Control of Leading Edge Vortices using Apex Flap over

Non-Slender Delta Wing

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Abstract

This thesis presents the experimental investigation of vortex flow structure over nonslender delta wing with leading edge sweep angle, Λ =45°. A comprehensive investigation has been conducted in a wind tunnel at Reynolds numbers ranging between 247,000 and 445,000. Seven-hole pressure probe measurements for flow characteristics such as normalized axial vorticity, normalized axial velocity, vortex trajectory and pressure variations are presented at various chordwise stations, angles of incidences and Reynolds number. The focus was on critical vortex flow characteristics and their dependency on the angle of attack, stream-wise movement and Reynolds number. The movement of vortex breakdown was also documented for both baseline model and modified apex flap model over the delta wings. In addition the effects of passive deflections of the apex flap over non-slender delta wings were investigated. The changes produced by the apex flap on the flow characteristics and vortex trajectory is also reported. The effects of apex flap over vortex breakdown progression and its control are also documented. Vortex flow evolution in the wake downstream of trailing edge is also discussed. It is demonstrated that weak leading edge vortices are generated in the proximity of wing surface with strong shear layer which move upward and outboard with apex flap deflection. It is recognized that vortex breakdown was delayed by 8% by downward apex flap deflection.

Résumé

Cette thèse présente l'étude expérimentale de la structure de l'écoulement tourbillonnaire sur aile delta non-mince avec les principaux angle de balayage de pointe, $\Lambda = 45^{\circ}$. Une enquête approfondie a été menée en soufflerie au nombre de Reynolds allant de 247,000 -445,000. Les mesures de la sonde pression sept trous pour les caractéristiques d'écoulement comme tourbillon normalisée axiale, vitesse axiale normalisée, la trajectoire de tourbillons et les variations de pression sont présentés à différents postes de sens de la corde, des angles d'incidences et le nombre de Reynolds. L'accent a été maintenu sur les caractéristiques de propagation des tourbillons critiques et leur dépendance à l'angle d'attaque, le mouvement streamwise et nombre de Reynolds. Le mouvement de rupture du vortex a également été documenté à la fois pour le modèle de base et modèle modifié du volet sommet sur les ailes delta. En plus de cela, l'objectif était d'étudier les effets réalisés par des détournements passifs de rabat sommet sur les ailes delta non-minces. Les changements produits par rabat sommet sur les caractéristiques de débit et de la trajectoire de vortex sont également signalés. Les effets de volet sommet sur la progression de la rupture du vortex et son contrôle sont également documentées. L'évolution des flux de vortex dans le sillage en aval du bord de fuite est également parlé dans une certaine mesure. Il est démontré que la faiblesse des grands tourbillons de bord sont générés à proximité de la surface de l'aile avec couche de fort cisaillement qui se déplacent vers le haut et à l'extérieur avec déviation du volet sommet. Il est reconnu que la rupture du vortex a été retardé de 8% en baisse déviation du volet sommet.

Table of Contents

A	ckno	owle	dgements	ii
Al	ostra	act		iii
Re	ésun	né		iv
Та	ble	of C	contents	v
Тε	ble	of F	igures	vii
No	ome	encla	ture	ix
1	Ir	ntrod	luction	1
	1.1	E	Ingineering Perspective and Background	1
	1.2	C	Dbjective	2
2	А	eroc	lynamics of Delta Wings	4
	2.1	L	iterature Survey	4
	2.	.1.1	Leading Edge Vortex	4
	2.1.2		Vortex Breakdown	13
	2.	.1.3	Aerodynamic Characteristics	16
3	E	xper	imental Procedures and Apparatus	20
	3.1	E	Experimental Flow Facility	20
	3.2	I	nstrumentation and Data Processing	21
	3.	.2.1	Seven-hole Pressure Probe	22
	3.3	Γ	Data Acquisition and Reduction	23
	3.4	Γ	Delta Wing Models	25
	3.4.1		Wing Models and Specifications	25
	3.	.4.2	Experimental Method	27
4	R	lesul	ts and Discussion	29
	4.1	E	aseline model	

	4.1.1	Variation of vortex characteristics with chordwise stations	30
	4.1.2	Variation of vortex characteristics with angles of attack	37
	4.1.3	Variation of vortex characteristics with Reynolds number	40
	4.1.4	Comparison of baseline model with apex flap model	40
	4.1.5	Variation of vortex characteristics with apex flap	45
	4.1.6	Wake vortex progression	49
	4.1.7	Vortex breakdown	52
5	Conclus	ions and Future work	55
	5.1 Coi	ntributions of present thesis	55
	5.2 Coi	nclusions	55
	5.2.1	Baseline model	55
	5.2.2	Apex flap model	56
	5.3 Fut	ure work	57
6	Referen	ces	58
Aj	ppendix A	: Seven-hole pressure probe calibration technique	62

Table of Figures

Figure 1: Effect of leading edge shape on vortex breakdown location7
Figure 2: Variation of normalized axial velocity with angle of attack
Figure 3: Variation of normalized tangential velocity with angle of attack9
Figure 4: Normalized axial vorticity distribution at various chord-wise stations11
Figure 5: radial distribution of normalized circulation12
Figure 6: Variation of vortex breakdown with angle of attack over slender delta wing14
Figure 7: Lift coefficient variation with angle of attack
Figure 8: Breakdown of aerodynamic drag [2]19
Figure 10: Schematic of wind tunnel [1, 2, 59]20
Figure 11: Wind tunnel (a) inlet (left top) (b) outlet (c) test section (d) acoustic silencer [2]
Figure 12: Seven-hole pressure probe [1, 2]22
Figure 13: Flow chart of seven-hole apparatus setup23
Figure 14: Geometrical drawing of experimental model26
Figure 15: Experimental model, measuring sensor and traverse system
Figure 16: Working scheme of present experimental study27
Figure 17: Chordwise distribution of normalized axial vorticity contours at Re = $371,000$ and $\alpha = 6^{\circ}$
Figure 18: Chordwise distribution of normalized axial velocity contours at Re = $371,000$ and $\alpha = 6^{\circ}$
Figure 19: Chordwise distribution of normalized axial vorticity contours at Re = 247,000 and $\alpha = 10^{\circ}$
Figure 20: Chordwise distribution of normalized axial velocity contours at Re = 247,000 and $\alpha = 10^{\circ}$
Figure 21: Variation of flow characteristics with chordwise stations (a, c and e) before vortex breakdown (b, d and f) after vortex breakdown at $\alpha = 10^{\circ}$ and Re = 247,000

Figure 22: Variation of flow characteristics with angle of attack (a, c and e) before vortex breakdown (b, d and f) after vortex breakdown at $\alpha = 10^{\circ}$ and Re = 247,00038
Figure 23: Absence of dual vortex structure (normalized axial vorticity) at $\alpha = 2^{\circ}$ and $\alpha = 6^{\circ}$
Figure 24: Normalized axial vorticity contours at $x/c = 0.4$, $\alpha = 10^{\circ}$ (a-c) Reynolds number effect (d-h) Apex flap effect
Figure 25: Normalized axial velocity contours at $x/c = 0.4$, $\alpha = 10^{\circ}$ (a-c) Reynolds number effect (d-h) Apex flap effect
Figure 26: Reynolds number effect on different flow characteristics at $\alpha = 10^{\circ}$ 43
Figure 27: Comparison of baseline and apex flap model at $\alpha = 10^{\circ}$ and Re = 247,00044
Figure 28: Effect of negative apex flap deflections on flow characteristics at Re = 247,000 and $\alpha = 10^{\circ}$
Figure 29: Effect of positive apex flap deflections on flow characteristics at $Re = 247,000$ and $\alpha = 10^{\circ}$
Figure 30: Normalized axial vorticity contour at $x/c = 1.05$, Re = 371,000 and $\alpha = 4^{\circ}$ 50
Figure 31: Normalized axial velocity contour at $x/c = 1.05$, Re = 371,000 and $\alpha = 4^{\circ}$ 50
Figure 32: Normalized axial vorticity contours at $x/c = 1.05$, Re = 247,000 and $\alpha = 10^{\circ} \dots 51$
Figure 33: Variation of vortex breakdown location with angles of attack

Nomenclature

AR	Aspect ratio, b^2/s
b	wing Span
c	wing chord
LEV	leading-edge vortex
Р	total pressure
р	static pressure
q	dynamic pressure, $\frac{1}{2} \rho U_{\infty}^{2} S$
Re	Reynolds number, $\rho c U_{\infty}/\mu$
S	wing planform area
S	wing semi span, b/2
t/c	thickness to chord ratio
u, v, w	streamwise, traverse and spanwise velocity component
VBD	vortex breakdown
X	streamwise direction
У	traverse direction
Z	spanwise direction
Г	vortex circulation
α	angle of attack
β	apex flap deflection
Δ	grid resolution
Λ	leading-edge sweep angle

Subscripts

c	core vortex value
0	outer vortex value
max	maximum value
00	free stream or ambient value

Chapter 1

1 Introduction

1.1 Engineering Perspective and Background

Over the past several decades, extensive research has been conducted over delta wings. Delta wings are used in high speed military and commercial aircrafts. Most of the modern aircrafts with delta wings have to go through various unsteady motions especially during takeoff and landing. Delta wings satisfy all requirements of military aircrafts such as high speed, super-maneuverability, control and power [3]. Delta wings achieve this due to their ability to stall at higher angle of attack and produce more lift, which is necessary during landing, takeoff and combat maneuvering [3].

Future aircrafts may be required to operate at angles of attack beyond static stall in order to increase their combat effectiveness (Herbst 1983). Therefore, the understanding of the complex flow developed over delta wings become important. Extensive research work has been carried out over both steady and unsteady delta wings but the emphasis was mainly on the slender delta wings. Different types of geometrical modifications to delta wings and various measuring techniques including both intrusive and non-intrusive have been used for better understanding.

Delta wings are broadly classified in to two main groups depending upon their sweep angle (Λ), slender delta wings or high sweep delta wings ($\Lambda > 55^{\circ}$) and non-slender delta wings also called low sweep delta wings ($\Lambda < 55^{\circ}$). Highly swept delta wings are found more useful at supersonic speeds whereas more recently non-slender delta wings gain popularity for their use in low Reynolds number applications such as unmanned air vehicles (UAV) and micro air vehicles (MAV). This has renewed the researchers' interest to understand the flow topology and vortical structure over low sweep delta wings. Delta wings differentiate themselves from others due to the unique vortical flow structure formed over the leeward side of wing. The vortical structure allows them to generate suction pressure on top side even at higher angles of attack causing a delay in stall angle [3-5]. In addition to these advantages delta wings have some drawbacks; massive flow separation over the leeward

side increases the drag and reduces the effectiveness of control devices [3, 5-7]. The problem becomes grave for non-slender delta wings which have earlier vortex breakdown and lower stall angles. Increase in drag will yield in lower L/D ratio which will lead to lower performance and longer takeoff time [2].

Leading edge vortices generated over the delta wings are complex in nature due to interaction of various vortices over leeward side and therefore need to understand in detail. This complexity gains more gravity when leading edge vortices interact with boundary layer over non-slender delta wings [8-10]. The transient nature of flow even on static delta wings with various geometrical factors and measuring techniques bound the researchers to work at specific locations and angles of incidences depending upon their requirements. The aerodynamic forces created by such complicated flows are nonlinearly related to the instantaneous angles of attack, sideslip and roll angle, as well as their rates of change and furthermore are likely to depend on the history of these quantities [3]. This requires extensive insight into vortical flow behavior and aerodynamics of non-slender delta wings and to apply a controlling mechanism to delay the vortex breakdown and delay the stall angle.

1.2 Objective

The primary objective of this study was to investigate and control leading edge vortices. This involves the provision of a standard set of accurate aerodynamic data on static and apex flap models undergoing a range of passive apex flap deflections for a range of angles of attack. To obtain this prime objective, there were sub tasks serving the purpose of milestones. From designing the wing support structure to finalizing and machining both experimental models, from extensive literature review to choosing various flow conditions and from calibrating the pressure probe to actually collecting and post-processing the data. Seven-hole pressure probe measurements for axial vorticity, axial velocity, vortex trajectory and pressure variations were acquired and presented at various chordwise stations and angles of incidences for different passive apex flap deflections. This set of aerodynamic data collected both in pre and post breakdown regions was then used to locate the vortex breakdown unlike the conventional methods which provide only qualitative results. Extensive wake survey analyses were performed downstream of trailing edge to examine the progression of leading edge vortices behind the trailing edge of both static and apex flap

models. This also allowed understanding the effect of apex flap on the leading edge vortex trajectory and shape.

Chapter 2

2 Aerodynamics of Delta Wings

2.1 Literature Survey

As mentioned earlier, there has been much work performed on the vortical flow over both static and unsteady slender delta wing. On the other hand, less work has been done over low sweep delta wings. In the following sections a brief review of the steady vortex flow studies will be presented.

2.1.1Leading Edge Vortex

The leading edge vortices generated by slender delta wings have been the subject of numerous experimental studies staring in the mid-1950s and continuing until today. The flow structure on the upper side of a delta wing at angle of attack is extremely complex and in many cases remains to explored [3]. It is necessary to comprehensively describe the structure of leading edge vortices before unfolding the aerodynamic characteristics of delta wings.

Flow passing over delta wing attaches itself to the lower side of the delta wing and starts moving towards the leading edge due to pressure variation on both sides. Flow remains attached to both upper (leeward) and lower (windward) sides at low angles of attack and hence potential lift is produced like the conventional wings. Flow separates at the leading edges and eventually rolls up to provide free shear layers in the form of two counter rotating vortices at moderate angles of attack. The primary reason for the flow separation at the leading edge is that the flow is unable to adjust to the sharp edge. The free shear layer moves inward and up to form a strong and coherent vortex. The counter rotating vortices originating from the apex and subsequently getting fed from the leading edges are formed over the leeward side [11]. Similar results have also been observed for slender delta wings where boundary layer separation is often fixed at apex by sharp leading edge which results in formation of three dimensional shear layers [12].

The surface flow on the upper surface is directed outward due to the vortex, and because of the large pressure gradient which exists between the suction peak and leading edge, the flow separates to form a small secondary vortex [13]. The secondary vortex pushes the inward and upwards. At the leading edge the flow mixes with the free shear layer. It is possible that there are additional vortices near the surface in this region [3]. A sketch of leading edge vortices representing both primary and secondary vortices. Leading edge vortex over delta wings can be divided in three distinct regions, the viscous subcore, rotational core and shear layer. The viscous subcore which is approximately 5% of local semi span in diameter is a region in which the gradients of local head, static pressure and velocity are very high. The axial velocities in this region are reported to be three times the free stream velocities [14-16]. The rotational core is approximately 30% of the local semi span in diameter, wherein the traces of the vortex sheet produce only minor perturbations on the circumferential and longitudinal velocity distribution. The shear layer, or vortex sheet, is generated at the wing's leading edge and feeds vorticity into the vortex core [2, 3, 17]. Both size and strength are directly linked with the angle of attack until the vortex breakdown occurs which will be discussed later.

At low angles of attack the flow reattaches itself to the wing upper surface. At moderate values of angles of attack, the roll up vortex reattaches itself to the wing surface which can be observed using oil flow pattern which creates a reattachment line in the streamwise direction over leeward side [18]. This creates a zone of attached flow under the primary vortex and moving the shear layers towards the lower pressure region near the leading edge. The spanwise progression of attached shear layer is then hindered by an adverse pressure gradient near the leading edge and thereby caused the secondary flow separation. The adverse gradient rolls up this separated shear layer in an opposite direction of that of primary vortex to form secondary vortex [2]. Theoretically, this process can continue to a more detailed stage producing a tertiary vortex.

2.1.1.1 Factors affecting Leading Edge Vortex

There are various parameters which affect the flow and the formation of leading edge vortices which are discussed as under:

2.1.1.1.1 Angle of Attack

This is most critical factor amongst the rest of the factors which has been studied extensively for both static and unsteady cases. Increasing angle of attack is equivalent to increasing adverse pressure gradient in the streamwise direction [18]. Leading edge vortex grows in size and strength with increasing angle of attack till the vortex bursts, this phenomenon is referred as vortex breakdown. This burst first occurs in the wake downstream of trailing edge and moves upstream with the increase in angle of attack. Further increase in incidence moves the breakdown location passed the trailing edge of delta wing causing a sharp increase in normal velocity fluctuations producing a destabilizing pitching moment [18]. At some critical value of angle of attack the breakdown reaches apex and the wing stalls.

2.1.1.1.2 Aspect Ratio

Increasing aspect ratio or lowering wing sweep means moving attachment line, peak suction pressure and secondary separation line further outboard. With the increase in sweep angle the breakdown becomes increasingly unsteady and crosses the trailing edge at higher angles of attack [3, 5, 15, 18, 19]. Vortex breakdown is a transient phenomenon and is found [20] to fluctuating by as much as 50% of root chord over 85° sweep delta wing.

2.1.1.1.3 Reynolds Number Effect

Effect of Reynolds Number has been studied significantly and found independent for slender delta wings whereas the effect was found significant over low sweep delta wings.[18, 19, 21-26] The turbulent boundary layer on the leeward side of the wing moves the vortices outward resulting in higher pressure peaks than in the laminar case. The secondary vortex becomes smaller and closer to leading edge vortex with much smaller pressure peak than laminar case. Therefore, the overall leading edge delta wings are nearly independent of Reynolds number [18]. No consistent effect due to Reynolds number was detected in the steady breakdown location over the range of 150,000 - 450,000 [3, 27].

2.1.1.1.4 Thickness Effect

Thickness can significantly affect the pressure distribution over delta wings. It has been found that root thickness to chord $\binom{t}{c_r}$ of 12% causes a 5% mean chord rearward shift in the centre of pressure compared to the thin (< 2%) delta wings [3, 18].

2.1.1.1.5 Leading Edge and Trailing Edge Shape Effects

Increasing the convexity of leading edge planform shape has the effect of moving the attachment line, peak section pressure and secondary separation line further outboard [18]. The effect of leading edge geometry on vortex breakdown is shown in figure 4. It also has the effect of moving the cores farther away from the wing. The round and elliptically shaped leading edges tend to have attached flow to significantly higher angles of attack, which makes the flow characteristics Reynolds number dependent.





2.1.1.2 Flow Features of Leading Edge Vortices

2.1.1.2.1 Velocity

The velocity field has a vital role in the characterization of leading edge vortices and has been studied extensively. Both intrusive (pressure probes, hot wire and Particle image velocimetry) and non-intrusive (flow visualization and laser doppler anemometry etc.) methods have been used to explain the velocity flow field before and after vortex breakdown [2, 3]. Figure 5 shows both axial and tangential velocity fields [3, 28]. Hall [29, 30] has worked on the velocity field and presented a theoretical model which had some assumptions of low being continuous and rotational and the viscous diffusion is confined to

a relatively slender subcore [2]. His model was in good agreement with the experimental results in the pre breakdown region. Leading edge vortices are defined to be as swirling jet flows as can be seen in the figures 2 and 3 where velocity measurements have been shown against angles of attack. Similar results have been presented for non-slender delta wings in the present study which will be shown later. The size of the leading edge vortices, the axial velocity, the tangential velocity and the suction pressure magnitudes was in direct relationship with angles of attack [2]. This trend continued till the vortex breakdown reached and the velocity magnitudes fall appreciably. The peak axial velocity to free stream velocity values $\left(\frac{u}{U_{\infty}}\right)$, are reported [2, 3, 31] to be 3 times for slender delta wings. For low sweep delta wings, it has been reported in the range of 1.2-1.3 [32, 33].



Figure 2: Variation of normalized axial velocity with angle of attack

Erickson and R.C. Nelson [3, 13] explained the axial velocity distribution within leading edge vortices and further elaborated to be spiraling in nature with downstream convection. Due to downstream progression of the flow these vortex lines are inclined to the vortex axis

and hence have a streamwise component supplementing the axial flow [2]. Nelson and Visser [34, 35] used cross wire to obtain velocity field distribution for sweep angle, $\Lambda = 75^{\circ}$ and angle of incidence, $\alpha = 20^{\circ}$. An important feature is the relative size of jet cone vs. the core associated with maximum normalized tangential velocity $\begin{pmatrix} V_{\theta} \\ U_{\infty} \end{pmatrix}$, also termed

as peak-to-peak distance. The jet cone is reported to be approximately 50% of wing semi span (b), whereas core based peak-to-peak of V_{θ} is 5-10% of local semi span. Jet cone distance changes with angle of attack while the peak-to-peak distance remains constant and are in good agreement with Earnshaw [17] proposed results [3]. Erickson [13] proposed that the majority of the flow field phenomena observed over delta wing is dominated by potential flow effects associated with external field, that is external pressure gradient.

Non-slender delta wings got focused after their ability to be use in unmanned air vehicles (UAV) and micro air vehicles (MAV). Researchers [4-7, 32, 36, 37] showed their interest on low sweep delta wings recently and employed different techniques to solve the mystery related to these wings.



Figure 3: Variation of normalized tangential velocity with angle of attack

2.1.1.2.2 Vorticity

Vorticity field provide additional insight to the leading edge vortices structures. Various investigators have put forward different theories associated with vorticity distribution and

adverse $\left(\frac{dP}{dx} > 0\right)$ or favorable $\left(\frac{dP}{dx} < 0\right)$ pressure gradient about vortex sub structure and vortex breakdown. Lee and Ho [38] stated that "a stationary leading edge vortex is achieved only when the convection of vorticity along the core axis balances the vorticity generation

from the boundary layer of the leading edge" and the swirl angle $\left(\phi = \tan^{-1} \frac{V_{\theta}}{U_{\infty}}\right)$ is an

indicator of the balance [3].

Payne, Visser and Nelson [15, 35] postulated that vortex breakdown is caused due to reduction in axial convection of vorticity and the main source is adverse pressure gradient. This gives importance to the critical value of vorticity responsible for vortex breakdown and vorticity distribution across the vortex cross section over both types of delta wings. Pagan and Solignac [28] and Delery et al. [39] have studies the effects of adverse pressure gradient oven slender delta wing with sweep angle, Λ =70° at angle of attack, α =27.5°. It was concluded that swirl angle and adverse pressure gradient are the two main sources of promoting vortex breakdown. Nelson and Pelletier [3] interpreted that "the maximum amount of vorticity or circulation at a given station is limited by the ability of the flow to move downstream, which in turn is regulated by the pressure gradient".

Visser and Nelson [35] used a cross wire anemometer to measure all three axial, radial and azimuthal components and suggested that axial vorticity distribution confined within 20% of the semispan on either side over 75° sweep delta wing. The magnitude drops to less than 10% of the peak value of vorticity within $\pm 10\%$ of local semi span. This holds true for slender delta wings but not much information is available for non-slender delta wings. It was concluded [35] that "Employing the maximum value of axial vorticity in determining the local strength or state of the vortex structure is deceptive. Grid-resolution dependence and the locally steep gradients deter this type of quantification. The majority of pre breakdown positive axial vorticity is concentrated about the vortex axis in a region approximately twice the diameter of the subcore". Figure 4 shows the normalized axial vorticity distribution at various chordwise stations. Mitchell and Molton [31] presented vorticity contours over slender delta wings. Vorticity contours were found coherent and

concentrated in centre and peak non-dimensionalized value of vorticity $\left(\frac{\zeta_z c}{U_{\infty}} > 200\right)$ was

reported in pre vortex breakdown region. The value dropped to around 140 just after vortex breakdown and eventually fell to 80 at 0.21c downstream of vortex breakdown. At that location the vortex concentration was lost and only a diffused vortex was left.



Figure 4: Normalized axial vorticity distribution at various chord-wise stations

Non-slender delta wings have their own significance that leading edge vortices are formed at much smaller angles of attack. Since the leading edge vortices are closer to the surface and more complicated, therefore, the interaction with boundary layer is predominant [9, 38, 40, 41]. As a result of this the vortex breakdown occurs earlier and the vorticity cannot obtain higher magnitudes. This can be seen in figure 7 [7] where lower magnitudes have been reported compared to the slender delta wings. Ol and Gharib [42], Gursul, Gordnier and Visbal [7] performed their experiments over delta wing with 50° sweep angle at low Reynolds number.

The diffusion of the primary vortex peak was found at lower angles of attack suggesting earlier vortex breakdown. The results thus obtained further strengthened the theoretical knowledge of vorticity balance between its production and downstream convection. This information is revealed by observing both signs of vorticity in the post breakdown region. The information thus collected ruled out the need of vorticity sinking downstream by virtue of counter rotating vortex structures. The detailed vortex structures over non-slender delta wings are shown in figure 7 indicating the dual vortex structures.

2.1.1.2.3 Circulation

Circulation confined to leading edge vortex is important not only for calculating strength but, also aerodynamic loads, such as $\lim_{c} \Gamma = \int_{c} V dr = \int_{A} (\nabla x V) dA = \int_{A} \zeta x dA$ [3]. The

spanwise distribution of circulation is plotted against non-dimensional radial distance starting from vortex centre as shown in figure 5. Visser and Nelson [35] mentioned that "the circulation profiles exhibited a logarithmic dependence over a given radial distance from vortex axis pointing to a possibly substantial influence of Reynolds stress terms in the flow field".



Figure 5: radial distribution of normalized circulation

2.1.2Vortex Breakdown

The strength of leading edge vortices increases with downstream distance and angle of attack until some critical value has been reached. The vortex breakdown stops as this limiting condition has reached. The post breakdown flow can be characterized by massive dilatation of the vortex structure, a profound alteration of the velocity field along with large scale fluctuations [39]. In process the primary vortex loses its coherence and rapid exchange of momentum results in large scale turbulence [43]. The detrimental phenomenon of vortex breakdown is typically characterized by an increase in vortex diameter whereas the non-linear vortex lift is a strong function of vortex breakdown moves upstream of trailing edge. This also causes the decrease in the lift curve and simultaneously increasing the drag and decreasing lift to drag L/D value and downgrading the wing performance. The onset of vortex breakdown plays important role in limiting the high value of lift and high angle of attack of delta wing performance [2]. Once the vortex breakdown reaches apex, the delta wing stalls resulting in complete separation of flow over leeward side.

2.1.2.1 Vortex Breakdown Dependence

People [29, 30, 39, 44-46] have postulated different competing explanations and counter proposals related to vortex breakdown and factors upon which it depends. These explanations can be divided into following main categories which include the phenomenon which is like the separation of a two-dimensional boundary layer [18, 47], the phenomenon is a consequence of hydrodynamic instability [18], this phenomenon depends in an essential way on the existence of a critical state [18, 48, 49] and the breakdown is a wave propagation phenomenon [18, 45].

Hall [30] has proposed an analytical model for vortex breakdown and postulated that two

main factors such as 1) Adverse pressure gradient
$$\frac{dP}{dx} > 0$$
, 2) Swirl angle $\phi = \tan^{-1} \frac{V_{\theta}}{U_{\infty}}$. All

the above mentioned theories emphasized the vortex breakdown to occur within observed range of swirl angle and the dependency of vortex breakdown over adverse pressure gradient. In addition to these factors other factors like angles of attack, apex, leading edge shape, trailing edge shape, thickness effects, Reynolds number (low Reynolds number on non-slender delta wings), aspect ratio or sweep angle and wind tunnel geometry with wall effects also play important role in defining the vortex breakdown location. The combination of all the above mentioned factors makes it a transient phenomenon and therefore the exact location of vortex breakdown always has some question marks.

Luckring [50] conducted surface pressure measurements on a 65° delta wing and revealed that unlike sharp leading edge, where separation is fixed at apex, blunt edge delayed the shear layer separation to about 30% of the chord. Conversely, it has been proved that the leading edge radius reduces the size and strength of the vortex [2]. The vortex breakdown location moves upstream with the increase in angle of attack while sweep angle affects the onset angle of vortex breakdown. The onset angle increases with the increase in sweep angle.

Most of the work related to vortex breakdown has been done over slender delta wings for the obvious reasons that leading edge vortices formed are more coherent, strong, distinct and away from wing body in contrast to non-slender delta wings. Figure 9 shows the scatter in the vortex breakdown locations over a same delta wing but under different tunnel and flow conditions [2].



Figure 6: Variation of vortex breakdown with angle of attack over slender delta wing

It has been observed that the vortex breakdown trend between leading and trailing edge increases with sweep angle while it shows linear trend for low sweep delta wings. These factors in addition to the above mentioned points delay the vortex breakdown to a much higher value of angles of attack. In case of non-slender delta wings the leading edge vortices formed much closer to wing surface and a strong interaction develops with boundary layer yielding an earlier breakdown [9, 10, 38, 41].

The intriguing aspect of vortex breakdown along with its practical implications makes it an interesting field of research. The unsteadiness associated with breakdown is well documented and even involves an out of phase oscillation of breakdown points along the vortex axis which results in periodic roll motion, or wing rock [2, 3, 39]. This kind of phenomenon is not found over non-slender delta wings.

2.1.2.2 Types of Vortex Breakdown

There are two major types of vortex breakdown observed over delta wings at higher values of angles of attack bubble type and spiral type [30], although in reality they may just represent the extremes in a continuum of breakdown forms [3]. Both these types of vortex breakdown contained within volume whose radial dimension is close to that of the upstream core radius.

Leading feature in bubble type is sudden and large axisymmetric expansion and generation of recirculation zone. The cone after passing through this zone sheds in the form of vortex rings. Swirl angle plays a key role whose first critical value produces spiral type vortex breakdown to exist. Further increase in swirl will move vortex breakdown upstream and second critical value will convert it to bubble type. Flow visualizations have revealed that the direction of this recirculation is opposite to that of leading edge vortex before breakdown. The magnitude of axial velocity drops from jet like to wake like within few core diameters [2]. Hall [29, 30, 47] and Leibovich [44, 45] carried forward the work done by Sarpkaya [51] and concluded the following results [18]. Vortex breakdown of either bubble or spiral resembles a solid obstruction changing from an upstream jet like flow to wake like flow. This change occurs across the vortex breakdown. The wake regions are observed to be unstable to nonaxisymmetric disturbances that produce coherent periodic low frequency oscillations. The above result was thought to be due to waves propagating

azimuth. Flows upstream of the breakdown are stable to nonaxisymmetric disturbances, but flows downstream of the breakdown are not.

2.1.3Aerodynamic Characteristics

Aerodynamic characteristics of delta wings have always been vital. Various techniques like theoretical, analytical, experimental, numerical and computational have been employed to investigate the load coefficients (lift, drag and moment) over delta wings both in pre and post stall regions [2, 3]. The aerodynamic characteristics of slender delta wings are non linear in nature as shown in figure 7.



Figure 7: Lift coefficient variation with angle of attack

A major portion of the lift generated over delta wing is attributed to vortex lift, generated due to suction pressure of leading edge vortices. The comparison reveals that lift curve gradient with respect to angle of attack has inverse relationship with sweep angle. The direct relation between circulation and angle of attack has already been established but, keeping angle of attack fixed, circulation has opposite connection with sweep angle [3]. This is due to the fact that leading edge vortices are established over non-slender delta wings than slender delta wings at lower angles of attack. The positive lift curve gradient starts to decrease as vortex breakdown hits the wing surface causing a decrease in suction

pressure. Soltani [52] concluded that "The nonlinear vortex lift and the movement of the burst point over the wing surface were related to changes in the measured lift-curve slope." Although the main objective of this research is not the aerodynamic characteristics but, still some basic information is presented here for load coefficients.

2.1.3.1 Lift Estimation

Various direct and indirect methods are in use to estimate the lift over delta wings.

2.1.3.1.1 Direct Method

Force balance has been used as direct method of estimating lift coefficient. It can be easily observed that at lower values of angles of attack the lift coefficient (C_L) trend is linear. As it reaches higher values it becomes non linear indicating the presence of vortex lift. This factor is dominant in high sweep delta wings where the leading edge vortices dominate with their presence unlike in non-slender delta wings. Erickson [13], Soltani [23, 52], Wentz and Kohlman [19, 41, 53] have worked extensively to estimate the lift coefficients for a wide range of Reynolds number. It has been observed that the presence of vortex breakdown over wing decreases the rate of growth of lift coefficient but overall value still increase until the wing stalls before maximum lift coefficient (C_{Lmax}) is achieved.

Not much work has been done for non-slender delta wings due to the complexity associated with them. The maximum value obtained in case of non-slender delta wings is much lower than high sweep delta wings and creating direct relationship with normal force coefficient. The explanation provided by Nelson and Pelletier [3] is "for a given angle of attack, the strength of leading edge vortices increases with increasing aspect ratio or lower leading-edge sweep". This essentially means that keeping angle of attack fixed for relatively lower values, lift coefficient for non-slender delta wings will be higher than slender delta wings [10, 38, 40, 41]. Taylor and Gursul recently [10] have worked over flexible non-slender delta wings in an attempt to device a method of delaying vortex breakdown and to improve lift coefficient. They concluded that "passive lift enhancement over a flexible, low sweep delta wing has been demonstrated as a potential method for control of vortex-dominated wing flows. Lift enhancement was achieved in the post stall region, and increased the lift coefficient by up to 45%, and delayed stall by up to 9°".

2.1.3.1.2 Indirect method

Enormous research has been conducted to approximate the lift coefficient (C_L) from the flow field information over slender delta wings unlike non-slender delta wings. Quantitative wake surveys have been performed by various researchers to estimate the lift and drag. These gain popularity due to their ability to approximately estimate different components of drag. Kusonose [54] provided the universal wake data analysis based upon the theories postulated by Maskell [55] and Brune [56]. Both theories used spanwise circulation and control volume approach to estimate the lift over wing in addition to planar wake assumption [55, 56]. These hypotheses were limited to main wing body but, could not grip the tip vortices properly. The situation becomes complex with shear layer rolling up creating a change in actual wing span and effective span. The error was profound for slender delta wings in contrast to low sweep delta wings [2]. To resolve this issue, Kaplan [57] introduced the concept of effective span which was applied in conjunction with Kutta-Joukowski. This assumption developed better compliance between Digital Particle Image Velocimetry (DPIV) data, wake surveys data and the date presented in literature [2].

2.1.3.2 Drag Estimation

Drag force acting on delta wings can be decomposed in to different components as shown in figure 8. This decomposition of aerodynamic drag provided an insight to the different components of drag and its impact on lift estimation. It is very difficult and critical to predict exact value of drag using wake survey and flow field information. The section downstream of trailing edge has all the disturbances coming from wall effects and wing support. The situation becomes grave once the vortex breakdown moves over the delta wing.

A brief literature survey about the production and growth of leading edge vortex and different flow characteristics upon which it depends were presented in this chapter. Flow characteristics such as axial velocity, tangential velocity, axial vorticity and circulation were discussed. Vortex breakdown phenomenon was also presented with its types and its dependency on certain flow and geometrical conditions were also talked about in this chapter. Finally, a little summary has been presented about the aerodynamic load coefficients.

The following chapter will elaborate the experimental facility, procedure, apparatus and experimental procedure. This will also illustrate the geometrical parameters of experimental model and the flow conditions to be used.



Figure 8: Breakdown of aerodynamic drag [2]

Chapter 3

3 Experimental Procedures and Apparatus

3.1 Experimental Flow Facility

Quantitative flow measurement experiments were performed in the Joseph Armand Bombardier wind tunnel available in the Experimental Aerodynamics Laboratory of the McGill University located at MD-162 in the Department of Mechanical Engineering. A schematic diagram of the wind tunnel and photographs of the contraction and exhaust are provided in figure 14. The open loop wind tunnel suits to study the basic characteristics of complex flow field. The wind tunnel is equipped with acoustic silencer to avoid noise generation in addition to 16 blades, 2.5m diameter fan. This fan is controlled computer controlled variable speed AC motor to provide necessary suction. The test section dimensions are $1.2 \times 0.9 \times 2.7m$ in the z, y and x directions respectively.



Figure 9: Schematic of wind tunnel [1, 2, 58]

The wind tunnel is 19m long in total with 3.3m of contraction section, 2.7m of test section, 9.1m of diffuser section followed by 0.3m, 1.2m and 2.4m of vibration absorber, power section and acoustic silencer respectively. These dimensions result in contraction ratio of 10:1. The turbulence intensity provided is 0.05% at a free stream of 35m/Sec which is essentially provided by 10mm honeycomb and a series of 2mm anti turbulence screens. The chamfer of wind tunnel test section wall corners decreases with downstream distance to counter the negative pressure gradient developed [58]. The start of test section is equipped with miniature static-pitot tube, connected to a Honeywell DRAL501DN differential

transducer with a maximum water head of 50mm. This transducer was already calibrated against the fan speed by precisely regulating using digital controller. Both baseline and apex flap delta wing experimental models were mounted over the wing support for all the test conditions. This wing support was erected from the wind tunnel bottom surface.



Figure 10: Wind tunnel (a) inlet (left top) (b) outlet (c) test section (d) acoustic silencer [2]

3.2 Instrumentation and Data Processing

In the present study one intrusive method was used to measure the velocity field. The information collected over the wing was then be used to calculate the load coefficients. Flow measurement probes were mounted on computer controlled, five degree-of-freedom traverse, which was actuated by an actuator, controlled by a data acquisition system, enabling full automation of the scanning process [1]. The spatial resolution of the traverse was 20µm along each of the x, y and z axes, and the total test section blockage from the traverse was approximately 8%.

3.2.1Seven-hole Pressure Probe

Seven-hole pressure probe was used for measuring and investigating time-average velocity vectors to quantify the formation and growth of leading edge vortices over delta wings. The seven-hole pressure probe is sensitive for flow angles of 70° for the incoming flow. In addition to that the additional two holes will increase the effective area of cone as compared to five holes pressure probe in case of flow separation. The seven-hole pressure probe consists of three main components which includes probe sting assembly, an array of transducers and signal conditioner unit.

The 2.8 mm diameter brass tip of seven-hole pressure probe has seven holes drilled in a closely packed configuration along its axis. Each hole is approximately 0.5 mm in diameter and ground to a 30° cone angle at the probe tip. The probe assembly and configuration of holes are shown in figure 16. The probe tip is held by 130 mm long probe shaft which is then fixed to the end of 400 mm long and 12 mm diameter probe sting. Flexible tygon tubing of 1.6 mm diameter and 550 mm length connects each pressure tap of probe to an array of pressure transducer.

The pressure transducers array is a series of seven Honeywell DC005NDR5 differential

transducers with a maximum water head of approximately 127 mm (5 in), fixed to a rigid sub-frame to ensure that all transducers membranes remain in same plane. This collection of transducers was firmly attached to the traversing mechanism whereas the ambient atmospheric pressure for all these transducers was provided from inside a cover damping unit.

The signal conditioner unit for this

a) 0.28 cm 13 cm 13 cm 13 cm 1.2 cm1.2 cm

Figure 11: Seven-hole pressure probe [1, 2]

transducers array was a custom-built, seven-channel analogue signal differential amplifier. This uses an external DC offset of 3.5 Volts which provides a fixed gain of 5:1. The transducers were found highly linear within 2% with a resolution of 125 Pascal/Volt over the whole calibration range. Seven-hole pressure probe measurements are time average and

steady enough so no analogue to digital filter is required. Every attempt was made to keep

the length of the tygon tubing short but, it was still sufficiently long to provide hydraulic damping of any noise more than 5 Hz. The signal conditioning output was fed into data acquisition system programmed by LabVIEW and the signals were monitored using an oscilloscope. The probe was then calibrated in situ, using the methods presented by Wenger and Devenport [59] and Birch [1], the details are presented in appendix A. The seven-hole pressure probe is mounted on 2-axis traverse system with Sanyo Denki model 103-718-0140 stepper motor to move in y-direction and Bionode model 2013MK2031 stepper motor for z-direction. The system was run by NI PCI-7344 4-axis motion controller operated through LabVIEW.

3.3 Data Acquisition and Reduction

Data was gathered using a 16-channel, 16 bit NI-6259 A/D board which was powered by Dell Dimension E100 PC and a NI BNC-2110 connector box that accepted the transducer outputs. The sampling frequency determines the processing time and quality of data. Choosing a low value of frequency will propagate the chance of missing useful flow field information. On the other hand using high frequency will result in capturing some unwanted high tone noises. A sampling rate of 500Hz was chosen for a total of 3000 samples. Figure 17 shows the different steps in obtaining the data.



Figure 12: Flow chart of seven-hole apparatus setup

The adaptive scan grid was placed perpendicular to wind tunnel floor. The grid resolution and boundaries of adaptive grid was case-dependent. The grid points were varied between 500-7000 depending on the different chordwise stations and angles of attack to get a better picture of flow field information. The finer grid of $\Delta y = \Delta z = 1/16'' = 1.6$ mm, was used where useful information could be present while coarse grid up to $\frac{1}{4}'' = 6.35$ mm was used in the far field region to ensure capturing a larger scan area. The physical limitation on finer grid was the probe size so the minimum size of grid attained was 1/16'' = 1.6 mm. Since the delta wing span was large 30'' = 765 mm which resulted in longer wind tunnel scan times which could be counterproductive for the flow field facility.

The final results were presented on the finest resolution so coarser grid was revised to finest resolution using interpolation between the actual raw data points. The set of data obtained from seven-hole pressure probe has u, v, w, p_{static} and p_{total} and these parameters were later used to convert into other derived quantities such as axial vorticity (ζ) and circulation (Γ). Axial vorticity can be calculated using 2nd order difference scheme. Different types of techniques were adopted primarily depending upon the location of grid point. The following formula was used to calculate the vorticity.

$$\zeta_{i,j} = -\left(\frac{\partial v}{\partial z} - \frac{\partial w}{\partial y}\right) \approx -\left(\frac{v_{j+1} - v_{j-1}}{2\Delta z} - \frac{w_{i+1} - w_{i-1}}{2\Delta y}\right)$$

Where i = 2, 3, ..., n-1 and j = 2, 3, ..., m-1 and n and m are the number of measurement points in spanwise and transverse direction. Grid resolution becomes important to calculate the value of axial vorticity due to its numerical sensitivity. The vortex core and total circulation can be calculated using Stoke's theorem.

$$\Gamma_o = \sum \sum \zeta_{i,j} \times \Delta y \Delta z \qquad r_{i,j} < r_o \qquad \qquad \text{Equation 1}$$

$$\Gamma_c = \sum \sum \zeta_{i,j} \times \Delta y \Delta z \qquad r_{i,j} < r_c \qquad \qquad \text{Equation } 2$$

$$\Gamma_o = \sum \sum v_{\theta,j} \times r_i \qquad r_{i,j} < r_o \Gamma_c = \sum \sum v_{\theta,j} \times r_i \qquad r_{i,j} < r_c \qquad \text{Equation 3}$$

where

$$r_{i,j} = (z_j - z_c)^2 + (y_i - y_c)^2 r_o = r(\zeta = 0.01\zeta_{i,j\max})$$
 Equation 4

$$v_{\theta,j} = (v_{i,j} - v_c) \sin \theta - (w_{i,j} - w_c) \cos \theta$$
 Equation 5

3.4 Delta Wing Models

3.4.1Wing Models and Specifications

Two non-slender delta wing models have been used in the present study, both have leading edge sweep angle, Λ =45°. The difference between the two is that one has a moveable apex flap at x/c=0.28, as shown in figure. The baseline model is to confirm the procedural accuracy with the published literature while the modified model would be used to control the vortex breakdown location and to get information about the vortex flow field information.

Both wings were made from aluminum alloy 6061, 3/16'' = 4.75 mm thick plate. The geometrical details and parameters are mentioned in the table 1. All the leading and trailing edges were beveled at 15° windward to fix separation point. This will also ensure strong shear layer emanating from the leading edges to roll up.

Parameter	Notation	Units	Static model	Dynamic
				model
Wing root chord	с	in / m	15 = 0.381	15 = 0.381
Wing Sweep Angle	Λ	0	45	45
Wing Span	b	in / m	30 = 0.762	30 = 0.762
Aspect Ratio	AR		4	4
Wing thickness	t	in / m	3/16 = 4.75e-3	3/16 = 4.75e-3
Thickness to chord ratio	t/c	%	1.25	1.25
Wing area	S	in^2/m^2	225 = 0.145161	225 = 0.145161
Wing volume	V	in^3/m^3	42.188 = 6.9e-4	42.188 = 6.9e-4
Wing Bevel Angle	σ	0	15	15
Apex Flap location		%		28

Table 1: Geometrical parameters of experimental models

The support structure was especially designed in order to ensure a solid grip with the rigidity of a wing model. Analytical calculations were performed for column buckling based upon the static and aerodynamic loadings as a first step of support design. The wing was fastened to the semicircular disk with a circular array of holes to change the angle of attack. Vertical support of 1'' = 25.4 mm width was fastened to aluminum block which was then bolted to $\frac{1}{2}'' = 12.7$ mm thick aluminum base plate. Vertical support was then connected to the above mentioned semicircular disk through shoulder bolt and loosely fit positioning pin. The location of this wing support was at x/c = 0.67, which allowed the trailing edge to play its normal part with respect to pressure gradient. Therefore a support

structure was then finalized which allowed the angle of attack to be changed by every 1°. An aerodynamic fairing was used to keep the disturbance to minimum. The main driving force is to mitigate the effects of flow disturbances while not compromising on wing rigidity.



Figure 13: Geometrical drawing of experimental model



Figure 14: Experimental model, measuring sensor and traverse system
3.4.2Experimental Method

Since the primary aim of this study was to investigate the flow characteristics and flow behavior and the dependency of these on factors such as angle of attack, Reynolds number and chordwise locations. Another important motive behind is that to trace the vortex breakdown location and to effectively control it to improve wing performance. Figure 20 shows the general approach followed in the present study. The apex of the wing was considered to be the origin and all the distances in the x, y and z directions were measured from that point in the streamwise, vertical and horizontal directions respectively. Nonslender delta wings were under consideration so that lower angles of attack were the prime focus at both fore and aft of trailing edge. The large wing span allowed to bring the pressure probe as far upstream as x/c = 0.2. The mean pressure and velocity measurements thus collected were used to calculate the various flow parameters based on these data sets.



Figure 15: Working scheme of present experimental study

In the streamwise directions, measurement planes were separated by 0.05c but at locations in the vicinity to vortex breakdown the measurement planes were only 0.01c apart. This allowed to develop better understanding about the vortex breakdown and to locate its position.

The next chapter will describe the results and discussion based upon the data sets collected above and behind the delta wings. Mean flow measurements for both baseline and modified apex flap model results are presented.

Chapter 4

4 Results and Discussion

This chapter primarily presents the results and discussion for both baseline and apex flap models. It has already been mentioned in the previous chapter that the sensor used was a seven-hole pressure probe and the scans were performed both upstream and downstream locations of trailing edge. The first part of this chapter deals with the baseline model results and the comparison established with the published literature. The development of leading edge vortex with respect to Reynolds number, angle of attack and various chordwise stations was investigated in detail. With the help of these results, a better understanding can be developed about the initiation and growth of leading edge vortex over non-slender delta wing. These results will include the contour plots obtained of different flow characteristics such as axial velocity, tangential or swirl velocity, axial vorticity and pressure contours.

The later part will focus on the apex flap model and the control it provides over the leading edge vortex through passive deflections and also its contributions to delay vortex breakdown. This will also be presented based upon the results obtained on the basis of above mentioned vortex flow field characteristics.

As a first step the seven-hole pressure probe was calibrated at free stream velocity of 15m/s for pitch and yaw angles of $\pm 60^{\circ}$. The following results give insight of calibration and the details of the method are available in the appendix A [1]. In order to establish conformity between results, a detailed analysis was carried out to serve as benchmark data. To serve this purpose, extensive investigations were carried out at various chordwise stations ranging from x/c = 0.2 to x/c = 1.85. One of the measurement plane was selected at x/c = 1.05 to capture the complete flow field information from both leeward and windward sides. By doing so an effort was made to avoid the wake entrainment effects coming from wind tunnel walls and wing support system. The following subsections will discuss the variation of different flow parameters and their control on both the experimental models used in the present study.

4.1 Baseline model

4.1.1Variation of vortex characteristics with chordwise stations

The initiation and growth of leading edge vortex with respect to chordwise stations is discussed in this section. The leading edge vortex grows in size and strength with downstream movement. There is an important point to make here that minimum cross location; maximum axial velocity location and maximum axial vorticity do not coincide over non-slender delta wings as they do over high sweep delta wings. This makes core selection an important and critical decision. Minimum cross flow location has been selected as the core centre in the present study as different parameters will be calculated on the basis of this selection.

Figure 17 (a-f) shows the contour plots of normalized axial vorticity distribution at $\alpha = 6^{\circ}$ and for a Reynolds number of 371,000. The information obtained is more qualitative and the contours are weaker and diffused. It is obvious that the leading edge vortices are formed much closer to the wing, making it difficult for the flow to properly roll as it is the case for slender delta wings. Both sides of wings can be seen ruling out any geometrical unbalance. At this angle of attack, vortex breakdown reaches the trailing edge or slightly upstream of it. It is difficult to locate as the trailing edge plays its role in changing the adverse pressure gradient which controls the location of vortex burst. The magnitude of the vorticity has different signs due to opposite direction of vorticity. With vortex moving downstream, the size increases gradually at the cost of strength. The decrease in magnitude suggests the effect of adverse pressure gradient. Figure 18 (a-h) shows the contour plots of normalized axial velocity distribution at $\alpha = 6^{\circ}$ and Reynolds number of 371,000. The earliest information these contours produce is that there is jet flow or maximum axial velocity (u_{max}) just at the vortex outer limits. Its magnitude is around 1.25 times the free stream velocity (U_{∞}) and it consistent irrespective of the chordwise station before the vortex breakdown. The acceleration responsible to move this jet like flow can be streamwise escalation of leading edge vortex. The other important information which can be extracted is the mismatch of (u_{max}/U_{∞}) and maximum axial vorticity.

Figure 19 (a-h) shows the contour plots of axial vorticity distribution normalized by local chord length to incorporate geometrical effects, plotted at different locations upstream



Figure 16: Chordwise distribution of normalized axial vorticity contours at Re = 371,000 and $\alpha = 6^{\circ}$



Figure 17: Chordwise distribution of normalized axial velocity contours at Re = 371,000 and $\alpha = 6^{\circ}$



Figure 18: Chordwise distribution of normalized axial vorticity contours at Re = 247,000 and $\alpha = 10^{\circ}$

of trailing edge. These contours are chosen at $\alpha = 10^{\circ}$ and Reynolds number of 247,000. The magnitude of vorticity is much lower than the slender delta wings. The value of incidence is chosen where the vortex breakdown is found over the delta wing. At x/c = 0.3, a skewed core can be visualized, the strength of this core is comparable in magnitude to strength of shear layer rolling up from leading edge. With the downstream movement, the size of leading edge vortex grows but, the magnitude decreased. This is because of the streamwise adverse pressure gradient and boundary layer interaction with leading edge vortex.

At x/c = 0.55, the weak core looks broken down and this argument is supported by the contour plots of axial velocity and pressure loss plots. A significant difference in the values can be observed upstream and downstream location of vortex breakdown. To be certain about the location of vortex breakdown, scans were performed at every 1% of chord length. Similar results were obtained at Reynolds number 371,000 and 445,000. The comparison between different values of free stream velocities will be shown in the upcoming topics. There is another important point to be mentioned here that leading edge vortex tries to roll up in the vicinity of wing surface, interaction of this vortex with boundary layer may cause the oncoming flow angle to exceed $\pm 60^{\circ}$. This may end up making it difficult for sevenhole pressure probe to resolve the angle and cause few bad points in the grid. Those points were not considered while generating the interpolated grid.

Figure 20 (a-h) shows the contour plots of axial velocity which is normalized by free stream velocity are plotted at different locations upstream of trailing edge. Here again a magnitude of $u_{max}/U_{\infty} = 1.25$ is observed in the form of jet like flow at the outer periphery or leading edge vortex which is comparable to the reported value [32]. The important point is that no significant change has been observed in this jet like flow even after the vortex breakdown.

The variation of normalized axial velocity (u/U_{∞}) , tangential velocity (v_{θ}/U_{∞}) and axial vorticity $(\zeta c/U_{\infty})$ against chordwise stations at Reynolds number 247,000 and $\alpha = 10^{\circ}$ is plotted as shown in figure 21. These plots are obtained by passing a horizontal line thorough the core centre at each contour plot. Both upstream and downstream locations of vortex breakdown are plotted in two separate graphs to have better comparison. In doing so the scales of both abscissa and ordinates are kept fixed. Figure 21 (a) and (b) show the normalized tangential velocity plots against the normalized core radius at both upstream



Figure 19: Chordwise distribution of normalized axial velocity contours at Re = 247,000 and α = 10°



Figure 20: Variation of flow characteristics with chordwise stations (a, c and e) before vortex breakdown (b, d and f) after vortex breakdown at $\alpha = 10^{\circ}$ and Re = 247,000

and downstream of vortex breakdown. It is evident from these plots that magnitude across the vortex has decreased significantly after the breakdown. Another important derivation is the asymmetry of vortex about the vortex centre suggesting the skewed shape of leading edge vortex. The effect of strong shear layer and its closeness to wing surface is indicated by the sinusoidal movement of values on the left side of the core centre. There is a drop of approximately 50% on each side when compared the magnitude before and after the vortex breakdown.

Figure 21 (c) and (d) show the variation of normalized axial velocity plotted against the normalized core radius at both pre and post vortex breakdown regions. Again the comparison before and after vortex bursts reveals handy information. The variation of axial velocity was concise within 10% on left side and 5% on right side of normalized radius, indicating the skewed form of vortex before breakdown. The left side suggests the dominating role of strong shear layer which the vorticity feed brings in. However, in the post breakdown region the variations spread in the wider range and the magnitudes were also reduced by as much as $10 \sim 15\%$.

Figure 21 (e) and (f) show the variation of normalized axial vorticity plotted versus the normalized core radius. The magnitude was higher at x/c = 0.3 and was reducing gradually in the pre breakdown region. The decrease in magnitude was not sharp like slender delta wings. The multiple peaks here indicate that strength of shear layer is comparable to leading edge vortex itself. Most of the data falls in the region of $\pm 10\%$ of non-dimensional local radius in the pre breakdown region. The magnitudes drop of around 20% and an increase in spread of more than 80% indicate the presence of vortex breakdown. This argument is further supported by the deviation in suction pressure contours and vortex trajectory modification plotted which will be shown later.

4.1.2Variation of vortex characteristics with angles of attack

This section will focus on the variations of vortex flow characteristics with the change in angles of incidence. The deviation of normalized axial velocity (u/U_{∞}) , tangential velocity (v_{θ}/U_{∞}) and axial vorticity $(\zeta c/U_{\infty})$ versus angle of attack is presented at Reynolds number = 247,000 and chordwise station, x/c = 0.3 is plotted as shown in figure 22. The growth in size, trajectory and other characteristics can be easily understood by viewing



Figure 21: Variation of flow characteristics with angle of attack (a, c and e) before vortex breakdown (b, d and f) after vortex breakdown at $\alpha = 10^{\circ}$ and Re = 247,000

these figures. Figures 22 (a) and (b) show the variation of normalized tangential velocity plotted against normalized local core radius at different angles of attack. The angles before and after the vortex breakdown are separated in two different plots at x/c = 0.3. The direct relationship between size of leading edge vortex and angles of attack can be easily

observed. Based on this information the skewed core size can be said to be around 5% of local normalized radius at x/c = 0.3, Reynolds number = 247,000 and $\alpha = 10^{\circ}$. The dominance of shear layer can again be observed when it generates sinusoidal waves. However, this supremacy features more after the breakdown has either passed this location or is in the vicinity. The size of the vortex was



Figure 22: Absence of dual vortex structure (normalized axial vorticity) at $\alpha = 2^{\circ}$ and $\alpha = 6^{\circ}$

found to be somewhat independent of angle of attack.

Figures 22 (c) and (d) demonstrate the variation of non-dimensional axial velocity plotted against normalized local core radius at different angles of attack. The peak magnitudes are in the vicinity of $u/U_{\infty} \approx 1.2$ and rest the whole field is at the free stream velocity before the vortex breakdown. In the post breakdown section, diffused wake like structure is found in $\pm 10\%$ of local normalized radius. Figure 22 (e) and (f) illustrate the non-dimensional axial velocity plotted against normalized local core radius at different angles of attack. The vorticity level has dropped by as much as 30% and spread has increased by approximately two times the width before vortex burst.

There is another important point to be established that the wavering movement of vortex level indicates the presence of strong shear layer. The strength of strong shear layer is comparable to that of leading edge vortex. This also rules out the presence of dual vortex structure which has been reported in literature at low angles of incidence ($\alpha \approx 5^{\circ}$) and low Reynolds number of the order of 10⁴. Figure 23 shows the absence of dual vortex structure for different lower values of angle of attack.

4.1.3Variation of vortex characteristics with Reynolds number

Vortex characteristics are reported to be independent of Reynolds number for slender delta wings where leading edge vortex is away from wing surface. However, it is reported to influence vortex flow characteristics over non-slender delta wings at low Reynolds number of the order of 104. The range of Reynolds number in the present study is from 247,000 -445,000 which is much higher than the value where Reynolds number affects the leading edge vortex. Figures 24 and 25 (a-c) show the contours of normalized axial vorticity and normalized mean axial velocity at $\alpha = 10^\circ$, x/c = 0.4 and free stream velocity (U_{∞}) varying from 10m/Sec - 18m/Sec. There is no significance difference between any of these contours as far as vortex size and strength of shear layer is concerned. A small and diffused vortex core whose location is different from the maximum vorticity location can be seen in all of these contours. Since there is no major difference found so it can be safely assumed that vortex characteristics are independent of Reynolds number. Figure 26 (a-f) shows the variation of normalized axial vorticity ($\zeta c/U_{\infty}$), suction pressure variation ($\Delta p_0/q_0$), normalized axial velocity (u/U_{∞}) , normalized maximum axial velocity (u_{max}/U_{∞}) and trajectory of vortex core location at different chordwise stations for different Reynolds number. It is evident that all of these lines collapse on each other emphasizing the independence of all these parameters from Reynolds number. A small variation in normalized axial vorticity at around x/c = 0.55 can be seen suggesting the vortex breakdown. Its location is further complimented by the sharp drop in suction pressure value and change in axial velocity magnitude. The trajectory of vortex core has no significant effect due to Reynolds number.

4.1.4Comparison of baseline model with apex flap model

Figure 27 shows the comparison between the baseline and apex flap model to see the effect of flap activation mechanism on the flow field. Different flow characteristics are plotted for both models. It is interesting to see that all the lines collapsed on each other. This suggests that there is no considerable effect of flap activation mechanism on different flow characteristics. Figure 27 (a) and (b) shows the normalized axial vorticity and nondimensional pressure loss versus different chordwise locations at $\alpha = 10^{\circ}$, Re = 247,000 and $\beta = 0^{\circ}$ for flap angle. There is no significant difference between the two and both



Figure 23: Normalized axial vorticity contours at x/c = 0.4, $\alpha = 10^{\circ}$ (a-c) Reynolds number effect (d-h) Apex flap effect



Figure 24: Normalized axial velocity contours at x/c = 0.4, $a = 10^{\circ}$ (a-c) Reynolds number effect (d-h) Apex flap effect



Figure 25: Reynolds number effect on different flow characteristics at $\alpha = 10^{\circ}$



Figure 26: Comparison of baseline and apex flap model at $\alpha = 10^{\circ}$ and Re = 247,000

follow the same pattern while moving downstream. Figure 27 (c) and (d) shows the normalized mean axial velocity and normalized maximum axial velocity variation. Both these figures show good relationship between the two models. Similar results can be seen for the vortex trajectory as shown in figure 27 (e) and (f).

4.1.5Variation of vortex characteristics with apex flap

Apex flap has been used as a method to control the leading edge vortex and to delay the vortex breakdown over non-slender delta wing. Figures 24 and 25 (d-h) show the effect of apex flap on normalized axial vorticity and normalized axial velocity for different passive flap deflections in the range of $\beta = \pm 10^{\circ}$ at x/c = 0.4, $\alpha = 10^{\circ}$ and Reynolds number = 247,000. The upward deflection is termed as positive while the downward deflection is referred as negative. Since the effect of Reynolds number was insignificant so contour plots at $U_{\infty} = 10$ m/Sec are shown as representative. The negative flap deflection makes the apex flap at smaller angle while the remaining wing remains at higher angle of attack. The upper deflection of flap increases the effective angle of attack for flap.

As the flap is 28% of root chord length so the plane selected to take measurements is at x/c = 0.4. Comparison of zero flap deflection with the baseline model shows no significant difference on vortex trajectory and size of vortex. Figure 24 (e) and (g) shows the effect of negative flap deflection on axial vorticity. Flap deflection of $\beta = -5^{\circ}$ shows interesting phenomenon as the size of vortex decreased with the change in trajectory. As the vortex moved up, this permitted the flow to roll properly under the vortex. Since the flap has 5° effective angle of attack while wing body is at 10°, so this kept the size of vortex smaller with higher magnitude of vorticity. The strong vortex will result in delay of vortex breakdown as the interaction with boundary layer will be minimal. Another reason for the change in trajectory and strength could be the wing to be at higher angle of incidence increasing the gap between vortex and wing body by few core diameters. Following this observation the flap was deflected to $\beta = -10^{\circ}$ to see the extent to which the negative deflection would be beneficial. The results were a little surprising as the size of vortex got even smaller with change in trajectory and once again it touched the wing surface. However the level of vorticity was higher than the previous case. This generated interesting points about the vortex structure and flow characteristics that up to which flap angle and chordwise station the results will be helpful. It was expected that the vortex will be further away from wing surface with more deflection but the adverse pressure gradient changed the trajectory of core significantly.

On the contrary, the size of the vortex increased tremendously but, at the cost of vorticity level as shown in figure 24 (f) and (h). The upward deflection increased the effective angle of attack of flap and adverse pressure gradient changed the path of vortex drastically. Vortex breakdown was promoted upstream leaving behind only weak and diffused flow structure. Another interesting feature was observed in the axial velocity plots where two jet like flow patterns were observed instead of just one. The shear layer feed was also strongly affected by the negative flap deflection and an inverse relationship was found between the two.

Figure 28 (a-f) shows the variation of normalized axial vorticity ($\zeta c/U_{\infty}$), suction pressure loss ($\Delta p_o/q_o$), normalized axial velocity (u/U_{∞}), normalized maximum axial velocity (u_{max}/U_{∞}) and trajectory of vortex core location at different chordwise stations for different passive and negative apex flap deflections. The angle of attack is kept constant at $\alpha = 10^{\circ}$ while Reynolds number was chosen to be 247,000. Figure 28 (a) shows the tremendous increase in the vorticity level before the vortex breakdown as the apex flap is at lower angle of incidence than the rest of the wing body. A sharp increase in vorticity magnitude is found just after the vortex breakdown indicating the reformation of vortex and reenergizing of flow. On the other hand similar observations are found for pressure loss plots which are consistent irrespective of flap deflections. Sharp variations in pressure loss and axial velocity plots indicate the delay of vortex breakdown location by as much as 8% compared to zero flap deflection [60, 61]. The trajectory was also affected by the flap deflection which moved towards root chord and also moved downward in the post vortex breakdown region.

Variation of normalized axial vorticity ($\zeta c/U_{\infty}$), suction pressure loss ($\Delta p_o/q_o$), normalized axial velocity (u/U_{∞}), normalized maximum axial velocity (u_{max}/U_{∞}) and trajectory of vortex core location at different chordwise stations are plotted as a function of chordwise stations for different positive and passive apex flap deflections. The variation of these parameters with apex flap deflections is shown in figure 29 (a-f). The angle of attack is kept constant at $\alpha = 10^{\circ}$ while Reynolds number was chosen to be 247,000. The changes in these parameters clearly indicate the promotion of vortex breakdown upstream by virtue of increase in adverse pressure gradient. Since the effective angle of attack for flap was much



Figure 27: Effect of negative apex flap deflections on flow characteristics at Re = 247,000 and $\alpha = 10^{\circ}$



Figure 28: Effect of positive apex flap deflections on flow characteristics at Re = 247,000 and $\alpha = 10^{\circ}$

higher than the wing body so it increased the frontal area and disturbed the flow. By doing so the feed was disturbed causing the trajectory to alter its path.

4.1.6Wake vortex progression

The growth and propagation of leading edge vortex downstream of trailing edge is also important. There were four different distinct chordwise stations chosen downstream of trailing edge at x/c = 1.05, 1.25, 1.45 and 1.85 for selected angles of attack to see the development of wake. Figures 30 and 31 shows the normalized axial vorticity and normalized axial velocity contours at x/c = 1.05, Reynolds number = 371,000 and $\alpha = 4^{\circ}$. The reason for choosing location at x/c = 1.05 is to capture the flow field information from both windward and leeward side and at the same time to keep the wall effects to minimum. This information can later be used to calculate the drag of the wing. Opposite signs of axial vorticity indicates the counter rotation of leading edge vortex whereas the same can be said about the flow coming from windward and leeward sides. The diffused vortex structure has no defined core centre and has low magnitude. The axial velocity contour reveals that there is strong wake like flow covering majority of the vortex portion.

To better understand the evolution of this weak vortex structure measurements were performed at other streamwise locations. Figure 32 (a-d) shows the variation of normalized axial vorticity contours at Reynolds number = 247,000 and $\alpha = 10^{\circ}$. Wake like flow is engulfed with the relatively weak jet like flow. This wake like flow pattern is understandably caused by the momentum deficit it has suffered suggesting the upstream vortex burst. At this angle of attack, vortex breakdown was in the vicinity of x/c = 0.55 so as a result a diffused vortex structure can be seen here. At x/c = 1.05, effects of shear layer are observed indicating the addition of low value of feed. As it reaches x/c = 1.25, the shear layer has faded away while the size of vortex has reduced. The amount of vorticity has dropped by ~30%. At x/c = 1.45, the magnitude is getting lower gradually with vortex getting rotated due to the tangential component of velocity. With the downstream movement, this skewed vortex is forming an elliptical vortex while the component of flow coming from windward side is fading away. At x/c = 1.85, the pattern looks comparable to wing tip vortex downstream of wing. The vorticity is confined to around 20% of semi span whereas the magnitude has dropped by \sim 50% of the preceding station value. The clockwise rotation of vortex is still observed at this location which will eventually form circular



Figure 29: Normalized axial vorticity contour at x/c =1.05, Re = 371,000 and α = 4°



Figure 30: Normalized axial velocity contour at x/c =1.05, Re = 371,000 and α = 4°



Figure 31: Normalized axial vorticity contours at x/c = 1.05, Re = 247,000 and $\alpha = 10^{\circ}$

shape. The magnitude of normalized axial velocity is found to be ~0.7 times the mean free velocity.

4.1.7Vortex breakdown

The term vortex breakdown has already been explained in the earlier discussion. Flow visualization was the main technique which was in use to report the vortex breakdown location. This procedure was a good technique but the information collected was qualitative in nature. Another important point which needs to be addressed is that much work has been done over slender delta wings where vortex breakdown occurs at higher angles of attack and the location is more stable as compared to low sweep delta wings. The prime reason for this is that maximum vorticity, maximum velocity and minimum cross flow location match for high sweep delta wings. This allows the vortex breakdown to be better defined over slender delta wings than non-slender delta wings.

It is important to mention that less work has been done over low sweep delta wings which can be due to the skewed form of leading edge vortex. In the present study, an attempt has been made to try and quantify the vortex breakdown location over delta wings with sweep angle, $\Lambda = 45^{\circ}$. To elaborate it in detail, scans were performed at every 1% of chordwise stations and at various angles of attack in the vicinity of vortex breakdown.

The normalized axial velocity and vorticity contours were less informative and decisive but, the normalized pressure loss contours and change in vortex trajectory further complimented the presence of vortex breakdown location. All this information can be extracted from figure 26 (a-f) where the vortex burst location has been highlighted. At $\alpha = 10^\circ$, the steeper drop in axial vorticity and velocity are found with sharp change in pressure contours at x/c = 0.55 indicating the vortex breakdown location. As not much work has been done on $\Lambda = 45^\circ$ at higher Reynolds number so a direct comparison has been made with 50° sweep delta wings and at low Reynolds number. It is important to mention that vortex breakdown location over non slender delta wings is highly sensitive and comes with a high degree of unsteadiness making it a transient phenomenon. It is therefore recommended to refer these readings with caution and keep numerical uncertainties and physical limitations in consideration. Figure 37 shows the vortex breakdown location for base line delta wing versus the angle of attack. A linear progression of vortex breakdown upstream follows the same pattern but it is a little off due to its comparison with different delta wings and at different flow conditions.



Figure 32: Variation of vortex breakdown location with angles of attack

It is important to point out that the progression of vortex breakdown upstream is linear over non-slender delta wings than slender delta wings. This could be due to wider local semi span for low sweep delta wings. In case of slender delta wings the vortex interaction causes the non linear trend while moving towards the apex.

Apex flap model reveal some interesting phenomenon by giving passive deflections. The downward or negative apex flap deflections delayed the effect of adverse pressure gradient and hence resulted in postponement of vortex breakdown. The delay in vortex breakdown can be observed from the axial vorticity, axial velocity and most importantly total pressure

loss plots which indicated the vortex burst location to be delayed by as much as 8% as specific angles of attack and specific apex flap deflections. On the other side the positive apex flap deflections propagates the vortex breakdown upstream due to increase in adverse pressure gradient. The other reason which could have promoted it can be the decrease in vorticity feed brought by shear layer.

Chapter 5

5 Conclusions and Future work

5.1 Contributions of present thesis

This experimental aerodynamics research study has investigated vortex flow characteristics over non slender delta wings and its major contributions include the experimental approach has been applied to quantify the vortex breakdown location over non-slender delta wings, apex flap has been used as a method to control the leading edge vortex and to delay the vortex breakdown, study and quantify the vortex flow characteristics over low sweep delta wings at both upstream and downstream of vortex breakdown and also trailing edge, attempt is made to quantify the vortex flow characteristics over the apex flap wing and downstream of trailing edge in the wake and experimental models of non-slender delta wing have been made apex flap modifications which can be used both as active and passive control

5.2 Conclusions

5.2.1Baseline model

- Complete and three dimensional information is not clear due to small vortex core size
- Peak velocity u_{max} was always on the peripheral region of vortex
- Axial vorticity drops from apex to trailing edge, the gradient becomes steeper near the vortex breakdown location
- Total pressure loss gets sharp change near vortex breakdown, this sharpness is irrespective of Reynolds number
- Variation of flow parameters with respect to streamwise locations and angles of attack is independent of Reynolds number
- Variation of vortex breakdown was also found to be independent of the changes made to Reynolds number
- Strong shear layer has its influence on the vorticity field creating multiple peaks

- Strength of shear layer was comparable to the strength of leading edge vortex
- The negative gradient in vorticity level was more in the pre vortex breakdown region than after the vortex burst
- No dual vortex structure was observed irrespective of angles of attack and Reynolds number
- Tangential velocity plots revealed the drop in vortex size by ~50% after vortex breakdown
- Leading edge vortex formed is skewed in shape irrelevant of flow conditions
- Leading edge vortex formed is either attached or is in the vicinity of wing surface
- Vortex trajectory was not influenced by Reynolds number
- Vortex breakdown does not apply on non-slender delta wing as it applies on slender delta wing. This is because the maximum axial vorticity, maximum axial velocity and minimum cross flow locations do not meet with each other.
- Pressure loss contours were handy in deciding the vortex breakdown location in addition to changes in axial vorticity and axial velocity
- Size and strength of leading edge vortex was directly coupled with angle of attack
- Size of leading edge vortex was directly linked with angle of attack where strength was indirectly related to angle of incidence
- u_{max} reached a value of $1.35U_{\infty}$ but, only in the peripheral region of vortex

5.2.2Apex flap model

- Vortex breakdown was observed to be delayed by as much as 8% by negatively deflecting apex flap
- Negative deflection of the apex flap reduced the size of vortex while increased the magnitude of vorticity
- Vortex core trajectory was shifted by passive apex flap deflections
- u_{max} was divided into two distinct regions with positive apex flap deflection
- Negative flap deflection moved the vortex breakdown downstream whereas it propagates it upstream with positive deflection. This suggests the delay of adverse pressure gradient for negative apex flap deflection and vice versa.

5.3 Future work

Two major areas which are not dealt with in this thesis are measurement of aerodynamic loads and to use apex flap as an active mean to control leading edge vortex. This study can be extended by making two more identical flaps at other two vertices of delta wing which can serve the purpose of dynamic tail. By controlling all three flaps will actively serve and replicate the concept of flexible non-slender delta wing. This can delay the stall angle when given small unsteady undulations of different frequencies and amplitudes for different flow conditions. In addition to this, these undulations will effectively control of leading edge vortex trajectory by maneuvering reattachment and separation lines. This will change the vortex flow characteristics and also the aerodynamic load coefficients by implementing these unsteady variations.

Seven-hole pressure probe provides only the time mean measurements at any cross section, a new advancement which can be done to see the transient behavior of flow passing over the unsteady delta wing movements. For this purpose, different types of sensors can be used such as hot wire anemometry. This will provide the velocity field variations with time and will also consider the hysteresis effects. By varying angle of attack and apex flap angles, amplitudes and undulation frequencies to compare with flexible non-slender delta wing. This will facilitate to understand in depth the initiation and growth of leading edge vortex both upstream and downstream of trailing edge.

Similarly, by making geometrically similar experimental models will allow to test in particle image velocimetry (PIV) equipped wind tunnel. Using this flow visualization facility will allow to compare the results obtained in the present study to compare with non-intrusive method. The results thus obtained can be compared with the information gathered through intrusive methods.

6 References

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Appendix A: Seven-hole pressure probe calibration technique

The seven-hole pressure probe (7HP) measures the time-averaged magnitude and direction of the flow in the vicinity of the probe tip. The probe tip is a series of seven pressure taps arranged in close-packed configuration in a truncated 30° cone. By convention, the holes are numbered 1 through 7, with hole 7 located at the center and holes 1 through 6 numbered clockwise from the bottom, as seen from upstream. By comparing the relative magnitudes of the pressures recorded at the different tap locations, the local flow velocity vector can be obtained. Due to the time-lag and damping effect of the length of tubing connecting the probe tip to the pressure transducer array, the 7HP is limited to measurement of time-mean values. Also, the probe is only sensitive to flow cone angles less than \approx 70° from the axis of the probe.

In order to calibrate the 7HP, an empirical function must be found which is homeomorphic within the range of measurement and can relate the pressure at the seven holes $\mathbf{P} = (P_1, P_2, ..., P_7)$ to the local velocity $\mathbf{v} = (u, v, w)$. In order to determine the values of this empirical function, the physics governing the sensor must be considered. First, since the magnitude of the velocity can be calculated from the difference between the static and stagnation pressures using Bernoulli's principle, and the 7HP provides local pressure measurements, the velocity magnitude can be computed directly rather than inferred from the calibration, reducing the order of the problem.

Since the magnitude of the velocity vector can be eliminated from the calibration parameter space, it becomes convenient to express the velocity in terms of magnitude and direction such that $\mathbf{v} = \{|\mathbf{v}|, \theta, \phi\}$ or $\mathbf{v} = \{|\mathbf{v}|, \alpha, \beta\}$, where θ is the cone angle, ϕ is the roll angle, α is the pitch angle and β is the yaw angle. The cone, roll, pitch and yaw angles are related to the orthogonal components of velocity as

$$\mathbf{u} = |\mathbf{v}| \cos(\beta) \cos(\alpha) = |\mathbf{v}| \cos(\theta) \tag{A1}$$

$$\mathbf{v} = |\mathbf{v}| \cos(\beta) \sin(\alpha) = |\mathbf{v}| \sin(\theta) \sin(\phi)$$
(A2)

w =
$$|\mathbf{v}| \sin(\beta)$$
 = $|\mathbf{v}| \sin(\theta) \cos(\phi)$ (A3)

These relationships are illustrated graphically in Figure A1.
Furthermore, at larger flow angles, the flow will separate from the tip of the 7HP and at least one of the pressure taps will be located in a wake region. Since surface pressures in wake regions are relatively insensitive to changes in flow magnitude or direction, the function relating P and v must be defined piecewise, depending on (a) whether or not there is flow separation over the probe tip, and if there is separation, (b) which taps are in the separated region. Since the probe tip is a 30° cone and the flow is known to remain attached everywhere on the probe tip when the probe is oriented parallel to the flow, and since the pressure in regions of separated flow is known to be higher than in regions of attached flow, the flow is assumed to be attached everywhere on the probe tip if the maximum pressure is recorded at the center tap.

In the case where the maximum pressure is recorded at the center tap and the flow is attached everywhere on the probe tip, the difference in the pressures between the top hole and the bottom hole will be a homeomorphic function of α (defined as C_{α}), and the difference in the pressure between the holes on the left and right sides will be a homeomorphic function of β (defined as C_{β}). Since the magnitude of the pressures will be a function of $|\mathbf{v}|$ and $|\mathbf{v}|$ has been eliminated from the parameter set, the pressures can be normalized against the local dynamic pressure $P_{TOT} - P_{STAT}$. The total pressure is taken as the pressure at the center tap, and the static pressure is approximated as the average of the pressure at all of the peripheral taps. Also, since there are six peripheral taps, there are two taps on each of the left and right sides and the average of the two side pressures are used. Thus,

$$C_{\alpha} = \frac{P_4 - P_1}{P_7 - P_{AV}}$$
(A4)

$$C_{\beta} = \frac{\frac{1}{2} \left(P_5 + P_6 \right) - \frac{1}{2} \left(P_3 + P_2 \right)}{P_7 - P_{AV}}$$
(A5)

$$P_{AV} = \sum_{k=1}^{6} \frac{P_k}{6}$$
(A6)

where k indicates the hole index number.

In the event that the maximum pressure is recorded at some hole *i* where $i \neq 7$, then the flow can only be assumed to be attached in the immediate vicinity of the stagnation point, where the hole *i* is located. As a result, only the hole *i* and the three holes adjacent to it are used to determine the values of the functions. The pressure difference between hole 7 and hole *i* will be a homeomorphic function of the cone angle θ (defined as C_{θ}), and the pressure difference between the holes located peripherally adjacent to the *i*th hole will be a homeomorphic function of the roll angle ϕ (defined as C_{θ}). The total pressure is taken as the maximum recorded pressure P_i , and the static pressure is approximated as the average of the pressures at the two peripherally adjacent holes P_{CW} and P_{CCW} (where the subscripts CW and CCW indicate the adjacent hole going around the probe tip clockwise and counterclockwise as seen from upstream, respectively).

$$C_{\theta i} = \frac{P_i - P_7}{P_i - P_{AV}} \qquad i = 1, 2, ..., 6$$
(A7)

$$C_{\phi} = \frac{P_{CW} - P_{CCW}}{P_i - P_{AV}} \tag{A8}$$

$$P_{AV} = \frac{P_{CW} + P_{CCW}}{2} \tag{A9}$$

The spherical co-ordinates are used in the case of $i \neq 7$ because the centers of the four holes used describe a 120° segment of a cone, and it is more convenient to describe this geometry in terms of cone and roll angles. The functions C_{θ} and C_{ϕ} are defined independently for each hole i = 1, 2, ..., 6.

Finally, since the magnitude of the velocity is computed based on approximate values of the static and total pressures, the measurement accuracy can be significantly improved by further defining functions C_{STATi} and C_{TOTi} , where i = 1, 2, ..., 7, which are

homeomorphic functions relating the approximate and actual values of the static and total pressures, respectively.

$$C_{STATi} = \frac{P_i - P_{STAT}}{P_i - P_{AV}} \qquad i = 1, 2, ..., 6$$
(A10)

$$C_{TOTi} = \frac{P_i - P_{TOT}}{P_i - P_{AV}} \tag{A11}$$

where P_{AV} is the approximated value of the static pressure, the definition of which depends on *i* and is given by equations A6 and A9.

To calibrate the probe, data is collected at a single free-stream velocity close to the expected velocities in the measurement region, and at many angles in pitch and yaw. The 7HP's measurement space is limited to the region $-70^{\circ} \le \alpha \le 70^{\circ}$, $-70^{\circ} \le \beta \le 70^{\circ}$. Above or below 70° in pitch or yaw, the functions described above are no longer sufficiently sensitive to the angles to be considered homeomorphic. The data is then used to construct seven calibration grids (Figure A2), with one grid associated with each of the seven holes. The calibration grids consist of a number of points in (C_{α}, C_{β}) or (C_{β}, C_{ϕ}) space of known flow angle, C_{STAT} , and C_{TOT} . Then, given any experimental pressure measurements, the hole registering the maximum pressure is determined and the corresponding calibration grid is used to interpolate the values of α and β (or θ and ϕ), together with C_{STAT} , and C_{TOT} . Then, substituting equations A10 and A11 into the Bernoulli equation yields the magnitude of the velocity,

$$|\mathbf{v}| = \sqrt{\frac{2}{\rho} (P_i - P_{AV}) (C_{STAT} - C_{TOT})}$$
(A12)

The orthogonal components of the velocity vector can then be calculated from equations A1, A2 and A3.



Figure A1 Representation of angular co-ordinate systems. $\alpha =$ pitch angle; $\beta =$ yaw angle; $\theta =$ cone angle; $\phi =$ roll angle.



Figure A2 Typical low-angle calibration data for seven-hole probe. α = pitch angle; β = yaw angle.