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Final Report for Individual Project ENG 5041P

Delta wing flow manipulation by unsteady, spatially varying leading-edge blowing

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UNIVERSITY OF GLASGOW
SCHOOL OF ENGINEERING

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Summary

One of the important developments in the capabilities of some post-WWII period fighter airplanes is “supermaneuverability”, allowing the pilot to change the heading faster and decrease the maneuver space requirements. This is achievable with flight at very high angles of attack, where delta wings are known to have superiority over conventional wing designs. In addition to this, active flow control is known to increase the post-stall capabilities of delta wings even further.

The current report investigates flow changes on a sharp edged delta wing due to unsteady leading-edge blowing with different actuation frequencies. Constant frequency and two spatially varying frequency configurations were considered. Experimental force and particle image velocimetry (PIV) measurements were done on a half model delta wing with 12 blowing slots located on the leading-edge. From force measurements it was noted that up to 40% increase in lift is possible with actuation at post-stall angles of attack. Little or no gain was seen at stall and pre-stall regimes. From PIV measurements, change in flow structure was discovered for stall regime, with actuation forming separate flow structures and not transporting the actuation momentum well into the main flow structure. For post-stall regime, where the biggest changes in force coefficients were observed, the actuation completely reshaped the flow, recreating a structure similar to the one seen in stall region angles of attack.

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Nomenclature

Latin Symbols

c	Chord length	$[m]$
C_μ	Actuation momentum coefficient	$[-]$
C_D	Drag coefficient	$[-]$
C_L	Lift coefficient	$[-]$
C_X	Axial force coefficient	$[-]$
C_Z	Normal force coefficient	$[-]$
$C_{L_{max}}$	Maximum lift coefficient	$[-]$
d	Particle displacement	$[m]$
f_a	Actuation frequency	$[Hz]$
f_c	Dominant frequency for the helical mode instability	$[Hz]$
g_i	Acceleration due to gravity component in i direction	$[m/s^2]$
$I(x, \Gamma)$	Image intensity field	$[-]$
k_{dom}^*	Universal helical mode instability frequency parameter	$[-]$
k_E	Turbulent specific kinetic energy	$[J/kg]$
l_C	Length scale	$[m]$
M	Factor of magnification	$[-]$
M	Mach number	$[-]$
p	Pressure	$[Pa]$

Q	Q-criterion	[1/s]
$R_{II}(\mathbf{s}, \mathbf{\Gamma}, \mathbf{D})$	Cross-correlation function between the two intensity fields	[−]
Re_{μ}, Re_{MAC}	Reynolds number	[−]
$R_F(\mathbf{s}, \mathbf{\Gamma}, \mathbf{D})$	Fluctuating noise component of the cross-correlation function between the two intensity fields	[−]
$R_C(\mathbf{s}, \mathbf{\Gamma}, \mathbf{D})$	Convolution of the mean intensities after cross-correlation of the two intensity fields	[−]
$R_D(\mathbf{s}, \mathbf{\Gamma}, \mathbf{D})$	Cross-correlation function component in between identical particles from the two intensity fields	[−]
$\mathbf{S}$	Rate of strain tensor	[1/s]
s	Wing half span	[m]
t	Time	[s]
$\overline{u}_i$	Mean velocity component in i direction	[m/s]
u'_i	Fluctuating velocity component in i direction	[m/s]
u/U_{∞}	Normalized velocity component in i direction	[−]
U_{∞}	Freestream velocity	[m/s]
U_C	Velocity scale	[m/s]
$V_0(\mathbf{X}_i)$	Light energy into electronic signal conversion function	[−]
v_i	Velocity component in i direction	[m/s]
x_i	Position component in i direction	[m]

Greek Symbols

α	Angle of attack	[deg]
Γ	Circulation	[m ² /s]
$\mathbf{\Gamma}$	State ensemble of the particle distribution	[−]
δ	Vortex core radius	[m]
δ	Wing thickness	[m]
λ	Bulk viscosity coefficient	[kg/m/s]
λ	Wing aspect ratio	[−]
λ_2	λ_2 -criterion	[1/s ²]
μ	Dynamic viscosity coefficient	[kg/m/s]
$\mathbf{\Omega}$	Vorticity tensor	[1/s]

ρ	Density	$[kg/m^3]$
$\tau(\mathbf{x})$	Point spread function of the imaging-lens	$[-]$
ϕ_w, φ_w	Wing sweep angle	$[deg]$
Ω_i	Normalized vorticity component in i direction	$[-]$
ω_i	Vorticity component in i direction	$[1/s]$

Subscripts

∞	Freestream value
----------	------------------

Abbreviations

<i>BLC</i>	Boundary layer control
<i>FV1</i>	Frequency variation 1
<i>FV2</i>	Frequency variation 2
<i>PIV</i>	Particle image velocimetry
<i>RMS</i>	Root-mean-square value
<i>TKE</i>	Turbulent kinetic energy

Chapter 1

Introduction

One of the important developments in the capabilities of some post-WWII period fighter airplanes is “supermaneuverability”. It allows the pilot to conduct maneuvers at post-stall angles of attack or to use thrust vectoring to yaw or pitch the aircraft independently of the flight path. For high thrust-to-weight ratio fighters the minimum-time turning maneuvers are performed faster if high angles of attack are utilized [37]. In addition, the space requirement for maneuvers is highly decreased (figure 1.1). That is why designing a fighter aircraft with post-stall capabilities is so important.

A conventional fighter aircraft can not achieve supermaneuverability, since the separated flow over the wing moves the neutral point so far back, that the conventional control surfaces lose their efficiency. This imposes trim and stability problems and control is lost. In contrast, for a delta wing aircraft the leading-edge vortex formation allows it to fly at angles of attack exceeding 30 deg. Additionally, this increases even further with canard control surfaces. Such aircraft have nonlinear lift curve slope and a very wide flight envelope. But, there exists a major limiting phenomena for reaching angles of attack even higher- vortex breakdown. This instability is majorly responsible for the loss of lift as well as tail buffeting. Consequently, control of such phenomena is of very high interest.

Flow characteristics over a delta wing, can be improved with passive or active flow control, mainly by delaying the vortex breakdown. Double delta wings, strakes, vortex generators, flaps, blowing or suction and many more methods are under research, and most are very well summarized by Rosencher and Appriou [33]. Recent developments in sensors, actuators and flow measurement techniques created new possibilities for controlling the unsteady flow phenomena too.

The purpose of this thesis is to investigate the possible high angle of attack flow improvements

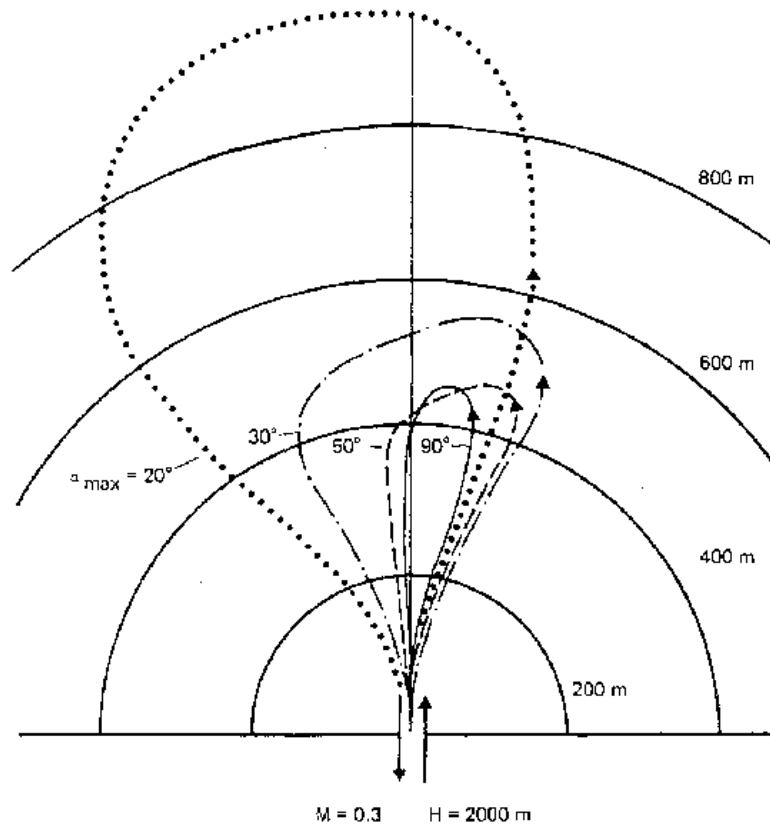


Figure 1.1: Thrust supported minimum time maneuvers at different maximum angles of attack. [36]

on delta wings while utilizing an unsteady active flow control system. A widely researched delta wing model from International Vortex Flow Experiment (VFE-2) [16] is used for testing. The model has a leading-edge blowing system installed, with spatially variable frequency, duty cycle and pressure. Unsteady blowing with spatially different frequency configurations is used to delay the vortex breakdown or reform the flow at specific angles of attack. Only experimental (wind tunnel testing) techniques are used to evaluate the effects of the control system.

Organisation and management of the project was split into following steps:

1. Topic research
2. Wind tunnel experiments (Force measurements)
3. Wind tunnel experiments (Particle Image Velocimetry measurements)
4. Analysis and comparison of the results

Chapter 2

Theory

The theoretical basis for the thesis is explained here. It includes the basic equations of fluid dynamics (Section 2.1), introduction to flow over delta wings (Section 2.2) and mechanisms of flow control (Section 2.4).

2.1 Fundamental equations of fluid dynamics

The aerodynamic forces, resulting from a flow moving around a body, are due to two main sources: pressure and shear stress distribution over the surface. The distribution of these two sources on a piece of fluid or a control volume is governed by two most important equations in fluid dynamics:

1. *Continuity* equation, also known as *Conservation of Mass* equation (equation 2.1), which states that the mass can not be created nor destroyed. Here it is written using the Cartesian tensor notation.

$$\frac{\partial \rho}{\partial t} + \frac{\partial}{\partial x_i}(\rho v_i) = 0 \quad (2.1)$$

2. *Conservation of Momentum* equation (equation 2.2), also known as the *Navier-Stokes* equation, which is developed from Newton's second law.

$$\rho \left(\frac{\partial v_i}{\partial t} + v_j \frac{\partial v_i}{\partial x_j} \right) = \rho g_i - \frac{\partial p}{\partial x_i} + \frac{\partial}{\partial x_i} \left(\lambda \frac{\partial v_k}{\partial x_k} \right) + \frac{\partial}{\partial x_j} \left(\mu \frac{\partial v_i}{\partial x_j} \right) + \frac{\partial}{\partial x_j} \left(\mu \frac{\partial v_j}{\partial x_i} \right) \quad (2.2)$$

Here ρ is the density of the continuum, v_i is the velocity component of the particle, x_i is the directional component, g_i is the gravitational component, p is the static pressure and λ and μ are the viscosity coefficients.

For the purpose of this thesis the aforementioned equations can be simplified, since the gravitation and compressibility plays a minor role (velocities tested are $M_\infty \ll 0.3$):

$$\frac{\partial v_i}{\partial x_i} = 0 \quad (2.3)$$

$$\rho \left(\frac{\partial v_i}{\partial t} + v_j \frac{\partial v_i}{\partial x_j} \right) = -\frac{\partial p}{\partial x_i} + \frac{\partial}{\partial x_j} \left[\mu \left(\frac{\partial v_i}{\partial x_j} + \frac{\partial v_j}{\partial x_i} \right) \right] \quad (2.4)$$

For solving turbulent flows, which exhibit high flow fluctuations, it is standard practice to split the characteristics into average and fluctuating components. For example, the velocity u_i would be split into the mean component $\bar{u}_i$ and the fluctuating component u'_i (equation 2.5). This is known as Reynolds Averaging.

$$u_i = \bar{u}_i + u'_i \quad (2.5)$$

The Navier-Stokes equation for unsteady flow then becomes Unsteady Reynolds-Averaged Navier-Stokes equation (equation 2.6).

$$\rho \left(\frac{\partial \bar{v}_i}{\partial t} + \bar{v}_j \frac{\partial \bar{v}_i}{\partial x_j} \right) = -\frac{\partial \bar{p}}{\partial x_i} + \frac{\partial}{\partial x_j} \left[\mu \left(\frac{\partial \bar{v}_i}{\partial x_j} + \frac{\partial \bar{v}_j}{\partial x_i} \right) - \rho \overline{u'_i u'_j} \right] \quad (2.6)$$

Here $-\rho \overline{u'_i u'_j}$ is the Reynolds stress tensor. The continuity equation remains the same.

The process of Reynolds Averaging increases the number of unknowns, which requires additional equations to “close” the system. This is the fundamental problem of turbulence, where the new unknown Reynolds stress tensor has to be somehow modeled or computed.

Vorticity of the flow is defined as:

$$\omega_i = \frac{1}{2} \left(\frac{\partial \bar{v}_k}{\partial x_j} - \frac{\partial \bar{v}_j}{\partial x_k} \right) \quad (2.7)$$

In this thesis, normalized velocity u/U_∞ and vorticity $\Omega_x = \omega_x s / 2U_\infty$ values are used.

To categorize the turbulent fields, commonly used are velocity RMS values:

$$u_{i_{RMS}}/U_\infty = \frac{\sqrt{\overline{u_i'^2}}}{U_\infty} \quad (2.8)$$

Furthermore, turbulent kinetic energy (TKE) is defined as:

$$k_E = \frac{1}{2} \left(\overline{u'^2} + \overline{v'^2} + \overline{w'^2} \right) \quad (2.9)$$

2.2 Flow over a delta wing

For straight or moderately swept wings flow separation is the reason for stall, which usually is an undesirable phenomena. For highly swept delta wings, separation already occurs at low angles of attack. But unlike the straight wings, delta wings make use of this separation. Because of the high sweep angle, separated shear layer rolls up over the wing, creating the so called “leading-edge vortex”. Because of these vortices, the lift curve gains a nonlinear shape and maximum lift coefficient is increased in comparison to conventional high aspect ratio wings.

The physics behind the leading-edge vortex structure, evolution and instabilities are explained in the following sections.

2.2.1 Formation and structure of leading-edge vortices

For a highly swept delta wing the flow separates from the leading-edge and turns up towards the suction side. Multiple discrete vortices are shed in a process similar to the classical Kelvin-Helmholtz instability [25], [9]. The vortices are almost identical and parallel to the leading-edge (figure 2.1), as the local velocity for all the points on the leading edge is nearly the same. The two neighboring vortices interact with each other through a mechanism of vortex pairing, which decreases the vortex frequency and roughly doubles the scale (figure 2.1). However, the downstream vortices interact not only with the neighboring vortices shed from the leading edge, but as well with the already developed vortex structure from upstream. Thus the structure further increases in scale.

To control the location of the separation, a sharp leading-edge can be employed. It increases the pressure gradient rapidly, forcing the separation, whereas a blunt leading-edge allows some freedom with more gentle pressure gradient variation. When the separation occurs over the whole leading-edge, then the primary vortex is said to be completely developed.

The primary vortex sheet rolls inside towards the centerline of the wing. There, a rotational core is created with low static pressure, where the viscous effects start to increase. As the circumferential velocities decrease towards the center of the vortex, high axial velocities start to appear in the core, following continuity. The velocities are roughly axis-symmetric and can be as large as four or five times the free-stream flow velocity (figure 2.2) [12]. The formation of the primary vortex can be considered to be a result of potential flow, as it can already be predicted by numerical inviscid Euler simulation [30].

Inboard of the primary vortex axis, the flow reattaches to the surface of the wing (figure 2.2). Flow streamlines point outwards and downstream towards the leading edge. There, additional velocity cross-component is induced by the high circumferential velocities of the vortex,

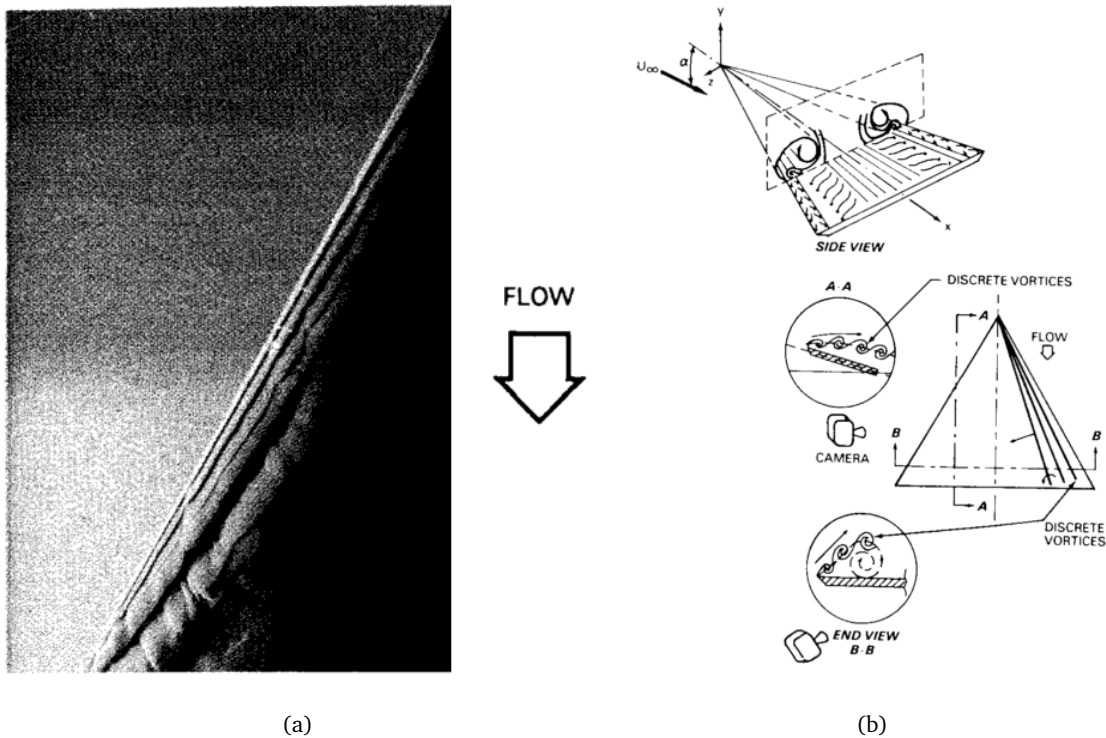


Figure 2.1: a) Top view of a delta wing at a fixed angle of attack $\alpha = 10^\circ$. b) Flow over a sharp leading-edged delta wing. [9]

thus creating additional suction on the surface. The lateral position of the suction peak is just around the same as the center of the vortex axis.

After passing the suction peak, flow experiences a big lateral pressure gradient due to vortex velocity component normal to the surface and as well the separated shear layer from the leading-edge. The flow again separates, creating a secondary vortex. The secondary vortex rotates against the primary vortex, but is far weaker in its strength. However, as the lateral flow velocities under the secondary vortex are again increased, another suction peak can be spotted, though not as strong as the first one. The development of the secondary vortex strongly depends on the type of boundary layer [8], hence in numerical simulations it only appears while using a turbulent solver.

2.2.2 Evolution of leading edge vortices

For a delta wing, leading-edge vortex already appears at low angles of attack with its apex starting at the rear end of the wing (figure 2.3, Zone 1). With increasing the angle of attack, apex of the vortex moves forward along the edge and the vortex axis moves inward as the vortex strength and size increase. At some angle of attack, the vortex axis reaches the apex point of the delta wing and can not move up any further. The vortex is considered to be fully

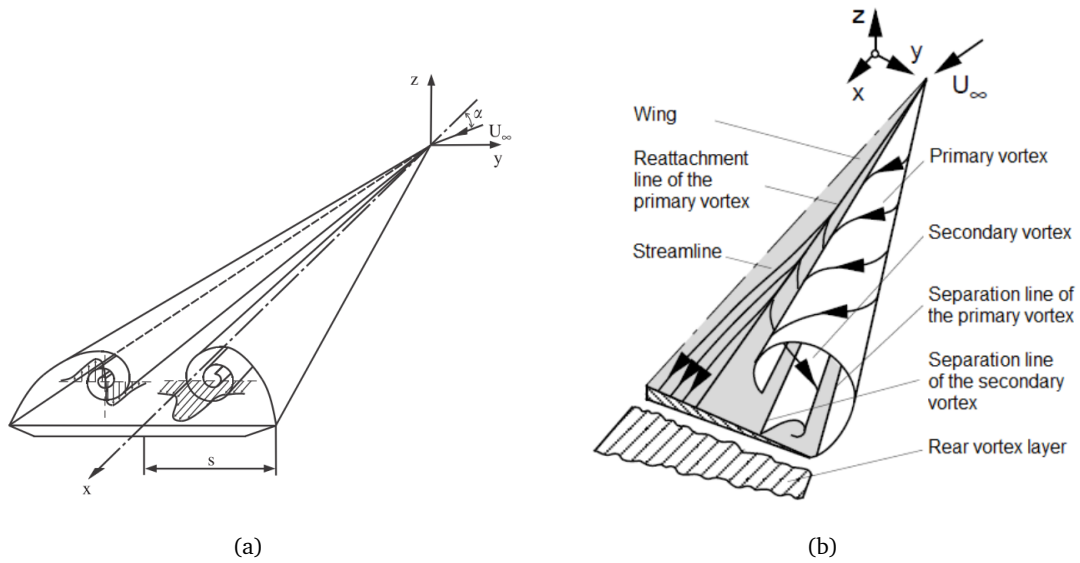


Figure 2.2: a) Schematic of shear layer and leading-edge vortices over a delta wing. [14]
b) Delta wing leading-edge vortex structure. [3]

developed (figure 2.3, Zone 2). With a laminar boundary layer, the vortex is fully developed at a lower angle of attack than with a turbulent one.

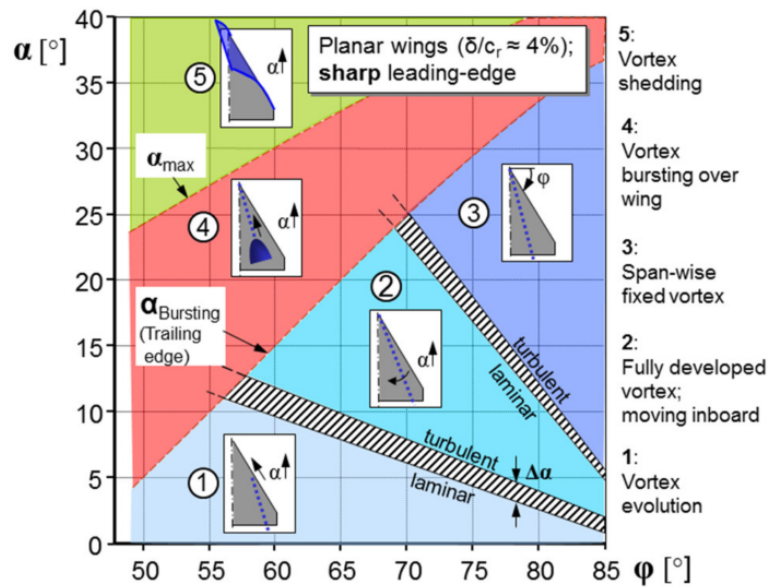


Figure 2.3: Sharp, planar delta wing vortex development. [21]

While further increasing the angle of attack, the separated shear layer pushes the growing vortex inboard. Also, the growing secondary vortex pushes the primary vortex axis upwards [30]. At some point, vortices from both wings start to block each other and no further movement

inboard is possible. Hence the vortex axis only moves upward, until the pressure gradient becomes too great for the vortex to overcome. Just before reaching the maximum lift coefficient, vortex breakdown occurs. The vortex core at the rear-end of the wing suddenly expands, decreasing the velocities and increasing the turbulence intensity throughout (figure 2.4).

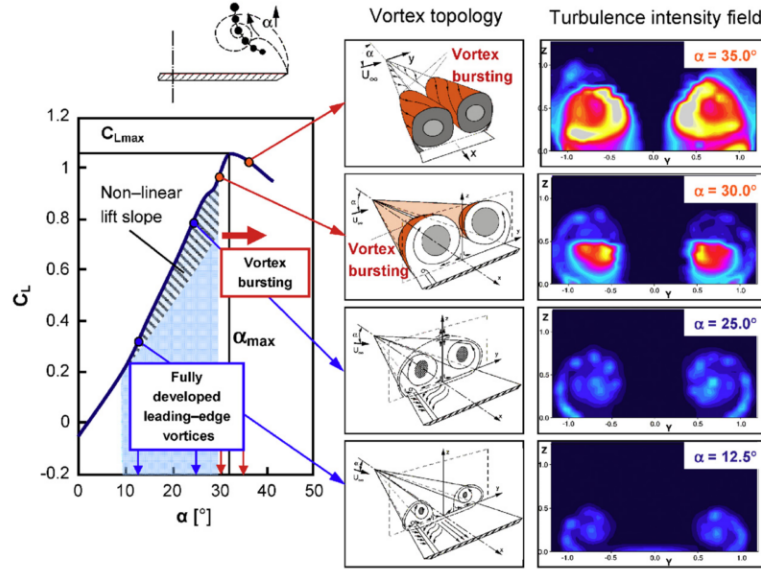


Figure 2.4: Typical delta wing turbulence intensity fields for a fully developed and burst leading edge vortices. $\phi_w = 76^\circ$, $U_\infty = 37\text{m/s}$, $Re_\mu = 1.07 \times 10^6$ [4]

With increasing the angle of attack further, the breakdown location moves towards the apex, until the vortex is completely burst (figure 2.3, Zone 4). At this point, the vortex structure is not visible anymore. The volume under the shear layer, which now covers the whole wing, has low or negative axial velocities and the lift coefficient drops to a local minimum.

2.2.3 Vortex breakdown and unstable phenomena

Theories about the vortex breakdown

As explained in the previous section, after reaching some geometry-specific angle of attack, the leading-edge vortex breaks down. Review of the recent works on vortex breakdown summarizes that the primary factor of vortex breakdown location is the adverse axial pressure gradient [26]. In vortex flows, such pressure gradient consists of two components: external pressure gradient, which depends on the angle of attack and the airfoil shape, and the contribution due to swirl, which is defined as ratio of the axial flux of momentum to the axial flux of axial momentum, all divided by the radius away from the axis. As stated by Escudier [7],

for an inviscid flow the pressure gradient can be defined as:

$$\left(\frac{dp}{dx}\right)_0 = \left(\frac{dp}{dx}\right)_\infty + \rho \int_0^\infty \frac{\Gamma}{r^3} \frac{\partial \Gamma}{\partial r} \frac{v}{u} dr \quad (2.10)$$

Here $dp/dx|_0$ is the pressure gradient on the vortex axis ($r = 0$), $dp/dx|_\infty$ is the external pressure gradient (at $r = \infty$), i.e. due to surface curvature, and $\Gamma = 2\pi r w$ is the local circulation. For a simple Rankine vortex, the equation becomes:

$$\left(\frac{dp}{dx}\right)_0 = \left(\frac{dp}{dx}\right)_\infty + \frac{2\rho}{\delta^3} \left(\frac{\Gamma_\infty}{2\pi}\right)^2 \frac{d\delta}{dx} \quad (2.11)$$

Here δ is the vortex core radius. It can be seen that for delta wings, where the core size keeps increasing downstream, a positive pressure gradient component is created, thus leading to the vortex breakdown.

Most of the theories explaining vortex breakdown support the theory of criticality, comparing the breakdown to the wave phenomena [26]. 'Critical state' is said to exist, which creates a boundary between the supercritical and subcritical flows, just like a shock wave separates supersonic and subsonic flows. Subcritical flow, where the disturbances can propagate both upstream and downstream, is said to exist downstream of the breakdown location. Upstream of the breakdown, supercritical flow is said to exist, where disturbances can only propagate downstream. This statement is as well widely supported by the experiments, where downstream conditions do appear to be far more crucial in vortex breakdown than the upstream conditions.

Vortex breakdown over delta wings

The vortex breakdown over delta wings usually occurs as one of the two types: bubble-type or spiral-type breakdown (figure 2.5) [13]. The breakdown decreases velocity fluctuations in the vortex axis and increases it in an annular region around the axis [4]. Turbulence intensities are as well highly increased.

Axisymmetric bubble breakdown can be characterized by a sudden stagnation point on the vortex axis, after which the vortex instantly expands into a bubble shape. The region inside the bubble acts as a low frequency recirculation zone. The bubble breakdown is usually followed by an expanded and weaker vortex core, which then deflects into a 'spiral tail' [26]. In some cases, multiple bubbles are observed both numerically [30] and experimentally [34]. However, in delta wing flows, spiral-type is more common. Transformation from bubble to spiral breakdown have as well been observed in experiments [13].

The spiral-type breakdown can be identified by deceleration of the vortex axis causing stagnation, followed by a sharp transformation of the vortex axis into a corkscrew-shaped twisting. A

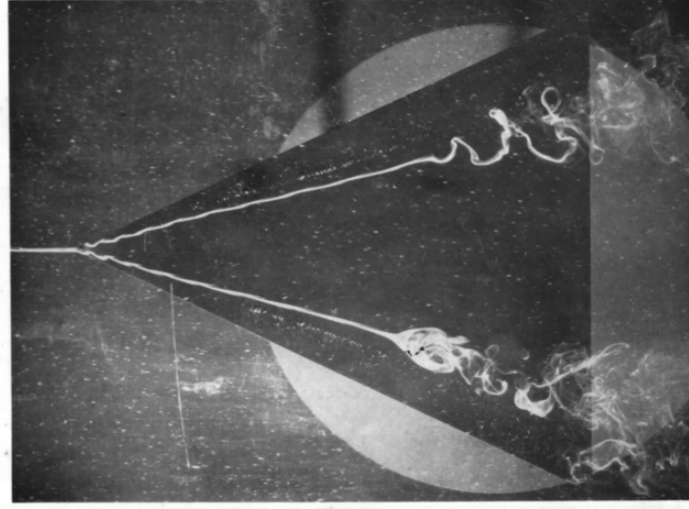


Figure 2.5: Flow visualization over a delta wing in a water tunnel, $U_\infty = 0.05 \text{ m/s}$. Top wing- spiral-type, bottom wing- bubble-type breakdown. [24]

helix is formed, which is twisted in the opposite, but rotates in the same sense as the vorticity lines upstream of the breakdown [19]. After a few turns it bursts into large-scale turbulence. The helix as well stretches downstream, increasing in pitch.

From spectral analysis of turbulence intensity [4] citeKramer2015 and from unsteady pressure measurements [2] it has been shown, that the frequency of helix rotation is concentrated within a narrow frequency band. The dimensionless frequency $k_{dom} = f x / U_\infty$ is almost constant for a given geometry and flow conditions [11]. Furthermore, Breitsamter [4] derived a universal frequency parameter k_{dom}^* based on length scale for the lateral expansion of the burst vortex core and the characteristic velocity given by the wing leading-edge normal velocity component:

$$k_{dom}^* = \frac{f_c l_C}{U_c} = \frac{f_c x \cdot \varphi_W \cdot \sin^2 \alpha}{U_\infty \cdot \sin \alpha} = \underbrace{\frac{f_c x}{U_\infty}}_{k_{dom}} \cot \varphi_W \sin \alpha = 0.28 \pm 0.025 \quad (2.12)$$

The dominant frequency at any point then can be calculated from:

$$f_c = \frac{1}{c \cot \varphi_W \sin \alpha} U_\infty (0.28 \pm 0.025) \quad (2.13)$$

2.3 Vortex identification methods

In simple terms, a vortex can be defined as ‘a multitude of material particles rotating around a common center’. The vortex core then is defined as something that is well localized in

space, has a net vorticity and circulation and the geometry of the vortex core must be Galilean invariant [18]. Multiple vortex identification methods are known to exist.

The most simple one uses the vorticity magnitude as the basis of identification. It is known to work well in shear-free flows, but fails to recognize the vortices in shear flows, for example flows close to the wall or close to the shear layers, that are always present within delta wing flows.

The criterion used quite commonly today is the Q-criterion. It was historically the first criterion to use velocity gradient decomposition ($\nabla \mathbf{v} = \mathbf{S} + \mathbf{\Omega}$, where $\mathbf{S} = 1/2[\nabla \mathbf{v} + (\nabla \mathbf{v})^T]$ is the rate-of-strain tensor and $\mathbf{\Omega} = 1/2[\nabla \mathbf{v} - (\nabla \mathbf{v})^T]$ is the vorticity tensor) to identify vortices in 3D flows. It is defined as:

$$Q = \frac{1}{2}[|\mathbf{\Omega}|^2 - |\mathbf{S}|^2] > 0 \quad (2.14)$$

The vortex core is said to exist where the rotational part of the velocity gradient tensor (vorticity magnitude) is higher than the shear strain rate part. However, the criterion is known to fail in some regions of conically symmetrical vortices (tornadoes) and other types of complex flows.

A comparably new mathematical definition of the vortex core is the λ_2 -criterion. The λ_2 -criterion corresponds to the pressure minimum in a plane after contributions of unsteady irrotational straining and viscosity are discarded [18]. It is defined as the median of the three eigenvalues of $\mathbf{\Omega}^2 + \mathbf{S}^2$.

Comparison of the three methods is presented in figure 2.6.

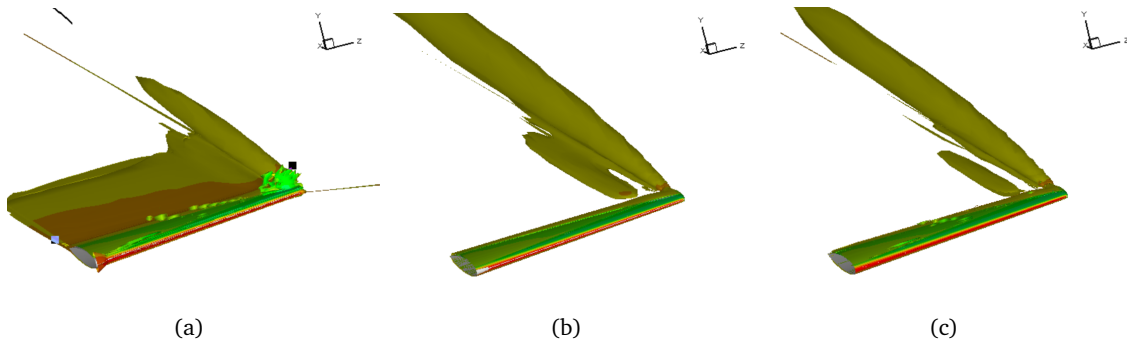


Figure 2.6: Comparison of the three vortex identification methods: a) Vorticity magnitude b) Q-criterion c) λ_2 -criterion. [17]

2.4 Flow control

In the beginning of aviation, Wilbur Wright stated that the problem of human flight is going to be solved when airplane roll control mechanisms will be developed [29]. The Wright broth-

ers tackled this problem with the first patented flow control mechanism, wing wrap, which was used on all of their “Flyers”. After that, invention of ailerons as they are known today, followed. Such were the first flow control mechanisms, developed to allow the first manned flight to be conducted.

During the last hundred years of aviation research, many great improvements in the topic followed. Slotless ailerons and turbulator tapes for gliders, different kinds of trailing- and leading-edge devices for commercial and military aircraft improved flying and handling qualities of all the aircraft. Two main types of flow control can be distinguished.

2.4.1 Passive flow control

Passive flow control methods are rigid and non-adaptive structures/features, that constantly effect the airflow. The biggest disadvantage is that passive flow control methods usually improve the aircraft’s performance at only one or a couple design points, but not over the whole flight envelope. Vortex generators are designed to increase low-speed performance of the aircraft, but during cruise it is seen only as a “useless” drag-generator. Same goes with rigid leading-edge slats, as in PZL-104 Wilga, or strakes on the McDonnell Douglas F/A-18 Hornet, both of which are required during take-off or other high angle of attack maneuvers, but are unnecessary during cruise.

However, some passive control methods are used in mass-production aircraft, which shows that the advantages are more important than the disadvantages. Examples of such would be Boeing 737-800 series aircraft, the most popular commercial aircraft in use, which have a set of turbulators in between the vertical and horizontal tails. Obviously, none of the passive flow control methods is a perfect solution to the dynamic environment of flight. Hence, active flow control methods, that can be tuned according to the flight mode, are preferred.

2.4.2 Active flow control

Active flow control methods are flexible structures, jets and other types of actuators that can be bent, moved or switched on and off according to the flight regime or pilot’s needs. This type of actuation allows to take advantage of the control mechanism precisely when needed without having the disadvantage of using it at an off-design point.

One of such methods is boundary layer control (BLC), like suction or blowing on the aerodynamic surfaces, increasing the aerodynamic quality of the airplane. For the first time, BLC was used on a production aircraft in 1958, in the shape of blown flaps on the Lockheed F-104 Starfighter. However, the auxiliary systems for such applications were heavy, complex and inefficient and the technology did not satisfy the expectations. There were a few other airplanes that made use of BLC technology, but nothing gave results too satisfying.

After the introduction of the idea that unsteady flow manipulation is more efficient than steady, and a big jump in micro-actuator technologies, the research in the area was renewed. And even though the innovations in aviation today are facing stagnation due to all the certification requirements, biggest airplane manufacturers are still investing money into development of such technologies. One of the examples would be Boeing's 757 ecoDemonstrator aircraft, which uses jet actuators on the vertical tail to decrease the required rudder size, hence decreasing the parasitic drag.

As of today, main objectives of active flow control are separation control, keeping the flow laminar and reduction of cavity fluctuations. All of the topics can highly reduce the drag of the aircraft, but the concept of flow manipulation, either passive or active, is applicable for any drag-generating object [10].

2.4.3 Active flow control over delta wings

Many interesting types of flow control specifically for delta wings exist, mainly due to the number of phenomena that can be controlled. Such control mechanisms include multiple vortex systems (i.e. canard configurations), various blowing and suction setups [35] [6] [21], leading-edge or apex flaps [20] and plasma actuators [28]. The actuation can be steady or unsteady, with the latter usually having actuation frequencies adjusted according to the phenomena to be controlled (figure 2.7).

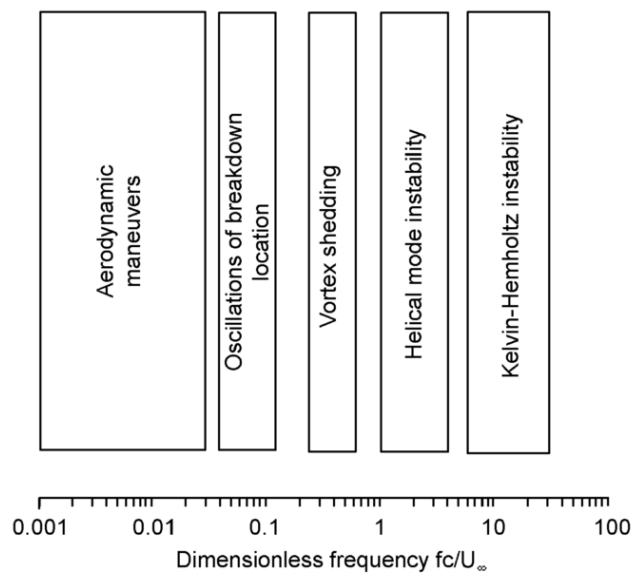


Figure 2.7: Spectrum of unsteady flow phenomena over slender delta wings as a function of dimensionless frequency. [14]

For the purpose of this thesis, systems with blowing are of highest interest. The most pop-

ular ones are stationary jets, either along the leading- or trailing-edges or along the vortex core. However, movable jets along the core have as well been successful [27]. Systems with tangential blowing along the leading-edge that make use of Coanda effect have as well been studied [14]. All of these systems are very dependent on external parameters like sweep angle or Reynolds number, hence have to be tuned not only for every application, but for flight regime as well. However, most of the systems successfully adjust the vortex breakdown position or improve the force coefficients, especially in post-stall regime. Unsteady actuation is usually found to be far more effective, hence the unsteady systems are seen as the ones with the most potential. For a more thorough review of flow control mechanisms for delta wings, see Gursul, 2007 [14].

2.5 Particle Image Velocimetry

Particle Image Velocimetry (PIV) is a technique used to record the velocity information of the flow field with help of tracer particles (usually oil droplets in aerodynamics) distributed in the flow. The particles are illuminated twice with a very short delay and their instantaneous velocity vectors can be determined from the pathlines. As well defined by Adrian [1], PIV is *accurate, quantitative measurement of fluid velocity vectors at a very large number of points simultaneously*. The standard two-dimensional measurement system can be seen in Figure 2.8. A more complex system with multiple cameras and lasers can be used to capture not only a two-dimensional, but also a three-dimensional flow velocity data. A traverse system can be employed to change the location of the light sheet and create an observation volume.

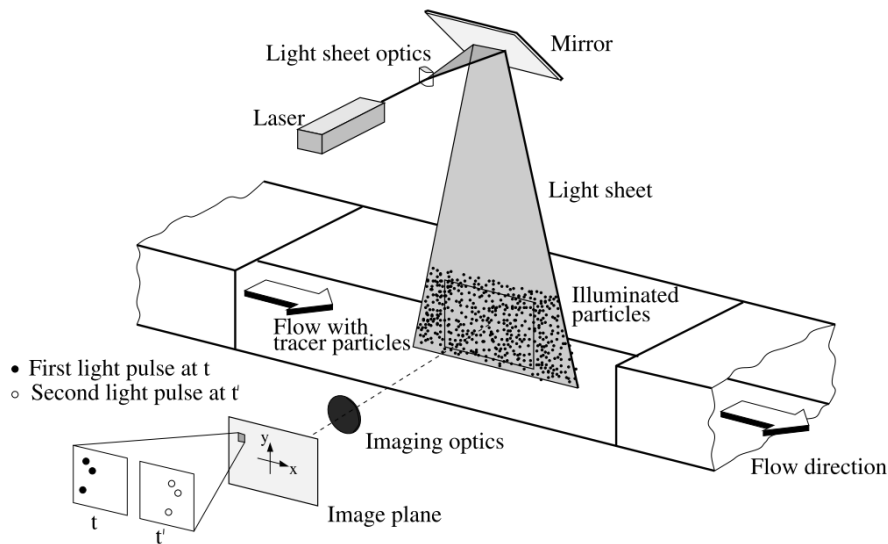


Figure 2.8: Experimental arrangement for 2D particle image velocimetry in a wind tunnel. [31]

2.5.1 Equations and algorithms for statistical PIV evaluation

When the laser light illuminates the particles, each particle reflects or diffracts the light into the camera. If each particle has a position vector $\mathbf{X}_i = (X_i, Y_i, Z_i)^T$ in a three dimensional space, then the coordinates of all the particles in the image are captured with state of ensemble Γ at a given time t [31]:

$$\Gamma = \begin{pmatrix} \mathbf{X}_1 \\ \mathbf{X}_2 \\ \vdots \\ \mathbf{X}_N \end{pmatrix} \quad (2.15)$$

Under assumption, that the particle position in measurement plane and its position in image plane are related by constant factor of magnification M , each particle has a position vector in the image plane, described as $\mathbf{x}_i = (x_i, y_i)^T = (X_i M, Y_i M)^T$. In reality, a more complex model for perspective effects would have to be taken into account. With the assumption that the particles do not overlap, the image intensity field $I(x, \Gamma)$ can be written as:

$$I(x, \Gamma) = \sum_{i=1}^N V_0(\mathbf{X}_i) \tau(\mathbf{x} - \mathbf{x}_i) \quad (2.16)$$

Here $\tau(\mathbf{x} - \mathbf{x}_i)$ is the convolution product of the Dirac delta-function, shifted to position $\mathbf{x}_i$, and the point spread function $\tau(\mathbf{x})$ of the imaging-lens, which usually is well approximated by the Gaussian distribution. In simple terms, the term $\tau(\mathbf{x} - \mathbf{x}_i)$ describes how the light energy coming from one particle is registered by the camera. The term $V_0(\mathbf{X}_i)$ converts this light energy into an electronic signal.

Most commonly, cross-correlation of intensity fields of two singly exposed recordings is used [31] (figure 2.9). With this method, two recordings, that are offset by some small delay, are cross-correlated using Fast Fourier Transform algorithms (figure 2.10). This method is as well going to be used in the current experiment. The mathematical background for cross-correlating the two frames is presented below.

Assuming that the ensemble of the second frame was recorded at time $t' = t + \Delta t$ and that all particles inside the plane were moved by constant displacement $\mathbf{D}$, the position of the particle i is now described as:

$$\mathbf{X}'_i = \mathbf{X}_i + \mathbf{D} = \begin{pmatrix} X_i + D_x \\ Y_i + D_y \\ Z_i + D_z \end{pmatrix} \quad (2.17)$$

And the displacements are defined as:

$$\mathbf{d} = \begin{pmatrix} M D_x \\ M D_y \end{pmatrix} \quad (2.18)$$

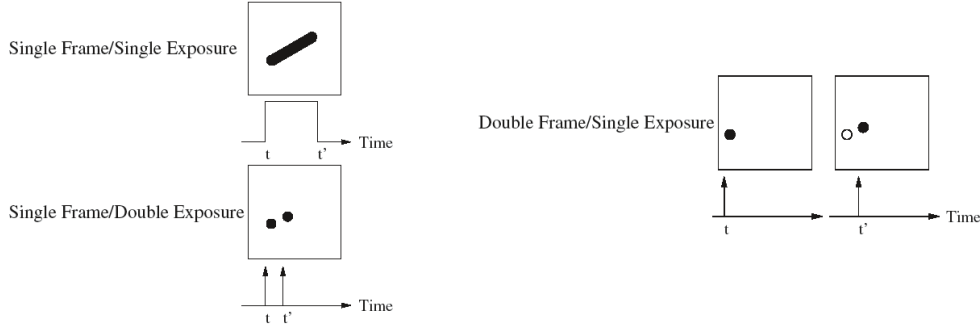


Figure 2.9: Single and double frame recording techniques. [31]

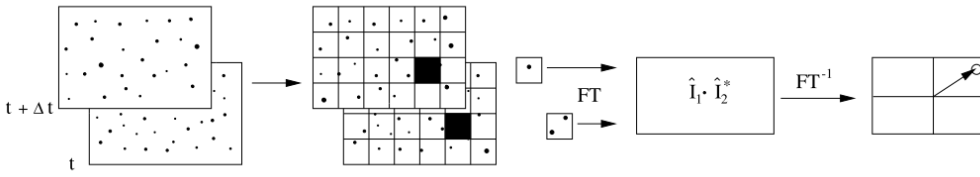


Figure 2.10: Double frame/single exposure recording technique and the digital cross-correlation method. [31]

Hence the image intensity field for the second exposure is defined as:

$$I'(x, \Gamma) = \sum_{j=1}^N V'_0(\mathbf{x}_j + \mathbf{D}) \tau(\mathbf{x} - \mathbf{x}_j - \mathbf{d}) \quad (2.19)$$

Here $V'_0(X)$ is the interrogation volume during the second exposure. Now, when the light intensity maps from the two frames are registered within the computer, cross-correlation is done:

$$R_{II}(\mathbf{s}, \Gamma, \mathbf{D}) = \langle I(x, \Gamma) I'(x, \Gamma) \rangle \quad (2.20)$$

Here $R_{II}(\mathbf{s}, \Gamma, \mathbf{D})$ is the cross-correlation function between the two intensity fields. It can be split into 3 components:

$$R_{II}(\mathbf{s}, \Gamma, \mathbf{D}) = R_C(\mathbf{s}, \Gamma, \mathbf{D}) + R_F(\mathbf{s}, \Gamma, \mathbf{D}) + R_D(\mathbf{s}, \Gamma, \mathbf{D}), \quad (2.21)$$

where $R_F(\mathbf{s}, \Gamma, \mathbf{D})$ is the fluctuating noise component and $R_C(\mathbf{s}, \Gamma, \mathbf{D})$ is the convolution of the mean intensities [31]. $R_D(\mathbf{s}, \Gamma, \mathbf{D})$ represents the component that corresponds to the correlation in between the identical particles of the first and second frame. Therefore it is the peak of R_D that leads towards the vector of the average in-plane particle displacement in between the two frames. Therefore, a velocity vector for the displaced particles can be calculated.

Chapter 3

Experimental Analysis

The experimental part of the report deals with force and Particle Image Velocimetry (PIV) measurements on the model placed in a wind tunnel. Subsequent chapters describe the methodology and setup of the experiment.

3.1 Experimental setup

3.1.1 Wind tunnel setup

Experimental measurements of the project are conducted in the wind tunnel facilities of the Institute of Aerodynamics and Fluid Mechanics, Technical University of Munich (Lehrstuhl für Aerodynamik und Strömungsmechanik, Technische Universität München). Wind Tunnel A is used, the specifications for which can be found in table 3.1.

Specification	Value
Cross section of the test section	$1.8m \times 2.4m$
Contraction ratio	7 : 1
Test section length	4.8m
Maximum power	420kW
Maximum velocity	65m/s
Turbulence intensity	$Tu_x = Tu_y = Tu_z < 0.4\%$

Table 3.1: Specifications of the Wind Tunnel A in the Institute of Aerodynamics and Fluid Mechanics of Technical University of Munich

3.1.2 Model setup

A half delta-wing model is used for the experiments. The geometry of the model is identical to the one used in International Vortex Flow Experiment 2 (VFE-2) [16], during which a big database on the flow phenomena around such a configuration, both steady and unsteady, was gained.

The geometry of the model is presented in table 3.2.

Parameter	Value
Root chord, c	$0.977m$
Half span, s	$0.457m$
Thickness, δ	$0.033m$
Sweep angle, ϕ	65 deg
Aspect ratio, λ	1.865

Table 3.2: Geometry of the half-delta wing model.

The upper and lower surfaces of the model are made from aluminum plates, one of which is covered with orange film to decrease the laser light reflections. Leading- and trailing-edges are made from glass fiber reinforced plastic. The model has a 55mm peniche, which elevates the model out of the wind tunnel floor boundary layer.

The leading-edge is sharp to fix the separation line geometrically. It has 12 pairs of slots, each being $2 \times 10mm$, split into 3 sections. The arrangement of the slots was selected so that the blowing would generate pairs of vortices that would interact with the flow structures shed from the leading-edge [22]. The air is supplied to the pressure chambers via internal tubes, that are connected to the facility's compressed air supply. The blowing jet is approximately perpendicular to the wing surface. The jet speed depends on the pressure supplied. Arrangement of the slots can be seen in figure 3.1.

12 independent actuators open and close the electromagnetic air release valve. The actuation frequencies of up to $1000Hz$ are possible. A frequency for each actuator can be selected independently. However, due to the data transfer limitations from the Labview actuation control software, not all combinations of frequencies are possible. It was found that a low least common multiple between the frequencies is needed. Phase angles as well as duty cycles can also be selected for each actuator.

To summarize, actuation parameters that can be selected for each slot pair are blowing frequency, phase angle and duty cycle. Jet speed (air pressure) can be adjusted for a whole section (4 slot pairs) only. The methodology behind selecting the parameters is described in Section 3.3.

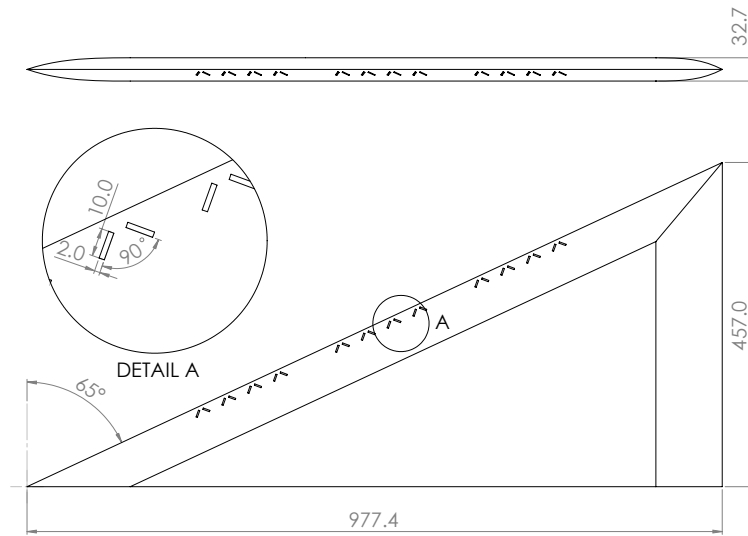


Figure 3.1: Geometry of the model and arrangement of the slots.

3.1.3 Measurement equipment setup

Force polars are measured with a high-accuracy 6 axis external balance, stored under the wind tunnel floor. Baseline and actuated cases are measured. During the measurements it was assured that the wing itself does not touch the peniche and only the forces acting on the wing surface are recorded.

A stereoscopic particle image velocimetry (2D3C) setup is used, consisting of 2 cameras, a laser and an external trigger (figure 3.2). The laser is a $325mJ$ Nd:YAG double cavity laser with rotatable light sheet optics, wavelength of $532nm$ and maximum double pulse frequency of $15Hz$. It is positioned on a traverse system above the test section. The external trigger is connected to the actuation system. The seeder is located in the diffuser and uses air pressure to vaporize oil into particles.

Two sCMOS cameras are used, each with resolution of 2560×2160 pixels. They are equipped with optical lenses ($50mm$ or $135mm$) and Sheimpflug adapters. The cameras are positioned in two different setups. The first one is employed to measure the vortex planes. For this purpose, cameras are positioned on the side of the open wind tunnel section, facing the upper surface of the wing and mounted on a traverse system. For measuring the cross-planes, cameras are mounted on aluminum fixtures that are connected to the same traverse as the laser.

LaVision's Davis software is used for making the measurements and post-processing.

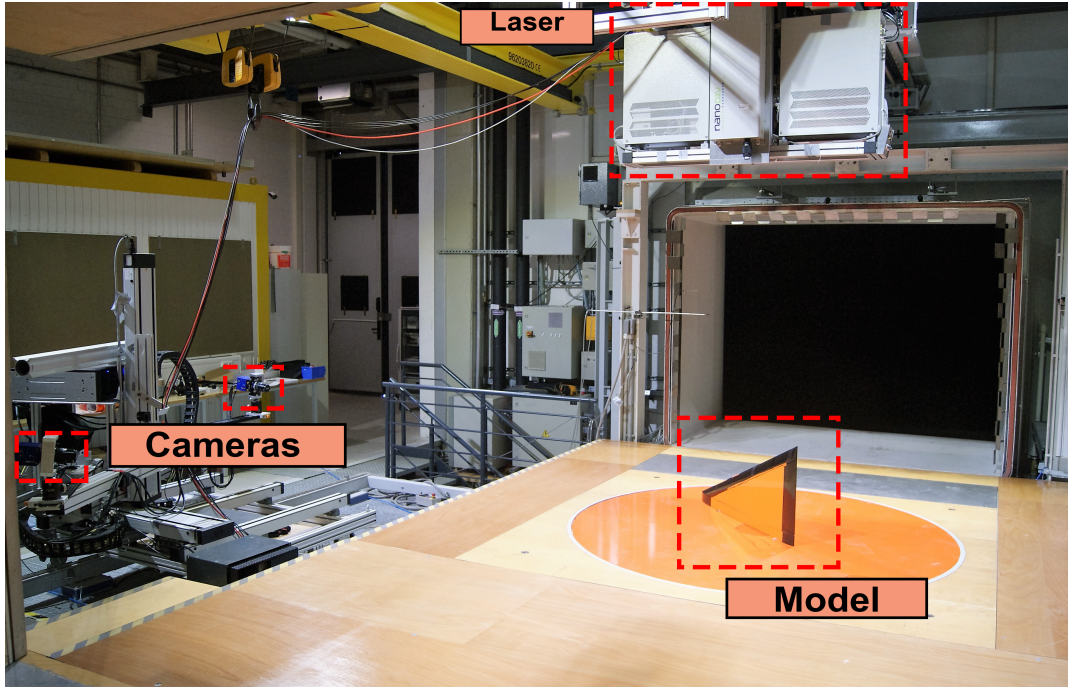


Figure 3.2: Experimental setup for vortex plane PIV measurements

3.2 Test cases

In relation to previous experiments done on the same model, three most important test cases are chosen: $\alpha = 23$ deg, 35 deg and 45 deg. The relevant free-stream velocities and Reynolds numbers can be found in table 3.3. Force measurements are done with all selected actuation parameters over the relevant angle of attack range.

PIV measurements of vortex planes focused around the breakdown location as well as cross-section planes from $x/c = 0.05$ to $x/c = 0.95$ with increments of $\Delta x/c = 0.05$ are conducted.

α, deg	$U_\infty, \text{m/s}$	Re_{MAC}	Constant actuation frequency , Hz	Measurement planes
23	24	1×10^6	65	$x/c = 0.05 - 0.95$ with increments of $\Delta x/c = 0.05$, planes on the vortex axis
35	12	5×10^5	32	
45	12	5×10^5	12	

Table 3.3: Test cases for the PIV measurements.

3.3 Actuation parameter selection

A set of actuation parameters had to be defined for testing:

- 12 frequencies (one for each blowing slot)
- 12 duty cycles
- 12 phase angles
- 3 jet velocities (directly proportional to the supplied pressure)

3.3.1 Actuation frequency selection

Actuation frequencies are selected based on the local helical mode instability frequency, as well as on previously tested constant actuation frequency configurations.

Constant actuation frequency are presented in table 3.3. Two more spatially varying frequency selection strategies are employed:

Experimental frequency selection method (FV1)

First frequency selection method is based on previous time-accurate hot-wire and unsteady surface pressure measurements on the same model. The hot-wire measurements were done with two probes inside the flow. One of the probes was stationary, located at $x/c = 0.4$ plane and the other one was mounted on the traverse system. More details can be found in the original work by Kramer [23].

The points that are used in determining the dominant frequencies are both from inner core and rotational core region. An example of the points investigated can be seen in figure 3.3.

Description of unsteady pressure measurements can be found in work by Blanchard [2]. Table 3.4 presents the recorded dominating frequencies for all the planes from both sources.

It was decided to shift the selected actuation frequencies upstream. This decision was based on a thought, that there exists some time delay between the pulse actuation and actual dissipation of the jet into the vortex structure, which occurs downstream of the slot. As no actual data on the dissipation times is available, the shift was based on personal judgment. Table 3.5 presents the selected actuation frequencies. Frequency configurations selected with this method are called Frequency Variation 1 (FV1).

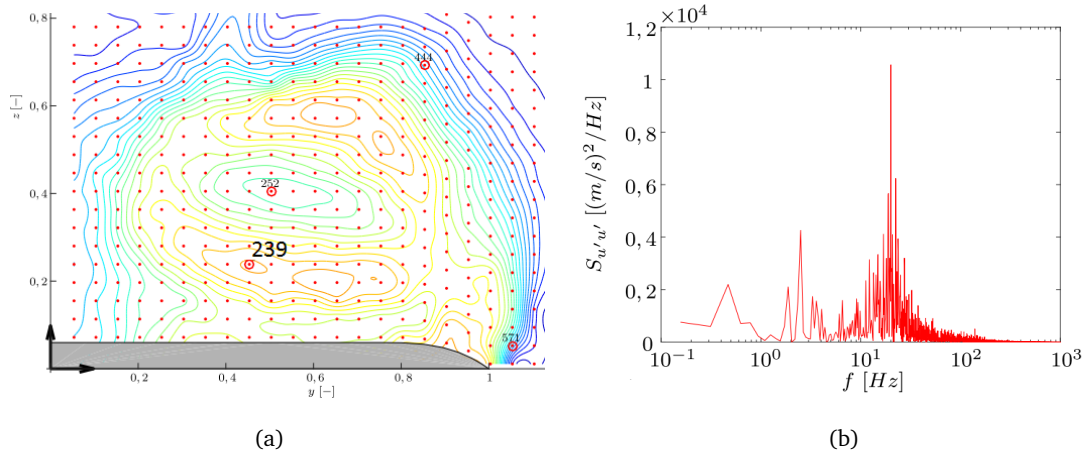


Figure 3.3: a) Measurement grid. b) Spectral distribution for point 239. $x/c = 0.6$, $U_\infty = 12m/s$, $\alpha = 35$ deg

Measurement plane, x/c :	0.4	0.6	0.8
23deg:			
Surface pressure measurements, Hz:	-	-	85
Hot-wire measurements, Hz:	-	80	46
35deg:			
Surface pressure measurements, Hz:	31	23	18
Hot-wire measurements, Hz:	29	20	-
45deg:			
Hot-wire measurements, Hz:	-	24	-

Table 3.4: Dominant frequencies in the plane from surface pressure [2] and hot-wire measurements [23].

Theoretical frequency selection method (FV2)

The second frequency selection method is based on semi-empirical equations of local dominant vortex frequency provided by Breitsamter (equation 3.1) [4].

$$f_c = \frac{1}{x \cot \varphi_w} \frac{U_\infty}{\sin \alpha} (0.28 \pm 0.025) = \frac{U_\infty}{x \sin \alpha} (0.6 \pm 0.054) \quad (3.1)$$

Corresponding dominant frequency for each slot location is calculated. However, as mentioned in section 3.1.2, not all desired frequencies could be used. Hence the actuation frequencies are somewhat different from the calculated ones. The calculated and actuation frequencies can be found in the table 3.6. Frequency configurations selected with this method are called Frequency Variation 2 (FV2).

Slot Nr.	Slot location, m	Pre-stall, 23deg		Stall, 35deg		Post-stall, 45deg	
		f_c, Hz	f_a, Hz	f_c, Hz	f_a, Hz	f_c, Hz	f_a, Hz
1	0.24	152	80	52	33	42	16
2	0.28	132	80	45	33	36	16
3	0.32	117	80	40	33	32	16
4	0.35	105	80	36	33	29	10
5	0.44	84	58	29	22	23	10
6	0.47	78	58	27	22	22	10
7	0.51	73	58	25	22	20	10
8	0.54	68	58	23	22	19	10
9	0.64	58	58	20	22	16	8
10	0.67	55	58	19	22	15	8
11	0.71	52	58	18	22	14	8
12	0.74	50	58	17	22	14	8

Table 3.5: Experimental actuation frequency selection (FV1). Here f_c is the calculated frequency (equation 3.1), f_a is the actuated frequency.

Figure 3.4 presents the distribution of actuation frequency parameter $f x \sin \alpha / U_\infty$ for all angles of attack for both frequency selection methods.

3.3.2 Duty cycle and jet velocity

Following the previous experiments on the model, the most effective duty cycle at 45 deg angle of attack has already been determined to be in the region of 20 – 60%, only changing the lift coefficient by 0.05 in between (baseline value is 0.6) for actuation frequencies of 12Hz and 18Hz [38]. For frequencies 6Hz – 36Hz, the optimal DC range is between 20 – 70%. The effect in changing the DC is again quite small- lift coefficient with each variation varies by 0.1 at most. As most of the previous measurements were made with $DC = 25\%$, the same value is kept. Phase angle was not varied in this experiment.

Previous experiments have as well defined the most efficient blowing jet momentum coefficient C_μ (equation 3.2) to be around $C_\mu = 3 \times 10^{-4}$ for pulsed leading-edge blowing [28]. In the current experiment, two supply pressures are used- $p_{jet} = 4bar$ and $p_{jet} = 5bar$, corresponding to jet velocities of $U_{jet} = 55m/s$ and $U_{jet} = 60m/s$. For freestream velocities $U_\infty = 12m/s$ and $U_\infty = 24m/s$ the momentum coefficients vary from $C_\mu = 2.4 \times 10^{-4}$ to $C_\mu = 11.2 \times 10^{-4}$.

$$C_\mu = \frac{\dot{m}_{jet} U_{jet}}{\frac{\rho_\infty}{2} U_\infty^2 S} = 2 \left(\frac{U_{jet}}{U_\infty} \right)^2 \frac{th}{\frac{1}{4}cb} \quad (3.2)$$

Slot Nr.	Slot location, m	Pre-stall, 23deg		Stall, 35deg		Post-stall, 45deg	
		f_c, Hz	f_a, Hz	f_c, Hz	f_a, Hz	f_c, Hz	f_a, Hz
1	0.24	152	184.5	52	61.5	42	51.25
2	0.28	132	123	45	41	36	41
3	0.32	117	123	40	41	32	30.75
4	0.35	105	123	36	41	29	30.75
5	0.44	84	123	29	41	23	20.5
6	0.47	78	61.5	27	20.5	22	20.5
7	0.51	73	61.5	25	20.5	20	20.5
8	0.54	68	61.5	23	20.5	19	20.5
9	0.64	58	61.5	20	20.5	16	20.5
10	0.67	55	61.5	19	20.5	15	20.5
11	0.71	52	61.5	18	20.5	14	20.5
12	0.74	50	61.5	17	20.5	14	10.25

Table 3.6: Theoretical actuation frequency selection (FV2). Here f_c is the calculated frequency (equation 3.1), f_a is the actuated frequency.

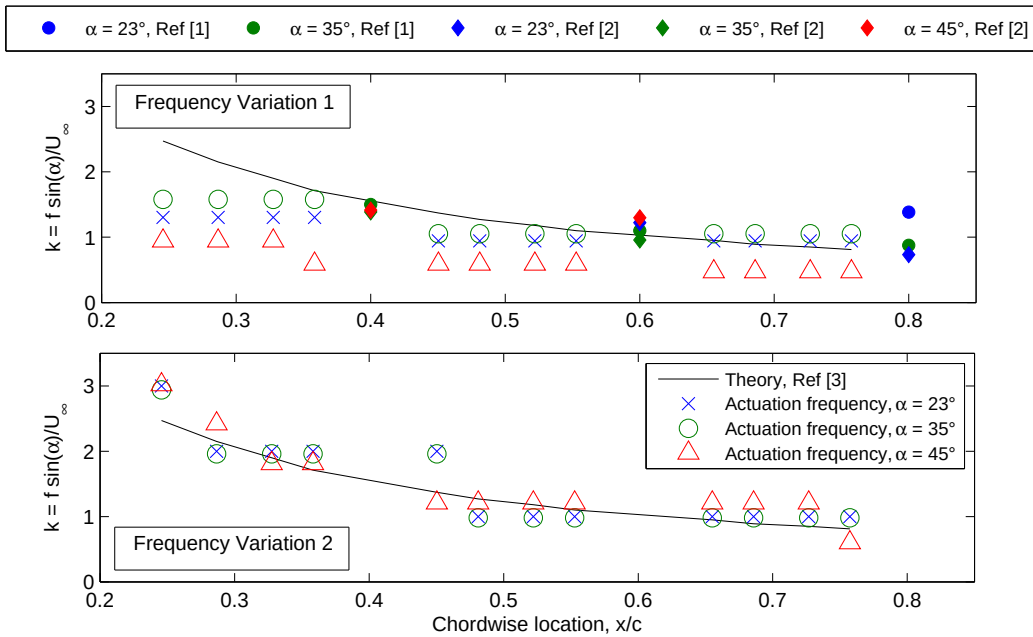


Figure 3.4: Actuation frequency parameter $f x \sin(\alpha)/U_\infty$ for Frequency Variation 1 and 2. Here Ref[1] are data points from surface pressure measurements [2], Ref[2] are data points from hot-wire measurements [23]. Ref[3], the theoretical values, are calculated according to formula 3.1 [4].

3.4 Methodology for Particle Image Velocimetry measurements

In the following sections, the methodology of the experiments is described.

3.4.1 Preparation

Equipment setup

The PIV equipment setup has already been described in section 3.1.3. The test section base is covered with orange film to decrease the reflections coming from the laser light. A black permanent marker is used for edges of the model and exposed boltheads. The control room is completely isolated from the laser light during all times.

One of two optical lenses is chosen for each calibration. 50mm lenses offer less detailed flow pictures (particle size $\approx 1px$). However, as the entire structure of the vortex is investigated, the level of details is decided to be enough. Some of the cross-section measurements are done with these lenses.

130mm lenses (particle diameter $\approx 3px$) are used for vortex planes, as the 50mm lenses presented some difficulties with calibration. They are as well used for most of the cross-section measurements. Where the field of view is not enough, multiple pictures focused at different points are stitched into one plane to capture all the required details. This later appeared to be a problem (section 4.4).

An example in differences for the two lenses' field of view can be seen in figure 3.5, taken from $\alpha = 45$ deg measurement. The green border represents the 50mm lenses, where field of view is 475mm in height and 560mm in width (measurement plane area is $0.266m^2$). The red border represents the 130mm lenses, where field of view is 248mm in height and 295mm in width (measurement plane area is $0.073m^2$). The view with the smaller lenses here is tilted upwards (+50mm shift upwards in the measurement plane).

Calibration

Calibration is done for every new set of measurement planes. Mostly, one calibration is done for one vortex measurement plane. Vortex axis is determined from previous PIV measurements on the same model. The axis location is compared with data provided by Furman and Breitsamter [8], where vortex axis was determined by the use of smoke. The agreement between both methods for all measured angles of attack is very good. The calibration plate is then placed on the vortex axis.

For cross-planes the same calibration is kept for multiple measurements, as long as the cameras and the laser setup could follow the required cross-plane location. The movement is

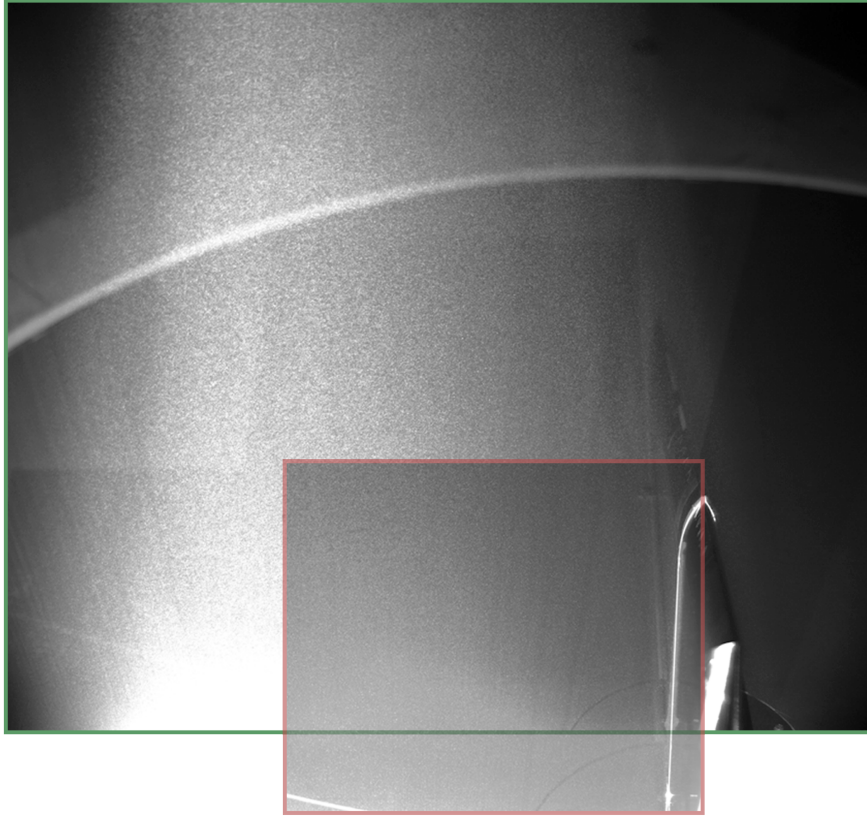


Figure 3.5: Field of view comparison for both lenses. Green border- 50mm lenses (475mm × 560mm), red border- 130mm lenses (248mm × 295mm).

mostly restricted by lack of room for the supporting structure of the setup. Hence, system has to be recalibrated towards the trailing edge. Another recalibration took place after finishing the measurements of a set, to include the effects of change of focus, which had to be readjusted manually after every calibration with the wind tunnel already running.

3.4.2 Conduction

Wind tunnel speed is adjusted according to the Betz manometer. Digital barometer and thermometer are used for finding out the required dynamic pressure. The wind tunnel fan is kept on for some seconds for the flow to become homogenous. Measurement log with atmospheric parameters is kept during the tests.

Oil particles are used as a tracer for the measurements. The oil is Bis(2-ethylhexyl) sebacate ($C_{76}H_{50}O_4$), with density of 0.91kg/l. Particle diameter varies around 0.5 – 5.0 μ m. The model is cleaned from time to time from the oil particles that get stuck on the surfaces. 400 Samples are taken at 12Hz (13Hz for $\alpha = 45$ deg case).

Double frame single exposure recording technique (section 2.5.1) is used.

3.4.3 Post-processing

Parameter selection

Processing parameters in LaVision's Davis software are adjusted for different lenses and cross-sections. It was discovered that the best way to control the resolution of the measurement plane is to use bigger interrogation windows with bigger overlap. Worse correlation factors are given by the small interrogation windows with no overlap. 'Multiple pass' option is used, with an initialized flow field reference equal to freestream conditions for the first pass. The high intensity locations, that could not be eliminated with orange film or marker physically, are masked with an algorithmic mask, setting the pixel intensity as a threshold.

Sliding background option, even though suggested by the processor developer, is disabled, since selecting the option creates many uncorrelated spots. This is especially significant in areas around the edges of the laser sheet.

Self-calibration wizard is used for some measurements. However, the difference between processed measurements with and without self-calibration appeared to be minimal.

The processing parameter configurations that are used are presented in table 3.7.

	Parameter name	Parameter value
1	Particle intensity normalization	13px
2	Define intensity threshold mask	Intensity above 64000
3	Initial pass shift	u and v values according to the test case
4	Vortex plane interrogation window size:	
	First pass	$64 \times 64px$, No overlap
	Second pass	$32 \times 32px$, No overlap
5	Cross-plane interrogation window size:	
	First pass	$128 \times 128px$, 50% Overlap
	Third and fourth pass	$64 \times 64px$, 50% Overlap

Table 3.7: PIV processing parameters.

Export and final post-processing in Tecplot

Average time to process a measurement plane is around 100 minutes. Then the data is exported to a Tecplot-compatible format (.dat). Since the coordinate systems of measurement

planes are always different from the default one of the wing, a Matlab script is used to transform the coordinates of all measurements into the desired system. All the final adjustments are done within the Tecplot software.

Table 3.8 summarizes the most important parameters of the PIV measurements.

Type of laser:	325mJ Nd:YAG double cavity laser
	Wavelength of 532mm
PIV setup technique:	2D3C, double frame, single exposure
Number of samples:	400
Recording frequency:	12Hz (13Hz for $\alpha = 45$ deg case)
Spatial resolution after post-processing:	3.3 – 5.1mm for 130mm lens
	8.9mm for 50mm lens
Planes measured:	Vortex planes
	Cross-planes every $\Delta x/c = 0.05$

Table 3.8: Description of the PIV measurements.

Chapter 4

Results

4.1 Force coefficients

4.1.1 Comparison of the results

The measured baseline normal and axial force coefficients ($Re_{MAC} = 1 \times 10^6$, $c = 0.977m$, half model) and other institutions' measurements on the same configuration are compared in figure 4.1. ONERA (French National Aerospace Research Centre) carried out low speed tests at the Lille center, ONERA L1 wind tunnel ($Re_{MAC} = 1 \times 10^6$, $c = 0.490m$) [32] and the University of Glasgow carried out force and oil flow measurements in the Argyll Wind Tunnel ($Re_{MAC} = 2 \times 10^6$, $c = 1.059m$) [5]. Both institutions conducted the experiments on full models. At low angles of attack very good agreement between the measurements in normal coefficient C_Z is seen. At $\alpha = 18 \text{ deg}$ differences between current experiment and other institutions' measurements appear. Much lower coefficient is reached by the current experiment and the peak is reached at a lower angle of attack. These differences are most likely due to other institutions carrying out the experiments on full models where interaction between leading-edge vortices from both wings exists, while the model used in the current experiment is only a half model.

As the angle of attack for a half model is increased, the reattachment line moves inwards. Larger area of the wing surface is affected by the spanwise velocity components appearing in between the reattachment line and the leading-edge, hence a bigger normal force component. In a full model case, the line stops moving at the centerline, where now it is the reattachment line for both vortices. However, for a half-model it could be the case, that the reattachment line moves further inboard and on to the peniche. As stated before, the forces acting on the

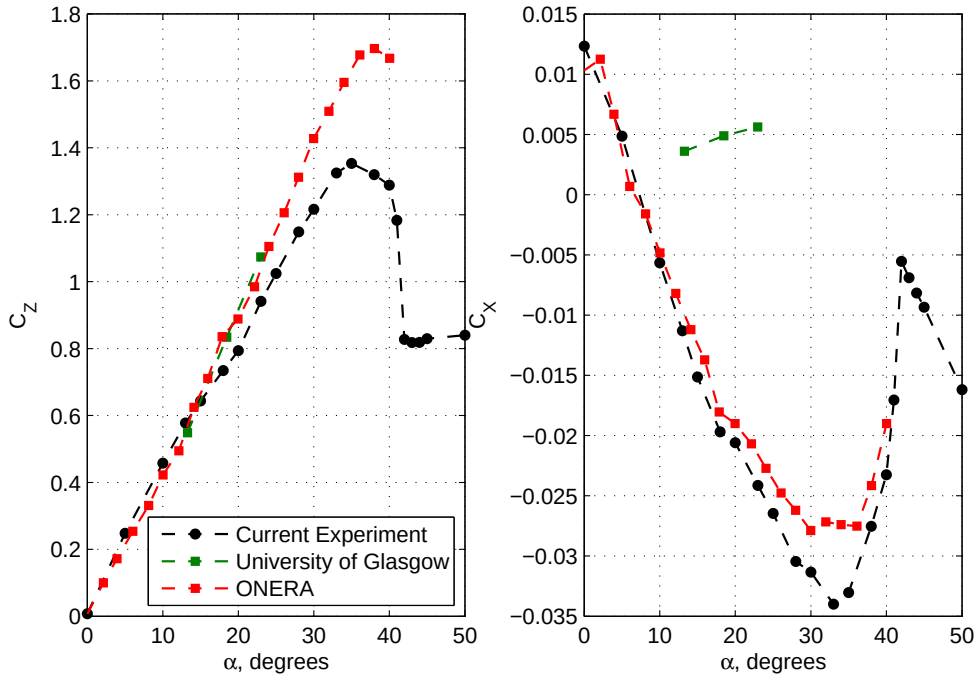


Figure 4.1: Comparison of normal C_Z and axial C_X force coefficients from the current experiment, ONERA [32] and University of Glasgow [5].

peniche are not measured, hence a part of cross-sectional velocity component that should induce additional suction on the wing surface now induces suction on the peniche and it is not reflected in the results.

To summarize, good correlation between the axial coefficients C_X in the current experiment and ONERA is reached. At ≈ 7 deg axial force becomes negative and peaks at $\alpha = 33$ deg. A sharp change in coefficient magnitude can be seen at $\alpha = 40$ deg. University of Glasgow presents positive C_X values for all of their measurements (sharp and round leading-edges). No explanation is found for why such a difference exists.

4.1.2 Lift and drag coefficients

The measured lift and drag coefficients for $U_\infty = 24\text{m/s}$ and $U_\infty = 12\text{m/s}$ of the baseline, the cases with constant frequency actuation and the spatially varying frequency configurations can be found in figures 4.2,4.3,4.4,4.5.

At $U_\infty = 24\text{m/s}$ ($Re_{mac} = 1 \times 10^6$) the maximum lift coefficient for the baseline is $C_{L_{max}} = 1.129$ at $\alpha = 33$ deg. For stall-region angles of attack, no valuable gain is achieved from any of the actuated cases. For constant frequency actuation with $f_a = 65\text{Hz}$, lift coefficient does not change. A very slight increase in drag can be seen for the same case. For configurations

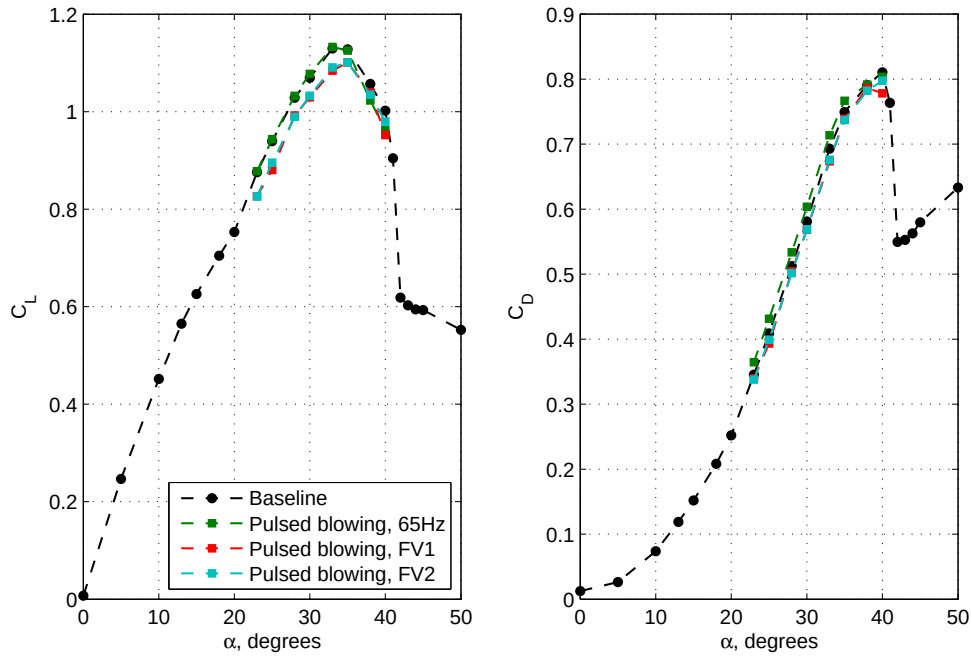


Figure 4.2: Lift and drag coefficients of the baseline and the cases with unsteady blowing. Pre-stall test cases at $U_\infty = 24 \text{ m/s}$.

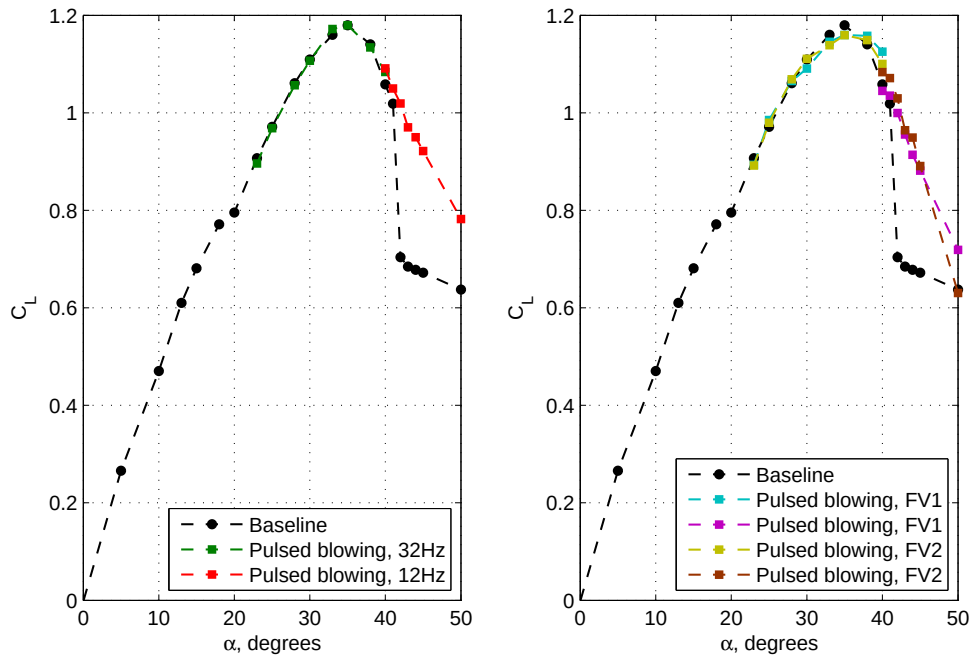


Figure 4.3: Lift coefficient of the baseline and the cases with unsteady blowing. Stall and post-stall cases at $U_\infty = 12 \text{ m/s}$. Constant blowing frequency on the left and spatially varying frequency on the right.

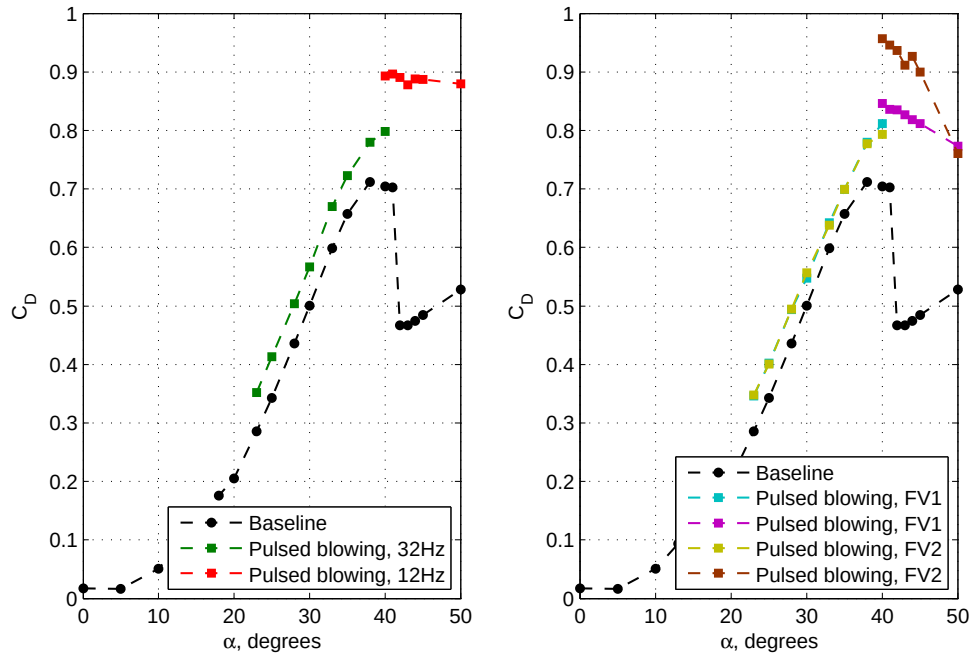


Figure 4.4: Drag coefficient of the baseline and the cases with unsteady blowing. Stall and post-stall cases at $U_\infty = 12\text{m/s}$.

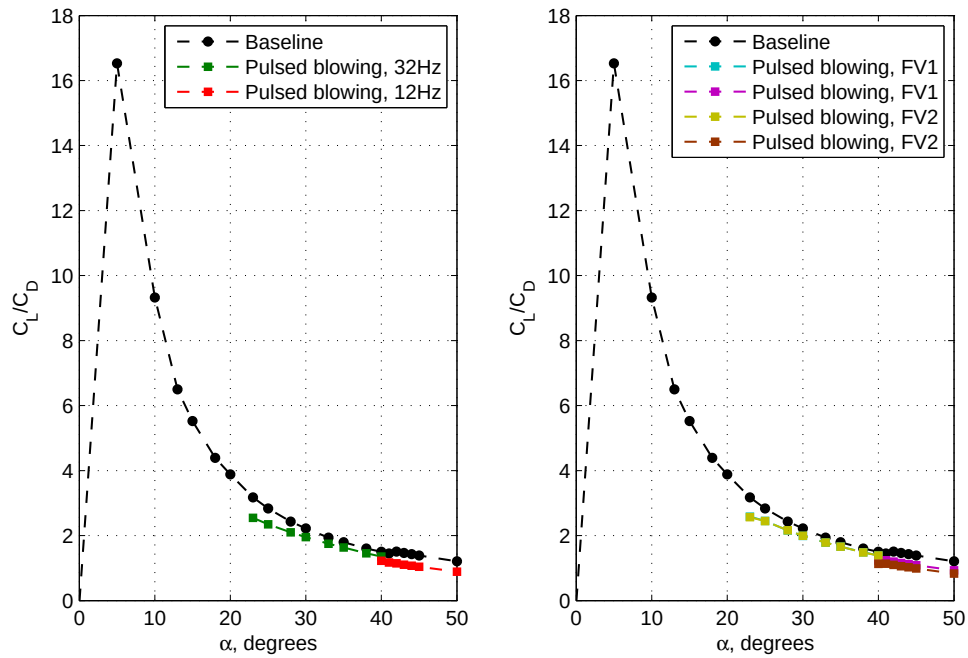


Figure 4.5: Lift-to-drag ratio of the baseline and the cases with unsteady blowing. Stall and post-stall cases at $U_\infty = 12\text{m/s}$.

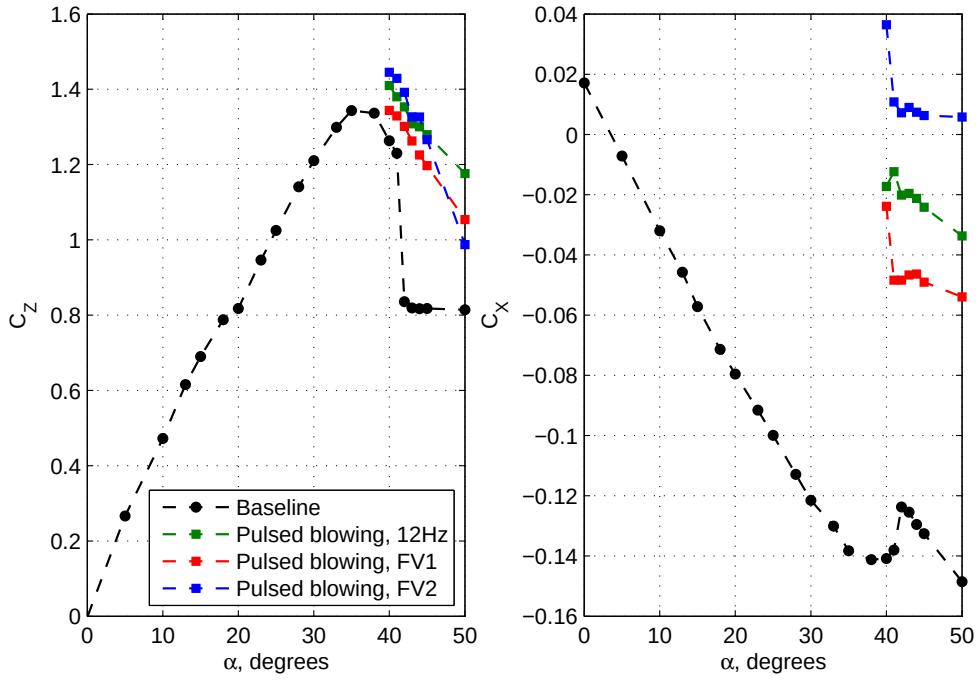


Figure 4.6: Comparison of normal and axial force coefficients.

FV1 and FV2, lift coefficient decreases by $\Delta C_L = -0.033$. Drag coefficients decrease for the same cases is negligible.

At $U_\infty = 12\text{m/s}$ ($Re_{mac} = 5 \times 10^5$) the baseline maximum lift coefficient is reached at the angle of attack of $\alpha = 35^\circ$ and equals to $C_{L_{max}} = 1.180$. In post-stall angles it decreases, leveling out at around $\alpha = 42^\circ$ with $C_L = 0.7$. With constant frequency actuation, only negligible differences can be seen in the stall region. In this region, the biggest gain with $f_a = 32\text{Hz}$ is at $\alpha = 40$, where $\Delta C_L = +0.026$. With $f_a = 12\text{Hz}$ at the same angle of attack, the gain increases to $\Delta C_L = +0.033$. For spatially varying frequency actuation cases at the same angle of attack, maximum increase is observed from Frequency Variation 1, with $\Delta C_L = +0.067$.

In the post-stall region, influence of unsteady blowing is considerably larger. With constant actuation frequency of $f_a = 12\text{Hz}$ the fall of C_L is more damped upon comparison to the baseline. At $\alpha = 42^\circ$, where the gain is largest, $\Delta C_L = +0.315$. For the spatially varying frequency configurations, the effect is found to be slightly reduced. FV2 configuration is found to achieve better results than FV1 in the range of $\alpha = 40 - 45^\circ$. However, at $\alpha = 50^\circ$ the FV1 configuration shows better improvement in C_L .

Drag coefficient graphs can be found in figure 4.4. The baseline drag peaks at $\alpha = 38^\circ$ with $C_D = 0.712$. In actuated cases, drag is significantly increased, especially at post-stall region. This is expected, since the reconstructed vortex structure increases the suction on the top surface of the wing, which at higher angles of attack translates into increased drag.

While lift coefficient does not differ too much for all the actuated cases in the post-stall region, more obvious differences can be observed while comparing drag coefficients. For $f_a = 12\text{Hz}$ frequency blowing case, the post-stall drag remains more or less constant at around $C_D = 0.88 - 0.89$. For spatially varying frequency configurations drag coefficient decreases significantly faster than for the constant frequency actuation. The peak values at $\alpha = 40^\circ$ for FV1 and FV2 are $C_D = 0.846$ and $C_D = 0.957$ respectively. At $\alpha = 50^\circ$ the values for both configurations drop to $C_D = 0.765$. FV1 clearly displays lower drag characteristics than the other two actuation cases. This furthermore leads to better lift-to-drag ratio, which on average is 4.5% higher in post-stall region when compared with constant actuation.

4.1.3 Normal and axial force coefficients

Post-stall region normal and axial force coefficients of the baseline, constant frequency actuation ($f_a = 12\text{Hz}$), FV1 and FV2 cases are presented in figure 4.6. At $\alpha = 40^\circ$ highest normal force coefficient is reached with FV2 actuation. Large normal coefficient gradient differences can be observed between all three actuation cases, with $f_a = 12\text{Hz}$ actuation exhibiting the lowest gradient and FV2 having the highest. Axial force gradients switch places, with now constant frequency actuation demonstrating the steepest negative gradient and FV2 having very small gradient.

Two explanations were found for the singularities in axial force coefficient appearing at $\alpha = 40^\circ$. First one is that the high jump in values is a result of hysteresis. As this is the first measuring point in the measuring sequence, it is possible, that the interaction between blowing and vortical structure only started when the angle of attack is being increased by one degree. The second explanation could be connected to the nature of the flow. For the baseline, a big change in values is seen at $\alpha = 41^\circ$. That is the angle of attack where the flow type changes from slender body to blunt body. Hence, with actuation this limiting angle of attack could possibly move in one direction or the other. Measurements over wider range of angles should be conducted to prove one theory or the other.

To summarize, reasonably good agreement with measurements done by ONERA is reached. Obvious gain in lift at post-stall angles of attack is reached with all actuation configurations, while at pre-stall and stall regions no significant change is achieved. Actuation not only affects the normal force coefficient, but it as well leads to noticeable changes in the axial force coefficient.

4.2 PIV measurements, the baseline case

In this section, baseline measurements at angles $\alpha = 23^\circ$, $\alpha = 35^\circ$ and $\alpha = 45^\circ$ are presented. The spatial layout of the measurement planes for $\alpha = 23^\circ$ can be seen

in figure 4.7. Here the averaged and normalized axial velocity component u/U_∞ fields are displayed. The measurements planes are blanked further away from the wing surface. For angles $\alpha = 35$ deg and $\alpha = 45$ deg the layout is very similar.

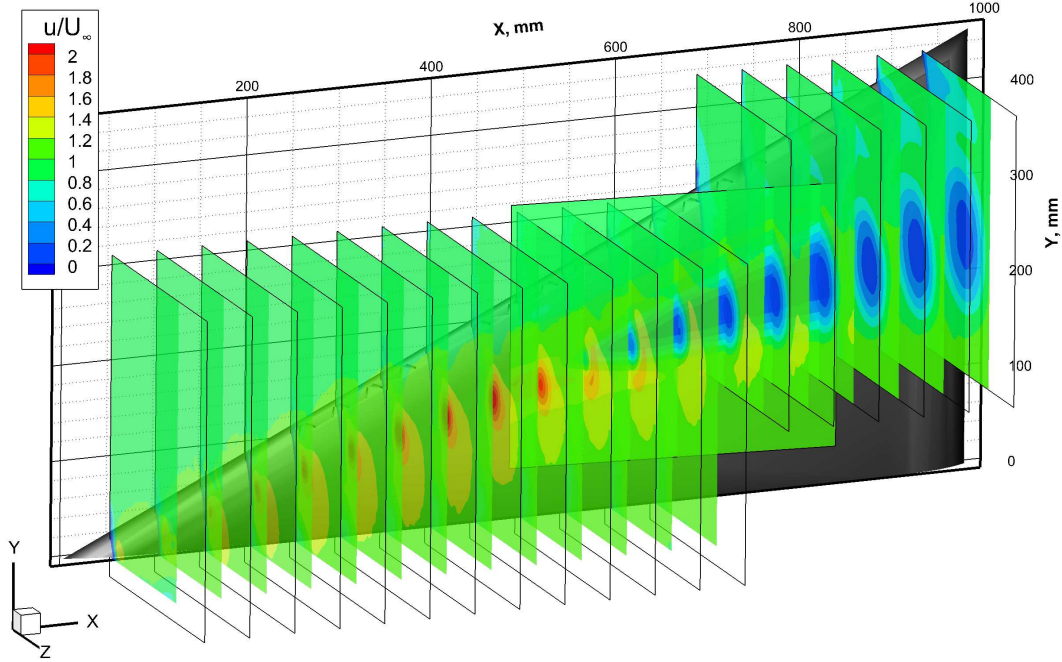


Figure 4.7: All PIV measurement planes visualized. Normalized axial velocity field, baseline case. $U_\infty = 24\text{m/s}$, $\alpha = 23$ deg.

4.2.1 $\alpha = 23$ deg

Measurements for $\alpha = 23$ deg case were done at $u/U_\infty = 24\text{m/s}$ ($Re_{MAC} = 1 \times 10^6$). Complete flow field was recreated by interpolating between the measurement planes. In figure 4.8, four isosurfaces are shown, depicting the vortex core and region after the breakdown. It has to be noted, that the flow close to the wing surface here is blanked only to show the primary vortex structure. Pathline of a core particle is visualized. It can be seen, how conservation of momentum for the particle is satisfied. Before breakdown, the particle has high axial velocities and it moves in an almost straight path. After the breakdown, the axial momentum is transformed into angular momentum and helical movement becomes apparent.

Selected planes are shown in more detail in figure 4.9 to represent the vortex evolution.

The maximum averaged axial velocity in the core is reached at $x/c = 0.45$ with $u/U_\infty = 2.2$. The high-velocity part can be clearly seen up to $x/c = 0.5$ (figure 4.9). Here normalized vorticity $\Omega_x = \omega_x s / 2U_\infty$ contour lines indicate that the vorticity peak does not coincide with maximum axial velocity peak. However, after investigating the v/U_∞ and w/U_∞ velocity

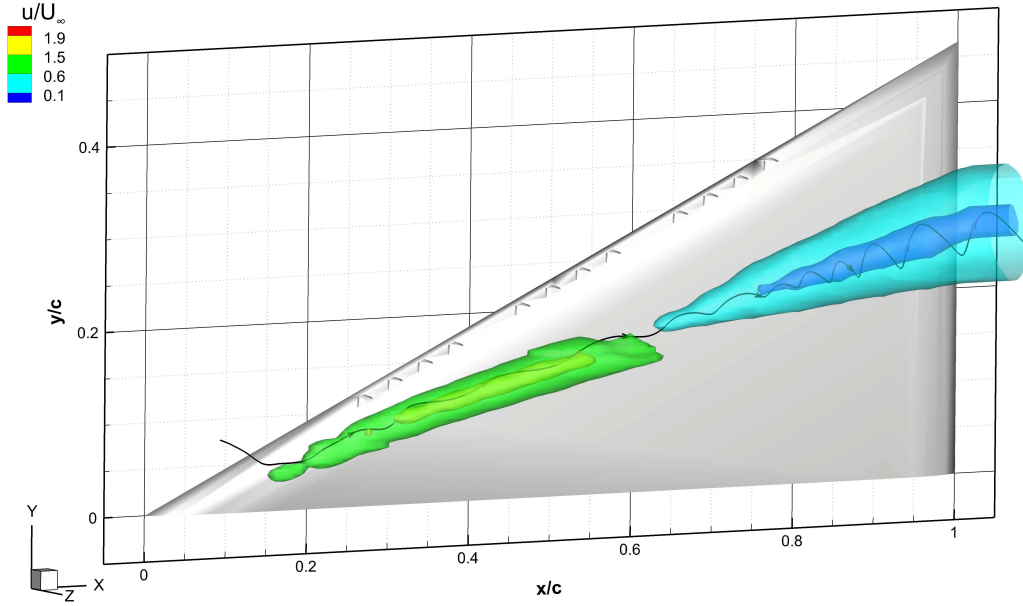


Figure 4.8: Isosurfaces of normalized axial velocity at $u/U_\infty = 0.1, 0.5, 1.5, 1.9$. Stream-trace inside the vortex core is visualized. The flow close to the wing surface is blanked.

fields, it has been deduced that this difference is smaller than the measurement plane finite grid size and most probably is due to the finite difference schemes used for calculating the vorticity. In between the vortex core and the wing surface, patches of negative vorticity exist. It is not exactly clear from the figure if this is all created by a single secondary vortex that appears because of separation induced by the vortex core. However, using the λ_2 -criterion (figure 4.10, (a)), four discrete vortices are visible. The primary vortex, with lowest λ_2 value, two discrete vortices before pairing and the secondary vortex. This is a good example how a vortex, that would be ambiguous with vorticity plot only can be detected using the λ_2 -criterion.

At the apex, big fluctuations appear around the leading-edge (figure 4.11). As the leading-edge vortex here is not so strong, the structure is heavily influenced by the shear layer. Between $x/c = 0.15$ and $x/c = 0.20$ the fluctuations become concentrated at the axis of the vortex, where high velocity gradients exist.

Vortex core details

Flow in the vortex core is inspected by looking at plane $x/c = 0.50$. Circumferential (v/U_∞ and w/U_∞) and axial (u/U_∞) velocity values are extracted into figure 4.10, (b).

At $x/c = 0.50$, the vortex core is located at $y/c = 0.159$ and $z/c = 0.069$. Here the circumferential velocities are equal to zero. The maximum axial velocity locations do not exactly

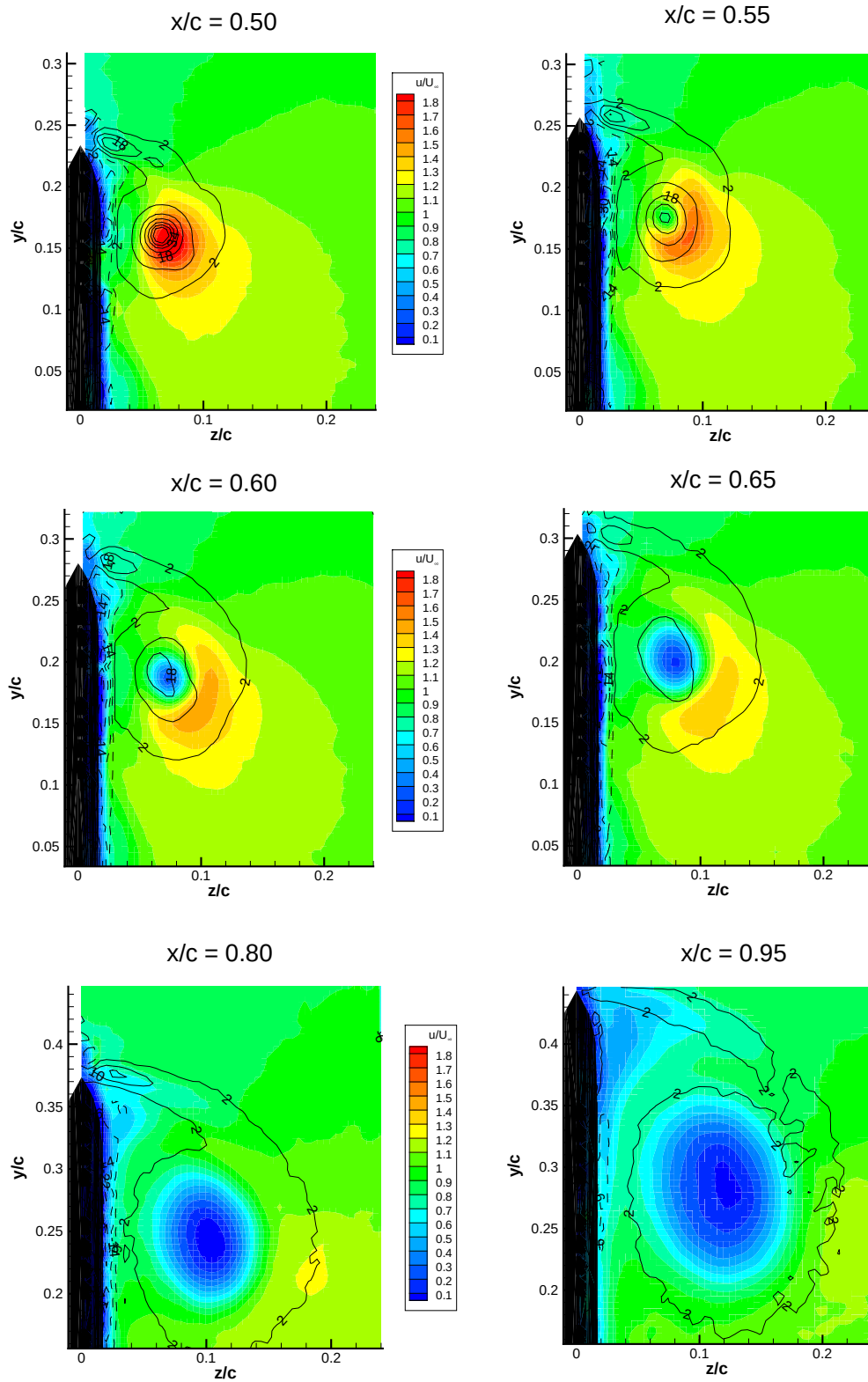


Figure 4.9: Baseline vortex evolution. Contour plots represent normalized axial velocity. Contour lines represent normalized vorticity $\Omega_x = \omega_x s / 2U_\infty$ every 8 units. Dashed lines represent negative vorticity values. $U_\infty = 24\text{m/s}$, $\alpha = 23\text{deg}$

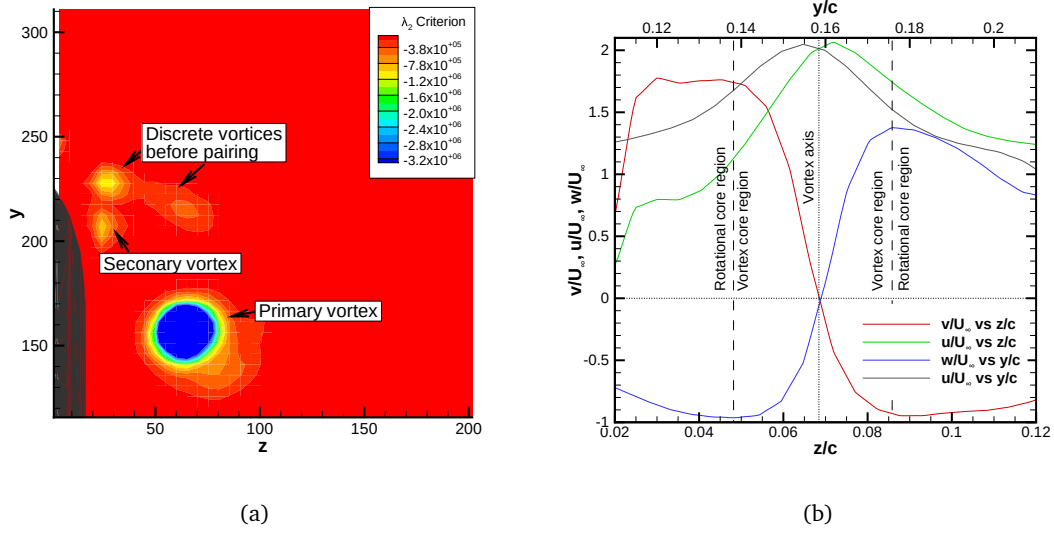


Figure 4.10: a) λ_2 criterion plot at $x/c = 0.50$, no actuation. b) Vortex core details.
 $x/c = 0.50$, $U_\infty = 24\text{m/s}$, $\alpha = 23^\circ$

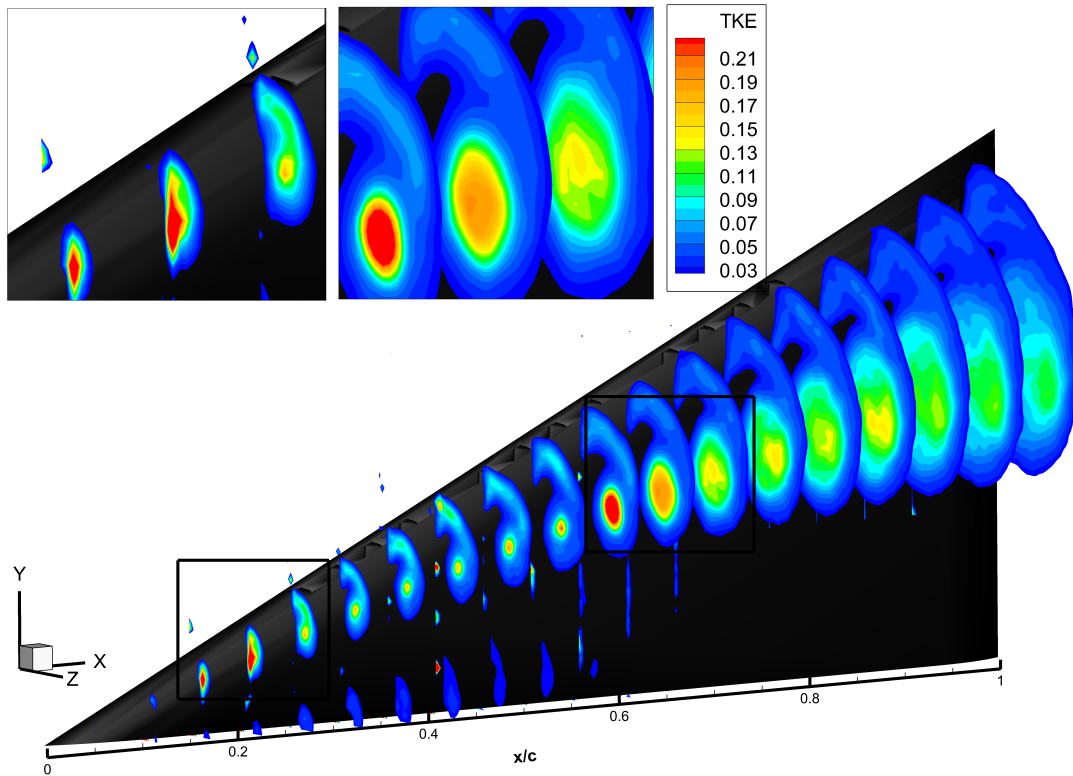


Figure 4.11: Turbulent kinetic energy distribution for the baseline case. The values below $TKE = 0.025$ are not shown here. $U_\infty = 24\text{m/s}$, $\alpha = 23^\circ$

correspond to the vortex axis due to spatial resolution. The distance between the maximum axial velocity and the zero of circumferential velocity here is 3.5mm for both coordinates, whereas the spatial resolution for this specific plane is 5mm .

Vortex core region here is identified according to w/U_∞ profile. Vortex core diameter is 37mm . A symmetrical velocity distribution, typical to an ideal decaying Lamb-Oseen vortex here is not apparent as there are strong interactions between the vortex core and the wall, and the shear layer as well. However, velocity distribution in y/c direction (spanwise) is more symmetrical than the z/c direction (normal to the wing surface). Hence it could be deduced that the vortex core velocity distribution is more influenced by the presence of the surface than of the shear layer.

Vortex breakdown

Between planes $x/c = 0.5$ and $x/c = 0.55$ the pressure gradient becomes too adverse for the vortex to overcome and vortex breakdown occurs. Retarded axial flow region starts to form on the vortex axis, indicated by the vorticity peak as seen in figure 4.9. The fluctuations at the breakdown location reach the maximum. A kidney-shaped high axial velocity flow zone appears, indicating precession of the vortex core around the axis of symmetry [26]. Therefore, at $x/c = 0.55$ plane the spiral structure of the burst vortex is already present.

Two main indications for the spiral-type breakdown can be made from analysis of velocity fluctuations. First, the TKE distribution in figure 4.11 becomes asymmetric and as well off-centered after the breakdown. As the vortex core now rotates in a spiral, it never (in ideal case) passes the rotational axis. Hence the fluctuations in the axis region come only from fluctuations of the retarded flow. However, in the annular region further away from the axis, more fluctuations are expected. This is due to the unsteady nature of the spiral vortex and as well the interaction between the vortex and the retarded flow region. Second indication can be seen in the vortex plane u_{RMS}/U_∞ distribution 4.12. The peak is seen at the breakdown point, with higher values then following the contour of retarded region creating a conical shape, typical to the spiral breakdown type.

In comparison, from hot-wire measurements on the same model and same angle of attack, the breakdown location was measured to be in between $x/c = 0.30$ and $x/c = 0.35$ [22]. The forward breakdown location from hot-wire measurements could be the result of measuring probe intruding the flow in the subcritical flow area. Blanchard [2] did unsteady pressure measurements on the same model and found the breakdown location to be between $x/c = 0.60$ and $x/c = 0.80$. It has to be noted, that both measurements were done in the smaller wind tunnel of the institute (Wind Tunnel B, Technische Universitat Munchen). Furman and Breitsamter [8] did full model surface pressure measurements and determined the breakdown location to be at $x/c = 0.68$ ($Re_{MAC} = 2 \times 10^6$). From computational simulations with adapted

Euler code, Muller and Hummel [30] records the breakdown location at $x/c = 0.70$.

The differences in breakdown locations as well could result from the ambiguity of the definition of breakdown location. For example, if the breakdown is said to exist where the vortex core axial velocities start to decrease, then the location could be recorded as $x/c = 0.50$. If it is where TKE reaches maximum, from cross-plane measurements it would be $x/c = 0.55$. From vortex plane measurements it would be $x/c = 0.55$ (figure 4.12) as well.

Vortex “wandering”

At the breakdown location, high axial velocity vortex core changes direction (figure 2.5). Also, the breakdown location oscillates front and aft along the axis, which is usually referred to as vortex “wandering”. Hence u_{RMS}/U_∞ values grow significantly, peaking at $x/c = 0.55$. Even though the exact reason for these axial oscillations is unknown, correlation between the breakdown location movements and appearance of Kelvin-Helmholtz instability is proved to exist [13].

The frequency of the breakdown location oscillations was suspected to be within frequency range of $f c/U_\infty = 0.04 - 0.15$ ($f = 1 - 3.7Hz$) [14]. Following Nyquist limit criterion, frequencies up to $f = 6Hz$ can be detected with the current measuring system, which has a sampling frequency of $f_s = 12Hz$. A rectangular window just in front of the breakdown location was selected and mean of the axial velocity V_x was calculated for each time step. The result was transformed into frequency domain using Fast Fourier Transform algorithm. Resulting frequency spectrum can be seen in figure 4.12.

Four noticeable peaks are visible, that are very similar in amplitude. One of them is located at $k = f c/U_\infty = 0.142$ ($f = 3.5Hz$), which falls in the suspected frequency range. The origin of the other peaks is unknown. However, with the current number of samples (400 frames) and the low sampling frequency, the results have to be interpreted with care.

Post-breakdown region

Following the vortex breakdown, a retarded flow region appears, which increases in size towards the trailing edge. Here the minimum averaged axial velocity value drops down to $u/U_\infty \approx 0$ at $x/c = 0.85$. Kidney-shaped high axial velocity region slowly decreases in size and almost completely disappears at $x/c = 0.95$. This can be explained by viscous dissipation, as the spiral vortex core is already broken down into small scale turbulence. Vorticity values drop significantly as well. Another recirculation zone in between the shear layer and the burst vortex structure here becomes more visible, with decreasing velocity magnitude in all components.

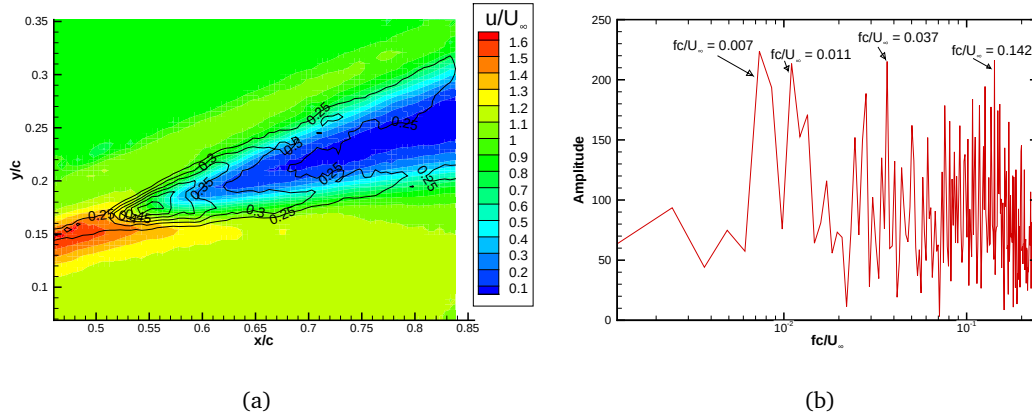


Figure 4.12: a) Vortex plane measurement. Contour colours represent the normalized axial velocity u/U_∞ values. Contour lines represent u_{RMS}/U_∞ values. b) Spectra of breakdown location variation. Significant peaks are marked. $U_\infty = 24\text{m/s}$, $\alpha = 23\text{deg}$

A big change is as well seen while comparing turbulent kinetic energy before and after the breakdown. Before the breakdown, most of the turbulent kinetic energy is concentrated within the vortex core (figure 4.11). After the breakdown, the energy becomes distributed within regions of bigger diameter and maximum values drop significantly. Asymmetrical turbulent energy distribution is apparent, which is usually attributed to the spiral mode breakdown. Fluctuations decrease downstream, as viscous dissipation takes effect.

4.2.2 $\alpha = 35\text{deg}$

Measurements of $\alpha = 35\text{deg}$ case are done at $U_\infty = 12\text{m/s}$ ($Re_{MAC} = 5 \times 10^5$). Selected planes are presented in figure 4.13. Downstream cross-sections at this angle of attack are measured with three different calibrations. As the calibrations were not perfect, separating lines in between the different calibration planes can be seen at $y/c = 0.31$ and $z/c = 0.25$.

At $x/c = 0.10$ plane the averaged flow already presents axial velocity values of $u/U_\infty = 0.17$ and at $x/c = 0.25$ negative velocities appear with values as low as $u/U_\infty = -0.2$. Negative velocity region spans all the way to the trailing-edge conically increasing in size. At planes $x/c = 0.80 - 0.90$ the region remains the same size and at the plane $x/c = 0.95$ it slightly decreases.

The turbulent kinetic energy contours for all the planes are pictured in figure 4.14. 3 distinct regions can be identified: $x/c = 0.05 - 0.15$, $x/c = 0.15 - 0.60$ and $x/c = 0.60 - 0.95$.

Up to plane $x/c = 0.15$, high values of turbulent energy ($TKE > 0.2$) are present, mostly concentrated in the outwards point of a bow shaped region, starting from the leading-edge.

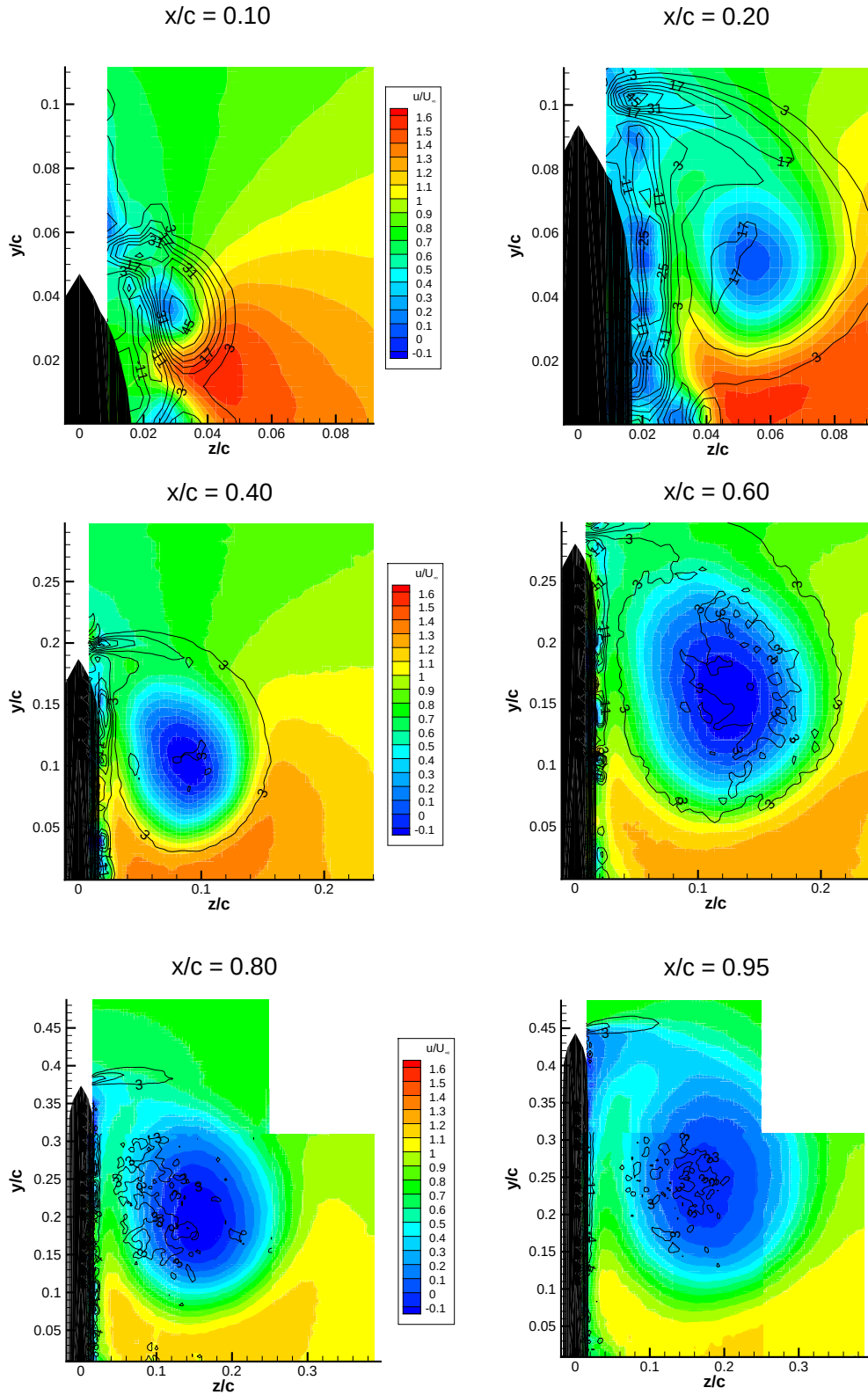


Figure 4.13: Baseline vortex evolution. Contour plots represent normalized axial velocity. Contour lines represent normalized vorticity $\Omega_x = \omega_x s / 2U_\infty$ every 7 units. Note the change in scale for different rows. $U_\infty = 12\text{m/s}$, $\alpha = 35^\circ$

The basis of high apex TKE values might be the same as was for $\alpha = 23$ deg case. As the rotational structure (vortex core in $\alpha = 23$ deg case) is a weak one, the shear layer and penicillate interaction instabilities are more influential on it.

At plane $x/c = 0.20$ the TKE values drop significantly, starting the second region, even though no comparable changes in the mean velocities can be seen. The energy is now distributed within the retarded flow region, with local maximums close to the wing surface and leading-edge. Fluctuations in this region might be a result of mixing of three distinct flow regions- wall bounded flow, rotational burst vortex flow and shear layer that is being shed from the leading-edge. The wall region flow has positive velocities in axial direction and velocity gradients due to boundary layer. The rotational burst vortex structure has big recirculation zones of negative axial velocities which constantly mix with the wall bounded flow. High gradient vortical shear layer flow is introduced near the leading-edge leading to the concentration of turbulence at the meeting point of the three flows.

The turbulence maximums increase in magnitude until plane $x/c = 0.60$ (highest value at $x/c = 0.55$, $TKE = 0.19$), where it starts dropping again. Now the energy is clearly seen to be distributed in an annular region, with local minimum in the center of the structure. Both the minimum and maximum values within plane decrease.

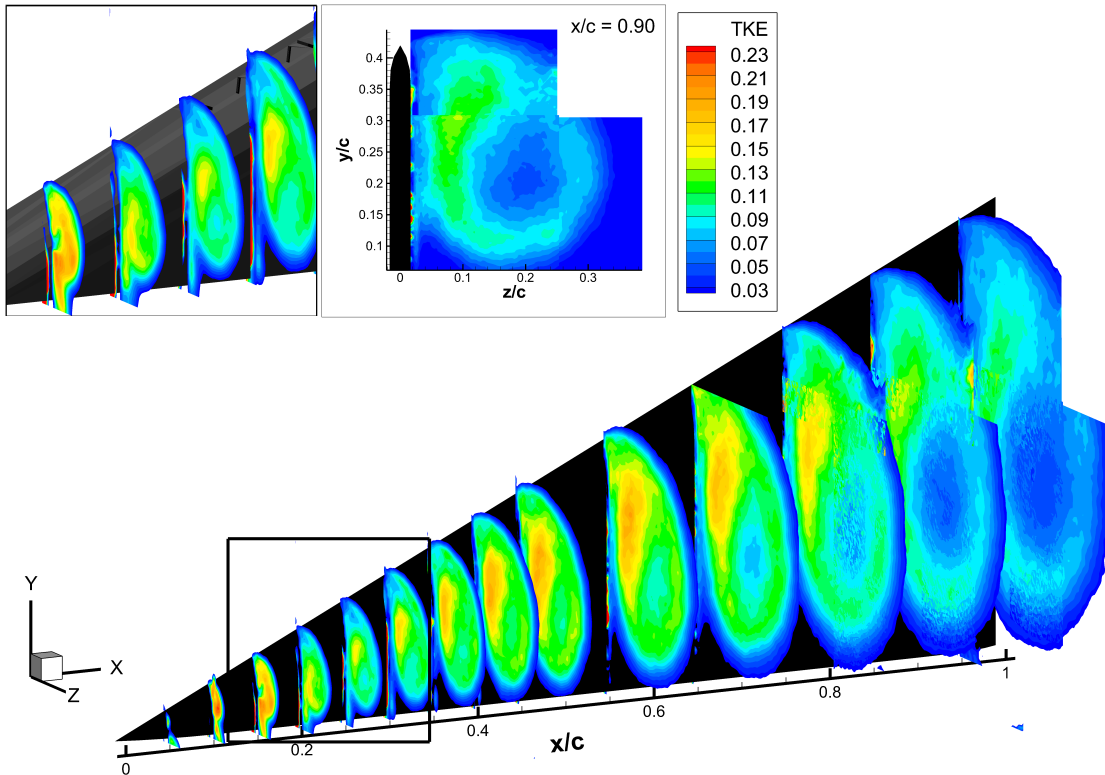
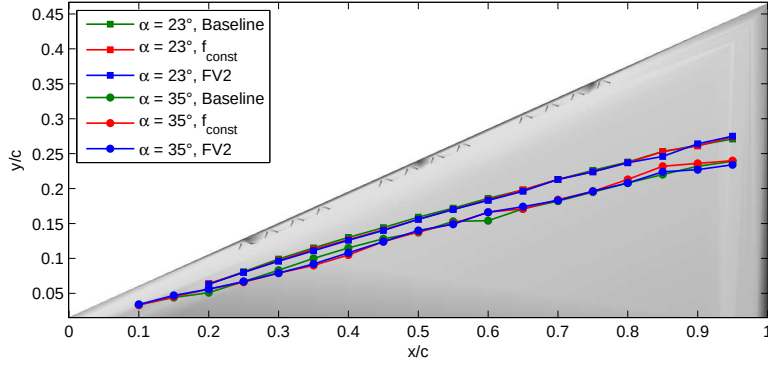


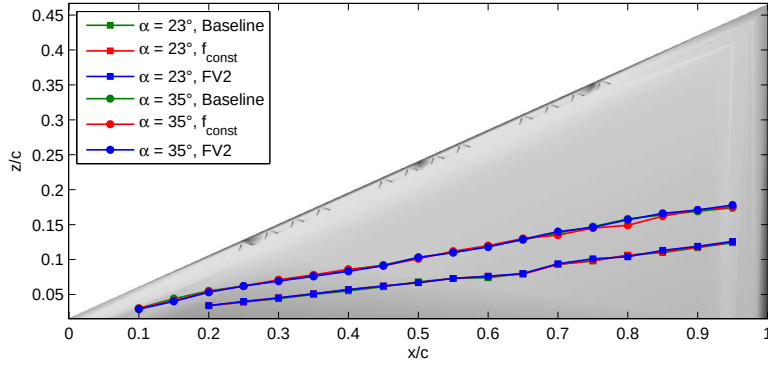
Figure 4.14: Turbulent kinetic energy distribution for the baseline case. The values below $TKE = 0.1$ are not shown here. $U_\infty = 12m/s$, $\alpha = 35$ deg

Axis of rotation position

Axis position for $\alpha = 23$ deg and $\alpha = 35$ deg cases are presented in figure 4.15.



(a)



(b)

Figure 4.15: Axis of rotation position. a) y/c - spanwise b) z/c - normal away from the wing

For $\alpha = 23$ deg case, the rotational axis (vortex core and burst vortex structure axis) is linear in y/c direction. In z/c direction (normal away from the wing), a change in axis position can be seen around the breakdown location. As the retarded flow diameter increases and velocities in the region decrease, the mixing layer between the retarded flow and the wall flow increases in thickness. Hence the conical post-breakdown region is pushed slightly away from the wing.

For $\alpha = 35$ deg case the axis is very slowly turning inwards (increase in y/c direction is getting smaller) up to $x/c = 0.60$ and then becomes linear. This could be a result of the shear layer pushing the big burst vortex structure inwards. Linear trend is visible for z/c direction.

For comparison, investigation on vortex axis position was done by Furman and Breitsamter ($Re_{MAC} = 2 \times 10^6$) on a full VFE-2 model [8]. Lateral position was described by θ_{VA} , angle

between the axis of symmetry and the vortex axis. For $\alpha = 23^\circ$, the angle was $\theta_{VA} = 18.4^\circ$, whereas for current measurements the angle is slightly lower, $\theta_{VA} = 16.4^\circ$. This is expected, as for a full model each leading-edge vortex makes sure that the other one does not get too close.

For the vertical position, parameter h , being the height of the axis above the wing at the trailing-edge, is used. From other experiments it was equal to $h = 0.075c$ [8]. Current experiment recorded the value of $h = 0.13c$, almost twice as high. The vertical position of the axis is dependent on the size of the burst vortex structure, hence large deviations can be achieved with different vortex breakdown locations, which are sensible to many external conditions. One explanation for the axis location being so high might be because the vortex in the current experiment bursts further up front than for the experiment done by Furman and Breitsamter, hence creating a much bigger burst structure, and at Reynolds number twice as high as well. Also, other experiment recorded the axis position with using smoke as an indication, whereas in the current experiment vortex center was defined precisely from PIV measurements, where both v and w values were equal to zero.

For $\alpha = 35^\circ$ case, the position parameters were $\theta_{VA} = 13.9^\circ$ and $h/c = 0.17$. From previous experiments on the same model, from which the vortex axis angle was calculated, value of $h/c = 0.19$ was used. Extrapolating data from Furman and Breitsamter, values $\theta_{VA} = 16.1^\circ$ and $h/c = 0.12$ are obtained [8] (their data ends at $\alpha = 30^\circ$). From these differences it can be concluded that the flow axis position, is heavily dependent on external parameters.

4.2.3 $\alpha = 45^\circ$

Averaged axial velocity field is presented in figure 4.16.

At this high angle of attack, the leading-edge vortex does not form anymore. Adverse pressure gradient over the wing surface is very high thus creating a massive blockage region with negative axial velocities. Near the leading-edge, the shear layer can be easily identified from the positive vorticity lines (see the top left corner of figure 4.16). The shear layer closely follows the retarded flow region.

Shear layer vorticity profiles are very similar for cross-planes at the front half of the wing. However, because of the laser reflections, the flow details near the leading-edge for planes $x/c = 0.55 - 0.95$ were lost, hence the values of vorticity after plane $x/c = 0.50$ seem to be smaller.

The change in shape of the flow structure, starting from plane $x/c \approx 0.50$ arises from flow reattachment to the wind tunnel floor. A clearer picture can be seen when looking at velocities in spanwise direction v/U_∞ (figure 4.17). Two iso-surfaces are shown, one at $u/U_\infty = -0.2$ and the other at $v/U_\infty = 0.05$. A region of positive spanwise velocities is seen to form at the

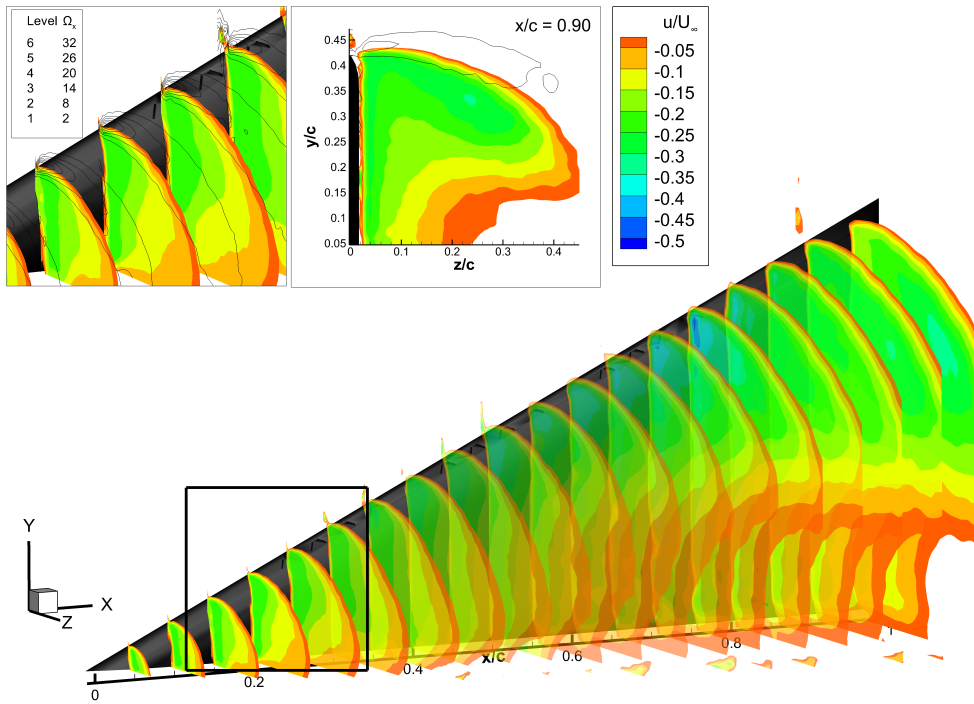


Figure 4.16: Averaged axial velocity field of the baseline. Only negative velocities are shown here. The contour lines in corner figures represent positive normalized vorticity lines. $U_\infty = 12\text{m/s}$, $\alpha = 45\text{deg}$

bottom, where the peniche and wind tunnel floor is. Additional momentum is injected from the “symmetry plane” into the flow, which would not happen for a full-scale model. Judging from the shape of the iso-surface (widening downstream), it is a separated flow from the wind tunnel floor.

Vorticity feeding

Comparison of vorticity feeding through the shear layer for all cases is included here. The analysis was done by integrating normalized axial vorticity Ω_x values across the line, crossing the shear layer where values of vorticity are $\Omega_x \geq 5$.

It was discovered that vorticity feeding for $\alpha = 23\text{deg}$ and $\alpha = 35\text{deg}$ cases is not constant throughout the leading-edge, but is actually decreasing downstream 4.18. The same trend is seen while analyzing only the peak values of the vorticity inside the shear layer. The explanation for this could lie within the interaction of the free-stream flow and the flow on the bottom surface. In figure 4.19 it is visible, that the angle between the boundary layer flow on the bottom surface at the leading-edge and the free-stream (vertical direction) is getting lower downstream. Hence the cross-flow component, on which magnitude of vorticity depends the most, decreases downstream as well.

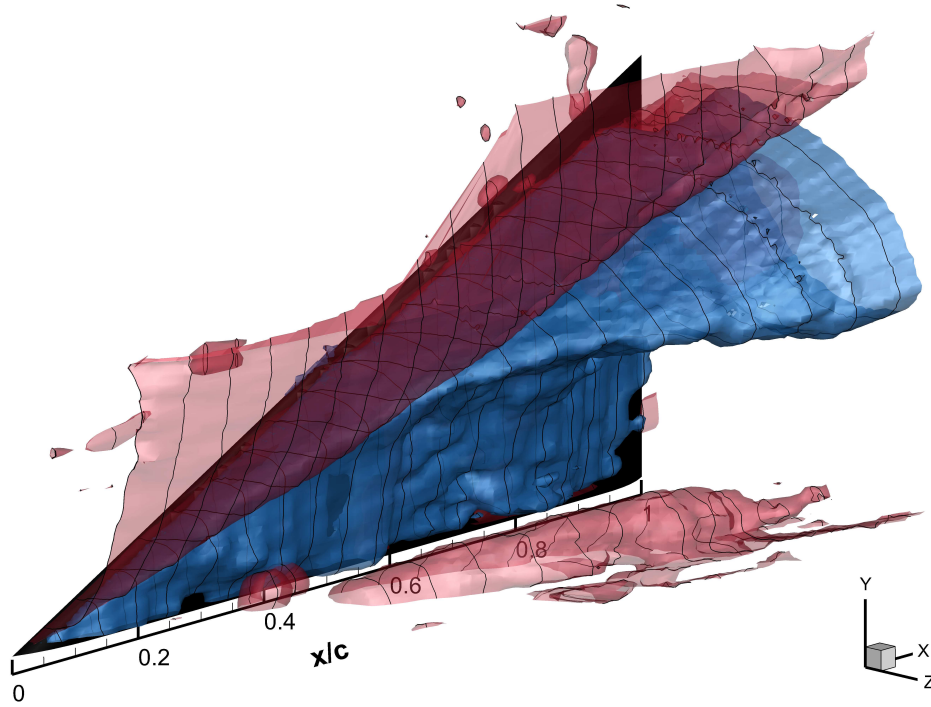


Figure 4.17: Interpolated baseline velocity field. Here shown are two iso-surfaces. Blue is at $u/U_\infty = -0.2$ and red is at $v/U_\infty = 0.05$. Contour lines show x/c planes. $U_\infty = 12\text{m/s}$, $\alpha = 45\text{ deg}$

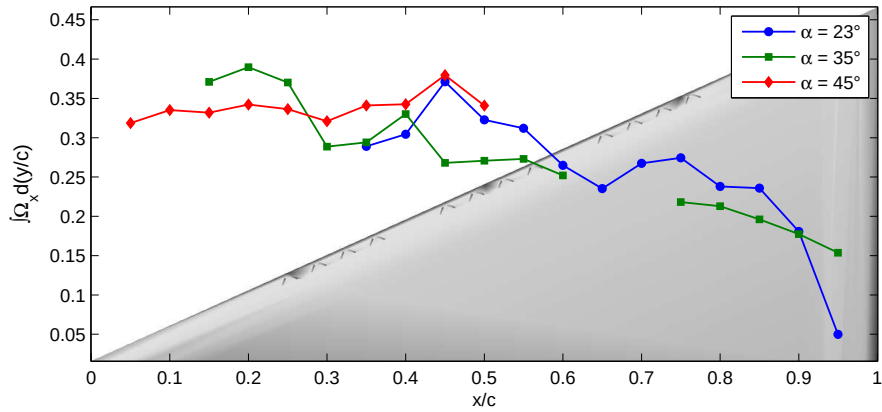


Figure 4.18: Vortex evolution for the actuated cases. $f_a = 65\text{Hz}$ on the left, FV2 on the right. $U_\infty = 24\text{m/s}$, $\alpha = 23\text{ deg}$

For $\alpha = 45\text{ deg}$ case, the values show an almost linear and rising trend. However, as no data near the leading-edge was available for planes $x/c = 0.55 - 0.95$, it is hard to estimate the further development.

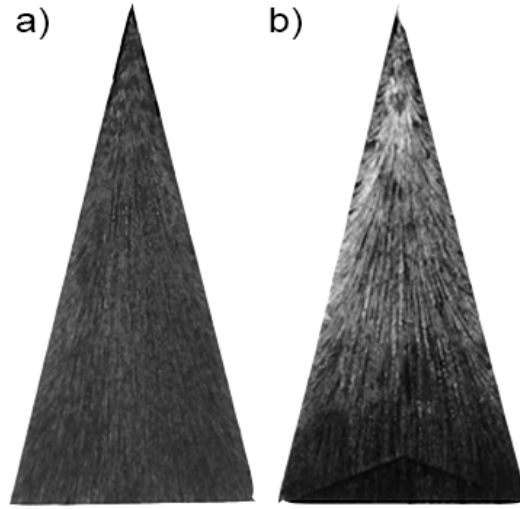


Figure 4.19: Oilflow patterns at $\alpha = 20.5$ deg. a) $Re = 0.9 \times 10^6$, b) $Re = 2 \times 10^6$. [15]

4.3 PIV measurements, the actuated cases

PIV measurements only for constant frequency and the Frequency Variation 2 actuations were done. FV2 was selected based on its better performance in lift coefficient at the three investigated angles of attack.

4.3.1 $\alpha = 23$ deg

Actuation for $\alpha = 23$ deg reveals minor changes in velocity fields. Selected cross-planes are presented in figure 4.20.

No major differences can be seen in between the two actuation flow fields. However, in comparison to the baseline, vortex breakdown position moves downstream. At $x/c = 0.55$, no breakdown is yet visible, as it was the case with the baseline. From comparing the values of u_{RMS}/U_∞ for the three cases along the vortex axis (figure 4.21), the peaks for both actuated cases are seen to move downstream to $x/c = 0.58$ for constant frequency actuation and to $x/c = 0.60$ for FV2.

Small differences can as well be seen while comparing the averaged axial velocity fields (figure 4.21). Until the breakdown, lower axial velocities in the core are seen than for the baseline. However, the retarded region velocities increase almost everywhere, and especially near the rotational axis. By comparing the $u/U_\infty = 0.2$ contour lines a clear difference is seen.

To summarize, with actuation the breakdown location is moved backwards and retarded region velocities increase, but the high-speed vortex core axial velocities get lower. It might be explained with the following. The extra momentum coming from the actuation before the

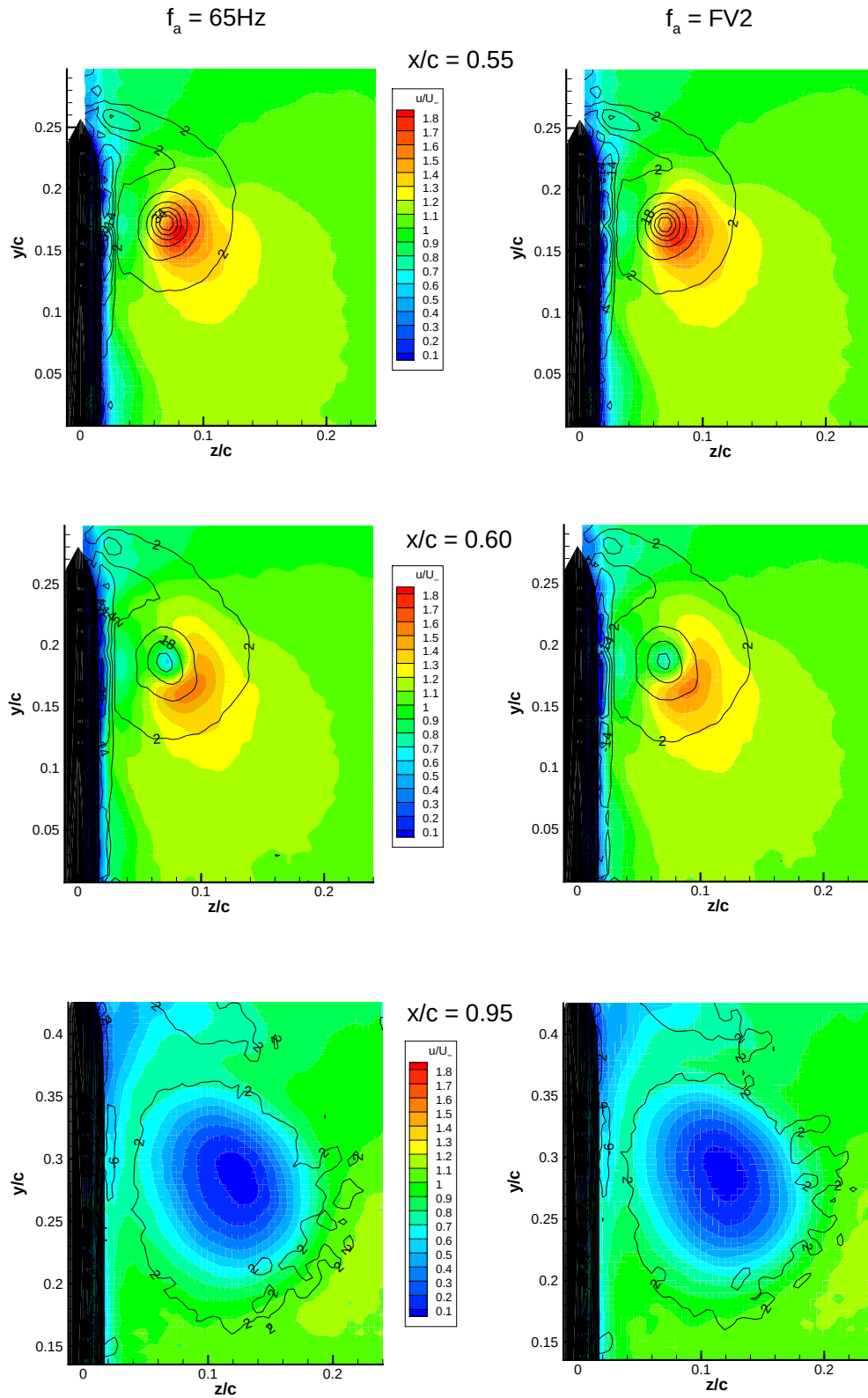


Figure 4.20: Vortex evolution for the actuated cases. $f_a = 65\text{Hz}$ on the left, FV2 on the right. Contour lines represent normalized vorticity $\Omega_x = \omega_x s / 2U_\infty$ every 8 units. $U_\infty = 24\text{m/s}$, $\alpha = 23^\circ$

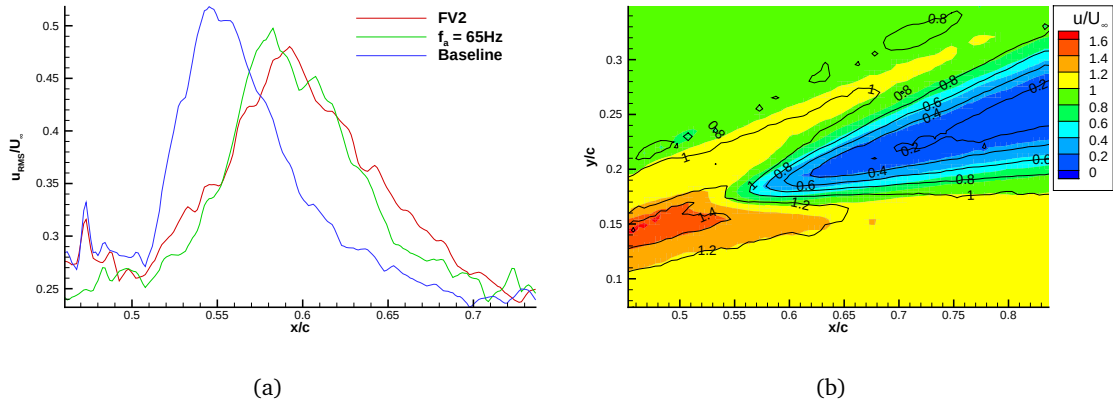


Figure 4.21: a) Plots of u_{RMS}/U_∞ values along the vortex axis for all measured cases. b) Axial velocity comparison between the baseline (contour plots) and actuated case, $f_a = 65\text{Hz}$ (contour lines). $U_\infty = 24\text{m/s}$, $\alpha = 23\text{deg}$

breakdown (first slot segment) does not get diffused with the leading-edge vortex structure. It could be, that the unnatural vortices created by the blowing do not pair with the natural ones coming from the leading-edge. Hence they are left rotating further away around the vortex core disturbing the stable core flow and hence decreasing the axial velocities. Then the second segment, which is exactly at the breakdown location, adds momentum to an already weak and dissipated vortex and hence the breakdown point moves a little bit downstream. The third segment interacts with the spiral vortex core. By comparing the RMS fields (not shown here) it can be seen that the higher RMS values span further downstream along the edges of the retarded flow region- spiral structure becomes longer and doesn't break up as fast, hence increasing the velocities along the rotational axis of the structure.

4.3.2 $\alpha = 35\text{deg}$

With the actuation on, a few important changes are seen within the flow structure.

Firstly, the blowing jet becomes visible in the measurement planes. In $\alpha = 23\text{deg}$ it could not be seen, maybe due to the freestream velocity being too high. However, in the current case, when comparing individual frames of plane $x/c = 0.30$ (just behind the second blowing slot, not shown here), with freestream velocity now being $U_\infty = 12\text{m/s}$, a very high w component velocity region ($w \approx 25\text{m/s}$) roughly 30mm in front of the blowing slot can be seen. The region coincides with highest w_{RMS} region as well. Interestingly, a vortical structure created by the actuation rotates in direction opposite to the leading-edge vortex ($\Omega_x \leq 0$).

Figure 4.22 presents turbulent kinetic energy distribution across the flow field for constant frequency actuation. Selected planes from both actuations with normalized averaged axial velocity and vorticity are presented in figure 4.23. The differences are described below.

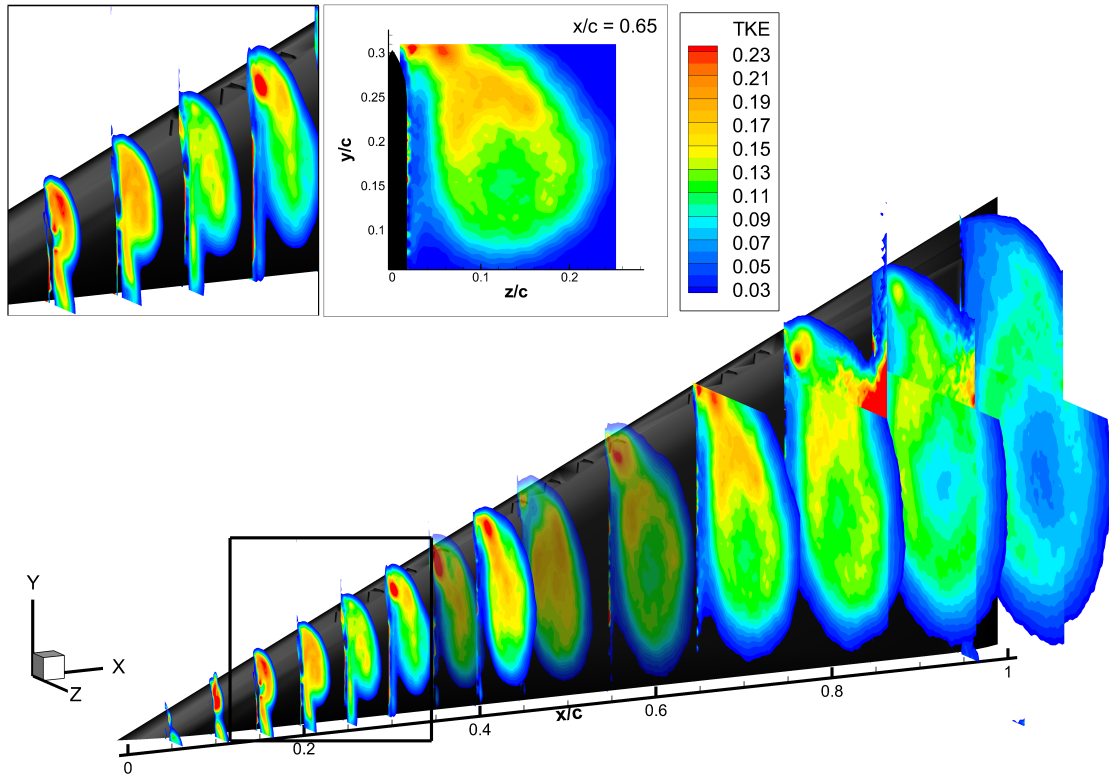


Figure 4.22: Turbulent kinetic energy distribution for the actuated case, $f_a = 35\text{Hz}$. The values below $TKE = 0.1$ are not shown here. $U_\infty = 12\text{m/s}$, $\alpha = 35^\circ$

The distribution of TKE up to plane $x/c = 0.20$ is very similar in between the actuated and un-actuated cases. However, the apex region of the wing has slightly bigger fluctuations (higher TKE values) than the baseline, even upstream of the actuation segments, meaning that the disturbances created by the actuation propagate both ways, upstream and downstream. This is expected, as the flow is subsonic and the flow behind the breakdown point is subcritical (section 2.2.3).

For constant actuation case, bigger changes are seen further on. At plane $x/c = 0.30$, where two blowing slots are already upstream, two regions of high TKE exist. The region with $TKE \geq 0.23$ is definitely created by one or both slots, as similar shapes are found in the planes close to the actuation segments downstream. It is hard to tell if the second region, where $TKE \approx 0.2$, is created by the very first blowing slot or if it is just a structure from the burst leading-edge vortex. At planes $x/c = 0.35$ and $x/c = 0.40$, the most fluctuating region of the flow again is very close to the leading-edge. The same question arises, if the region is a result of a single actuation slot or if it is a combined structure created by the actuation. One way or another, the structure is clearly seen to exist up to $x/c = 0.45$ where the second actuation segment starts, meaning that not all the momentum from the actuation is transported to the burst leading-edge vortex structure. However, some of it is, as the middle

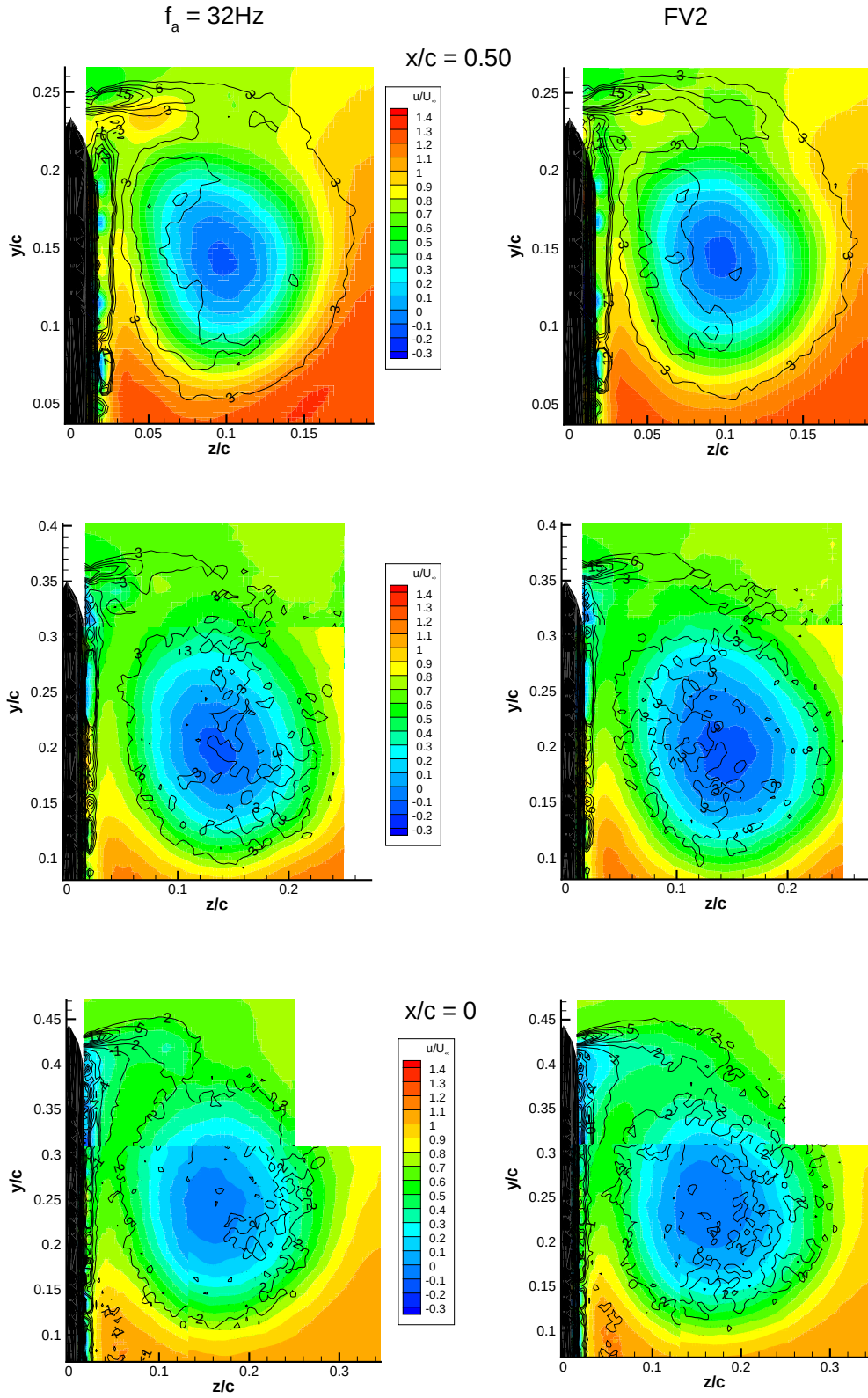


Figure 4.23: Selected planes for the actuated cases. $f_a = 32\text{Hz}$ on the left, FV2 on the right. Contour lines represent normalized vorticity $\Omega_x = \omega_x s / 2U_\infty$ every 3 units. $U_\infty = 12\text{m/s}$, $\alpha = 35\text{deg}$

range values ($TKE \approx 0.17$) are seen to exist in a bigger region than for the baseline.

Starting from $x/c = 0.50$, two clear structures can be identified. One is the rotational structure, which has TKE values and distribution very similar to the baseline case (higher TKE close to the wing surface distributed in an annular structure). The second one is the structure created by the actuation with very high TKE values. It would seem that the two structures do not mix with each other, as there exists a boundary of low TKE values in between. However, the rotational structure is pushed inwards by the actuation. This is backed up by the change in rotational axis position (figure 4.15), where the biggest spanwise (y/c direction) difference in between the baseline and actuated cases exists at $x/c = 0.60$. This hypothesis could as well be supported by the fact, that the structure created by the actuation becomes similar to the secondary vortex created by the vortex core (they both rotate against the main vorticity feeding direction), which pushes the vortex (here- the burst vortex structure) inwards and away from the wing. Part of such structure is seen for both actuations (figure 4.23, plane $x/c = 0.50$), where close to the leading-edge a higher axial velocity ($u/U_\infty \approx 1.1$) region exists.

The chordwise resolution of the measurements is too low to see if the region from plane $x/c = 0.50$ is actually the same as from $x/c = 0.55$. Hence, another hypothesis could be that the region with high TKE values is actually created from each blowing slot separately. In such case, the region in plane $x/c = 0.50$ would come from the second slot from the second segment, it would dissipate until $x/c = 0.55$ and the region seen in $x/c = 0.55$ would already come from the third blowing slot. A closer investigation should be made to get more details about the flow region just next to the blowing slot.

After the third actuation segment, the TKE distribution within the main rotational structure is very similar to the baseline. At planes $x/c = 0.80 - 0.90$ a high TKE region close to the leading-edge suggests that the structure, created by the third actuation segment spans from $x/c = 0.76$, where the 12th slot is located, at least $14cm$ downstream. This would be another argument supporting the hypothesis two paragraphs above, that the actuation structures actually last to overcome the distance in between two measurement planes.

Figure 4.24 presents the TKE distribution with the second actuation case, FV2 (actuation frequencies $61.5Hz$, $41Hz$ and $20.5Hz$).

For the apex region up to plane $x/c = 0.20$ both actuations result in same flow pattern. While comparing planes $x/c = 0.40$ it can be seen that changes coming from actuation FV2 results in a much more axis-symmetrical structure than the baseline or when actuating with constant frequency. The TKE is as well distributed within the structure more evenly and the peaks visible in constant frequency actuation case here are not that obvious. This is expected, as now the actuation momentum is injected not all at once, but split over some time. No gaps in between the actuation structures and the rotational structure can be seen, meaning that the actuation momentum gets transported within the flow faster than with single frequency

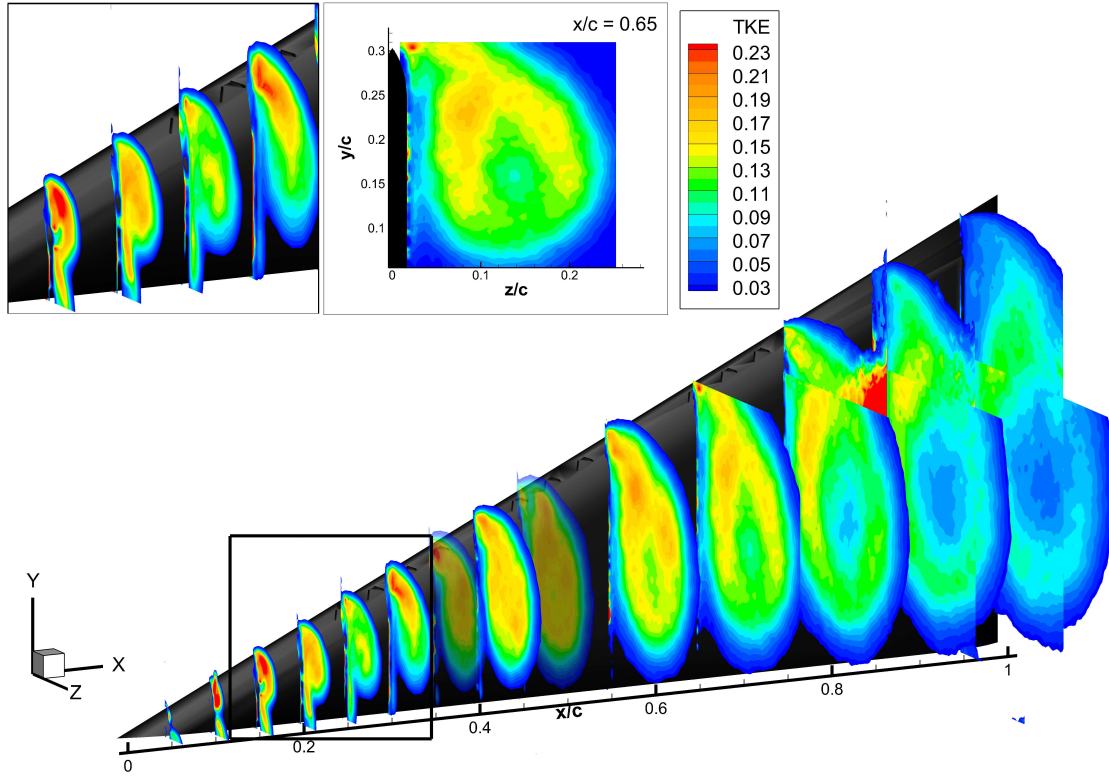


Figure 4.24: Turbulent kinetic energy distribution for the actuated case, FV2. The values below $TKE = 0.1$ are not shown here. $U_\infty = 12\text{m/s}$, $\alpha = 35\text{deg}$

actuation. This is as well supported by the fact that the actuation structure from the very last blowing slot can only be seen (although barely) until plane $x/c = 0.80$, whereas for constant frequency actuation it was seen even at $x/c = 0.90$.

It has been shown, that minor changes in lift coefficient occur with actuation at $\alpha = 35\text{deg}$, but a big increase in drag is seen (section 4.1.2). This difference comes from changes in axial force component. For both actuations the axial suction (“thrust”) decreases, but no changes in normal force are seen. No connection between these changes and the PIV measurements is found.

4.3.3 $\alpha = 45\text{deg}$

Actuation at $\alpha = 45\text{deg}$ angle of attack presents biggest transformation in the flow structure, which as well is resembled in the force measurements. Averaged axial velocity field for the constant frequency actuation is shown in figure 4.25.

The flow field is clearly different than it was for the baseline case, figure 4.16. With actuation, a burst vortex structure is recreated, more similar to the one seen in $\alpha = 35\text{deg}$ case. Almost perfect conical structure is formed, showing rotational motion. No signs of flow reattachment

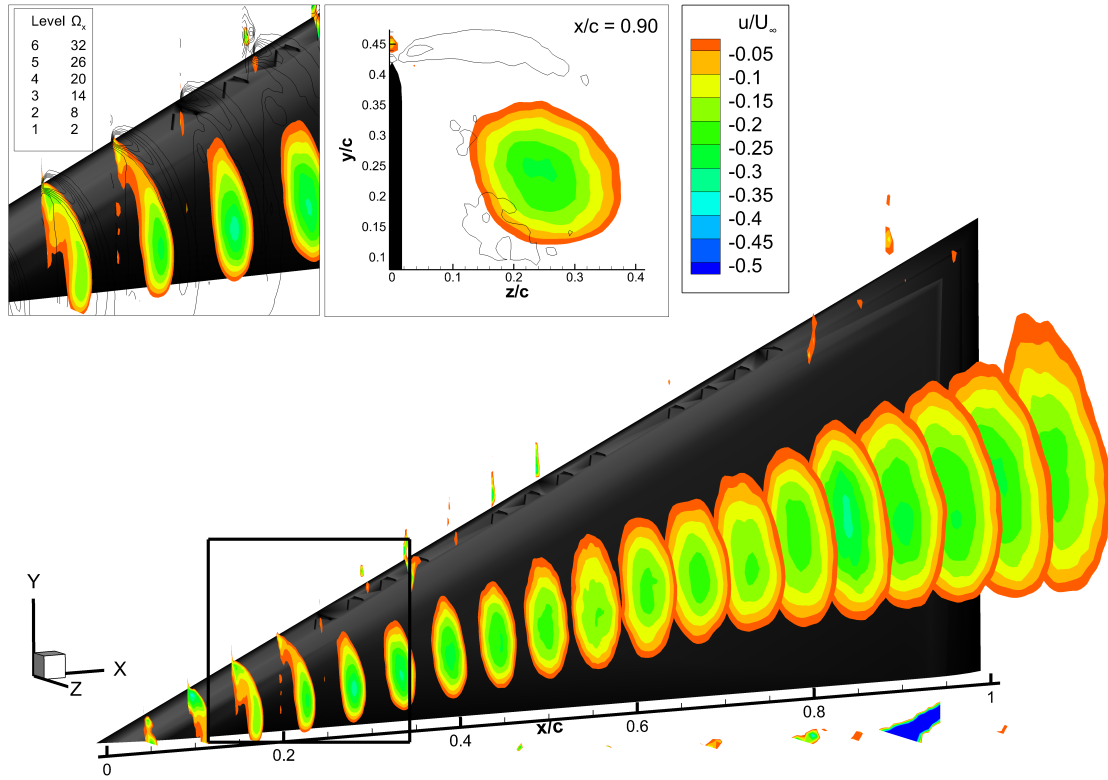


Figure 4.25: Averaged axial velocity field of the constant frequency actuation case. Only negative velocities are shown here. The contour lines in corner figures represent positive normalized vorticity lines. $U_\infty = 12\text{m/s}$, $\alpha = 45\text{deg}$

to the wind tunnel floor are seen. In the front region, the rotational structure clearly starts not from the apex of the wing, but from somewhere on the peniche, showing another disadvantage of using a half model only.

The actuation creates fluctuation structures similar to the $\alpha = 35\text{deg}$ case, with high TKE values just behind the blowing slots. However, now these structures are transported faster into the main structure and no gaps in between the two are seen. Also, no “trail” is seen to span from the very last blowing slot as it was in the previous case. This could be explained with the flow now having much higher w (normal to the wing) and much lower u (axial) velocity components, hence the actuation momentum is transferred in-plane rather than downstream.

No major differences in between the two actuations are seen. However, while comparing the TKE distribution, it can be spotted that FV2 distributes the energy in an annular region (minimum values at the center $TKE \approx 0.10$), whereas constant frequency actuation distributes the energy more evenly across the plane (minimum values at the center $TKE \approx 0.15$). Constant frequency actuation as well has axial vorticity Ω_x slightly higher than the FV2.

Another difference in between the two actuations is seen while comparing the axis of rotation

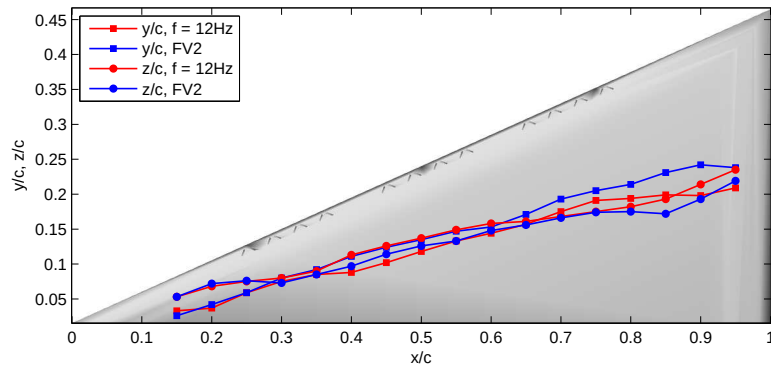


Figure 4.26: Axis of rotation position. y/c - spanwise, z/c - normal away from the wing.
 $U_{\infty} = 12m/s$, $\alpha = 45$ deg

position, figure 4.26. Up to plane $x/c = 0.30$ the position is nearly the same for both actuations, but further downstream the axis for FV2 is seen to move closer to the wing surface and to the leading-edge as well. This means that the constant frequency actuation creates rotational structure stronger than the FV2, as velocity component v at the wing surface for constant frequency is slightly higher.

Core axial velocity profiles comparison

Axial velocity comparison across the rotational structure core between the two actuation cases is done.

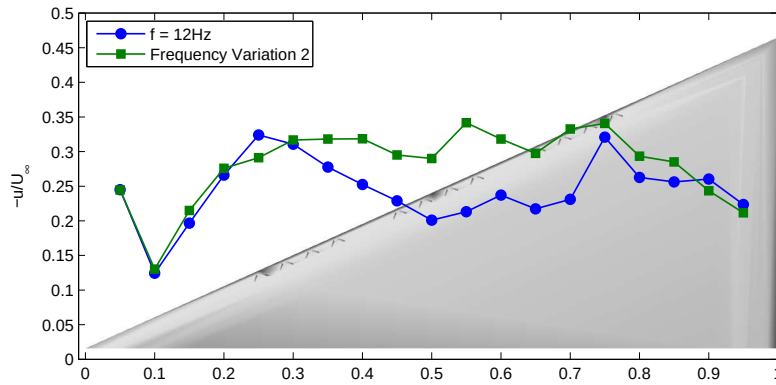


Figure 4.27: Rotational structure core normalized axial velocity local minimums of the two actuated cases. $U_{\infty} = 12m/s$, $\alpha = 45$ deg

As before, the apex region (up to $x/c = 0.20$) is almost identical for both actuations. However, further downstream, FV2 has core velocity values lower than the constant frequency actuation. Interestingly, a correlation between the velocity peaks and the actuation slot locations can be seen in figure 4.27, where local axial velocity minimums are shown. Where

the blowing segments are located, there exists a local axial velocity minimum, with a higher velocity region following it. For constant frequency actuation these peaks are located at $x/c = 0.25, 0.60, 0.75$ and for the FV2 at $x/c = 0.30, 0.55, 0.75$. The centers of the blowing segments are at $x/c = 0.30, 0.50, 0.70$. This spatial influence coming from the actuation could again be explained by the fact, that with higher angles of attack the transportation of actuation momentum happens more in-plane rather than out-of-plane (downstream). Therefore, the zones that are affected the most are closer to the actuation segments than it was for the other angles of attack.

4.4 Error analysis

Care was taken to decrease possible errors during the measurements. However, some were unavoidable. Only systematic errors are analyzed here, and all are split into two types- setup errors, and methodological errors. Methodological are the ones that could have been avoided by changing the steps of how a measurement is taken, whereas the setup errors mostly arise because of the current availability of equipment.

- Methodological errors
 - For vortex axis measurements, the vortex axis position was set based on measurements from previous experiments, as they were measured before the cross-planes. However, if the cross-planes were to be measured first, the vortex axis could be identified for exactly the same setup.
 - For $\alpha = 35$ deg case, planes $x/c = 0.75 - 0.95$ were stitched from multiple measurements, hence clear vertical and horizontal lines can be seen (figure 4.13). It is apparent, that measurements were done at somewhat different velocities. A better option would have been to use 50mm lenses.
 - For 50mm lenses ($\alpha = 45$ deg case, planes $x/c = 0.55 - 0.95$), the reflected light sheet increases the light intensity in some parts of the frame, but not near the leading-edge where the shear layer is shed, hence part of the measurement area was lost. This could be fixed with moving the laser closer to the wing (in $-z$ direction) or by fine-tuning the PIV post-processing parameters.
- Setup errors
 - Low correlation values for some of the PIV measurements were seen. At higher velocities, some vibrations from the downstream camera were present and as well the calibration was never perfect, as the camera focus had to be adjusted by hand.
 - Use of the half model presented some differences, most of which are stated before.

- For force measurements, some error was made in determining the freestream dynamic pressure, as for higher velocities/angles-of-attack the dynamic pressure values read from Betz manometer were unsteady. Electronic dynamic pressure measurement was unavailable at the time.

4.5 Summary

Force and particle image velocimetry measurements were done on a half delta wing model with unsteady leading-edge blowing. The PIV measurements were done at three angles of attack: $\alpha = 23$ deg, 35 deg and 45 deg.

For the baseline, good agreement in between force coefficients for low angles of attack ($\alpha \leq 20$ deg) from other institutions is found. The maximum lift coefficient is found to be $C_{L_{max}} = 1.129$ at $\alpha = 33$ deg, $U_\infty = 24$ m/s ($Re_{mac} = 1 \times 10^6$) and $C_{L_{max}} = 1.180$ at $\alpha = 35$ deg, $U_\infty = 12$ m/s ($Re_{mac} = 5 \times 10^5$). From PIV measurements, the vortex breakdown is found to happen around the mid-point of the wing ($x/c = 0.55$) at $\alpha = 23$ deg and the vortex bursts completely at $\alpha = 35$ deg. No rotational structure is seen at $\alpha = 45$ deg, but a part of flow is found to reattach to the wind tunnel floor.

With actuation, best improvement is seen at post-stall angles of attack. With constant frequency actuation of $f_a = 12$ Hz the fall of C_L is more damped in comparison to the baseline. At $\alpha = 42$ deg, where the gain coming from the actuation is the biggest, $C_L = 0.7$ for the baseline and the gain is $\Delta C_L = +0.315$.

From PIV measurements it is found that for $\alpha = 23$ deg case the breakdown location is slightly moved downstream by $\Delta x/c = 0.04$ – 0.05 . For $\alpha = 35$ deg case the structures appearing from the actuation can be recognized as regions of high turbulent kinetic energy in cross-planes. They were seen to last through a few planes and the effect from the last blowing slot spans at least 14 cm downstream. The created structures as well push the main burst vortex structure inward. These observations lead to a conclusion that the momentum from the actuation does not transport into the main rotational structure instantly and actually is left outside. More transportation is seen from FV2.

With actuation at $\alpha = 45$ deg, a complete reformation of the flow is seen. A conical structure similar to the burst vortex is created. Much more transportation of the actuation momentum is seen, and now more of it happens in-plane. Therefore, spatial correlation in between the flow velocities inside the recreated structure and the blowing slot segments exists. Constant frequency actuation distributes the *TKE* almost evenly throughout the structure, whereas for FV2 the distribution is more concentrated within an annular region around the edges of the structure. A stronger rotational structure is created by the constant frequency actuation and its axis is seen to move slightly inwards and further away from the wing.

Chapter 5

Conclusions and suggestions

Effects of unsteady leading-edge blowing on a delta wing model were investigated in this report. Two types of unsteady actuation were considered- constant frequency for all 12 blowing slots and spatially varying frequency configurations. Theory behind the delta wing flows and phenomena like leading-edge vortex, vortex breakdown and helical mode instability was presented.

In the experimental part of the thesis, methodology behind selection of specific frequencies for actuation was explained. Equipment setup for PIV and force measurements was presented.

Force measurements at relevant angle of attack ranges and particle image velocimetry measurements at three angles of attack ($\alpha = 23$ deg, 35 deg and 45 deg) were conducted. Review of systematic errors in the measurements was included. The main conclusions from the results were:

- Good agreement between full and half models for low angles of attack ($\alpha \leq 20$ deg) is found.
- Biggest effect from the actuation is seen at post-stall angles of attack.
- Biggest gain comes with $f_a = 12\text{Hz}$ actuation at $\alpha = 42$ deg, $\Delta C_L = +0.315$ (baseline $C_L = 0.7$).
- Complete vortex breakdown happens at $\alpha = 35$ deg.
- Separate structures are created by the constant frequency actuation at $\alpha = 35$ deg, that do not completely dissipate within the burst vortex structure and can sometimes last for more than 14cm downstream.

- At $\alpha = 35$ deg, spatially varying actuation frequency configuration (FV2) momentum dissipates more than the momentum coming from constant frequency actuation.
- For the baseline, part of flow is found to reattach to the wind tunnel floor at $\alpha = 45$ deg.
- Major flow changes occur at $\alpha = 45$ deg with the actuation on. Conical structure, similar to a burst vortex, is created by the actuation.
- Spatial correlation between the flow velocities inside the structure and the blowing slot segments exists.
- Constant frequency actuation momentum is dissipated more than spatially varying actuation frequency configuration (FV2) momentum.

A few suggestions for reducing systematic errors away were already made in section 4.4. Other suggestions would be:

- Do time-accurate PIV measurements, especially for the $\alpha = 35$ deg case, since most of the flow is dominated by unsteady phenomena and structures that get lost in the process of averaging.
- Tune the post-processing parameters and try to get more details of the flow for areas where small spatial resolution was an issue. Such areas would include separate frames with blowing jets visible and areas near the leading-edge for $\alpha = 45$ deg.
- As hysteresis is known to exist in delta wing flows, try out unsteady actuation strategies that would vary over time, i.e. one frequency actuation that would change into a different frequency actuation after some time.

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