ABSTRACT

ALEMAN CHONA, MARIA AUXILIADORA. Airfoil Flow-Separation and Stall Detection Using Surface-Mounted Pitot Tubes. (Under the direction of Dr. Ashok Gopalarathnam.)

A two-probe sensor concept for the detection of flow separation is introduced in this research. The conventional flight envelope of transport-type aircraft follows the linear aerodynamic region where the flow remains attached and the boundary layer is thin. However, as the angle of attack is increased, the adverse pressure gradient on the surface causes the flow to lose momentum and eventually separate resulting in significant loss of lift, leading to aircraft stall accidents. In this work, a real-time pressure measurement system is explored that consists of two opposite-facing pitot tubes mounted on the surface of an airfoil to detect flow separation or stall. Experimental investigations were conducted in the subsonic wind tunnel at North Carolina State University (NCSU) to assess the effectiveness of such a technique. Tests were performed at three different Reynolds numbers (0.39, 0.5, and 0.56 million) on a 12%-thick low-Reynolds number airfoil in the NCSU subsonic wind tunnel. An estimate of the separation-point location was obtained from the flat-lining behavior or constant pressure region on the airfoil surface pressure distribution to confirm the presence of separation region at the pitot tubes location. The results showed that the pressure difference between the forward and backward-facing pitot tubes can be successfully used to determine the location of the separation point as the angle of attack changes. For any airfoil, the location of the separation point on the airfoil surface can be correlated with the stall angle of attack. Hence, mounting the pitot tubes at a chordwise location that corresponds to flow separation at stall can serve as an effective technique for stall detection. The two-probe concept holds the potential to be further developed into an entirely self-contained device that can be externally mounted on any aircraft system.

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Airfoil Flow-Separation and Stall Detection Using Surface-Mounted Pitot Tubes

by
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DEDICATION

To my parents, sister, and uncle for their unconditional support, love, and motivation, which encourage me to pursue all my goals.

BIOGRAPHY

Maria Auxiliadora Aleman Chona was born in San Cristobal, Venezuela on November 24, 1992. She began playing tennis at the age of eight, and soon after, she started competing nationally representing her state. She occupied the third position in the national rankings, and she got the chance to compete internationally representing her country. She had the goal of obtaining an athletic-scholarship in the United States to play Tennis for a Division I school once she finished high school. She completed her secondary education at Unidad Educativa Colegio Metropolitano in 2008, where she was the valedictorian, and then she spent a year preparing for the entrance exams and competing internationally representing her country. In the Fall of 2009 she moved to Jonesboro, AR, to pursue a bachelor's degree in Mechanical Engineering at Arkansas State University. A year after, she transferred to Arkansas Tech University, where she completed her bachelor's degree with honors in 2014. Shortly afterwards, she began working at Baldor Electric in Clarksville, AR as a Manufacturing/Lean Engineering Intern. Her college and industry experience motivated her to pursue a Master's degree. She began working on her Master's degree in Aerospace Engineering in the Fall of 2015 at North Carolina State University. In the Spring of 2016 she joined Dr. Ashok Gopalarathnam's Applied Aerodynamics Research Group where she conducted research under his guidance. Shortly after finishing her Master's degree she will began working towards a Ph.D. degree in Aerospace Engineering at North Carolina State University.

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Chapter 1

Introduction

Aircraft generally operate in the linear regime of aerodynamics. In this region, the flow is mostly attached and the boundary layer is thin. This results in a linear relation between lift and angle of attack and has been widely studied by a variety of experimental, computational, and analytical methods. As the angle of attack is increased, the adverse pressure gradient on the upper surface and the wall shear cause the flow to lose momentum leading to boundary-layer separation [1,2]. The flow separation progresses towards the leading edge of the wing as the angle of attack is further increased, thereby causing stall, which is characterized by a noticeable loss in lift, a significant increase in drag, and is often associated with loss of control.

Although some aerodynamic applications such as wind turbines and helicopters routinely operate in or near the stalled regime, it is generally undesirable for transport-type aircraft where the conventional flight regimes remain in the linear aerodynamic region. Early detection and avoidance of stall is crucial to preventing many loss-of-control-induced accidents, and has been the subject of extensive research. The research presented in this thesis is an investigation of the use of surface-mounted pitot tubes as a flow-separation detection and stall-prevention system on a low-Reynolds number airfoil.

1.1 Flow Separation and Stall

As flow passes over an airfoil, the airfoil experiences a pressure gradient on its upper surface. That pressure gradient exhibits a portion of high to low pressure near the leading edge (LE) region, called a favorable pressure gradient, and the suction peak is the value of the lowest pressure. After the suction peak, the pressure increases in the flow direction and attempts to return to approximately the freestream static pressure. This resultant pressure gradient is called an adverse pressure gradient. There are two necessary factors that cause flow separation, one is the increasing pressure in the flow direction (adverse pressure gradient), and the other is the laminar or turbulent viscosity effects [3]. The flow in the boundary layer suffers a greater deceleration due to the viscosity effects, which reduces the momentum of flow near the wall, and it limits the ability of air to move forward against the pressure [3]. At the onset of separation both the velocity gradient and the shear stress are zero, and at a point downstream of separation flow reversal occurs due to the viscous effects. Figure 1.1 shows the velocity profile inside the boundary layer before the separation point, at the onset of separation, and once flow has separated.

Figure 1.1: Boundary layer velocity profiles.

When the flow is attached to the surface of an airfoil, the airfoil can produce high lift and low drag. However, when the angle of attack is increased, the flow on the upper surface of the airfoil separates due to a high adverse pressure gradient and the stream structure over the upper surface is no longer the most optimum for the best performance. The higher the angle of attack, the more the separated flow region shifts towards the LE and the more it increases in size. The consequence of separated flow at a high angle of attack is a large loss in lift and a significant increase in drag, which are the conditions at which the airfoil is said to be stalled [4].

There exists a close relationship between flow separation and stall, and several modern low-order methods use flow separation as a central element in predicting the loss of lift and stall at high angles of attack [5]. Due to this relationship, knowledge of the separation point is an important element of stall detection and avoidance. The critical flow feature associated with flow separation is the reversal of flow direction at the airfoil surface, which is also accompanied by a change in direction of shear stress. At the exact location of flow separation, the flow speed at the airfoil surface parallel to the wall becomes zero. Numerous sensors based on different transduction mechanisms can successfully capture the changes in the flow direction and effectively identify the critical aerodynamics flow features like separation-point location, flow reattachment, and stagnation point. For instance, researchers have developed shear-stress sensors based on anemometric principle [6]. Hot-wire sensors [7], thermal flow sensors [8] and hot-film sensors [9,10] have been demonstrated by researchers to be successful means of sensing flow features. Self-governing smart plasma slats have been developed for sensing and controlling flow separation and incipient wing stall [11,12].

MEMS technology has opened new avenues with on-skin sensors for the identification of flow features and measurement of shear stress [13]. Artificial microstructures immersed in the boundary layer can successfully respond to changes in the surface shear stresses. Cylindrical micro-pillars have been explored by researchers for quantifying shear stress [14,15] and for identification of flow separation in experimental investigations and computational simulations [16,17]. Micro-fences have been successfully shown to be capable of sensing flow reversal on a flat-plate

wind tunnel setup using optical transduction [18]. Although optical measurement techniques have shown potential in laboratory investigations, the technology is still not entirely feasible for practical applications on aircraft wings due to the complicated camera setup required.

Direct conversion of pillar deflection to electrical output is a more practical approach and these techniques have been explored by exploiting the electromechanical behavior of smart material systems, such as carbon nanotube arrays [19–21] and piezoelectric microstructures [22,23]. More complicated sensor designs and transduction mechanisms involve cylindrical microstructures attached on flexible platforms [24–26], where the rotational moment of the base of the pillar displaces the membrane underneath due to the influence of external force, to induce a detectable electrical response in the form of capacitance, inductance or resistance change.

The boundary-layer separation is also reflected as a flat-lining in the surface pressure distributions as the suction pressure drops to low values resulting in the loss of lift. Surface-pressure signatures have been studied by researchers to identify the connection of boundary-layer behavior with surface pressures [27,28]. Tracking the surface pressures at different chord locations can also serve as a useful tool for the identification of flow separation and stall detection as adopted by Yeo et al. [29,30]; in their work, differential pressures at several chordwise locations on the airfoil were monitored, and the relative differences between pressure measurements from the leading-edge and trailing-edge regions were compared against each other, allowing a degree of stall prediction.

1.2 Research Objectives

This research aims to present a comparatively straightforward, low-cost approach to detect flow separation using a set of two thin pitot tubes mounted on the airfoil surface. The pitot probes are placed tangential to the airfoil surface, with one probe facing the leading-edge (forward-facing), similar to a Preston tube [31], and the second probe pointed towards the trailing-edge (backward-facing) to measure a pressure value that is expected to be close to the local static pressure value. Together, these two probes can provide critical distinct pressure

measurements characteristic of separation-point location. With some knowledge of the type of airfoil under consideration, the location of the pitot probes can be carefully determined such that the stall angle coincides with the flow separation at the device location. Hence, the knowledge of separation-point crossing the location of the pitot probes can be effectively used as a stall-warning technique. With further development, this system has the promise to lead to the future possibility of developing an entirely self-contained device that can be externally mounted on any aircraft system.

1.3 Outline of Thesis

The current thesis is organized as follows. Chapter 2 gives an overview of the pitot-tubes concept and it explains why a system of two pitot tubes was utilized instead of simply using a forward-facing thin pitot tube. Chapter 3 contains specifications of the pitot tubes, testing facility and setup, experimental technique, and uncertainty analysis. Chapter 4 presents the results of wind-tunnel experiments to examine the potential of such a simple two-probe stall sensing concept. Finally, in Chapter 5 conclusions are discussed and the scope for future work is presented.

Chapter 2

Concept

A pitot probe mounted on the airfoil surface and facing the incoming attached flow will measure the total pressure, i.e. a sum of the local dynamic pressure (depending on the location of the probe within the boundary layer) and the local static pressure corresponding to the surface pressure on the airfoil at that chord location. The total pressure inside the region of separated flow is generally significantly lower than that outside the region, where the pressure is comparable to the freestream total pressure. The same pitot probe inside a separated-flow regime will register a reduced dynamic pressure due to flow reversal and low static pressure. So, as the angle of attack of the airfoil increases, a surface-mounted pitot probe is expected to observe a drop in the pressure measurements. A critical value of the pressure sensed by the probe can be identified as the onset of flow separation, and this value will correspond to the static pressure on the airfoil at that chord location. Figure 2.1a shows a sketch of a forward-facing probe mounted on the upper surface of an airfoil at 50% chord at an angle of attack of zero degrees, and an image of the same forward-facing probe mounted on the upper surface of an airfoil at an angle of attack of 20 degrees.

A forward-facing probe alone can practically provide a signature that marks the flow reversal at the probe location. The problem associated with just one probe is that the critical-pressure value will vary for different values of the freestream dynamic pressure for the same airfoil,

Figure 2.1: Pitot tubes on the airfoil outside the separated regime at low angles of attack and within the separation regime at high angles of attack.

thereby requiring the knowledge of the freestream dynamic pressure for normalization. This calls for a pitot-static probe at a different location that is either always facing the freestream or requires necessary corrections to compensate for angle of attack change. Although such a procedure will work, this defeats the idea of having a single, self-sufficient unit mounted externally on the airfoil surface which can detect stall for any freestream condition and without any prior calibration or external sensor knowledge.

For this purpose, the additional pressure measurement from the backward-facing probe (Figure 2.1b) was explored. Aligned with the direction of the flow, such a probe is expected to measure a pressure that is close to the local static pressure, as the dynamic pressure would be zero or a small negative value in the wake of the probe itself [32]. Hence, a difference between the pressures measured by the forward-facing probe and the backward-facing probe at the same chord location would drop to zero as the separation-point reaches the probe location, irrespective of the freestream velocity. This observation aligns with the expected behavior and is exploited in this research.

Chapter 3

Experimental Setup

The forward-facing and backward-facing pitot probes were mounted on the upper surface of a low-Reynolds number airfoil, and tests were conducted in the subsonic wind tunnel at North Carolina State University (NCSU). Probe specifications, wind-tunnel facility, test setup, and uncertainty analysis are presented in the following sections.

3.1 Probe Specifications

The pitot probes mounted on the airfoil were fabricated by bending stainless steel tubes of 1.59 mm (0.063 inch) outer diameter and 0.254 mm (0.010 inch) wall thickness, with a probe diameter ratio of 0.68 (shown in Figure 3.1). The tubes were configured such that one tube was facing the direction of the freestream flow (towards the leading edge) and the second tube was pointed towards the trailing edge as shown in Figure 3.2. In addition, the pitot tubes were installed such that both tubes were capable of recording measurements at the same chord location along the airfoil. It was ensured that the probes made full contact with the airfoil wall to not have discrepancies in the readings due to a small clearance between the airfoil wall and the probes. The probes were estimated to be completely immersed in the boundary layer for lower angles of attack when the flow is attached, and within the separated regime at higher angles of attack. The length of both probes along the flow direction was 2.2 cm. This length

was chosen to be larger than approximately three times the diameter of the stainless steel tubes to minimize any interference [33] caused due to the 90 degrees bend of the probe itself. Figure 3.2 shows the pitot tubes mounted on the airfoil at 50% of the chord.

Figure 3.1: Pitot tubes specifications.

Figure 3.2: Pitot tubes mounted on the airfoil surface at 50% chord location. Forward-facing probe (top), Backward-facing probe (bottom).

3.2 Wind-Tunnel Setup

To evaluate the performance of the surface-mounted pitot tubes, the probes were attached to an airfoil model and tested in a low-speed wind tunnel. The tests were conducted in the NCSU Subsonic Wind Tunnel (shown in Figure 3.3 [34]), which is a closed-circuit tunnel with a 0.81-m high, 1.14-m wide, and 1.17-m long test section. The tunnel fan is equipped with variable-pitch blades allowing the velocity in the test section to be continuously varied up to a maximum speed of approximately 40 m/s at a dynamic pressure of 15 psf. The tunnel has a settling chamber to test section contraction ratio of 3:1, and the setting chamber is equipped with an aluminum honeycomb screen and two stainless steel anti-turbulence screens with the intention of reducing the freestream turbulence in the test section [34]. Turbulence levels were obtained

from experiments performed by fellow a PhD student using a hot-wire anemometer placed at the center of the test section and traversed horizontally ± 20 cm across the test section. At each location tested, the turbulence intensity was determined to be 1.3%. The turbulence intensity is equivalent to the ratio of the root mean square of the velocity fluctuations to the average velocity [35]. The tunnel turbulence factor was calculated using a turbulence sphere test according to the methods described by Barlow and Pope [36], and it was found to be 1.4. An 8-inch diameter sphere was utilized and the tunnel speed was varied in order to obtain measurements at different Reynolds numbers. The high turbulence intensity in this facility increases the chance of having a premature transition from laminar to turbulent flow on the airfoil [36]. This is critical for laminar flow airfoils, especially when attempting to detect flow separation. In order to decrease the need for low turbulence levels it is suggested to install a trip strip near the model's leading edge to fix the transition point on the model [36]. However, a trip strip was not installed on the model used in this exploratory study, but it is being considered for a follow-on extension.

Figure 3.3: North Carolina State University Subsonic Wind Tunnel [34].

The airfoil used to test the probes is a 12%-thick low-Reynolds-number airfoil, designed specifically for the low Reynolds numbers experienced in the NCSU subsonic wind tunnel [37]. The model is composed of 9 identical interchangeable sections of chord 30.5 cm, span of 9 cm,

and a 20% chord trailing edge flap. The flap was maintained at zero degrees deflection for the current work. The airfoil was mounted vertically and the sections were assembled together to span the entire test section from the floor to the ceiling. This configuration simulates a 2-D airfoil by eliminating any tip interference in the test. Each of the sections was fabricated using stereolithography and has pressure ports on the surface along the entire chord length. There are 22 pressure taps on each surface of the airfoil. The taps are unequally spaced around the airfoil so as to obtain more detailed pressure distribution near the leading edge. These taps are able to communicate the instantaneous pressure values to the measurement system via urethane pressure tubing that runs within the airfoil and connects to the wind-tunnel data-acquisition setup. The pressure measurement system consists of three Scanivalve multi-point electronic pressure-scanning (ESP) modules. Each unit is a DSA3217 digital sensor array that incorporates 16 temperature-compensated piezoresistive pressure sensors, and for a sensor pressure range of ± 10 inch H₂O its static accuracy (% F.S) is $\pm 0.20\%$ (± 52.02 psf $\pm 0.20\%$).

The pressure data was collected from the mid-span section of the airfoil model to ensure that there was no interference due to the wind tunnel walls. The forward-facing and backward-facing pitot probes were attached at the desired chordwise location on the upper surface of the airfoil (either 50%, 60%, and 70% of the chord) and at a spanwise distance of 4.5 cm from the location of the airfoil pressure taps, as shown in Figure 3.4. The airfoil surface pressure measurements were also taken simultaneously with each probe. The airfoil model was mounted on a vertical support through the floor of the wind-tunnel test section. This support has a Velmex stepping motor that controls the angle of rotation of the vertical support. The data-acquisition process was automated for every run. A National Instruments LabVIEW[®] interface was used to acquire the pressure measurements and to control the airfoil rotation by sending commands to the Velmex stepping motor software (COSMOS). A screen shot of the LabVIEW virtual instrument (VI) is shown in Figure 3.5. The angle of attack was varied from 0 to 10 degrees in steps of 2 degrees and from 10 to 25 degrees in steps of 0.5 degrees. At each angle of attack, the flow was allowed to stabilize for 16 seconds and then pressure data was collected at

a rate of 100Hz for 5 seconds, which resulted in 500 data points per pressure tap/probe being recorded at each angle of attack. Once a run was complete, a MATLAB program was used to identify sections of the data corresponding to each angle of attack, and then calculate the mean and the standard deviation of the measurements. The reduced data from the pitot tubes was then plotted for inspection, and the airfoil surface pressures were reduced to pressure coefficients using the tunnel dynamic pressure, which were subsequently integrated using the trapezoid method to calculate the airfoil lift coefficient (C_l). It is worth noting that drag data was not collected in the current work because it was not essential to predict airfoil flow separation and/or stall at different chord locations.

Figure 3.4: (Left) Location of the forward and backward-facing pitot tubes with respect to the airfoil pressure taps. (Right) Airfoil model used for mounting the pitot tubes.

Figure 3.5: A screen shot of the LabVIEW virtual instrument (VI) used for the automated data-acquisition process.

A separate LabVIEW interface was utilized simultaneously to acquire tunnel raw data such as temperature, dynamic pressure, air speed, and time. The freestream dynamic pressure was calculated using the pressure difference between the tunnel settling-section and the test-section static pressures. The tunnel settling-section static pressure was obtained from a static pressure port located downstream of the two stainless steel anti-turbulence screens, and the test-section static pressure was taken from a static pressure port located slightly upstream of the test section. The static pressures were connected pneumatically to an Omega differential pressure transducer Model PX653-10D5V. An ambient pressure measurement was required to obtain the freestream static pressure, but this measurement was not taken during the tests. Therefore,

separate freestream dynamic pressure measurements were obtained using a pitot-static tube placed at the center of the test section and connected to the wind-tunnel data-acquisition setup. The freestream dynamic pressure measurements taken with the pitot-static tube do not account for the change in freestream dynamic pressure due to the tunnel temperature increase during the time span of each run. The ambient temperature was measured using an Omega Type T Thermocouple that was mounted right after the the two stainless steel anti-turbulence screens, and connected to an Omega 1/8 DIN Thermocouple and RTD Panel Meter Series DP25B.

The equations used to calculate flow conditions, airfoil C_p distribution, and pitot tubes pressure difference are shown in Table 3.1

Table 3.1: Definitions and equations for various flow properties

Property	Symbol	Equation
Dynamic Pressure	q_∞	$q_\infty = P_{t,\infty} - P_{s,\infty}$
Ambient Density	ρ_{amb}	$\rho_{amb} = \frac{P_{amb}}{RT}$
Dynamic Viscosity	μ	$\mu = \mu_0 \left(\frac{T}{T_0} \right)^{3/2} \left(\frac{T_0 + S}{T + S} \right)$
Freestream Velocity	V_∞	$V_\infty = \sqrt{\frac{2q_\infty}{\rho_{amb}}}$
Reynolds Number	Re	$Re = \frac{\rho_{amb} c V_\infty}{\mu}$
Airfoil Pressure Coefficient	C_p	$C_p = \frac{P - P_{s,\infty}}{q_\infty}$
Pitot Tubes Pressure Difference	P_{Diff}	$P_{Diff} = P_F - P_B$

In Table 3.1, $P_{t,\infty}$ is the freestream total pressure, $P_{s,\infty}$ is the freestream static pressure, P_{amb} is the ambient pressure, R is an air gas constant, T is the ambient temperature, μ_0 is a reference viscosity, T_0 is a reference temperature, S is the Sutherland temperature, V_∞ is

the freestream velocity in the test section derived from Bernoulli’s incompressible equation and conservation of mass, c is the airfoil chord, P is the airfoil surface pressure at a particular static port, P_∞ is the freestream static pressure, and P_F and P_B are the pressures measured by the forward- and backward-facing pitot tubes respectively.

3.3 Zero Angle Of Attack Validation

XFOIL [38] is an analysis and design computational software used for subcritical airfoils. XFOIL uses an inviscid linear-vorticity panel method with a two-equation lagged dissipation integral method that is used to represent the laminar and turbulent layers, and it is coupled with an e^9 -type amplification formulation to determine the transition point. The implementation of a global Newton method aids in computing the viscous performance of the airfoil. This computational method has proven to be effective in the rapid analysis of low Reynolds number airfoil flows including those with transitional separation bubbles [38]. The XFOIL boundary layer formulation is well founded for attached flows or for a thin separated boundary layer. Therefore, XFOIL predictions are anticipated to be unreliable beyond stall. In addition, previous work done by Gopalarathnam et al [39] showed that in comparison to wind tunnel results, XFOIL tends to over-predict the $C_{l_{\max}}$, stall angle of attack (α_{stall}), and C_l values at angles of attack close to stall.

XFOIL predictions have been used in the current work to validate the model’s zero-angle-of-attack angular alignment in the tunnel. XFOIL results were obtained for a Reynolds number of 0.5 million and subsequently experimental airfoil surface pressure measurements were collected to match the computational pressure-coefficient distribution. Once the experimental results matched the XFOIL predictions, a best case was established and the validation of future test runs was done by comparing with both XFOIL and the experimental best case. In addition, XFOIL predictions were utilized to compare with the experimental C_l curve.

3.4 Data Correction

When a lifting airfoil is in a flow that is constricted between wind tunnel walls, the experimental measurements may not always compare to the airfoil's performance in free-air. The alteration in the experimental pressure and lift coefficient measurements happen because of blockage and a distortion of the streamlines [40]. There are various correction techniques for airfoil surface pressures and lift coefficient measurements such as the ones discussed by McAlister et al [40]. However, no corrections were made to the data obtained in the present work due to the primary focus being the development of a system to detect flow separation and/or stall, instead of characterizing the airfoil.

3.5 Uncertainty Analysis

Experimental measurements have a certain degree of uncertainty due to the total error given by the instrument. The total error consists of precision and bias error. Precision or random errors are caused by unknown sources that affect the measurements, and are non-repeatable. A way to reduce precision errors is to average multiple sample sets. Contrarily, bias errors are repeatable and caused by known sources such as incorrect calibration and over/under estimation of measurements. Uncertainties in the experimental flow condition, airfoil C_p distribution, and pitot tubes were calculated using the Kline McClintock method [41], which accounts only for bias uncertainties. Precision and wind tunnel corrections uncertainties are not accounted for.

Assume product R is a function of the independent variables $x_1, x_2, x_3, \dots, x_n$, and it is represented by Eq. 3.1. To calculate the experimental uncertainty in the product R , the root-sum-square (RSS) method is utilized, because multiple uncertainties have to be combined. This expression is shown in Eq. 3.2.

$$R = R(x_1, x_2, x_3, \dots, x_n) \tag{3.1}$$

$$w_R = \sqrt{\left(\frac{\partial R}{\partial x_1} w_1\right)^2 + \left(\frac{\partial R}{\partial x_2} w_2\right)^2 + \left(\frac{\partial R}{\partial x_3} w_3\right)^2 + \dots + \left(\frac{\partial R}{\partial x_n} w_n\right)^2} \quad (3.2)$$

where w_r is the uncertainty in product R , and $w_1, w_2, w_3, \dots, w_n$ are uncertainties in the independent variables that make up product R . Using this method, the flow condition, airfoil C_p distribution, and pitot tube measurements uncertainties are calculated using the equations shown in Table 3.2.

Table 3.2: Equations for uncertainty analysis

Property	Symbol	Uncertainty Equation	Partial Derivatives
Dynamic Pressure	q_∞	$w_{q_\infty} = \sqrt{\left(\frac{\partial q_\infty}{\partial P_{t,\infty}} w_{P_{t,\infty}}\right)^2 + \left(\frac{\partial q_\infty}{\partial P_{s,\infty}} w_{P_{s,\infty}}\right)^2}$	$\frac{\partial q_\infty}{\partial P_{t,\infty}} = 1$ $\frac{\partial q_\infty}{\partial P_{s,\infty}} = -1$
Ambient Density	ρ_{amb}	$w_{\rho_{amb}} = \sqrt{\left(\frac{\partial \rho_{amb}}{\partial T} w_T\right)^2}$	$\frac{\partial \rho_{amb}}{\partial T} = -\frac{P_{amb}}{RT^2}$
Dynamic Viscosity	μ	$w_\mu = \sqrt{\left(\frac{\partial \mu}{\partial T} w_T\right)^2}$	$\frac{\partial \mu}{\partial T} = \frac{\mu_0(S+T_0)(3S+T)\sqrt{T/T_0}}{2T_0(S+T)^2}$
Freestream Velocity	V_∞	$w_{V_\infty} = \sqrt{\left(\frac{\partial V_\infty}{\partial q_\infty} w_{q_\infty}\right)^2 + \left(\frac{\partial V_\infty}{\partial \rho_{amb}} w_{\rho_{amb}}\right)^2}$	$\frac{\partial V_\infty}{\partial q_\infty} = \frac{\sqrt{2}}{2\rho_{amb}\sqrt{q_\infty\rho_{amb}}}$ $\frac{\partial V_\infty}{\partial \rho_{amb}} = -\frac{\sqrt{2}q_\infty}{2(\rho_{amb})^2\sqrt{q_\infty/\rho_{amb}}}$
Reynolds Number	Re	$w_{Re} = \sqrt{\left(\frac{\partial Re}{\partial \rho_{amb}} w_{\rho_{amb}}\right)^2 + \left(\frac{\partial Re}{\partial V_\infty} w_{V_\infty}\right)^2 + \left(\frac{\partial Re}{\partial \mu} w_\mu\right)^2}$	$\frac{\partial Re}{\partial \rho_{amb}} = \frac{V_\infty c}{\mu}$ $\frac{\partial Re}{\partial V_\infty} = \frac{c\rho_{amb}}{\mu}$ $\frac{\partial Re}{\partial \mu} = -\frac{V_\infty c\rho_{amb}}{\mu^2}$
Airfoil Pressure Coefficient	C_p	$w_{C_p} = \sqrt{\left(\frac{\partial C_p}{\partial P} w_P\right)^2 + \left(\frac{\partial C_p}{\partial P_{s,\infty}} w_{P_{s,\infty}}\right)^2 + \left(\frac{\partial C_p}{\partial q_\infty} w_{q_\infty}\right)^2}$	$\frac{\partial C_p}{\partial P} = \frac{1}{q_\infty}$ $\frac{\partial C_p}{\partial P_{s,\infty}} = -\frac{1}{q_\infty}$ $\frac{\partial C_p}{\partial q_\infty} = \frac{-P+P_{s,\infty}}{(q_\infty)^2}$
Pitot Tubes Pressure Difference	P_{Diff}	$w_{P_{Diff}} = \sqrt{\left(\frac{\partial P_{Diff}}{\partial P_F} w_{P_F}\right)^2 + \left(\frac{\partial P_{Diff}}{\partial P_B} w_{P_B}\right)^2}$	$\frac{\partial P_{Diff}}{\partial P_F} = 1$ $\frac{\partial P_{Diff}}{\partial P_B} = -1$

The Scanivalve multi-point electronic pressure-scanning (ESP) module specifications are listed in Table 3.3.

Table 3.3: Scanivalve ESP module specifications.

Scanivalve Ethernet Pressure Scanner DSA3217			
Property	Values	Units	Static Accuracy (% Full Scale)
Range	± 5	inch H ₂ O	$\pm 0.40\%$
	± 10	inch H ₂ O	$\pm 0.20\%$
	± 1	psid	$\pm 0.12\%$
	± 2.5	psid	$\pm 0.08\%$
	± 5 to 500	psid	$\pm 0.05\%$
	± 501 to 750	psid	$\pm 0.08\%$
	15 to 250	psia	$\pm 0.05\%$ (with calibration performed)
	15 to 250	psia	$\pm 0.10\%$ (without calibration performed)
(including linearity, hysteresis, and repeatability)			

The Scanivalve ESP modules utilized for data acquisition had a pressure range of ± 10 inch H₂O, which is equivalent to ± 52.02 psf with relative and absolute uncertainties of $\pm 0.20\%$ and ± 0.1040 psf respectively. Let's consider an example test case to perform uncertainty analysis: the 12%-thick low-Reynolds-number airfoil tested at a Reynolds number of 0.39 million and angle of attack of 6 degrees. For the example test case, the constants used for the uncertainty analysis are shown in Table 3.4 and the single measurement uncertainties are shown in Table 3.5. Table 3.6 shows the calculated flow condition and pressure uncertainties for the example test case.

Table 3.4: Given constants

Property	Units	Value
Airfoil Chord (c)	ft	1
Gas Constant (R)	ft-lb/(slug-deg R)	1716
Reference Temperature (T_0)	deg R (K)	491.6 (273.11)
Sutherland Temperature (S)	deg R (K)	199.8 (110.56)
Reference Dynamic Viscosity (μ_0)	lb-s/ft ²	3.58E-7
Ambient Pressure (P_{amb})	lb/ft ² (lb/in ²)	2097.36 (14.57)

Table 3.5: Single Measurement Uncertainties

Property	Units	Reference Value	Absolute Uncertainty	Relative Uncertainty
Ambient Temperature (T)	F (R)	72 (531.67)	± 0.9 (± 0.9)	$\pm 1.25\%$ ($\pm 0.17\%$)
Freestream Static ($P_{s,\infty}$) Pressure	psf	0.09	± 0.1040	$\pm 115.60\%$
Freestream Total ($P_{t,\infty}$) Pressure	psf	4.9	± 0.1040	$\pm 2.12\%$
Pitot Forward-Facing (P_F) Pressure	psf	1.4	± 0.1040	$\pm 7.43\%$
Pitot Backward-Facing (P_B) Pressure	psf	-4.05	± 0.1040	$\pm 2.59\%$
Airfoil Surface Pressure (P)	psf	-5.756	± 0.1040	$\pm 1.81\%$

Table 3.6: Flow Condition and Pressure Uncertainties

Property	Units	Reference Value	Absolute Uncertainty	Relative Uncertainty
Freestream Dynamic Pressure (q_∞)	psf	4.81	± 0.1471	$\pm 3.06\%$
Ambient Density (ρ_{amb})	slug/ft ³	0.0023	$\pm 3.8915\text{E-}6$	$\pm 0.17\%$
Dynamic Viscosity (μ)	lb-s/ft ²	3.8059E-7	$\pm 4.9811\text{E-}10$	$\pm 0.13\%$
Freestream Velocity (V_∞)	ft/s	64.69	± 0.99	$\pm 1.53\%$
Reynolds Number (Re)	-	390736	± 6043	$\pm 1.55\%$
Airfoil Pressure Coefficient (C_p)	-	-1.215	± 0.048	$\pm 3.96\%$
Pitot Pressure Difference (P_{Diff})	psf	5.45	± 0.15	$\pm 2.70\%$

Airfoil C_l uncertainties were not calculated using the Kline McClintock method due to its complexity. Instead, precision uncertainties were calculated by integrating the C_p distribution data to obtain C_l values, and then the mean and standard deviation of the C_l values was calculated. The confidence interval was set at 95% (2σ). The C_l precision uncertainty for the example test case at all angles of attack described in Section 3.2 is shown in Figure 3.6.

Figure 3.6: Airfoil C_l precision uncertainty. Error bars at 95% confidence intervals.

Chapter 4

Results

This chapter presents the results from the tests conducted in the subsonic wind tunnel at North Carolina State University. The results from the pitot-static tube pressure measurements, flow separation prediction obtained from the airfoil C_p distribution, airfoil lift curve, and flow-separation and stall detection using the surface-mounted pitot-tubes are discussed in the following sections.

4.1 Pitot-Static Tube Measurements

The results for the pitot-static pressure measurements taken at the center of the test section for three freestream dynamic pressure cases (5, 8, and 10 psf) are presented in Table 4.1.

Table 4.1: Pitot-static tube pressure measurements

Tunnel Dynamic Pressure (psf)	Pressure Type	Mean Pressure (psf)	Absolute Uncertainty (psf)	Relative Uncertainty
5	Total	4.9201	± 0.1040	$\pm 2.11\%$
	Static	0.0823	± 0.1040	$\pm 126.37\%$
8	Total	7.9780	± 0.1040	$\pm 1.30\%$
	Static	0.2042	± 0.1040	$\pm 50.93\%$
10	Total	10.0282	± 0.1040	$\pm 1.04\%$
	Static	0.2889	± 0.1040	$\pm 36.00\%$

The freestream total pressure measured by the pitot-static tube is equal to the total pressure minus the atmospheric pressure, which is equivalent to the tunnel dynamic pressure. And the measured freestream static pressure is equal to the static pressure minus the atmospheric pressure, which should be a value close to zero.

4.2 Airfoil Surface Pressure Distributions And Lift Curve

4.2.1 Flow Separation Prediction From C_p Distribution

The pressure gradient on the upper surface of an airfoil changes with angle of attack. As the angle of attack is increased, the pressure gradient becomes more adverse or less favorable on the upper surface, representing an increase in pressure in the flow direction (positive dP/dx). At low angles of attack, the pressure at the trailing edge reaches a value minimally higher compared to the freestream static pressure, but at high angles of attack the pressure near the trailing edge becomes smaller than the freestream static pressure [4] and a constant pressure region is originated. The chordwise location where the pressure plateau meets the start of the adverse pressure gradient serves as an indicator of the location of the onset of flow separation. This behavior was looked for in the C_p distributions at different angles of attack to determine the onset of flow separation at three different chordwise locations (50%, 60%, and 70% chord). The angles of attack at which the surface pressure distributions reflected the onset of flow separation at 50%, 60%, and 70% chord were used to corroborate the predictions from the pitot tubes.

The model utilized in this work does not have static pressure ports at the exact desired chordwise locations (50%, 60%, and 70% chord). Therefore, two separate surface pressure distributions, at two different angles of attack, that reflected the onset of flow separation slightly downstream and upstream of the desired chordwise location were obtained. Then, the angle of attack at which the onset of flow separation reached the desired chordwise location was linearly interpolated. Figures 4.1, 4.2, and 4.3 show airfoil surface pressure distributions at three different freestream dynamic pressures for two angles of attack. Figures 4.1a, 4.2a, 4.3a, and

Figures 4.1b, 4.2b, 4.3b show the location of the onset of separation slightly downstream and upstream of the 50% chord location. The surface pressure distributions reflecting the location of the onset of separation slightly downstream and upstream of the 60% and 70% chord locations at a freestream dynamic pressure of 5 psf are shown in Figures 4.4 and 4.5.

(a) AoA = 14.5 degrees

(b) AoA = 15 degrees

Figure 4.1: $q_{\infty} = 5$ psf. Pressure coefficient C_p distributions.

(a) AoA = 14.5 degrees

(b) AoA = 15 degrees

Figure 4.2: $q_\infty = 8$ psf. Pressure coefficient (C_p) distributions.

(a) AoA = 14.5 degrees

(b) AoA = 15 degrees

Figure 4.3: $q_\infty = 10$ psf. Pressure coefficient (C_p) distributions.

(a) AoA = 13.5 degrees

(b) AoA = 14 degrees

Figure 4.4: $q_\infty = 5$ psf. Pressure coefficient (C_p) distributions.

(a) AoA = 12.5 degrees

(b) AoA = 13 degrees

Figure 4.5: $q_\infty = 5$ psf. Pressure coefficient (C_p) distributions.

The linearly interpolated angles of attack at which the surface pressure distributions reflect the onset of flow separation at 50%, 60%, and 70% chord are shown in Table 4.2.

Table 4.2: Onset of flow separation based on surface pressure distributions

Chordwise Location	Tunnel Dynamic Pressure (psf)	Angle of Attack (deg)
50%	5	14.5
	8	14.8
	10	14.8
60%	5	13.8
70%	5	12.9

4.2.2 Lift Coefficient Versus Angle of Attack Curve

The C_l versus angle of attack curves from experiments at three different freestream dynamic pressures can be seen in Figures 4.6a, 4.7a, and 4.8a. The predicted curves obtained from XFOIL computations are also shown in these figures. It is seen that for angles of attack greater than 8 degrees, XFOIL over-predicts C_l compared to the experimental values, which is expected for angles of attack close to and beyond stall, as mentioned in Section 3.3. However, for the 5, 8, and 10 psf freestream dynamic pressure cases the stall angle of attack was 13.5, 14, and 14.5 degrees respectively, thus experimental C_l values between 8 degrees and the stall angles of attack are slightly less compared to XFOIL. In addition, the linear slope of the C_l versus angle of attack curve does not match entirely between the computational and experimental results. These discrepancies could be attributed to test setup influences, wind tunnel turbulence levels, and not applying correction factors. Some of the test setup influences are errors in the zero angle of attack location, the model's placement not being at the center of the test section, and the clearance size between the model and the test section's roof. The model was placed far back in the test section and the trailing edge was approximately 3 cm away from the test section's gap, where air is suctioned, which could have allowed for 3D effects (spanwise flow). When the clearance size between the model and the test section top was very small, the leading edge dragged at the top of the test section, making it difficult for the Velmex stepping motor to rotate the model to the correct angle of attack. These test setup influences were recognized at a late stage and were not resolved due to time constraints.

The C_l precision errors for the 5, 8, and 10 psf freestream dynamic pressure cases are shown

in Figures 4.6b, 4.7b, and 4.8b respectively. For all three cases, the C_l value limits are very close to the mean C_l values for angles of attack from zero to the stall angle of attack, and for angles of attack beyond stall the fluctuations in C_l values increase. This is expected since surface static pressure is likely to start deviating as the separated flow region increases.

Figure 4.6: $q_\infty = 5$ psf. (a) C_l vs. angle of attack (b) C_l precision error with error bars at 95% confidence intervals.

(a)

(b)

Figure 4.7: $q_\infty = 8$ psf. (a) C_l vs. angle of attack (b) C_l precision error with error bars at 95% confidence intervals.

(a)

(b)

Figure 4.8: $q_\infty = 10$ psf. (a) C_l vs. angle of attack (b) C_l precision error with error bars at 95% confidence intervals.

4.3 Pitot-Tubes Tests

Wind-tunnel tests were performed for three different freestream dynamic pressures, as listed in Table 4.3. For each case, separate runs were conducted with the pitot tubes at three different chordwise locations (50%, 60%, and 70% chord) on the upper surface of the airfoil.

Table 4.3: Cases with the forward and backward-facing pitot probes placed separately at 50% chord and 70% chord on the airfoil upper surface.

Case Number	Reynolds Number	Freestream Velocity (m/s)	Tunnel Dynamic Pressure (psf)
1	0.39 million	20	5
2	0.5 million	24	8
3	0.56 million	28	10

4.3.1 Flow Separation Detection

The results for case 1, where the freestream dynamic pressure is 5 psf, are presented in Figures 4.9 and 4.10. In this run, the pitot probes were mounted on the upper surface of the airfoil at 50% chord location. The mean and the standard deviation of the pressures recorded by the pitot probes at each angle of attack are plotted in the results.

Figure 4.9a shows the measurements made by the forward-facing pitot probe (P_F). This probe senses a sum of the dynamic pressure at the probe location inside the boundary layer and the local static pressure at the airfoil chord location. For low angles of attack, the flow remains attached at the probe location but the boundary layer grows as the angle of attack is increased. This behavior results in the pitot probe observing lower dynamic pressure because the probe's distance from the wall remains the same. The local static pressure also varies at the probe location and the overall total pressure measured by the forward-facing pitot probes follows a decreasing trend. At higher angles of attack, a greater standard deviation is observed in the recorded pressure data. This behavior is to be expected because, as the separation point

approaches the probe location, unsteady flow structures are sensed by the pressure sensors resulting in greater fluctuations in the recorded data.

The backward-facing pitot probe measurements (P_B) were compared to the static pressure measurements at the 50% chord location. The results display a trend that is close to the local static pressure, as shown in Figure 4.9b. The pressure measured by the backward-facing probe was expected to be close to the local static pressure except for some deviation due to the probe wake, as mentioned in Section 2.

The results from the pressure difference between the forward- and backward-facing probes ($P_F - P_B$) exhibit a decreasing trend as shown in Figure 4.9c. The pressure difference reaches zero at an angle of attack of 15.5 degrees which suggests the onset of the separation at the probe location. This was confirmed by the observing the flat-lining or the development of a constant pressure region in the surface pressure distribution on the airfoil, as mentioned in the Section 4.2.1. The surface pressure distribution reflected the onset of flow separation at 50% of the chord for the angle of attack of 14.58 degrees. The prediction from the pitot probes is within 1 degree of the estimated angle of attack using the surface pressures. The slight over-prediction could be attributed to the interference caused due to the probes themselves. Overall, the estimate is reasonable and the results prove that the pitot-probe system has the potential to detect the onset of flow separation at the probe location.

The same test case was repeated with the pitot probes placed at 70% of the chord location and the results are plotted in Figure 4.10. The probes show a similar trend as in the 50% chord location case. The difference between the forward and backward probes drops to zero at an angle of attack of 14 degrees, suggesting that the flow separation has reached the probe location. For this test run, the surface pressures indicated that the separation-point coincides with 70% chord location at an angle of attack of approximately 12.85 degrees. In this case also, the pitot-probe system estimates the angle of attack for onset of flow separation to within 1.2 degrees

Figure 4.9: Case 1, $q_\infty = 5$ psf. Pressure (psf) versus AoA (deg) at 50% chord (x/c). (a) forward facing probe (P_F), (b) backward-facing probe (P_B), and (c) the pressure difference between the forward and backward facing probe ($P_F - P_B$). The vertical dashed line represents the AoA at which flow separates at 50% chord based on the experimental pressure distribution over the airfoil. P_∞ is the freestream static pressure. Error bars at 95% confidence intervals.

Figure 4.10: Case 1, $q_\infty = 5$ psf. Pressure (psf) versus AoA (deg) at 70% chord (x/c). (a) forward facing probe (P_F), (b) backward-facing probe (P_B), and (c) the pressure difference between the forward- and backward-facing probe ($P_F - P_B$). The vertical dashed line represents the AoA at which flow separates at 70% chord based on the experimental pressure distribution over the airfoil. P_∞ is the freestream static pressure. Error bars at 95% confidence intervals.

4.3.2 Effect of Freestream Dynamic Pressure

The pressure measurements from the forward- and backward-facing probes and the pressure difference between the two at 50% chord for cases 1–3, are shown in Figure 4.11. As expected, the results for the pressure difference exhibit decreasing trends and the uncertainty behavior is comparable between all three cases. For cases 1–3, the pressure difference reaches zero at angles of attack of 15.5, 16, and 15.5 degrees respectively. In addition, for the same cases, the experimental pressure distribution over the airfoil without the presence of the probes suggests that flow separated at the 50%-chord location at angles of attack of 14.58, 14.75, and 14.75 degrees as shown in Figure 4.12. Although the change in freestream dynamic pressure produced slight different predicted angles of attack, all three cases exhibited a zero pressure difference, which were also relatively close to one another. This proves that the flow separation can be predicted irrespective of the freestream velocity. The results are encouraging and support the use of such pitot-tube setups regardless of the freestream dynamic pressure.

Figure 4.11: Pressure (psf) versus AoA for cases 1–3 at 50% chord (x/c). (a) Forward-facing probe (P_F), (b) backward-facing probe (P_B), and (c) pressure difference between the forward and backward facing probes ($P_F - P_B$). The AoA at which flow separates at 50% chord based on the experimental pressure distribution over the airfoil is denoted with a vertical dashed line color coded with its corresponding case. P_∞ is the freestream static pressure. Error bars at 95% confidence intervals.

Figure 4.12: Pressure difference ($P_F - P_B$) zero reference crossing for cases 1–3 at 50% chord. The AoA at which flow separates at 50% chord based on the experimental pressure distribution over the airfoil is denoted with a vertical dashed line color coded with its corresponding case.

4.3.3 Stall Detection

Based on the results, it is clear that separation point reaches 70% chord at an angle of attack of 14 degrees and the separation further moves till 50% when the angle of attack has been increased to 15.5 degrees. The next step is to correlate the location of separation point with the airfoil stall angle of attack. To study that relationship, the coefficient-of-pressure (C_p) distribution was integrated to calculate the lift coefficient (C_l) and determine the stall angle. The stall angle of attack for this airfoil is 13.5 degrees as shown in Figure 4.13a. To use the pitot probe system as a stall detection device, the probes should be mounted at the location of the separation point when stall occurs. The actual separation location at stall is close to approximately 60% of the chord as can be seen from the surface pressure distribution shown in Figure 4.13b. Therefore, the pitot probes were placed at 60% percent of the chord and tests were conducted.

Figure 4.13: (a) C_l versus angle of attack curve, (b) Airfoil C_p distribution at stall (13.5 degrees).

The forward- and backward-facing pitot tube measurements at 60% of the chord for case 1 are presented in Figure 4.14. The probes show similar trends compared to the 50% and 70% chord location cases. The results from the difference between the forward- and backward-facing probes exhibit zero pressure at an angle of attack of 14 degrees, and for this run, the surface pressures pointed that the onset of flow separation reached the 60% chord location at an angle of attack of approximately 13.81 degrees. This suggests that for the current airfoil the onset of stall is at an angle of attack of 13.81 degrees for this particular test run. The stall angle of attack prediction from the pitot-probe system is within 0.2 degrees. The results from the pitot-probe system demonstrate that it is possible to detect flow-separation at the location when stall occurs, which makes it a system with the potential to detect the onset of stall.

Figure 4.14: Case 1, $q_\infty = 5$ psf. Pressure (psf) versus AoA (deg) at 60% chord (x/c). (a) forward facing probe (P_F), (b) backward-facing probe (P_B), and (c) the pressure difference between the forward and backward facing probe ($P_F - P_B$). The vertical dashed line represents the AoA at which the stall onset occurs based on the experimental pressure distribution over the airfoil. P_∞ is the freestream static pressure. Error bars at 95% confidence intervals.

Chapter 5

Conclusions and Future Work

5.1 Conclusions

A real-time pressure measurement system (consisting of two opposite-facing pitot tubes) was investigated in this research for detecting surface-flow separation and identifying stall onset. The pitot probes were mounted on the upper surface of a low-Reynolds number airfoil that was designed for the NCSU subsonic wind tunnel. Pressure measurements from the pitot probes were recorded, along with the surface pressures on the airfoil and tunnel raw data such as freestream dynamic pressure, temperature, air speed, and time, for a complete sweep of angles of attack. In addition, data was recorded for test runs with pitot probes mounted at different chordwise locations on the airfoil. The difference between the individual measurements of the two pitot probes followed a decreasing trend as the angle of attack was increased, and the difference dropped to zero as the separation-point location coincided with the pitot-probe location. At higher angles of attack beyond stall, when the pitot probes were inside the separation region, the measurements recorded greater fluctuations but the difference between the forward and backward-facing probes exhibited mean values close to zero or negative. A reasonable estimate of the separation-point location was also extracted from the surface-pressure measurements to confirm the presence of separation region at the probe location and validate the observations.

The results presented in this thesis demonstrate the capability of a two-probe system to capture the onset of flow separation at the probe location irrespective of the freestream velocity. For any airfoil, the location of the separation-point on the surface can be correlated with the stall angle. Hence, the knowledge of the separation-point location obtained from the pitot probes can be effectively used for stall detection and avoidance. These results support the possibility of a completely self-sufficient device comprising of the opposite-facing pitot tube system similar to that used in this research.

5.2 Future Work

A next step in research would be to make slight changes to the setup and perform oil flow visualization to characterize the low-Reynolds number airfoil used in this work. This would be done to eliminate the errors in the C_l values at low angles of attack, and to check the repeatability of the flow separation and stall detection results obtained from the pitot tubes.

A following step would be to fabricate a completely self-sufficient device. Such a device would have in-built pressure transducer and wireless data-transfer capability to communicate with the on-board control system and provide information in real-time. The focus would be on identifying the design of such a device that can have sufficient space to accommodate the pressure transducers and necessary electronics. A cylindrical unit is shown as a prototype in Figure 5.1, with pressure taps at diametrically opposite ends to be used as the two-probe measuring system. A transducer can be integrated that can directly measure the difference between the pressure taps and the device could be externally mounted on any wing/blade section without compromising the structural integrity of the wing.

(a) 3D Prototype.

(b) 3D Prototype inside view.

(c) Prototype inside view.

(d) Prototype bottom view.

Figure 5.1: Cylindrical unit prototype.

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