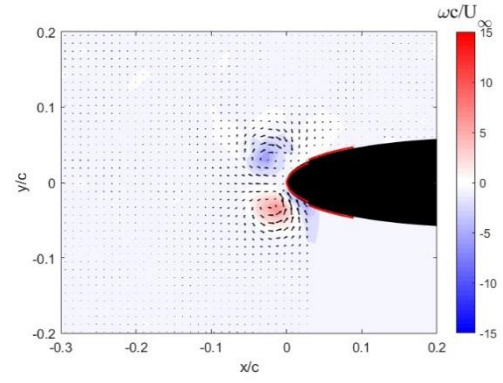
In our study, PIV measurements were also conducted to investigate the dynamics of the induced flow by pulsed actuation of a DBD plasma actuator with two opposing exposed electrodes sharing the same encapsulated electrode. The dual exposed electrodes were actuated on and off synchronously by identical sinusoidal voltages in this study. Instantaneous flow behaviours by pulsed activation of the linear plasma synthetic jet actuator are presented in Figure 5-12. After plasma is activated, a vortex pair of opposite vorticity and symmetric about wall-normal direction appears in front of aerofoil leading edge where the DBD plasma actuator is mounted, which can be clearly seen in Figure 5-12 (a). Then this vortex pair travels away from plasma actuator and grows in size. As for the vorticity, it initially increases from Figure 5-12 (a) to (e) while it starts to reduce at later time instants. The behaviour of this vortex pair is very similar to the starting vortex pair with steady actuation of plasma actuator in Section 5.1 although the vorticity and size are much smaller. For the case of steady actuation of DBD plasma actuator, momentum is continuously injected into the ambient air and more fluids are entrained into the induced flow than that by pulsed actuation of plasma actuator, creating a larger and stronger starting vortex pair. The second vortex pair could be

clearly seen in Figure 5-12 (f) and the third vortex pair appear in Figure 5-12 (g). As the exposed electrodes are activated on and off alternately, successive vortex pairs will be generated, as demonstrated in Figure 5-12 (h), (i) and (j). The coherence of these vortices decreases when they travel further away from plasma actuator, and a jet with self-similar characteristics is developed after $t = 0.22\text{s}$. Jukes et al. (2006) also finds a jet with self-similar characteristics is formed after the initiation stage through investigating the flow induced by pulsed actuation of a symmetric plasma actuator on a flat plate. It should be noted that the size and vorticity of the first vortex pair are larger than successive vortex pairs produced at following instants, which is a very interesting phenomenon. After ambient flow starts moving, velocity gradient between induced flow and ambient fluid is smaller compared with that by the first pulsed actuation, so weaker vortex pairs are generated by the successive pulsed actuation of plasma actuator. This study also confirms that a train of vortices are produced by unsteady actuation of DBD plasma actuator.

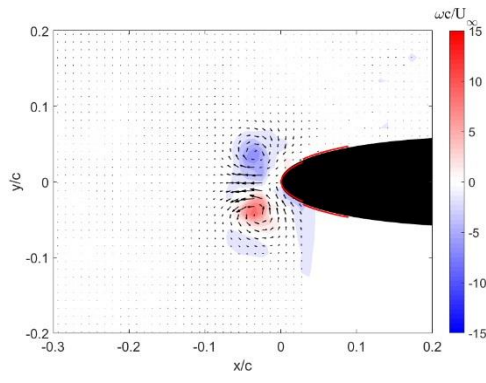
Power spectral analysis was also conducted to investigate the coherent structures within the wall-normal plasma jet by unsteady actuation of DBD plasma actuator. Figure 5-13 represents the power density spectra of streamwise velocity at varying x/c positions with y/c located at jet centreline. For clarity, each successive spectrum is shifted by a magnitude of 100 relative to that at its upstream location. A dominant peak at $f = 29\text{Hz}$ appears in all the spectra listed in the figure, the fundamental frequency corresponds to the pulsed modulation frequency, which indicates that coherent structure dominates the flow and a vortex pair is generated by each pulsed actuation of plasma actuator. The fundamental peak value increases from $x/c = -0.05$ to $x/c = -0.10$ while it reduces as it travels away from $x/c = -0.15$, this is in good consistence with the phenomenon in Figure 5-12. The vortex pair grows while it travels from $x/c \approx -0.05$ to $x/c \approx -0.12$, contributing to an increase in the peak at the dominant frequency. As the vortex pair travels further away from plasma actuator, the vortices start to dissipate, which is demonstrated in Figure 5-12 (h), (i) and (j). This leads to a lower power spectral density peak at $x/c = 0.2$ and 0.25 . Super-harmonics of the fundamental frequency appear in the power spectra at $x/c = -0.05$ and $x/c = -0.1$, suggesting a nonlinear interaction between the vortex pairs in the near field of plasma actuator.



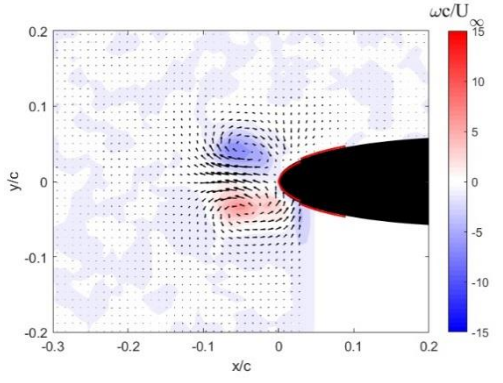
(a) $t = 0.012s$



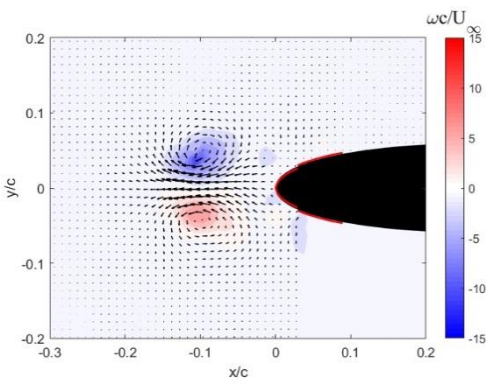
(b) $t = 0.02s$



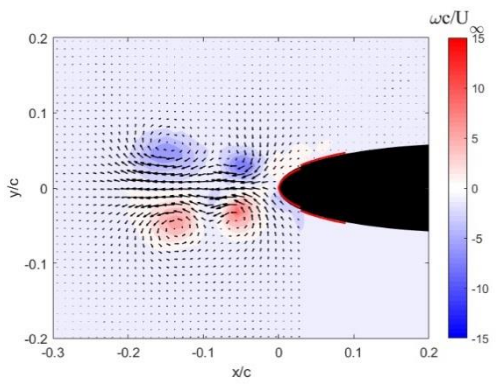
(c) $t = 0.03s$



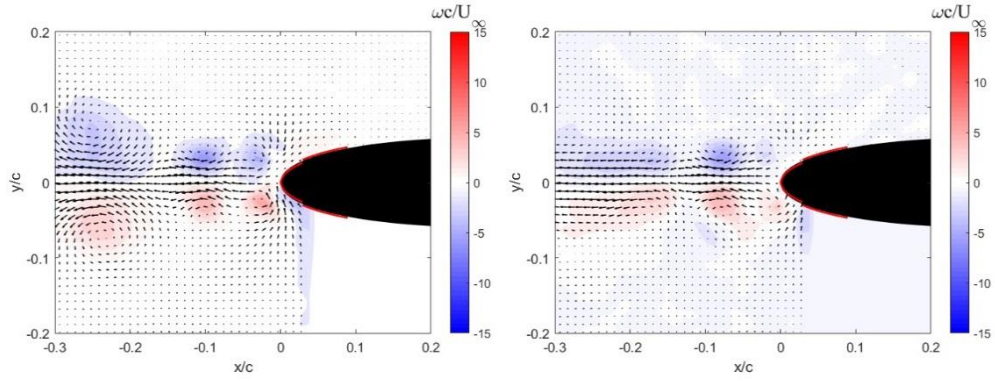
(d) $t = 0.05s$



(e) $t = 0.08s$

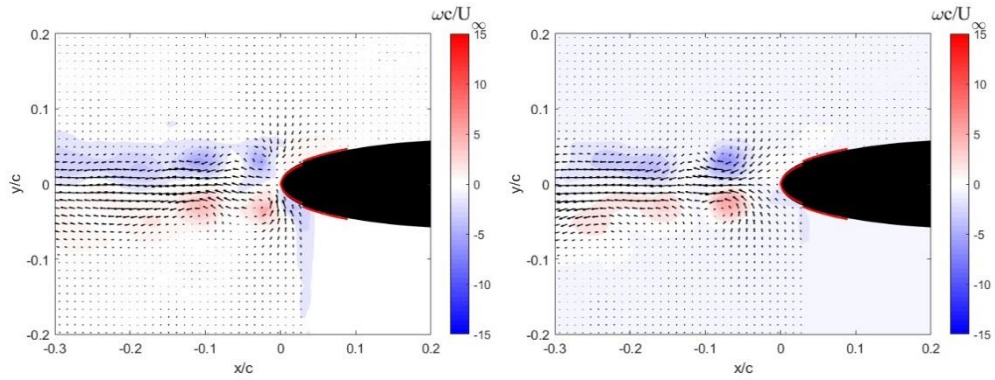


(f) $t = 0.10s$



(g) $t = 0.16s$

(h) $t = 0.22s$



(i) $t = 0.40s$

(j) $t = 0.80s$

Figure 5-12 Instantaneous flow field induced by unsteady actuation of the three-electrode DBD plasma actuator. $t = 0$ refers to the moment plasma is about to be on. Modulation pulse frequency is 29Hz and duty cycle is 25%.

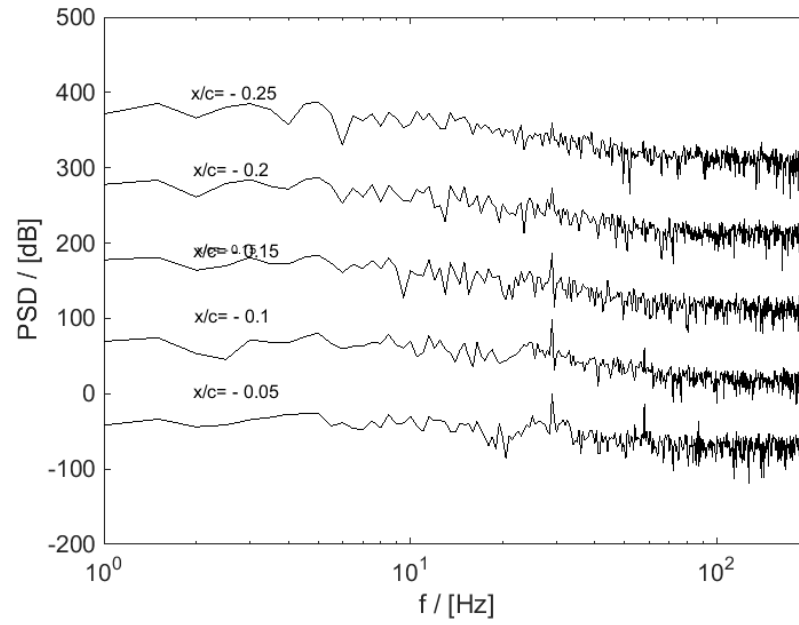


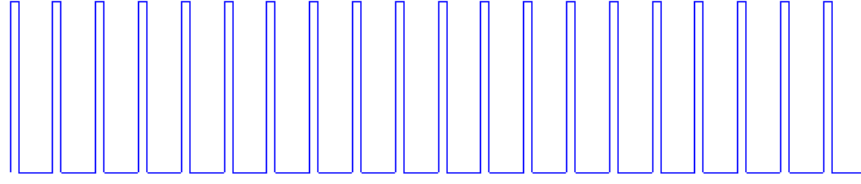
Figure 5-13 Power Spectral analysis of streamwise velocity at different streamwise locations with y/c at jet centreline.

6 Flow Separation Control With Steady Actuation of DBD Plasma Actuator

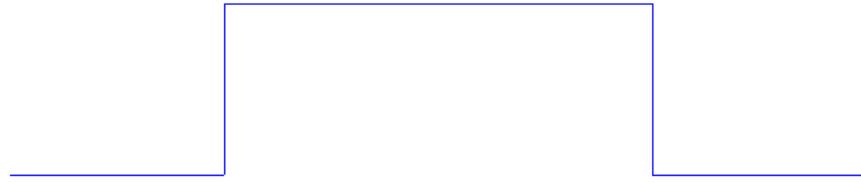
6.1 Introduction

After an understanding of baseline flow around NACA 0012 aerofoil and wall-normal plasma jet by linear plasma synthetic jet actuator, flow separation control was conducted with steady actuation of plasma actuator in this chapter. The setup of PIV system and wind tunnel is the same as that in the baseline flow characteristics study in Chapter 4. Two PSI-MCPG2503C power amplifier devices were used to provide the high AC voltages to the dual exposed electrodes with the embedded electrode connected to ground. The switches of the power amplifier devices are remotely controlled by a BNC 575 Pulse Generator, which is triggered by the signal sent from synchronization box to high-speed camera. When the pulse generator is triggered, a pulse of 1s will be sent to power amplifier devices which would start to provide high AC voltages to plasma actuator. A time delay of 0.5s is set in the pulse generator so that plasma actuator starts 0.5s after high-speed camera, this time delay enables a comparison of the flow behaviours before and after plasma is on. The sampling time of high-speed camera is 2s, in other words, high-speed camera keeps capturing the flow around aerofoil for another 0.5s after plasma is off, thus the dynamics of flow field after plasma is off could be captured as well. The relative timing between camera triggering signal, pulse generated by BNC 575 Pulse Generator and high AC voltage signal is illustrated in Figure 6-1. Here, to clearly illustrate the signals and the relative timing between them, a much lower signal frequency is used in Figure 6-1 (a) and (c) than the actual frequency. In this chapter, flow separation control over a NACA 0012 aerofoil with a wall-normal plasma jet was investigated at varying angles of attack to demonstrate its effectiveness on separation control and the maximum delay in stall angle this flow control technique could achieve. Then, the effect of plasma jet direction on flow separation control was studied to show the advantage of a wall-normal plasma jet in separation control compared with a wall plasma jet. Thirdly, parametric study of plasma jet strength's effect on flow separation control was also conducted. Finally, mechanism for flow separation control with a steady wall-normal plasma jet was

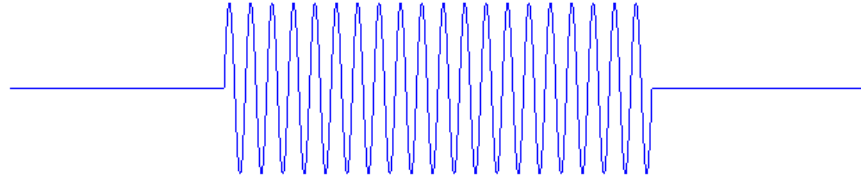
exploited and the possible mechanism was proposed in the final section of this chapter.



(a) signal sent from synchronization to camera



(b) pulse produced by pulse generator



(c) power supply voltage signal

Figure 6-1 Signals from the measurement system devices.

6.2 Flow Separation Control at Varying angles of attack

Flow separation control over a NACA 0012 aerofoil with steady actuation of a linear plasma synthetic jet actuator was investigated at different angles of attack. For all the cases, DBD plasma actuator was driven by sinusoidal voltages of $E_{up\ p-p} = 5.5\text{ kV}$, $f_{up} = 34\text{kHz}$ and $E_{low\ p-p} = 5.5\text{ kV}$, $f_{low} = 34\text{kHz}$. As discussed in Section 5.1, a wall-normal jet would be produced in still air at this voltage, therefore, this wall-normal jet will interact with freestream in the current flow control studies. The corresponding streamwise velocity contours superimposed by streamlines of the time-averaged flow fields with plasma are illustrated in Figure 6-2. When angle

of attack is set at 0° , the flow field with plasma is similar to that without plasma, in other words, the wall-normal plasma jet creates little impacts on the flow field around NACA 0012 aerofoil at this small angle of attack. This is consistent with the findings over a NACA 0015 aerofoil by M. Post and Corke (2004) that lift coefficient is not affected by a single plasma actuator placed at leading edge at small angles of attack. At angle of attack of 8° , the recirculation region disappears and flow goes along the aerofoil surface when plasma is on, demonstrating that flow separation is effectively controlled by this wall-normal plasma jet. As angle of attack increases to 10° and 12° , the severer flow separation could still be controlled by means of interaction between the wall-normal plasma jet and freestream. With angle of attack set at 14° , the wall-normal plasma jet can not make the separated flow reattach to the aerofoil surface although the recirculation region becomes smaller compared with that for the baseline flow. Therefore, leading-edge flow separation is delayed quite a lot with this flow control technique and the limit of this method to reattach the separated flow is $\alpha = 14^\circ$.

The normalized Reynolds stress contours are presented in Figure 6-3 for the flow around a NACA 0012 aerofoil at varying angles of attack with steady actuation of a linear plasma synthetic jet actuator. When the aerofoil is set at an angle of attack of 0° , the Reynolds stress seems not to be affected by the application of a wall-normal plasma jet, once again confirming the wall-normal plasma jet has little effect on the flow around a NACA 0012 aerofoil at angle of attack of 0° . At angles of attack of 8° , 10° and 12° , Reynolds stress locates in a region just above the suction surface of aerofoil, indicating a momentum exchange in the near-wall region, which results in a control of flow separation with steady actuation of plasma actuator. Although the application of a wall-normal plasma jet can not reattach the separated flow at post-stall angle of attack of 14° , Reynolds stress moves closer to the aerofoil surface, corresponding to an alleviation of flow separation. The r.m.s. velocity contours are shown in Figure 6-4. Both the streamwise and vertical r.m.s. velocity fluctuations seem not to be affected by the application of plasma for the flow around the aerofoil at angle of attack of 0° . At angles of attack of $\alpha = 8^\circ$, 10° and 12° , the r.m.s. velocity fluctuation peaks locate at a much smaller distance to aerofoil surface compared with the baseline cases, corresponding to a flow separation control around the aerofoil. When the aerofoil is set at angle of attack of

$\alpha = 14^\circ$, the r.m.s. velocity fluctuation peak is located further away from the suction surface than that at smaller angles of attack, which is due to the existence of the alleviated flow separation.

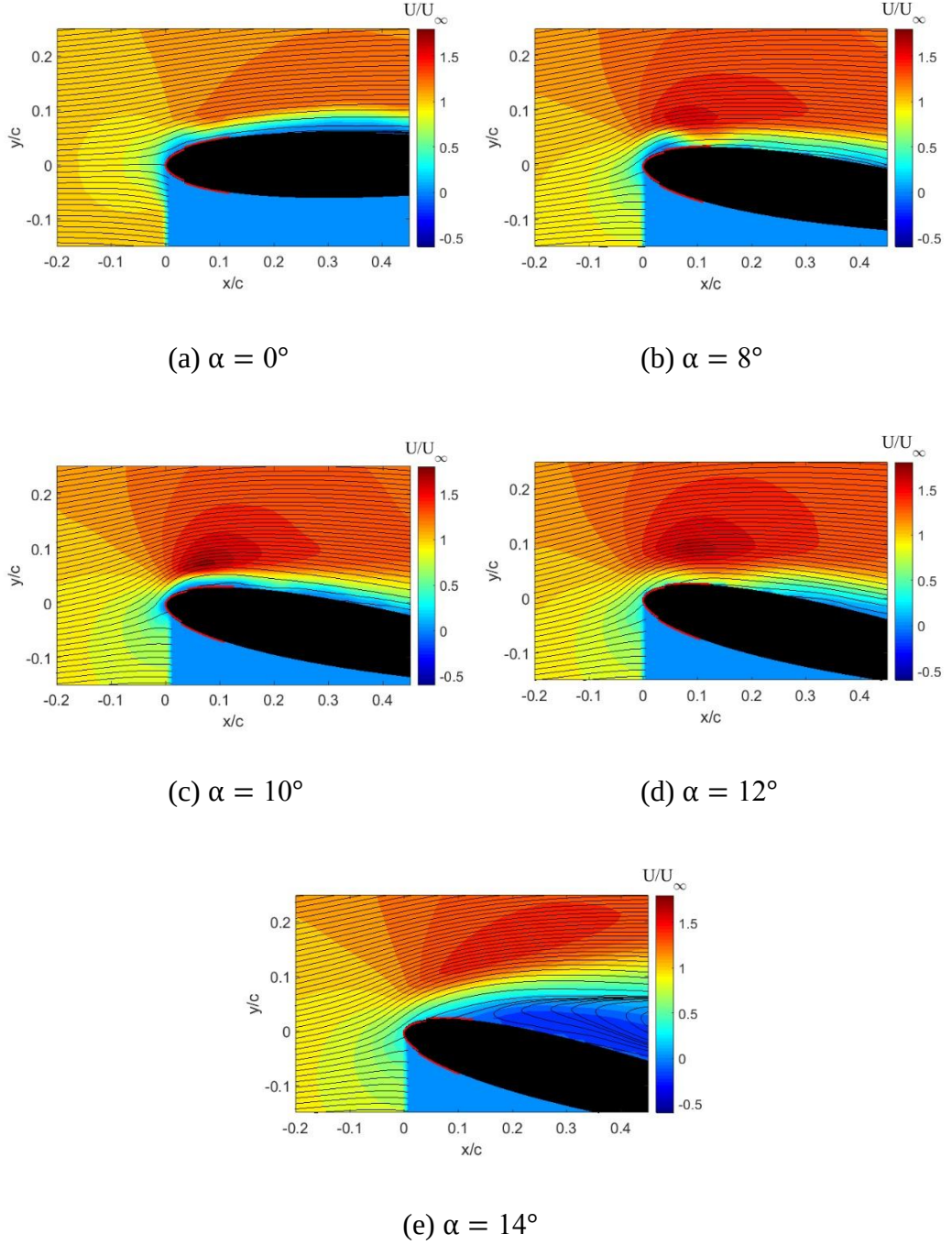
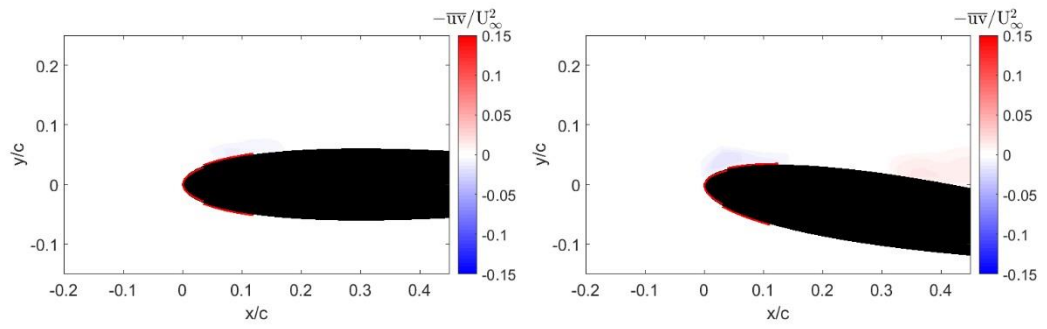
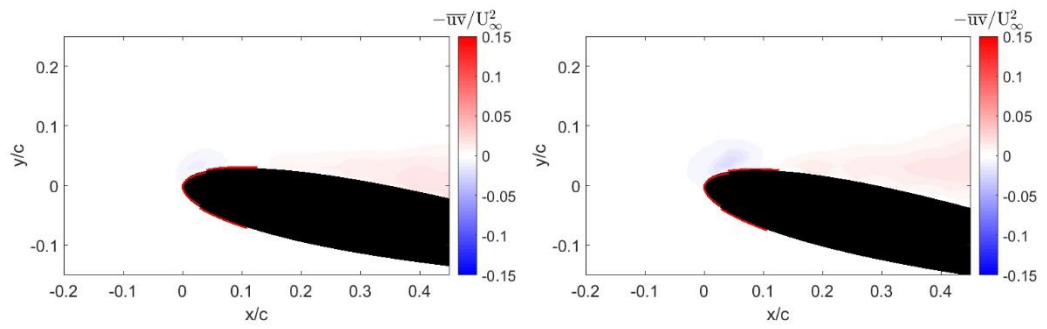


Figure 6-2 Normalized streamwise velocity contour superimposed by streamline of the time-averaged controlled flow fields with plasma at varying angles of attack. Freestream velocity is 3m/s and the corresponding chord Reynolds number is 20,000.



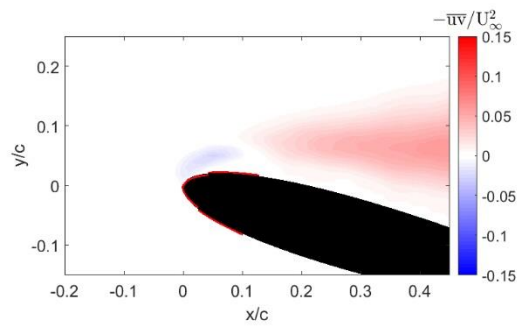
(a) $\alpha = 0^\circ$

(b) $\alpha = 8^\circ$



(c) $\alpha = 10^\circ$

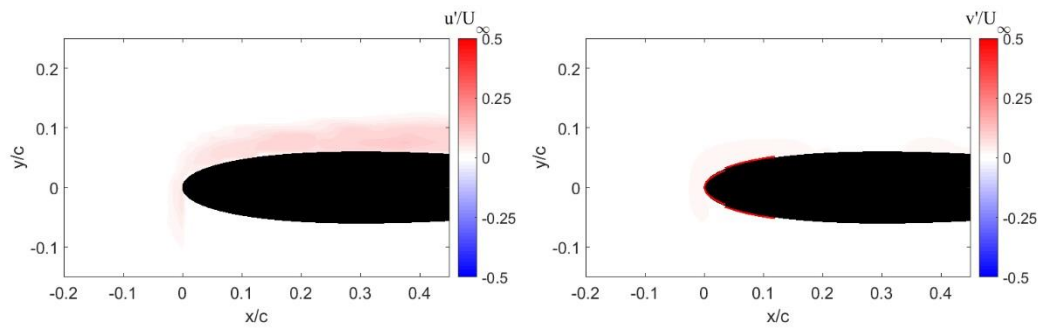
(d) $\alpha = 12^\circ$



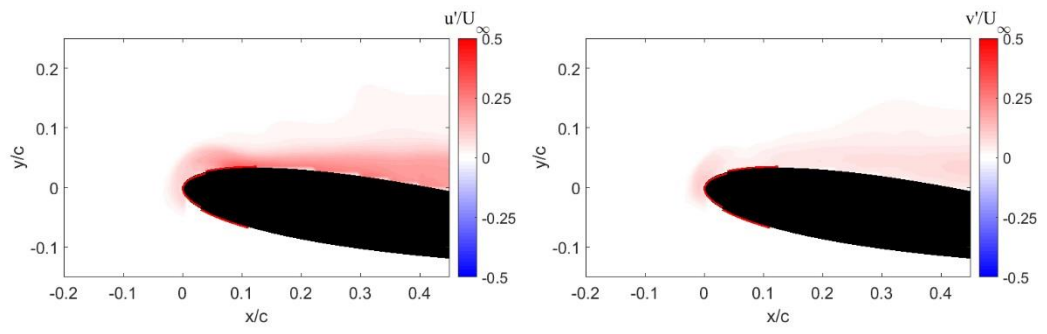
(e) $\alpha = 14^\circ$

Figure 6-3 Normalized Reynolds stress contours for the mean controlled flow fields with plasma at varying angles of attack.

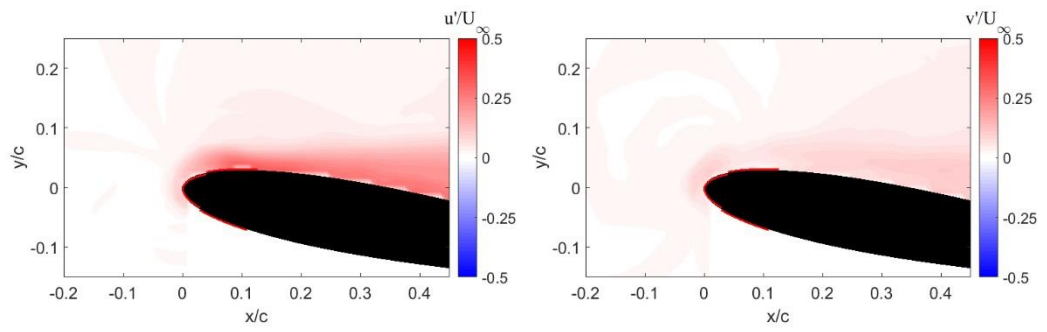
a) $\alpha = 0^\circ$



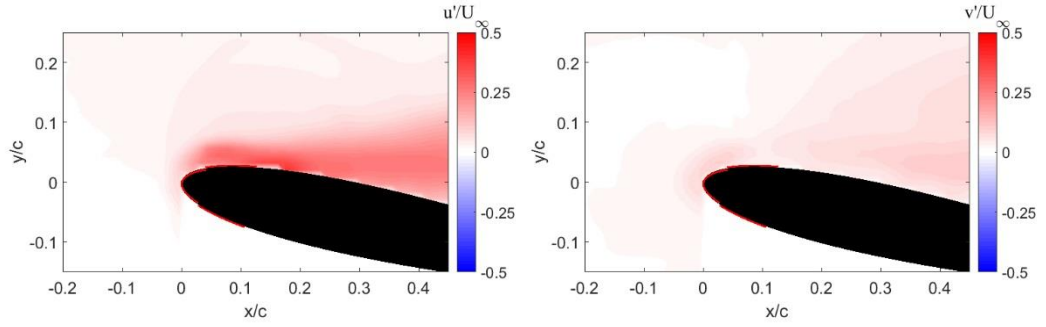
b) $\alpha = 8^\circ$



c) $\alpha = 10^\circ$



d) $\alpha = 12^\circ$



e) $\alpha = 14^\circ$

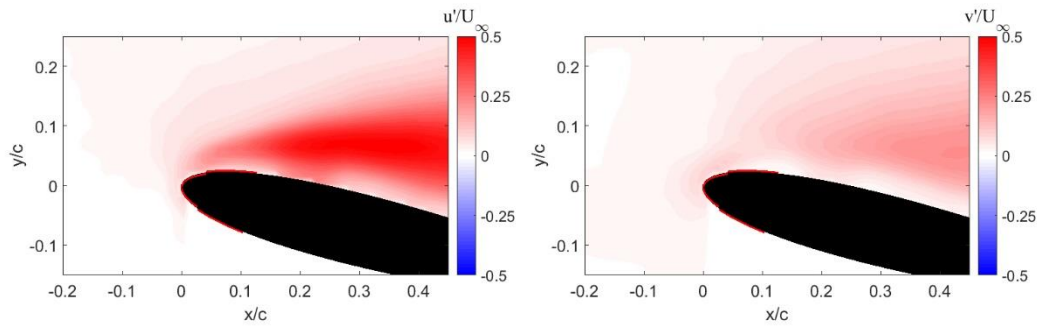


Figure 6-4 Contours of streamwise (left column) and vertical (right column) root-mean-square velocities with steady activation of plasma.

The dynamics of the controlled flow fields around the NACA 0012 aerofoil with steady actuation of plasma actuator was investigated as well. Figure 6-5 shows the velocity vectors, spanwise vorticity contours and swirling strength isolines for the instantaneous flow fields at angle of attack of 0° and freestream velocity of 3m/s. Plasma is about to be turned on at $t^+ = 0.00$ and lasts for a time period of $t^+ = 30.00$, therefore, Figure 6-5 (a) shows the baseline flow while Figure 6-5 (b), (c) and (d) represent the instantaneous controlled flows with plasma. It seems that plasma has little effect on the flow field around NACA 0012 aerofoil at angle of attack of 0° by comparing the flow fields with and without plasma. This shows the wall-normal plasma jet does not even affect the flow field in a transient sense at angle of attack of 0° .

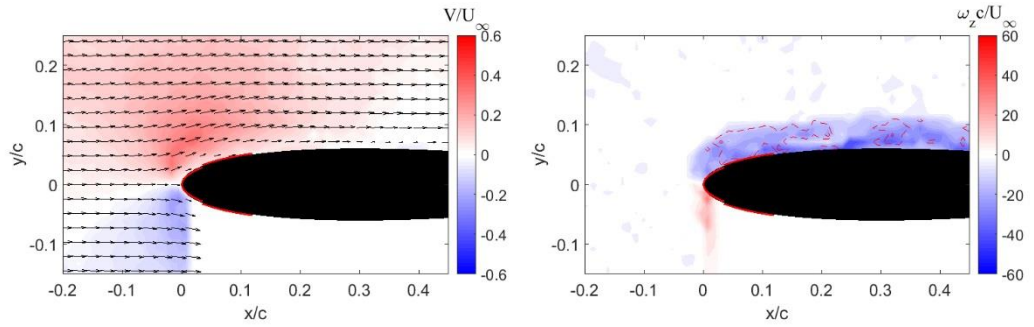
At angle of attack of 8° , the spanwise vorticity located away from aerofoil surface in Figure 6-6 (a) indicates a separated shear layer, confirming a flow separation before plasma is on. When plasma actuator is activated, the interaction between the

wall-normal plasma jet and the ambient flow leads to a mixing between the high-momentum fluids and boundary layer, so the boundary layer will be capable of withstanding adverse pressure gradient and keeps attached to the aerofoil surface. The research of flow control around a circular cylinder using DBD plasma actuator by Jukes and Choi (2009) finds that a discontinuity is caused between the attached boundary layer and the separated shear layer when plasma is actuated, which results in a roll-up of the tail of the old shear layer as it moves downstream. In this study, a similar phenomenon is observed as well, as can be seen in Figure 6-6 (b), the separated shear layer rolls up into a clockwise vortex at $x/c \approx 0.33$, downstream of the attached boundary layer. Similar observations of transient separation control were also made by other researchers (Brzozowski et al., 2010; Siau and Bonnet, 2017; Woo and Glezer, 2013) after an activation of an actuator. As time goes on, the vortex due to the rollup of the separated shear layer travels out of the field of view and the attached boundary layer occupies the region above the upper surface, which can be confirmed by the velocity vectors and attached vorticity region in Figure 6-6 (c) and (d). Therefore, flow separation around the NACA0012 aerofoil at this angle of attack is effectively controlled with steady actuation of plasma actuator. In addition, negative spanwise vorticity occupies a thicker region close to the leading edge when plasma is activated, implying a generation of vorticity due to the interaction between plasma jet and the local flow. At larger angles of attack of 10° and 12° , similar phenomenon was observed around the upper surface of aerofoil after plasma actuator is switched on.

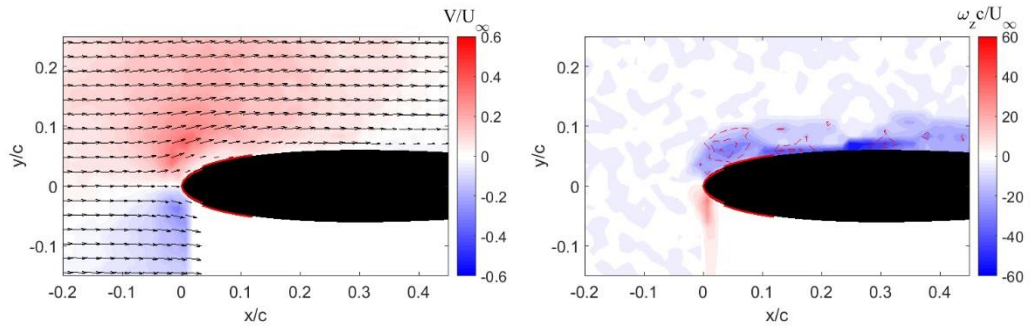
When the aerofoil is set up at a post-stall angle of attack of 14° , flow separation is so severe that it can not be eliminated with a steady wall-normal plasma jet. After plasma is on, the old separated shear layer breaks away from the upstream boundary layer and forms into a large-scale clockwise vortex which rolls up and travels downstream. The entrainment by this vortex enable a momentum exchange between the high-momentum fluids and the boundary layer, as a consequence, the local velocity gradient and pressure gradient would be modified. According to Darabi and Wygnanski (2004), the outwards movement of the near-wall fluids leads to a decrease in local static pressure. Consequently, the resulting favourable pressure gradient and momentum entrainment together make the flow upstream of this vortex almost reattach to the aerofoil surface in Figure 6-9 (b) and (c). After

the vortex travels away, even the re-energized boundary layer is not capable of tolerating the severe adverse pressure gradient any more, flow starts to separate again although it is alleviated compared with the baseline flow at angle of attack of 14° , which is demonstrated in Figure 6-9 (d), (e) and (f). As most of the flow separation control studies with unsteady actuation (switch on and off alternatively) of plasma actuator were conducted at angle of attack of 14° in Chapter 7, flow behaviours were also investigated after plasma is turned off at this angle of attack. As shown in Figure 6-9 (g), the re-energized shear layer breaks away from the upstream boundary layer and sheds off into another large-scale vortex after plasma is stopped. An almost attached boundary layer is followed probably due to the local favourable pressure gradient by this large-scale vortex. As the vortex travels downstream, flow gradually starts to recover to its baseline state again, as demonstrated in Figure 6-9 (h) and (i).

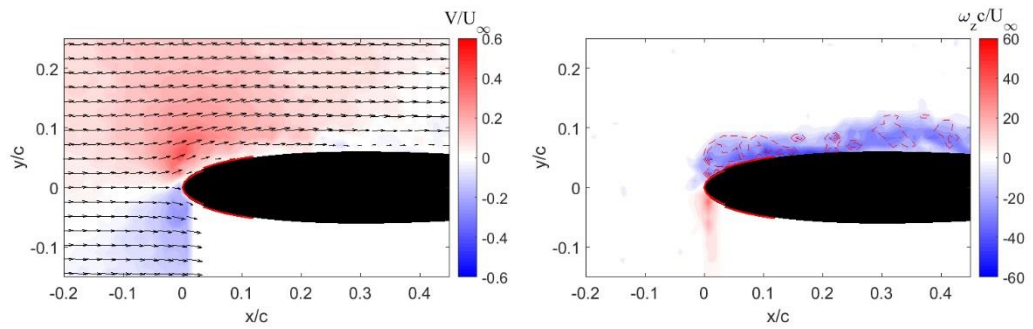
a) $t^+ = 0.00$



b) $t^+ = 0.30$



c) $t^+ = 2.40$



d) $t^+ = 4.80$

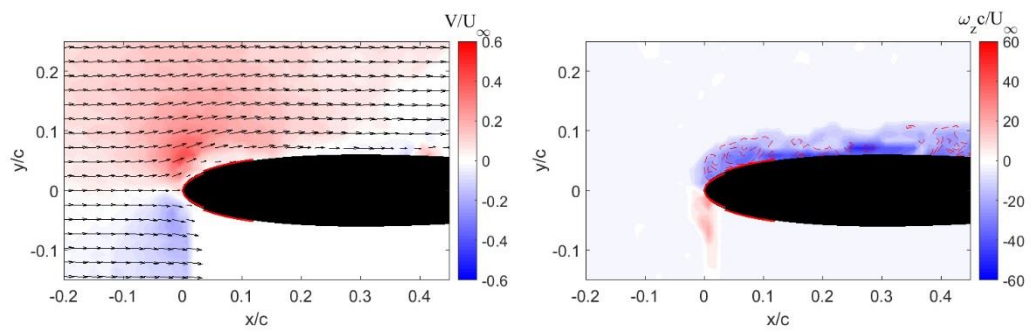
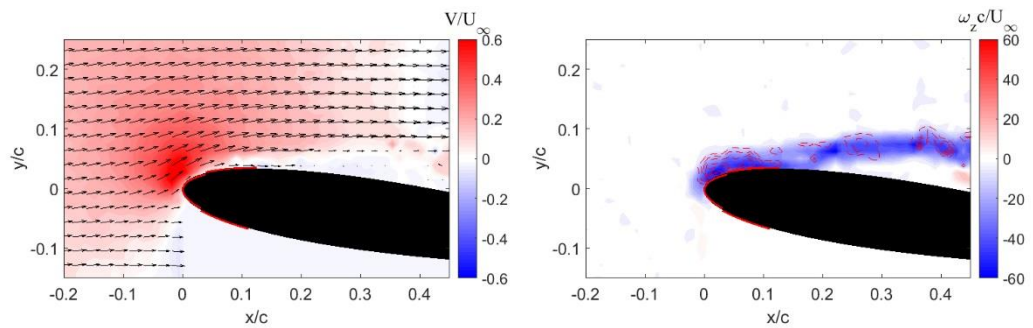
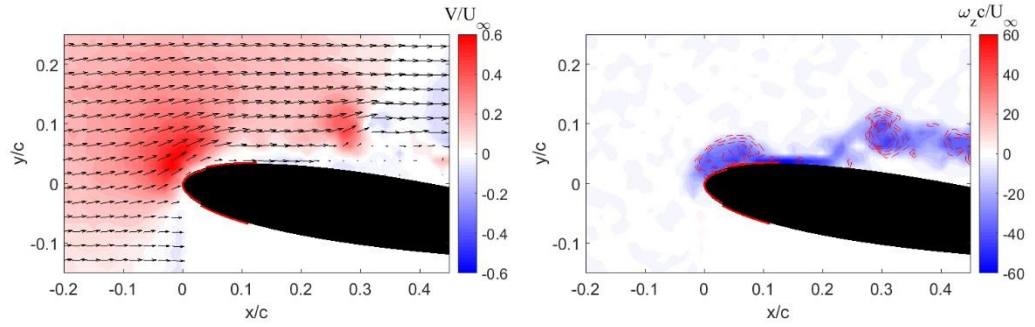


Figure 6-5 Instantaneous spanwise vorticity field superimposed by velocity vectors and swirling strength contours around NACA 0012 aerofoil: $t^+ = 0.00$ refers to the moment plasma is about to be on. Freestream velocity is 3m/s and angle of attack is $\alpha = 0^\circ$.

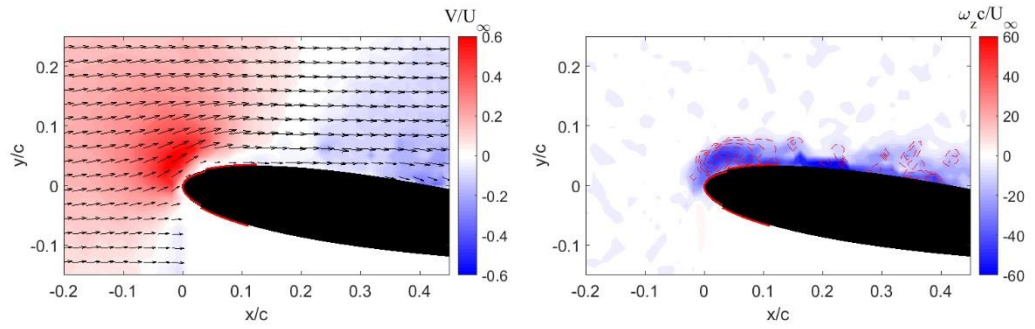
a) $t^+ = 0.00$



b) $t^+ = 0.30$



c) $t^+ = 2.40$



d) $t^+ = 4.80$

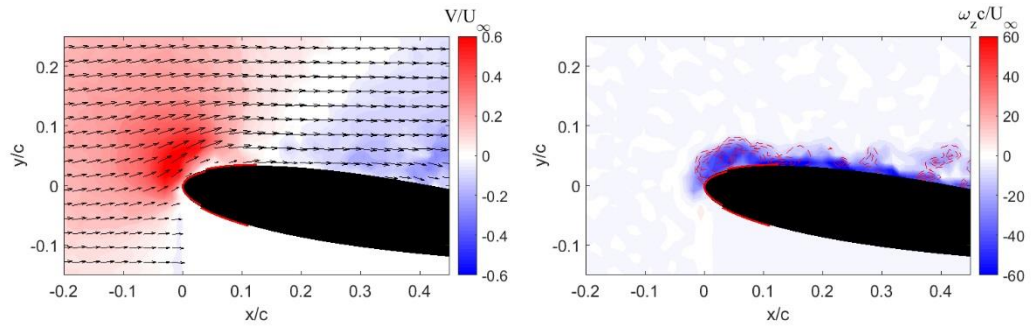
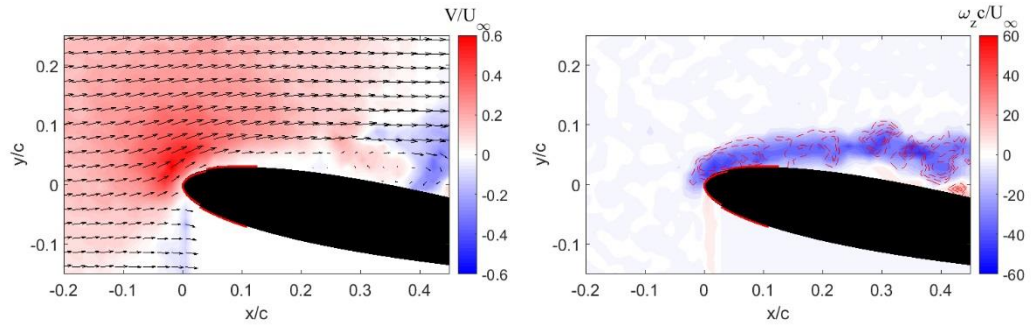
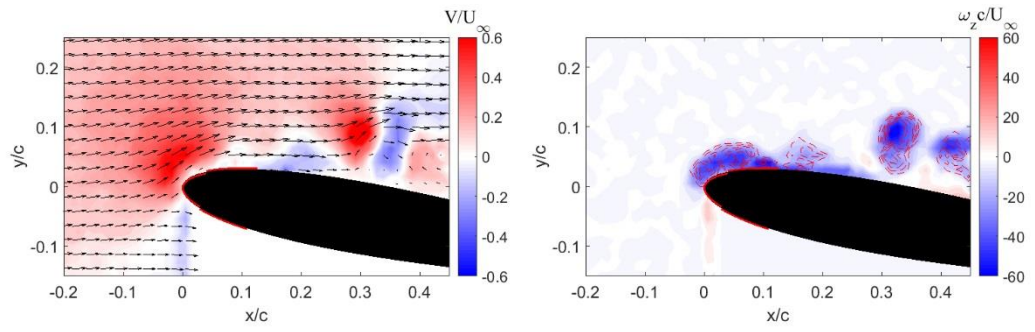


Figure 6-6 Instantaneous spanwise vorticity field superimposed by velocity vectors and swirling strength contours around NACA 0012 aerofoil. $t^+ = 0.00$ refers to the moment plasma is about to be on. Freestream velocity is 3m/s and angle of attack is $\alpha = 8^\circ$.

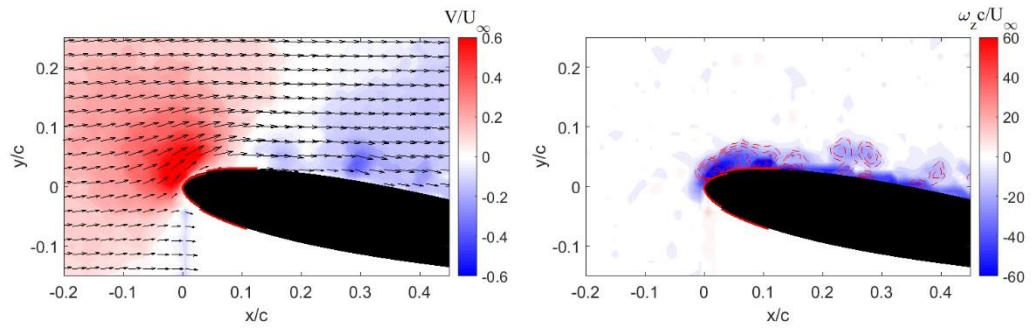
a) $t^+ = 0.00$



b) $t^+ = 0.30$



c) $t^+ = 2.40$



d) $t^+ = 4.80$

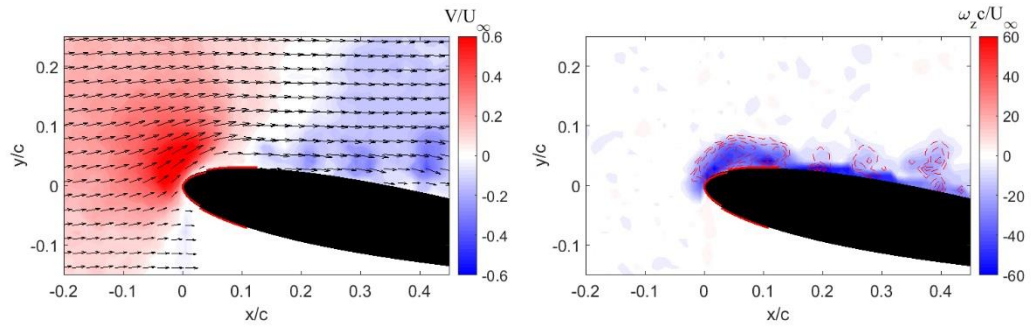
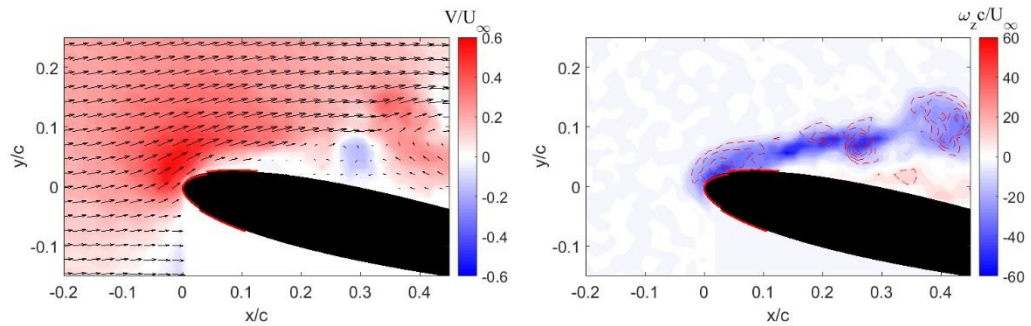
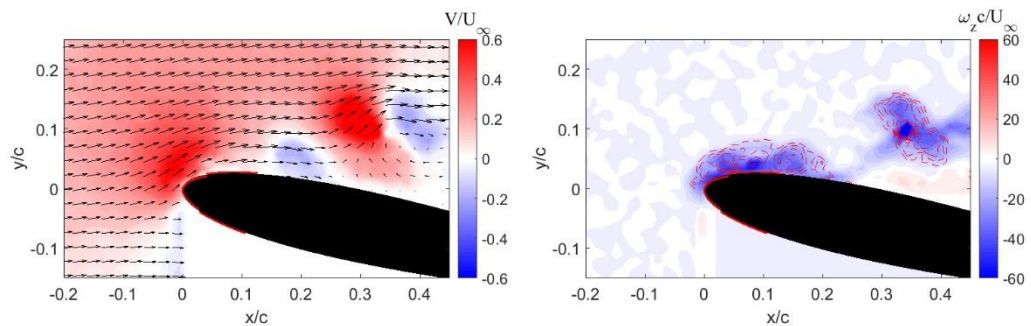


Figure 6-7 Instantaneous spanwise vorticity field superimposed by velocity vectors and swirling strength contours around NACA 0012 aerofoil. $t^+ = 0.00$ refers to the moment plasma is about to be on. Freestream velocity is 3m/s and angle of attack is $\alpha = 10^\circ$.

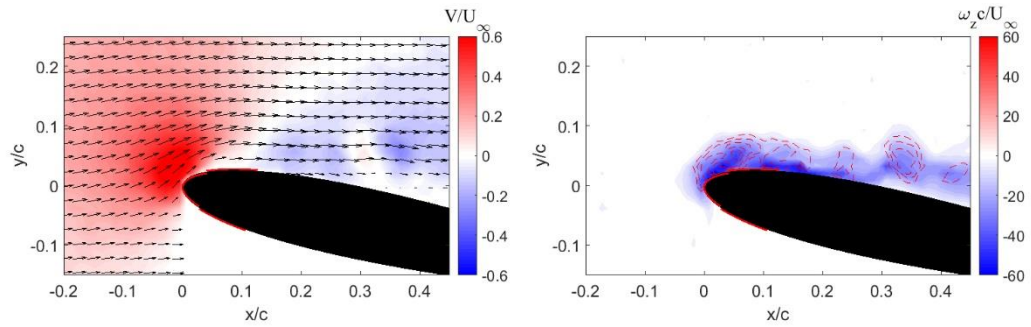
a) $t^+ = 0.00$



b) $t^+ = 0.30$



c) $t^+ = 2.40$



d) $t^+ = 4.80$

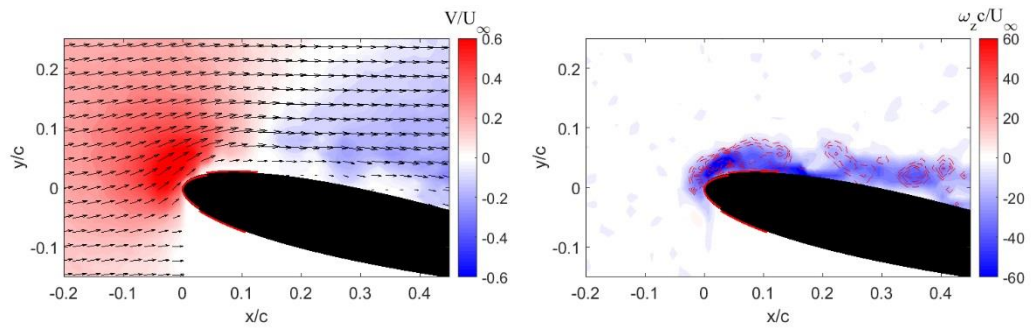
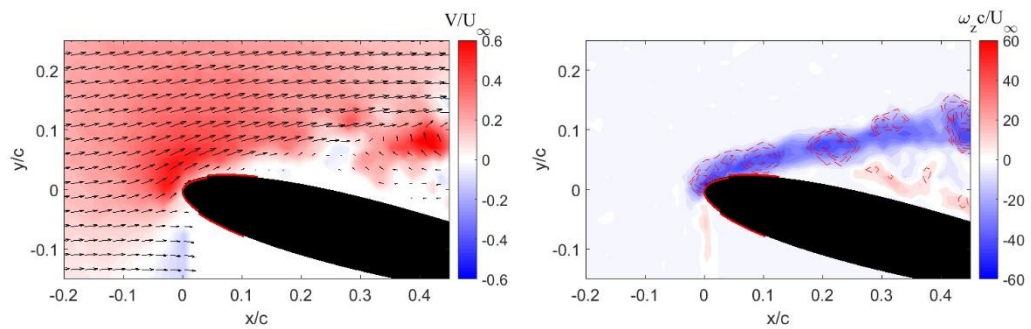
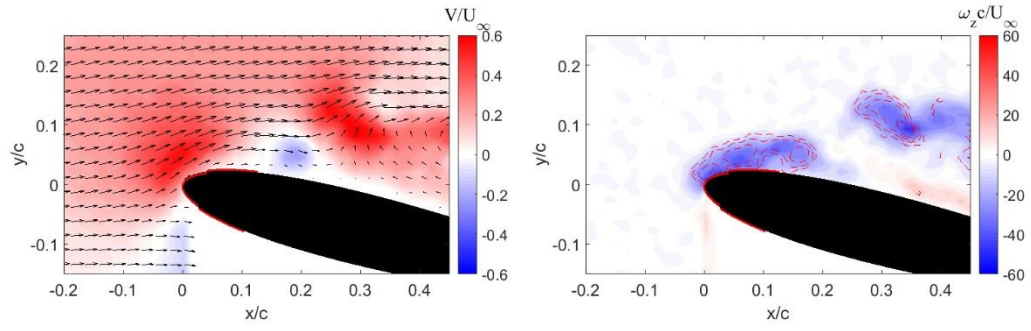


Figure 6-8 Instantaneous spanwise vorticity field superimposed by velocity vectors and swirling strength contours around NACA 0012 aerofoil. $t^+ = 0.00$ refers to the moment plasma is about to be on. Freestream velocity is 3m/s and angle of attack is $\alpha = 12^\circ$.

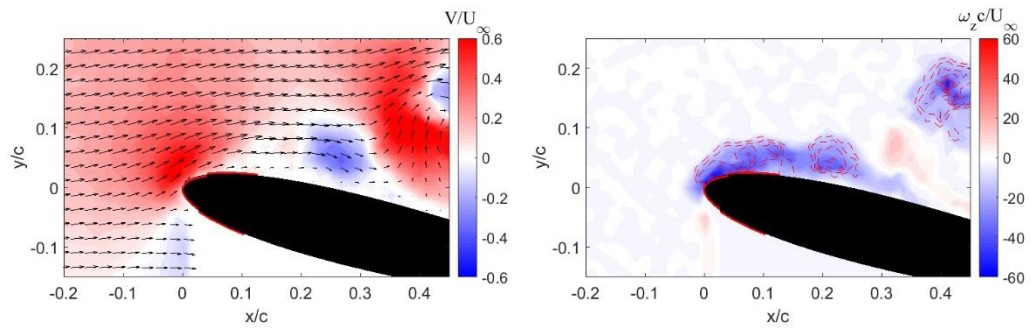
a) $t^+ = 0.00$



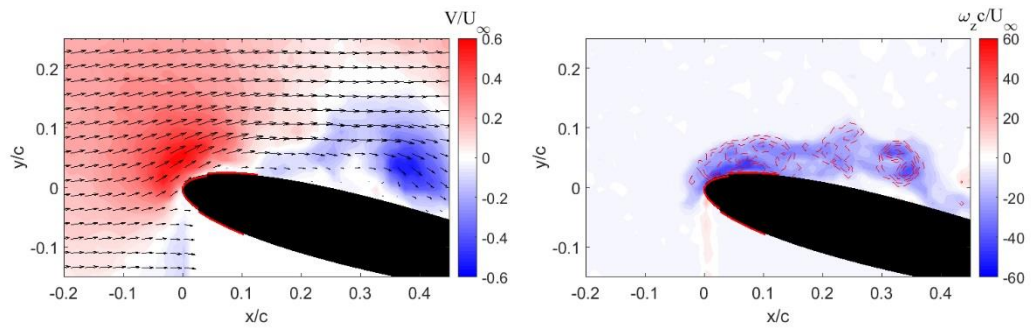
b) $t^+ = 0.30$



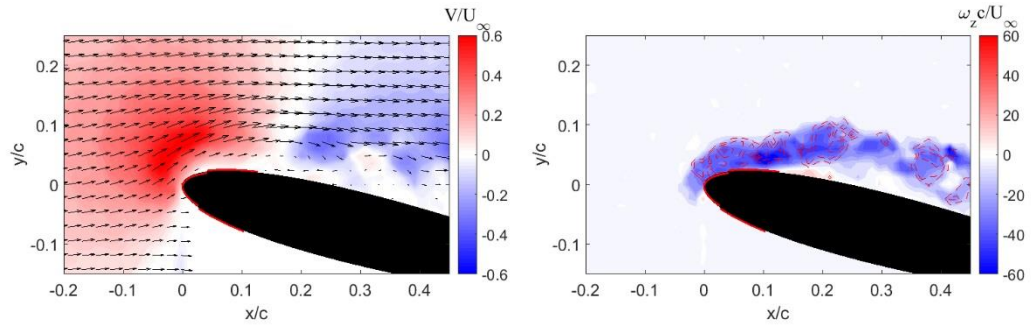
c) $t^+ = 0.45$



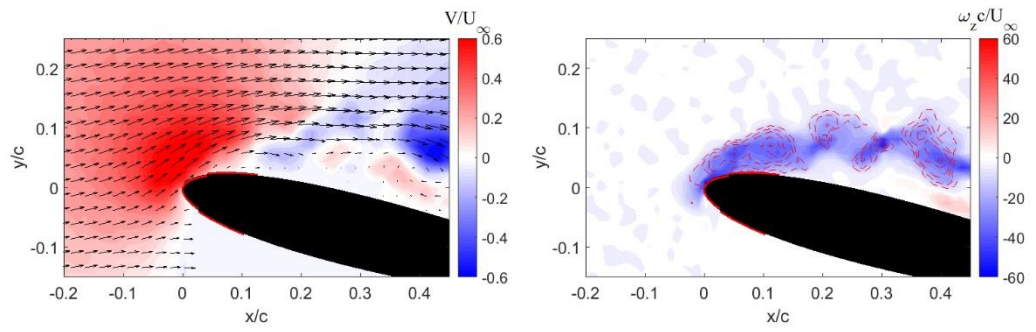
d) $t^+ = 0.75$



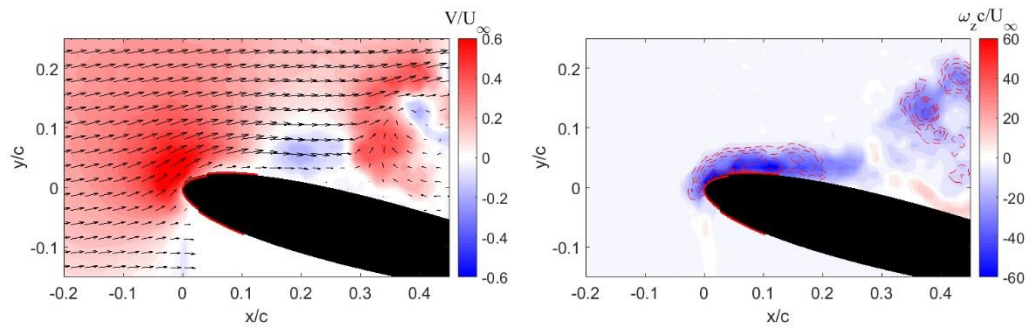
e) $t^+ = 2.40$



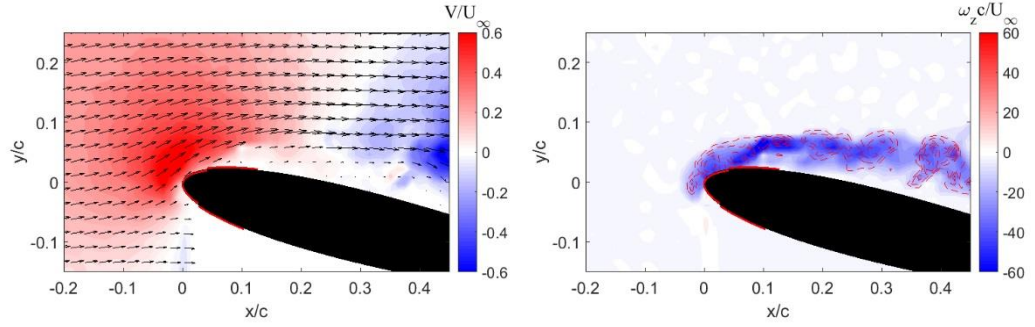
f) $t^+ = 4.80$



g) $t^+ = 30.60$



h) $t^+ = 31.50$



i) $t^+ = 33.00$

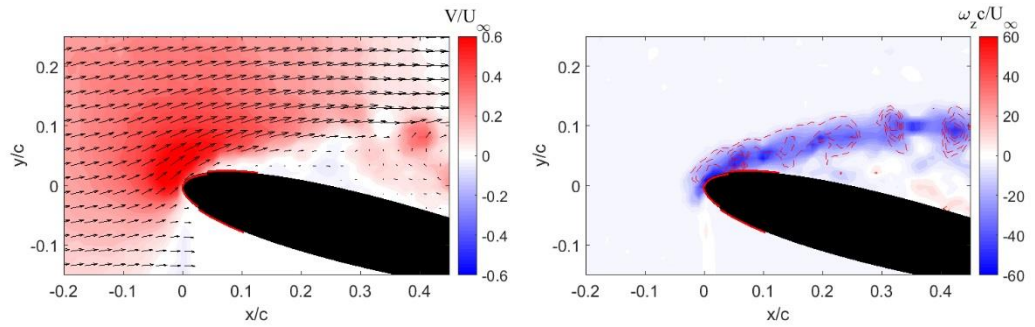


Figure 6-9 Instantaneous spanwise vorticity field superimposed by velocity vectors and swirling strength contours around NACA 0012 aerofoil. $t^+ = 0.00$ refers to the moment plasma is about to be on and plasma is switched off at $t^+ = 30.00$. Freestream velocity is 3m/s and angle of attack is $\alpha = 14^\circ$.

6.3 Effect of Plasma Jet Direction On Flow Separation Control

To study plasma jet direction's effect on flow separation control over a NACA 0012 aerofoil, PIV measurements were conducted with high-amplitude sinusoidal voltages applied to only the lower-surface or upper-surface exposed electrode. In other words, controlled flow fields with upstream-directed wall plasma jets were measured for comparison with that controlled by a wall-normal plasma jet. Here, freestream velocity is 3m/s and plasma momentum keeps fixed at 0.25%.

Figure 6-10 shows the streamwise velocity contour superimposed by streamline of the time-averaged flow fields around NACA 0012 aerofoil at angle of attack of 14° with high AC voltage applied to only the lower-surface plasma actuator (Figure 6-10 (a)) and the upper-surface plasma actuator (Figure 6-10 (b)) respectively. As can be seen from the streamlines in Figure 6-10, flow still separates from aerofoil

surface with wall plasma jet applied, in other words, flow separation can't be eliminated through the use of wall plasma jet. As for the size of the recirculation region, it becomes smaller with wall plasma jets than that for the baseline flow, indicating a positive effect on flow separation control effect. However, the reverse flow region is larger relative to that with wall-normal plasma jet, therefore wall-normal plasma jet has a better capability of controlling leading-edge flow separation than the upstream-directed wall plasma jets. This is probably due to the greater velocity gradient between freestream and the wall-normal jet, which might result in stronger vortical structures and momentum exchange.

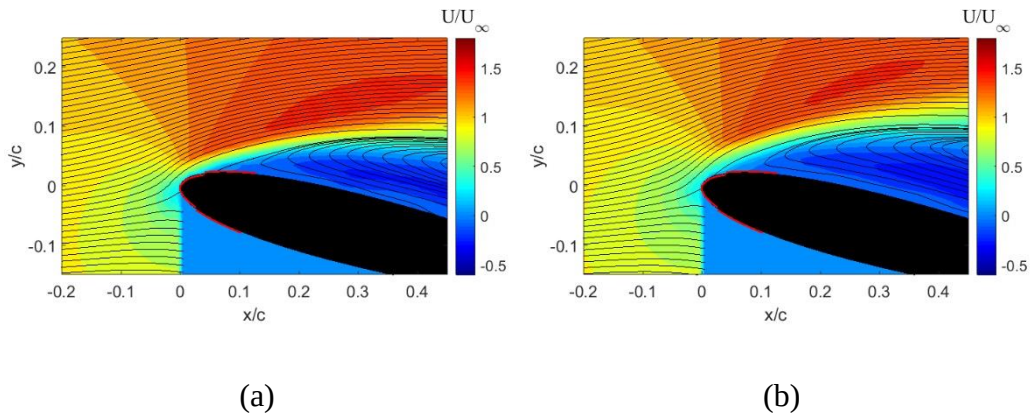


Figure 6-10 Normalized streamwise velocity contour superimposed by streamline of controlled flow fields with different plasma jet direction: (a) $\alpha_j = -67^\circ$, $C_\mu = 0.25\%$; (b) $\alpha_j = 63^\circ$, $C_\mu = 0.25\%$. The angle of attack is 14° and freestream velocity is 3m/s, corresponding chord-based Reynolds number is 20,000.

6.4 Effect of Plasma Jet Strength On Flow Separation Control

Study on the effect of plasma jet strength on flow separation control around NACA 0012 aerofoil was also conducted and the results will be presented in this section. Specifically, PIV measurements of flow field around a NACA 0012 aerofoil with wall-normal plasma jets of different strength were conducted. In this part, freestream velocity is still 3 m/s and the corresponding chord-based Reynolds number is 20,000. The sampling frequency of high-speed camera is 2,000 Hz, basically, the setup is consistent with that in the previous section except that the wall-normal plasma jets are of different strengths.

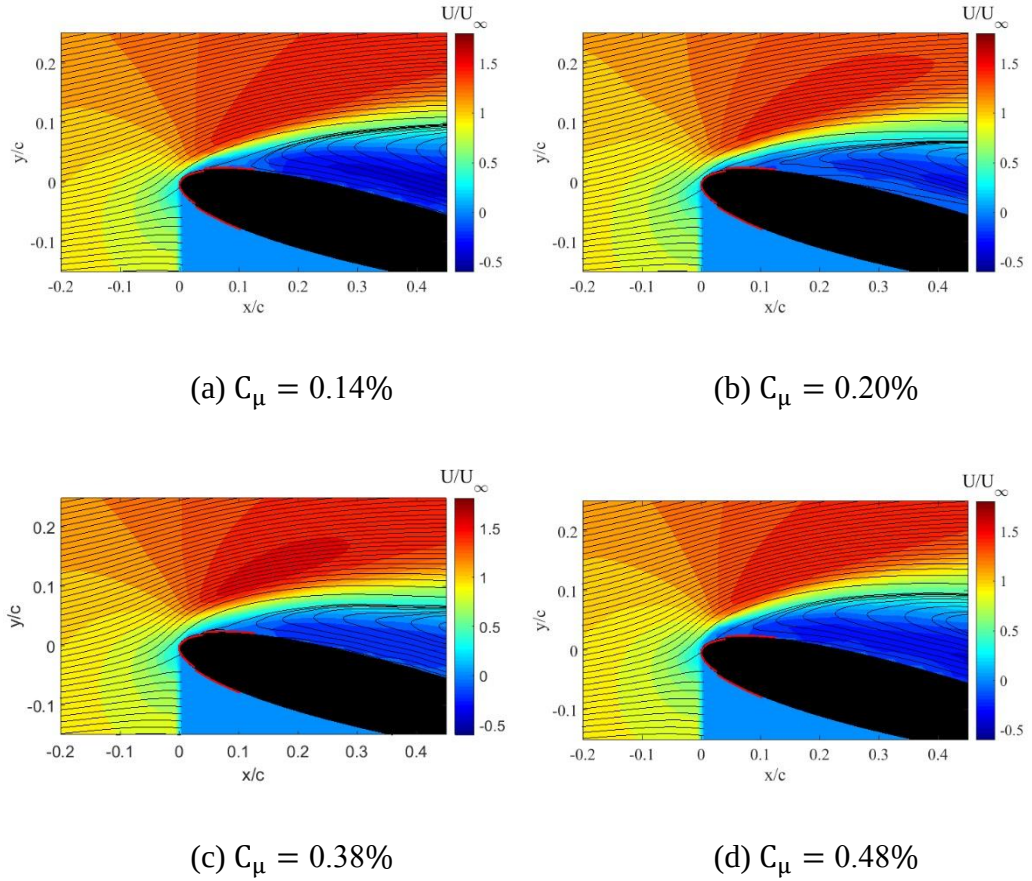


Figure 6-11 Normalized streamwise velocity contour superimposed by streamline of controlled flow fields with different plasma jet strength. The angle of attack is $\alpha = 14^\circ$ and freestream velocity is 3m/s, corresponding to a chord Reynolds number of 20,000.

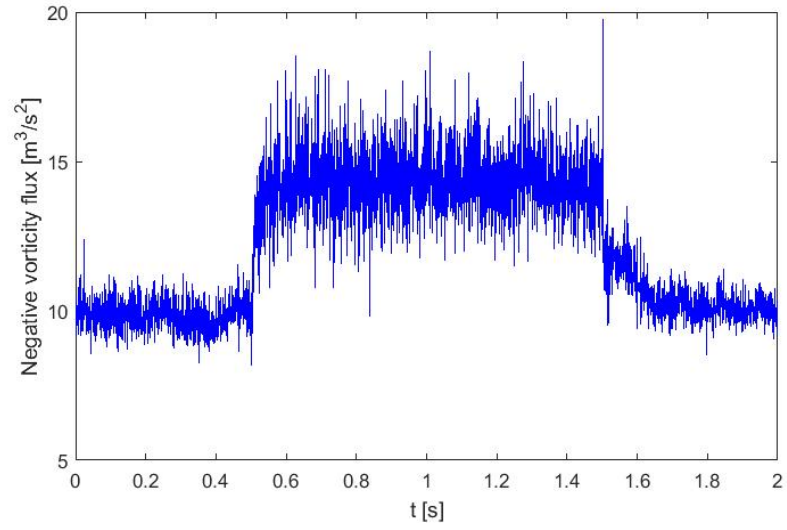
Figure 6-11 shows the streamline of the time-averaged flow field around a NACA 0012 aerofoil with steady actuation of plasma actuator by different sinusoidal voltage amplitudes. When a very weak wall-normal plasma jet with momentum coefficient of 0.14% is applied at the leading edge of aerofoil, a severe flow separation is observed. This shows that plasma can't give a prominent effect on the flow separation around aerofoil at post-stall angle of attack of 14° when the plasma momentum coefficient is 0.14%. As expected, the flow separation is much more alleviated with stronger plasma jets applied in Figure 6-11 (b) and (c) where plasma momentum of $C_\mu = 0.20\%$ and 0.38% are injected respectively. However, a further increase in plasma jet strength will not result in a better flow separation control effect. This is demonstrated in Figure 6-11 (d), which shows that a plasma jet with a momentum coefficient of 0.48% damages the beneficial effect on flow separation

by weaker plasma jets and causes a severer flow separation. This is consistent with the findings over a laminar aerofoil by Packard et al. (2013). Up to now, the reason for this phenomenon is still not very clear.

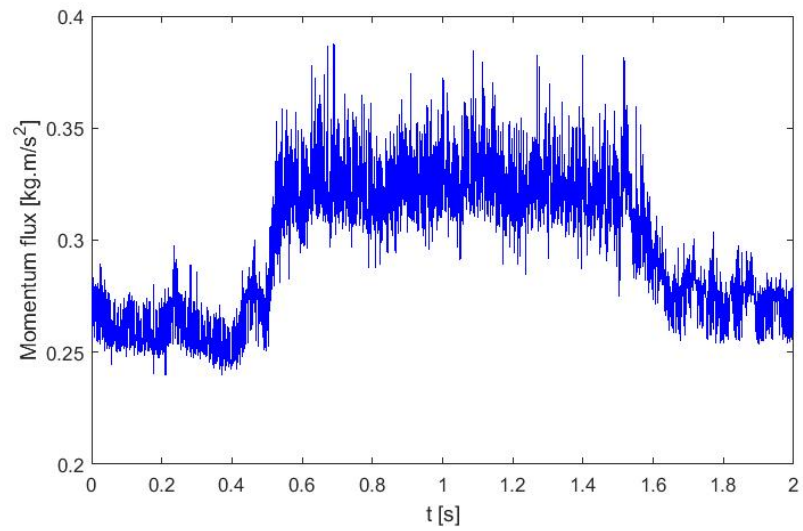
6.5 Mechanism Exploration

In the previous sections of this chapter, the separation control effect of a steady wall-normal plasma jet was demonstrated. It clearly shows that the separation around a NACA 0012 aerofoil could be entirely eliminated at Reynolds number of 20,000 and angles of attack of $\alpha = 8^\circ$, 10° and 12° when a steady wall-normal plasma jet with $C_\mu = 0.25\%$ is applied. At a higher angle of attack of $\alpha = 14^\circ$, since the adverse pressure gradient is severer, the interaction between freestream and this plasma jet leads to an alleviation of flow separation rather than a flow reattachment. Therefore, a wall-normal plasma jet could affect flow separation significantly in a positive way. In this section, the mechanism accounting for flow separation control with wall-normal plasma jet will be explored.

Vorticity flux $\int \omega_z U dy$ and momentum flux $\int \rho U^2 dy$ were utilized to study the mechanism for flow separation control around the NACA0012 aerofoil with steady actuation of a linear plasma synthetic jet actuator. Here, ω_z is spanwise vorticity, U is streamwise velocity and ρ is air density. Integration was conducted at $x/c = 0.065$ and y is from the upper surface to $0.15c$ above it. Figure 6-12 shows the development of vorticity and momentum flux over time at angle of attack of 8° with steady actuation of plasma actuator starting at $t = 0.5s$ and stopping at $t = 1.5s$. As can be seen in Figure 6-12 (a), vorticity flux magnitude increases quickly from around $10 \text{ m}^3/\text{s}^2$ to approximately $14.2 \text{ m}^3/\text{s}^2$ after plasma is turned on. This indicates an addition of vorticity to the flow above the suction surface with the application of the wall-normal plasma jet. After plasma is switched off, the vorticity magnitude drops to $10 \text{ m}^3/\text{s}^2$ again within $0.13s$. A similar trend is observed in momentum flux as well, which might suggest that the wall-normal plasma jet works like a vortex generator which introduces high-momentum fluids into the boundary layer, resulting in a more energetic flow in terms of momentum flux.



(a)

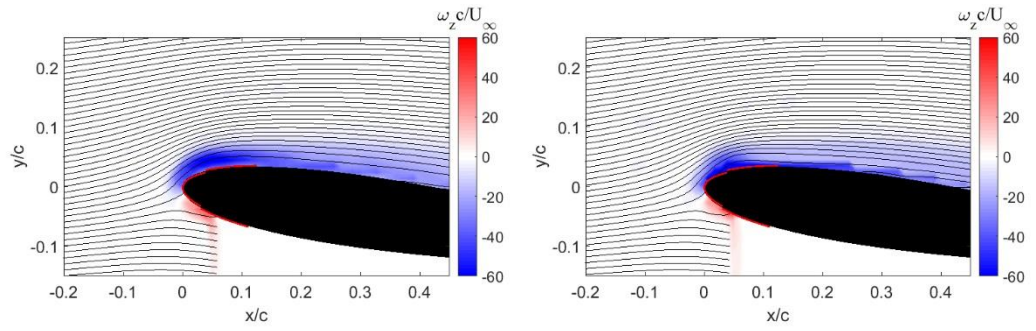


(b)

Figure 6-12 Transient profiles at $x/c = 0.065$: (a) vorticity flux; (b) momentum flux. Activation of plasma took place at $t = 0.50s$ and termination of plasma at $t = 1.50s$.

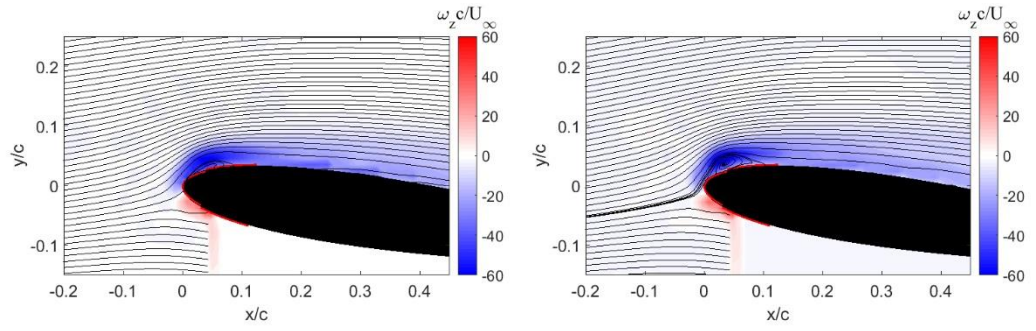
To validate the conjecture that the wall-normal plasma jet works in a similar way to vortex generator, PIV measurements were conducted at higher jet-to-freestream velocity ratios to investigate how this plasma jet interacts with the flow around the leading edge of the NACA 0012 aerofoil. In other words, reduced freestream velocities and a stronger plasma jet were used, specifically, freestream velocities of 3.0 m/s, 2.5 m/s, 2.0 m/s, 1.5 m/s, 1.0 m/s and 0.8 m/s were used while the wall-normal plasma jet was produced by higher AC voltages of 6.0 kV and has a maximum mean axial velocity of around 0.9 m/s. The corresponding wall-normal plasma jet momentum coefficient is $C_{\mu} = 0.48\%$, 0.58% , 0.72% , 0.96% , 1.44% and 1.80% , and velocity ratio is 0.3, 0.36, 0.45, 0.6, 0.9 and 1.125 respectively. The sampling frequency and time are consistent with those in Sections 6.2, 6.3 and 6.4. Although measurements were conducted at different angles of attack, only the results at angle of attack of 8° are presented in this section to demonstrate how the wall-normal plasma jet affects the adjacent flow around NACA 0012 aerofoil.

Time-averaged analysis of flow fields with plasma was conducted to study the effect of the wall-normal plasma jet. Figure 6-13 shows the spanwise vorticity contour superimposed by streamline for the mean flow fields around the aerofoil with the application of a wall-normal plasma jet at angle of attack of 8° and varying freestream velocities. With plasma on, separation is entirely controlled and flow goes along the aerofoil surface when the freestream is set at 3 m/s and 2.5 m/s, which is similar to the controlled case at angle of attack of 8° using a wall-normal plasma jet ($U_{\infty} = 3.0$ m/s and $C_{\mu} = 0.25\%$) in Section 6.2. A very tiny recirculation region starts to form behind the wall-normal plasma jet when the freestream velocity is 2.0 m/s, and the size of this recirculation bubble increases with jet-to-freestream velocity ratios, as shown in Figure 6-13 (c), (d), (e) and (f). The wall-normal plasma jet is weaker than freestream, so it would be bent towards the freestream direction, resulting in the formation of a recirculation bubble behind the plasma jet. This is similar to the findings of steady actuation of linear plasma synthetic jet actuator in crossflow over a flat plate by Santhanakrishnan et al. (2006). The streamline in Figure 6-13 also confirms a reattachment of the separated flow downstream of the recirculation bubble created by the interaction between the wall-normal plasma jet and the freestream. Therefore, the application of wall-normal plasma jet could lead to a control of flow separation around aerofoil.



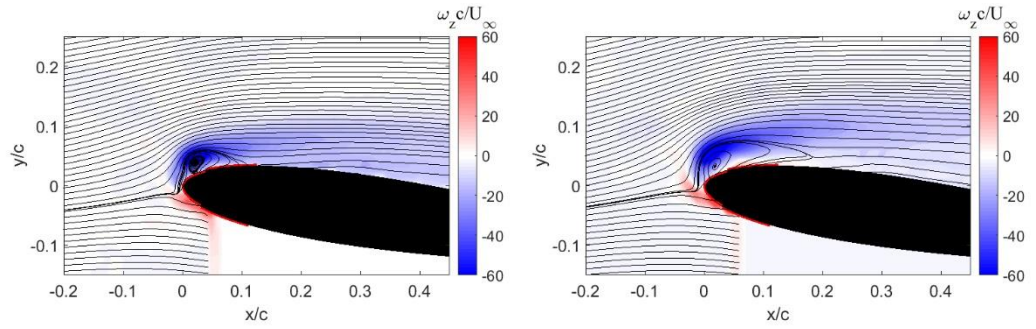
(a) $U_{\infty} = 3.0$ m/s

(b) $U_{\infty} = 2.5$ m/s



(c) $U_{\infty} = 2.0$ m/s

(d) $U_{\infty} = 1.5$ m/s

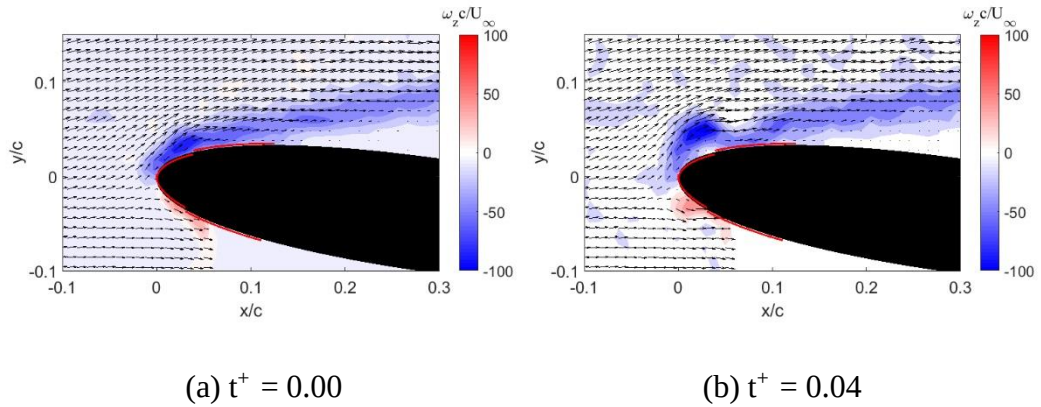


(e) $U_{\infty} = 1.0$ m/s

(f) $U_{\infty} = 0.8$ m/s

Figure 6-13 Mean flow field around NACA0012 aerofoil with plasma at different freestream velocities.

Transient analysis of the flow field with plasma was implemented to obtain a deeper understanding of how the steady wall-normal plasma jet interacts with the flow around NACA 0012 aerofoil. Figure 6-14 shows the instantaneous flow fields around aerofoil after plasma is on at angle of attack of 8° and freestream velocity of 0.8 m/s. At $t^+ = tU_\infty/c = 0$, the moment plasma is just about to be on, flow separates from the suction surface of aerofoil. And the shear layer, which could be indicated by concentrated negative vorticity, is shifted away from aerofoil surface, as shown in Figure 6-14 (a). After plasma is on, two starting vortices appear around the exposed electrodes on both suction and pressure sides in Figure 6-14 (b), then the vortices develop and become stronger, indicated by a higher vorticity magnitude at $t^+ = 0.12$, in the meanwhile, the vortices travel away from aerofoil surface. Afterwards, the positive vorticity created around the lower surface goes to the flow field above the upper surface (Figure 6-14 (e, f)), which may indicate the wall-normal jet due to the collision between the two wall jets is formed as the starting vortices travel away. The starting process is finished at $t^+ = 0.4$. This wall-normal plasma jet is bent towards the freestream direction and blown into the flow field above the suction surface. After the starting process, a “quasi-steady” state is developed. As can be seen in Figure 6-14 (g), (h) and (i), there is always a recirculation region behind the jet, consistent with the observations of jet in crossflow at velocity ratio of 0.5 by Gopalan et al. (2004). In addition, some vortical structures shed from the recirculation region into the downstream flow field, thus mixing could be promoted and high-momentum fluids could be introduced into the boundary layer.



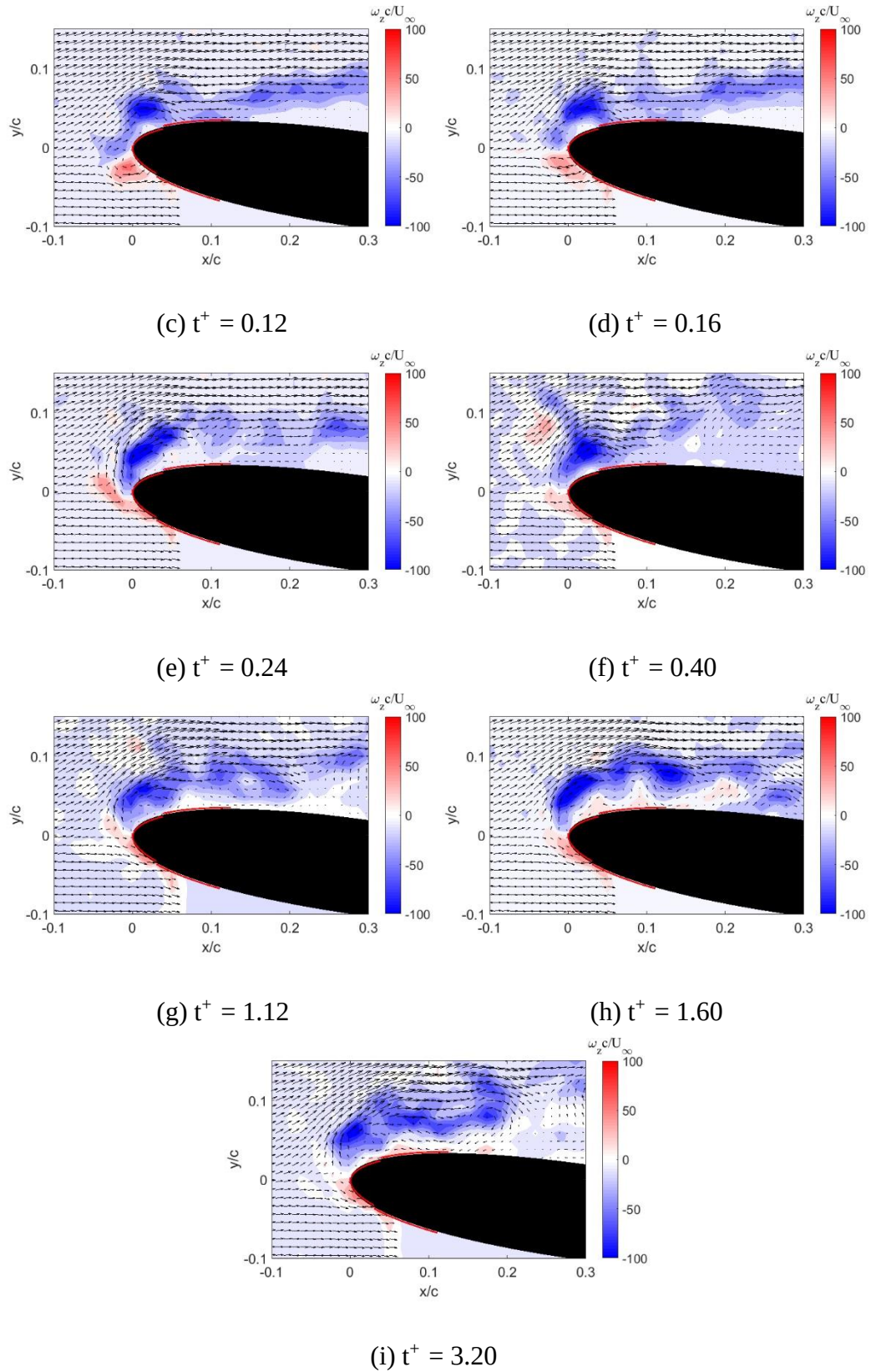


Figure 6-14 Instantaneous flow field with plasma at angle of attack of 8° . The freestream velocity is 0.8m/s. $t^+ = 0$ is the moment when plasma is just about to be on.

On the basis of the above analysis, the mechanism accounting for the flow separation control around a NACA 0012 aerofoil with a wall-normal plasma jet might be related to the creation of vortical structures created by the interaction between freestream and the wall-normal plasma jet. The vortical structures could promote mixing, thus high-momentum freestream fluids could be transported into the boundary layer, enabling the boundary layer to withstand the adverse pressure gradient. This is a different mechanism from the direct momentum injection by downstream-directed wall jets, which was used in previous researches to control the flow separation around a bluff body.

7 Flow Separation Control with Unsteady Actuation of DBD Plasma Actuator

7.1 Introduction

Periodic excitation was proved to be more effective in flow separation control than steady excitation (Wu et al. 1991; Greenblatt & Wygnanski 2000; Seifert et al. 1993; Seifert et al. 1996; He et al. 2009). Unsteady actuation always involves an oscillatory momentum addition to flow while momentum is added continuously with steady actuation. The excitation amplifies the flow instability, causing some large coherent structures to form which entrain high-momentum fluid towards the wall and energize the flow so that it's capable of withstanding the adverse pressure gradients (Seifert et al. 1993).

Excitation was first introduced as a flow control technique by Schubauer and Skramstad (1947) who utilized periodic perturbations to initiate the instability in a laminar boundary layer. This sheds a light on laminar flow separation control. Periodic excitation could trigger transition to turbulence, which is more capable of withstanding adverse pressure gradient, therefore laminar separation can be controlled by initiating transition (Mangiarotty 1989; Hans U. Meier, Alois Maier 1991; Greenblatt & Wygnanski 2000). Turbulent separation control could also be controlled by periodic excitation as large coherent structures are produced when periodic excitation is applied, and these coherent structures would promote mixing between the high-momentum fluids and the near-wall flow (Oster et al. 1978; Ho & Huang 2006; Browand & Ho 1983).

Flow separation control was achieved via addition of momentum to the boundary layer near the aerofoil surface until 1960s (Greenblatt and Wygnanski, 2000), however, in the year of 1975, it was demonstrated that separated flow could also be reattached to the aerofoil surface through acoustic excitation without momentum injection (Collins and Zelenevitz, 1975). After that, a great number of researches have been conducted on the effects of acoustic excitation on flow separation and it was proved that acoustic excitation is effective over a variety of aerofoils and even flat plates at a range of frequencies (Langley 2016; Ahuja et al. 1983; Ahuja & Burrin 1984; Marchman J. F. et al. 1987; Nishioka et al. 1990; Zaman 1992).

Apart from acoustic excitation, unsteady actuation is also applied via mechanical blowing, suction, synthetic jet et al. As a promising flow control technique, DBD plasma actuators are electronic devices that respond very fast and could be flush-mounted to the aerofoil surface without any damages to the original profiles. Moreover, wall-normal jet could penetrate the separated flow easily and transfer high-momentum fluid into the near-surface region. So the present study investigated the effect of periodic activation of a linear plasma synthetic jet actuator on flow separation control and the corresponding mechanism.

7.2 Separation control at varying angles of attack

Time-resolved PIV measurements were conducted to investigate how the unsteady actuation of DBD plasma actuator affects the separated flow around a NACA 0012 aerofoil at stall, light post-stall, moderate post-stall and deep post-stall angles of attack. The sampling frequency of PIV system is 2,000 Hz. The freestream velocity is 3m/s and the corresponding chord-based Reynolds number is 20,000. Turbulence intensity at this freestream velocity is less than 0.3%. DBD plasma actuator, which consists of two exposed electrodes and one embedded electrode, is positioned at the leading edge of the NACA 0012 aerofoil.

Unsteady actuation of DBD plasma actuator is achieved by low-frequency pulsed modulation of the high-frequency sinusoidal voltage signal (see Figure 7-1), causing the plasma actuator to switch on and off alternately at the pulse frequency and duty cycle. In the present study, a non-dimensional frequency of $f^+ = f_e / f_0 = 1$ was used at varying angles of attack of 10° , 12° , 14° and 16° to investigate the plasma separation control effect on the flow field around NACA 0012 aerofoil at different angles of attack. Here, f_e is the excitation forcing frequency and f_0 is the estimated natural vortex shedding frequency. At the post-stall angles of attack, the separated flow around an aerofoil behaves like a bluff-body flow and the natural vortex shedding frequency corresponds to a Strouhal number of $St = f_0 d / U_\infty \approx 0.2$, where U_∞ is the freestream velocity and d is length of normal projection of aerofoil on cross-stream plane (Yarusevych et al., 2009). The duty cycle of the modulation pulse is 10% and the corresponding momentum coefficient is $C_\mu = 0.025\%$.

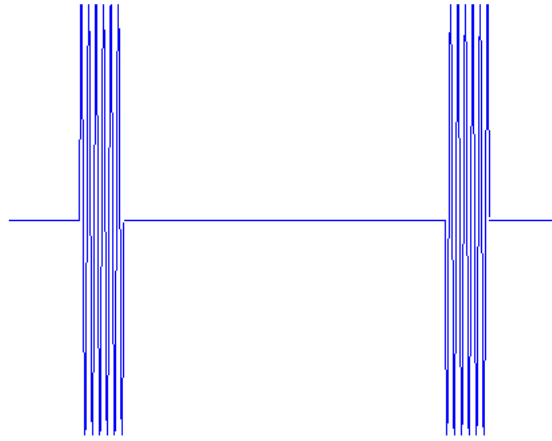
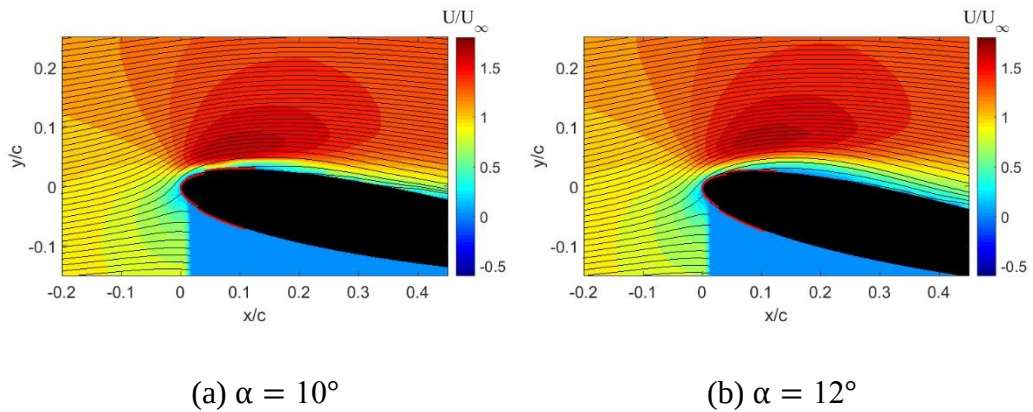


Figure 7-1 Waveform of unsteady actuation voltage signal.

Figure 7-2 presents the normalized streamwise velocity contours superimposed by streamlines of the time-averaged flow fields around a NACA 0012 aerofoil with periodic activation of plasma actuator at Reynolds number of 20,000 and varying angles of attack. With plasma excitation at the natural vortex shedding frequency, namely, $f^+ = f_e/f_0 = 1$, the recirculation region disappears and the separated flow reattaches to the aerofoil surface at angles of attack of 10° , 12° and 14° , in other words, flow separation is effectively controlled with plasma excitations. It's worth mentioning that steady actuation of plasma actuator at momentum coefficient of $C_\mu = 0.25\%$ could only alleviate flow separation around aerofoil but can not reattach the separated flow at the angle of attack of 14° (discussed in Section 6.2). With unsteady actuation of plasma actuator, separation completely disappears at post-stall angle of attack of 14° with only 10% energy consumption of that for steady actuation, demonstrating a more effective and economical way of leading-edge flow separation control around an aerofoil. Similar finding was also observed by He et al. (2009b), who performed a study over a NACA 0015 aerofoil with plasma actuator through aerodynamic force measurements and demonstrated that unsteady actuation of plasma actuator at the leading-edge could delay the stall angle of attack to 22° at Reynolds number of 217,000 while steady actuation could only postpone the stall angle of attack to 18° . Meanwhile, the power consumption for the unsteady actuation case was also 10% that of steady actuation in their study, demonstrating a better flow separation control effect with periodic excitations as

well. Moreover, an enhanced-velocity region appears over the upper surface within the chordwise range of $0 \leq x/c \leq 0.3$, which has a stronger capability of withstanding the adverse pressure gradient than the baseline cases. According to Sato et al. (2015), this accelerated flow around the leading edge leads to a suction peak, which contributes to a higher lift coefficient C_L and smaller drag coefficient C_D .

At post-stall angle of attack of 16° , separation is attenuated but still exists above the suction surface of aerofoil with unsteady actuation of plasma actuators (Figure 7-2 (d)), therefore, the limitation of flow separation control around NACA 0012 aerofoil with unsteady actuation of plasma actuator at Reynolds number of 20,000 is $\alpha = 16^\circ$, two degrees higher than that with steady actuation of plasma actuator, moreover, the energy consumption is only ten percent of that with steady actuation. Although flow separation is not entirely eliminated with excitations of plasma at angle of attack of 16° , an enhanced-velocity region still appears above the upper surface. However, it seems that the magnitude of the enhanced-velocity is smaller than that at smaller angles of attack and the enhanced-velocity region locates at a higher vertical position from aerofoil surface due to the existence of reverse flow at angle of attack of 16° .



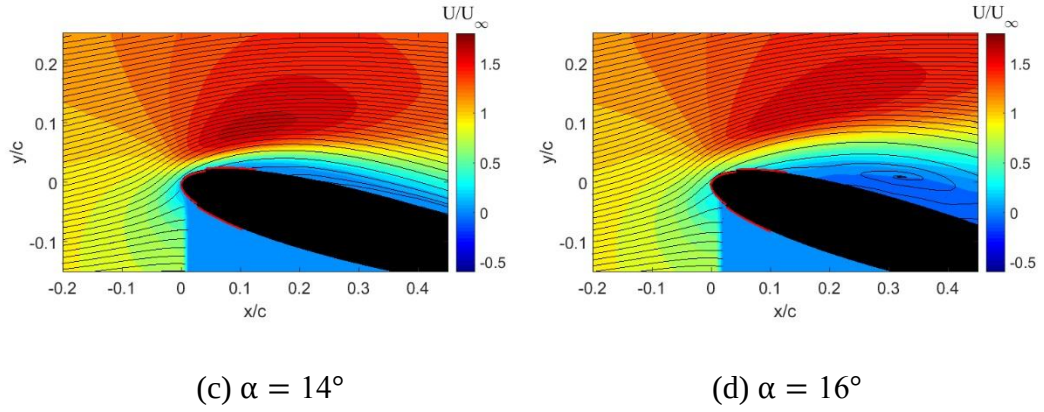
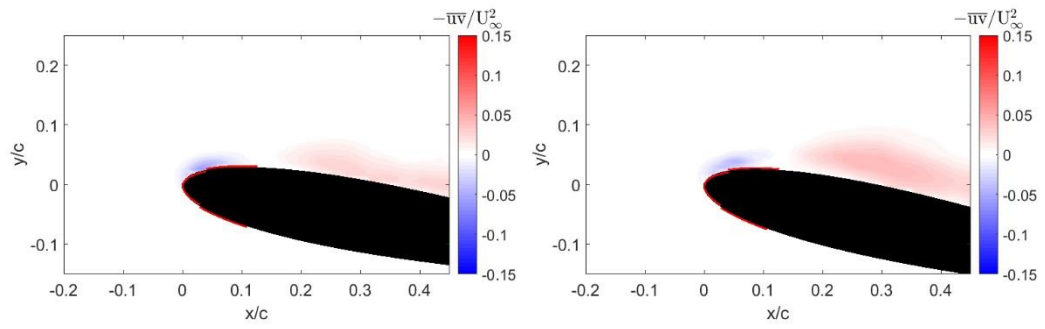


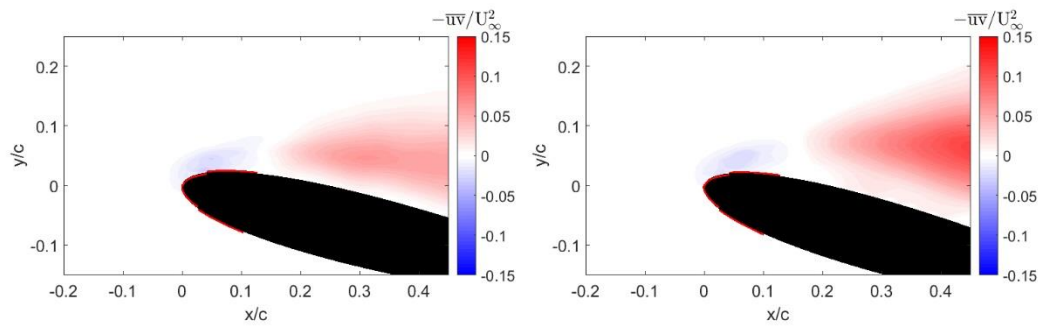
Figure 7-2 Normalized streamwise velocity contour superimposed by streamline of the time-averaged flow field around a NACA 0012 with plasma excitations at a frequency of $f^+ = 1$ and duty cycle of 10%. The freestream velocity is $U_\infty = 3\text{m/s}$ and the corresponding chord-based Reynolds number is 20,000.

Figure 7-3 illustrates the normalized Reynolds stress contours for the controlled flow fields with oscillatory excitations of plasma at various angles of attack of $\alpha=10^\circ, 12^\circ, 14^\circ$ and 16° and freestream velocity of $U_\infty = 3\text{m/s}$. Here, the excitation frequency is $f^+ = 1$ and duty cycle is 10%, the corresponding momentum coefficient is 0.025%. For the baseline cases, the concentrated Reynolds stress is located away from the aerofoil surface as shown in Figure 4-4, which might be caused by the separated shear layer and shedding vortices. With unsteady addition of wall-normal plasma jet momentum at a normalized momentum coefficient of $C_\mu = 0.025\%$, high Reynolds stress region attaches to the upper surface of aerofoil at angles of attack of $10^\circ, 12^\circ$ and 14° , indicating that momentum exchange occurs in the near-surface region, see Figure 7-3 (a ~ c), therefore high-momentum fluid could be transported into the boundary layer, enabling it to overcome the adverse pressure gradient and controlling the flow separation around the aerofoil. At angle of attack of 16° , the concentrated Reynolds stress region is away from aerofoil surface, however, it is closer to the suction surface compared with the baseline case, corresponding to an attenuation of flow separation. In the meanwhile, the Reynolds stress magnitude is amplified on the application of plasma excitations relative to the baseline cases, indicating a stronger momentum transfer effect, both the greater magnitude and lower position of Reynolds stress are beneficial for flow separation control around aerofoil at post-stall angles of attack.



(a) $\alpha = 10^\circ$

(b) $\alpha = 12^\circ$

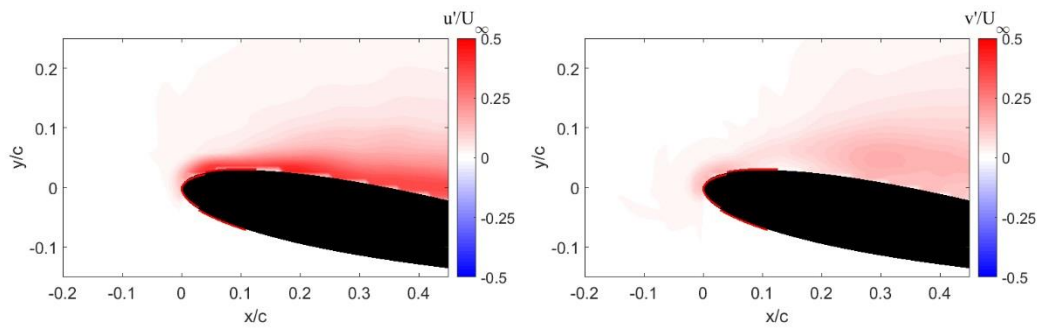


(c) $\alpha = 14^\circ$

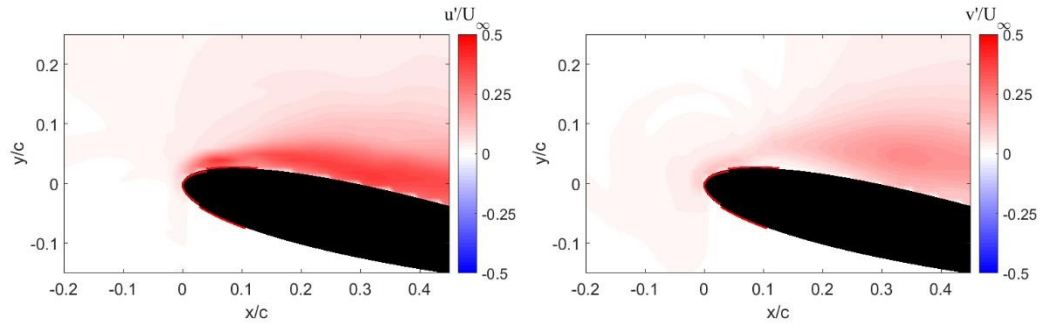
(d) $\alpha = 16^\circ$

Figure 7-3 Normalized Reynolds stress contour for the controlled flow fields around NACA 0012 aerofoil with unsteady actuation of DBD plasma actuator at different angles of attack.

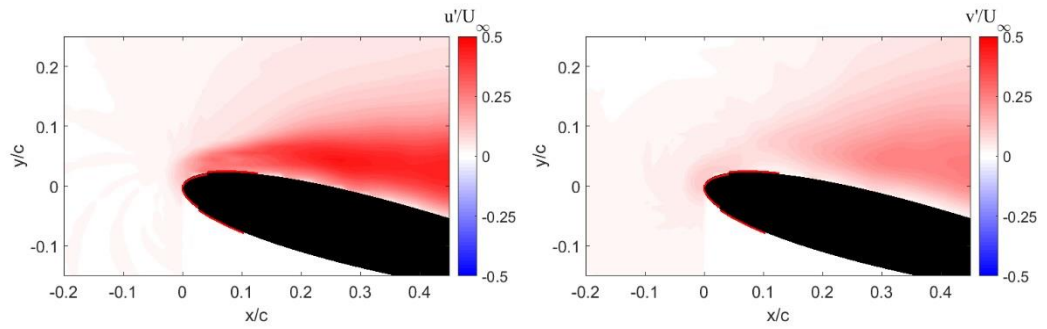
a) $\alpha = 10^\circ$



b) $\alpha = 12^\circ$



c) $\alpha = 14^\circ$



d) $\alpha = 16^\circ$

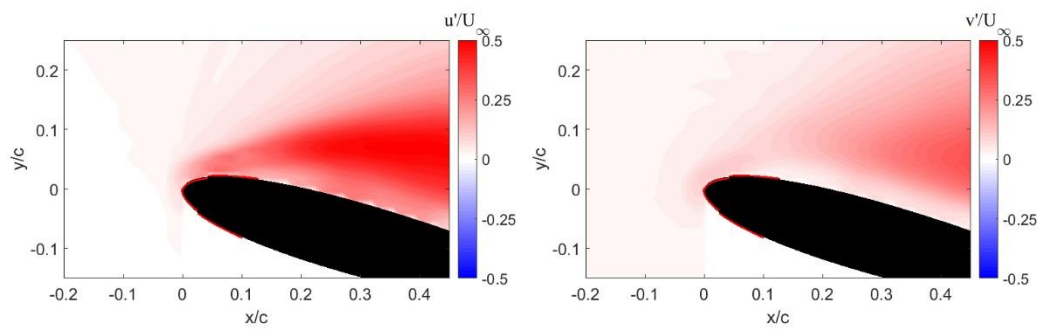


Figure 7-4 Contours of streamwise (left column) and vertical (right column) root-mean-square velocities with unsteady activation of plasma at different angles of attack.

R.M.S. streamwise and vertical velocity fluctuations above the upper surface of aerofoil were calculated and presented in Figure 7-4. For the baseline case, the maximum r.m.s. velocity locates away from the aerofoil, and the magnitude of the r.m.s. velocity peak increases with the chordwise distance (see Figure 4-5), similar to the previous findings (Boutillier & Yarusevych 2012; Kiya & Sasaki 1983). With periodic plasma excitations, the maximum r.m.s. velocity fluctuations locate much closer to the suction surface of aerofoil, which is due to that the shear layer is moved closer to the wall as a result of flow separation control. An amplification of rms velocity is observed on the application of plasma excitations relative to the baseline case, which indicates that flow instability is amplified at excitation frequency of $f^+ = 1$. Moreover, a stronger amplification is developed at higher angles of attack, suggesting it is easier for excitation to amplify in a severer adverse pressure gradient. The r.m.s. streamwise velocity is always larger than the r.m.s. vertical velocity.

Phase-averaged analysis was conducted to investigate the flow behaviours with unsteady actuation of DBD plasma actuator. Figure 7-5 shows the phase-averaged streamlines, vertical velocity contours and spanwise vorticity contours superimposed by swirling strength isolines for the controlled cases with periodic activation of plasma actuator at angle of attack of 10° . Phase angle of $\varphi = 0^\circ$ refers to the start of each plasma pulse, in other words, plasma is about to be on at phase angle $\varphi = 0^\circ$, and lasts until $\varphi = 36^\circ$ with a duty cycle of 10%. At phase angle of $\varphi = 0^\circ$, flow is attached to the aerofoil surface, which can be clearly seen from the streamlines in Figure 7-5 (a). After plasma is turned on, a break in spanwise vorticity is observed in Figure 7-5 (b), this is probably because of a discontinuity between the re-energized boundary layer by plasma and the original boundary layer. Then the old boundary layer rolls up into a vortex and travels downstream (Figure 7-5 (c), (d) and (e)). The positive and negative vertical velocities around the vortex indicate an entrainment effect by this vortex. The “starting vortex” formed on the initiation of an actuator in flow separation control studies was also observed by other researchers (Brzozowski et al., 2010; Darabi and Wygnanski, 2004; Siau and Bonnet, 2017; Woo and Glezer, 2013). In Figure 7-5 (e), a second vortex is formed at $x/c \approx 0.28$, which could be indicated by both the spanwise vorticity contour and swirling strength isolines. This is in very good agreement with previous

findings of flow separation control over a circular cylinder by Jukes and Choi (2009). They found a vortex is created due to the break of the re-energized boundary layer from its upstream boundary layer after the end of an excitation. Darabi and Wygnanski (2004) did researches on the control of separated flow over a solid surface, who also found a vortex located upstream of the “starting vortex” over a forcing cycle. They believed this vortex is created through the concentration of vorticity by actuation of an actuator. In the current study, the formation of the second vortex occurs around 144° after plasma is turned off, so there is a “phase lag” here because it takes time for the vortex to form. Flow is always attached to aerofoil surface over the entire plasma forcing cycle, contributing to an attached flow field in time-averaged sense as shown in Figure 7-2 (a).

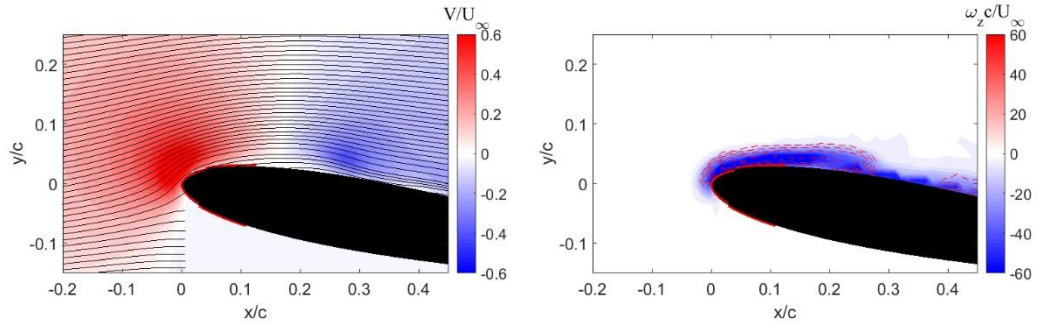
At angle of attack of 12° , as the adverse pressure gradient is severer, flow transiently separates from the upper surface with plasma excitations at frequency of $f^+ = 1$ and duty cycle of 10% although flow separation is eliminated in a time-averaged sense. This is demonstrated in Figure 7-6 which shows the phase-averaged vertical velocity, streamline, spanwise vorticity and swirling strength λ_{ci} . At phase angle of $\varphi = 0^\circ$, as can be seen from the streamline, flow separates and shear layer is shifted away from the aerofoil surface due to a heavier adverse pressure gradient at angle of attack of 12° . After plasma is turned on, the vorticity above the upper surface is fragmented into two segments, in other words, the separated vorticity region and the attached vorticity layer are created on both sides of a break in Figure 7-6 (b). The separated vorticity sheds from the attached vorticity layer and the tail of the separated shear layer rolls up into a vortex, which is shown in Figure 7-6 (b) and (c). The spanwise vorticity corresponding to this vortex is stronger than that at angle of attack of 10° , indicating a stronger vortex. In addition, another break is formed at phase 90° , inducing the formation of another vortex at phase 135° , which seems to be slightly larger than that formed at angle of attack of 10° . This second vortex is much weaker than the one created on the start of plasma, accordingly it can not be observed from the streamline because it is submerged in the mean flow, however, it could be identified by the swirling strength and the spanwise vorticity. The second vortex appears in Figure 7-6 (c), (d) and (e), and an attached boundary layer is always developed upstream of it.

After the vortices travel away, separation control effect is lost and separation starts to appear again at $\varphi = 315^\circ$.

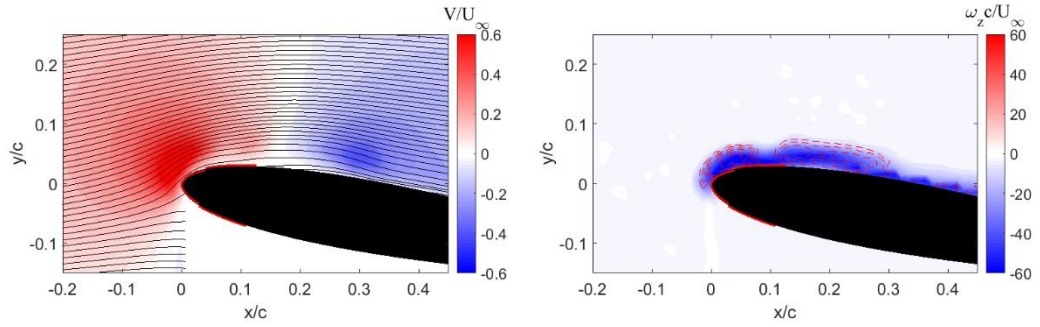
When angle of attack increases to 14° , as shown in Figure 7-7, a similar phenomenon is observed over the entire cycle with plasma excitations as that at angle of attack of 12° . However, the two spanwise vortices generated over a forcing cycle are larger in size. At higher angles of attack, the adverse pressure gradient is severer, kinetic energy of the boundary layer will be lost in order to move against the adverse pressure gradient, therefore, the near-wall fluid is weaker in momentum at a higher angle of attack. Low-momentum fluid is easier to be entrained, so larger vortices are formed at higher angles of attack over the cycle. In the meanwhile, the vortices are located at a higher vertical position from the suction surface relative to those at smaller angles of attack. At phase angle of $\varphi = 0^\circ$, flow separates more heavily compared with that at smaller angles of attack, indicated by a recirculation region with a larger size and a separated shear layer shifted further away from aerofoil surface. Accordingly, the vortex due to evacuation of the separated shear layer is located at a vertical position further away from the upper surface. As demonstrated in 6.5, vorticity is added to the flow via the interaction between freestream and the wall-normal plasma jet, a concentrated vorticity layer is formed upstream of the “starting vortex”, shown in Figure 7-7 (b), (c) and (d). After plasma is turned off, the concentrated vorticity is released and forms another vortex at phase 72° in Figure 7-7 (g). Entrainment was induced by these vortices, and the outwards transportation of the near-wall fluids results in a decrease in static pressure. Consequently, a favourable pressure gradient is caused, which together with momentum entrainment force the flow to reattach to the aerofoil surface. Therefore, flow separation disappears in Figure 7-7 (j), (k) and (l). When both vortices travel away, flow starts to separate again at $\varphi = 270^\circ$ and 315° , as shown in Figure 7-7 (m) and (n). Under this condition, flow separates over a larger portion of the plasma forcing cycle compared with that at angle of attack of 12° due to the existence of a greater adverse pressure gradient. Even flow separation appears transiently over the forcing cycle, flow separation is entirely controlled and eliminated in a time-averaged sense, shown in Figure 7-2 (c).

At deep post-stall angle of attack of 16° , the dynamics of the two clockwise vortices formed over the forcing cycle are similar to those at angle of attack of 14° . However, the vortices are located further away from aerofoil surface due to a severer adverse pressure gradient at this higher angle of attack. Flow separation occurs all over the entire cycle although the size of the reverse flow region is reduced after plasma is on, this is consistent with an alleviation but not elimination of flow separation with unsteady actuation of plasma actuator at frequency of $f^+ = 1$ and duty cycle of 10% in a time-averaged sense (Figure 7-2 (d)).

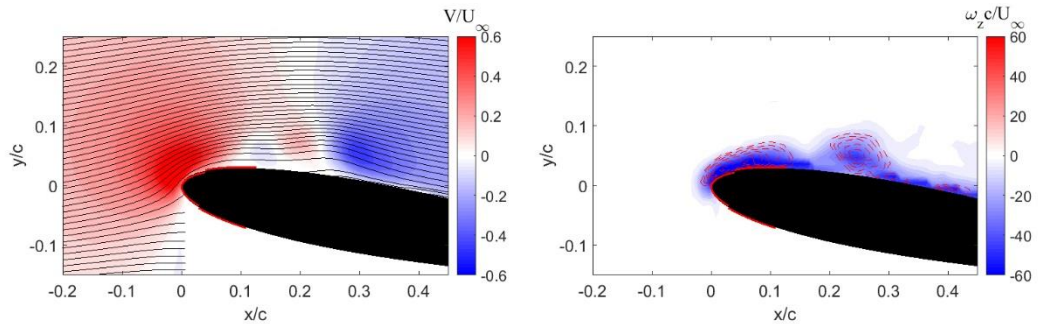
a) $\varphi = 0^\circ$



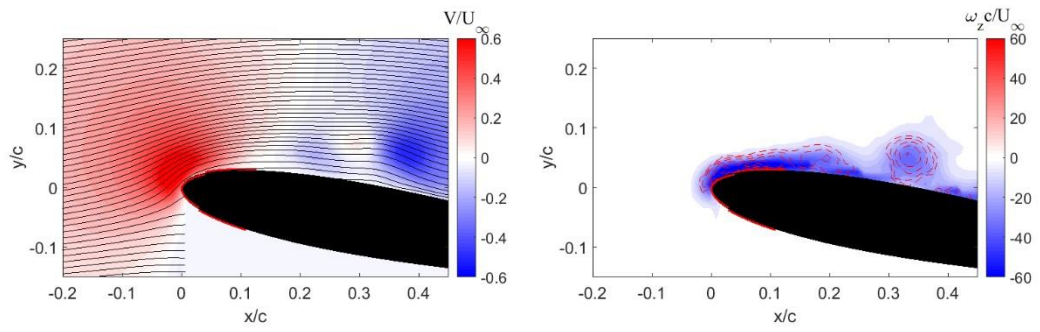
b) $\varphi = 45^\circ$



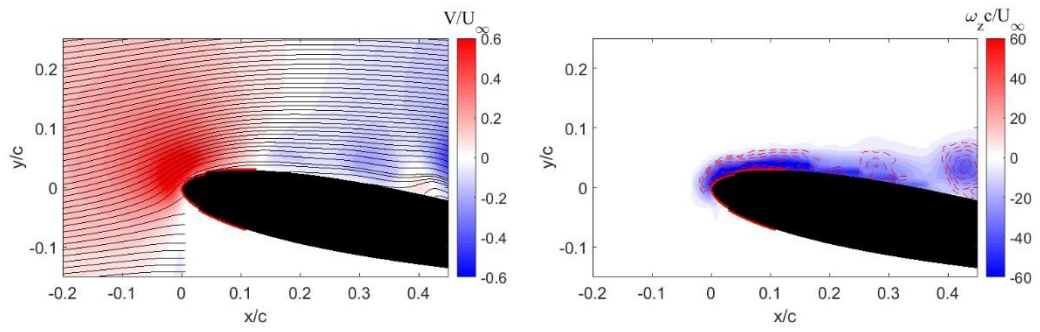
c) $\varphi = 90^\circ$



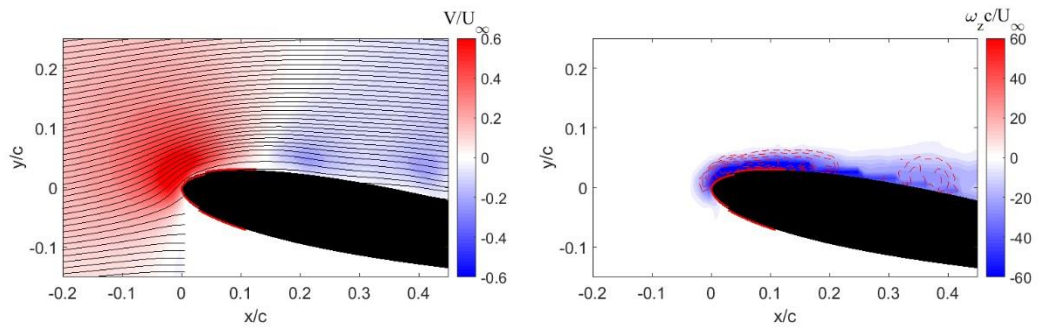
d) $\varphi = 135^\circ$



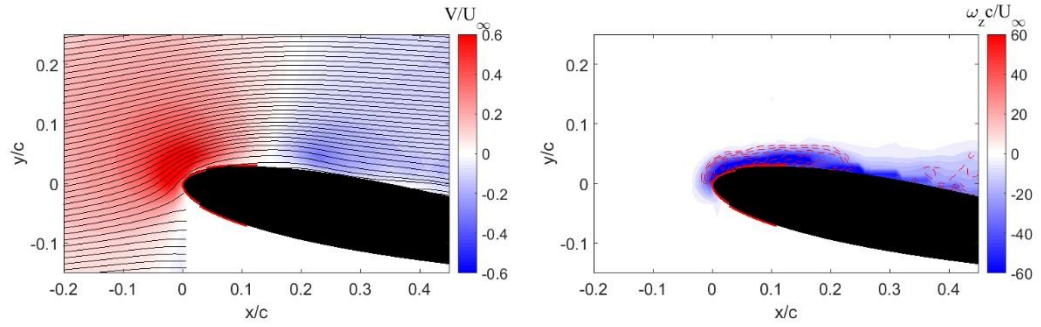
e) $\varphi = 180^\circ$



f) $\varphi = 225^\circ$



g) $\varphi = 270^\circ$



h) $\varphi = 315^\circ$

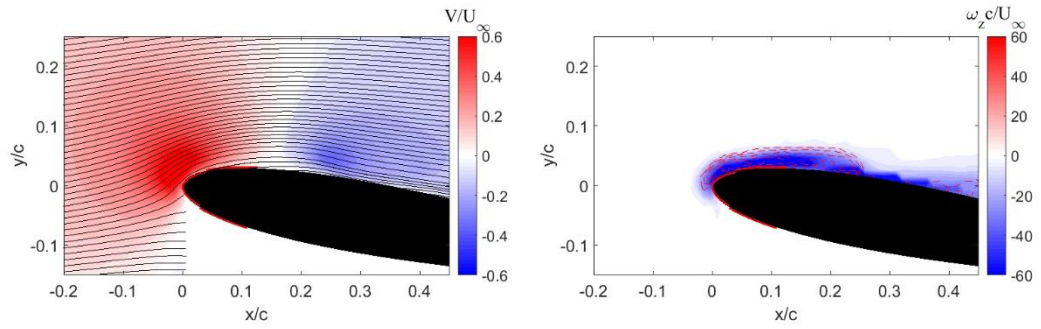
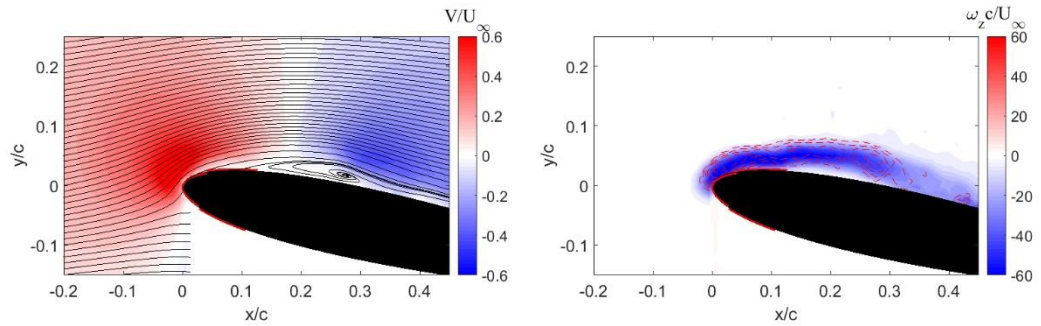
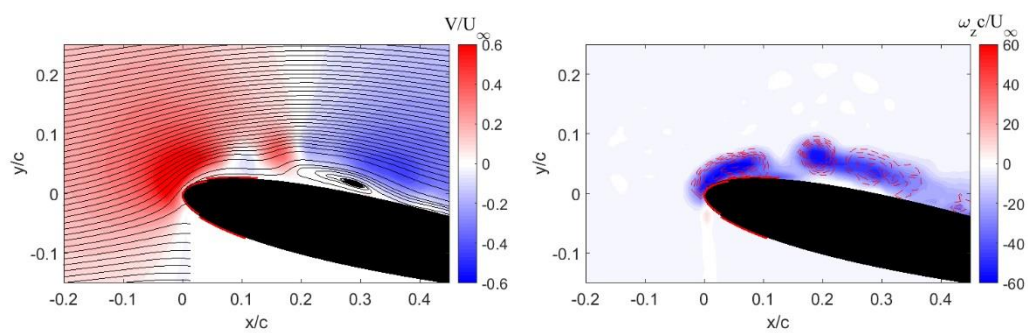


Figure 7-5 Phased-averaged analysis of the flow field around NACA 0012 aerofoil at angle of attack of $\alpha = 10^\circ$ with plasma excitations at frequency of $f^+ = 1$ and duty cycle of 10%. Plasma is on at phase 0° . Freestream velocity is $U_\infty = 3\text{m/s}$. The red dashed lines represent swirling strength isolines.

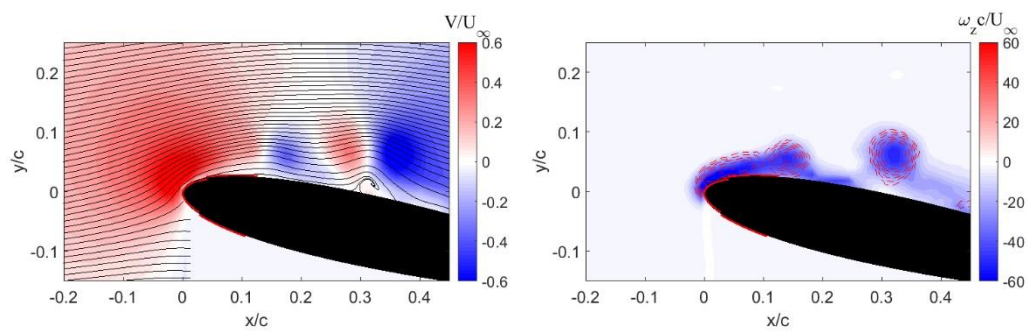
a) $\varphi = 0^\circ$



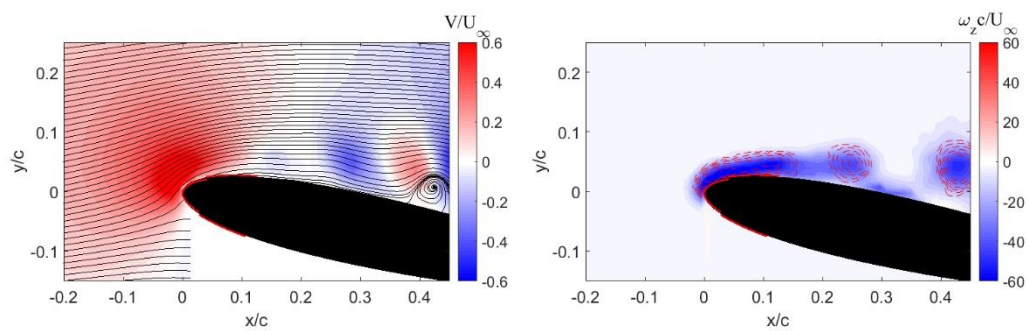
b) $\varphi = 45^\circ$



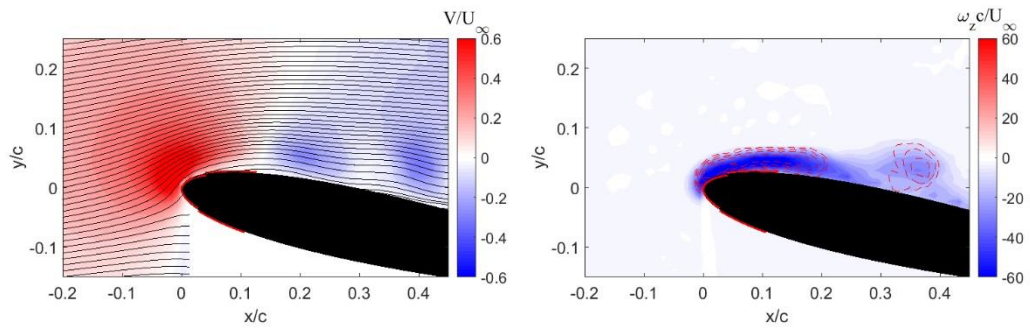
c) $\varphi = 90^\circ$



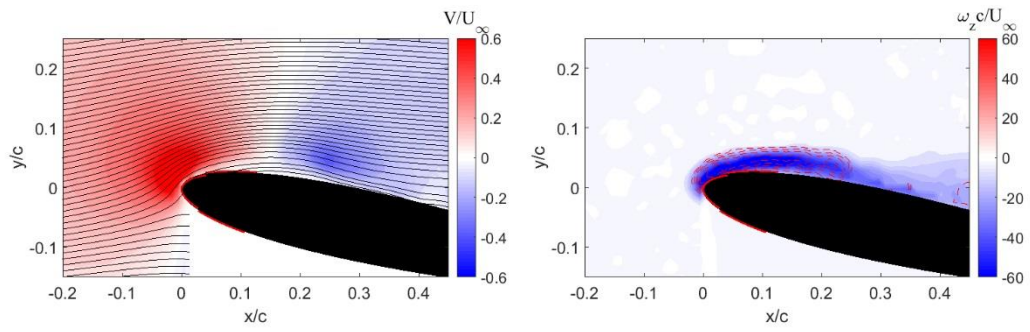
d) $\varphi = 135^\circ$



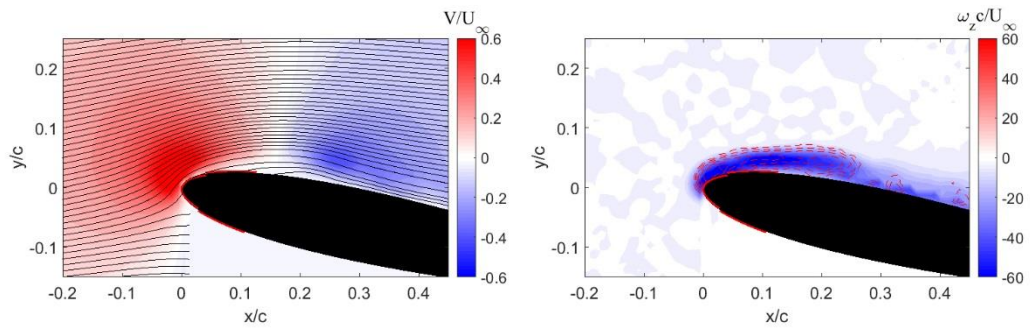
e) $\varphi = 180^\circ$



f) $\varphi = 225^\circ$



g) $\varphi = 270^\circ$



h) $\varphi = 315^\circ$

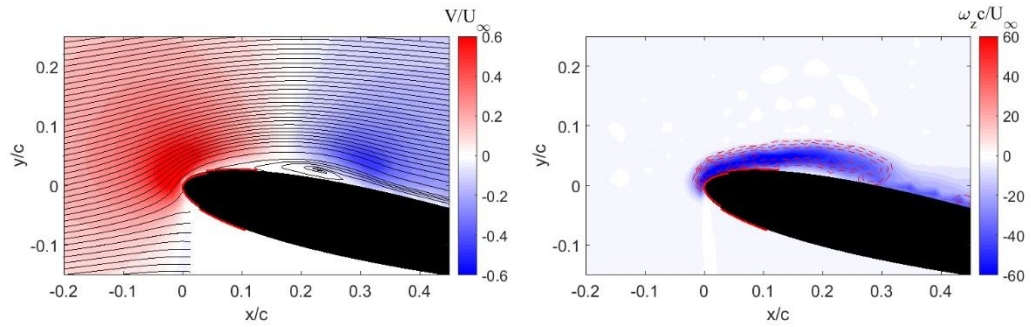
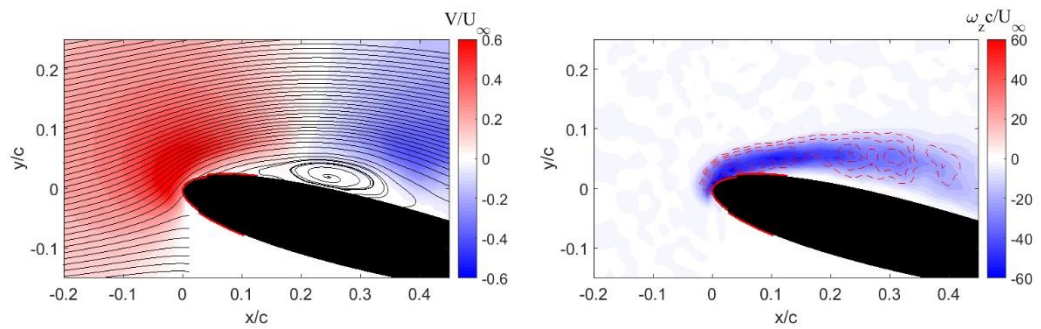
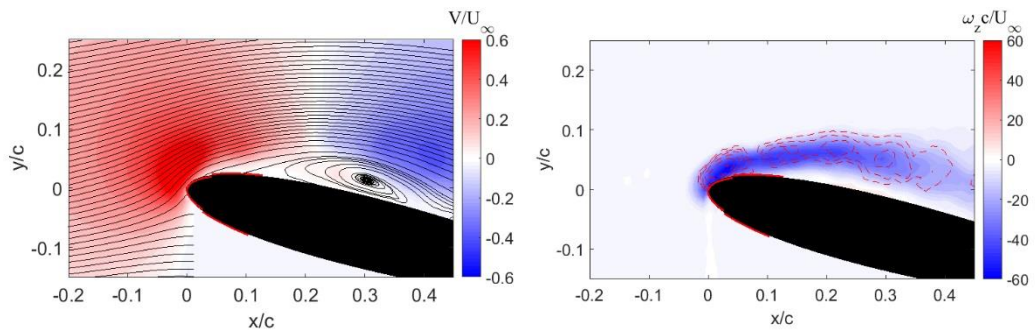


Figure 7-6 Phased-averaged analysis of the flow field around NACA 0012 aerofoil at angle of attack of $\alpha = 12^\circ$ with plasma excitations at frequency of $f^+ = 1$ and duty cycle of 10%.

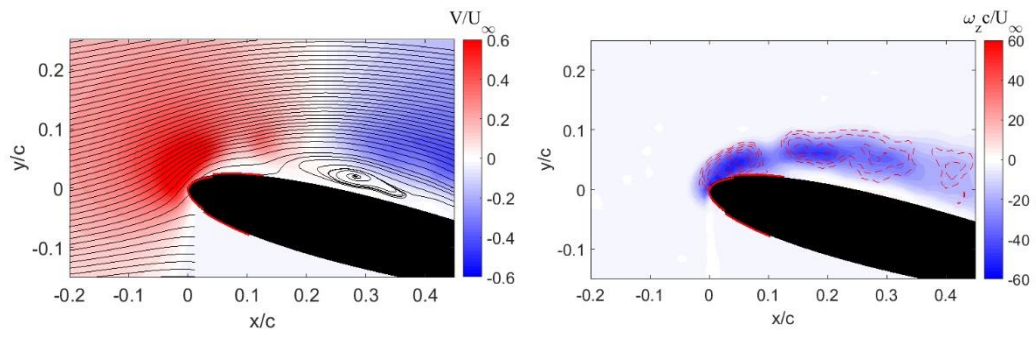
a) $\varphi = 0^\circ$



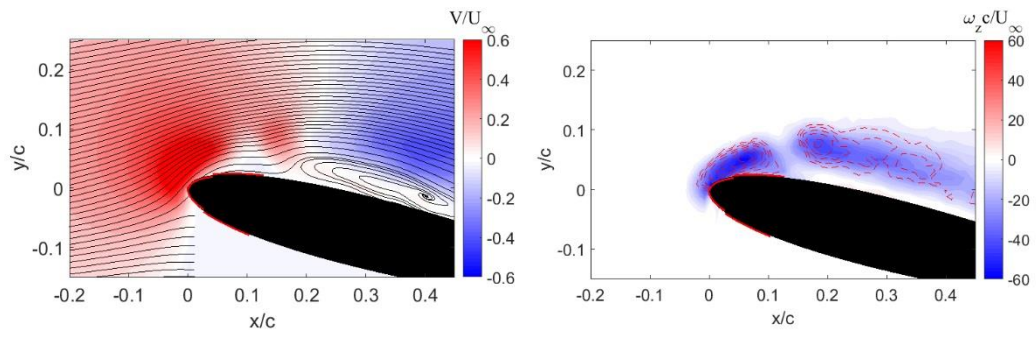
b) $\varphi = 13.5^\circ$



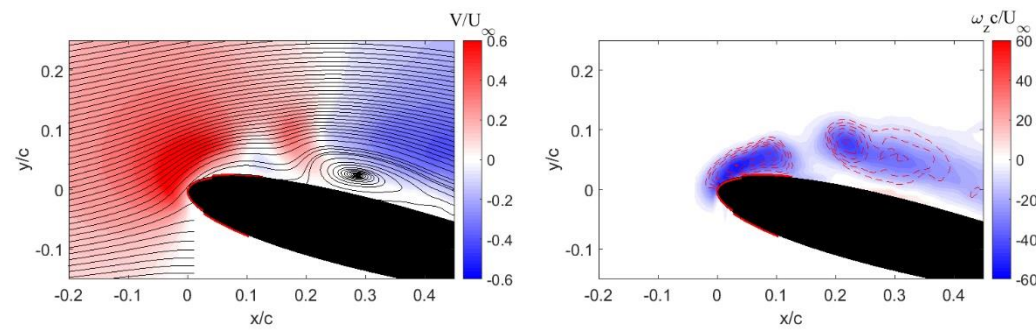
c) $\varphi = 27^\circ$



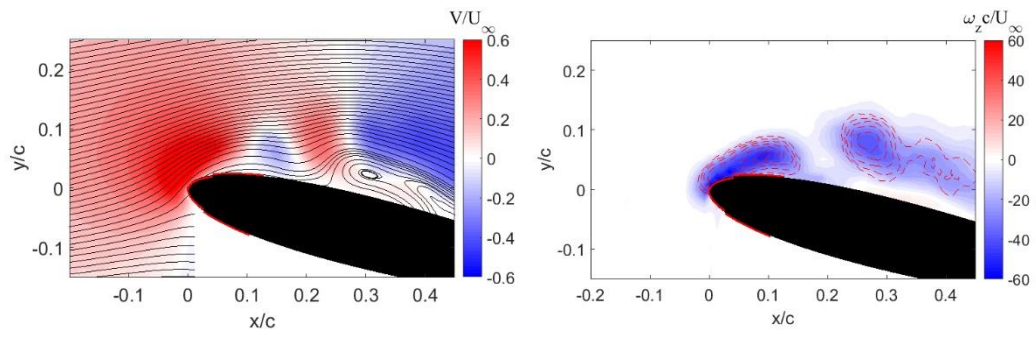
d) $\varphi = 36^\circ$



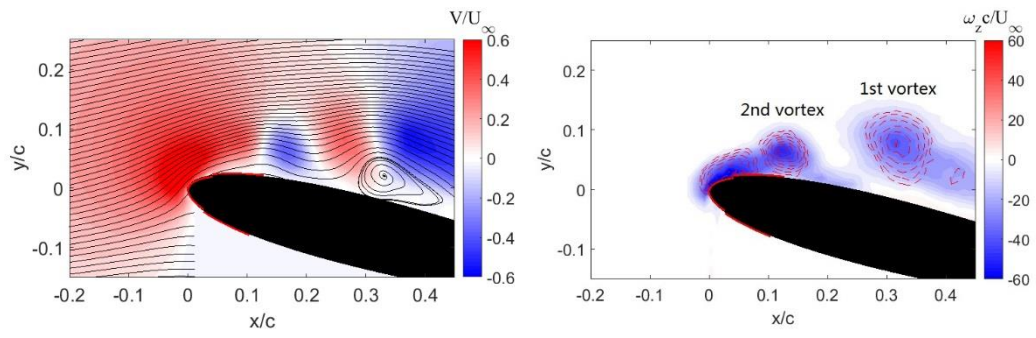
e) $\varphi = 45^\circ$



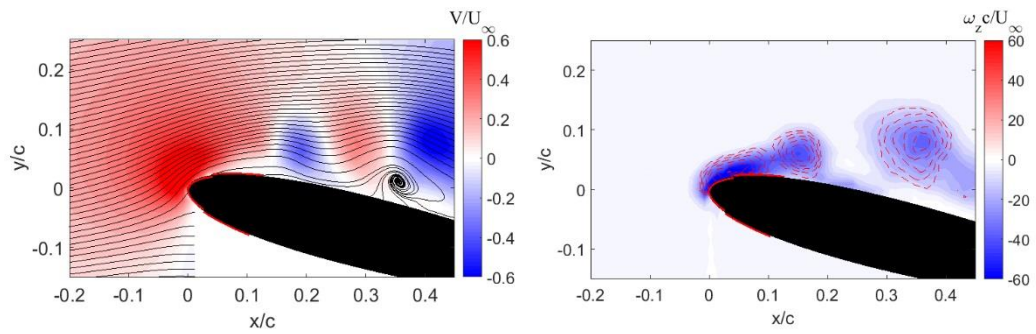
f) $\varphi = 58.5^\circ$



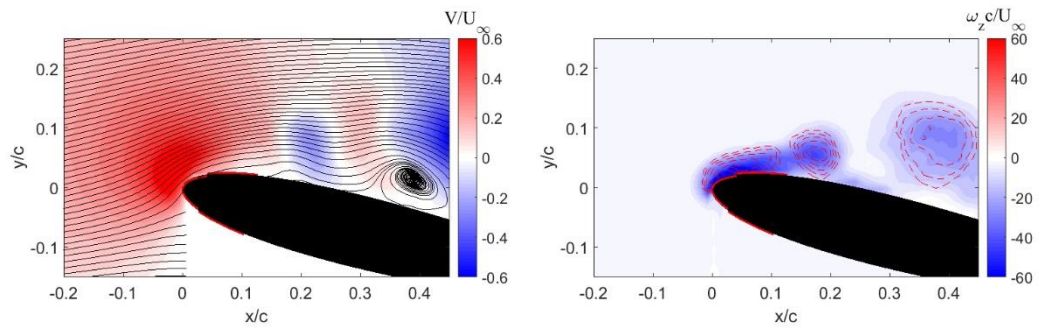
g) $\varphi = 72^\circ$



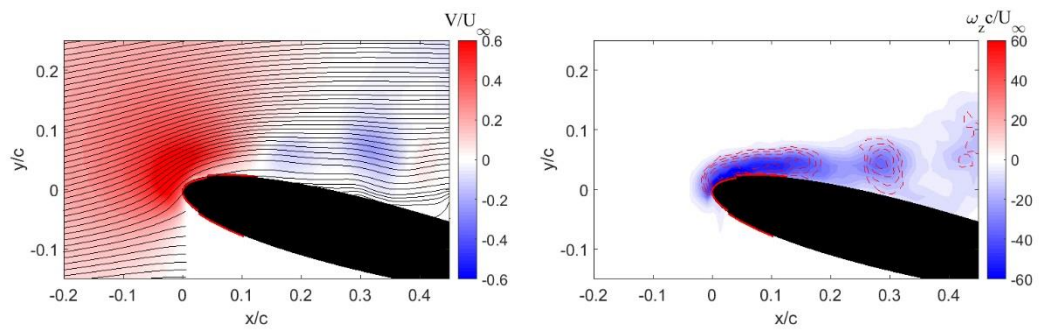
h) $\varphi = 81^\circ$



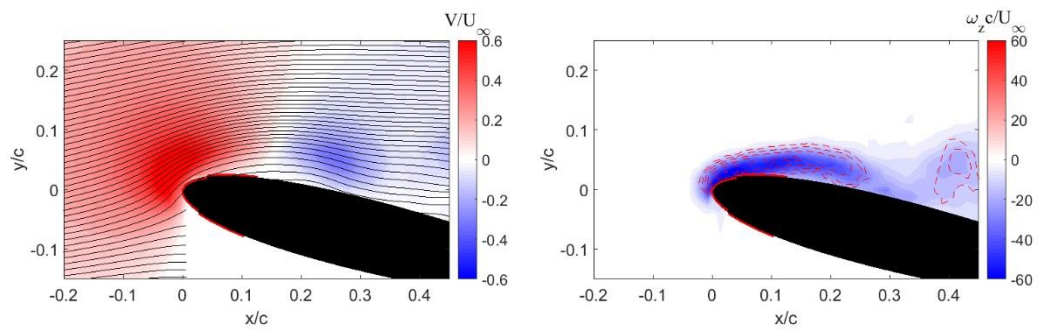
i) $\varphi = 90^\circ$



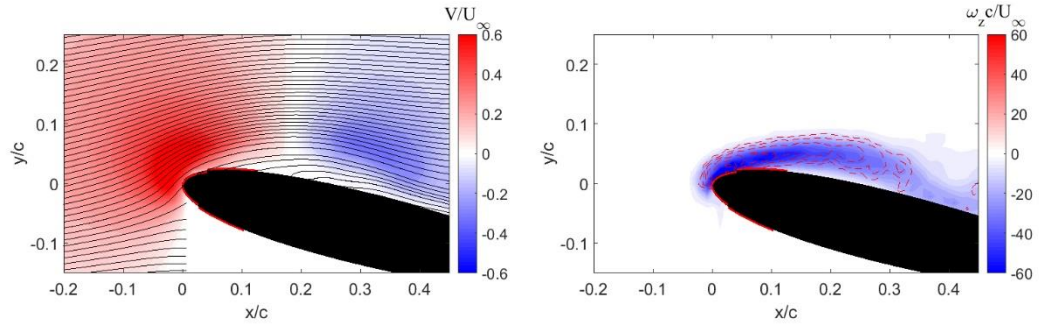
j) $\varphi = 135^\circ$



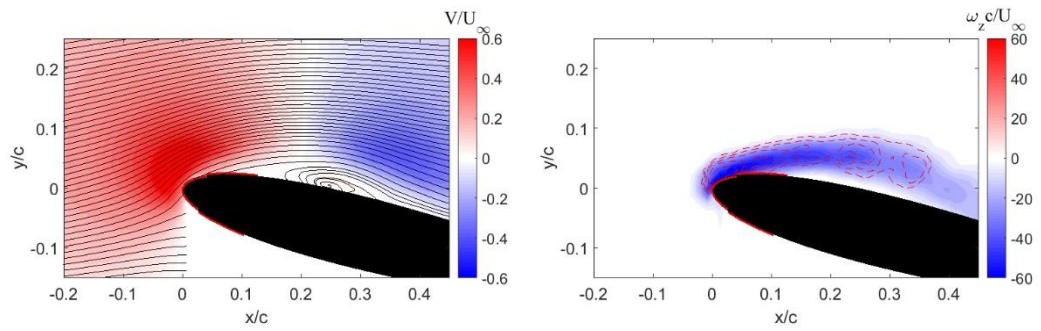
k) $\varphi = 180^\circ$



l) $\varphi = 225^\circ$



m) $\varphi = 270^\circ$



n) $\varphi = 315^\circ$

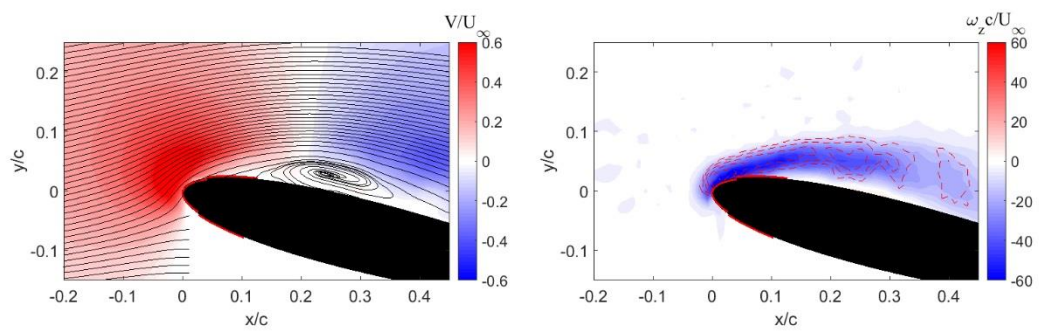
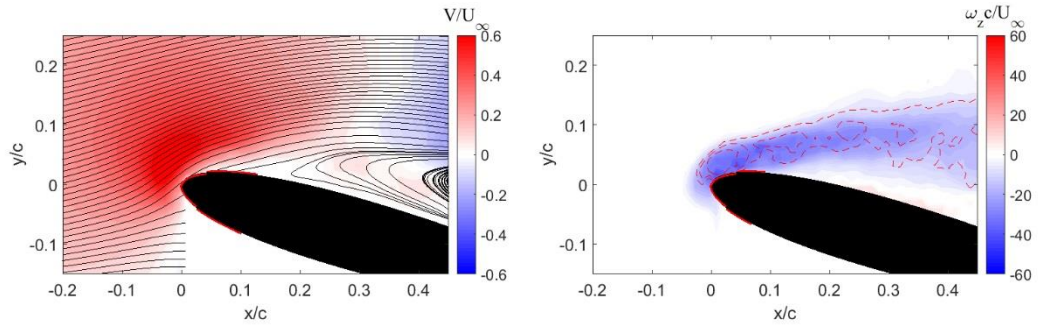
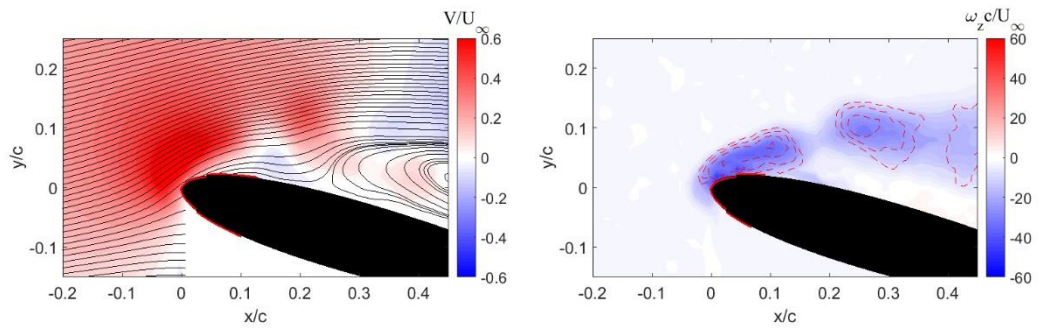


Figure 7-7 Phased-averaged analysis of the flow field around NACA 0012 aerofoil at angle of attack of $\alpha = 14^\circ$ with plasma excitations at frequency of $f^+ = 1$ and duty cycle of 10%.

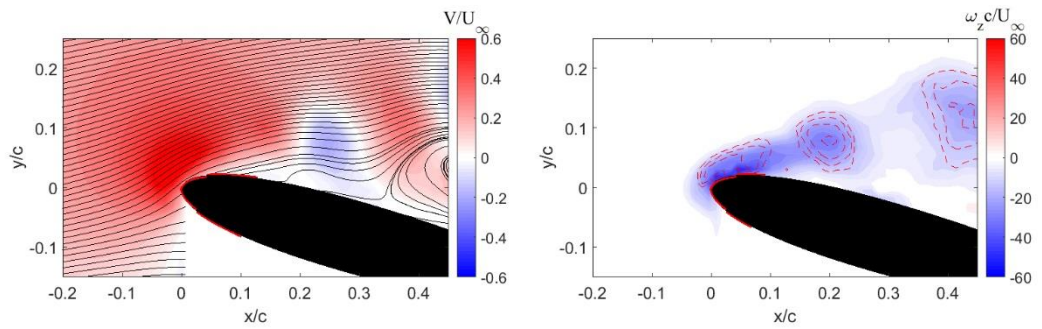
a) $\varphi = 0^\circ$



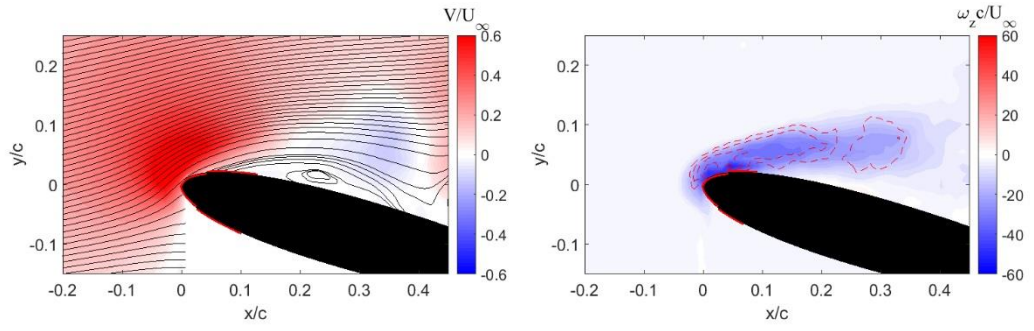
b) $\varphi = 45^\circ$



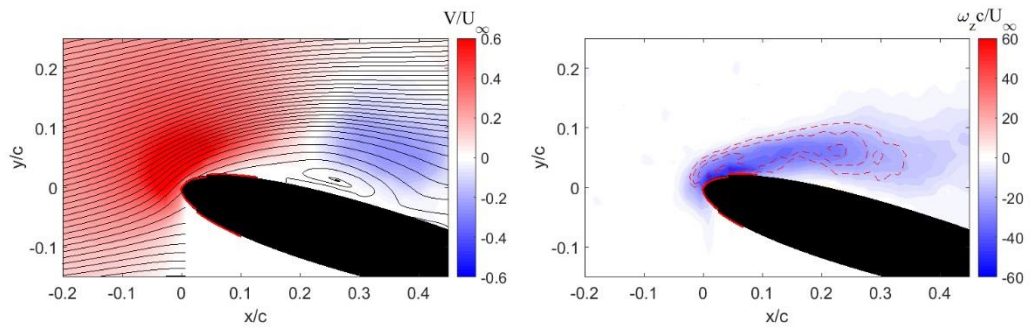
c) $\varphi = 90^\circ$



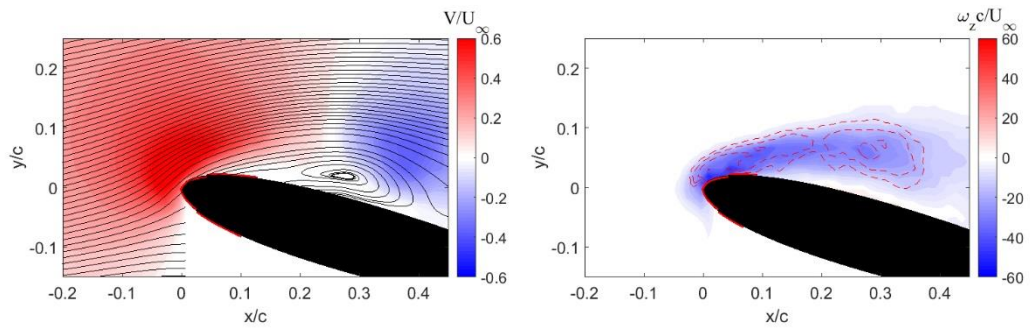
d) $\varphi = 135^\circ$



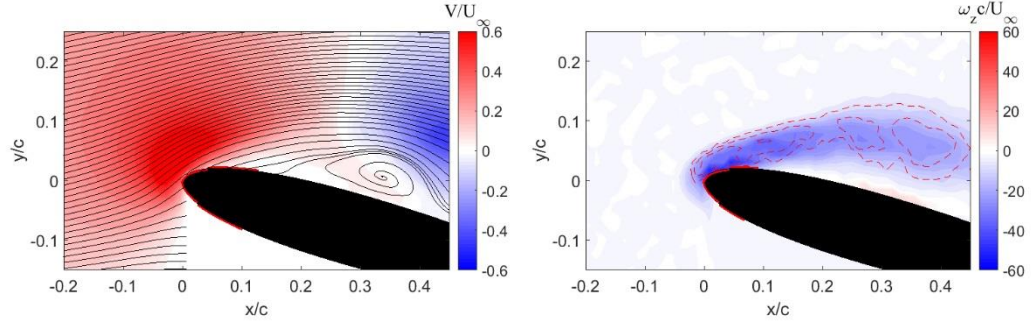
e) $\varphi = 180^\circ$



f) $\varphi = 225^\circ$



g) $\varphi = 270^\circ$



h) $\varphi = 315^\circ$

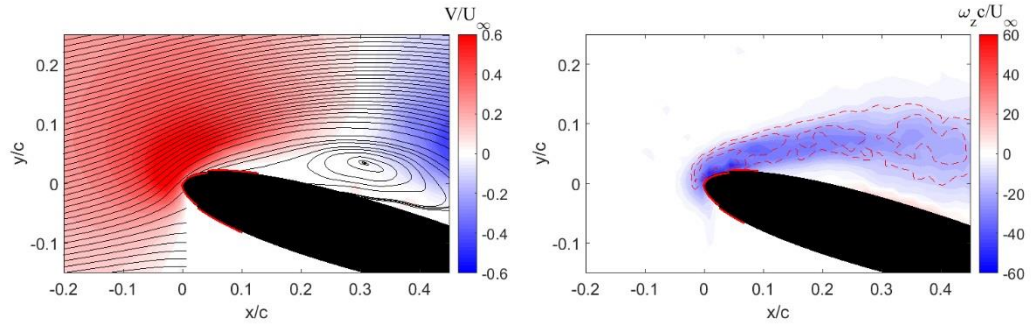


Figure 7-8 Phased-averaged analysis of the flow field around NACA 0012 aerofoil at angle of attack of $\alpha = 16^\circ$ with plasma excitations at frequency of $f^+ = 1$ and duty cycle of 10%.

7.3 Effect of modulation frequency on flow separation control

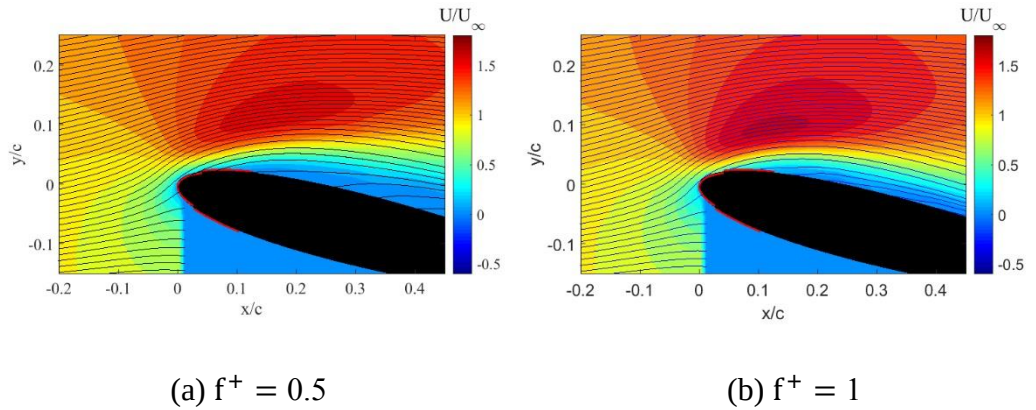
Flow separation control with unsteady actuation of linear plasma synthetic jet actuator at post-stall angle of attack of 14° and varying excitation frequencies was also investigated. PIV measurements were conducted with a sampling frequency of 2,000 Hz and freestream velocity of 3m/s, the corresponding chord-based Reynolds number is 20,000. Here, the sinusoidal voltage signal was modulated by pulses with a broad range of frequencies of $f^+ = f_e/f_0 = 0.5, 1, 5, 10, 50$ and 100 to study the effect of excitation frequency on flow separation control. The duty cycle of the pulses tested in this section is always 10% and the corresponding plasma momentum coefficient is $C_\mu = 0.025\%$.

Time-averaged streamwise velocity contour superimposed by streamlines are plotted in Figure 7-9 for the controlled cases with various excitation frequencies. When unsteady actuation of DBD plasma actuators is applied at frequencies of $f^+ = f_e/f_0 = 0.5, 1, 5, 10$ and 50 and momentum coefficient of $C_\mu = 0.025\%$, the separation bubble disappears and flow reattaches to aerofoil surface, therefore flow separation is effectively controlled with periodic activation of plasma actuator at these frequencies. This demonstrates that the separated flow is sensitive to external disturbances and could be suppressed with excitations at a wide range of frequencies, specifically, $O(0.1) \sim O(10)$. Moreover, an enhanced-velocity region shows up above the upper surface, and this region locates closer to the aerofoil surface at higher excitation frequencies within the range of $f^+ = 0.5 \sim 5$ while position of enhanced-velocity region with unsteady excitation frequency of $f^+ = 10$ and 50 are almost the same as that with excitations at a frequency of $f^+ = 5$. When plasma actuator was excited at the highest frequency tested in this study, namely, $f^+ = 100$, flow separation is alleviated relative to the baseline cases but can not be entirely eliminated, in other words, excitation at frequency of $O(100)$ is not able to reattach the separated flow to the aerofoil surface at momentum coefficient of 0.025% and Reynolds number of $20,000$. Therefore, flow separation control with plasma excitation at this frequency is not as effective as that at a smaller excitation frequency.

Streamwise velocity contours show a velocity gradient above the suction surface, which is due to a boundary layer developed in that region. A thinner boundary layer is produced when unsteady actuation is performed at frequencies of $f^+ = 5$ and 10 . As thinner boundary layer is more capable of withstanding adverse pressure gradient (Smith, 1975), thus a better flow separation control effect is achieved at these excitation frequencies. This can be confirmed by the time-averaged velocity profiles in Figure 7-10, which shows that a fuller and thinner boundary layer is developed at a higher excitation frequency within the frequency range of $f^+ = 0.1 \sim 5$, then it seems that the flow separation control effect saturates within the range of $f^+ = 5 \sim 10$ since similar velocity profiles are obtained at excitation frequencies of $f^+ = 5$ and 10 . Within the excitation frequency of $f^+ = 10 \sim 100$, the separation control effect seems to decrease with excitation frequency, as can be seen in Figure 7-10, the velocity profile for the controlled case with actuation

frequency of $f^+ = 50$ is flatter than that at excitation frequency of $f^+ = 10$, however, it's still fuller than those for the controlled flow fields with unsteady actuation of plasma actuation at frequencies of $f^+ = 0.1$ and 1. All the velocity profiles with plasma excitations at frequencies of $f^+ = 0.1, 1, 5, 10$ and 50 are positive, which demonstrates separation is fully controlled for these controlled cases. When plasma actuator is actuated at a frequency of $f^+ = 100$, a velocity profile with negative velocity and inflectional point is developed, confirming the existence of a flow separation. This velocity profile is the weakest among all the controlled cases, so the least effective flow separation control is achieved at frequency of $f^+ = 100$ among all the tested frequencies. However, velocity profile at frequency of $f^+ = 100$ is fuller than that for baseline flow, therefore, it still helps control the flow separation around NACA 0012 aerofoil.

On the basis of the above analysis, it seems the optimal excitation frequency is $f^+ = O(10)$ in the current study since the forcing at this frequency excites the shear layer instability of the baseline flow. This is consistent with the findings using a synthetic jet at around an aerofoil leading-edge (Glezer et al., 2005). However, Greenblatt and Wygnanski (2000) claim that the optimal frequency for flow separation control by periodic excitation is $f_e L_{te}/U_\infty \approx 1$, which is around the wake instability frequency. The optimal frequency reported in literature is quite spread (Bar-Sever, 1989; Chang et al., 1992; Hsiao et al., 1994; HSIAO et al., 1990), and our understanding of this is still very limited. In the author's opinion, the optimal frequency might be affected by baseline flow regimes, excitation level and location and so on.



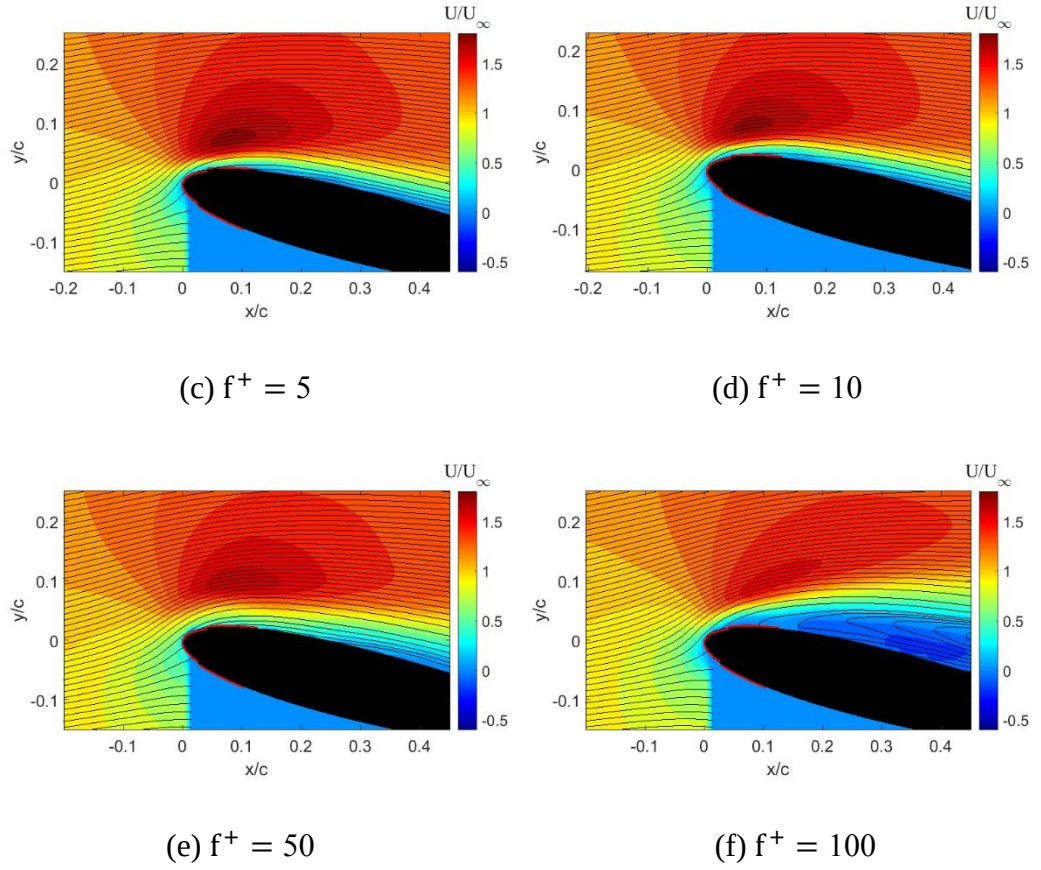


Figure 7-9 Normalized streamwise velocity contour superimposed by streamline of the time-averaged flow field for the controlled cases with unsteady actuation of DBD plasma actuator at different frequencies. Plasma momentum coefficient is 10%. The freestream velocity is $U_\infty = 3\text{m/s}$ and angle of attack is $\alpha = 14^\circ$.

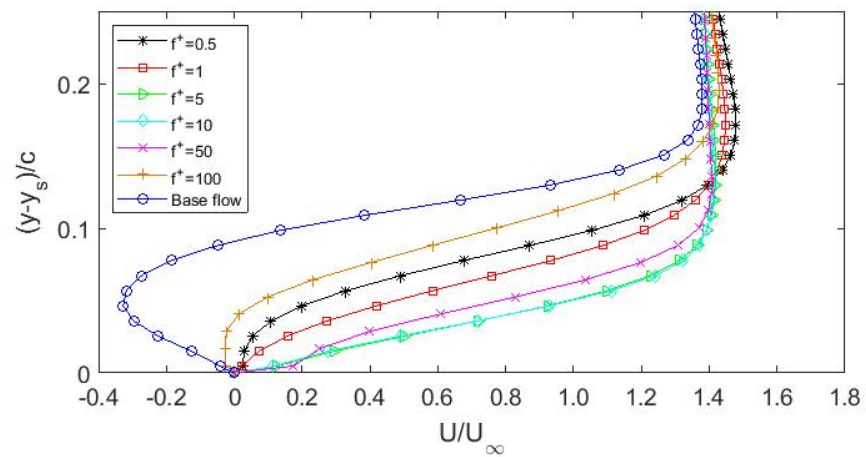
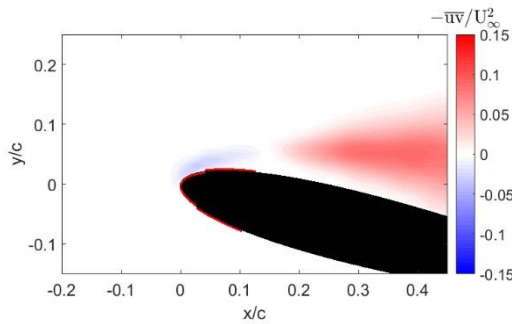


Figure 7-10 Mean velocity profiles at $x/c = 0.35$ for the baseline case and controlled cases with plasma excitations at different frequencies.

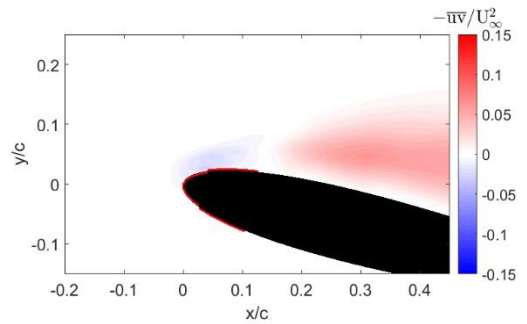
The normalized Reynolds stress contours for the controlled flow fields with plasma excitations at different frequencies are plotted in Figure 7-11. As can be seen, when DBD plasma actuator is actuated periodically at frequencies of $f^+ = 0.1, 1, 5, 10$ and 50 , the Reynolds stress locates closer to the aerofoil surface compared to the baseline case in Figure 4-4, suggesting momentum exchange is promoted in the near-wall region, accordingly, flow separation is well suppressed with plasma excitations at these frequencies. When excitation frequency increases to $f^+ = 100$, Reynolds stress is located away from the suction surface, which might correspond to a separated shear layer and vortex shedding, suggesting an existence of flow separation with plasma excitations at such a high frequency. The vortical structures by plasma lead to an increase in Reynolds stress compared with the uncontrolled flow. The lower position and greater magnitude of the Reynolds stress relative to the baseline case account for an alleviation of separation when excitations are applied at frequency of $f^+ = 100$. Moreover, the Reynolds stress magnitude decreases with excitation frequency with the range of $f^+ = 0.5 \sim 5$ while a similar Reynolds stress magnitude is observed at excitation frequencies of $f^+ = 5, 10$ and 50 , then it increases to a higher value at $f^+ = 100$.

Figure 7-12 shows the root-mean-squared streamwise and vertical velocity contours for the controlled flow fields with plasma excitations at different frequencies. Moreover, the r.m.s. streamwise velocity profiles at chord-wise position of $x/c = 0.35$ are extracted and plotted in Figure 7-13, in the meanwhile, the r.m.s. streamwise velocity profile for the baseline flow is plotted in the figure as well for comparison. Without any control, the vertical location of the r.m.s. velocity peak at $x/c = 0.35$ is around $0.12c$ vertically away from the upper surface, and the maximum r.m.s. streamwise velocity magnitude u' is around $0.32U_\infty$ at angle of attack of 14° . For the controlled cases, the vertical location of the maximum streamwise r.m.s. velocity moves closer to the aerofoil surface, confirming a suppression of flow separation. The velocity fluctuation is greater in the near-wall region with periodic excitations than that for the baseline flow, which demonstrates an amplification of flow instability by plasma excitation. The vertical location of the maximum streamwise velocity fluctuation is located closer to the aerofoil surface at excitation frequency of $f^+ = 5, 10$ and 50 than other controlled cases, which implies that momentum exchange occurs nearer to the suction surface

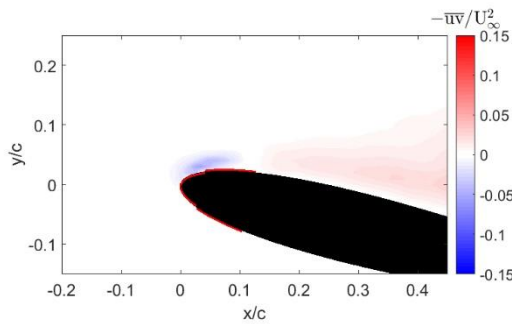
so that momentum is easier to be transformed into the boundary layer, accordingly, better flow separation control effect is achieved at these excitation frequencies. The maximum u' at excitation frequency of $f^+ = 100$ locates furthest from the upper surface among all the controlled cases. This is ascribed to the existence of a separated shear layer. In addition, the maximum r.m.s. streamwise velocity magnitude decreases with excitation frequency within the range of $f^+ = 0.5 \sim 5$, however, a similar maximum r.m.s. streamwise velocity magnitude u' is obtained at excitation frequency of $f^+ = 5, 10$ and 50 . At the highest excitation frequency tested in this study of $f^+ = 100$, the maximum streamwise velocity fluctuation increases to around $0.5U_\infty$, almost the same as that with plasma excitations at frequency of $f^+ = 0.5$. It's interesting to note that the trend of the maximum r.m.s. streamwise velocity magnitude u' against excitation frequency is the same as the Reynolds stress magnitude versus excitation frequency.



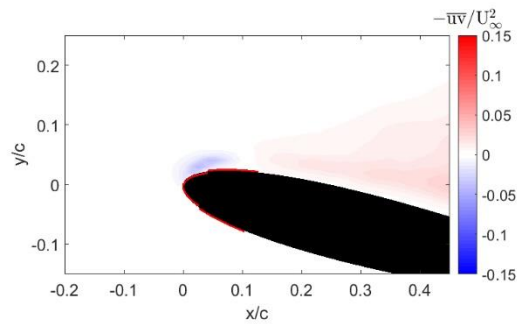
(a) $f^+ = 0.5$



(b) $f^+ = 1$



(c) $f^+ = 5$



(d) $f^+ = 10$

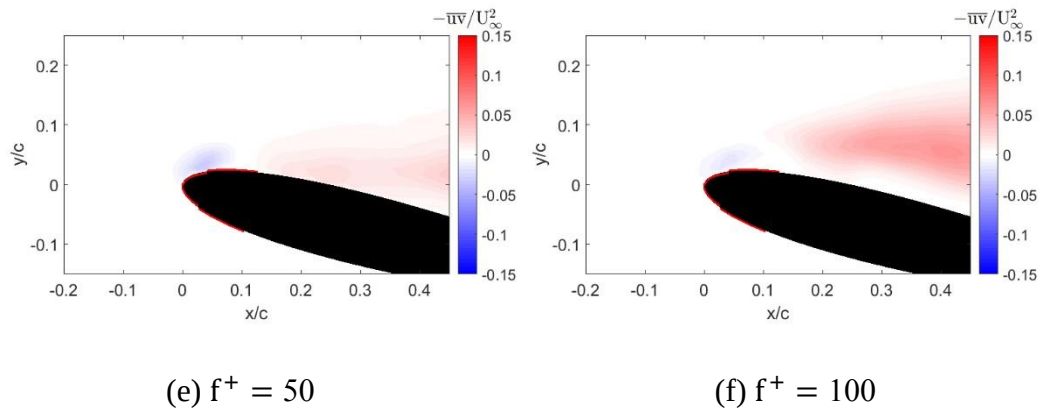
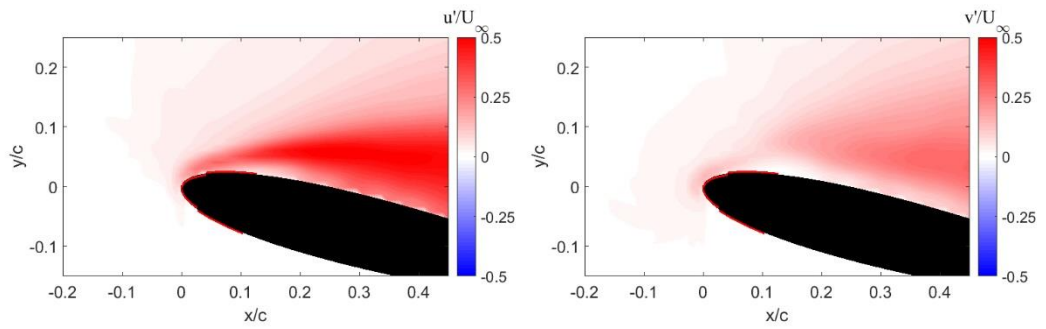
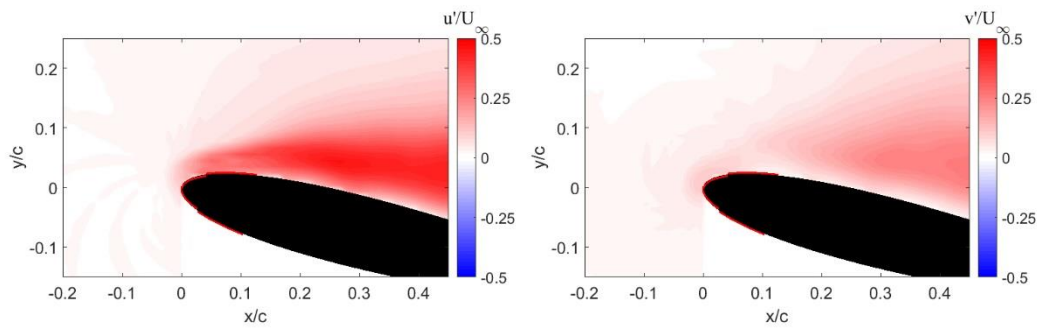


Figure 7-11 Normalized Reynolds stress contours of the time-averaged flow field around a NACA 0012 aerofoil for the controlled cases with unsteady actuation of DBD plasma actuator at different frequencies.

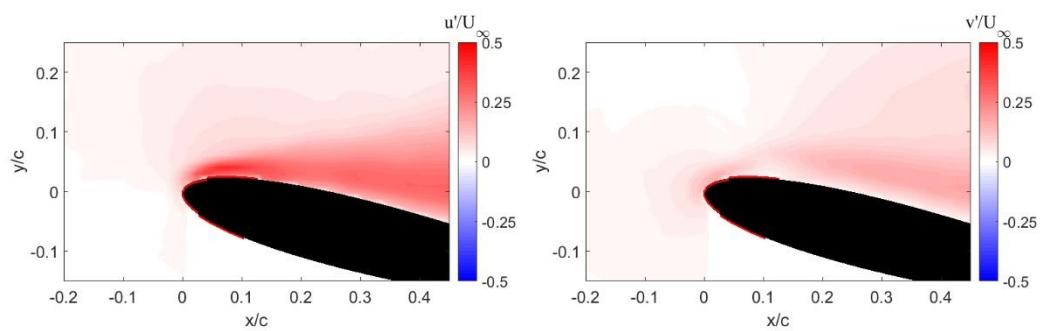
a) $f^+ = 0.5$



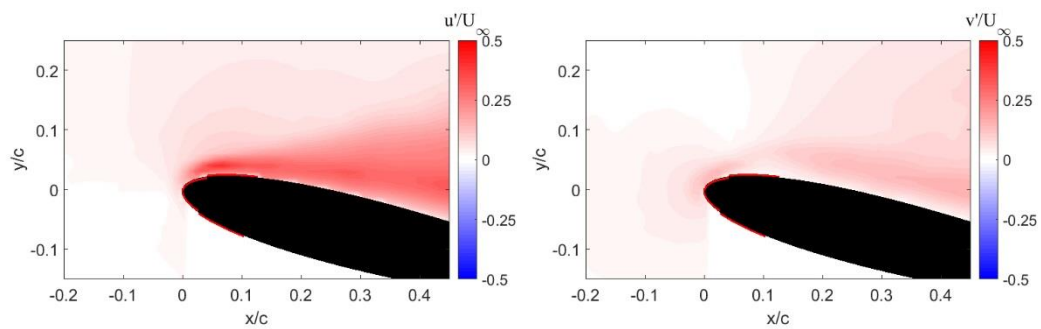
b) $f^+ = 1$



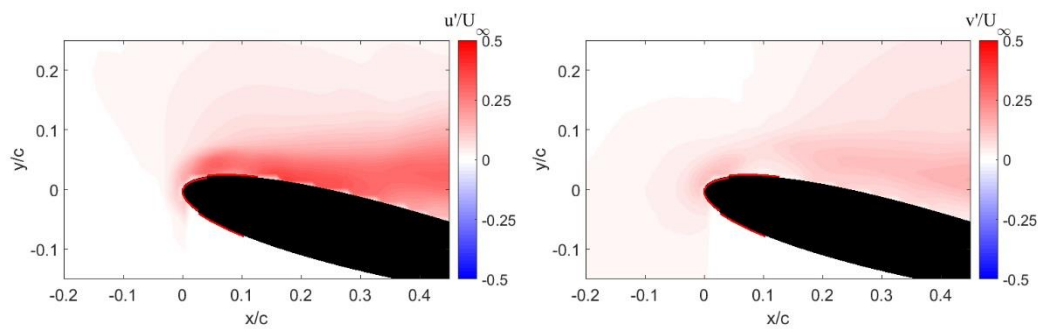
c) $f^+ = 5$



d) $f^+ = 10$



e) $f^+ = 50$



f) $f^+ = 100$

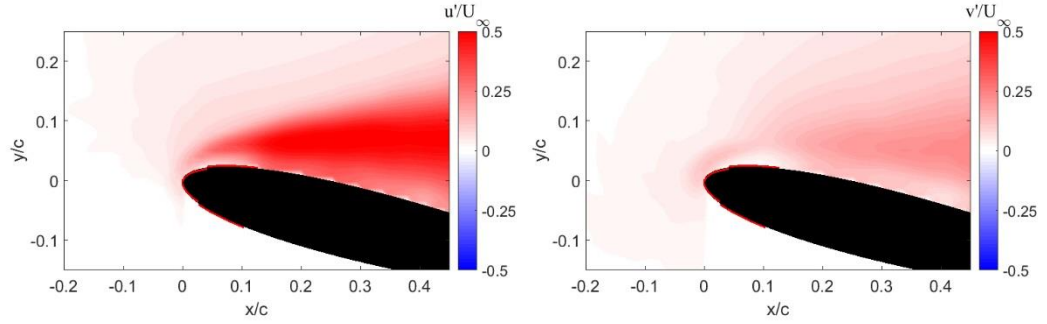


Figure 7-12 Contours of streamwise (left column) and vertical (right column) root-mean-square velocities with unsteady activation of plasma at different excitation frequencies.

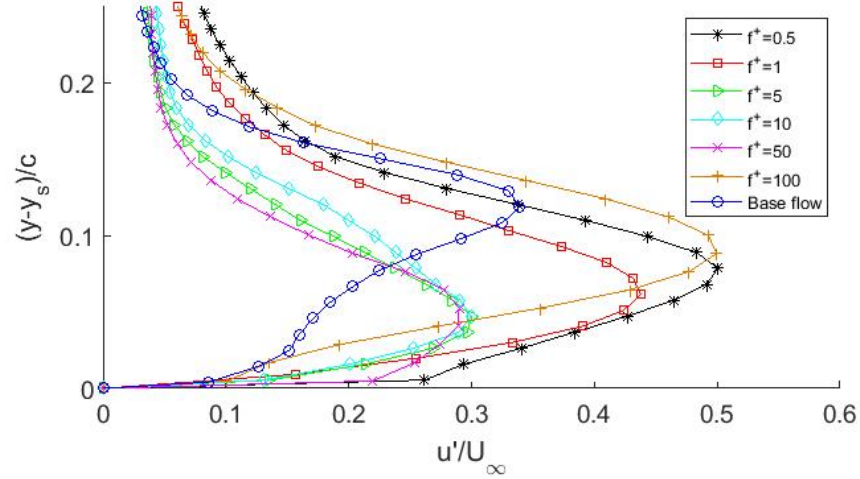


Figure 7-13 Root-mean-squared streamwise fluctuation velocity profiles at $x/c = 0.35$ for the natural case and controlled cases with unsteady actuation of DBD plasma actuator at different excitation frequencies. Duty cycle is 10%. The freestream velocity is $U_\infty = 3\text{m/s}$ and angle of attack is $\alpha = 14^\circ$.

To gain deeper insights into flow behaviours with plasma excitations at different frequencies, phase-averaged analysis was conducted and the corresponding results are presented. Figure 7-14 shows the phase-averaged vertical velocity, streamline, spanwise vorticity and swirling strength for the controlled flow field with unsteady actuation of plasma actuator at a frequency of $f^+ = 0.5$ and duty cycle of 10%. At

phase angle of 0° , the streamline clearly shows a flow separation from aerofoil surface, which can also be confirmed by the positive vertical velocities above the upper surface of aerofoil shown in Figure 7-14 (a). The spanwise vorticity away from the suction surface corresponds to a separated shear layer. It is worth mentioning that the positive vertical velocity magnitude at phase angle 0° is greater and the recirculated flow occupies a larger region than that with excitation at frequency of $f^+ = 1$, which is shown in Figure 7-7 (a), so a severer separation is developed in the beginning of each plasma pulse cycle. The vortex due to the evacuation of the separated shear layer on the initiation of plasma can be observed at phase 45° in Figure 7-14 (b), as indicated by the swirling strength isolines and vorticity contours. This vortex locates further away from the aerofoil surface than that with excitations at unity non-dimensional excitation frequency because flow is more severely separated and the separated shear layer locates at a larger vertical position in the beginning of the forcing cycle. The vortex by a release of the concentrated vorticity after plasma is turned off could be identified by swirling strength and positioned at $x/c \approx 0.28$ in Figure 7-14 (c). These vortices induce an entrainment effect between the freestream and boundary layer, as indicated by the positive and negative vertical velocities around the vortices. Accordingly, a fully attached flow is obtained at phase angle of 135° in Figure 7-14 (d). After both of the vortices travel downstream, the time interval between the successive plasma durations is so large at plasma pulse frequency of $f^+ = 0.5$ that flow recovers and starts to separate again in Figure 7-14 (e), (f), (g) and (h). At excitation frequency of $f^+ = 0.5$, flow starts to separate at phase angle of 180° , much earlier than that at $f^+ = 1$, which is 270° . Moreover, a severer flow separation is developed at the end of the cycle due to a larger time interval between forcing cycles at this small excitation frequency.

When plasma excitations are applied into the flow field around NACA 0012 aerofoil at a higher frequency of $f^+ = 5$, flow is attached to aerofoil surface at phase of 0° , as shown in Figure 7-15 (a). In the meanwhile, two vortices created during the previous plasma forcing cycle still exist in the field of view and are located at around $x/c = 0.3$ and 0.4 respectively, which can be indicated by the swirling strength isolines. At excitation frequency of $f^+ = 5$, the period between plasma forcing cycles is so small that plasma-induced vortices could only travel a

small distance and keep within the field of view when the next plasma cycle begins. The trajectories of the two vortices produced over a plasma forcing cycle is presented in Figure 7-16, and the convection velocity can be obtained from the slope of the phase evolution. The convection velocity of the vortices is around 2.9 m/s. Under the condition of flow separation control with plasma excitations at this frequency, the space between the two vortices is very small. Accordingly, these two vortices are almost merged together in their initial stage due to the small distance between them, this is well demonstrated in Figure 7-15 (b) and (c). Then these two vortices travel downstream and break away at phase angle of 135° , as shown in Figure 7-15 (d). The first vortex formed on the initiation of plasma develops in the beginning, then it starts to weaken and become smaller due to the dissipation within the flow. Since vortices are densely packed and no flow separation appears at the beginning of a forcing cycle at $f^+ = 5$, less flow is evacuated when plasma is initiated, accordingly, a weaker vortex is formed at this excitation frequency. The second vortex formed during a plasma forcing cycle develops over the entire cycle, the reason accounting for this might be that it is more energetic and capable of overcoming the dissipation in the flow as it's created due to a release of the concentrated vorticity that promotes mixing and energizes the flow. The vortices produced at excitation frequency of $f^+ = 5$ are much smaller than that produced at non-dimensional excitation frequency of $f^+ = 0.5$ and 1, confirming that a smaller vortex is produced at a higher excitation frequency, consistent with the observations of other flow separation control studies (Zhang and Samtaney, 2015) and (Jukes and Choi, 2009b). The vortices are closely packed at a higher excitation frequency, therefore the vortex does not have enough space to be fully developed, as a result, smaller vortices are created at higher excitation frequencies. It can be clearly seen in all the eight phases that flow is always attached over the entire cycle period, demonstrating a better flow separation control effect at excitation frequency of $f^+ = 5$ compared with that at non-dimensional excitation frequency $f^+ = 0.5$ and 1. The time interval between plasma durations is very small at excitation frequency of $f^+ = 5$, consequently, flow does not have time to recover and separate between the successive plasma forcing durations.

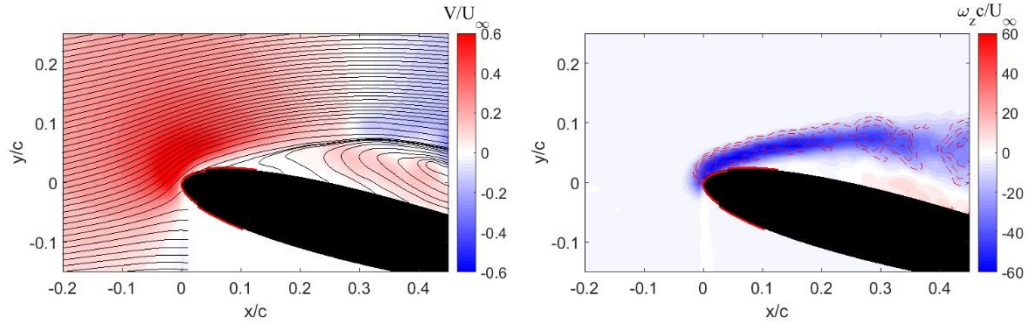
Phase-averaged behaviours of the flow with periodic addition of plasma momentum at frequency of $f^+ = 10$ are presented in Figure 7-17. At phase angle

of 0° , three vortices generated during previous plasma pulse cycles exist in the field of view and locate at $x/c = 0.19, 0.3$ and 0.42 respectively. The time duration of a plasma forcing cycle is very small and the vortices can only travel a short distance within a cycle, thus more vortices could be seen in the field of view at this higher excitation frequency. As can be seen in Figure 7-17, it seems that only one vortex is created over a plasma pulse cycle, which is quite different from that at smaller excitation frequencies, at which two vortices are produced within one cycle. At excitation frequency of $f^+ = 10$, since duty cycle is fixed at 10%, the plasma forcing duration is very small relative to that at smaller frequencies, as a consequence, the evacuated boundary layer on the initiation of plasma and the released concentrated vorticity layer merge together and rolls into one vortex in Figure 7-17 (d). Since there are always quite a few vortices modifying the flow above the aerofoil surface, the flow keeps attached to the suction surface over the entire cycle. Therefore, a very good flow separation control effect is achieved at this excitation frequency. As demonstrated in previous discussions, smaller vortices are created at higher excitation frequencies, accordingly, momentum entrainment induced by these vortices is weaker. Therefore a further increase in excitation frequency does not necessarily give a better flow separation control effect although the time interval between excitation forcing durations is smaller.

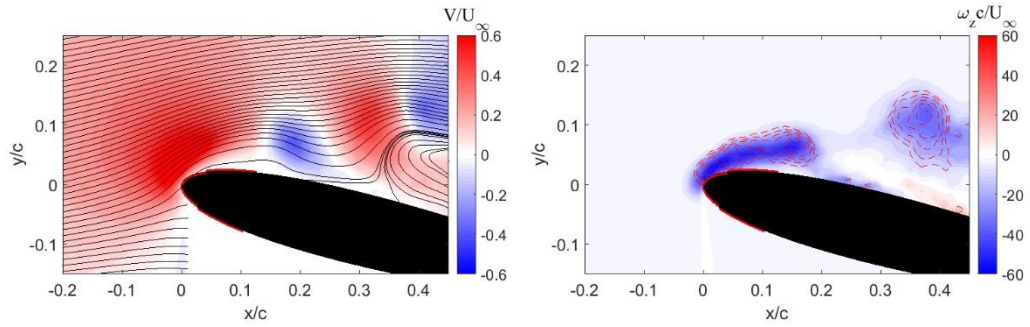
As for the controlled flow fields with unsteady actuation of plasma actuator at frequencies of $f^+ = 50$ and 100 , the excitation frequency is so high that it's impossible to perform phase-averaged analysis at a sampling frequency of $2,000$ Hz. Therefore, phase-averaged analysis was not applied to these two cases and instantaneous analysis was utilized to investigate the transient flow behaviours with plasma excitations at frequencies of $f^+ = 50$ and 100 . As shown in Figure 7-18, after plasma is turned on, the separated shear layer rolls up into a vortex and travels downstream. The subsequent flow locates closer to the aerofoil surface and reattaches at $t^+ = 4.50$ when plasma excitations were applied at a frequency of $f^+ = 50$. This is probably due to an amplification of the disturbances induced by plasma excitations, which promotes momentum exchange. With plasma excitation at a frequency of $f^+ = 100$, the flow is not able to reattach to the aerofoil surface, moreover, it behaves similar to the flow with steady actuation of plasma actuator,

which indicates that excitation frequency of $f^+ = 100$ is too large to excite the instability of the flow around a NACA 0012 aerofoil at Reynolds number of 20,000.

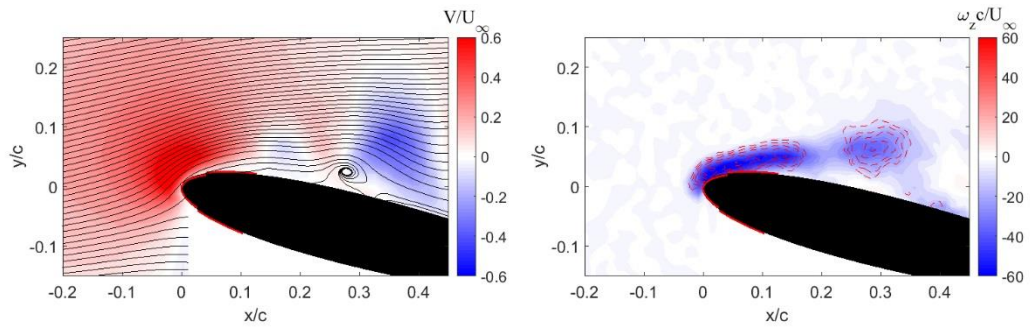
a) $\varphi = 0^\circ$



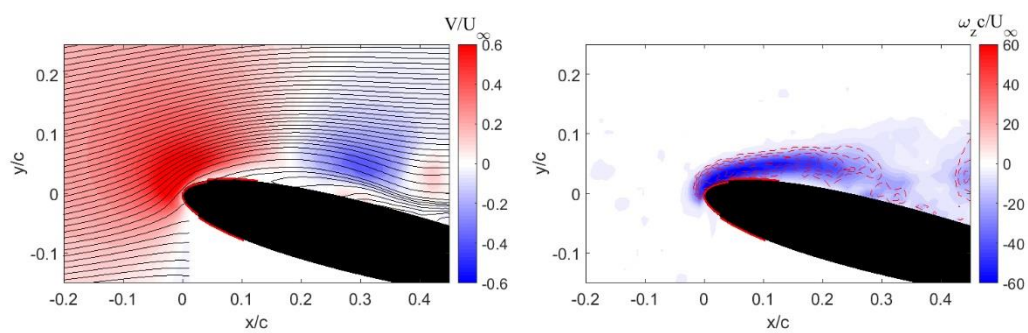
b) $\varphi = 45^\circ$



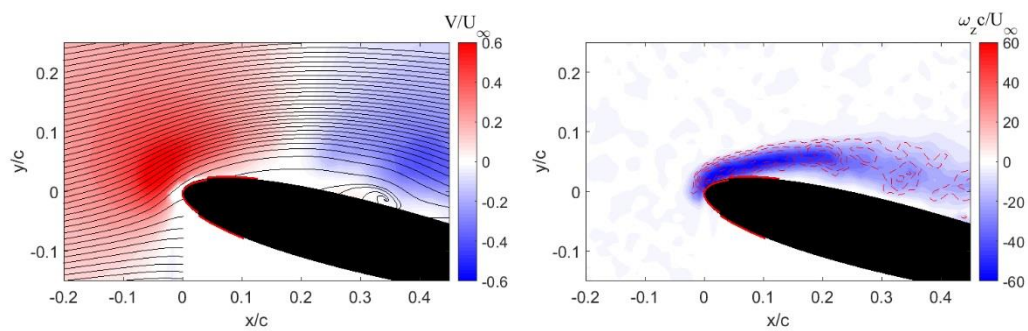
c) $\varphi = 90^\circ$



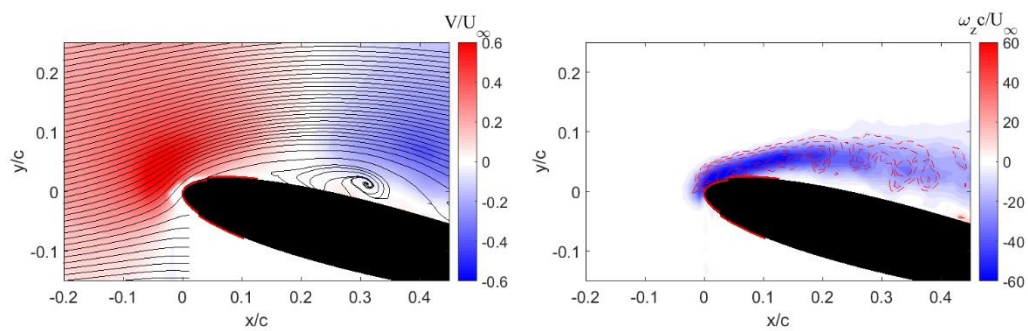
d) $\varphi = 135^\circ$



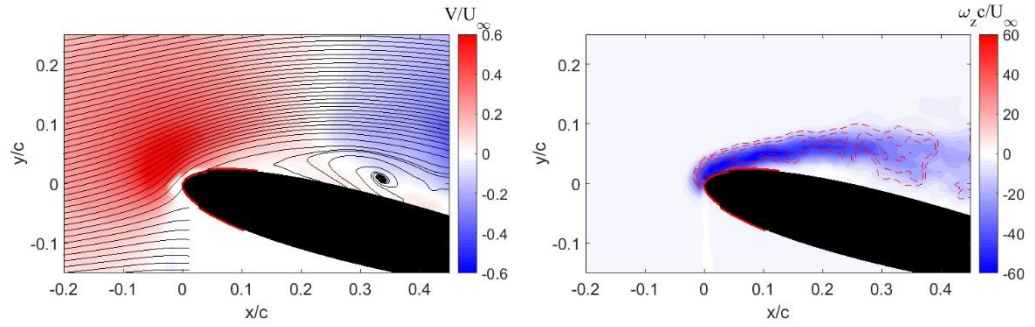
e) $\varphi = 180^\circ$



f) $\varphi = 225^\circ$



g) $\varphi = 270^\circ$



h) $\varphi = 315^\circ$

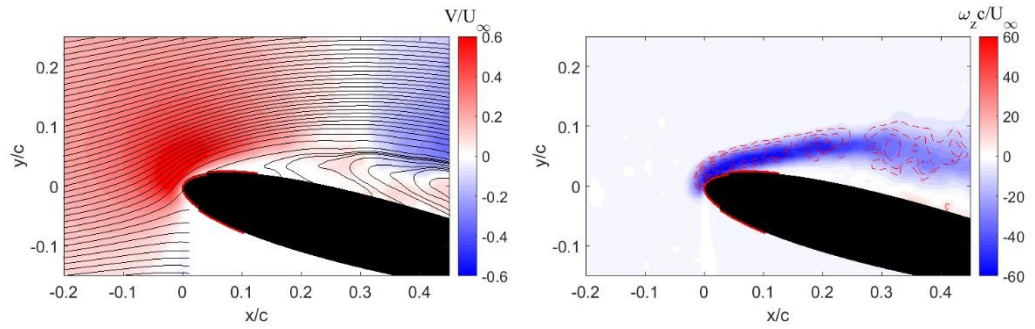
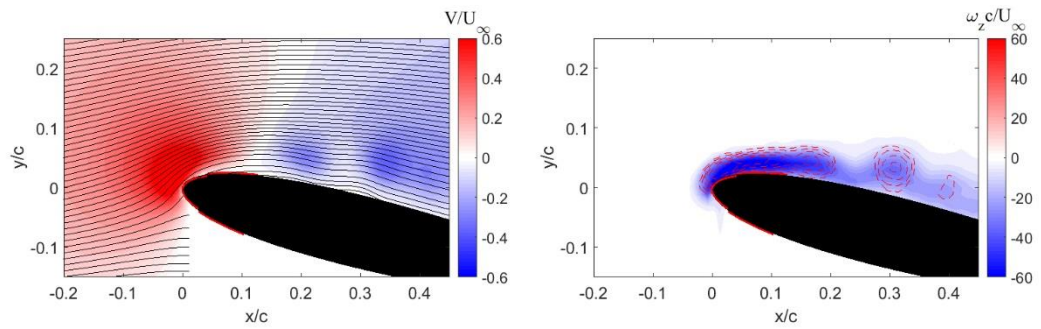
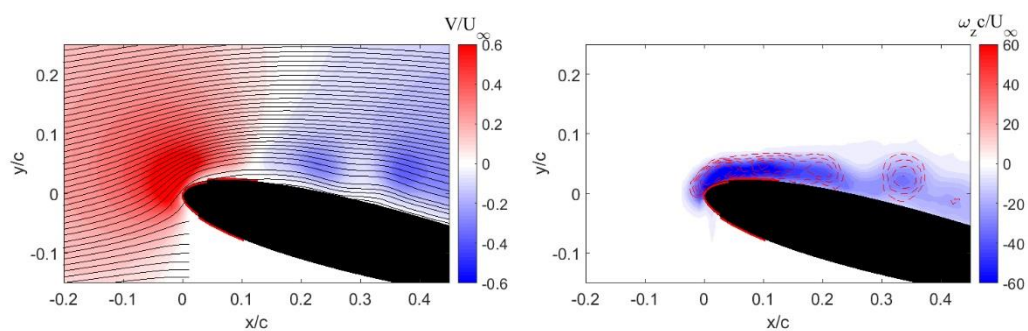


Figure 7-14 Phased-averaged analysis of the flow field around NACA 0012 aerofoil at angle of attack of $\alpha = 14^\circ$ with plasma excitations at frequency of $f^+ = 0.5$ and duty cycle of 10%. Freestream velocity is $U_\infty = 3\text{m/s}$.

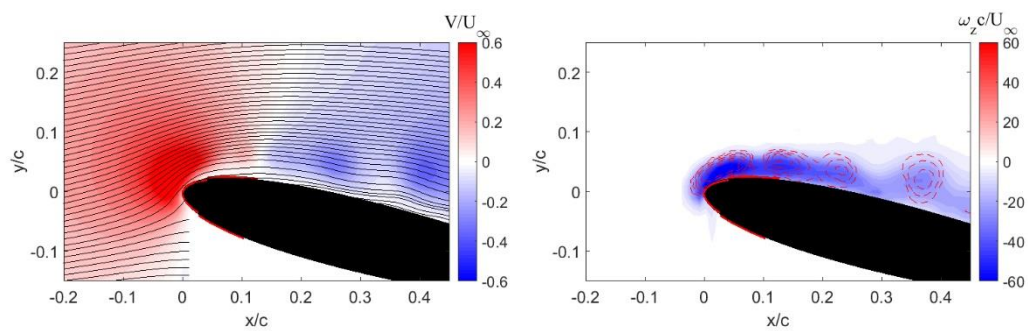
a) $\varphi = 0^\circ$



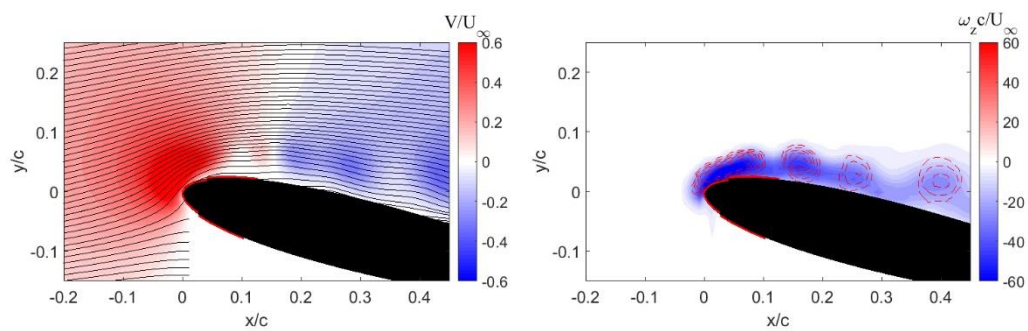
b) $\varphi = 45^\circ$



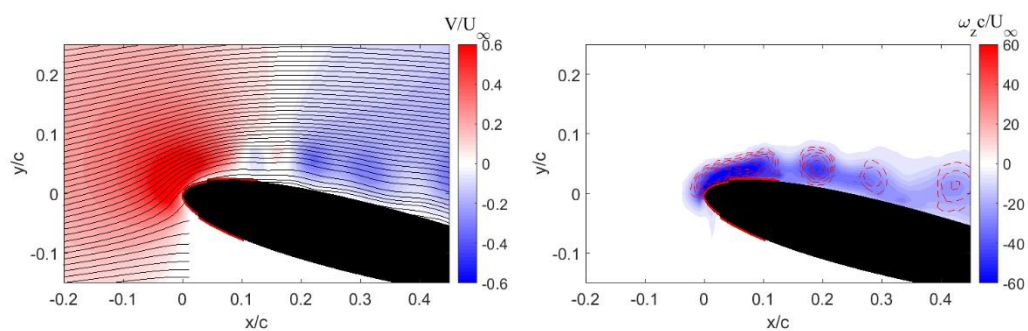
c) $\varphi = 90^\circ$



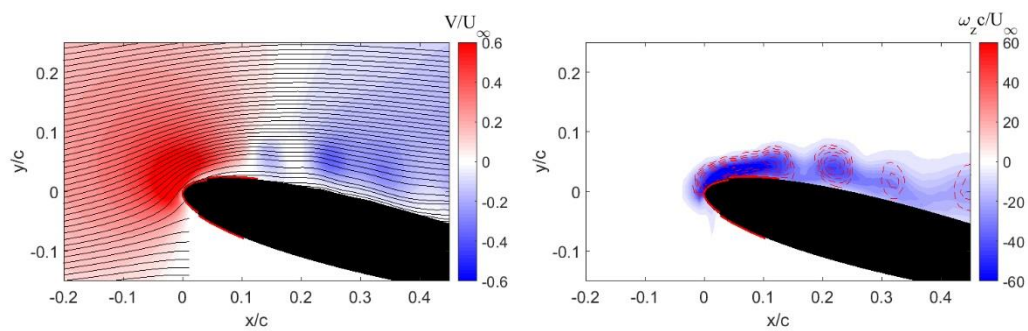
d) $\varphi = 135^\circ$



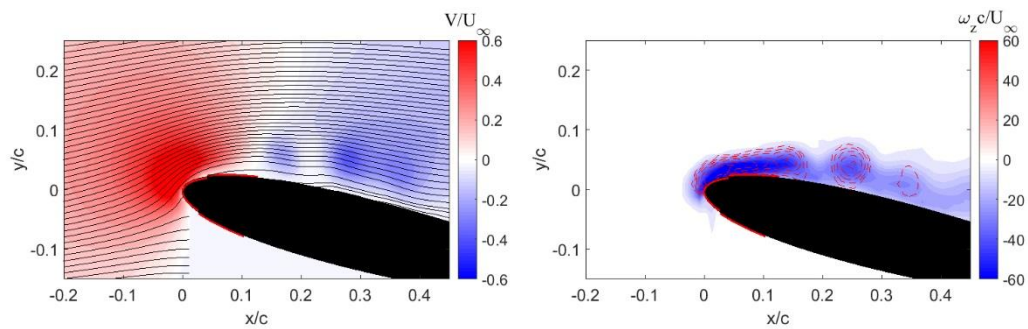
e) $\varphi = 180^\circ$



f) $\varphi = 225^\circ$



g) $\varphi = 270^\circ$



h) $\varphi = 315^\circ$

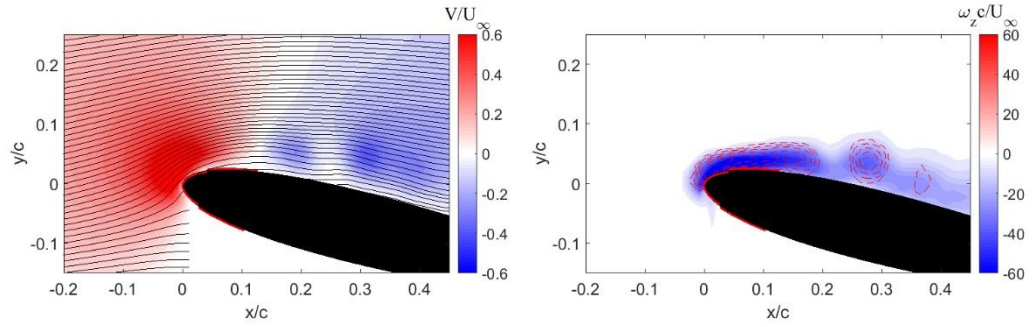


Figure 7-15 Phased-averaged analysis of the flow field around NACA 0012 aerofoil at angle of attack of $\alpha = 14^\circ$ with plasma excitations at frequency of $f^+ = 5$ and duty cycle of 10%.

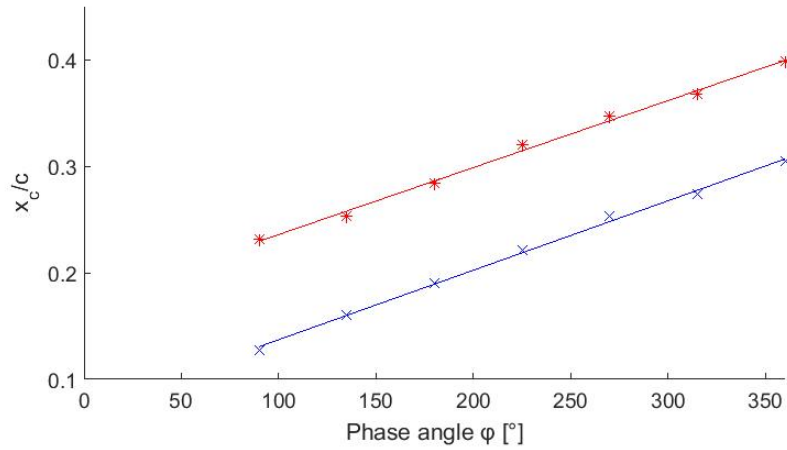
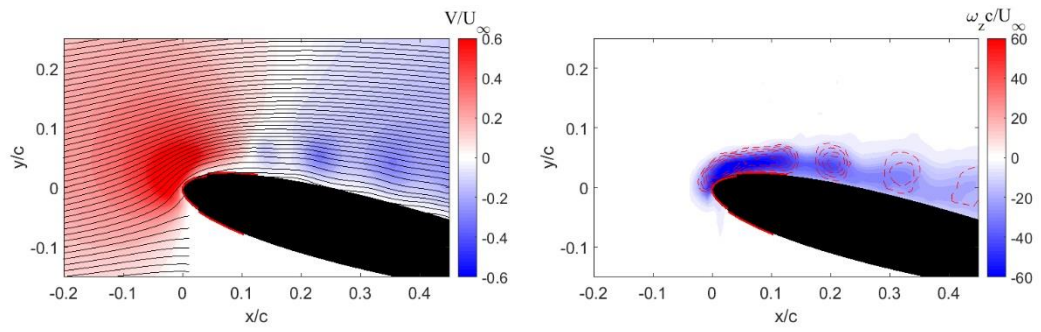
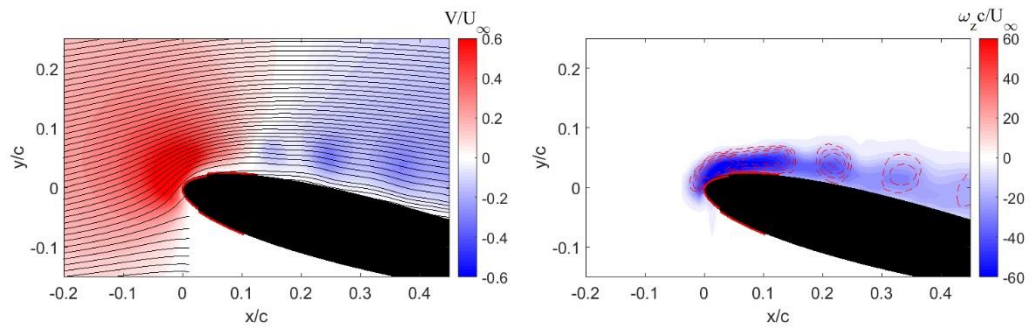


Figure 7-16 Trajectories of the vortices produced over a plasma forcing cycle at frequency of $f^+ = 5$ and duty cycle of 10%: *, trajectory of the first vortex; x, trajectory of the second vortex. The solid lines represent the linear fits of the vortex trajectories.

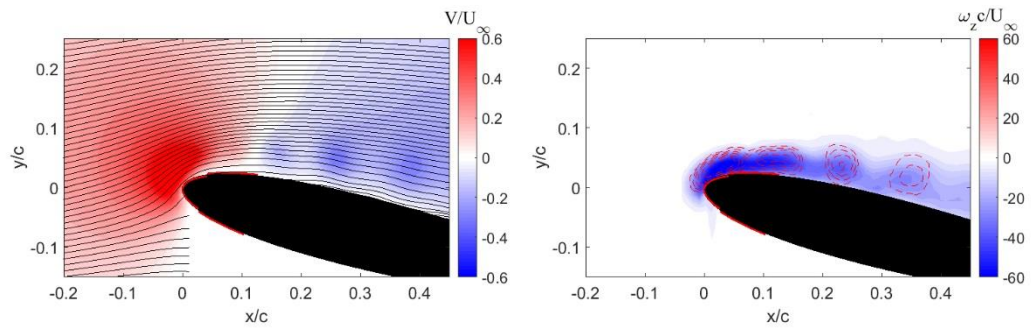
a) $\varphi = 0^\circ$



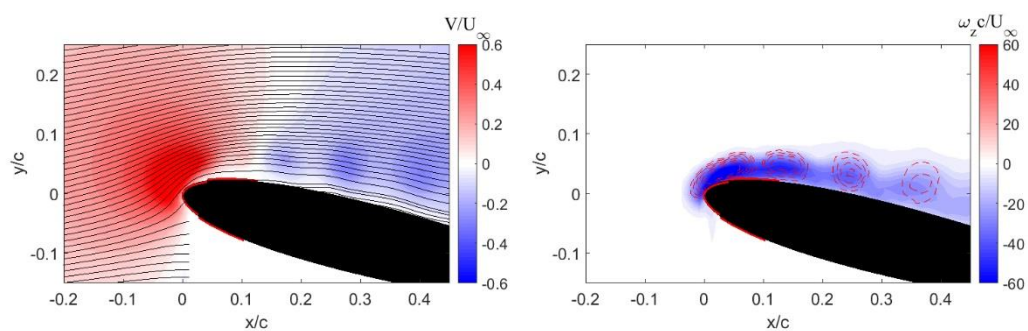
b) $\varphi = 45^\circ$



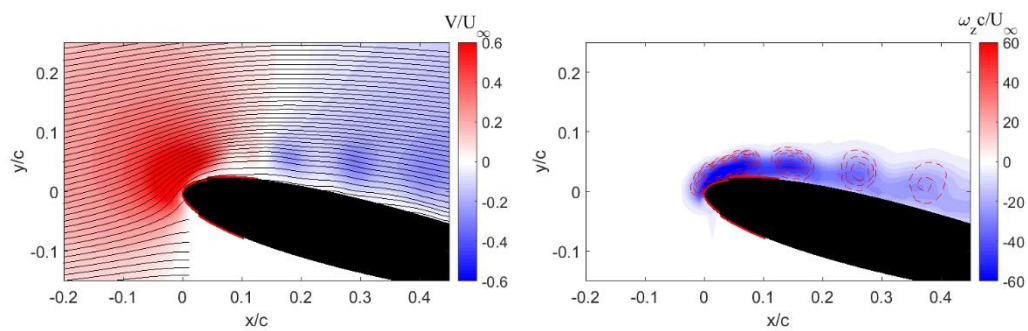
c) $\varphi = 90^\circ$



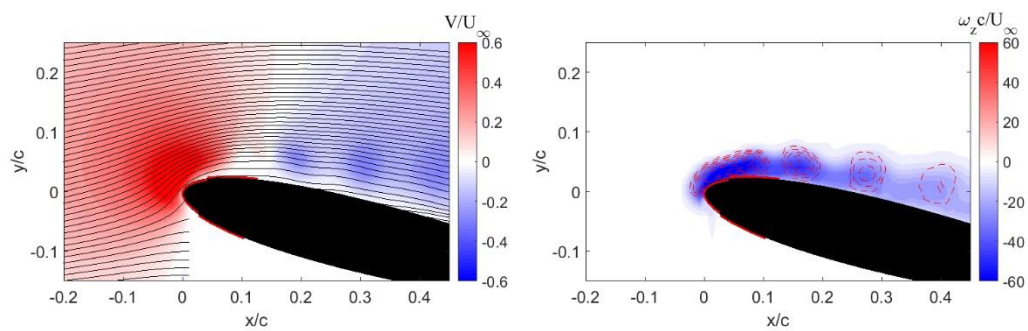
d) $\varphi = 135^\circ$



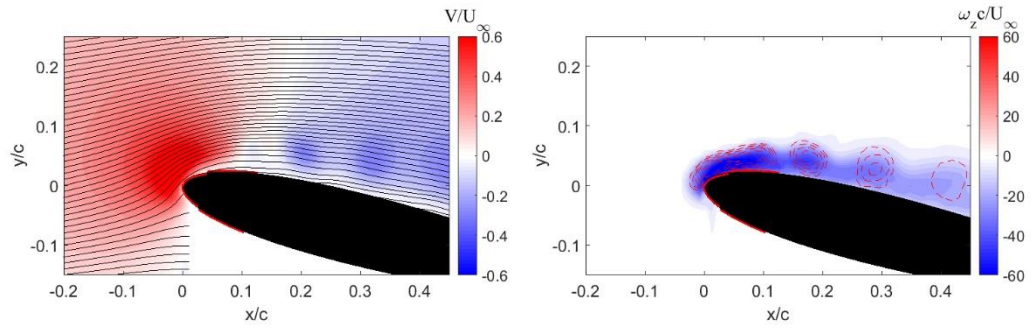
e) $\varphi = 180^\circ$



f) $\varphi = 225^\circ$



g) $\varphi = 270^\circ$



h) $\varphi = 315^\circ$

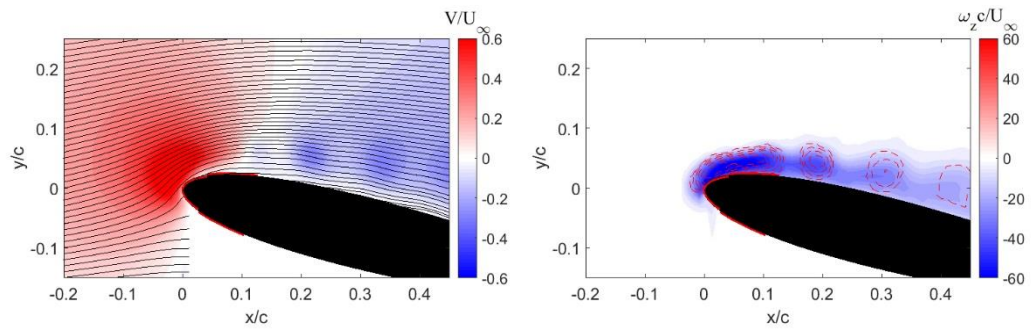
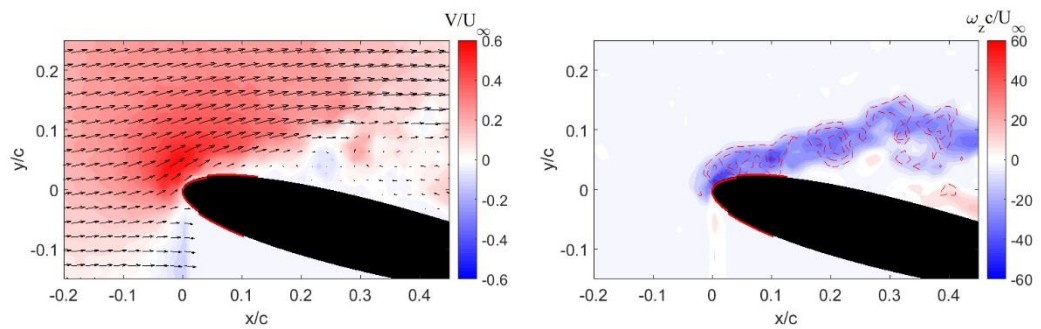
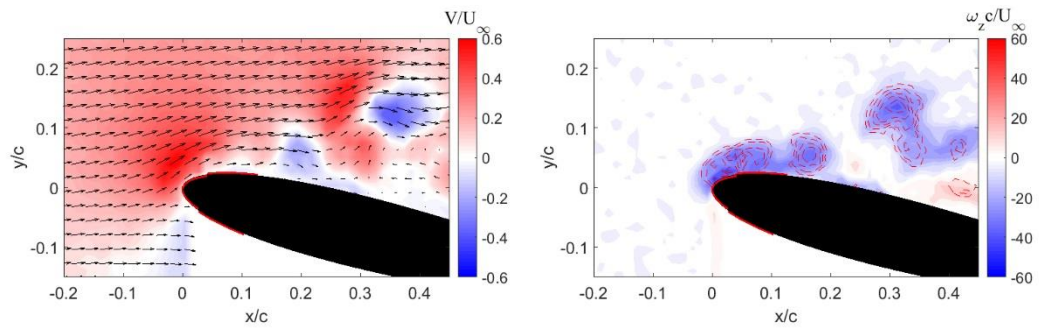


Figure 7-17 Phased-averaged analysis of the flow field around NACA 0012 aerofoil at angle of attack of $\alpha = 14^\circ$ with plasma excitations at frequency of $f^+ = 10$ and duty cycle of 10%.

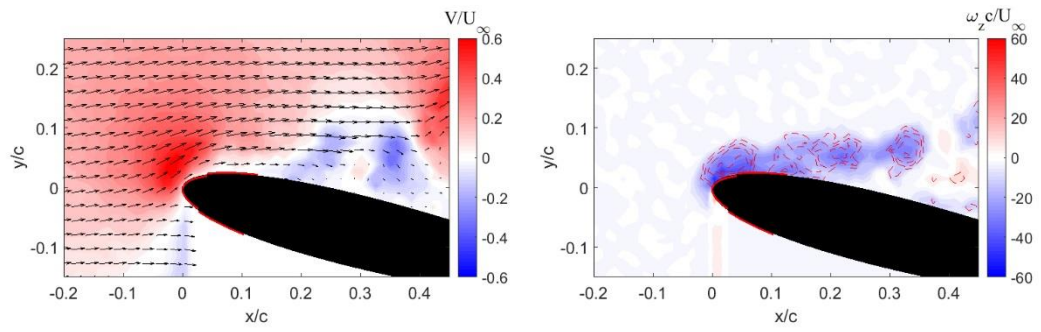
a) $t^+ = 0.00$



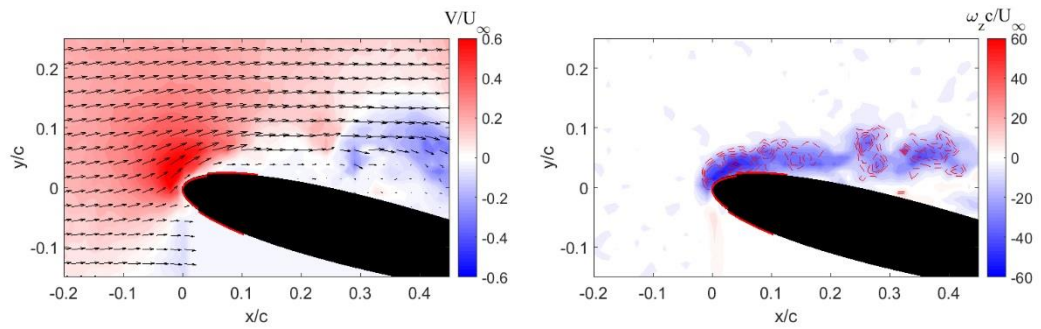
b) $t^+ = 0.30$



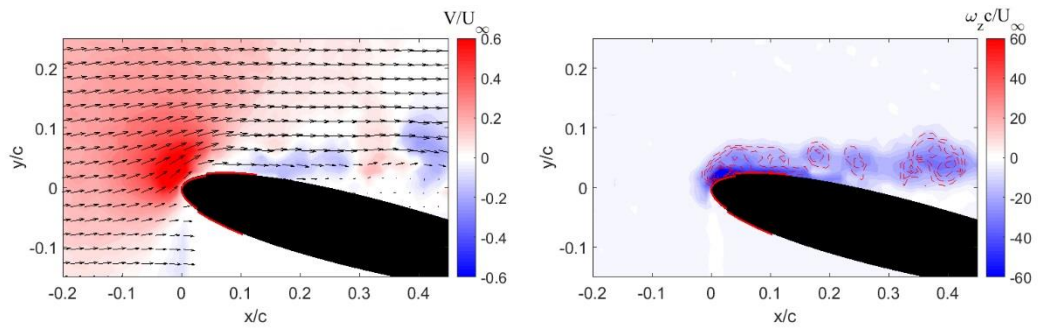
c) $t^+ = 0.60$



d) $t^+ = 0.75$



e) $t^+ = 1.35$



f) $t^+ = 4.50$

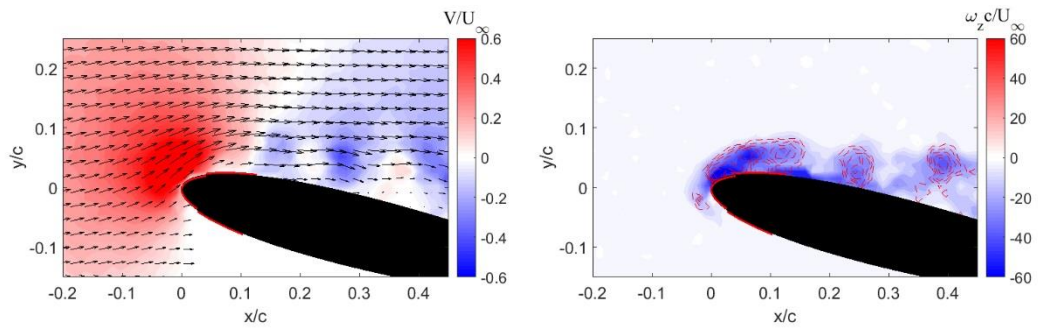
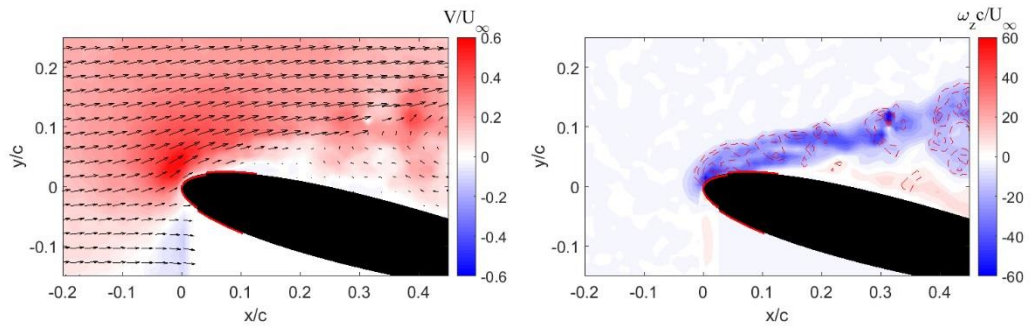
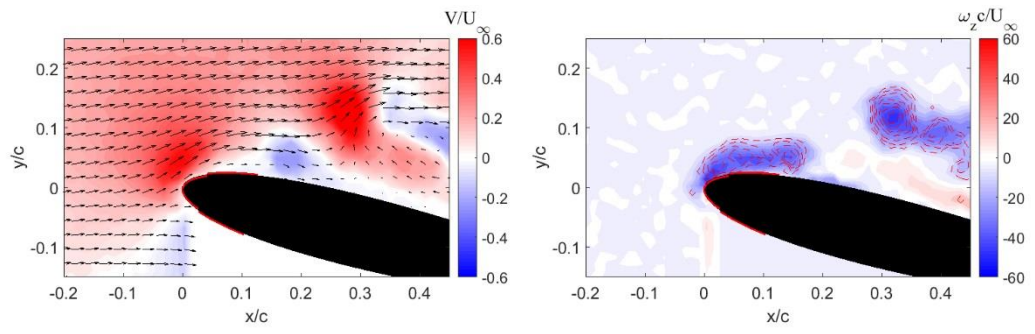


Figure 7-18 Instantaneous flow field around a NACA0012 aerofoil with plasma excitation at frequency of $f^+ = 50$. The red dashed lines represent swirling strength isolines.

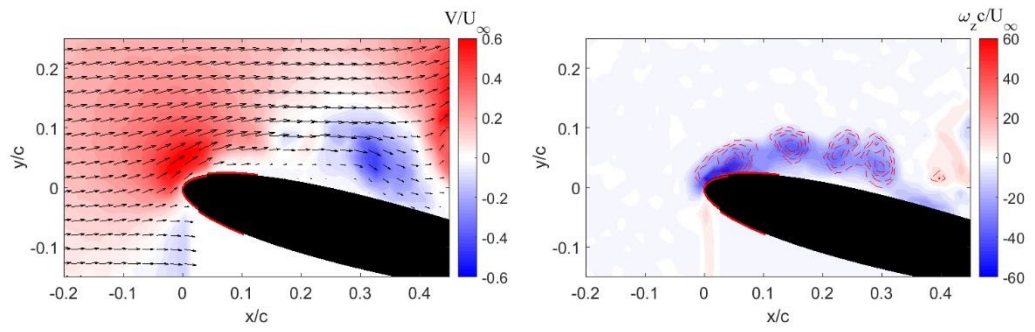
a) $t^+ = 0.00$



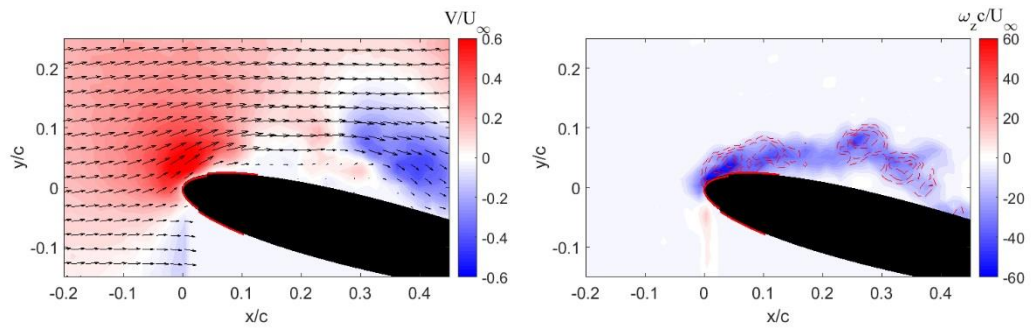
b) $t^+ = 0.30$



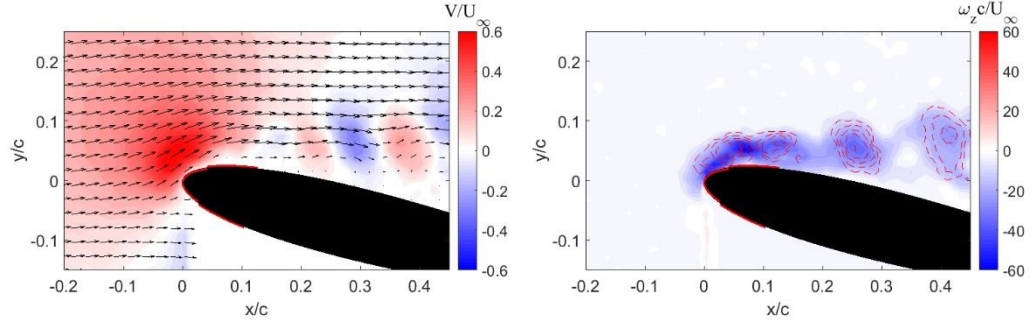
c) $t^+ = 0.60$



d) $t^+ = 0.75$



e) $t^+ = 1.35$



f) $t^+ = 4.50$

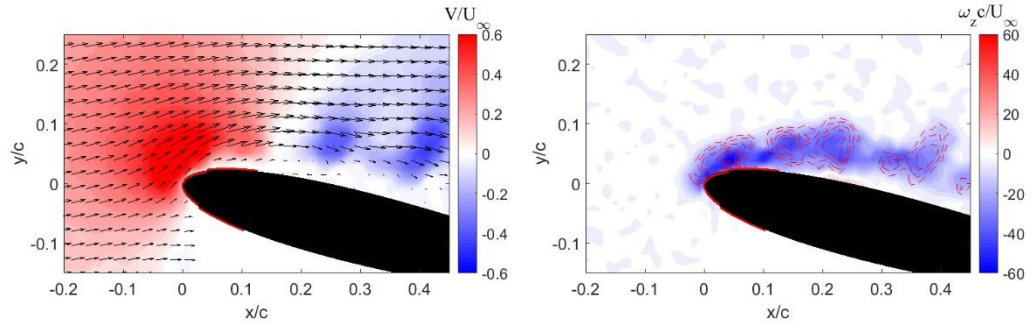


Figure 7-19 Instantaneous flow field around a NACA0012 aerofoil with plasma excitation at frequency of $f^+ = 100$.

7.4 Effect of plasma momentum on flow separation control

Effect of the plasma momentum coefficient on flow separation control around NACA 0012 aerofoil at post-stall angle of attack of 14° was studied in this section. PIV measurements of flow field around an aerofoil with unsteady actuation of DBD plasma actuator were conducted at a sampling frequency of 2,000 Hz and freestream velocity of 3m/s. Both the carrier voltage amplitude and unsteady actuation duty cycle could affect plasma momentum, in the current study, unsteady actuation duty cycle was used to adjust plasma momentum coefficient while amplitude is fixed. The unsteady actuation frequency is $f^+ = 1$ and the various duty cycles of unsteady actuation pulses tested in this study are $\tau = 50\%$, 25% , 5% , 2% , 0.2% and 0.01% , corresponding to a momentum coefficient of $C_\mu = 0.125\%$, 0.0625% , 0.0125% , 0.005% , 0.0005% and 0.000025% respectively.

The streamwise velocity contours superimposed by streamlines of the time-averaged flow field around an aerofoil with periodic excitation of plasma at fixed frequency of $f^+ = 1$ and different plasma momentum coefficients of $C_\mu = 0.125\%$, 0.0625% , 0.0125% , 0.005% , 0.0005% and 0.000025% are presented in Figure 7-20. At angle of attack of 14° , flow separates from the suction surface of aerofoil due to the severe adverse pressure gradient for the baseline case, shown in Figure 4-1 (e). As can be clearly seen from the streamlines, periodic excitations of plasma actuator with momentum coefficients of $C_\mu = 0.125\%$, 0.0625% , 0.0125% and 0.005% enable the separated flow to reattach to the aerofoil surface and completely eliminate the flow separation around aerofoil. Flow separation control could be achieved by plasma excitations with even 2% of the energy consumption by steady actuation of plasma actuator, indicating a much more effective flow separation control technique with unsteady actuation of the linear plasma synthetic jet actuator than steady actuation. However, when plasma momentum is reduced to $C_\mu = 0.0005\%$, the separated flow can not be reattached, instead, a separation bubble appears above the aerofoil surface. With a smaller plasma momentum coefficient of $C_\mu = 0.000025\%$, flow totally separates from the upper surface, it seems that unsteady actuation of plasma actuator with such a small momentum coefficient has little influence on the flow field around aerofoil at angle of attack of 14° and Reynolds number of 20,000.

The mean streamwise velocity profiles of the flow field with unsteady actuation of plasma actuator at different duty cycles are plotted in Figure 7-21 to compare the flow separation control effectiveness of plasma excitations at different duty cycles (momentum coefficients). With plasma momentum of $C_\mu = 0.125\%$, 0.0625% , 0.0125% and 0.005% ($\tau = 50\%$, 25% , 5% and 2%), all the velocities are positive in the profile, indicating an elimination of flow separation, moreover, velocity is only slightly greater at higher plasma momentum coefficients, confirming plasma momentum just affects the flow separation control effect minimally within the momentum coefficient range of $C_\mu = 0.005\% \sim 0.125\%$ ($\tau = 2 \sim 50\%$). When the plasma momentum is reduced to $C_\mu = 0.0005\%$ ($\tau = 0.2\%$), the velocity profile looks much flatter and negative velocity starts to appear, demonstrating a flow

separation around the aerofoil, therefore, flow separation control with plasma excitation at plasma momentum coefficient of $C_\mu = 0.0005\%$ is less effective than

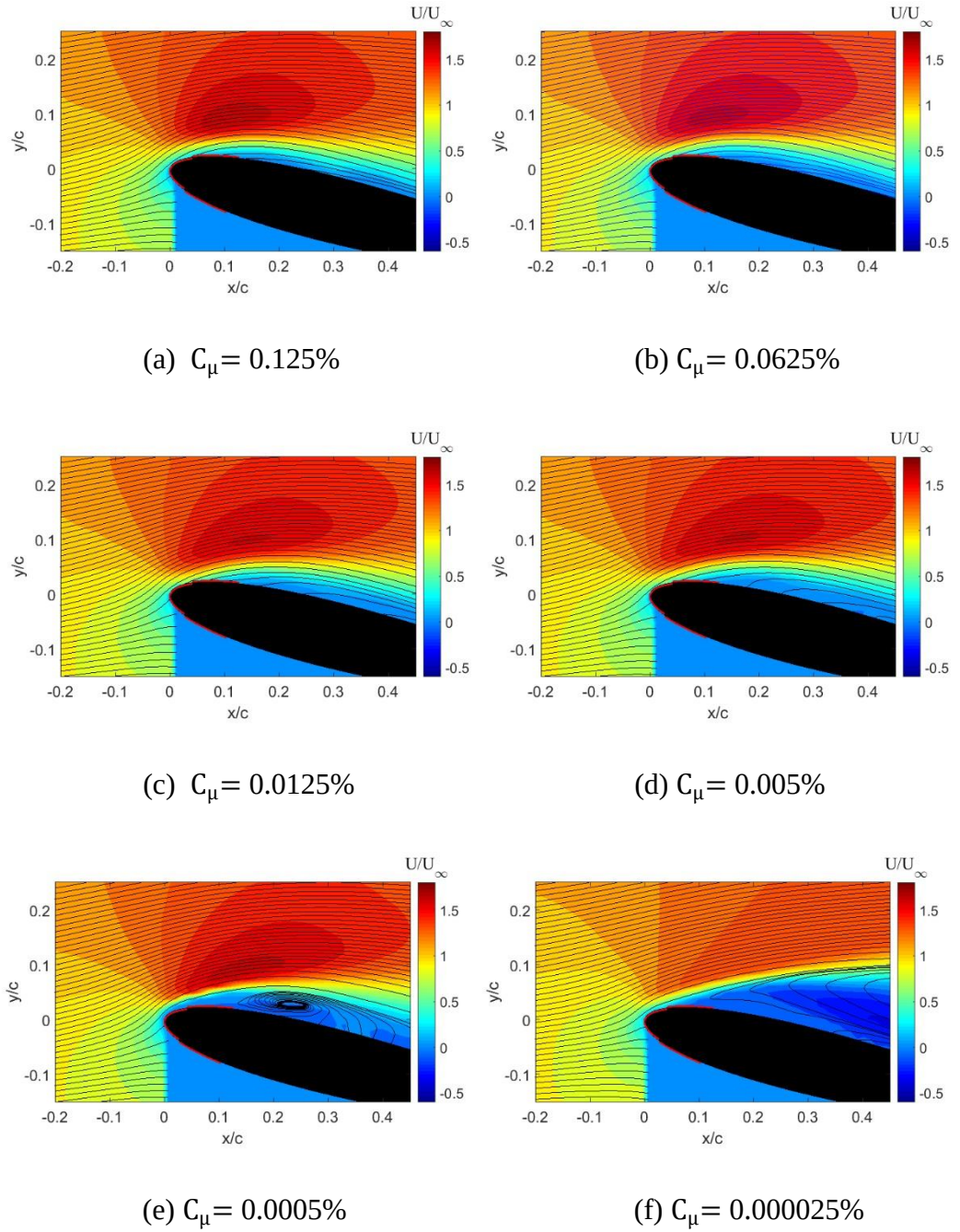


Figure 7-20 Normalized streamwise velocity contour superimposed by streamline of the time-averaged flow field around a NACA 0012 aerofoil for the controlled cases with unsteady actuation of DBD plasma actuator at different momentum coefficients. The pulsed excitation frequency is $f_e/f_0 = 1$. The freestream velocity is $U_\infty = 3\text{m/s}$ and angle of attack is $\alpha = 14^\circ$.

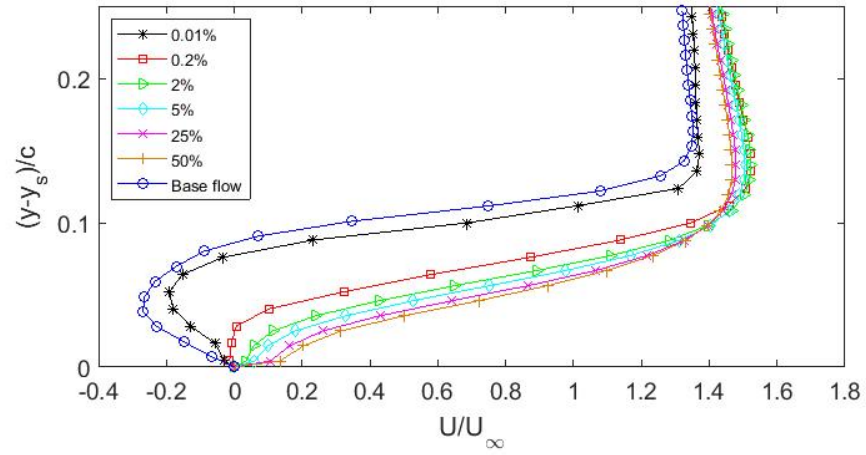


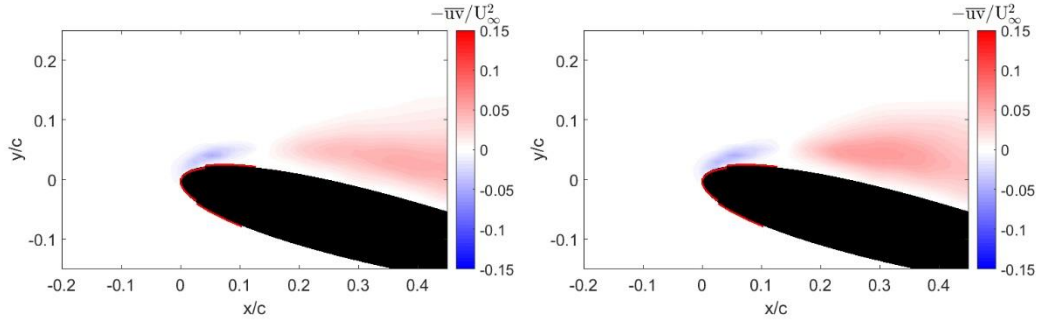
Figure 7-21 Mean streamwise velocity profiles at $x/c = 0.3$ for the baseline case and controlled cases with unsteady actuation of DBD plasma actuator at different momentum coefficients (duty cycles). Excitation frequency is $f^+ = 1$. The freestream velocity is $U_\infty = 3\text{m/s}$ and angle of attack is $\alpha = 14^\circ$.

those at momentum coefficients of $C_\mu = 0.125\%$, 0.0625% , 0.0125% and 0.005% , and excitation momentum could affect flow separation control significantly when it is very low. The velocity profile with plasma excitation at momentum coefficient of $C_\mu = 0.000025\%$ ($\tau = 0.01\%$) is only slightly fuller than that for the baseline case, confirming that periodic excitation of plasma at such a low momentum only has a very weak capability of controlling the flow separation around NACA 0012 aerofoil at post-stall angle of attack of 14° .

Figure 7-22 shows the normalized Reynolds stress contours over an aerofoil at angle of attack of 14° for the controlled flows with periodic activations of plasma actuator at various momentum coefficients of $C_\mu = 0.125\%$, 0.0625% , 0.0125% , 0.005% , 0.0005% and 0.000025% with excitation frequency fixed at $f^+ = 1$. With plasma excitations at momentum coefficients of $C_\mu = 0.125\%$, 0.0625% , 0.0125% , 0.005% and 0.0005% , the Reynolds stress locates closer to the aerofoil surface compared with that for the baseline case, this might lead to a control of flow separation on the application of plasma. The Reynolds stress increases significantly with plasma excitations relative to the baseline case, suggesting an amplification of the flow instability. Moreover, an interesting phenomenon is found that the Reynolds stress magnitude reduces with momentum within the range of $C_\mu = 0.005$

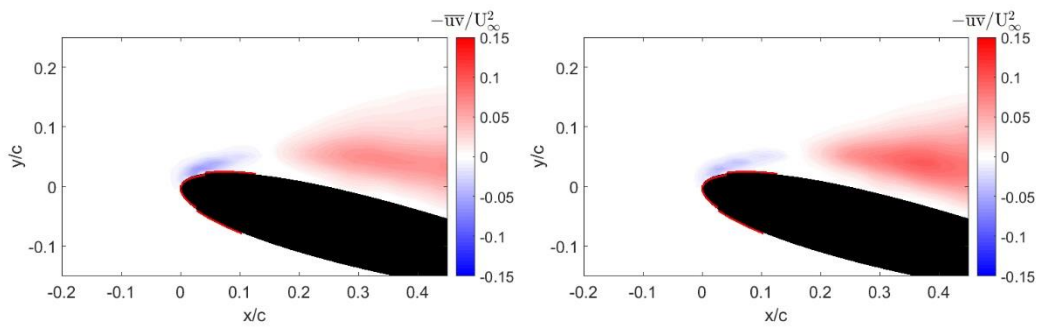
$\sim 0.125\%$. With periodic excitation of plasma actuator at momentum coefficient of $C_\mu = 0.000025\%$, a concentrated Reynolds-stress region is observed away from aerofoil surface, which is very similar to the baseline flow. However, it seems that the maximum Reynolds stress magnitude is slightly larger than that for the uncontrolled case. So perturbations are still caused to the flow around the aerofoil even the plasma momentum is pretty low.

R.m.s. velocity fluctuations for the flow fields around a NACA 0012 aerofoil controlled with periodic excitations of plasma actuator at various momentum coefficients (duty cycles) and frequency of $f^+ = 1$ were calculated and presented in Figure 7-23. Within the plasma momentum range of $C_\mu = 0.005\% \sim 0.125\%$, the r.m.s. streamwise velocity peak decreases with plasma momentum and the vertical position of the peak moves closer to the upper surface, which is further confirmed by the r.m.s. streamwise fluctuation velocity profiles at $x/c = 0.3$ in Figure 7-24. This is in good agreement with the trend of Reynolds stress in Figure 7-22. At a smaller plasma momentum of $C_\mu = 0.0005\%$ ($\tau = 0.2\%$), the streamwise fluctuation velocity profile has a peak of $u'/U_\infty \approx 0.45$ and locates at $y/c \approx 0.062$, almost the same as that with plasma momentum of $C_\mu = 0.005\%$ ($\tau = 2\%$). However, the r.m.s. streamwise velocity fluctuation at plasma momentum of $C_\mu = 0.0005\%$ is smaller in the near-wall region of $0 \leq y/c \leq 0.04$ compared with that at larger plasma momentum coefficient, corresponding to a weaker momentum exchange effect in the near-wall region, which might account for a less effective flow separation control effect. The r.m.s. streamwise velocity profile for the flow with plasma excitations at momentum coefficient of $C_\mu = 0.000025\%$ is slightly greater than that for the baseline flow, consistent with the observations that plasma excitation at momentum coefficient of $C_\mu = 0.000025\%$ has only minimal effect on the control of the flow separation around NACA 0012 aerofoil at angle of attack of 14° .



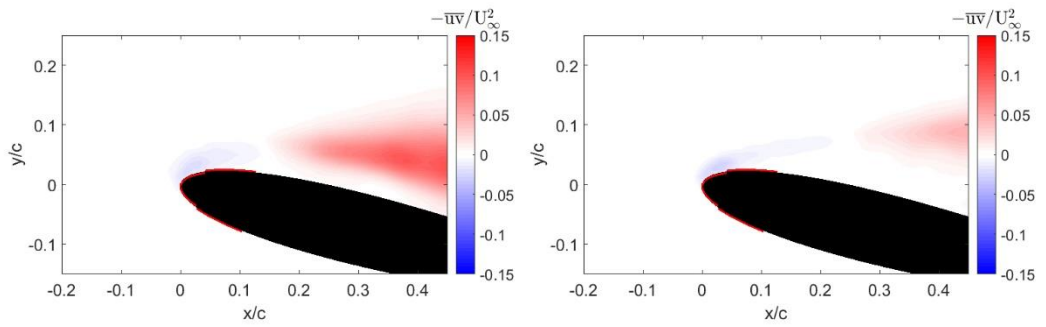
(a) $C_\mu = 0.125\%$

(b) $C_\mu = 0.0625\%$



(c) $C_\mu = 0.0125\%$

(d) $C_\mu = 0.005\%$



(e) $C_\mu = 0.0005\%$

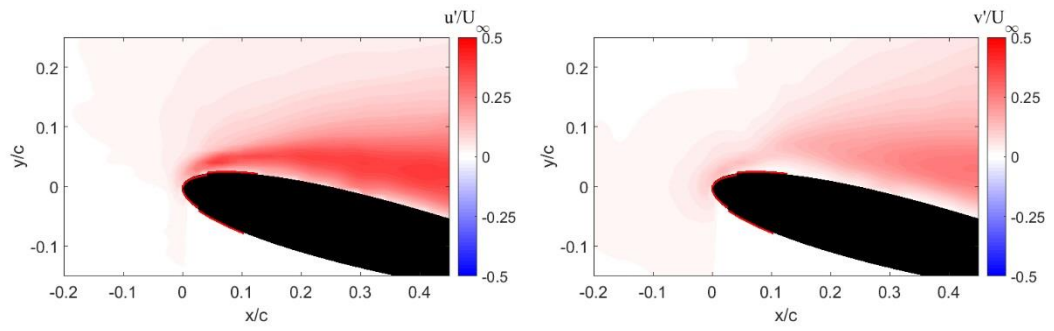
(f) $C_\mu = 0.000025\%$

Figure 7-22 Normalized Reynolds stress contour for the controlled case with unsteady actuation of DBD plasma actuator at different momentum coefficients.

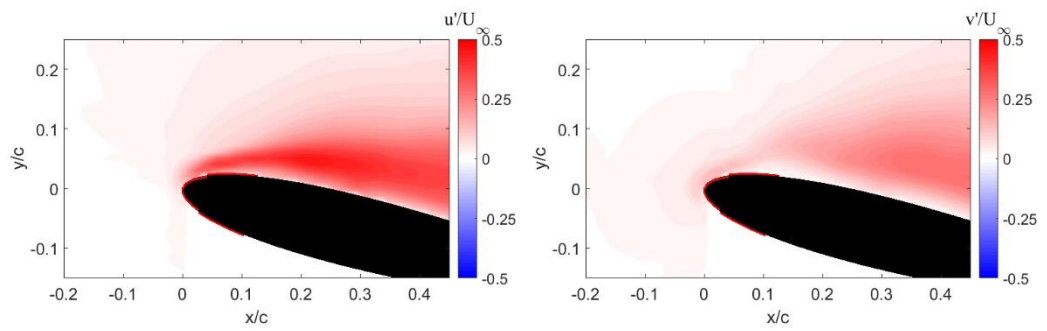
The pulsed excitation frequency is $f_e/f_0 = 1$. The freestream velocity is

$U_\infty = 3\text{m/s}$ and angle of attack is $\alpha = 14^\circ$.

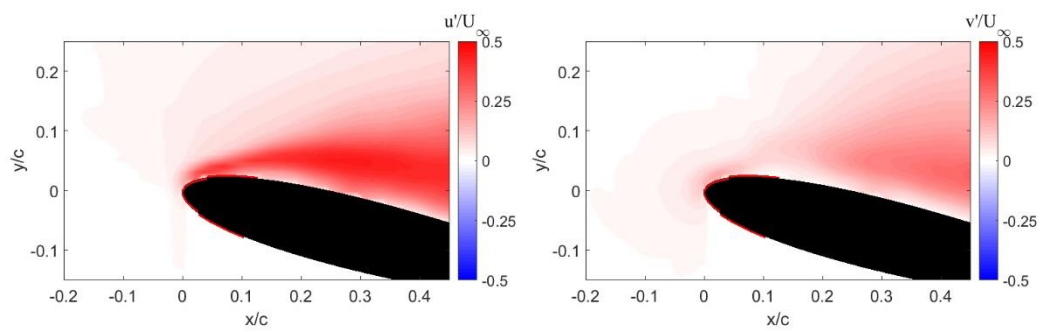
a) $C_\mu = 0.125\%$



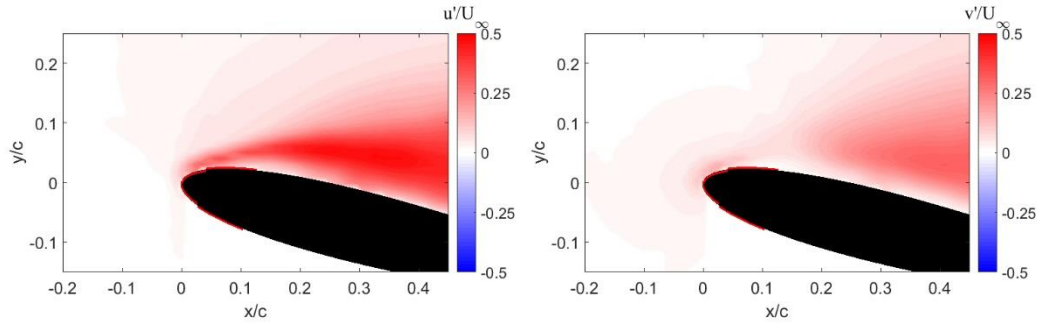
b) $C_\mu = 0.0625\%$



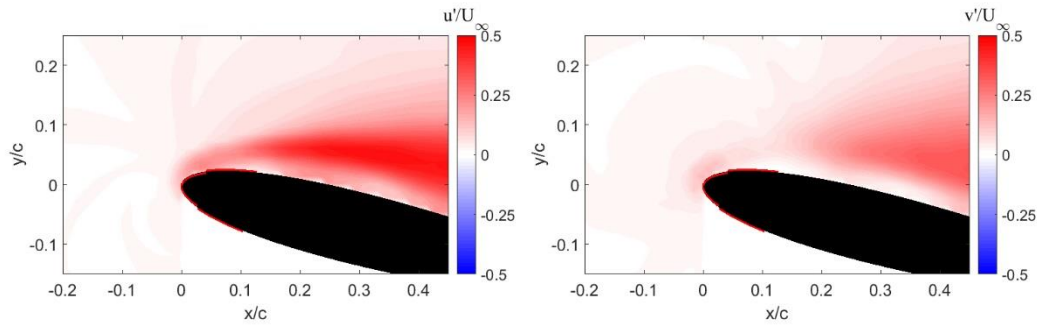
c) $C_\mu = 0.0125\%$



d) $C_\mu = 0.005\%$



e) $C_\mu = 0.0005\%$



f) $C_\mu = 0.000025\%$

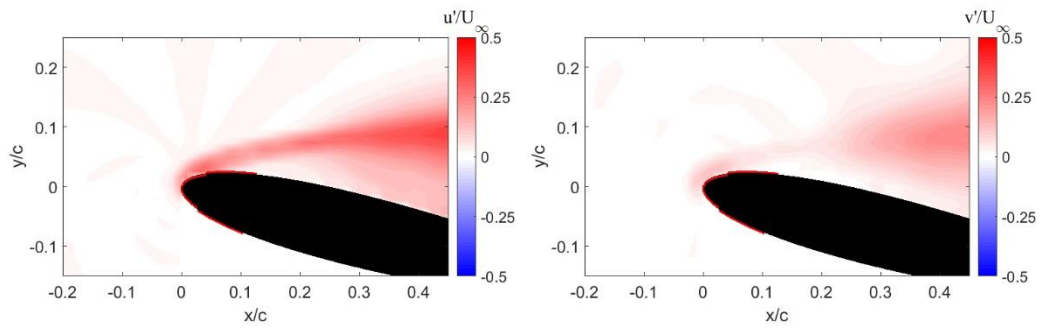


Figure 7-23 Contours of streamwise (left column) and vertical (right column) root-mean-square velocities with unsteady activation of plasma at different momentum coefficients.

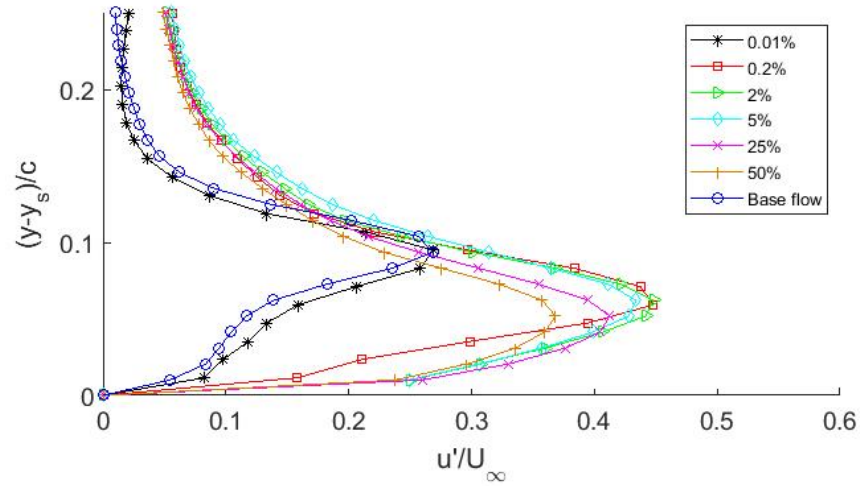


Figure 7-24 Root-mean-squared streamwise fluctuation velocity profiles at $x/c = 0.3$ for the baseline case and controlled cases with unsteady actuation of DBD plasma actuator at different momentum coefficients (duty cycles). Excitation frequency is $f^+ = 1$. The freestream velocity is $U_\infty = 3\text{m/s}$ and angle of attack is $\alpha = 14^\circ$.

Phase-averaged analysis was also conducted to investigate the transient flow behaviours with periodic excitations of plasma actuator at different duty cycles and frequency of $f^+ = 1$. At a momentum coefficient of $C_\mu = 0.125\%$, a slight flow separation is observed at phase of 0° , which can be demonstrated by the streamline in Figure 7-25 (a). After plasma is turned on, the tail of the separated shear layer rolls up into a vortex and travels downstream, shown in Figure 7-25 (b) and (c). The entrainment by this vortex results in velocity changes around it, which leads to a change in static pressure. In other words, a favourable pressure gradient is formed due to the upwash effect of the vortex, as a result, an attached flow is developed above the aerofoil surface at phase of 135° and 180° . By the way, the concentrated vorticity could help reattach the separated flow as well. Actually, the concentrated vorticity consists of a number of small-scale vortical structures as demonstrated in Section 6.5, however, these vortical structures are filtered out by phase-average. It seems that a second vortex is formed at phase of 225° due to a release of the concentrated vorticity after plasma is turned off. This second vortex grows and travels downstream at phase of 270° and 315° , however, it seems that smaller modifications are made to the flow field by this second vortex compared with the

“starting vortex” formed when plasma is activated, as indicated by the vertical velocity and streamline, therefore, it might play a smaller role in flow separation control at this frequency-duty cycle combination than the vortex generated on the activation of plasma. At smaller plasma momentum coefficients of $C_{\mu} = 0.0625\%$ and 0.0125% , a similar phenomenon is observed, however, a severer separation is developed in the beginning of each plasma forcing cycle, accordingly, a larger region of separated flow will roll up when plasma excitation is applied, resulting in a larger and stronger vortex. In addition, the vortex formed after plasma is turned off is smaller and the two vortices generated within each forcing cycle are more densely packed at smaller duties.

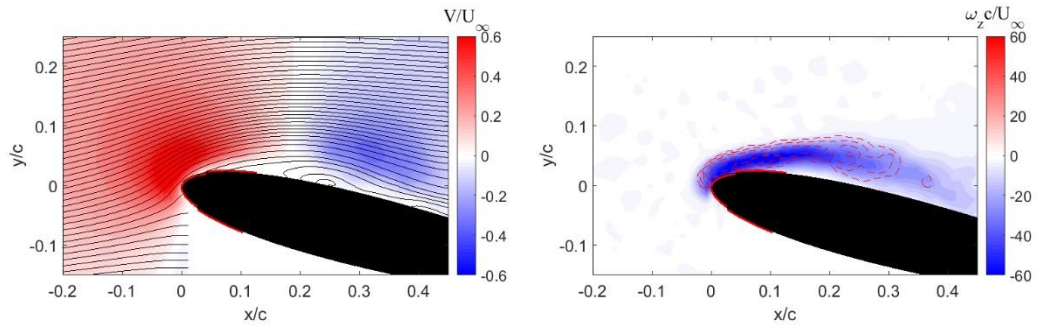
At a smaller plasma momentum of $C_{\mu} = 0.005\%$, the phase-averaged streamlines over the plasma forcing cycle suggest that a slightly heavier separation is developed due to a smaller duty of plasma excitation. It is interesting to note that only one vortex is formed over the entire plasma forcing cycle, probably this is because the duty cycle is so small that the evacuated separated shear layer and the released concentrated vorticity layer merge together, which roll up into one vortex. When plasma is excited at a momentum coefficient of $C_{\mu} = 0.0005\%$, a similar process is observed from the phase-averaged analysis except that reverse flow always exists all over the whole plasma forcing cycle. This results in a separated flow in time-averaged sense in Figure 7-20 (e).

When plasma excitations are applied at a pretty small momentum coefficient of $C_{\mu} = 0.000025\%$, as can be seen from the eight phases over a plasma forcing cycle, the excitation of plasma is not able to cause a rollup of the separated shear layer and flow always behaves similarly to the baseline flow. So plasma excitation at such a small momentum coefficient is not that effective in flow separation control and only has minimal effect on the control of flow separation around a NACA 0012 aerofoil at post-stall angle of attack of 14° .

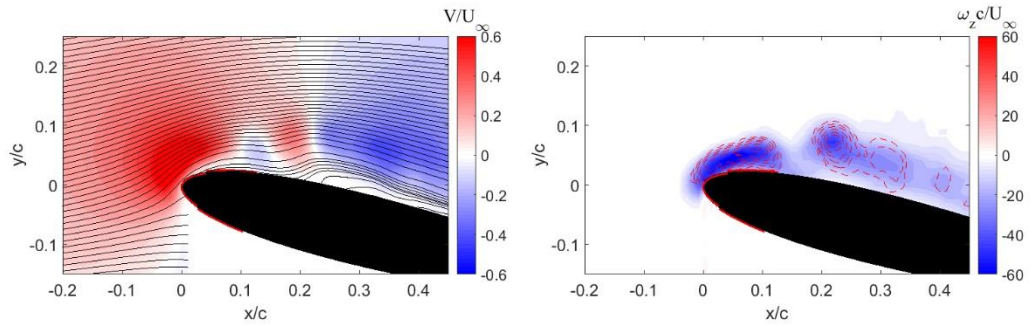
Figure 7-31 shows the trajectories of the two vortices generated by plasma excitations at momentum coefficients of $C_{\mu} = 0.0125\%$, 0.025% , 0.0625% and 0.125% . The vortex trajectories were obtained from the phase-averaged data. As can be clearly seen, the trajectories of the “starting vortex” are similar even when plasma excitations were applied with different durations. Moreover, the “starting

vortex” is always formed around 45° in phase after plasma is activated when aerofoil is set at angle of attack of $\alpha = 14^\circ$ and plasma excitation is exerted at $f^+ = 1$. As for the second vortex, it occurs at quite different timing with different plasma excitation duty cycles. Specifically, it always starts to appear $36^\circ \sim 50^\circ$ in phase after plasma is turned off. So it takes $36^\circ \sim 50^\circ$ in phase for the second vortex to form. Since this vortex is produced due to a switch-off of plasma, it could be termed as “stopping vortex”. The slope of the vortex trajectory can be interpreted as convection velocity. Generally, the “stopping vortex” travels faster for the controlled cases with larger plasma durations.

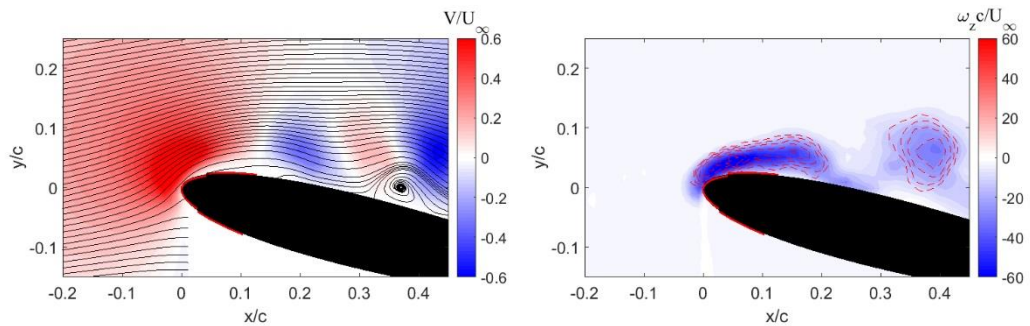
a) $\varphi = 0^\circ$



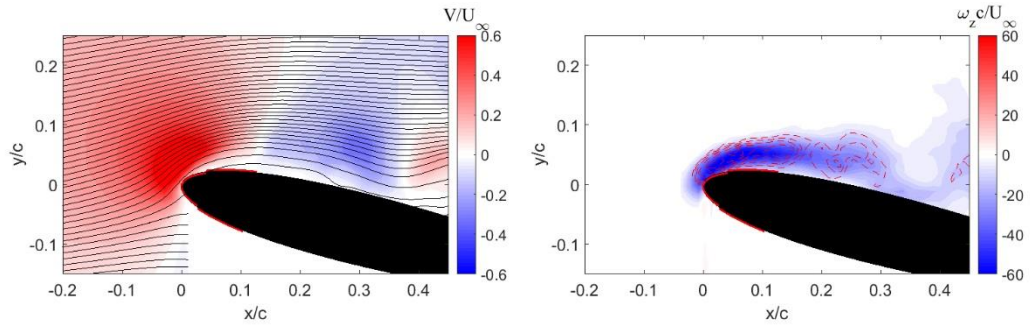
b) $\varphi = 45^\circ$



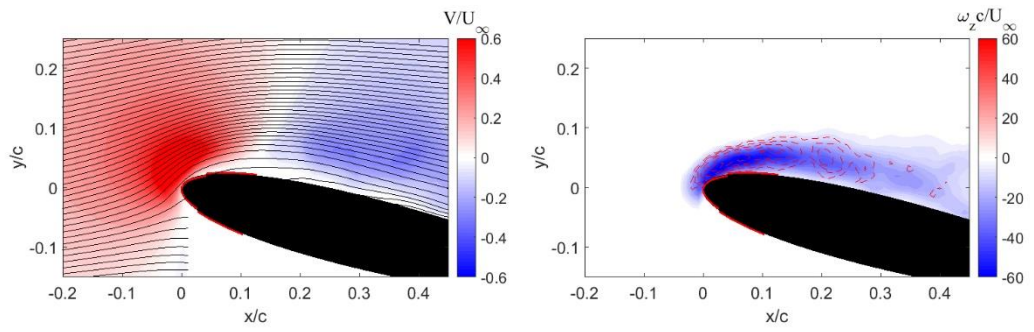
c) $\varphi = 90^\circ$



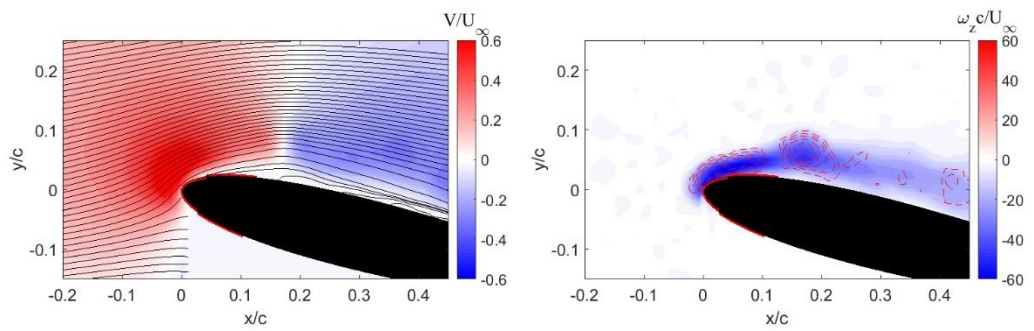
d) $\varphi = 135^\circ$



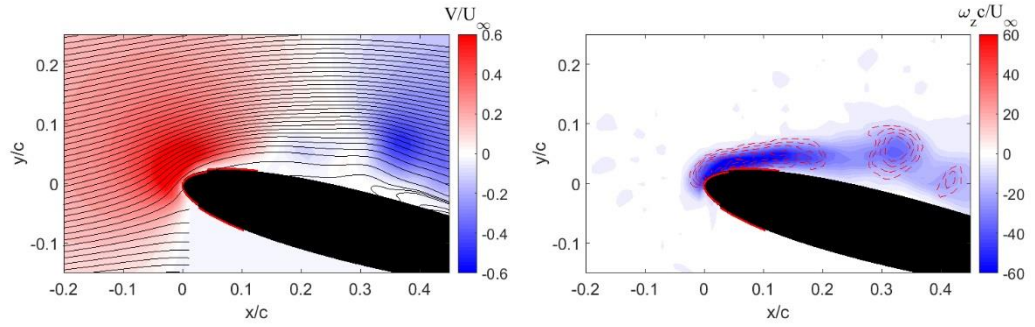
e) $\varphi = 180^\circ$



f) $\varphi = 225^\circ$



g) $\varphi = 270^\circ$



h) $\varphi = 315^\circ$

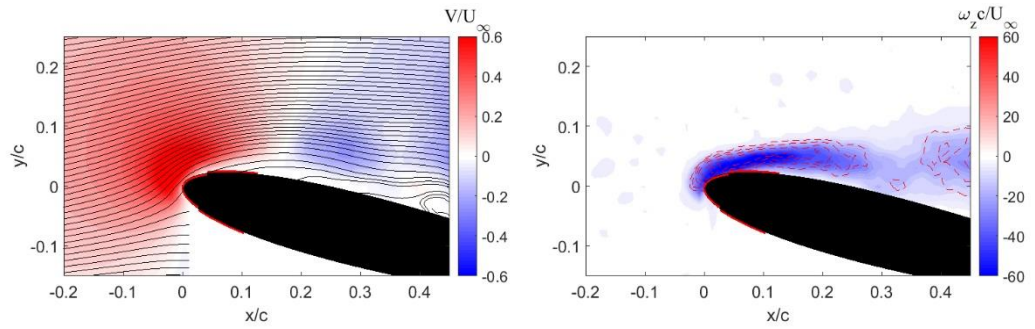
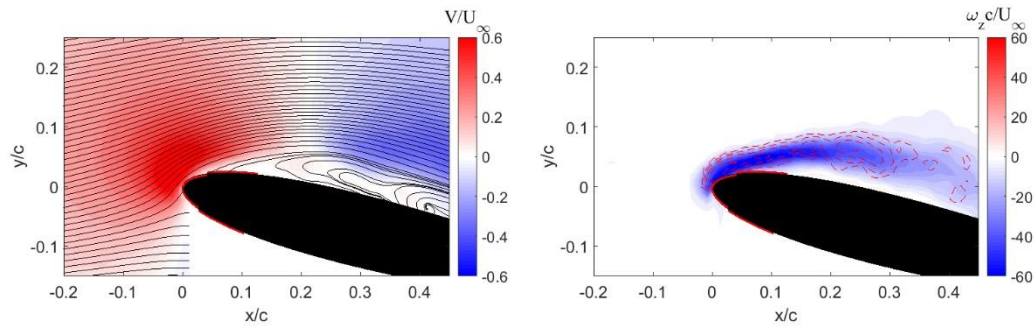
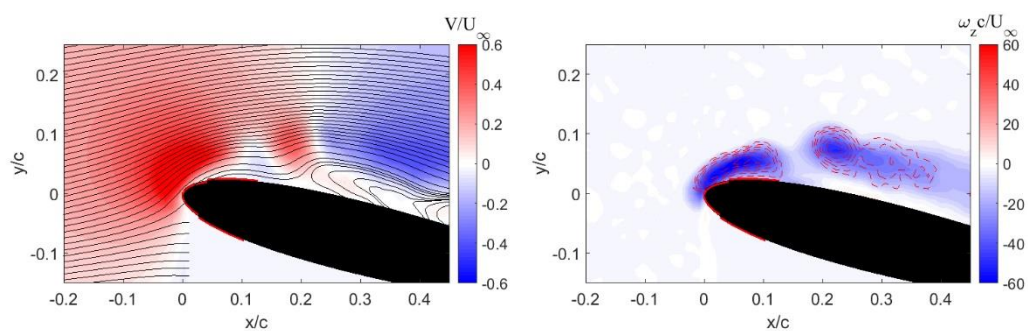


Figure 7-25 Phased-averaged analysis of the flow field around NACA 0012 aerofoil at angle of attack of $\alpha = 14^\circ$ with plasma excitations at frequency of $f^+ = 1$ and momentum coefficient of $C_\mu = 0.125$. Freestream velocity is $U_\infty = 3\text{m/s}$.

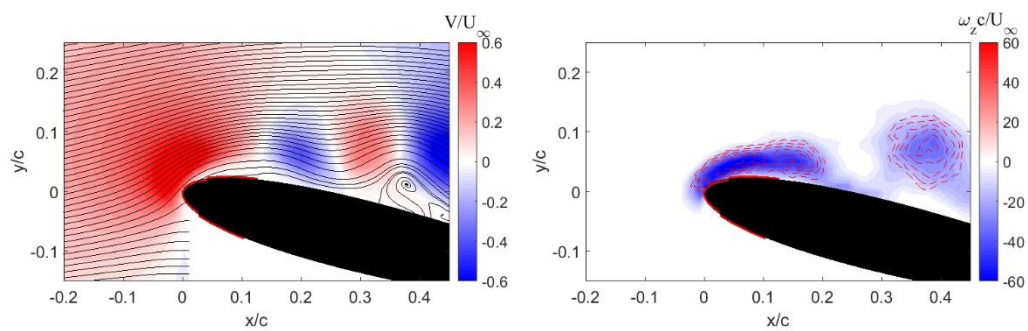
a) $\varphi = 0^\circ$



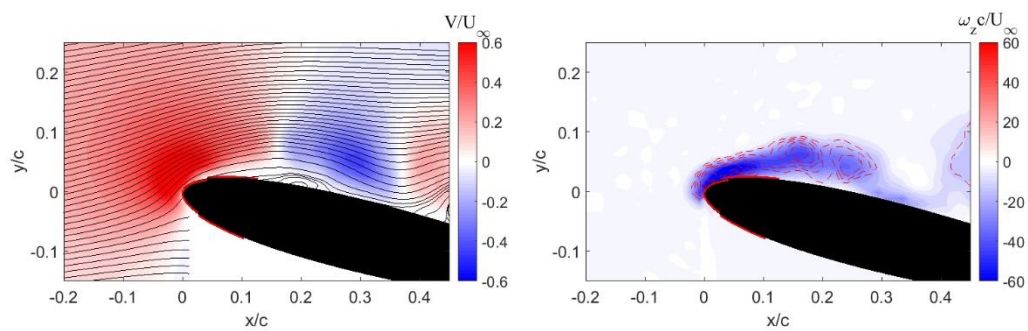
b) $\varphi = 45^\circ$



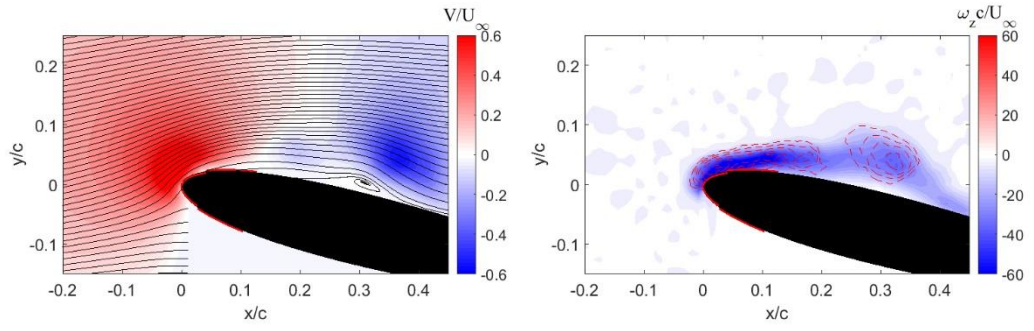
c) $\varphi = 90^\circ$



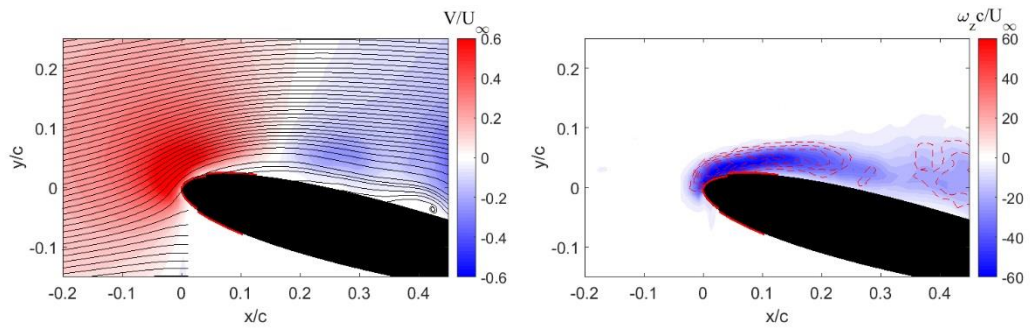
d) $\varphi = 135^\circ$



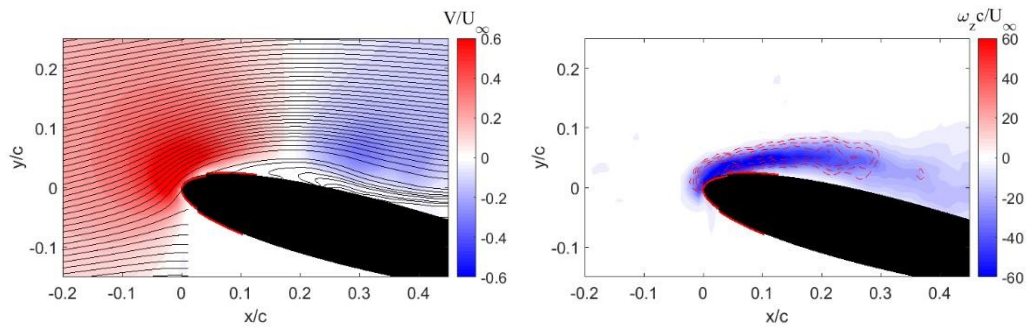
e) $\varphi = 180^\circ$



f) $\varphi = 225^\circ$



g) $\varphi = 270^\circ$



h) $\varphi = 315^\circ$

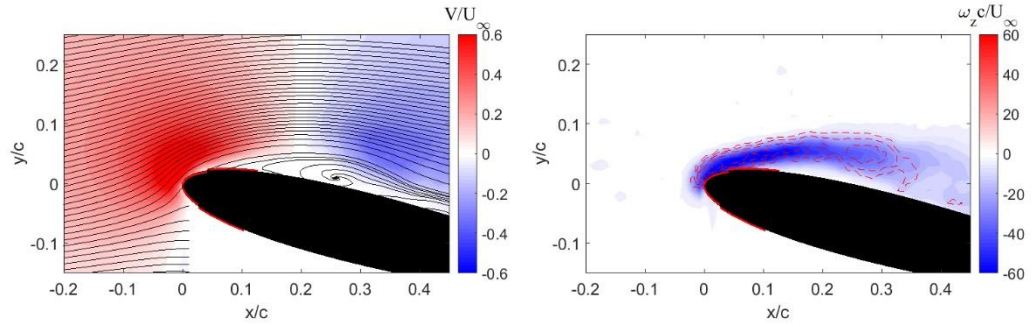
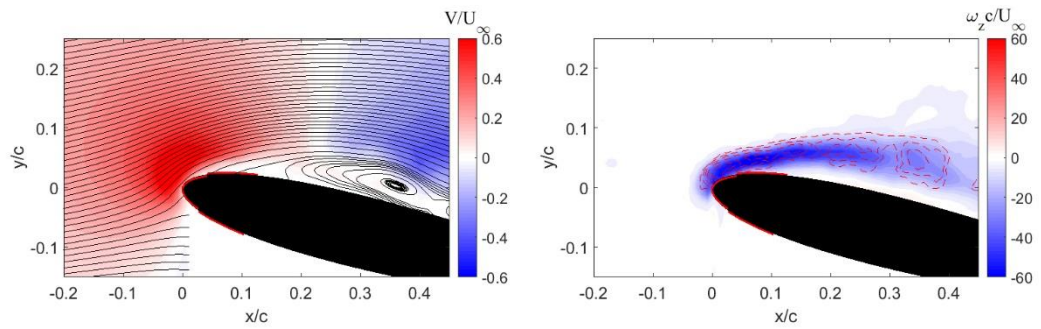
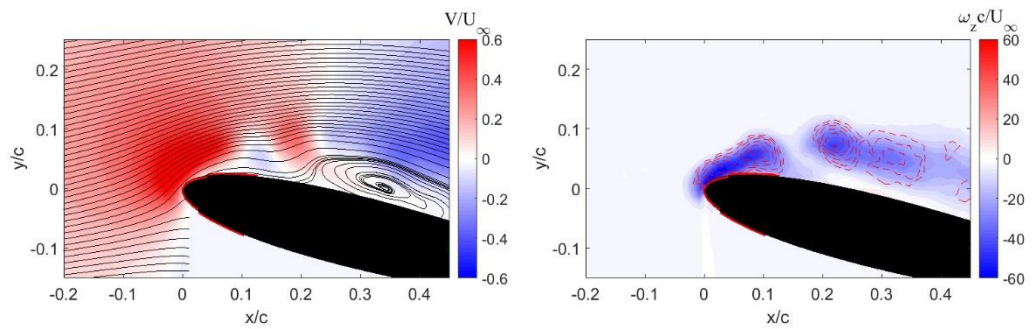


Figure 7-26 Phased-averaged analysis of the flow field around NACA 0012 aerofoil at angle of attack of $\alpha = 14^\circ$ with plasma excitations at frequency of $f^+ = 1$ and momentum coefficient of $C_\mu = 0.0625\%$.

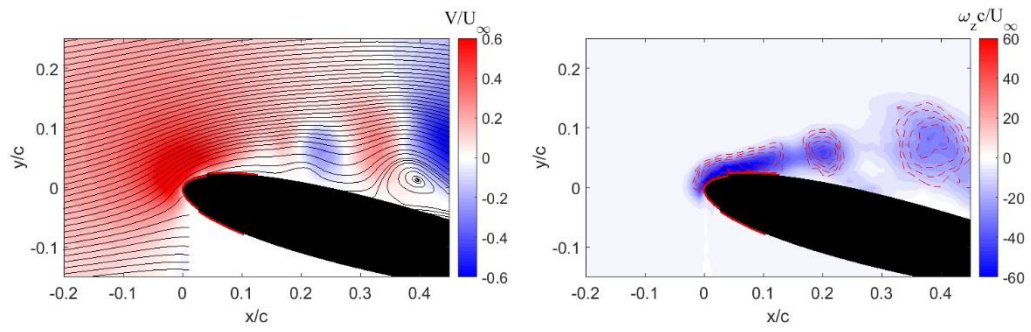
a) $\varphi = 0^\circ$



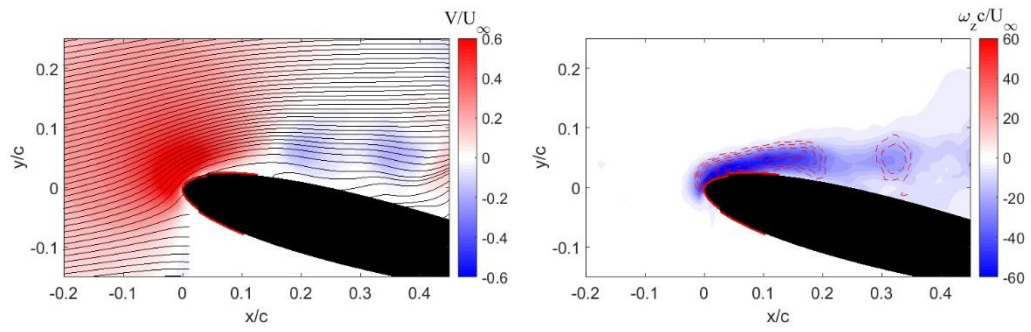
b) $\varphi = 45^\circ$



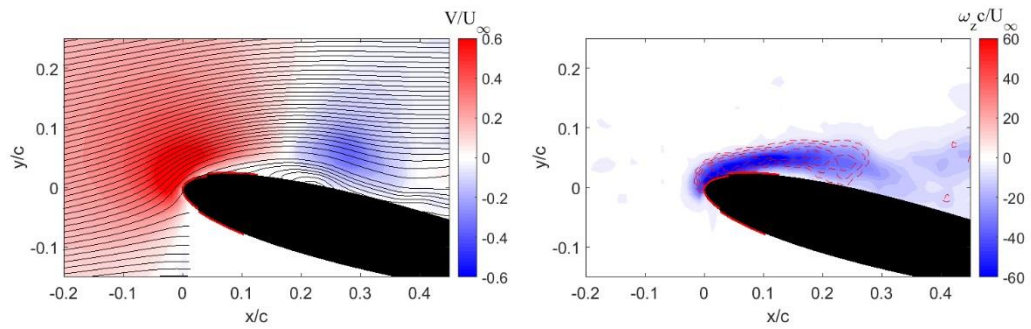
c) $\varphi = 90^\circ$



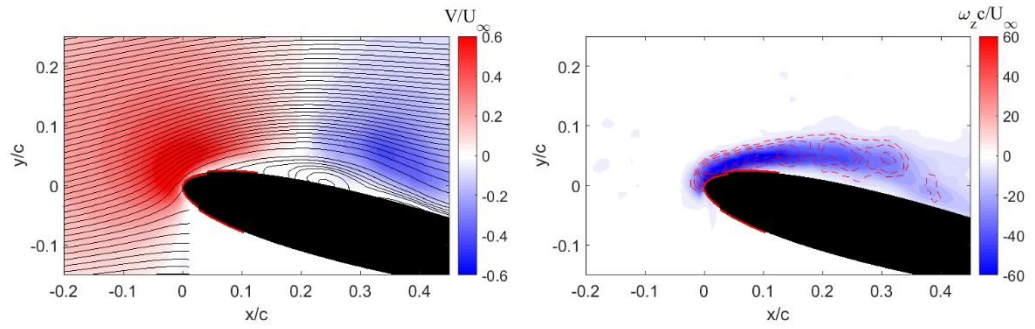
d) $\varphi = 135^\circ$



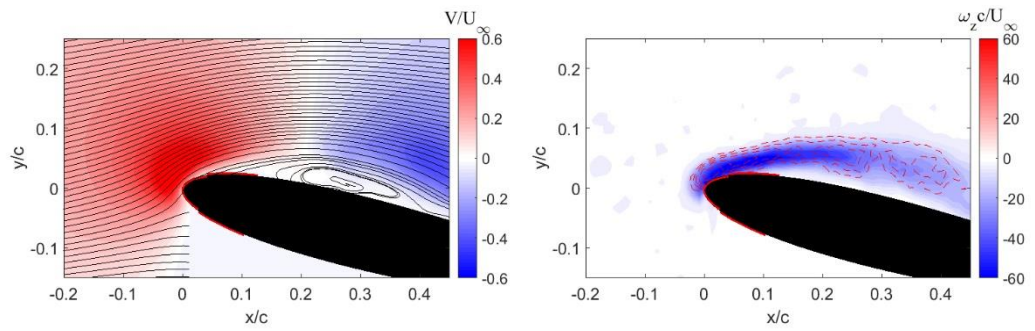
e) $\varphi = 180^\circ$



f) $\varphi = 225^\circ$



g) $\varphi = 270^\circ$



h) $\varphi = 315^\circ$

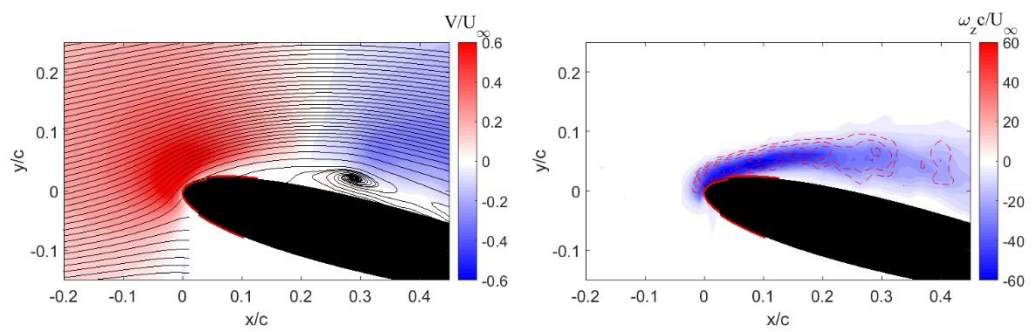
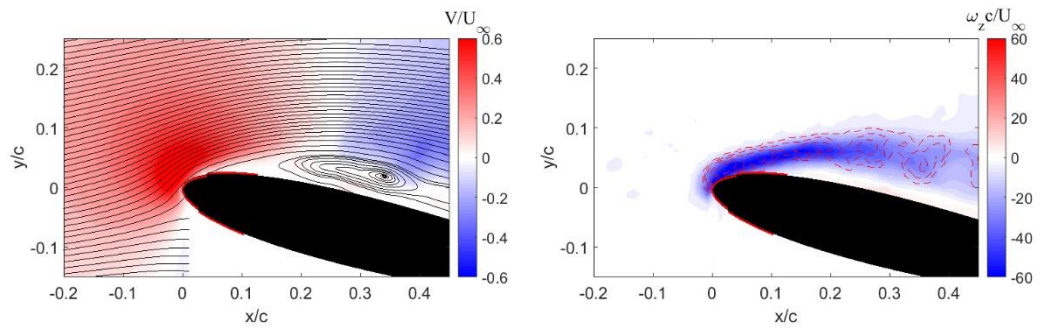
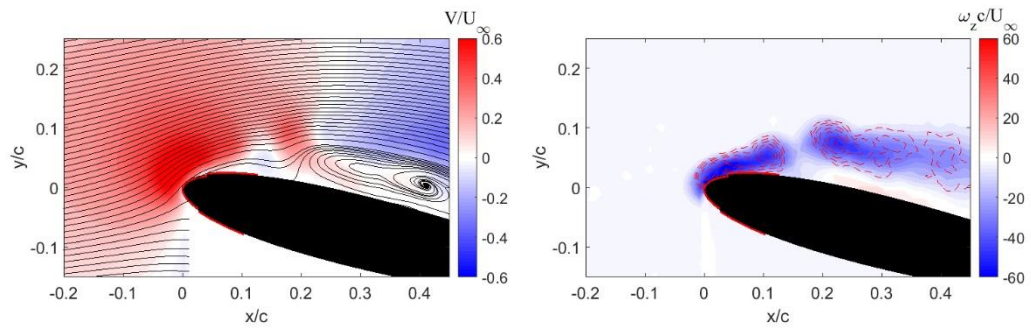


Figure 7-27 Phased-averaged analysis of the flow field around NACA 0012 aerofoil at angle of attack of $\alpha = 14^\circ$ with plasma excitations at frequency of $f^+ = 1$ and momentum coefficient of $C_\mu = 0.0125\%$.

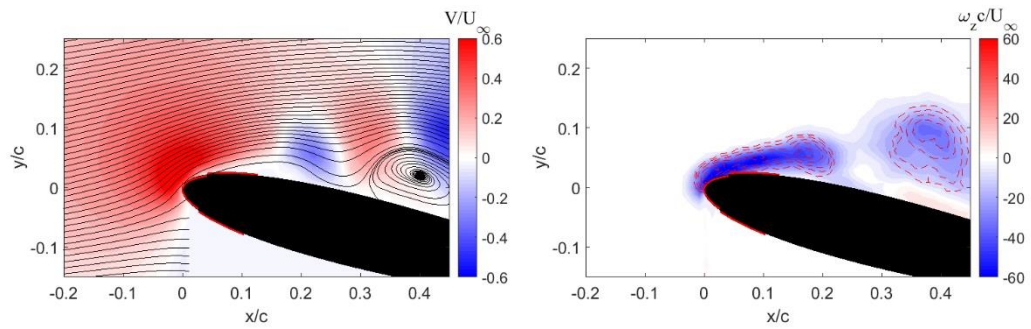
a) $\varphi = 0^\circ$



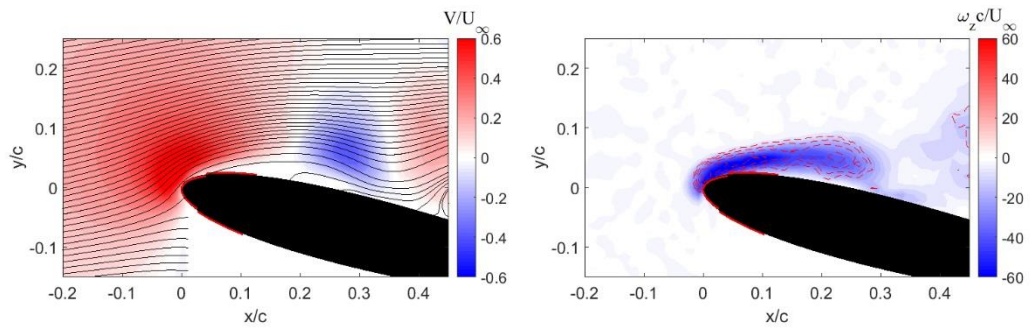
b) $\varphi = 45^\circ$



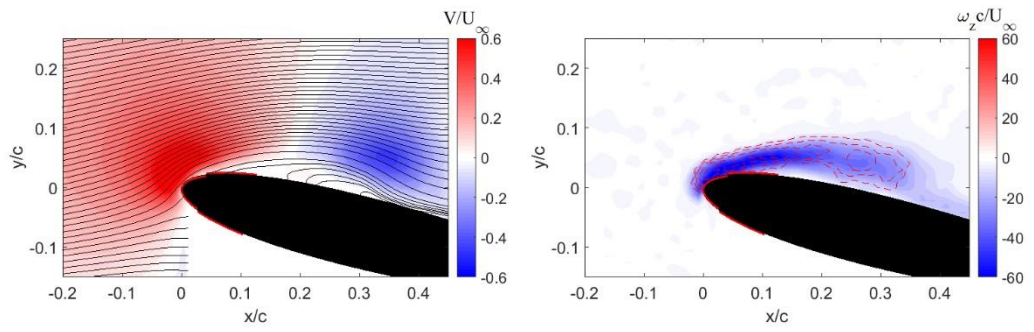
c) $\varphi = 90^\circ$



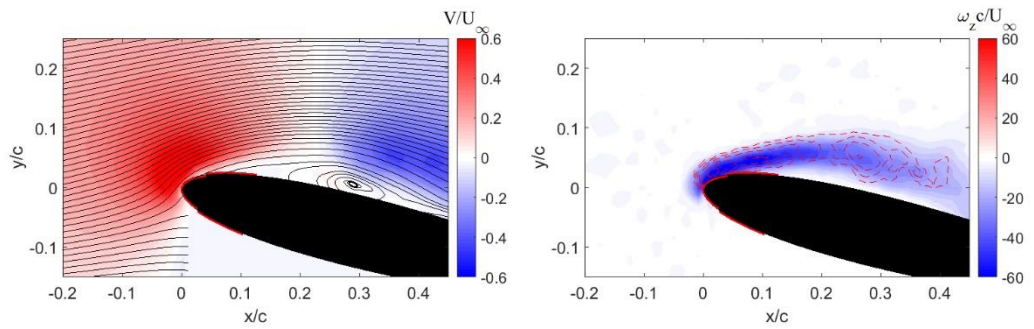
d) $\varphi = 135^\circ$



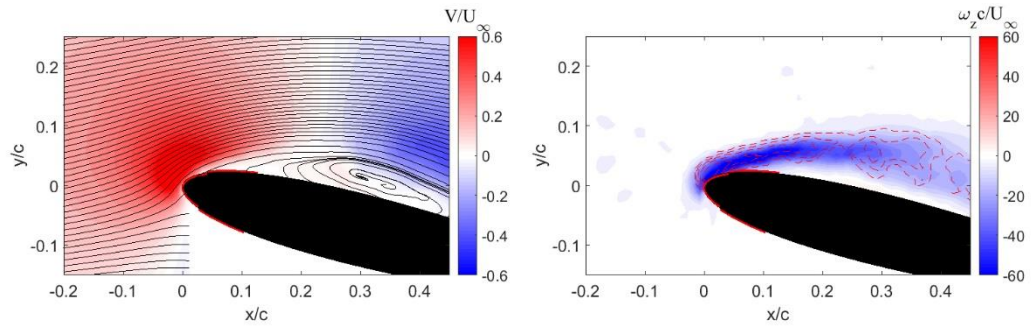
e) $\varphi = 180^\circ$



f) $\varphi = 225^\circ$



g) $\varphi = 270^\circ$



h) $\varphi = 315^\circ$

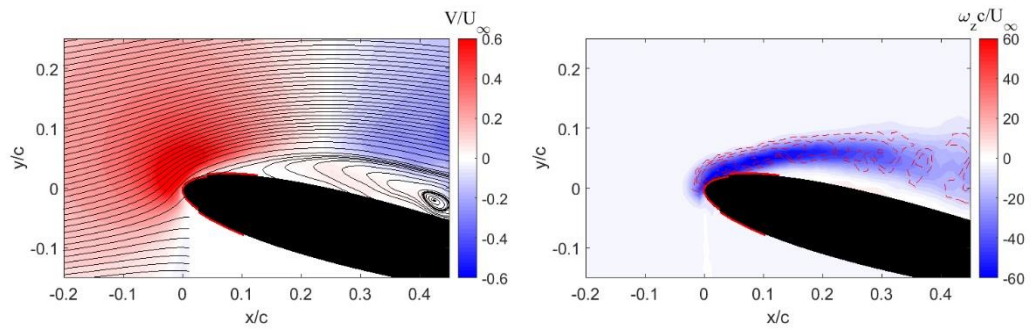
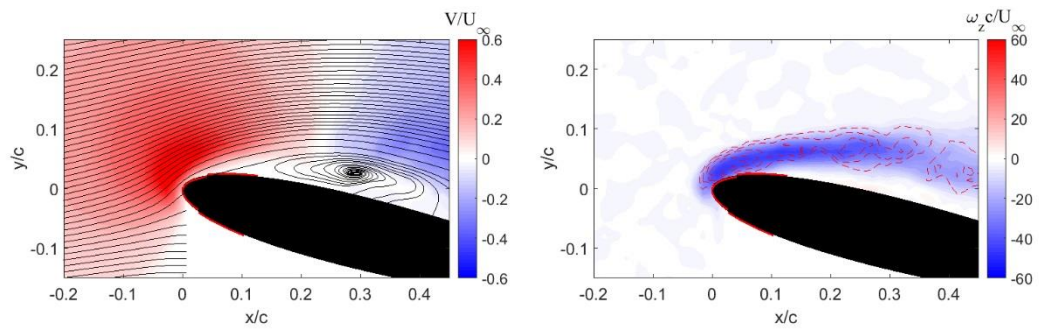
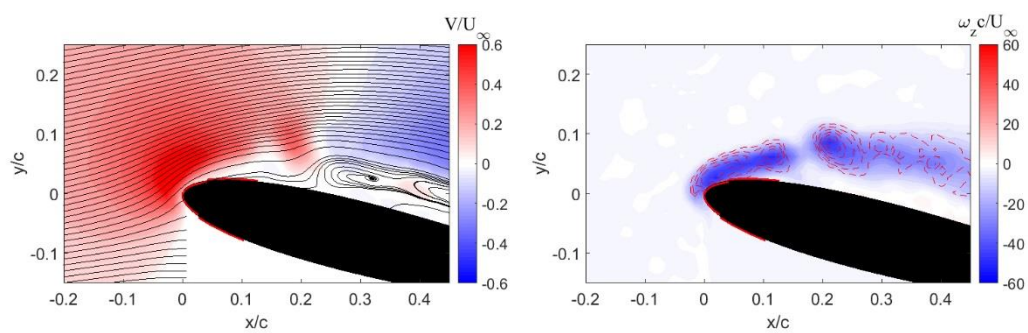


Figure 7-28 Phased-averaged analysis of the flow field around NACA 0012 aerofoil at angle of attack of $\alpha = 14^\circ$ with plasma excitations at frequency of $f^+ = 1$ and momentum coefficient of $C_\mu = 0.005\%$.

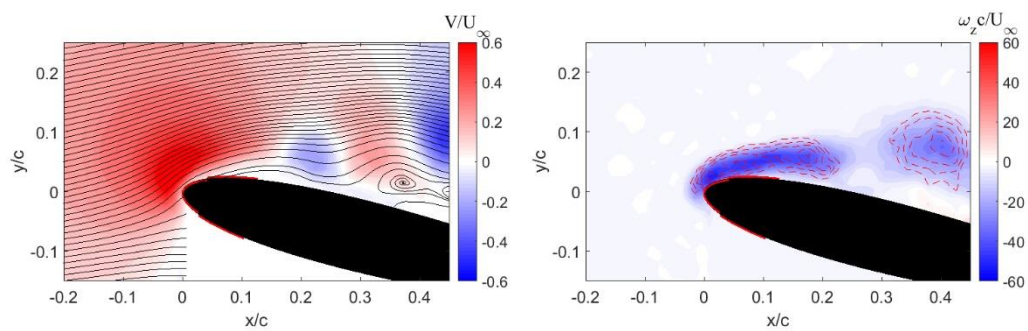
a) $\varphi = 0^\circ$



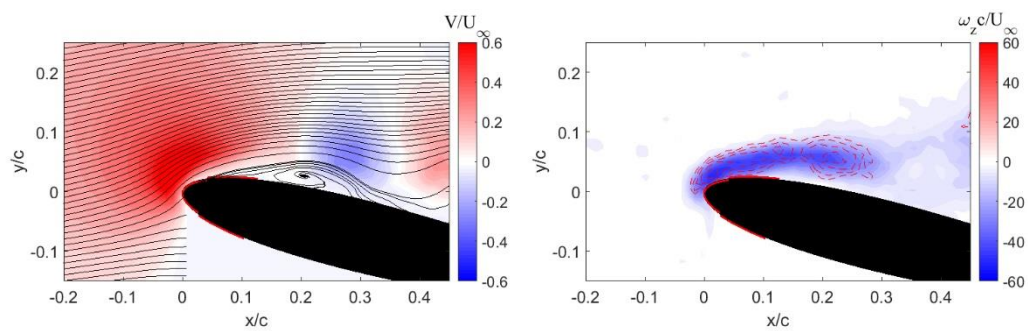
b) $\varphi = 45^\circ$



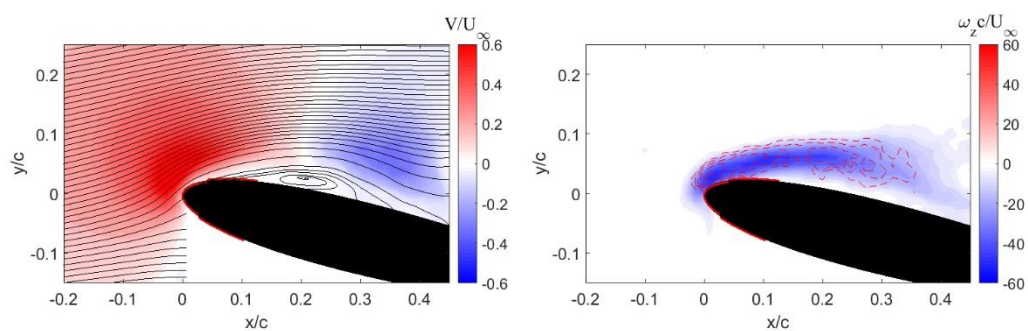
c) $\varphi = 90^\circ$



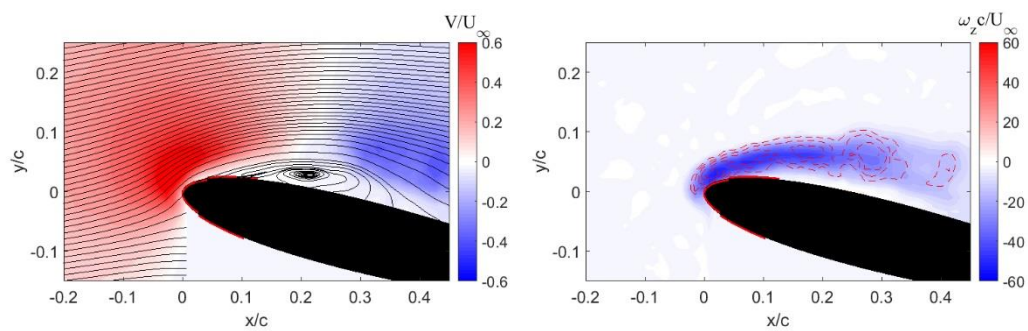
d) $\varphi = 135^\circ$



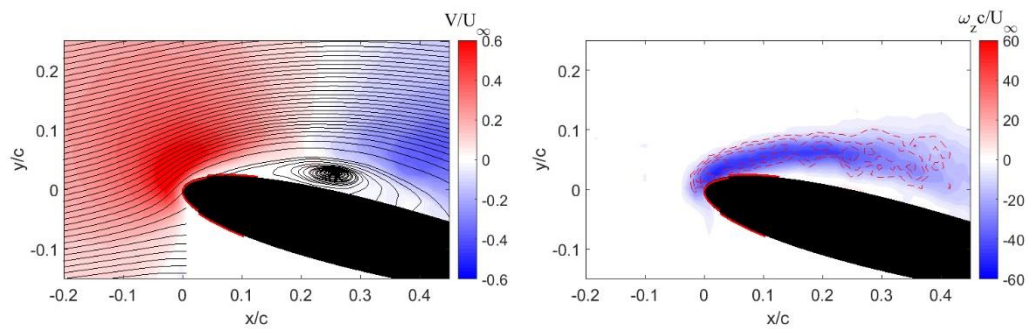
e) $\varphi = 180^\circ$



f) $\varphi = 225^\circ$



g) $\varphi = 270^\circ$



h) $\varphi = 315^\circ$

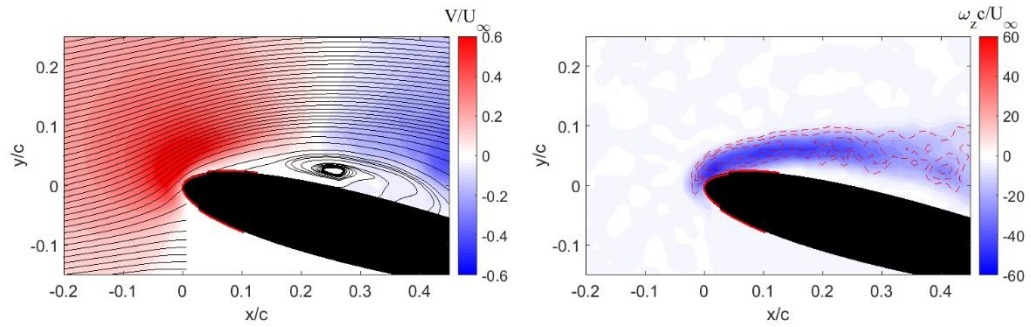
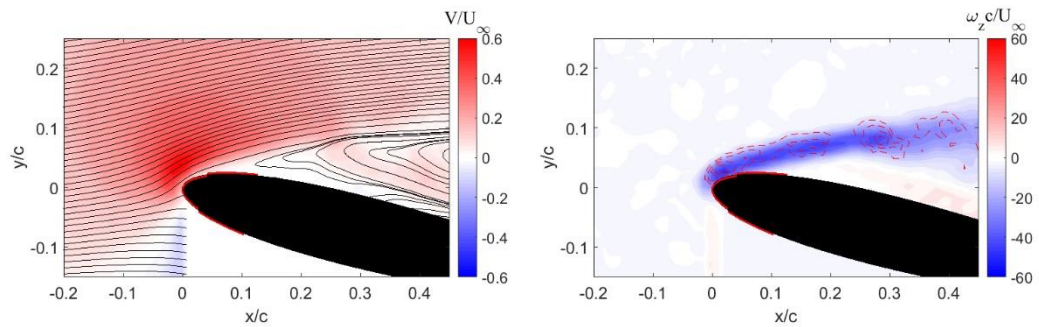
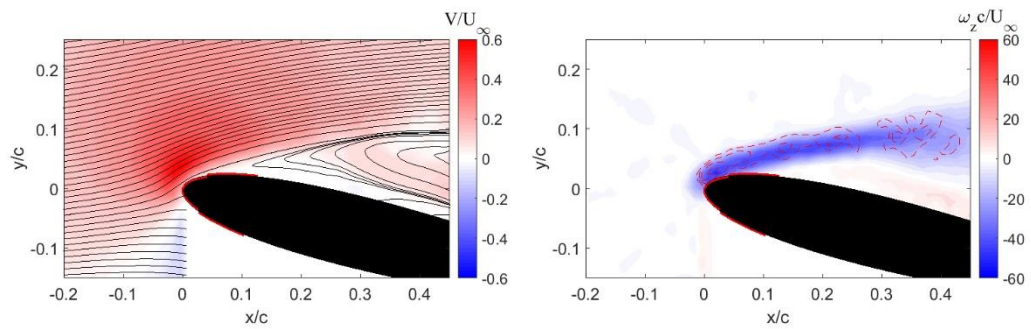


Figure 7-29 Phased-averaged analysis of the flow field around NACA 0012 aerofoil at angle of attack of $\alpha = 14^\circ$ with plasma excitations at frequency of $f^+ = 1$ and momentum coefficient of $C_\mu = 0.0005\%$.

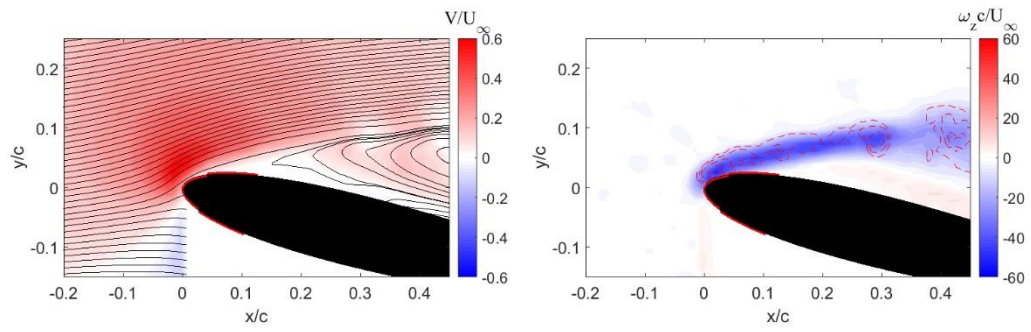
a) $\varphi = 0^\circ$



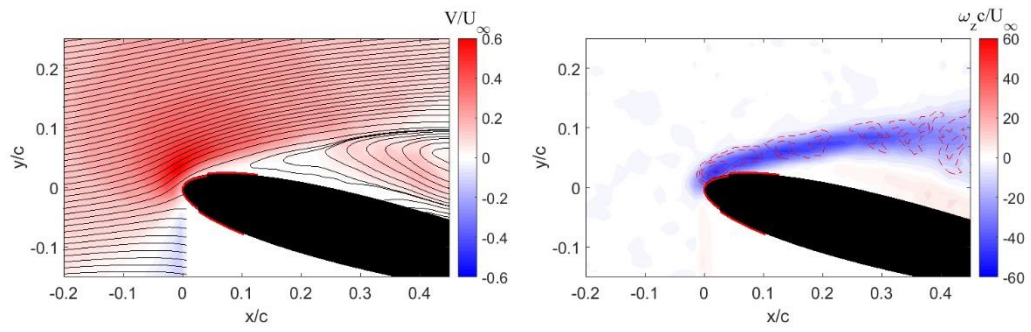
b) $\varphi = 45^\circ$



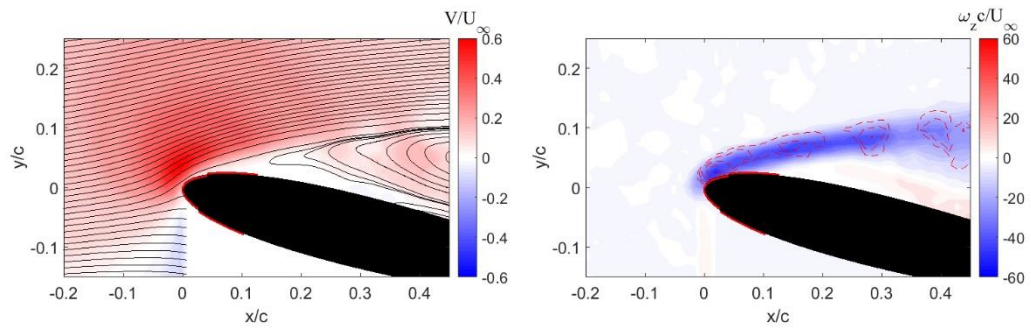
c) $\varphi = 90^\circ$



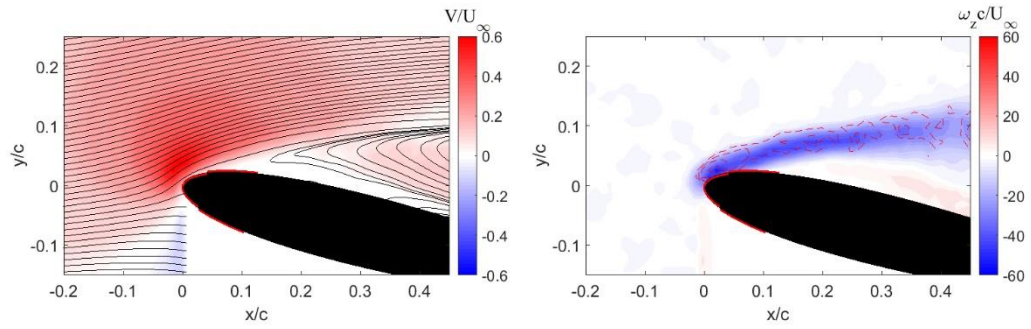
d) $\varphi = 135^\circ$



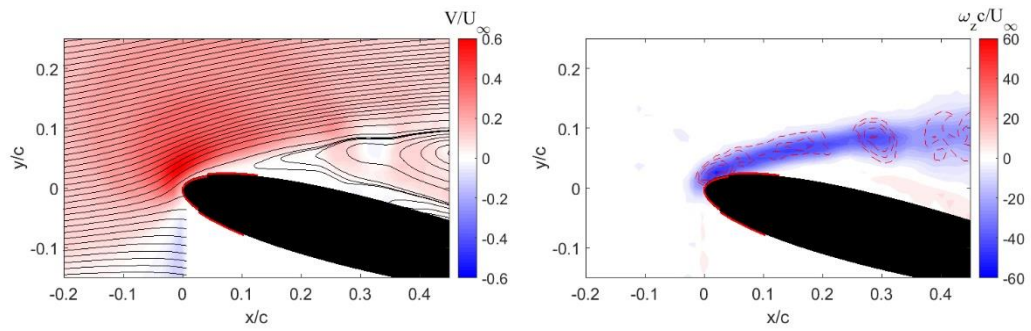
e) $\varphi = 180^\circ$



f) $\varphi = 225^\circ$



g) $\varphi = 270^\circ$



h) $\varphi = 315^\circ$

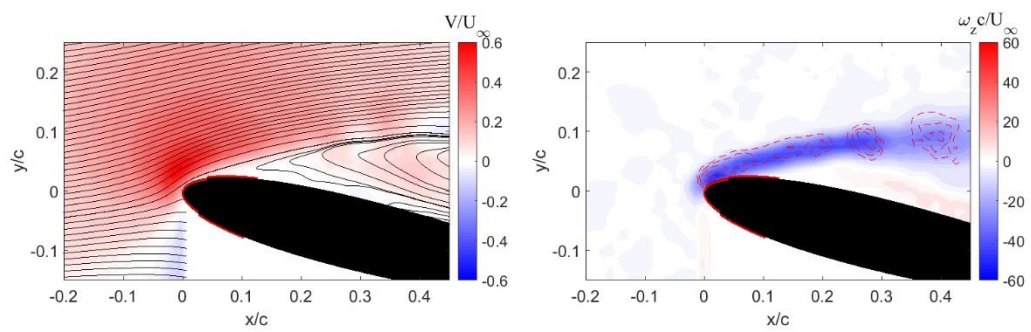


Figure 7-30 Phased-averaged analysis of the flow field around NACA 0012 aerofoil at angle of attack of $\alpha = 14^\circ$ with plasma excitations at frequency of $f^+ = 1$ and momentum coefficient of $C_\mu = 0.000025\%$.

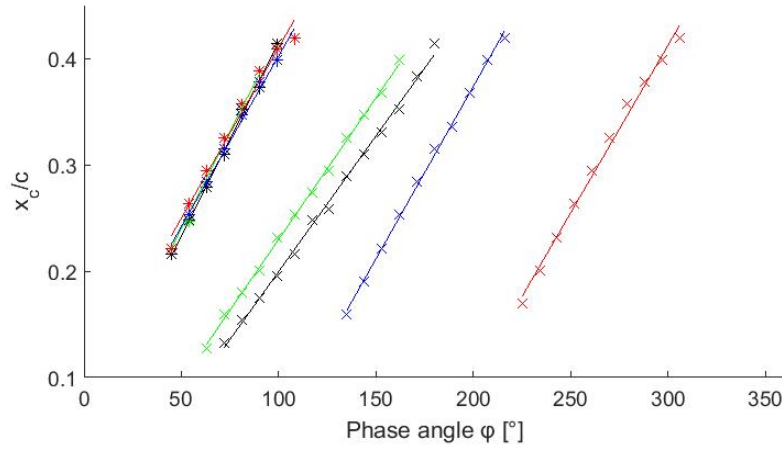


Figure 7-31 Trajectories of vortices created by plasma excitations at $f^+ = 1$ and various duty cycles. *, first vortex; x, second vortex. The colours of green, black, blue and red represent the controlled cases with plasma excitations at momentum coefficients of $C_\mu = 0.0125\%$, 0.025% , 0.0625% and 0.125% respectively. The sold lines are linear fits of vortex trajectories.

7.5 Mechanism Exploration

In Chapter 7, periodic activation of a linear plasma synthetic jet actuator was utilized to control the flow separation around NACA 0012 aerofoil at stall and post-stall angles of attack and Reynolds number of 20,000. PIV results demonstrate a better effectiveness in flow separation control than steady actuation of plasma actuator, in the meanwhile, energy consumption could be reduced significantly. In this section, efforts will be made to explore the mechanism for leading-edge flow separation control over an aerofoil with unsteady actuation of a linear plasma synthetic jet actuator.

As shown in previous sections, a train of discrete vortices are created when unsteady actuation of a linear plasma synthetic jet actuator is applied. In the author's point of view, these vortices are the key to understand the mechanism of flow separation control in the current study. Greenblatt and Wygnanski (2000) argue that the excited mixing layer could be treated as a basis for separation control, and it was also demonstrated that periodic excitation is an effective way to control mixing in the mixing layer, in other words, the excitation re-organizes the flow field into a series of large coherent structures which transfers high momentum fluids

across the mixing layer (Oster et al., 1978; Oster and Wygnanski, 1982; Wygnanski and Petersen, 1987). The separation control research over an aerofoil by Wu et al. (1998) also shows that a number of large coherent structures are created when periodic excitations are applied to the post-stall flow around aerofoil, and they believe that flow separation is controlled due to a stronger entrainment by the periodic vortices.

The phase-averaged analysis of the flow field with periodic activation of plasma actuator in Figure 7-7 shows that two vortices are generated over each plasma forcing cycle. Specifically, the separated shear layer rolls up into a large coherent structure after plasma is excited; the concentrated vorticity layer is released and forms another one when plasma is turned off. This is also clearly illustrated in the schematics in Figure 7-32. These vortices alter the flow field fundamentally, they promote stronger entrainment, re-organize the flow field and transfer the momentum between the freestream and the near-wall region. In the meanwhile, the outwards entrainment of the near-wall fluids causes a decrease in static pressure, and the resulting favourable pressure gradient forces the flow to reattach as well. This is believed to be the primary mechanism for flow separation control around aerofoil with periodic activation of a linear plasma synthetic jet actuator in current study. At this frequency-duty cycle combination, flow starts to separate again at phase angle of 270° after both of the vortices travel away and it recovers to a weaker separation relative to the baseline case before the next forcing cycle starts.

Figure 7-33 shows the power spectral analysis of spanwise vorticity at various chordwise positions for the controlled flow field with periodic activation of plasma actuator at frequency of $f^+ = 1$, duty cycle of 10% and angle of attack of 14° . To facilitate the spectrum presentation, the successive spectrum is shifted by a magnitude of 100. The spectrums at different chordwise positions show a clear dominant peak at $f/f_0 = 1$, which corresponds to the excitation frequency, so a frequency lock-in is observed here. This demonstrates that vortices are created during each periodic cycle and they dominate the flow field, confirming vortices by periodic excitation play a significant role in flow separation control. This is further investigated with the vorticity flux $\int \omega_z U dy$ and momentum flux $\int \rho U^2 dy$ against time, which are presented in Figure 7-34. Integration was performed at the

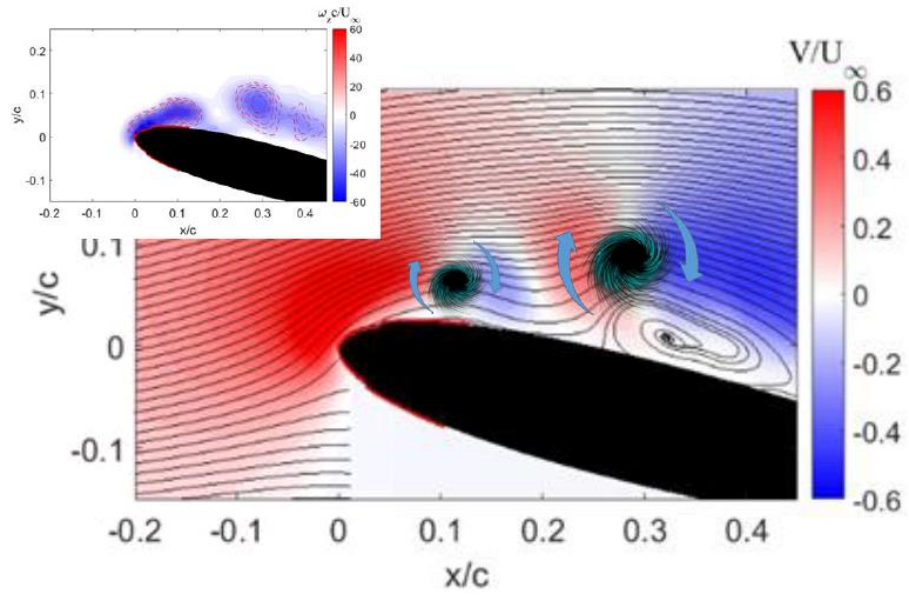


Figure 7-32 Schematic of vortices generated by periodic excitation of plasma actuator. $f^+ = 1$, $\tau = 10\%$, $\alpha = 14^\circ$.

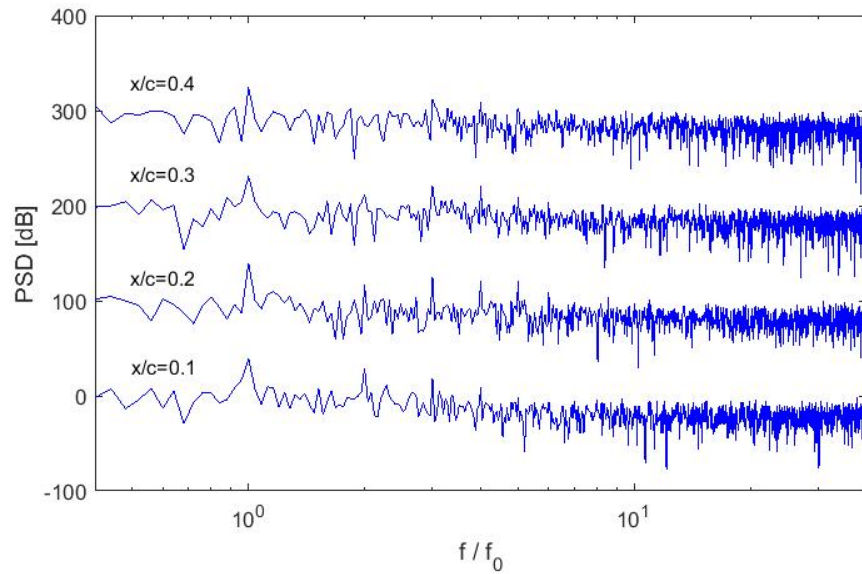
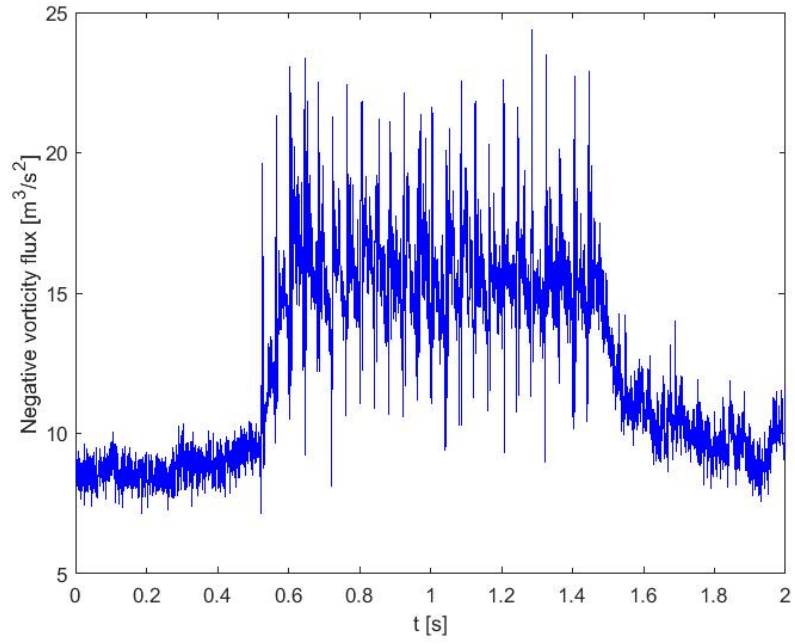
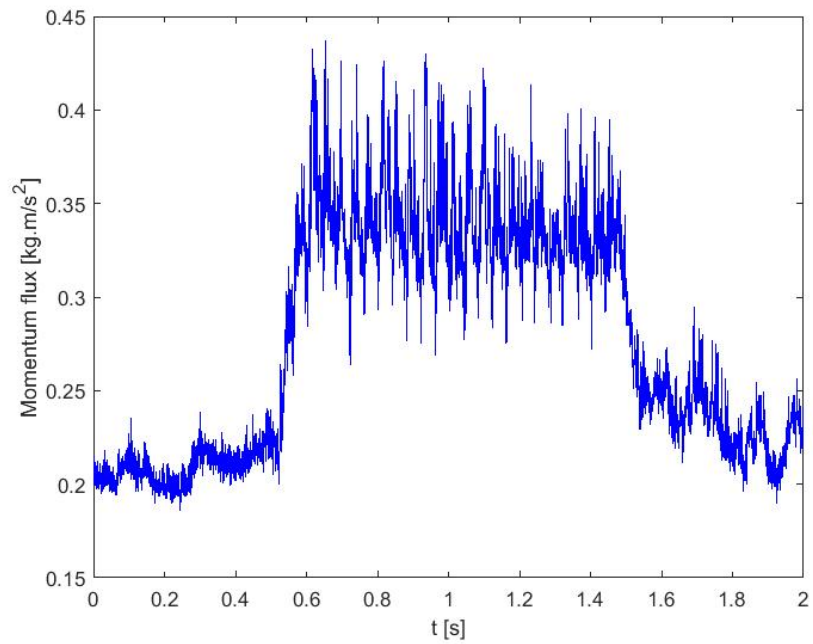


Figure 7-33 Power spectral analysis of spanwise vorticity at chordwise positions from $x/c = 0.1$ to 0.4 with an interval of 0.1 for the controlled flow field with unsteady actuation of plasma actuator at frequency of $f^+ = 1$ and duty cycle of 10% . The angle of attack is $\alpha = 14^\circ$ and freestream velocity is $U_\infty = 3\text{m/s}$.



(a) Vorticity flux



(b) Momentum flux

Figure 7-34 Transient profiles at $x/c = 0.065$: (a) vorticity flux; (b) momentum flux. Activation of plasma took place at $t = 0.52s$ and termination of plasma at $t = 1.48s$.

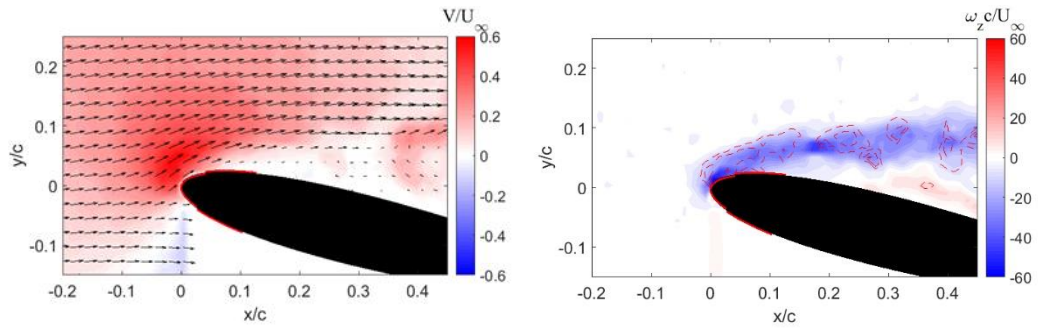
same chordwise position as that in Section 6.5. Plasma was fired at $t = 0.52\text{s}$ and switched off at $t = 1.48\text{s}$. As can be clearly seen, after plasma is activated, the vorticity flux increases by around 80% very quickly and momentum flux rises by approximately 70%, this demonstrates that vorticity and momentum are added by the activation of plasma actuator. The increases in vorticity flux and momentum flux occurs synchronously, indicating that the added vorticity enhances entrainment and transports high momentum into the boundary layer. Moreover, a number of peaks with a time interval of around 0.04s appear in both vorticity flux and momentum flux curves, this is due to the passage of vortices by plasma excitations, this also suggests that vortex could affect momentum entrainment significantly.

Since two vortices are normally created over a plasma forcing cycle, analysis was conducted by taking both of them into consideration. As suggested by Darabi and Wygnanski (2004), volume-flow-rate q_e was used to quantify the net flow crossing between the adjacent vortices over a forcing cycle. Specifically, the velocities normal to the axis connecting the adjacent vortex cores were extracted and integration was conducted over distance along the axis to obtain the volume flux. Both of the vortices are captured at the phase of 90° , therefore, volume flux was calculated from the phase-averaged PIV velocity data at phase of 90° . A volume flux of $0.0035\text{ m}^3/\text{s}$ per unit span was obtained, which clearly demonstrates a movement of the fluids away from the aerofoil surface. This is ascribed to that the outwards entrainment induced by the downstream vortex exceeds the opposed entrainment by the upstream vortex, which is directly related to the size differences between them. The transport of fluids away from aerofoil surface causes a decrease in the local static pressure, the resulting favourable pressure gradient enables flow to reattach.

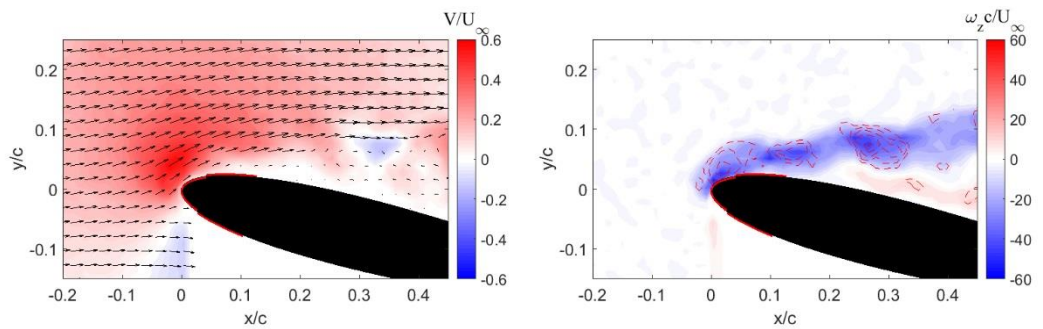
Flow responses to a single plasma pulse at post-stall angle of attack of 14° were also investigated to further confirm the role of vortex in flow separation control. Figure 7-35 shows the instantaneous flow field with a single plasma pulse applied. The pulse width is 0.004s , corresponding to a duration of $t^+ = 0.12$. Flow separates from aerofoil surface at angle of attack of 14° before plasma is fired, as shown in Figure 7-35 (a). When plasma is activated, the tail of the old separated shear layer rolls up into a vortex and the upstream flow keeps almost attached to the suction

surface (Figure 7-35 (b)). A second vortex is created after plasma is turned off in Figure 7-35 (c) due to a release of the concentrated vorticity layer that is created by the interaction between plasma jet and freestream. These vortices cause a strong entrainment, which modifies the local pressure gradient. According to Brzozowski et al. (2010), a favourable pressure gradient is observed where vortex-induced up-moving occurs. Consequently, the upstream flow is attached in Figure 7-35 (d). After the vortices travel downstream, flow starts to separate again, and reaches its baseline status at $t^+ = t_0 + 4.13$ in Figure 7-35 (g). This process is similar to what's been observed in the phase-averaged results of the flow field with periodic excitations of plasma. This process of transient flow behaviours with single pulsed excitation of plasma clearly demonstrates that vortices by excitation of plasma play a significant role in flow separation control. The vortices induce strong entrainment, and the transport of fluid away from the aerofoil surface leads to a favourable pressure gradient. The momentum entrainment together with the resulting favourable pressure gradient help achieve a flow separation control over NACA 0012 aerofoil with unsteady actuation of plasma actuator.

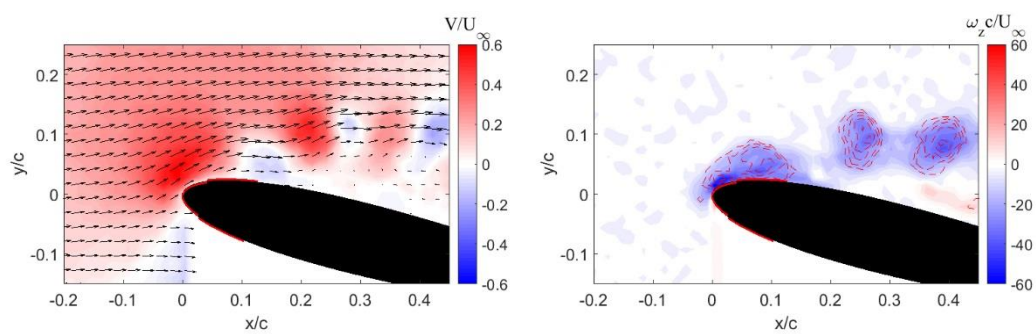
a) $t^+ = t_0 - 0.03$



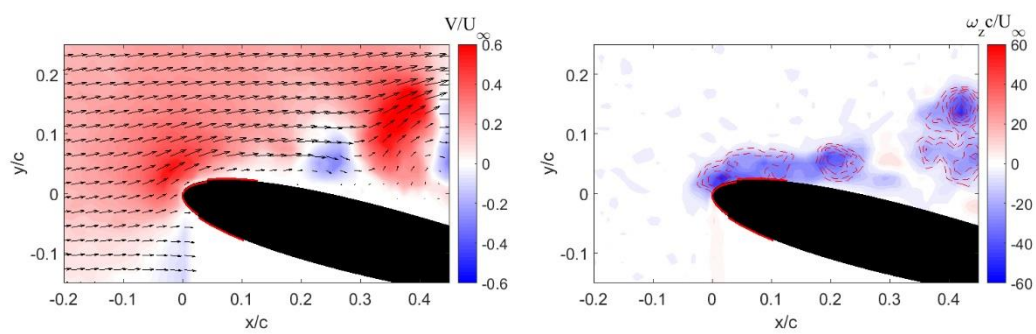
b) $t^+ = t_0 + 0.03$



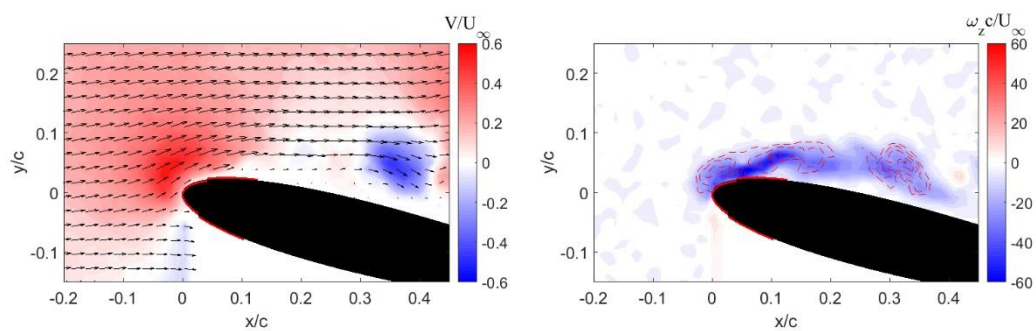
c) $t^+ = t_0 + 0.21$



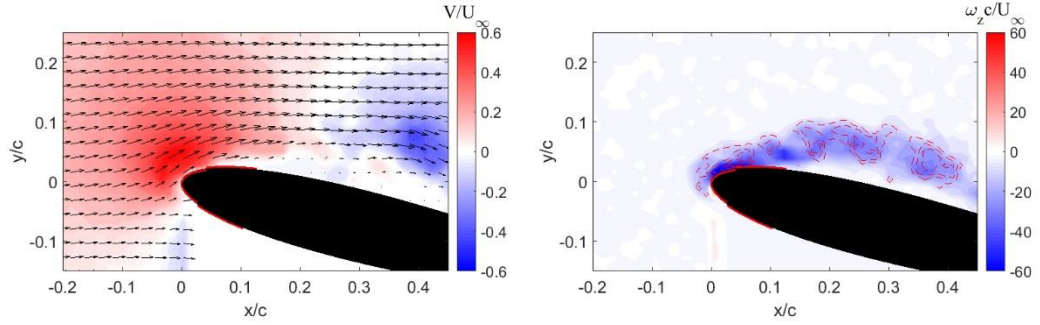
d) $t^+ = t_0 + 0.39$



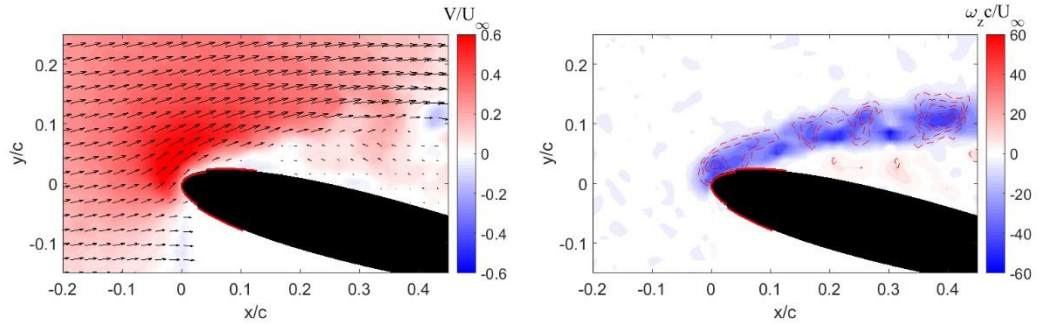
e) $t^+ = t_0 + 0.57$



f) $t^+ = t_0 + 1.05$



g) $t^+ = t_0 + 4.13$



h) $t^+ = t_0 + 4.50$

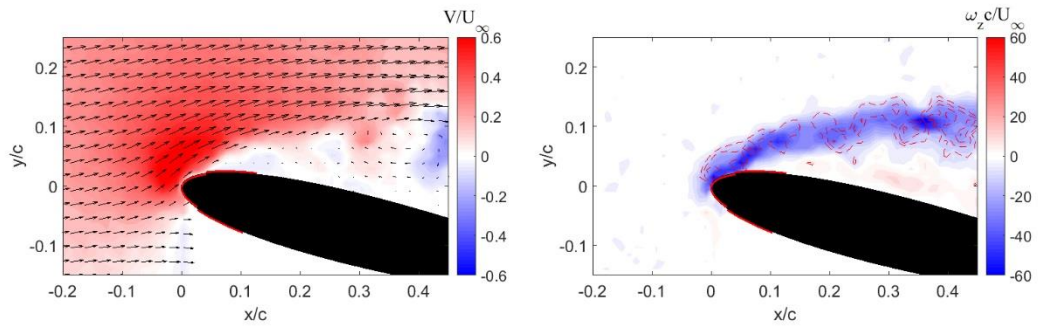


Figure 7-35 Instantaneous flow field controlled a single plasma pulse with a duty of 0.004s. t_0 is the instant when plasma is activated. Freestream velocity is

$$U_\infty = 3\text{m/s and angle of attack is } \alpha = 14^\circ.$$

In summary, large-scale clockwise vortices are created by plasma excitations, accordingly, entrainment is induced and the outwards transport of fluids from aerofoil surface causes a decrease in local static pressure gradient. The resulting favourable pressure gradient helps reattach the flow to aerofoil surface. The

momentum entrainment and the resulting negative pressure effect of the large-scale vortex are believed to be the primary mechanism for flow separation control around a NACA 0012 aerofoil with plasma excitations in the present study. Moreover, the concentrated vorticity due to the smaller-scale vortical structures by the interaction between plasma jet and freestream could help separation control as well, which is demonstrated in Section 6.5.

8 Conclusion and Future Recommendations

This chapter will give a summary of the key findings in each chapter, based on which recommendations for future work are provided. Specifically, the conclusion are summarized in Section 8.1, the novelties are presented in Section 8.2, some suggestions for future study are offered in Section 8.3 and publications on the basis of this work are listed in Section 8.4.

8.1 Conclusions

Time-resolved PIV measurements of the flow induced by a linear plasma synthetic jet actuator were conducted in still air, which shows that a wall-normal plasma jet is formed due to the collision between the two wall plasma jets by the two exposed electrodes when high identical AC voltages are applied. Two starting vortices of opposite vorticity are produced on the initiation of the plasma actuator, followed by a quasi-steady plasma jet. The half width profile suggests that the wall-normal plasma jet spreads linearly within distance from $x/c = -0.08$ to $x/c = -0.3$ and a spreading rate of $|d\delta_{1/2}/dx| \approx 0.31$ is obtained. The spreading rate is larger than that of a wall-normal plasma jet formed on a flat plate, for example, Santhanakrishnan et al. (2009) reported a spreading rate of 0.1 and N Benard et al. (2008) obtained a value of 0.13. This might be because that the two wall plasma jets are inclined when they meet and collide. Moreover, another region is also reported in this study, namely, $-0.08 \leq x/c \leq -0.035$, where the wall-normal plasma jet doesn't spread linearly with distance, this region could be termed as jet establishment zone, where the wall jets collide and wall-normal plasma jet forms. It spreads slowly in this region compared with the fully developed jet zone. Momentum coefficient was used to quantify the strength of the wall-normal plasma jet, an overshoot is observed after plasma is activated, namely, the momentum coefficient increases very quickly to 0.8% and then drops to quasi-steady value of 0.25%, implying a quasi-steady plasma jet. Apart from these, modulation of the wall-normal plasma jet strength was also studied, here, voltage amplitude was adjusted to affect the jet strength. It seems that weaker plasma jets are more sensitive to disturbances or buoyancy and inclined plasma jets are formed when voltage with an amplitude of 3.36 kV and 5.00 kV are applied, however, the plasma jet are more or less normal to the aerofoil surface with higher voltages applied.

Finally, pulsed modulation of the wall-normal plasma jet in still air was investigated as well. PIV results show that a train of vortices are produced at the pulsed excitation frequency when plasma actuator is actuated periodically. These vortices are of great significance to flow separation control in Chapter 7.

Characteristics of the baseline flow around a NACA 0012 aerofoil at Reynolds number of 20,000 and various angles of attack were studied before efforts were made on the control of flow separation around the aerofoil. PIV results demonstrate that flow separates from aerofoil surface at angle of attack of 8° , 10° , 12° , 14° and 16° while flow is fully attached to the upper surface at angle of attack of 0° . Within the range of $\alpha = 8^\circ \sim 16^\circ$, the height of reverse flow region increases with angle of attack and flow is shifted further away from the suction surface. Moreover, the magnitude of negative velocity rises as well at a higher angle of attack, all these indicate a severer flow separation. This is confirmed by separation angles of the flow around aerofoil at different angles of attack. Here, separation angle is defined as the angle between the linear fit of the recirculation region height and the chordwise position, which is 5.7° , 8.9° , 12.2° , 17.9° , and 19.8° for the separated flows at angle of attack of 8° , 10° , 12° , 14° and 16° respectively. Similar to the observations by Brendel and Mueller (1988), the inflection point height increases with chordwise positions.

Flow separation control over a NACA 0012 aerofoil with steady actuation of DBD plasma actuator was investigated in Chapter 6. On the addition of a wall-normal plasma jet with a momentum coefficient of $C_\mu = 0.25\%$, the separated flow reattaches to aerofoil surface at angles of attack of 8° , 10° and 12° because of the interaction between the wall-normal plasma jet and freestream. However, the adverse pressure gradient is so severe that this wall-normal plasma jet can't make the separated flow reattach to aerofoil surface at post-angle of attack of 14° although flow separation is alleviated with plasma. At angle of attack of 0° , where no flow separation occurs, it seems that flow is almost not affected by the wall-normal plasma jet. PIV measurements at reduced freestream velocities and increased plasma jet strength show that the wall-normal plasma jet is bent towards the freestream velocity direction. Moreover, some vortical structures are created and shed into the flow downstream due to the interaction between the wall-normal

plasma jet and freestream. The mixing effect of these vortical structures are thought to contribute to a control of flow separation around the NACA 0012 aerofoil.

In Chapter 7, investigation of flow separation control over a NACA 0012 aerofoil with plasma excitations was conducted. It has been demonstrated that flow separation at angle of attack of 14° could be entirely eliminated with plasma excitations at frequency of $f^+ = 1$ and only 10% of the energy consumed by steady actuation, which could not reattach the separated flow at this angle of attack. Therefore, periodic excitation of plasma is able to achieve a better flow separation control effect than steady actuation, even with less energy consumption. An increase in Reynolds stress magnitude has been found with periodic activation of plasma relative to that for the baseline case, this suggests a stronger mixing is caused by unsteady actuation of plasma actuator. In the meanwhile, the Reynolds stress region moves closer to aerofoil surface, accordingly, mixing occurs in the near-wall region so that high-momentum fluids are easier to be transported into the boundary layer. The stronger mixing in the near-wall region helps to achieve a flow separation control around aerofoil. Phase-averaged analysis clearly demonstrates that vortices are created with unsteady actuation of plasma actuator, specifically, a “starting vortex” is formed due to the evacuation of the old separated shear layer on the activation of plasma and a “stopping vortex” is produced because of a release of the concentrated vorticity after plasma is turned off. These vortices entrain the high-momentum fluids into the boundary layer, so they could play a significant role in flow separation control. In addition, Darabi and Wygnanski (2004) argued that a decrease in the local static pressure would be caused by the outwards entrainment of near-wall fluids, and the resulting favourable pressure gradient forces flow to reattach. It was found that separation around a NACA 0012 aerofoil at post-stall angle of attack of 14° could be controlled with periodic excitations of plasma at a wide range of frequencies, $O(0.1) \sim O(10)$, in other words, reattachment of the separated flow was achieved with excitations at frequencies of $f^+ = f_e/f_0 = 0.5, 1, 5, 10$ and 50 . Excitation of plasma at a frequency of $f^+ = 100$ could not eliminate separation above the aerofoil surface, instead, an alleviation of flow separation is observed. A fuller velocity profile is developed at a higher excitation frequency within the range of $f^+ = 0.1 \sim 5$, suggesting flow separation control effect increases with excitation frequency within this range. It seems that flow separation

control effect saturates from $f^+ = 5$ to $f^+ = 10$ since similar velocity profiles are obtained at excitation frequency of $f^+ = 5$ and $f^+ = 10$. Further increase in excitation frequency results in a flatter and weaker boundary layer above aerofoil surface. Parametric study of plasma momentum's effect on flow separation control was also performed. Periodic excitation of plasma at frequency of $f^+ = 1$ and momentum coefficient of $C_\mu = 0.125\%$, 0.0625% , 0.0125% and 0.005% could make the separated flow reattach to the upper surface. Additionally, a slightly fuller velocity profile is obtained at a larger plasma momentum within this range. When the plasma momentum is reduced to $C_\mu = 0.0005\%$, an alleviation of flow separation is obtained and a separation bubble appears above the aerofoil surface. Periodic excitation of plasma almost loses its effect on flow separation control at a plasma momentum of $C_\mu = 0.000025\%$, resulting in a flow separation comparable to the baseline flow. The entrainment and the negative pressure effect of the large-scale vortices produced via periodic excitation of plasma actuator are believed to be the primary principle for the control of flow separation around the aerofoil with periodic excitations in current study. The concentrated vorticity could help control flow separation around the aerofoil as well.

8.2 Novelty

1. Separation angles were calculated for the flow separation around a NACA 0012 aerofoil at a pretty low Reynolds number of 20,000.
2. The spreading rate of the wall-normal plasma jet is 0.31.
3. Linear plasma synthetic jet actuator placed at aerofoil leading edge could effectively control the flow separation at post-stall angles of attack.
4. Wall-normal plasma jet achieves a better flow separation control effect than upstream-directed wall plasma jet.
5. The interaction between the wall-normal jet by LPSJA (with steady actuation) and freestream produces a number of vortical structures, therefore, it acts as a vortex generator.
6. A standing vortex is formed behind the steady plasma jet at velocity ratios of 0.6, 0.9 and 1.125, moreover, separation is controlled due to an interaction between the steady plasma jet and freestream flow.
7. Plasma excitation is quite effective in flow separation control around aerofoil within a wide range of frequencies, in other words, $O(0.1) \sim O(10)$.

8. Flow separation could be controlled with plasma excitations at a duty cycle of 0.2%, corresponding to a momentum coefficient of $C_\mu = 0.0005\%$.
9. Momentum flux rises by approximately 70% with plasma excitations applied.
10. A “starting vortex” and “stopping vortex” are generated in the flow field above the aerofoil surface over a plasma forcing cycle.

8.3 Future Recommendations

DBD plasma actuator was utilized to control the leading-edge flow separation around a NACA 0012 aerofoil in this study. Basically, the interaction between wall-normal plasma jet and freestream velocity leads to flow separation control at Reynolds number of 20,000, which is a pretty small Reynolds number. The authority of this plasma actuator to control separation at higher Reynolds numbers needs to be investigated in the future. Additionally, to improve its capability of controlling flow separation at a higher Reynolds number, further researches on DBD plasma actuator are also required since the maximum jet velocity obtained up to now is around 8 ~ 10 m/s which is still very low compared to the flight speed of commercial aeroplanes.

Flow behaviours around a NACA 0012 aerofoil with steady actuation and unsteady actuation of DBD plasma actuator were examined, however, the aerodynamic performances with plasma were not measured in current study. Therefore, it is recommended that aerodynamic lift, drag and pressure distribution be measured to quantitatively demonstrate the improvement of aerodynamic performance caused by the application of plasma actuator.

Another recommendation is closed-loop control of flow separation over aerofoil with plasma actuator. Because of “hysteresis”, flow doesn’t separate immediately when plasma is off, in other words, it would keep attached for some time. Surface-mounted devices are suggested to be used to detect flow separation, when flow separates again, then activate plasma actuator, and switch it off after flow reattaches to aerofoil surface. This is a more economical way to control flow separation with the lowest power consumption.

8.4 Publications

1. Longfei Song and Kwing-So Choi. Control of the Leading-edge Flow Separation using DBD Plasma Actuators. European Drag Reduction and Flow Control Meeting – EDRFCM 2015, March 23–26, 2015, Cambridge, UK.
2. Longfei Song, Kwing-So Choi, Xin Zhang and Wanbo Wang. LEADING-EDGE FLOW SEPARATION CONTROL WITH PLASMA ACTUATORS. XXIV ICTAM, 21-26 August 2016, Montreal, Canada.

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