Experimental Testing of Low Reynolds Number Airfoils for Unmanned Aerial Vehicles

by

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A thesis submitted in conformity with the requirements for the degree of Master of Applied Science
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Abstract

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This work is focused on the aerodynamics for a proprietary laminar flow airfoil for Unmanned Aerial Vehicle (UAV) applications. The two main focuses are (1) aerodynamic performance at Reynolds number on the order of 10,000, (2) the effect of a conventional hot-wire probe on laminar separation bubbles. For aerodynamic performance, pressure and wake velocity distributions were measured at $Re = 40,000$ and $60,000$ for a range of angles of attack. The airfoil performed poorly for these Reynolds numbers due to laminar boundary layer separation. 2-D boundary layer trips significantly improved the lift-to-drag ratio. For probe effects, three Reynolds numbers were investigated ($Re = 100,000, 150,000, \text{ and } 200,000$), with three angles of attack for each. Pressure and surface shear distributions were measured. Flow upstream of the probe tip was not affected. Transition was promoted downstream due to the additional disturbances in the separated shear layer.
Dedication

To my maternal grandfather, who has always wanted to have a scientist in the family.
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Chapter 1

Introduction

1.1 Background

Ever since the Wright brothers first took to the sky in 1903, development of aviation technology has been driven relentlessly by mankind’s desire to fly better, faster, higher. Combined with advances in materials and propulsion, the post-WWII era saw the greatest boom to aviation research and industry in its short history. The Cold War saw an explosion of research into high speed aircraft, and as a result, much attention was paid to the development of high-performance high Reynolds number airfoils. With the development of automated control technology and micro-fabrication, however, smaller unmanned aircraft are steadily gaining popularity [40]. These Unmanned Aerial Vehicles (UAVs) offer unparalleled advantages in manoeuvrability, endurance, flexibility, and safety over their manned counterparts. As aerial missions are becoming more complex and dangerous in both the civilian and military sectors, the importance of these UAVs are becoming more evident. Micro-Aerial Vehicles (MAVs) are an area of particular interest for development because of their abilities to performing many types of mission under different
environments.

Because of the relatively low speed or high altitude that these aircraft often operate in, the flow they experience is mostly in the low Reynolds number regime, between $10^4$ and $10^6$ [40]. Figure 1.1 shows a spectrum of low Reynolds number airfoil applications. As it can be seen, the most prevalent applications within these Reynolds numbers are for radio-controlled (RC) model aircraft. Most RC airfoils were designed from empirical data gathered from trial-and-error experiments. Although there has been excellent examples of high-performance airfoils for low Reynolds number that uses numerical flow solvers coupled with an optimization method [14, 48, 52], there is a general lack of systematic design methods for such airfoils compared to their high Reynolds number counterparts. The main difficulty is the accurate and efficient modelling of flow at low Reynolds numbers. The aerodynamic design of such small aircrafts cannot simply be scaled down from larger ones. Flow characteristics at low Reynolds numbers are vastly different from those found in high Reynolds numbers. The main difficulty associated with analyzing flow in this regime is laminar flow separation. In high Reynolds number flow over an airfoil, the onset of adverse pressure gradient caused by the suction side pressure recovery is typically after boundary layer transition. The flow is therefore more resistant to separations. The same is not true for low Reynolds number flow, where the boundary layer usually remains laminar for a significant portion of the airfoil chord length. In the extensive summary of by Carmichael [7], airfoil lift-to-drag ratio ($L/D$) at low Reynolds number is often very poor compared to $L/D$ at high Reynolds numbers. Despite significant efforts made to design high-performance low Reynolds number airfoils, particularly those done by Lissaman [31], Olejniczak and Lyrintzis [48], and Selig and Guglielmo [52], low Reynolds number airfoil design remains an non-rigorous process due to the lack of reliable computational fluid dynamic (CFD) predictions and experimental data.

One of the most important hurdles in designing low Reynolds number airfoils is the
accurate modelling of the behaviour of laminar separation bubbles (LSBs). They are formed when the separated laminar shear layer undergoes transition and reattachment, forming a pocket of recirculating air. The effect of LSBs on airfoil performance is debated [37]. They keep the flow attached to the airfoil by transferring momentum from freestream to the boundary layer, but on the other hand, separation bubbles disrupts the flow over the airfoil by modifying its inviscid shape, causing additional drag. The balance between drag caused by the bubble and skin friction drag caused by turbulent boundary layer is important. Figure 1.2 shows the general trend of $L/D$ of smooth and rough airfoils over a spectrum of Reynolds numbers. It can be seen that at lower Reynolds number, rough airfoils have better performance due to attached turbulent boundary layer. As the Reynolds number increases, however, the additional skin friction drag becomes significant, causing their performance to be inferior to that of smooth airfoils. Mitra et al. [37] suppressed the formation of LSBs through leading edge boundary layer tripping, and concluded that in some cases, eliminating LSBs by forcing leading edge transition offers no overall drag reduction because of the additional skin friction. There is clearly a trade-off between reducing LSB size and maintaining laminar boundary layer. The understanding of the transition process in the separated shear layer is crucial to predicting the formation,
size, and behaviour of separation bubbles. There is currently a lack of consensus on the effects of different transition processes on the evolution of bubble formation. Hot-wire anemometry is a popular experimental method for investigations into transition because of its small spatial size and high temporal resolution. It is, however, an intrusive method. The effect of a hot-wire probe on LSBs is debated, and there has been no attempt to date to quantify any potential effects that the probe may have.

1.2 Objectives

The present work is based on previous experiments performed by Naghib-Lahouti [42]. The author conducted aerodynamic performance tests on a custom-designed laminar flow airfoil for Reynolds numbers ranging from 150,000 to 330,000. Based on the results obtained by Naghib-Lahouti, it was decided that laminar separation bubbles will be examined more closely for more Reynolds numbers. Hot-wire anemometry was chosen as the primary method for bubble characterization. Before such investigation could be
made, however, the effects of the probe itself on the bubble must be documented. As there is currently a lack of quantitative studies on such effects from the probe, two main objectives arose for the present work. The first is to extend the available data on aerodynamic performance and bubble formation to lower Reynolds number range, on the order of $10^4$. The second objective is to systematically investigate the effect of hot-wire probe on laminar separation bubbles. For the first objective, tasks include (1) identifying the separation, transition, and reattachment location through pressure distribution and (2) investigating the effectiveness of boundary layer tripping on improving aerodynamic performances for Reynolds number on the order of $10^4$. The results will aid future numerical models in transition prediction under free flow and forced transition conditions. The second objective will examine any potential effects of the presence of a hot-wire probe within LSBs on pressure and surface shear stress distributions. Attention will be paid to the flow upstream of the probe. Oil film interferometry, an optical measurement method, will be used for measuring shear stress.

The thesis is structured as follows. Chapter 2 reviews previous studies into the mechanics of laminar separation bubbles, the history and current state of low Reynolds number airfoil development, and an introduction into oil film interferometry. Chapter 3 details the setup of each experiment and the design of test spaces and experiment procedures. Results are presented in Chapter 4, which looks at aerodynamic performance improvements made by boundary layer tripping, as well as the effects of hot-wire probe on pressure and wall-shear stress distribution within the separation bubble. Chapter 5 summarizes the conclusion made from the results, and outlines proposed future investigations.
Chapter 2

Literature Review

2.1 Laminar Separation Bubbles

2.1.1 Formation and Dynamic Behaviour

In a typical flow over an airfoil, there is a point of minimum pressure, or suction peak, near the leading edge on the upper surface caused by accelerated flow over the suction (upper) side. After the peak, pressure gradually increases as the flow travels downstream, until it recovers to approximately the static pressure of the ambient air at the trailing edge. This gradual increase in pressure, called pressure recovery, causes an adverse pressure gradient to act on the boundary layer over the upper surface. In high Reynolds number flow, the boundary layer typically transitions before or right after the suction peak. The resulting turbulent boundary layer is often energetic enough to resist the adverse pressure gradient from causing it to separate. In low Reynolds number, however, the boundary layer is laminar over an extensive portion of the airfoil, which makes it therefore prone to separation. The separated boundary layer then mixes with the inviscid freestream,
Chapter 2. Literature Review

Figure 2.1: Structure of a laminar separation bubble, taken from Häggmark et al. [23]

forming a shear layer. This shear layer is highly unstable, which eventually results in transition. If transition happens quickly enough after separation, the boundary layer will reattach onto the airfoil, forming a recirculating region of low pressure air called laminar separation bubble (LSB) [20, 31]. This bubble modifies the effective shape of the airfoil, creating a “bump” as seen by the inviscid flow. Figure 2.1 shows the structure of a LSB.

LSBs are generally categorized into “short” and “long” types. A short LSB will typically only cause a local change to the pressure distribution as its length is only a few percent of the airfoil chord. A “flattening” of the pressure, followed by a steep recovery after transition, is usually observed. After reattachment, the pressure recovers to the same values as it would have in the absence of a bubble [7, 31]. A long LSB, in contrast, typically covers over 20 – 30% of the airfoil chord. This will cause a significant modification to the effective shape of the airfoil as seen by the outer inviscid part of the flow [31]. A long LSB will significantly reduce the peak suction pressure, and therefore the lift, of the airfoil [7]. Figure 2.2 shows the effect of long and short bubbles on the pressure distribution compared to flow without bubbles. The separation point is identified as the start of the low pressure plateau, transition as the onset of rapid pressure recovery, and
Figure 2.2: Effects of long and short bubbles on $C_p$ distributions. “S” and “R” denote separation and reattachment respectively. Taken from Gaster [20].

reattachment as the recovery of $C_p$ to a value that an attached turbulence boundary layer would obtain. As the angle of attack increases, LSBs will generally decrease in length and move upstream. At large angles of attack, however, the steep pressure recovery might cause an adverse pressure gradient strong enough to burst a short bubble into a long one. Gaster [21] characterized this phenomenon as “nose stall”. The bursting of the bubble will bring a very abrupt deterioration of the airfoil performance. Furthermore, if the angle of attack is reduced after bursting, a short bubble will not be immediately recovered. This is the cause for the hysteresis effects seen in airfoils at low Reynolds number [31].
2.1.2 Classification

The long and short categorization was first proposed by Owen and Klanfer [49]. The authors suggested using \((Re_\delta^*)_S\), which is the Reynolds number based on boundary layer displacement thickness at separation, as a parameter for long and short bubble distinction. A value of \((Re_\delta^*)_S\) between 400-500 is suggested by Owen and Klanfer [49] for short bubbles, and values below that for long bubbles. Tani [58] examined the upstream influence of the two types of bubbles in addition to the criteria proposed by Owen and Klanfer [49]. Long bubbles were classified as those that have a global effect on pressure distribution, namely reduced suction peak, while short bubbles effects are local. Gaster [20] suggested using the momentum displacement thickness at separation, \(\theta_s\), instead of \(\delta_S^*\) as the length scale. He also introduced a new parameter, \((\theta_S^2/\nu)(\Delta U/\Delta x)\), in addition to \(Re_{\theta_S}\) for a two-part criteria in determining the bursting of short bubbles into long ones. \(\Delta U/\Delta x\) is the average velocity gradient across the length of the bubble in the inviscid flow. In the numerical simulation of Pauley et al. [50], the authors found that bubbles considered short using Gaster’s criteria exhibited vortex shedding behind the reattachment point, causing it to oscillate at a regular frequency. Long bubbles, on the other hand, showed no sign of unsteadiness. Working along these results, Pauley et al. [50] classified short bubbles as having vortex shedding in their wake, whereas long bubbles do not exhibit that characteristic. These findings, however, were challenged by the experimental work done by Watmuff [62], who found no sign of vortex shedding behind a short bubble unless external excitation was applied. The difference was attributed by Watmuff to the modelling of adverse pressure gradient through suction that caused excitations to the bubbles in Pauley et al. [50]. It should, however, be noted that the works by Watmuff and Pauley et al. were performed on flat plates. Two-dimensional vortex shedding has been reported on airfoils [29, 65], although large vortex formation is inhibited by the shear layer’s close proximity to the surface [6]. In the DNS analysis by
Alam and Sandham \[1\] on short bubbles, the authors found that convective instability dominates the separated shear layer. Based on this, Alam and Sandham \[1\] suggested equating absolute instability with long bubbles and convective instability with short ones. This, however, is inconsistent with the slow approach to reattachment found in long bubbles \[35, 36\]. Up to date, there has been no consensus on the classification criteria for short and long bubbles. Recently, Diwan et al. \[10\] proposed a pressure gradient parameter term defined as

\[
P = \frac{h^2 \Delta U}{\nu \Delta x}.
\]

Here \(h\) is the maximum bubble height, \(\nu\) is the kinematic viscosity of air, and \((\Delta U/\Delta x)\) is the same average velocity gradient used by Gaster. This parameter gives a measure of the global nature of the bubble, instead of the localized criteria used in previous studies. It will be used in the present study to distinguish between the different bubble types.

### 2.1.3 Transition Mechanisms

Most studies have shown that the dominant mode of transition in short bubbles is Kelvin-Helmholtz (KH) instabilities in the free shear layer \[6, 24, 26, 35, 65\]. Häggmark et al. \[23\] found that the growth rate of wave disturbances within the shear layer is exponential, and that the amplitude of growth is in excellent agreement with DNS and classical linear instability theory. The exponential growth dominates the flow until reattachment, where coherent 3-D structures form. These structures appear to have the same spanwise wavelength as the wavelength of the 2-D wave disturbances. Above the shear layer, there is vortex shedding. Jones et al. \[29\] through DNS analysis performed on a NACA-0012 airfoil described the self-sustaining mechanism of the shedding process. Inside the
2-D vortices, small 3-D perturbations are created. These perturbations are amplified within the vortex before being convected upstream by the reverse flow. They are then convected to the next developing vortex, where the perturbations are further amplified, and hence repeating the cycle. Yarusevych et al. 65 showed that these vortices roll up from the separated shear layer, where they are then shed. Yarusevych et al. 69 further found that shedding occurs within a band of frequencies centred around a fundamental frequency. The vortex roll-up from the shear layer marks the final stage of transition 69. Although studies have shown that transition mechanism is different between long and short bubbles 35, the studies on short bubble transition greatly outnumber those on long bubbles. While short bubble transition is relatively well understood, it is unclear as to why long bubbles, under the same flow conditions, are generally slow to reattach even after transition 35, 36. Both numerical and experimental studies have been attempted, but they have not been successful in detailing the transition process governing long bubble. This is due to difficulties in maintaining a steady long bubble in laboratory conditions 62 and convergence issues in CFD modelling 35.

2.2 Low Reynolds Number Airfoils

2.2.1 History and Categorization

Low Reynolds number airfoils typically have natural laminar flow over most, if not all, of the chord length. These airfoils are primarily used in hand-launched gliders, hobby model aircrafts, and ultra-light hand gliders. Natural laminar flow also occurs for most birds in flight. Recently, both the civilian and military sectors have shown an increased interest in the development of these low Reynolds number airfoils, due to an increased usage of UAVs that operate at low speed or high altitude. In the extensive summary of the history of low
Reynolds number airfoil development by Carmichael [7], the author divided the Reynolds number spectrum into twelve regions, covering from fractional Reynolds numbers to over $10^9$. Five of those regions, ranging from 10,000 to 3,000,000 will be examined here.

From 10,000 to 30,000, complete natural laminar flow exists over the airfoil in most situations. In the free flight testings conducted by Bauer [3] on hand-launched glider models, it was found that the total drag coefficient with induced drag removed agrees with drag coefficient of laminar flow over a plate. The lift coefficient generated at these Reynolds number is relatively low, at 0.5 or less. The drag coefficient is also very low, however, due to the complete laminar flow over the wing. Trimming the model to a higher lift coefficient would cause laminar flow separation without reattachment, resulting in a marked increase in drag coefficient. It is difficult at this regime for the flow to transition, even with artificial tripping. A representative aircraft operating within these Reynolds number is a small rubber band powered model plane.

The region between 30,000 and 70,000 is when laminar separation bubble first appears. From the collected airfoil performance data, Carmichael [7] was able to estimate the distance between separation and reattachment, and expressed it as $Re_R - Re_S \approx 50,000$, where $Re_R$ and $Re_S$ are chord Reynolds number at reattachment and separation respectively. Thus generally speaking, for flow under 50,000, the airfoil is not long enough for a separated laminar boundary layer to transition and reattach. Many high-performance small model aircrafts operate within these Reynolds numbers. It is also interesting to note that future possible exploration of Mars by an aircraft would be flying at these Reynolds numbers [18].

From 70,000 to 200,000, airfoil performance markedly improves due to the elimination of laminar flow separation at low angles of attack. Boundary layer tripping can further improve the performance by promoting transition to decrease the size of separation
bubbles. This effectiveness decreases as the flow approaches the upper bound of this region. Within the next Reynolds number region between 200,000 and 700,000, laminar separation bubbles readily appear in most flow conditions. Boundary layer tripping loses its effectiveness. Typical examples of objects in these two Reynolds number regions are bats, large birds, large RC aircrafts, and hand-gliders.

The last flow region has Reynolds number ranging from 700,000 to 3,000,000. There are extensive experimental data for this region from NACA/NASA investigations performed post-WWII. Here the effects of laminar separation bubble are steadily reduced, until it is negligible toward the upper end of this region. Carmichael [7] considers this to be the upper bound for the operating environment of specifically designed low Reynolds number airfoils.

### 2.2.2 Flow Characteristics

The flow behaviours at low Reynolds number are fundamentally different from those at high Reynolds numbers [31, 38, 51]. Flow over airfoils at Reynolds number greater than 3 million is generally well known. The boundary layer remains attached over the airfoil, and typically transitions at around the mid-chord point. This transition and the flow behind it are generally well behaved [51]. When the boundary layer is extensively laminar, such as in the case of low Reynolds number flow, it is prone to separation caused by adverse pressure gradient. Because the transition process in the separated shear layer is highly sensitive to initial conditions, airfoil geometry is a large influencing factor in separation bubble formation and behaviour. Generally speaking, an airfoil designed for high Reynolds number have very poor $L/D$ in the $10^4 - 10^5$ Reynolds number range. In the comparative study of Counsil and Boulama [9], the NACA 0012 airfoil was found to have very poor aerodynamic performance compared to the Selig-Donovan 7003 airfoil,
which is optimized for low Reynolds number flight. The unsteady behaviour of the separated shear layer also causes some unique characteristics of aerodynamic performances at low Reynolds numbers. In the series of extensive wind tunnel testings done by Selig \cite{53, 54, 55, 56, 57}, it was found that in many instances between Reynolds number of 60,000 to 100,000, coefficient of lift ($C_l$) would increase slowly with angle of attack, much less than the predicted slope of $2\pi$ from inviscid theory. Upon reaching a critical angle before stall, the value of $C_l$ would suddenly increase, and its slope would be closer to $2\pi$. On the other hand, drag coefficient $C_d$ would increase rapidly at low angles of attack, and upon reaching the same critical angle, it would drop significantly. The resulting lift-drag polar exhibits a “zig-zag” shape as opposed to the C-shape commonly seen in high Reynolds number airfoils. An example of such a polar is shown in Figure 2.3. This phenomenon is caused by laminar flow separation and reattachment. At low angles, the boundary layer separates without reattachment. This causes the airfoil to be in a post-stall like state due to the massive flow separation, causing the slow increase in lift coefficient with angles of attack. The wake is also much wider than that of an attachment flow, causing more momentum loss and therefore an increased drag coefficient. As the angle of attack increases further, the separated shear layer becomes more unsteady, and in certain cases could transition and reattach itself unto the surface, forming a separation bubble. This causes the wake to decrease in width and revert the airfoil back to a pre-stall state, thus significantly increasing the aerodynamic efficiency. As the Reynolds number increases, transition in the shear layer occurs fast enough that a separation bubble forms even at low angles of attack, thereby eliminating the “zig-zag” pattern seen in the drag polar. This type of flow phenomenon has been observed visually through smoke wire tests in the studies by \cite{67, 68}.

The accurate prediction of laminar shear layer transition is thus a main design difficulty for low Reynolds number airfoils. Compared to transition in attached flows, there has been relatively little study into separated shear layer transition \cite{53}. While excellent low
Reynolds airfoils have been designed, notably the Selig S1223 [52] and the work done by Olejniczak and Lyrantzis [48], which was based on the optimization method by Liebeck, the prediction of their dynamic behaviour in real world operation is still lacking because of the difficulties in accurately modelling laminar separation bubbles. Therefore a detailed experimental characterization of separation bubbles, coupled with improvements in numerical estimation methodologies are crucial for the future development of low Reynolds number airfoil design.

### 2.2.3 Flow Control Schemes

Many studies have been done to examine various control schemes for laminar separation bubbles. The most popular method is passive boundary layer tripping devices such as trip wire [4, 7, 22, 27, 39], discrete bumps [7, 8, 30, 70], and distributed surface roughness [7, 41].
Zhou and Wang [70] applied a 4th order spectral difference Navier-Stokes Equations Solver to a SD7003 airfoil with discrete rounded bumps located near the leading edge. The Reynolds number was 60,000. In the study, the height and width of the bumps were varied to parameterize their effects on bubble formation. Compared with the smooth baseline airfoil, tripped airfoils had an overall reduction of separation bubble size. In some cases the bubble was attenuated completely. Lift-to-drag ratio increased up to 19%. Higher bumps were able to introduce more disturbances into the boundary layer, inducing an early onset of transition, and thereby more effective in reducing the effects of separation bubbles. The width of the bumps did not have as much influence. It was also found that the bumps were more effective at higher angles of attack, although only three angles, 2°, 4°, and 6°, were tested. 2-D trip wires are historically shown to be effective at improving airfoil performance at low Reynolds number as well. According to Carmichael [7], a rectangular trip measuring 0.3 mm in height by 2 mm in width located at 20% chord was able to increase the aerodynamic efficiency of the Wortmann FX 63-137 airfoil by 55% at $Re_c = 60,000$. Furthermore, trips located in front of 10% chord generally were able to significantly improve airfoil performance even for Reynolds numbers as low as 40,000. Distributed surface roughness showed improvements in airfoil performance as well. In the study by Mueller and Batill [41], $C_l$ increased more rapidly at $Re = 40,000$ with the use of a grit textured surface taped unto the leading edge of a NACA 663−018 airfoil. Although there was a penalty of decreased maximum lift. The grit also prevented flow separation at low angles of attack, making the airfoil better behaved at these angles.

External acoustic excitation is another popular method of experimental laminar separation control on an airfoil. In the aforementioned study by Mueller and Batill [41], the authors compared the effectiveness of acoustic excitation with surface grit textures over a range of acoustic frequencies. It was found that for $AoA = 1^\circ$ and $Re_c = 130,000$, there was an optimal excitation frequency at 388 Hz. Excitation at this frequency showed almost identical performance in lift coefficient improvements as the grit texture. Yaru-
sevych performed various more detailed studies into the effects of external acoustic excitation \cite{64,63,66}. In these studies, the optimal excitation frequency was found to be the fundamental frequency of the most dominant natural disturbances within the separated shear layer. Yarusevych et al. \cite{66} showed that the amplitude of such excitation is an influencing factor as well in laminar separation bubble development. Testing on a NACA 0025 airfoil at two Reynolds numbers of 100,000 and 150,000, it was found that there exist a minimum threshold, which was linked to background noise, for the acoustic wave amplitude to take effect. Once the threshold was exceeded, the acoustic excitation was able to delay laminar separation and promote transition for an early onset of reattachment, leading to an overall reduction in the size and influence of laminar separation bubbles. There also exist, however, a maximum amplitude level where the disturbance saturates the receptivity of the separated shear layer. In Yarusevych et al. \cite{64}, acoustic excitation was coupled with mechanical vibration to produce even greater attenuation of laminar separation bubble effects. In some test cases, the overall drag coefficient was reduced by 64%.

### 2.2.4 Experimental Difficulties

The difficulties with low Reynolds number airfoil experiments are mainly attributed to the sensitivity of laminar boundary layers to external factors such as freestream turbulence level, acoustic excitation, and surface finish. For most experiments, the dominant environmental factor is freestream turbulence level. Watkins et al. \cite{61} studied its effects on bubble formation on a flat plate with contoured leading edge and a sharp trailing edge. It was found that a turbulence intensity of 7.4\% was able to eliminate the formation of laminar separation bubbles on the flat plate airfoil. Post-stall behaviour are affected by freestream turbulence level as well. Marchman III et al. \cite{34} summarized four past studies on the aerodynamic hysteresis loop exhibited by the Wortmann FX 63-137 airfoil. Two of
the studies had disagreement on the size of the loop, while the other two reported no loop at all. The differences observed were attributed by Marchman III et al. [34] to variations in the flow environments. Hoffmann [25] investigated the effect of freestream turbulence on the lift and drag coefficient of a NACA 0015 airfoil. It was found that for an intensity of greater than 3%, hysteresis loop was eliminated, while below this threshold the loop grew in size as turbulence intensity dropped down. Airfoil surface finish have historically been a major source of experimental uncertainty. Mueller and Batill [41] observed asymmetric behaviour for a symmetry airfoil because of manufacturing defects. With modern computer-guided manufacturing techniques, however, these defects are becoming negligible. In regard to surface roughness, Traub [60] concluded that a finish equivalent to 400 grit sandpaper or higher can be considered aerodynamically smooth. This is well within the scope of modern manufacturing capabilities.

To understand the fundamental physics behind the behaviour of LSBs to external factors, one must characterize its natural transitional behaviour. Hot-wire anemometry is a popular method to study bubble transition because of its high temporal resolution and small spatial size. It remains, however, an intrusive technique. Brendel and Mueller [6] found that an approach angle of 10° or less with the tangent of the local surface produced negligible effects on the flow field. Similar results were also observed by Yarusevych et al. [68]. Elimelech et al. [17] used temperature-sensitive paint to qualitatively verify that the hot-wire probe used in their study had no noticeable effects on the flow. Häggmark et al. [23], however, cautioned on the use of hot-wire probes. The probe will need to be properly designed and be of an appropriate size for each experiment. This is particularly challenging for investigations of leading edge bubbles, where typical bubble could be a fraction of a millimetre in height and a few millimetres in length. The author, however, did not further elaborate on quantitative criteria for choosing appropriate probes for his study. Diwan and Ramesh [11] found through smoke flow-visualization that inserting a probe into the bubble modifies its shape and size, but no quantification of the probe
effects were made in the study either. To date, there is a lack of systematic investigation to quantify the effect of a hot-wire probe on LSBs.

2.3 Oil Film Interferometry

2.3.1 Theory

Oil film interferometry (OFI) is a non-intrusive analysis technique for direct shear stress measurements on a solid surface. A thin layer of oil film with known physical properties is applied to a model surface and then subjected to a flow. The motion of the oil film surface depends on shear stress, gravity, pressure gradient, surface curvature of the oil, and surface tension.

Squire \cite{56} derived the oil film equation relating the oil thickness distribution to shear stress and pressure gradient. He assumed that the motion of the oil film is governed entirely by the viscous equations of motion within the boundary layer. This assumption requires that the oil film be very thin so as to lie completely within the boundary layer. In most applications this is considered valid, where the oil film has a thickness on the order of 10 \( \mu m \) \cite{59}. Tanner and Blows \cite{59} derived their own equations for shear stress...
using a control volume approach, but it has been shown that they are equivalent to Squire’s formulations [47]. Although the motion of the oil film depends on many factors, Naughton and Sheplak [47] argued through order-of-magnitude analysis that for most applications, the dominant term is the surface shear stress. A detailed derivation of the oil film equation, culminating in a simplified version that depends only on surface shear stress, was published by Naughton and Sheplak [47]. Figure 2.4 shows the coordinate system and the control volume used for the oil film equation derivation. The resulting thin-film equation is reproduced below in (2.2), where \( h \) is the height of the oil film, \( \tau_{x,z} \) is the surface shear stress in the \( x \) and \( z \) direction respectively, and \( \mu_o \) is the oil’s dynamic viscosity.

\[
\frac{\partial h}{\partial t} = -\frac{\partial}{\partial x} \left( \frac{\tau_{x} h^2}{2\mu_o} \right) - \frac{\partial}{\partial z} \left( \frac{\tau_{z} h^2}{2\mu_o} \right). \tag{2.2}
\]

Both formulations of oil film equations proposed by Squire [56] and Tanner and Blows [59] require a time and space history of the oil film thickness. This can be obtained by taking advantage of the Fizeau fringe phenomenon [47, 57]. Figure 2.5 is a simple schematic showing the basic setup of an OFI experiment. An incident ray directed at the oil surface is split into two beams through refraction and reflection from the air-oil interface. The refracted beam is then reflected by the model surface, creating two reflected beams out of the oil surface with a phase offset between them that depends on the film height, \( h \). Equation 2.3 is reproduced from Born and Wolf [5]. It shows the relationship between film height \( h \), phase difference \( \phi \), source light wavelength \( \lambda \), and light incident angle \( \theta_i \). The index of refraction of film and air are denoted by \( n_f \) and \( n_a \) respectively.

\[
h = \frac{\lambda \phi}{4\pi} \left( \frac{1}{\sqrt{n_f^2 - n_a^2 \sin^2 \theta_i}} \right). \tag{2.3}
\]
Chapter 2. Literature Review

Figure 2.5: A basic schematic diagram of an OFI experiment setup. The essential components shown are: model, oil droplet, light source, and camera. This is reproduced from Stimson and Naguib [57].

The interference creates alternating light and dark bands called fringe lines. The image of the fringe lines are referred to as interferograms. By tracking the fringe lines, oil film height can then be calculated. This is the working principle behind the oil film interferometry technique. Figure 2.6 shows a schematic of the physical phenomenon, while Figure 2.7 shows an example of fringe lines forming near the reattachment point of a laminar separation bubble on an airfoil.

Tanner and Blows [59] proposed a simplified equation and technique for measuring shear stress at a specific location. Small droplets or sheets of oil can be deposited at the desired location, and the shear stress can be assumed to be constant at that location. This assumption is good provided that the oil film measurements are taken over a sufficiently small area, and that there is no large model surface curvature and pressure gradient around the region of interest.

Stimson and Naguib [57] used multiple interferograms taken at regular time intervals to construct a composite image that shows the streamwise progression of fringe line development at a single spanwise location. Since the flow was considered 2-D within the region of interest, a single location on the oil film that is free from surface contamination,
such as dust or air bubbles, can be chosen for analysis. The required time and spatial plot of the oil height was developed from the composite image, and by applying the constant shear assumption, an analytical relation was developed between surface shear and the velocity of the fringe lines. Using this simplified approach, Stimson and Naguib [57] showed good agreement of skin friction measurements with previous studies on a fenced flat plate. Driver [12] also used the constant shear assumption in his study of a 3-D swept wing model. Instead of having multiple snapshots, however, he used a single-image approach that required a temporal history of the freestream dynamic pressure and temperature. This method is advantageous for test facilities that do not have easy optical access, as only a single image needs to be captured immediately after tunnel shutdown. It should be noted that for both methods, the startup and shutdown time of the tunnel need to be relatively short compared to the test running time, as the unsteadiness during startup and shutdown violates the assumption of constant shear.

The simplifications made by Tanner and Blows [59] cannot be easily applied to test cases where a shear gradient exists. One could obtain approximate measurements by performing multiple tests, each over a small area, that span the entire region of interest.
This, however, is a very time-consuming method and does not work practically if the shear gradient is large, such as near regions of flow separation or reattachment. Furthermore, if the surface has significant curvature, then the simple post-imaging analysis by Stimson and Naguib [57] cannot be applied. Photogrammetry would have to be used to correctly map the image space to physical space. These two complications, shear gradient and surface curvature, are typically present in airfoil tests. Therefore the aforementioned constant shear simplifications are not applicable to OFI tests on airfoils.

To extend OFI to account for variable shear, Garrison and Ackman [19] developed what they called a “Global Interferometer”. This method uses an iterative solution, where the shear stress at one fringe line is based on the solution at the previous line. The initial solution is still based on a constant shear assumption near the leading edge of the oil film, but this method allows for measurements over a much larger area of interest with one interferogram. Naughton et al. [46] and Naughton and Liu [45] non-dimensionalized Garrison and Ackman’s equation to calculate the skin friction coefficient, $C_f$, in the form of

$$C_{f,i+1}^{\frac{1}{2}} = \frac{\int_0^{x_i} (n/C_{f,i})^\frac{1}{2} dx}{h\sqrt{n} \int_0^{i} (q/\mu) dt}, \quad (2.4)$$

Here $q$ and $\mu$ are the freestream dynamic pressure and oil viscosity respectively, and $n$ is streamline divergence, which is set to unity in a 2-D flow. The index $i$ denotes the $i^{th}$ fringe line. The initial solution is provided by

$$C_{f,1} = \frac{\mu x}{q h t}. \quad (2.5)$$

It can be seen that by using Eqs. 2.4 and 2.5 shear stress can be calculated at any fringe
line located at $x$, at any time $t$ as long as the integral $\int_0^t (q/\mu) dt$ can be calculated. By using this global approach, the restriction of constant shear imposed over the oil film is removed, and thus allows for a single interferogram to cover a large region, even one that includes flow separation and reattachment.

As mentioned before, photogrammetry is often used in test cases where the surface geometry is complex, or the camera is not perpendicular to the surface \[13, 16, 28, 43, 46, 45\]. Essentially, the purpose of photogrammetry is to map the image space onto physical space. Naughton and Liu \[45\] provided a detailed high-level description of the process. There are two main sets of information that need to be determined: camera interior parameters and exterior orientation. In Naughton and Liu, the camera lens is modelled as a single point called the perspective centre, which has coordinates $(X_c, Y_c, Z_c)$ with respect to the model space. Its rotation is represented by the Euler angles $(\omega, \phi, \kappa)$, which are essentially the pitch, roll, and yaw of the camera. These six parameters, referred to as the exterior parameters, completely define the camera orientation with respect to the model. On the other hand, the interior parameters include the camera principle distance $c$ and the photogrammetric principle-point location $(x_p, y_p)$. Figure 2.8 shows these parameters in relation to the image and physical space.

In addition to these parameters, non-linearity effect parameters due to camera lens distortion need to be taken into account. For a digital image, the pixel cell size and ratio of horizontal to vertical pixel spacing $S_h/S_v$ are needed \[44\]. To obtain these parameters, Naughton \[44\] proposed a two-stage approach to the analysis. First, a highly 3-D object with known markers is used to determine the lens distortion and camera interior parameters. The object should fill the entire image space to minimize errors when calculating for these parameters. An example of a suitable 3-D object is shown in Figure 2.9.

Next, with the camera’s interior setting locked in place, the exterior parameter would be determined by taking an image of the model with known physical markers. These
markers could be inherent on the model, such as pressure ports, or an added layer of grid pattern. The exterior parameters could then be calculated by mapping the image of the model markers onto physical space, through the interior and lens distortion parameters calculated before. Naughton et al. [46] used this two-stage approach in addition to an optimization method developed by Liu et al. [32]. The optimization refines the estimates of the parameters through an iterative method.

### 2.3.2 Practical Considerations

There are some practical considerations for obtaining high quality interferograms. The first is model surface illumination. From Eq. 2.3, it can be seen that the source light wavelength, $\lambda$, is needed to calculate oil film height. The best sources are monochromatic light, as a constant value of $\lambda$ simplifies film height calculation. Commonly used lights includ metal-vapour bulbs, such as sodium and mercury vapour lamps [12]. If monochromatic lights are unavailable, coloured filters could be used. The illumination also must be from a diffuse source to ensure even lighting throughout the region of interest. Glares
Table 2.1: Characteristics for surfaces used in OFI, reproduced from Naughton and Sheplak [47]

<table>
<thead>
<tr>
<th>Material</th>
<th>Quality</th>
<th>$n_s$</th>
</tr>
</thead>
<tbody>
<tr>
<td>SF 11 glass</td>
<td>Excellent</td>
<td>1.78</td>
</tr>
<tr>
<td>Mylar film</td>
<td>Good</td>
<td>1.67</td>
</tr>
<tr>
<td>Nickel</td>
<td>Good</td>
<td>1.6-2.0</td>
</tr>
<tr>
<td>Polished stainless steel</td>
<td>Acceptable</td>
<td></td>
</tr>
<tr>
<td>Polished aluminum</td>
<td>Poor</td>
<td>1.44</td>
</tr>
</tbody>
</table>

in interferograms reduce the contrast of the fringe lines. White paint [12], diffuser filters [57] and even model surface treatment [47] have been used in studies in the past. Practically speaking, any measure that prevents specular reflection on the model surface is sufficient.

The most commonly used type of oil is the Dow Corning 200 silicone oil [12, 13, 16, 28, 47, 43, 46, 45]. The oil is available in many different viscosities. The choices of which are dependent upon experiment temperature, freestream velocity, run time, and the desired spacing between fringe lines [12]. The oil viscosity requires calibration with respect to temperature. Since oil viscosity uncertainty constitutes a large part in the overall uncertainty, it is highly recommended that the viscosity be calibration with a viscometer [47, 43, 57]. Oil application could be done as lines or drops of oil with a flat edge or a point/dropper respectively [43]. Generally speaking, drops of oil are for qualitative measurements, while lines of oil are for quantitative measurements [46].

Model surface is also an important aspect of obtaining high quality interferogram. According to Zilliac [71], the theoretical optimum surface refractive index, $n_s$, is 2. Naughton and Sheplak [47] compiled a list of commonly used material for models, which is reproduced in Table 2.1. Black Mylar film is a popular surface treatment method because of its good surface qualities for OFI and ease of application.
Chapter 3

Experiment Details

3.1 Facilities and Setup

Measurements were made in the closed-loop wind tunnel at the University of Toronto Institute for Aerospace Studies. The tunnel has a test section measuring $1.2 \text{ m} \times 0.8 \text{ m} \times 5.0 \text{ m}$. The maximum freestream velocity is approximately $40 \text{ m/s}$. The freestream turbulence level measured at $10 \text{ m/s}$ is less than 0.05%. Figure 3.1 shows a picture of the wind tunnel’s test section. For all tests, freestream dynamic pressure and temperature were acquired. A Pitot-static tube connected to a MKS Type 225 Baratron differential pressure transducer was used to measure freestream dynamic pressure. Depending on the test Reynolds number, either the 1 torr or the 10 torr transducer was used. The accuracy of the freestream transducer was 0.5% of full scale. Temperature measurements were taken by a T-type thermocouple through a 16-bit National Instrument PCI-6259 A/D data acquisition card connected to a computer.

The airfoil section model tested was designed and constructed by Brican Inc., the cross-
section airfoil of which was the BE12037M10 laminar flow airfoil developed by Brian Eggleston [42]. It was a proprietary airfoil whose main design objective was to maintain a laminar boundary layer for as long as possible in order to minimize friction drag. The extensive laminar boundary layer made this airfoil susceptible to the formation of laminar separation bubbles over a wide range of Reynolds numbers, which made it an ideal airfoil for the purposes of this study. The model was constructed in two aluminum halves, encasing stainless steel tubings for 40 pressure ports. It has a span of 0.736 m and a chord length, $c$, of 0.254 m, which gives an aspect ratio of 2.9. The top of the model was secured to the ceiling of the test section through a custom-made aluminum plate that replaces one of the ceiling glass panels. An aluminum baseplate was used to support the model from the floor of the test section, as it could not be directly mounted through the glass floor of the tunnel. A rectangular endplate was added to the bottom of the model assembly to improve the two dimensionality of the flow at the lower portion of
the model. A Lin Engineering 8618M stepper motor coupled with a Thomson Micron 40:1 gearbox was used to change the angle of attack. The total angular resolution of the motor-gearbox system was approximately $0.01^\circ$ per step. Motion control was done through a 4-channel controller manufactured by The Motion Group. The controller was interfaced with Matlab. The 40 pressure ports are located at 58% of the span, measuring from the bottom of the model. The tubings for these ports extend out of the model from the top. A custom-built pressure transducer array was used to take pressure distribution measurements. The array contains 40 Freescale MPXV7002 0-2 kPa differential pressure transducers. After in-house recalibration of the array, the error in pressure measurements was estimated to be 1%. Pressure data acquisition was performed through a 16-bit Measurement Computing DaqBoard/3035USB data acquisition card, which allowed simultaneous sampling of all 40 transducers.

Wake velocity profiles were measured by a Pitot-static tube mounted on a traverse. The tube was connected to a MKS Type 120AD Baratron differential pressure transducer, which has an accuracy of 0.12% of the reading. Data acquisition was performed through the same system as the freestream dynamic pressure data. Depending on the Reynolds number and angles of attack, either a Velmex 12” X-Slide or a Velmex 24” B-Slide was used for spanwise traversing. The measurements were taken at 4c downstream from the trailing edge of the model, as per the suggestion by El-Gammal et al. [16]. The Pitot-static tube was positioned at the same height as the pressure ports so that the wake velocity profiles were measured at the same plane of measurement as the airfoil pressure distributions.

Measurements of the boundary layer profile were made with a Dantec conventional hot-wire probe, with a 5 $\mu$m copper-plated tungsten wire with a sensing length of 1 mm. Hot-wire velocity data were acquired with a constant temperature anemometer operating at an overheat ratio of 0.6. Velocity calibration was performed in-situ against a pitot-static
tube connected to a temperature compensated Baratron 120AD 0-1 torr pressure trans-
ducer. A calibration was performed before and after each test run. The final calibration
coefficients were calculated from the average of the two calibration runs. King’s law was
fitted to 14 calibration velocities between 1 m/s to twice the test freestream velocity.
Hot-wire data were acquired at 5 kHz for 5 s through the PCI-6259 A/D data acquisition
system. A T-type thermocouple was mounted in the freestream to provide measurements
for temperature correction of the hot-wire data. The typical tunnel temperature fluctu-
ation during a test run was ±0.5°C. The overall uncertainty in the mean velocity profile
is estimated to be less than ±1%. The hot-wire probe was mounted on a traverse system
with three degrees of freedom. The linear and angular resolutions were 60 µm and 0.2°,
respectively. The linear traverses were controlled through a computer, while the rotation
stage was set manually. The traverse system can place the probe to within 6° of the
model surface tangent from 0.1c to 1c, for 0° ≤ α ≤ 9°. Surface reflection technique was
used to find the probe’s initial wall-normal location, which was on average 0.7 mm, or
0.28% chord. The uncertainty in the wall-normal location is estimated to be ±90 µm.
Figure 3.2 shows the setup of the model, along with the hot-wire traverse system.

Figure 3.3 shows a diagram of the setup for the OFI tests conducted. Xiameter PMX-200
20CS silicone fluid was used for the experiment. The viscosity of the fluid was calibrated
with respect to temperature with a viscometer. A sodium-vapour lamp was used as
the monochromatic light source, which has two dominant spectral lines at 589.0 µm and
589.6 µm. For practical applications, however, the light was considered to be monochro-
matic at a wavelength of 589.3 µm. The outside of the test section was covered with
matt-white Bristol Boards for light diffusion. Black Mylar film was used to coat the
surface of the model to enhance the visibility of the fringe lines [47]. A Stingray F125B
charged-couple device (CCD camera was used to acquire images of the oil film. The
camera has a picture size of 1292 pixels × 964 pixels and a cell size of 3.75 µm × 3.75 µm.
At maximum zoom, each pixel corresponds to an area of approximately 90 mm × 90 mm
Chapter 3. Experiment Details

3.2 Procedures

3.2.1 Aerodynamic Force Tests

The test matrix for aerodynamic forces was based on requirements from Brican Flight Systems. Airfoil performance was tested at very low Reynolds numbers. Table 3.1 lists the low Reynolds number tests performed.
Table 3.1: Test matrix for very low Reynolds number performance

<table>
<thead>
<tr>
<th>Test Number</th>
<th>Re</th>
<th>$\alpha$ range ($^\circ$)</th>
<th>Turbulator</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>40,000</td>
<td>2 to 10 in 2$^\circ$ increments</td>
<td>No</td>
</tr>
<tr>
<td>2</td>
<td>60,000</td>
<td>2 to 10 in 2$^\circ$ increments</td>
<td>No</td>
</tr>
<tr>
<td>3a</td>
<td>40,000</td>
<td>2 to 10 in 2$^\circ$ increments</td>
<td>Rect</td>
</tr>
<tr>
<td>3b</td>
<td>40,000</td>
<td>2 to 10 in 2$^\circ$ increments</td>
<td>Rd</td>
</tr>
<tr>
<td>4a</td>
<td>60,000</td>
<td>2 to 10 in 2$^\circ$ increments</td>
<td>Rect</td>
</tr>
<tr>
<td>4b</td>
<td>60,000</td>
<td>2 to 10 in 2$^\circ$ increments</td>
<td>Rd</td>
</tr>
</tbody>
</table>

Plastic strips were used as passive devices to investigate the effects of boundary layer tripping at very low Reynolds numbers. These are polystyrene strips manufactured by Evergreen. Two cross section geometries, half round and rectangular, were used. Figures 3.4 and 3.5 show the geometry and size of the two types of strips used. The strips were attached at 8.5% c as per the request of Brican, using carpenter’s glue.

For each test run, the wind tunnel freestream velocity was set to match the target Reynolds number to within $\pm$1%. Atmospheric pressure was obtained online from Environment Canada. For each new angle of attack, the flow was allowed to settle for 30 s before any data were taken. The wake traverse used for low Reynolds number tests was the Velmex 12” X-Slide, which covered a range of $\pm$0.6 chord. Wake measurements were taken at every 3 mm, which was approximately the width of the Pitot-static tube used. Airfoil pressure distribution measurements were taken manually with a single MKS Type 120AD Baratron differential pressure transducer, because the available pressure transducer array did not have sufficient accuracy at these low Reynolds numbers. Lift and moment coefficients were calculated from the pressure distribution, while drag coefficient was calculated from the normalized wake velocity profile. Wind tunnel corrections accounting for solid and wake blockages were applied in accordance with Barlow et al. [2].
3.2.2 Hot-Wire Effect Tests

The objective of the hot-wire tests was to investigate the effect of the presence of the probe on the formation and size of LSBs. This section will focus on the procedure measuring the effect of the probe on pressure distribution. Table 3.2 summarizes the test space, which covers low, medium, and high angles of attack for three Reynolds numbers. In these sets of tests, low angles of attack are considered to be around 0°, medium around 5°, and high around 9°. First, reference pressure distributions were measured for all three Reynolds numbers at angles of attack (AoA) from 0° to 9°, in 1° increments. The range and location of LSBs were identified for each AoA, and one angle was chosen for each of the AoA ranges for all three Reynolds numbers, resulting in nine test cases. For each Re and AoA range, the angle for which the pressure distribution exhibited the most distinctive trace of a LSB was selected for further investigation.

As mentioned in 2.1.1 the separation and reattachment locations of the LSBs for each test case were determined from the pressure distributions. The entire bubble was then divided into six equally spaced measurement locations, with separation and reattachment as the end points. For each location, a surface-normal boundary layer velocity profile was taken with the hot-wire system. Pressure distribution data were taken at every three boundary layer profile points. Figure 3.6 shows a diagram of the procedure.
Table 3.2: Test matrix for hot-wire probe effects on pressure distribution

<table>
<thead>
<tr>
<th>Test Number</th>
<th>Re</th>
<th>$\alpha$ (°)</th>
<th>LSB chord range ($x/c$)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>100,000</td>
<td>0</td>
<td>0.62 - 1</td>
</tr>
<tr>
<td>2</td>
<td>100,000</td>
<td>6</td>
<td>0.38 - 0.67</td>
</tr>
<tr>
<td>3</td>
<td>100,000</td>
<td>9</td>
<td>0.16 - 0.37</td>
</tr>
<tr>
<td>4</td>
<td>150,000</td>
<td>0</td>
<td>0.62 - 0.85</td>
</tr>
<tr>
<td>5</td>
<td>150,000</td>
<td>5</td>
<td>0.38 - 0.67</td>
</tr>
<tr>
<td>6</td>
<td>150,000</td>
<td>9</td>
<td>0.16 - 0.31</td>
</tr>
<tr>
<td>7</td>
<td>200,000</td>
<td>0</td>
<td>0.62 - 0.85</td>
</tr>
<tr>
<td>8</td>
<td>200,000</td>
<td>5</td>
<td>0.44 - 0.62</td>
</tr>
<tr>
<td>9</td>
<td>200,000</td>
<td>8</td>
<td>0.16 - 0.37</td>
</tr>
</tbody>
</table>

Figure 3.6: Schematic of the measurement locations for hot-wire pressure distribution effect tests. Each boundary layer profile contains 40 data points; pressure distribution was measured for every three points.
At the start of each new chord location for the hot-wire probe, a reference pressure distribution without the presence of the probe was taken first, in order to minimize pressure array sensor drift. The hot-wire traverse then moved the probe into the desired location relative to the airfoil. To measure the normal distance of the probe tip from the airfoil surface, the Stingray F125B CCD camera was used to take images of the probe and its reflection on the airfoil surface, which was polished to increase the specular reflection of the probe. Images for the first five wall-normal locations were captured. To measure the position of the probe, the location of the tip in pixel coordinates was first identified on each of the five images. Because the physical distance of the movement of the tip is known, the conversion from pixel distance to physical distance could be made. Lastly, the pixel distance of the tip and its reflection at the initial starting location is measured and converted to physical distance. This distance was then divided by two to obtain the initial wall-normal location of the probe.

3.2.3 Oil Film Interferometry Tests

The design of OFI test space was made after the conclusion of the pressure distribution tests. In comparison with reference measurements, the probe locations that were observed to cause the most significant deviations in pressure distribution within the LSB were chosen for OFI tests. The runs at AoA = 0° for all Reynolds numbers and at 6° for Re = 100,000 were not included in OFI tests because the low magnitude of shear stress at these flow conditions was not large enough to develop fringe lines of sufficient quality with the oil available. Table 3.3 summarizes the test space for OFI.

At the start of each run, a known grid pattern printed on a piece of paper was taped to the airfoil surface. The camera was focused on the region of interest on the airfoil. After the zoom and focus was set, the camera was rotated to take a snapshot of the
Table 3.3: Test matrix for oil film interferometry

<table>
<thead>
<tr>
<th>Test Number</th>
<th>Re</th>
<th>( \alpha ) (°)</th>
<th>Probe chord location index ((x/c))</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>100,000</td>
<td>9</td>
<td>2 (0.202)</td>
</tr>
<tr>
<td>2</td>
<td>100,000</td>
<td>9</td>
<td>5 (0.328)</td>
</tr>
<tr>
<td>3</td>
<td>150,000</td>
<td>5</td>
<td>2 (0.438)</td>
</tr>
<tr>
<td>4</td>
<td>150,000</td>
<td>9</td>
<td>1 (0.16)</td>
</tr>
<tr>
<td>5</td>
<td>150,000</td>
<td>9</td>
<td>3 (0.22)</td>
</tr>
<tr>
<td>6</td>
<td>200,000</td>
<td>5</td>
<td>1 (0.44)</td>
</tr>
<tr>
<td>7</td>
<td>200,000</td>
<td>5</td>
<td>3 (0.512)</td>
</tr>
<tr>
<td>8</td>
<td>200,000</td>
<td>8</td>
<td>2 (0.202)</td>
</tr>
<tr>
<td>9</td>
<td>200,000</td>
<td>8</td>
<td>5 (0.328)</td>
</tr>
</tbody>
</table>

3-D camera calibration block for the purpose of determining the interior settings of the camera required by OFI analysis. The 3-D block was moved manually on its traverse until it came into focus for the camera. The last step of preparation was to rotate the camera back to face the region of interest and take a snapshot of the grid on the airfoil for photogrammetry calculations used by OFI analysis. The hot-wire probe was then moved into place, and lines of oil were applied to the surface. Throughout the entire test run, including the tunnel start up, freestream dynamic pressure and temperature were recorded at 1 s intervals. The images were taken manually. An average run lasts for 10 minutes and produces 5 to 6 images.
Chapter 4

Results

4.1 Aerodynamic Properties at Low Re

4.1.1 Surface Pressure and Wake Velocity Distribution

Baseline Case

The normalized surface pressure distribution for Re = 40,000 and 60,000 are shown in Figures 4.1 and 4.2 respectively. The $C_p$ distribution for $Re = 60,000$ shows some points of interest. At higher angles of attack ($\alpha \geq 8^\circ$), there is a distinct low pressure region on the suction side of the airfoil that strongly indicates the presence of a laminar separation bubble (LSB). A significant increase in peak suction was also observed with the formation of LSBs. These phenomena were not observed for angles of attack below $8^\circ$. The lack of pressure recovery observed for the lower angles of attack indicates that the boundary layer did not reattach once separated. For the pressure distribution at $Re = 40,000$ shown in Figure 4.1, the boundary layer was not observed to reattach.
Figure 4.1: $C_p$ distribution for Re = 40,000 without roughness elements

Figure 4.2: $C_p$ distribution for Re = 60,000 without roughness elements

The normalized wake velocity profiles are shown in Figures 4.3 and 4.4 for the baseline Re = 40,000 and 60,000 respectively. In Figure 4.4, the wake velocity deficits are all centred close to the centreline of the airfoil for all angles of attack below 8°. At higher angles, however, the wake is deflected down towards the pressure side of the airfoil, and the width is reduced by about 50%. This confirms that the flow reattached for angles of attack of greater than 8°. The stronger instabilities in the separated shear layer at higher angles of attack are believed to have caused the transition in the boundary layer that led to reattachment. With the reduced wake size and the increased peak suction observed for $\alpha \geq 8^\circ$ in the Re = 60,000 baseline case, the $L/D$ ratio significantly increased. This will be documented in more details in Section 4.1.2.

Additional tests were conducted for the region between $6.5^\circ \leq \alpha \leq 7.5^\circ$ to further investigate the flow reattachment observed. The pressure and wake velocity distributions are shown in Figures 4.5 and 4.6. The “higher Re” and “lower Re” in the legends denotes the slight deviations from the nominal Reynolds number found in the two runs. The average Reynolds number of the two tests were calculated to be $60368 \pm 175$ and $59903 \pm 172$, and the difference was $465 \pm 256$. From Figure 4.6, it is clear that the flow only attached at $\alpha = 7.5^\circ$ for the slightly higher Reynolds number run. This showed that
at $\alpha = 7.5^\circ$, the airfoil is very sensitive to the flow conditions. An increase of less than 1% in Reynolds number was able to trigger flow reattachment. Because this sensitivity to changes in Reynolds number was only observed for this angle, it is concluded that $\alpha = 7.5^\circ$ is the critical angle for flow reattachment at $\text{Re} = 60,000$. From examining the pressure distribution shown in Figure 4.5, there is an anomalous high pressure region along the pressure side of the airfoil for $\alpha = 7.5^\circ$ at the slightly higher Reynolds number. An additional measurement of pressure distribution was made for just $\alpha = 7.5^\circ$ at $\text{Re} = 60357 \pm 228$. Figure 4.7 shows the pressure distribution for $\alpha = 7.5^\circ$ for all three tests. Comparing the $C_p$ distribution of the two tests with the highest averaged Reynolds number, the anomalous high pressure region was not observed in the additional test even though the Reynolds numbers from the two tests were within each other’s error bounds. It is suspected that the high pressure region is due to experimental error rather than the nature of the flow. The effects of this anomaly on the lift coefficient are discussed in Section 4.1.2.
Figure 4.5: Additional $C_p$ distribution for $\text{Re} = 60,000$ at $6.5^\circ \leq \alpha \leq 7.5^\circ$ without roughness elements; “higher” and “lower” Re denotes slight deviations from the nominal Reynolds number.

Figure 4.6: Additional wake velocity distribution for $\text{Re} = 60,000$ at $6.5^\circ \leq \alpha \leq 7.5^\circ$ without roughness elements; “higher” and “lower” Re denotes slight deviations from the nominal Reynolds number.

Figure 4.7: $C_p$ distribution for $\alpha = 7.5^\circ$ for three tests at slightly deviated Reynolds numbers from $\text{Re} = 60,000$. 
Figure 4.8: $C_p$ distribution for Re = 40,000 with rectangular roughness element; the black dots denote the pressure port downstream of the roughness element; the trailing edge of the roughness element is located 1 mm upstream.

Figure 4.9: $C_p$ distribution for Re = 60,000 with rectangular roughness element; the black dots denote the pressure port downstream of the roughness element; the trailing edge of the roughness element is located 1 mm upstream.

With Rectangular Roughness Element

The $C_p$ distributions with the rectangular roughness element attached are presented in Figures 4.8 and 4.9 for Re = 40,000 and 60,000 respectively. Comparing the pressure distribution with the baseline case, the rectangular roughness elements were able to introduce sufficient instabilities into the separated shear layer, which caused the reattachment of the boundary layer downstream. Laminar separation bubbles were observed from examining the suction side pressure distributions. Also note that there is a clear trend that the separation bubble is moving upstream and decreasing in size with increasing angles of attack. Suction peaks were observed just downstream of the roughness elements, as labelled by the black dots. The measurements here could be contaminated as the trailing edge of the roughness elements were only 1 mm upstream of the pressure port, which is the width of the rectangular strip.

The wake profiles with rectangular roughness elements are presented in Figures 4.10 and 4.11 for Re = 40,000 and 60,000 respectively. Comparing with the wake profile of the
baseline case, the widths are clearly reduced, confirming that the flow was reattached by the disturbance introduced through the roughness elements. Also, it can be seen from the wake velocity distributions that the flow did not reattach in the $Re = 40,000$ case for $\alpha \leq 4^\circ$, but did for the high Reynolds number case. This implied that the rectangular roughness elements used was not as effective at the lower Reynolds number.

With Half-Round Roughness Element

The $C_p$ distributions for the half-round roughness elements are shown in Figures 4.12 and 4.13 for $Re = 40,000$ and 60,000 respectively. As with the case of the rectangular roughness elements, the half-rounds were also able to trigger the reattachment of the boundary layer downstream. Upon a closer examination of the pressure distribution for the test case at $Re = 40,000$ with half-rounds, the peak suction and the region around it at $\alpha = 10^\circ$ are lower than that at $\alpha = 8^\circ$. This indicated that the airfoil is stalling at $\alpha = 10^\circ$. The same suction peaks just downstream of the roughness elements are also observed here. Again, these peaks could be due to the contamination from the presence of the roughness elements.
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Figure 4.12: $C_p$ distribution for Re = 40,000 with half-round roughness element; the black dots denote the pressure port downstream of the roughness element; the trailing edge of the roughness element is located 1 mm upstream.

Figure 4.13: $C_p$ distribution for Re = 60,000 with half-round roughness element; the black dots denote the pressure port downstream of the roughness element; the trailing edge of the roughness element is located 1 mm upstream.

The wake profiles of the half-round test cases are presented in Figures 4.14 and 4.15 for Re = 40,000 and 60,000 respectively. Again, the reduced wake width compared with the baseline wake profiles indicated that the half-round roughness elements were also able to cause reattachment of the boundary layer.

Figures 4.16 and 4.17 show the trailing edge $C_p$ vs. $C_l$ for all six test cases. Regions of flow separation and reattachment can be clearly seen here. A convergence to $C_p = 0$ or a slightly positive value indicates attached flow, while divergence from it indicates detached flow. Again, it can be seen that the effectiveness of the roughness elements increase with Reynolds number. The flow was attached for all angle tested at Re = 60,000 with the roughness elements in place, but it was only attached for the higher angles for Re = 40,000. The two types of roughness elements had similar performance.
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Figure 4.14: Wake velocity distribution for $Re = 40,000$ with half-round roughness element

Figure 4.15: Wake velocity distribution for $Re = 60,000$ with half-round roughness element

Figure 4.16: Trailing edge $C_p$ vs. $C_l$ for all $Re = 40,000$ test cases

Figure 4.17: Trailing edge $C_p$ vs. $C_l$ for all $Re = 60,000$ test cases
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4.1.2 Aerodynamic Coefficients

The lift coefficient was determined by integrating the $C_p$ distributions of all of the test cases in the previous section. The surface of the airfoil was divided into panels, with the boundaries located at the pressure ports. Midpoint rule was used for the numerical integration. The lift coefficient for $Re = 40,000$ and $60,000$ are shown in Figures 4.18 and 4.19 respectively. The lift coefficients from the test cases with the roughness elements were significantly higher than those from the baseline case. This was due to the reattachment of the flow over the airfoil. Looking at the lift coefficients for the $Re = 60,000$ cases, there is a significant increase in the lift coefficient from $\alpha = 6^\circ$ to $8^\circ$. This is again due to the reattachment of the flow. Additionally, in Figure 4.18 the airfoil clearly stalls at $\alpha = 10^\circ$ in the half-round case, confirming the interpretation of reduction in peak suction going from $8^\circ$ to $10^\circ$ discussed in the previous section. The data from the two tests for $6.5^\circ \leq \alpha \leq 7.5^\circ$ are also presented here. As expected, the anomalous high pressure region increased the lift coefficient for $\alpha = 7.5^\circ$ for the higher Reynolds number cases, while the other data points shows a behaviour of $C_l$ that is consistent with detached flow.

The drag coefficients were calculated using the momentum integral method [16, 64].
Assuming that the static pressure in the wake has recovered to the free stream level, using force balance on a control volume over the airfoil, the following expression can be derived for the total drag coefficient:

$$C_d = \frac{2}{c} \int_{\text{wake}} \frac{u_{\text{wake}}}{U_{\infty}} \left(1 - \frac{u_{\text{wake}}}{U_{\infty}}\right) dx.$$  \hspace{1cm} (4.1)

To ensure the validity of the static pressure recovery assumption, the wake velocity measurements were carried out at 4c downstream from the trailing edge of the airfoil. This value was recommended by El-Gammal [15] as a suitable distance for airfoil testings at moderate Reynolds number.

The total drag coefficients for Re = 40,000 and 60,000 are shown in Figures 4.20 and 4.21 respectively. As expected, the drag coefficient for the Re = 60,000 baseline case dropped significantly for $\alpha \geq 8^\circ$ because of the flow reattachment. On the other hand, the drag coefficient for the Re = 40,000 baseline case did not drop because the flow did not reattach. The roughness elements significantly reduced the total drag coefficient in both Reynolds number cases as expected. The data points from the two tests for $6.5^\circ \leq \alpha \leq 7.5^\circ$ are also presented here. The $C_d$ at $\alpha = 7.5^\circ$ for the higher Reynolds number case is expectedly low as the flow was observed to reattach, while the other data points followed the trend for detached flow. It should be noted here that the error bounds for $C_d$ were not relative to their values. This is due to the fact that the dominant error from the experiments was from the Type 223B transducer, which has a constant accuracy of 0.5 Pa rather than a percentage of the reading. Because of this, the propagated relative errors for $C_d$ is large when the drag coefficient is small. This also leads to a large error bound in the L/D ratios, which are shown later.

The lift to drag ratios are shown in Figures 4.22 and 4.23. The improvement to the
aerodynamic performance of the airfoil is more evident here, as the L/D ratio increased by an order of magnitude with the addition of roughness elements. Table 4.1 summarizes the experimental maximum L/D ratio with error bounds for all of the six test cases. The drag polars for the two Reynolds numbers are shown in Figures 4.20 and 4.21. They basically show the same information presented in the previous plots. One thing to note here is that similar drag polars were observed the wind tunnel test data of Selig et al.\cite{33,55}.

Table 4.1: Max. L/D ratio of the different airfoil configurations at Re = 40,000 and 60,000

<table>
<thead>
<tr>
<th></th>
<th>Re = 40,000</th>
<th></th>
<th>Re = 60,000</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Max L/D</td>
<td>α(°)</td>
<td>Max L/D</td>
<td>α(°)</td>
</tr>
<tr>
<td>Clean</td>
<td>6.7 ± 1.1</td>
<td>6.0</td>
<td>31.9 ± 4.6</td>
<td>8.2</td>
</tr>
<tr>
<td>Rd. RE</td>
<td>32.1 ± 12.5</td>
<td>8.2</td>
<td>44.9 ± 12.9</td>
<td>6.1</td>
</tr>
<tr>
<td>Rect. RE</td>
<td>39.0 ± 16.3</td>
<td>8.2</td>
<td>42.0 ± 8.3</td>
<td>8.2</td>
</tr>
</tbody>
</table>

Figure 4.20: $C_d$ at Re = 40,000 for baseline and roughness element cases

Figure 4.21: $C_d$ at Re = 60,000 for baseline and roughness element cases
Chapter 4. Results

Figure 4.22: L/D at $Re = 40,000$ for baseline and roughness element cases

Figure 4.23: L/D at $Re = 60,000$ for baseline and roughness element cases

Figure 4.24: Drag polar at $Re = 40,000$ for baseline and roughness element cases

Figure 4.25: Drag polar at $Re = 60,000$ for baseline and roughness element cases
4.2 Hot-Wire Probe Effects on LSBs

4.2.1 Pressure Distribution

Figures 4.26 to 4.34 show the pressure coefficient difference for all nine test cases, viz. 
\[ \Delta C_p(x, y) = C_{p,p}(x, y) - C_{p,r}(x, y), \]
where \( C_{p,p} \) is the pressure coefficient with probe in place, and \( C_{p,r} \) is the reference pressure coefficient. The \( \Delta C_p \) values are presented in two different ways. The left column in each figure shows the contour in order to identify structures of \( C_p \) effects from the probe. The right columns show \( \Delta C_p \) averaged over wall-normal locations to show its magnitude.

Pressure coefficient difference for \( Re_c = 100k, 150k, \) and \( 200k \) at low AoA (around \( 0^\circ \)) are shown in Figures 4.26, 4.27, and 4.28 respectively. For the lowest \( Re_c \), no discernible differences in \( C_p \) can be observed. There are no identifiable coherent structures in the contour plots that show \( \Delta C_p \) is affected by the probe’s chord and wall-normal locations. The averaged \( \Delta C_p \) values are all within zero when error bounds are taken into account. As \( Re_c \) increases, a band of high pressure was observed to form behind the probe tip. This band structure becomes more organized, and its peak value increases as the probe moves further downstream. This region of high pressure is caused by an earlier onset of pressure recovery. Because transition in an LSB is associated with a sudden onset of pressure recovery, the appearance of this high pressure region suggests that the additional disturbance introduced by the probe was sufficiently strong to promote transition. This transition promotion effect was observed for all of the test cases. It is largely independent of the wall-normal location of the probe, except for a few isolated cases at high AoA, which will be discussed later.

Figures 4.29, 4.30, and 4.31 show the \( \Delta C_p \) plots for the medium AoA test cases (around \( 5^\circ \)). Similar to the low AoA results, an earlier onset of pressure recovery due to probe
Figure 4.26: $\Delta C_p$ for $Re_c = 100,000$ and $\alpha = 0^\circ$. Left column shows the contour plots; red dotted lines denote the probe tip. Right column shows the average over wall-normal locations; probe schematic shows its relative chord position.
Figure 4.27: $\Delta C_p$ for $Re_c = 150,000$ and $\alpha = 0^\circ$. Left column shows the contour plots, right column shows the average over wall-normal locations.
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Figure 4.28: $\Delta C_p$ for $Re_c = 200,000$ and $\alpha = 0^\circ$. Left column shows the contour plots, right column shows the average over wall-normal locations.
disturbance was observed here. The relative strength of the effect from the probe, however, is weaker, as can be seen from the lower values of peak $\Delta C_p$ and the less organized structures in the contour plots compared to those for the low AoA results. In addition to the band of high pressure downstream of the probe tip, a band of low pressure appears behind the promoted pressure recovery. This is associated with the low pressure wake region of the probe holder body.

The largest effects from the probe were observed for the high AoA test cases (around 9°), the results of which are shown in Figures 4.32, 4.33, and 4.34. Transition promotion was observed prominently for the lowest $Re_c$ test case, and less so as the Reynolds number increases. This is expected because as Reynolds number increase, the boundary layer transition point moves upstream, therefore the initial disturbance introduced by the probe, which stayed at the same chord location for all three test cases, contributes less to boundary layer transition.

For most test cases, pressure difference downstream of the probe was not observed to be strongly dependent on the probe’s wall-normal location. For $Re_c = 100,000$ at $\alpha = 9^\circ$, however, there is a strong dependency on the wall-normal location. From the contour plots in Figure 4.32, a 3-D structure can be seen forming at the transition point at around 0.3c when the probe is between 0.4% c and 0.65% c away from the surface. The structure culminated in a highly localized effect when the probe tip was placed right before transition. Figure 4.35 shows a slice of the $\Delta C_p$ contour at the most affected chord position, compared with the boundary layer velocity profile. An inflection point in the velocity profile was found near the localized effect. It is believed that near this unstable region in the boundary layer, the additional disturbances introduced by the probe amplified the instabilities already present in the shear layer, and therefore promoted transition. Only tests at the lowest Re showed this localized effect, which agrees with the diminishing of transition promotion effect with increasing Re seen in the $\Delta C_p$ results.
Figure 4.29: $\Delta C_p$ for $Re_c = 100,000$ and $\alpha = 6^\circ$. Left column shows the contour plots, right column shows the average over wall-normal locations.
Figure 4.30: $\Delta C_p$ for $Re_c = 150,000$ and $\alpha = 5^{\circ}$. Left column shows the contour plots, right column shows the average over wall-normal locations.
Figure 4.31: $\Delta C_p$ for $Re_c = 200,000$ and $\alpha = 5^\circ$. Left column shows the contour plots, right column shows the average over wall-normal locations.
Figure 4.32: $\Delta C_p$ for $Re_c = 100,000$ and $\alpha = 9^\circ$. Left column shows the contour plots, right column shows the average over wall-normal locations.
Figure 4.33: $\Delta C_p$ for $Re_c = 150,000$ and $\alpha = 9^\circ$. Left column shows the contour plots, right column shows the average over wall-normal locations.
Figure 4.34: $\Delta C_p$ for $Re_c = 200,000$ and $\alpha = 8^\circ$. Left column shows the contour plots, right column shows the average over wall-normal locations.
Figure 4.35: $\Delta C_p$ wall-normal profile of the localized 3-D region compared to the normalized boundary layer profile

Because long and short bubbles have different transition mechanism \cite{35}, it is expected that their receptivity to external disturbances will be different as well. As the probe moved downstream in each of the test cases, different trends were indeed observed on how the additional disturbances added by the probe were affecting transition. For the low and medium AoA test cases, transition promotion increases in intensity as the probe moved downstream. This is more prominently observed in the low AoA cases as the high pressure regions became a well-defined band as the probe moved downstream (Figures 4.27 and 4.28). A weaker increase of transition promotion occurred in the medium AoA cases. For the high AoA cases, however, the transition promotion effects decreased as the probe moved downstream. These two different trends are believed to be caused by the difference in receptivity of long or short bubbles. Short bubble transitions are dominated by Kelvin-Helmholtz shear layer instability, while long bubbles are more dominated by viscous effects due to an extensive part of the bubble being laminar. Disturbances introduced further upstream inside a long bubble could dissipate before reaching the more receptive transition region, thus moving the probe downstream can effectively introduce more disturbances to promote transition. In contrast, any disturbance introduced near the start of an inviscid shear layer would be amplified downstream. Therefore moving the probe back would decrease the effective additional disturbance received in the transition
Figure 4.36: Pressure gradient parameter vs. Reynolds number based on maximum bubble height. A parameter value less than the critical value of $-28$ indicates a long bubble.

region. It is thought that the difference between the receptivity of the two types of bubbles to external disturbance is responsible for the two trends observed here. The pressure gradient parameter introduced by Diwan et al. [10] is used to distinguish the bubbles found in the present study into long and short types. As stated in Section 2.1.2 the parameter is based on maximum bubble height and average streamwise velocity gradient over the bubble. The authors found that a critical value of $-28$ serves as the boundary between the two types, with values below indicating long bubbles. Figure 4.36 shows the pressure gradient parameters calculated for all the test cases. It can be seen that the bubbles at high AoA are short, and long at medium and low AoA. The observed differences in the receptivity of long and short bubbles are in agreement with the current understanding of laminar separation bubble transition mechanisms.
4.2.2 Separation and Reattachment Locations

Figure 4.37 shows a comparison of separation and reattachment locations of the separation bubbles measured from pressure distribution and qualitative OFI analysis. The two methods showed good agreement for locating the reattachment point. However, larger discrepancies were found in locating the separation point. This is attributed to the difficulties in locating the start of the low pressure plateau in the pressure distribution when the pressure suction peak is low. This shows that the reattachment location can be accurately measured from $C_p$ curve, while the separation point should be measured with oil film. This is advantageous to the examination of the effect of the probe on separation and reattachment locations of the LSBs, as the probe holder blocks visual access to the reattachment location for OFI analysis. Figure 4.38 shows the separation and reattachment locations of selected test cases compared with those of the reference cases. The separation points showed excellent agreement across all of the test cases, indicating that the probe did not have noticeable effect on the separation location of the bubble. This supports the evidence seen in the $\Delta C_p$ results that the flow upstream of the probe was not noticeably affected. More scattering is observed for the reattachment locations, which is caused by the interaction of the wake from the probe holder with the separation bubble.

4.2.3 Surface Shear Stress Distribution

Figures 4.39 to 4.43 show the skin friction coefficient, $C_f$, within the LSBs. Results from test cases with and without probe are both shown in comparison. The general trend of skin friction seen is that it decreases nonlinearly as the boundary layer approaches the separation point, and becomes negative once the flow is reversed. The skin friction then remains at a relatively constant negative value after separation for the majority
of the bubble length. The onset of transition is marked by the sudden increase in the magnitude of $C_f$. After reaching a local minimum, $C_f$ rapidly increases to a positive value. It can be clearly seen that the skin friction after reattachment is much higher than before separation, due to the fact that the reattached boundary layer is turbulent.

It is interesting to note that although transition happens in the separated shear layer, OFI, being a surface measurement technique, was able to capture its effect on surface shear stress. This is in agreement with the observations made by El-Gammal et al. [16], who found that skin friction is very sensitive to changes in pressure gradient. Transition in a separation bubble causes a sudden pressure recovery that results in a large adverse pressure gradient. Because of this, OFI was able to capture the effects of transition, which occurs away from the surface, on skin friction. Figure 4.44 shows the comparison of transition locations measured by $C_p$ and $C_f$ distributions. It can be seen that the two methods have good agreement. This implies that along with separation and reattachment, OFI can experimentally determine transition locations in a separation bubble.

The presence of a hot-wire probe did not noticeably affect shear stress distribution within
the bubble. Only measurements upstream of the probe tip were made due to optical restrictions. Figures 4.42 and 4.43 show that for Reynolds number of 200,000, all the shear distributions with the probe in place collapsed with the baseline distribution. It is evident that the hot-wire probe had no effects on shear for these test cases. The same conclusion can be made for $Re_c = 150,000$ at $\alpha = 9^\circ$, as shown in Figure 4.41. In Figure 4.40, some scatterings were seen after separation. This could be attributed to the poor quality of the fringe lines obtained. Because the actual magnitude of skin friction was the lowest for this test case, the fringe lines were noisier than the others. As a result, there is a larger degree of uncertainty for this particular flow condition. Figure 4.39 shows some apparent minor deviations near the separation point. Flow separation appears to be delayed slightly when the probe is at chord location 2 ($x/c = 0.202$). This could be caused by the disturbance from the probe being convected upstream by the reverse flow. These disturbances could energize the boundary layer near separation, causing a slight delay. However, due to the restriction imposed by the viscosity of the oil used, investigations closer to the separation point was not possible. Less viscous oil would be able to more accurately measure the small magnitude of shear stress near separation.

As with the results from $\Delta C_p$, the test case at $Re_c = 100,000$ and $\alpha = 9^\circ$ is found to be the most sensitive to the presence of a probe. For the same flow condition, when the probe was placed at chord location 5 ($x/c = 0.328$), which is the natural transition point, shear stress showed no noticeable difference. This is because the disturbance from natural transition in the separated shear layer dominates the flow behaviour at this region, making the additional probe disturbance negligible.
Figure 4.39: Comparison of skin friction coefficient $C_f$ within the LSB with and without probe for Re = 100,000 and $\alpha = 9^\circ$.

Figure 4.40: Comparison of skin friction coefficient $C_f$ within the LSB with and without probe for Re = 150,000 and $\alpha = 5^\circ$.

Figure 4.41: Comparison of skin friction coefficient $C_f$ within the LSB with and without probe for Re = 150,000 and $\alpha = 9^\circ$.

Figure 4.42: Comparison of skin friction coefficient $C_f$ within the LSB with and without probe for Re = 200,000 and $\alpha = 5^\circ$.

Figure 4.43: Comparison of skin friction coefficient $C_f$ within the LSB with and without probe for Re = 200,000 and $\alpha = 8^\circ$.

Figure 4.44: Comparison of transition point prediction using $C_p$ and OFI shear stress distributions.
Chapter 5

Conclusions and Future Works

5.1 Conclusions

The aerodynamic performance of a proprietary laminar flow airfoil for low Reynolds number UAV applications was tested. The formation of laminar separation bubbles on the suction side of the airfoil at Reynolds numbers of 40,000 and 60,000 has been investigated to aid the development of CFD prediction of LSB behaviour at low Reynolds numbers. The effectiveness of 2-D boundary layer trip to improve aerodynamic efficiency was studied at these Reynolds numbers. In the second part of the project, the response of laminar separation bubble to the intrusion of a conventional hot-wire probe was systematically investigated for Reynolds numbers of 100,000, 150,000, and 200,000 for a range of angles of attack between 0° to 9°. Pressure and shear stress distributions were examined in the region within and near the separation bubbles. Oil film interferometry was used to measure the time-averaged shear stress distribution.

The $L/D$ ratio of the airfoil investigated was very poor at $Re_c = 40,000$ and 60,000 as it
was designed for operations around $Re_c = 300,000$. The angles of attack tested spanned from $2^\circ$ to $10^\circ$ in increments of $2^\circ$. At $Re_c = 40,000$, there were massive laminar boundary layer separations for all angles. As a consequence, peak suction was low and the wake was wide, resulting in reduced lift and increased drag. The maximum lift-to-drag ratio at this Reynolds number was only 6.7, occurring at $6^\circ$. Test for $Re_c = 60,000$ also revealed large boundary layer separation for $2^\circ$ to $6^\circ$. For angles of attack of $8^\circ$ and higher, however, the boundary layer reattached downstream on the airfoil, forming a laminar separation bubble. This is attributed to the flow becoming more unstable in the separated shear layer, causing transition to happen early enough for reattachment. The critical angle of attack for boundary layer reattachment was found to be $7.5^\circ$. Beyond this angle of attack, aerodynamic performance increased significantly. The highest $L/D$ ratio was 31.9 for this Reynolds number, occurring at $8^\circ$. 2-D boundary layer trips were then used to investigate their effectiveness in improving aerodynamic performance. Two different cross section geometries were used: rectangular and half round. They both had identical height and width. The trips were polystyrene strips attached at $8.5\% c$. Significant improvements were observed for both Reynolds numbers. Both trips were able to introduce sufficient disturbances into the boundary layer to trigger transition early enough for reattachment. As a result, peak pressure increased and wake width decreased for all flow conditions. The improvement in lift-to-drag ratio was substantial for $Re_c = 40,000$, where the half round and rectangular trips increased it to 32.1 and 39.0 respectively, both occurring at $8^\circ$. The improvement for $Re_c = 60,000$ was less dramatic, but still noticeable. Both trips increased the lift-to-drag ratio to around 43.

The effect of hot-wire probes on laminar separation bubble was investigated in the second half of the project. Tests were performed for three Reynolds numbers, 100,000, 150,000, and 200,000, with three angles of attack for each. The angles of attack ranged between $0^\circ$ and $9^\circ$. They were separated into low (around $0^\circ$), medium (around $5^\circ$), and high (around $9^\circ$) angles. The probe was mounted on a traverse system that enabled it to approach
the airfoil surface at an angle of $6^\circ$ or less with the local surface tangent. No significant changes to the pressure distribution upstream of the probe tip were observed for all test cases. Downstream of the probe tip, pressure recovery within the separation bubble, which signifies the onset of transition, occurred earlier as compared to the baseline case without the probe. This implied that the additional disturbance introduced by the probe was sufficiently high to promote transition in the separated shear layer. As the probe tip moved downstream within the bubble, two different trends were observed. For the low and medium angle tests, the transition promotion effect increased as the probe tip moved closer to the natural transition point. The opposite was true for the high angle test cases. This implies that there are two different transition mechanisms at work. Using the pressure gradient parameter proposed by Diwan et al. [10], it was confirmed that long bubbles formed for medium and low angles of attack, while short bubbles were present for high angles. Short bubble transition is dominated by Kelvin-Helmholtz instability. Long bubble’s transition process is currently unclear, however it is suspected that viscous effects play a more important role than in short bubbles. It is speculated that the disturbance introduced to a short bubble is amplified as it travels downstream, thus moving the source of disturbance downstream would lessen its contribution to transition. On the other hand, the disturbance introduced to a long bubble is dampened by viscous effects as it travels downstream, therefore moving the source downstream would increase its contribution to transition.

The wall-normal location of the probe was found to have no large effect on transition promotion, except for the test case at $Re_c = 100,000$ and $\alpha = 9^\circ$. For this case, while the probe was upstream of the natural transition point, a very localized transition promotion, as a function of probe wall distance, was observed. Upon a closer examination of the boundary layer profile, an inflection point in the boundary layer velocity profile was discovered. When the probe is near the inflection point, transition promotion was most prominent. It is concluded that the disturbances introduced into this highly unstable
Chapter 5. Conclusions and Future Works

region caused the localized effects seen.

Finally, oil film interferometry was used both as a qualitative and quantitative measurement method to examine the effect of the probe on the flow upstream of the tip in the separation bubble. The separation point did not change noticeably with the addition of the probe. Shear stress distribution remained largely unchanged as well. There does appear to be a slight delay in separation in the case of $Re_c = 100,000$ at $\alpha = 9^\circ$. Conclusions regarding this cannot be drawn at this point, however, as more data is needed near the separation point. The viscosity of the oil used limited the test space to only include flow conditions with relatively high shear stress. As a result, OFI could not be performed on all of the low angle and some of the medium angle test cases.

5.2 Future Works

Additional improvements could be made to the probe effect experiments in order to gather more and higher quality data for more conclusive results.

1. Less viscous oil could be used for future tests. The current oil, which has a nominal viscosity of 20 centistokes (cSt), performed well for tests at $Re_c = 200,000$. It did not, however, develop high quality fringe lines near low shear regions such as the separation point for the other Reynolds numbers. A less viscous oil would be able to develop adequate fringe lines near these regions of interest. 10 cSt is recommended.

2. Lower Reynolds number could be tested to see if the probe would have more effects on the LSB as Reynolds number decreases. Current results show an apparent higher probe effect for lower Reynolds numbers, but only slightly. Testing for lower Reynolds number would get a more complete picture of any potential effect of the
probe. Recommended Reynolds number would be between 60,000 and 100,000.

3. Different airfoil geometries could be tested to see the sensitivity of probe effect to airfoil shapes. It is well known that LSB formation is highly sensitive to airfoil geometry, it would be interesting to see if probe effects on LSB would have similar sensitivity.

4. PIV could be used to measure the flow field of the separation bubble with the intrusion of a probe. The current OFI setup does not permit measurements downstream of the probe due to optical blockage. PIV would obtain a sense of the overall 3D effects of the probe on LSBs.
Bibliography


