ABSTRACT

SAINI, ADITYA. Determination of Section Aerodynamic Operating Condition on Wings and Rotor Blades From Leading-Edge Pressure Measurements. (Under the direction of Dr. Ashok Gopalarathnam.)

The identification of the inflow air data quantities such as the airspeed, angle of attack, and local lift-coefficient provides the capability for efficient control and operation of aerodynamic vehicles and devices. These quantities are significant for performance enhancement in various aerodynamic applications such as Unmanned Air Vehicles (UAVs), Micro-Aerial Vehicles (MAVs), and in wind-turbine aerodynamics. This thesis presents a novel method for the determination of the air data parameters for a section of a wing or rotor blade using a few pressure measurements in the vicinity of the leading edge. The approach approximates the leading-edge region of the airfoil as a parabola and then employs an aerodynamic model, which is derived from the inviscid equations of flow past a parabola. Pressures sensed at five discrete locations near the leading edge of an airfoil are consumed in the algorithm to solve the model using nonlinear regression techniques. The algorithm directly computes the inflow velocity and the stagnation-point location. The performance of the algorithm is assessed using experimental data in the literature, obtained from different wind-tunnel test campaigns. The results show good correlation between the actual and predicted values within the pre-stall regime. The algorithm is also tested using the measurements obtained on a rotating blade to evaluate the validity of the method in the presence of three dimensional flow. In addition, a leading-edge sleeve concept is introduced as a short term goal. The sleeve can be externally mounted on the leading edge of a wing/blade section and serve as an alternate pressure measuring unit that can be replaced with superior flush pressure-sensing technologies in the future. Wind-tunnel tests were conducted to evaluate the effectiveness of using such a device at the leading edge and study its effect on the aerodynamic behavior. The results show acceptable resemblance between the pressure taps on the sleeve and the actual pressure distribution. This result provides an impetus to further develop the leading-edge sleeve concept as a measuring system that delivers

discrete pressures to the algorithm for analysis of the aerodynamic conditions.

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Determination of Section Aerodynamic Operating Condition on Wings and Rotor Blades
From Leading-Edge Pressure Measurements

by
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DEDICATION

To my parents and my beloved sister for their unwavering support and love over all these
years.

BIOGRAPHY

Aditya Saini was born on December 11, 1990 in Faridabad, India. He completed his schooling at St. Anthonys Secondary School and received his senior secondary education from Modern Vidya Niketan School in Faridabad. Immediately after graduating from high school, Aditya began pursuing his Bachelor of Technology degree in Mechanical Engineering from the Indian Institute of Technology, Ropar. During his junior year, he got the opportunity to intern for the Engine Test-Bed Research and Design center (ETBRDC) at Hindustan Aeronautics Limited. The experience of working in the R&D department greatly motivated him to embark on the path of research. In August of 2012, Aditya enrolled at North Carolina State University to pursue a Master of Science degree in Aerospace Engineering. His research interest in aerodynamics led him to join the NCSU Applied Aerodynamics Research Group under the guidance of Dr. Ashok Gopalarathnam.

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Nomenclature

Symbols

AoA	angle of attack, deg
C_l	airfoil or section lift coefficient
C_p	pressure coefficient
FADS	flush airdata sensing
LEFS	leading-edge flow sensing
MAV	micro-aerial vehicle
NLF	natural-laminar-flow
P	local static pressure
RMS	root mean square
Re	Reynolds number based on airfoil chord length
r	thickness ratio
s_s	stagnation-point location
SEADS	shuttle entry airdata sensing system
SERI	Solar Energy Research Institute
UAE	unsteady aerodynamics experiment
UAV	unmanned aerial vehicle

Greek

α	airfoil angle of attack, deg
α_{ideal}	ideal angle of attack, deg
U_{∞}	freestream reference velocity
λ_{le}	initial camber of airfoil
λ_p	inclination of parabola
ρ_{le}	leading-edge radius of airfoil
ρ_p	radius of curvature of parabola
P_{∞}	freestream static pressure
ϱ	density of fluid

Chapter 1

Introduction

The instantaneous aerodynamic operating condition of flying bodies is an important input for purposes of flight control and performance evaluation. Unmanned Aerial Vehicles (UAVs) and Micro-aerial vehicles (MAVs) have tremendous potential in military and civilian flight operations due to their small scale and portability. However, they are greatly affected by the operational environment and complex flow conditions based on different mission requirements. For instance, UAVs capable of dynamic soaring require systems for computing the wind vector to provide the ability to exploit the naturally-occurring wind-shear gradients and to extract energy from the environment. Similarly, MAVs find applications in a variety of assignments, but the performance of their attitude control system is affected significantly by the flowfield characteristics. These small aerial vehicles are highly sensitive to turbulence owing to their small size and physical design, and this poses greater control challenges compared to the larger aircraft. Their utility is often limited by ineffective attitude control in turbulent environments, and rapid changes in wind gusts can cause sudden loss of control. Hence, aerodynamic sensing techniques are essential to address the guidance and control issues related to different flight operations and regimes.

More generally, the identification of freestream air data parameters is critical for any aircraft mission to achieve rapid flight envelope expansion and for the assessment of flight safety.

The flowfield characteristics include inflow-velocity magnitude, aerodynamic orientation (angle of attack), stagnation pressure, and sectional lift coefficient. These quantities are also integral to flight research and test data analysis, as they provide a deeper understanding of the actual aerodynamic phenomenon and its relationship to theoretical predictions. Wind-turbine aerodynamics is another application area where the determination of the aerodynamic characteristics plays a crucial role in optimum energy capture.

The conventional methods adopted for estimating these air data parameters, along with current developing technologies, are discussed at greater length in the following sections for different aerodynamic applications.

1.1 Aircraft Flight

Real time measurement of aerodynamic parameters during aircraft flight is critical for flight-control augmentation. The capability of determining the airdata parameters holds the key to accurately predict impending stall and for carefully guiding the aircraft throughout its flight envelope. Dangerous and unrecoverable stall situations like deep stall can be anticipated and successfully avoided by possessing accurate knowledge of the complex flow environment. Such stall-warning devices and algorithms mostly operate based on the information of local angle of attack and/or the stagnation-point location.

Conventionally, flow sensing and measurement systems employ intrusive airdata booms that extend beyond the local flowfield. These probes measure the freestream velocity by the direct stagnation of the flow using pitot-static tube and the angle of attack is estimated via calibrated mechanical vanes [1]. A typical aircraft boom that measures pitot-static pressures and flow angles is shown in Figure 1.1 and Figure 1.2 shows the air data boom of F-35C combat aircraft.

However, airdata probes are known to have measurement disadvantages. In addition to possible malfunctions, they are susceptible to inaccuracies in data estimation because of boom bending, alignment error, vibration and flow interference due to the probe itself. The size of such probes also limits its application in micro-air vehicles (MAV), where weight and cost are

Figure 1.1: Noseboom for measuring pressures and flow angles from Ref. [1]

critical design factors. To overcome these difficulties associated with intrusive airdata systems, the concept of flush airdata sensing (FADS) was developed as a cheaper alternative to the traditional techniques.

Figure 1.2: Air Data Boom of F-35C from http://commons.wikimedia.org/wiki/File:Cockpit_-and_Air_Data_Boom_F-35C.jpg

The FADS technology was first developed as the shuttle entry airdata sensing (SEADS) system for the space shuttle program by NASA, which demonstrated the feasibility of this concept [4]. Further endeavors were made to adapt this approach for transonic and subsonic flights [5, 6, 7]. The FADS system circumvents the disadvantages of conventional methods by deriving the complete aerodynamic state solely from the surface pressure measurements, thereby

eliminating the use of any intruding mechanical probes.

The pressure measurements in these investigations are acquired using a matrix of surface pressure ports located on the aircraft nose or on the wing sections in the vicinity of the leading edge. Figure 1.3 shows the nose cap of the F-18 research aircraft. In most of the research efforts, the pressure ports are typically accommodated in the vicinity of the fuselage nose-tip because the flow in this region remains unseparated over a greater angle-of-attack regime. But this approach has drawbacks associated with complex design of the avionics housed in the aircraft nose. The feasibility of locating the FADS system on a wing leading edge was demonstrated by researchers at NASA [8].

Processing this flow-field information gathered in real-time to yield the required quantities is the next challenging task that can be executed via different algorithm schemes. The most traditional method relies on the development of calibration curves and look-up tables. Comprehensive analysis of experimental or simulation data is required to select pressure orifice locations with adequate sensitivity towards particular airdata characteristics. Parameters are then formulated using combinations of suitable pressure values to translate their measurements to required aerodynamic features. For instance, two pressure taps in the vertical plane of symmetry normalized by dynamic pressure showed acceptable response towards flow angle [6]. A lot of these programs involved extensive in-flight and wind tunnel tests to compare the performance of FADS with conventional pitot-static tubes and mechanical vanes [9, 10, 11].

The second approach is to employ advanced mathematical models that are programmed to extract the flow-field parameters using the measurements obtained from the surface pressure ports. These methods are more popular in UAV development. A majority of these schemes are based on artificial neural networks that offer a means of modelling complex non-linear systems without the need of explicit knowledge about the functional correlation that exists between the input and output variables of the system. However, they require substantial amounts of training data to develop the relationship between the input pressure values and the required aerodynamic parameters. A comparison of the neural network method with look-up table approach

Figure 1.3: F-18 Systems Research Aircraft closeup of nose cap showing new flush air data system sensor holes from <http://www.dfrc.nasa.gov/Gallery/Photo/F-18SRA/HTML/EC97-43936-6.html>

Figure 1.4: Schematic of port locations on the RT-FADS nose cap from Ref. [2]

was demonstrated by researchers at the University of Leicester by mounting a FADS system on a micro-air vehicle and conducting wind tunnel experiments at low speed of Mach 0.07 [12, 13]. In 2013, Quindlen and Langelaan explored wind-vector sensing using flush pressure ports and established the viability of the method for dynamic soaring UAVs [14]. Similar other efforts have been made which mostly concentrate on optimizing the structure of these neural networks or developing other hybrid algorithms based on similar mathematical models such as Kalman filters [15, 16]. Though, these approaches show acceptable results, they need considerable amount of data for training. Also, they do not capture the actual physics of the flow and instead act as mathematically modelled “black-boxes” that give out air data parameters in response to input pressure measurements.

A third approach developed for flow analysis involves the formulation of aerodynamic models built on fundamental equations of potential flow for simplified geometrical shapes. Different models based on flow past a hemisphere and infinite two-dimensional cylinder have been developed and tested in various research efforts. The equations integrated in these algorithms attempt to capture the actual physical aerodynamic behavior. The pressure measurements are given as input to the model and the airdata quantities are estimated using nonlinear regression analysis [11]. A method based on such an approach for estimating the angle of attack of an airfoil from the a pressure distribution was presented by Brand [23]. This method is also discussed in detail in Chapter 3 and compared to the current research effort.

The flow sensing approach has also been demonstrated for cruise flap automation by McAvoy and Gopalarathnam [27], where the pressures measured on the airfoil surface were used to determine the best flap deflection angle. A trailing-edge cruise flap system, when automated, has the potential to result in low airfoil drag over wide life-coefficient range. The research effort used the nondimensional difference between the static pressures at two locations on the airfoil surface to set the cruise flap at the correct angle.

1.2 Wind Turbines

Wind-turbine aerodynamics is another application area, where one of the challenging issues is the real time determination of aerodynamic characteristics of the wind-turbine blade sections. Blade aerodynamic forces are responsible for power extraction and need to be maximized to ensure efficient energy capture. Wind turbines also face adverse time-varying wind gusts and perturbations during their life span. These undesirable overloads impose excessive stresses on the wind-turbine blades and gearboxes which appreciably shorten their service life. To overcome these issues and to maximize the power output, various passive and active control strategies like stall or pitch control and variable-speed concepts are employed for wind turbines. The most common of these is the blade pitch control which is generally regulated by optimizing the power output of the wind turbine. Generator rotor speed and power output are usually the controlling variables used in these techniques. Another alternative is the aerodynamic control technique, where the local flowfield information like the angle of attack and local dynamic pressure are used for better control of the blade pitch to improve power extraction as well as help to mitigate the effects of excessive blade aerodynamic overloads.

The complex flow environment in which wind turbines operate makes the determination of local dynamic pressure and angle of attack a challenging task. Current methods include direct measurements using excessive instrumentation like pressure taps all along the chord of the blade section and velocity probes to measure the local flow field characteristics [17]. As discussed in Section 1.1, these multi-hole flow intrusive devices suffer from plenty of drawbacks as they can alter the very flow field that they are trying to measure if they are located too close to the blade section. However, if they are too far, then they might not correctly imitate the flow-field they are trying to represent and this may also introduce a temporal phase between those measurements and the corresponding surface pressure data. The blade section has to be manufactured with pressure taps embedded in the blade along with sensors and pressure tubing installed inside the structure, thereby, adding much more to the cost of fabricating a single blade. Being inside the blade, these pressure sensors and related instrumentation become completely inaccessible,

making replacement of damaged components of the system highly difficult and often impossible without the complete tear-down of the blade structure.

1.3 Research Objectives

The current research derives inspiration from the benefits of the FADS systems discussed in Section 1.1 and introduces a novel method for extracting the aerodynamic quantities from the measured pressure values. In this approach, a simple aerodynamic model is formulated which is based on flow past a parabola and this model is used to develop a functional relationship between the input pressures and the output parameters. This model can be solved for the unknown coefficients by providing the measured pressure values at selected locations and using nonlinear curve-fit regression analysis.

The pressure values used in the algorithm need to be measured from the surface of the wing/blade section. This approach requires sensors to be accommodated within the blade making the fabrication process complicated and expensive as discussed in Section 1.3. Hence, this research also explores the feasibility of developing a minimally-intrusive leading-edge device that can be externally mounted on the leading edge of an airfoil. The device would eventually be able to house all the essential instrumentation to measure the pressure values and send the information wirelessly to the control unit, where that information can be analyzed in real time to derive the necessary aerodynamic parameters for effective control.

1.4 Thesis Outline

The second chapter discusses the inviscid equations of flow past a parabola that form the foundation of the complete aerodynamic model used in the algorithm along with the associated parameters and approximations. The chapter also covers the procedure followed for fitting a parabolic curve over the leading edge of any airfoil (cambered or symmetric). In Chapter 3, the methodology adopted in testing the algorithm is discussed along with the non-linear

regression method that acts as an effective tool for solving the aerodynamic model. It also explains the significance of the coefficients that appear in the equations and the process involved in extracting the required air data parameters from these coefficients. The method is tested for different airfoils using pressure data obtained from computations and from wind-tunnel tests. The results for all the test cases are summarized in Chapter 4. The concept of the leading-edge sleeve, followed by the design, fabrication and testing of the experimental prototype is explained in Chapter 5. The conclusions based on the results and the future scope of this research are presented in Chapter 6.

Chapter 2

Aerodynamic Model

This chapter presents the derivation of the aerodynamic model and describes the procedure adopted for defining a parabolic profile at the leading edge of the airfoil.

2.1 Formulation

The approach adopted to formulate the aerodynamic model is based on the analysis of flow past a simple geometrical configuration. The novelty of this research effort lies in approximating the leading edge of the airfoil as a parabola and then exploiting the available analytical solution for inviscid flow past a parabola to capture the dependence of the required airdata quantities on the airfoil surface pressures. The parameters of interest include the local freestream air velocity (U_∞), angle of attack (α), stagnation point location (s_s), and the local sectional lift coefficient (C_l). The discrete pressure measurements on the surface of the airfoil in the vicinity of the leading edge serve as the input variables that are used in the algorithm to return the local flow-field quantities.

2.1.1 Flow Past a Parabola

The argument of defining a parabola at the leading edge of the airfoil is justified by closely observing the leading edge region of any airfoil profile. Most airfoils designed for subsonic flow

have a finite leading-edge radius. The upper and the lower surfaces of an airfoil can be generally described by Equation 2.1 from Ref. [18].

$$y = \pm T_o x^{1/2} + C_1 x \pm T_1 x^{3/2} + C_2 x^2 + \dots \quad (2.1)$$

Where T_n and C_n are coefficients for the thickness and camber, respectively. And x is the distance measured from the leading edge to the trailing edge of the airfoil. The first coefficient T_o determines the nose radius (ρ_{le}) and the second coefficient C_1 corresponds to the initial camber (λ_{le}). The leading-edge radius (ρ_{le}) is proportional to the square of the thickness ratio (r) of the airfoil. In a small region in the proximity of the leading edge where $x = O(r^2)$, the airfoil can be expressed by the first two terms of Equation 2.1 using ρ_{le} and λ_{le} as follows:

$$y = \pm \sqrt{2\rho_{le}x} + \lambda_{le}x \quad (2.2)$$

This equation represents an inclined parabola and it can be described in rotated coordinates as:

$$\tilde{y} = \pm \sqrt{2\rho_{le}\tilde{x}} \quad (2.3)$$

where $\tilde{x}$ is the abscissa of the transformed axis. This equation exactly resembles the function that defines a parabolic curve. Hence, this assumption rationalizes the argument of representing the leading edge of any airfoil (symmetric or cambered) with a parabola. Here the leading-edge radius, ρ_{le} , and the initial camber, λ_{le} , of the airfoil are represented using the subscript ‘ le ’ to avoid any confusion with the radius of curvature, ρ_p , and the inclination of the parabola, λ_p , discussed later in Section 2.2.

The solution for inviscid flow past a parabola is discussed in literature by Van Dyke [18], who derives the exact velocity on a parabola in a uniform stream given as:

Figure 2.1: Parabola at an angle of attack.

$$\frac{v}{U_\infty} = \left(\frac{x}{x + \rho_p/2} \right) \left(1 \pm \frac{a'}{\sqrt{x}} \right) \quad (2.4)$$

Here, the positive sign refers to the upper surface and the negative sign is for the lower surface. The parameter a' plays a significant role as it directly relates to the stagnation-point location, as illustrated in Figure 2.1. As discussed later, this factor, a' , also serves as an effective calibration quantity to generate look-up tables for estimation of angle of attack and lift coefficient.

2.2 Fitting a Parabola

The velocity distribution, expressed in Equation 2.4, is valid for a parabola defined from the leading-edge parameters of the airfoil. This approximation restricts the extent of the parabola in the region where $x = O(r^2)$. However, current research attempts to stretch the limits of this parabola to envelope the airfoil upto a control point (which is approximately 6% of the chord in this research). The underlying purpose is to create adequate room for accommodating sufficient number of sensors that are required to measure the surface pressures. This objective is accomplished by adopting an approach, in which a “best-fit” parabolic profile is first generated

using the airfoil coordinates, based on the region of interest. The derived parabolic curve is first generated then compared with Equation 2.2 to deduce the leading edge quantities (ρ_p and λ_p) required in the aerodynamic model. These parameters are denoted with a subscript ‘ p ’ to denote that they are inferred from the equation of the parabola. The procedure followed in this approach is discussed for the both symmetric and cambered airfoils.

2.2.1 Symmetric airfoil

An iterative optimization scheme is applied that initializes with a starting design point (x_1, y_1) on the surface of the airfoil. This coordinate on the airfoil is used to generate a parabola by satisfying Equation 2.2 to obtain ρ_p as follows

$$\rho_p = \frac{y_1^2}{2x_1} \quad (2.5)$$

where λ_p is zero owing to the symmetric nature of the profile. The resulting parabolic curve is compared with the airfoil shape by calculating an error estimate (e) that tracks the difference between their y -coordinates for the range of x from 0 to 6% chord. The error estimate is the sum of squares of the deviation in the two profiles for all the points considered in the domain.

$$e = \sum (y_{airfoil} - y_{parabola})^2 \quad (2.6)$$

The process is reiterated with different design points until a minimum error estimate is reached. Essentially, the procedure conforms to an optimization algorithm that converges to an ideal design point, which results in the perfect parabola fit. For a symmetric Joukowski airfoil, the parabolic profile determined is shown in Figure 2.2.

2.2.2 Cambered airfoil

For a cambered airfoil, the procedure followed is analogous to that explained in the previous section. However, it accommodates certain modifications to calculate λ_p . The camber of the

Figure 2.2: Leading-edge parabola fit for a symmetric Joukowski airfoil

airfoil profiles is removed using the design environment in the XFOIL [19] code and a symmetric version of the same shape is generated. The process is triggered with an initial design point and ρ_p is calculated from Equation 2.5, while the inclination of the parabola is given by Equations 2.7 and 2.8.

$$\lambda = \frac{\Delta y}{\Delta x} \quad (2.7)$$

where Δx is the distance of the design point from the leading edge along the chord and Δy is given by

$$\Delta y = \left[\frac{y_{up} + y_{low}}{2} \right]_{cambered} - \left[\frac{y_{up} + y_{low}}{2} \right]_{symmetric} \quad (2.8)$$

Figure 2.3: Leading edge parabola fit for a cambered NLF airfoil

Here y_{up} and y_{low} correspond to the ordinate of the design point on the upper surface and lower surface of the airfoil respectively. The parabolic profile obtained using these calculated parameters is evaluated with the cambered airfoil analogous to the method discussed in the previous section. The parabola defined for a cambered natural-laminar-flow (NLF) airfoil is depicted in Figure 2.3.

The approach explained in this section clearly distinguishes between the actual leading-edge parameters of the airfoil and those based on the defined parabola. The latter quantities are plugged into the aerodynamic algorithm to determine the airflow characteristics. However, it is important to note the numerical difference between ρ_{le} and ρ_p , to understand that they

are of the same order. The values of ρ_{le} and ρ_p are tabulated in Table 2.1 for three SERI series airfoil (S8XX) for comparison, while Figure 2.4 shows the parabolic profiles for the same airfoils.

Table 2.1: Leading-edge radius comparison for SERI airfoils

Airfoil	ρ_{le}	ρ_p	λ_p
S809	0.01089	0.0141	0.0474
S812	0.00899	0.0127	0.1589
S825	0.0076	0.0133	0.1822

2.3 Inviscid equations of flow

The aerodynamic model is built on the theory of potential flow past a parabola, which was briefly highlighted in section 2.1. The velocity distribution on the surface of the parabola in Equation 2.4 is derived for inviscid flow at an angle of attack by Van Dyke [18] using conformal mapping. In a real viscous flow, the velocity varies normal to the surface until the edge of the boundary layer but the static pressure remains constant across the boundary layer irrespective of the velocity distribution. Because the flow is usually attached at the leading edge and the boundary layer is thin, it is reasonable to develop an aerodynamic model based on inviscid pressure distribution over a parabola to approximate the flow in the leading-edge region of the airfoil. The pressure distribution is usually expressed in terms of a non-dimensionlized pressure coefficient (C_p) given by Equation 2.9.

$$C_p = \frac{P - P_\infty}{\frac{1}{2}\rho U_\infty^2} \quad (2.9)$$

Where P is the local static pressure, P_∞ is the freestream static pressure and ρ is the density of the fluid. The velocity distribution can also be related to C_p using Bernoulli's equation as

(a) S809

(b) S812

(c) S825

Figure 2.4: Parabola defined for SERI airfoils

follows

$$C_p = 1 - \left(\frac{v}{U_\infty} \right)^2 \quad (2.10)$$

Equation 2.9 and 2.10 can be used to express $P - P_\infty$ as given by Equation 2.11.

$$P - P_\infty = \frac{1}{2} \rho U_\infty^2 \left[1 - \left(\frac{v}{U_\infty} \right)^2 \right] \quad (2.11)$$

$\frac{v}{U_\infty}$ in this relation can be substituted from the velocity distribution obtained for the inviscid flow past a parabola in Equation 2.4. With some algebraic manipulation, the relationship can be expressed in terms of the abscissa of the inclined axis as follows

$$P - P_\infty = \frac{1}{2} \rho U_\infty^2 \left[1 - A^2 \left(\frac{1}{\tilde{x} + \frac{\rho_p}{2}} \right) \left(\sqrt{\tilde{x}} \pm a' \right)^2 \right] \quad (2.12)$$

This relationship is valid near the leading edge of the airfoil but can be successfully used upto approximately 6% chord because the parabola has been matched over this region of the airfoil. Also, as shown in [18], the oblique abscissa $\tilde{x}$ can be related to the original abscissa x by:

$$\tilde{x} = x \pm \lambda_p \sqrt{2\rho_p x} \quad (2.13)$$

Substituting for $\tilde{x}$ in Equation 2.12, the final relationship between the surface pressure and the respective x -location results in Equation 2.14.

$$P - P_\infty = \frac{1}{2} \rho U_\infty^2 \left[1 - A^2 \left(\frac{1}{x \pm \lambda_p \sqrt{2\rho_p x} + \frac{\rho_p}{2}} \right) \left(\sqrt{x \pm \lambda_p \sqrt{2\rho_p x}} \pm a' \right)^2 \right] \quad (2.14)$$

Here, P is the static pressure on the surface of the airfoil measured at a known x location. The freestream static pressure P_∞ and the air density are assumed to be easily obtainable from the known atmospheric conditions. Leading edge radius, ρ_p , and slope, λ_p , are the design parameters, defined in Section 2.2, based on the parabola-airfoil matching. U_∞ is the local freestream

velocity that the airfoil encounters and a' is the parameter that determines the location of the stagnation point. Another coefficient of significance is A , which is entirely influenced by the airfoil shape and is a pre-determined design input for the algorithm. It is empirically established that this factor A is dependent on the location of the pressure points considered.

The stagnation-point location can be directly derived from the knowledge of a' without any calibration. The angle of attack and the lift force coefficient are calibrated with values of this parameter, a' , using computational data obtained from XFOIL code. The dependency of these flow-field parameters on a' is discussed at greater length in Chapter 3.

2.4 Chapter 2 Summary

The inviscid solution of flow past a parabola was presented and the equation derived by Vandyke [18] for a parabola is used as the foundation to build the aerodynamic model. Defining a parabola to fit the leading edge of any airfoil upto around 6% chord was discussed. The parabolic profile generated is essential in acquiring the quantities used in the final model, i.e. the nose radius for the parabola ρ_p and the inclination λ_p . The velocity distribution is converted to surface pressures and a final expression is obtained relating the surface pressures to the x -locations along with important coefficients that define the flowfield information.

Chapter 3

Algorithm Development

This chapter provides a detailed description of the leading-edge flow sensing algorithm (LEFS) developed in this research. The approach adopted to solve the aerodynamic model is presented to determine the flowfield parameters from the leading-edge pressures.

3.1 Nonlinear Regression

The analytical equations for flow past a parabola are used to establish a valid functional relationship between the aerodynamic parameters expressed in the form

$$\vec{P} = F(\vec{\phi}, \vec{X}) \quad (3.1)$$

where the vector $\vec{P}$ is the difference between the static surface pressure at the sensor locations and the freestream static pressure. The vector $\vec{X}$ corresponds to the respective locations on the airfoil profile and $\vec{\phi}$ includes the aerodynamic parameters that are required to be estimated.

Ideally, the function F should be invertible, which would facilitate the direct estimation of the output variables by feeding in the pressure differences. But considering the nature of the analytical function expressed in Equation 2.14, deriving the invertibility becomes cumbersome, if not impossible. An alternative approach to tackle this problem by adopting numerical methods

which employ nonlinear regression techniques.

The equations discussed in Section 2.3 form the structure of the aerodynamic model and is based on the airfoil surface pressure distribution in the vicinity of the leading edge. This equation can be numerically solved by formulating the problem as a minimization of the sum of the squares of the errors at each pressure point location on the airfoil. The error is defined as the difference between the pressures measured by the sensors and the pressures estimated from the function F .

$$\epsilon_i = P_i - F(x_i, a', U_\infty, A) \quad (3.2)$$

$$S = \sum_{i=0}^n \epsilon_i^2 \quad (3.3)$$

Here S is the residual function that is minimized to solve for the quantities a' and U_∞ . And n is the number of pressure data points measured on the airfoil surface. So, a total of n equations are derived and solved for extracting the unknown coefficients in the current work. Numerous optimization techniques are available that follow iterative procedures to solve such regression problems. The current algorithm uses the nonlinear least-squares function (*lsqnonlin*) from the optimization toolbox in MATLAB to estimate the aerodynamic variables. This function uses the trust-region-reflective method for the non-linear curve fitting operation, which is based on the interior-reflective Newton method described in Refs. [20, 21] and briefly discussed in the following subsection.

3.1.1 Trust-Region-Reflective Method

Iterative optimization techniques can be broadly studied under two categories: line-search methods and trust-region methods. Line-search methods calculate a search direction in each iteration and then advance along this path to solve for a better solution, which is utilized in the following iteration to identify the new search direction. The trust-region algorithms construct an approx-

imate model near the current best point and then solve the model to yield a better solution. The process is repeated until the final best solution is achieved. However, the approximate models are only valid in a “trust” region in the vicinity of the current iterate. Generally, a merit function is devised to track the performance of the model in the trust region. Consequently, it can limit or enlarge the size of the trust region. A review of different trust-region algorithms is presented by Yuan in Ref. [22]. The approach used in the *lsqnonlin* function is developed by forming a model with an appropriate quadratic function and scaling matrix. These methods have strong convergence properties, in addition to greater reliability and robustness. They also have the capability to effectively handle bound constraints.

3.2 Methodology

The methodology is summarized with the aid of a flowchart shown in Figure 3.1. Five pressure values sensed on the airfoil surface in the neighbourhood of the leading edge are used as inputs in the leading-edge flow sensing algorithm to return a' and U_∞ by employing nonlinear regression techniques. Here U_∞ is the local freestream velocity and the parameter a' is used to fetch the angle of attack and lift coefficient from pre-developed look-up tables. Hence, for any freestream condition, these are the only two variables calculated directly by the algorithm in real-time. This implies that a minimum of two pressure points are sufficient to determine the system at any flow condition.

An important input parameter required in the aerodynamic model is the parameter ‘ A ’. This quantity is a calibration variable that is strongly airfoil specific and depends on the pressure point locations under consideration. It is estimated empirically using simulation data.

A notable approximation in this simplified model is that it is derived from the inviscid equations of flow. Consequently, the direct input of all the measured pressure values in the algorithm for any aerodynamic orientation might not yield accurate results. A critical element of this LEFS algorithm is to use a strategic switching logic that exploits the nature of the pressure distribution on the airfoil. This is the prime reason for using five pressure points

Figure 3.1: Basic methodology of the LEFS algorithm

that envelope the region of interest near the leading edge. The following section presents a comparison of the viscous and inviscid pressure values to better understand the need for the switching operation.

3.2.1 Comparison of Inviscid and Viscous Pressure distribution

The pressure distribution calculated on the airfoil surface using inviscid potential flow solutions do not exactly match the actual pressures observed due to the real aerodynamic phenomenon. An example of deviation between inviscid and real pressure distribution in the vicinity of the leading edge is when there is a laminar separation bubble at the leading edge on the suction surface. A comparison of viscous and inviscid pressure behavior simulated from XFOIL is presented in this section. The inviscid formulation is based on a simple panel method with a linear distribution of surface vorticity on each panel. The solution is obtained by satisfying the boundary condition of zero normal flow on each panel and the Kutta condition at the trailing edge. The viscous analysis in XFOIL accommodates the effects of boundary layer by coupling the incompressible panel method with the integral boundary layer formulation via the surface transpiration model. This Viscous-Inviscid Interaction allows XFOIL to effectively capture

transitional flows (laminar and turbulent), predict laminar separation bubbles and essentially provide good representation of actual aerodynamic behavior.

The inviscid and viscous pressure values are generated using XFOIL for the S809 airfoil in Figures 3.2 and 3.3, for both positive and negative angle of attack regimes. For $\alpha > 0$, the stagnation point lies on the lower surface which exhibits positive pressures, while the upper surface experiences suction. The difference between the viscous and inviscid cases on the upper surface is clearly evident from the plots. This discrepancy is induced as a result of the boundary layer growth on the airfoil in the viscous case. The on-coming flow observes an airfoil contour slightly different from the actual shape because of this boundary layer formation. The “effective” airfoil shape continues to change at higher angles of attack as the boundary layer thickens because of adverse pressure gradients developing over the airfoil. This behavior is responsible for the increasing deviations at higher angles of attack. At these conditions, flow separation also plays a major role after the airfoil stalls. Similarly, greater deviations are observed on the lower surface for $\alpha < 0$, when the stagnation point moves on the upper surface and suction pressures are experienced on the lower surface.

However, an important observation is that in all flow conditions there is a greater matching between the viscous and inviscid pressures near the stagnation-point location. This feature can be attributed to the fact that the stagnation streamline intersects the airfoil normally at the stagnation point and the boundary-layer formation commences from this point onwards. Hence, the boundary layer is relatively very thin in this region, thereby exhibiting closer resemblance to the inviscid pressures in the vicinity of the stagnation point. This phenomenon is exploited by the switching operation integrated in the LEFS algorithm. The idea is to have a qualitative estimate of the stagnation-point location from the five pressure measurements and then target only those pressure data points that reside in the neighborhood of the stagnation point. The selected pressure values are then given as inputs to the LEFS algorithm.

(a) AoA = 0 deg

(b) AoA = 2 deg

(c) AoA = 4 deg

(d) AoA = 6 deg

(e) AoA = 8 deg

(f) AoA = 10 deg

Figure 3.2: Viscous and inviscid pressure comparison for S809 airfoil at positive angles of attack

(a) AoA = -2 deg

(b) AoA = -4 deg

(c) AoA = -6 deg

(d) AoA = -8 deg

(e) AoA = -10 deg

(f) AoA = -12 deg

Figure 3.3: Viscous and inviscid pressure comparison for S809 airfoil at negative angles of attack

3.2.2 Switching Operation

At the stagnation point, the streamline incidents normally on the airfoil surface and divides the flow on the upper surface from that on the lower surface. Since the flow is brought to rest, it exhibits maximum static pressure on the airfoil at this point. The LEFS algorithm takes input from the pressures sensed by the 5 pressure taps tagged as P_1, P_2, P_3, P_4 and P_5 ; where P_1 is aft of P_2 on the upper surface, while P_5 is aft of P_4 on the lower surface, and P_3 is at the leading edge as shown in Fig 3.1. The maximum pressure response among these measurements provides substantial indication that the stagnation-point is in the vicinity of that pressure tap.

The switching operation also takes advantage of the fact that stagnation-point location is very accurately predicted by the aerodynamic model. This feature is exploited in transitioning between the appropriate pressure points used when the angle of attack becomes greater than 8 or 9 degrees and less than -6 degrees. Numerous test cases were considered from XFOIL and the pressure variation with angle of attack was carefully analyzed to devise this switching logic. Since the knowledge of local angle of attack is unknown initially, the stagnation-point location is used as a suitable trigger to switch between the best pressure points for accurate velocity prediction. Figure 3.4 explains the criteria for pressure tap selection in the switching logic. For the S809 airfoil, the x_{stag} for switching was defined at 0.02 for upper surface and 0.01 for lower surface. Similarly, for other airfoils also, this location is determined using computational test cases.

(a) If $P_{max} = P_1$, then P_1 and P_2 are used in the algorithm

(b) If $P_{max} = P_2$, then P_1 and P_2 are used if $x_{stag} > 0.01$, else P_1, P_2 and P_3 are employed

(c) If $P_{max} = P_3$, then P_2, P_3 and P_4 are used in the algorithm

(d) If $P_{max} = P_3$, then P_2, P_3 and P_4 are used in the algorithm

(e) If $P_{max} = P_4$, then P_4 and P_5 are used if $x_{stag} > 0.02$, else P_3, P_4 and P_5 are employed

(f) If $P_{max} = P_5$, then P_4 and P_5 are used in the algorithm

Figure 3.4: Selecting the pressure points based on the maximum pressure information

3.3 Estimating Inflow Velocity and Stagnation-Point Location

The local freestream velocity is directly obtained by solving the aerodynamic model. The second output of the LEFS algorithm is a' , which directly relates to the stagnation-point location on the airfoil surface. The relationship is given in Equation 3.6 and 3.7.

$$x_{stag} = a'^2 \quad (3.4)$$

$$y_{stag} = -a' \sqrt{2\rho_{le}} \quad (3.5)$$

Here, a' is measured from the “ideal” angle, i.e. the angle at which the stagnation-point coincides with the vertex of the parabola. For a symmetric airfoil, this happens at $\alpha = 0$. However, in the case of a cambered airfoil, the stagnation-point coincides with the vertex at a small negative angle. Hence, the a' values are offset by a factor that corresponds to that ideal angle. Therefore, a compensating factor δ based on α_{ideal} is added to the a' values to correct the offset for calculating the stagnation-point locations. The equations then take the following form.

$$x_{stag} = (a' + \delta)^2 \quad (3.6)$$

$$y_{stag} = -(a' + \delta) \sqrt{2\rho_{le}} \quad (3.7)$$

where δ is derived using test cases and is a factor of the ideal angle of attack of the airfoil.

3.4 Estimating Angle of Attack and Lift Coefficient using Look-up Table Approach

The LEFS algorithm employs the look-up table (LUT) approach to estimate the AoA and C_l from the freestream airdata. The parameter a' obtained from solving the aerodynamic model

strongly relates to the AoA and can be effectively used to generate the look-up tables. The algorithm is run for several Reynolds-number cases to calculate a' for the entire regime of flow encompassing both positive and negative stall angles. In the current work, the pressure data for these test runs is acquired from XFOIL. Two separate 2-dimensional look-tables are generated with one accommodating the AoA vs. a' values and the other having the AoA vs. C_l data. For any airfoil under consideration, such tables can be generated from computational data and integrated into the algorithm prior to the application. In real-time operation, the flow velocity solved from the aerodynamic model at any instant is translated to the local Reynolds number, which is then used along with a' to extract the angle of attack and lift coefficient from the tables.

3.5 Estimating Angle of Attack using Stagnation-Point Relation

An alternative approach for estimating the angle of attack of an airfoil from the pressure distribution was explored by Brand [23], as a part of the ‘Field Rotor-Aerodynamics project undertaken by the Netherlands Energy Research Foundation (ECN). An attempt was made in this research to tackle the same problem of estimating the angle of attack from the pressure distribution, but by adopting a different strategy. This section summarizes the method used by Brand with the objective of employing his approach for a comparative analysis.

The method of Brand exploits the geometrical orientation of the dividing streamline incident on the airfoil contour. The stagnation point is located on the airfoil surface along the dividing streamline, and the angle of attack is defined as the angle between the freestream and the airfoil chord. This implies that a correlation exists between the chordwise position of the stagnation-point and the angle of attack of the airfoil. The intersection of the dividing streamline with the airfoil chord $(x_m, 0)$, as illustrated in Figure 3.5, was required to arrive at the relationship given by Equation 3.8.

Figure 3.5: Flow past an airfoil at an angle of attack

$$\tan \alpha = \frac{-y_{st}}{x_m - x_{st}} \quad (3.8)$$

Here, (x_{st}, y_{st}) is the position of the stagnation point on the airfoil profile. The chordwise coordinate x_m of the intersection point was determined empirically. It was shown that x_m is a weak function of α and depends on the thickness of the airfoil profile. Based on this observation mean values of x_m were utilized in his algorithm for the two regimes: $\alpha > 0$ and $\alpha < 0$. So, the stagnation-point location remains the only parameter required for the working of this concept. The practical pressure distribution on the airfoil surface is not continuous, but rather discrete. This implies that the maximum pressure point is not an accurate estimate of the stagnation-point location. The method devised by Brand employs a mathematical relationship to approximate the pressure distribution near the leading and solve that function for the stagnation point using the maximum pressure and the adjacent pressure measurements. This approach still requires a fairly dense spacing of pressure taps for greater accuracy.

Figure 3.6: x_m values for different Reynolds numbers for S809 airfoil

The LEFS algorithm developed in the current work determines the exact location of the stagnation point with just five pressure measurements near the leading edge. The ready availability of an accurate estimate of the stagnation-point location from the current work provided the motivation to evaluate the stagnation-point relation discussed by Brand and compare the technique with the look-up table approach adopted in the current LEFS algorithm. The values of x_m were determined for the S809 airfoil in this research for different Reynolds numbers, and they are plotted in Figure 3.6. It is seen that the x_m value does not change much with angle of attack. The mean values of x_m taken for the S809 airfoil for further analysis are: $x_m = 0.1768$ for $\alpha > 0$ and $x_m = 0.1681$ for $\alpha < 0$. So, this stagnation-point approach proposed by Brand was integrated as a modification to the LEFS algorithm. The results comparing both methods, LEFS and LEFS-mod, for α -prediction are presented in Chapter 3.

3.6 Chapter 3 Summary

The methodology adopted for solving the aerodynamic model was presented in this chapter. Nonlinear regression techniques were introduced as effective tools for determining the system

with a focus on the trust-region-reflective approach. The input and output parameters of the algorithm were discussed, along with the switching logic that tracks the stagnation-point location to select the most appropriate pressure measurements as inputs for the algorithm. Finally, relationships involving the aerodynamic parameters were defined and the methods employed for extracting the angle of attack and lift coefficient from the output parameter, α' , were explained. In the next chapter, the results from the algorithm are presented for the S809 airfoil at various operating conditions.

Chapter 4

Results

This section presents the results obtained by the application of the LEFS algorithm to steady-flow experimental data for a S809 airfoil model. The method was further assessed for rotating blade experimental data for the same airfoil gathered as a part of the Unsteady Aerodynamics Experiment (UAE) test campaign conducted by National Renewable Energy Laboratory (NREL).

4.1 Application of the LEFS Algorithm to Steady-Flow Experimental Results

The LEFS algorithm was applied to wind tunnel experimental data for SERI series airfoils. The pressure distribution was available for the S809 (21% thickness) airfoil model tested under steady-flow conditions in the subsonic wind tunnel of the Ohio State University Aeronautical and Astronautical Research Laboratory (OSU/AARL) [24]. The Reynolds numbers used in the study for steady state conditions were 0.75, 1, 1.25 and 1.5 million. The airfoil model was equipped with pressure taps on the surface all along the chord, but only five pressure port measurements were utilized for testing this technique. The pressure values were extracted from the upper surface at 1.4% and 4.5% chord, on the lower surface at 1.5% and 5.6% chord and

from the leading edge. These data points were fed into the algorithm to estimate the freestream velocity, angle of attack, and the lift coefficient.

4.1.1 Calibration and Look-Up Table Development

The initial setup for running the algorithm consists of developing the look-up tables and estimating the value of the parameter ‘ A ’ in accordance with the pressure point locations. Pressure values were extracted from the XFOIL computations at the required Reynolds numbers. This data was used to determine the values of ‘ A ’ for different combinations of pressure tap locations, as required by the switching algorithm. In this calibration setup, the freestream velocity was treated as a known input to obtain ‘ A ’ and these values are tabulated in the Table 4.1.

Table 4.1: Values of the parameter ‘ A ’ used for different pressure tap locations

Pressure point locations	A
P_1 and P_2	1.196
P_1 , P_2 and P_3	1.108
P_2 , P_3 and P_4	1.011
P_3 , P_4 and P_5	1.101
P_4 and P_5	1.039

Here, the pressure tap locations are labelled in the same manner as explained in Section 3.2.2, starting with the aft pressure tap on the upper surface and numbering them respectively until the aft pressure point on the lower surface.

The next step involved running the leading-edge algorithm for the entire flow regime to prepare the look-up tables. The Reynolds numbers considered were 0.75, 1, 1.25 and 1.5 million. A table of α vs. C_l was generated using XFOIL for $-16^\circ < \alpha < 18^\circ$. The a' values calculated by the algorithm for these test conditions were compiled in a second table for all the four

Reynolds number cases. The look-up tables designed by this procedure were based on just the computational information acquired from the XFOIL code. These tables were then integrated in the algorithm as reference data and the algorithm was tested with wind tunnel experiments.

4.1.2 Results for S809 airfoil

The LEFS algorithm was employed to estimate the local inflow velocity, angle of attack and the lift coefficient from the five pressure measurements extracted from the experimental data. These estimated aerodynamic quantities were compared to the actual values to evaluate the validity of the algorithm. The velocity predictions for the four Reynolds number cases are co-plotted with the actual freestream velocities in Fig A.8. The results match very closely to the real values with a RMS error of 0.5418 m/sec for the highest Reynolds-number case of 1.5 million.

The estimation of the angle of attack and lift coefficient for all the four cases is shown in Figures 4.2 to 4.5. The angle-of-attack plots present a comparative assessment of the LUT approach adopted in the leading-edge algorithm with the stagnation-point relation approach developed by Brand. The RMS errors for both these approaches are tabulated in Table 4.2.

Table 4.2: RMS errors for angle of attack estimations

Reynolds number	LEFS RMS error	LEFS-mod RMS error
0.75 million	2.159°	2.474°
1 million	1.6646°	2.0548°
1.25 million	1.5326°	1.912°
1.5 million	1.4884°	1.9249°

The angle of attack prediction by the LUT approach shows excellent matching with the actual α values within the range $-12^\circ < \alpha < 10^\circ$, with an absolute error of 0.689° in this range for the case of 0.75 million Reynolds number. Similarly, an absolute error of 0.481° , 0.545° , and

0.443° is observed within the same α -range for Reynolds numbers of 1, 1.25, and 1.5 million, respectively. This α - interval corresponds to the pre-stall regime of the flow. Greater deviations in the angle of attack estimation beyond the stall angle is attributed to the flow separation. The stagnation-point method shows analogous behavior by exhibiting close resemblance to the actual α for pre-stall conditions with larger errors induced in the post-stall domain. The absolute errors for this method in the range $-12^\circ < \alpha < 10^\circ$ are 1.014° , 0.954° , 0.974° and 1.029° for Reynolds number of 0.75, 1, 1.25, and 1.5 million respectively. It is evident from the results that the LUT approach outperforms the stagnation-point method.

The stall behavior is presented in the lift-estimation plots, where a comparison of the estimated C_l is made with the C_l values obtained from wind tunnel tests and XFOIL computations. The C_l estimation from the algorithm shows excellent correlation with the wind tunnel data.

Figure 4.1: Velocity estimation for different Reynolds numbers

(a) Angle of attack estimation from LEFS and LEFS-mod

(b) Lift coefficient estimation

Figure 4.2: Case - 1: Re = 0.75 million

(a) Angle of attack estimation from LEFS and LEFS-mod

(b) Lift coefficient estimation

Figure 4.3: Case - 2: Re = 1 million

(a) Angle of attack estimation from LEFS and LEFS-mod

(b) Lift coefficient estimation

Figure 4.4: Case - 3: Re = 1.25 million

(a) Angle of attack estimation from LEFS and LEFS-mod

(b) Lift coefficient estimation

Figure 4.5: Case - 4: Re = 1.5 million

4.2 Application of the LEFS Algorithm to NREL S809 Rotating Blade Experiments

The Unsteady Aerodynamics Experiment (UAE) was conducted by National Renewable Energy Laboratory (NREL) with the primary objective of quantifying the full-scale, three dimensional aerodynamic behavior of horizontal-axis wind turbines [3]. A 10-meter diameter research wind turbine was tested in the world's largest wind tunnel located at the NASA Ames Research Center in Moffett Field, California. The tunnel test section is 24.4 m x 36.6 m (80 ft x 120 ft). The wind turbine was extensively instrumented to accurately measure several aerodynamic quantities for numerous test sequences. Figure 4.6 shows the two bladed turbine in the wind-tunnel test section.

Figure 4.6: The research turbine with instrumented blade mounted in the wind-tunnel test section.

Blade surface pressures, angle of attack and local dynamic pressure were measured at five radial locations on one blade. Five-hole probes with a spherical tip were used to provide the dynamic pressure, local flow angle, and spanwise flow angle measurements at a distance of 80% chord ahead of the blade. These probes were mounted at five radial locations, 34%, 51%, 67%, 84% and 91% span. The surface pressure distribution was captured with an array of 22 pressure taps instrumented at five primary radial stations: 30%, 46.6%, 63.3%, 80% and 95% span.

Figure 4.7: Pressure taps on the blade surface and five-hole probe locations from Ref. [3]

Again, the pressure measurements from only 5 pressure taps (out of 22) were acquired from the dataset and given as input into the LEFS algorithm. The selected pressure taps were located at 2% and 4% chord on the upper surface, 2% and 6% chord on the lower surface and on the leading edge of the S809 airfoil. The radial station selected for analysis on the wind turbine blade was located at 46.6% span and had a chord length of 0.627 meters. The primary reason for analyzing this mid-span location was to observe the effect of 3-D flow to some extent on the functionality of the leading-edge algorithm. Three wind speed cases were studied in this research with the wind turbine blades rotating at a constant angular velocity of 70.38 rpm. The data from all the sensing units was recorded after every 1.9 milliseconds for a complete time duration of 306 seconds. These values were averaged over each 1-second interval and used in the algorithm.

4.2.1 Calibration and Look-Up Table Development

The application of the LEFS algorithm to the rotating blade data requires the same calibration procedure as discussed in Section 4.1.1. The wind turbine blade used in the NREL test campaign was a two-bladed turbine with tapered and twisted blades that use the S809 airfoil from the root to the tip. Although, the same airfoil was employed in the blade section as used in the OSU airfoil model, the pressure tap locations were different in these tests. Hence, the A parameter was estimated again using the simulation data from XFOIL for these new pressure tap locations. Table 4.3 shows the values of A used for the blade sections. The look-up tables were also developed by the same approach that was described in Section 4.1.1. The datasets from the test sequences in Ref. [3] provide variations of the normal force coefficient (C_n). To compare with the dataset listings, the lift coefficient in the look-up tables were transformed into normal force coefficients. The Reynolds number used in the look-up tables were 0.75, 1, 1.25 and 1.5 million.

Table 4.3: Values of the parameter ‘ A ’ used for different pressure tap locations

Pressure point locations	A
P_1 and P_2	1.237
P_1 , P_2 and P_3	1.115
P_2 , P_3 and P_4	1.028
P_3 , P_4 and P_4	1.103
P_4 and P_5	1.070

4.2.2 Results for Rotating S809 Blade Section

The pressure measurements at the 46.6% radial section were used in the algorithm to predict the local freestream velocity at that section. The actual inflow velocity observed by the blade was derived from the dynamic pressure measurements, which were obtained using two different techniques. The first technique used the information from the 5-hole probe that was corrected for centrifugal and hydrostatic effects. The second approach used the stagnation-point pressure from the pressure tap locations. The resolution of the pressure taps on the lower surface was assumed to be sufficient enough to extract the maximum positive pressure. The results for velocity estimations are summarized in Figures 4.8-4.10. These figures also present the normal force coefficients predicted from the LEFS algorithm coplotted with the actual values measured during the tests.

The angle-of-attack range for the test sequences extended to very high angles beyond stall. The implementation of the algorithm was focussed only in the domain that spanned from -20 degrees to 20 degrees because the LUTs were generated for this range. However, since the velocities are determined as a direct output, the algorithm could be assessed even at higher angles of attack for estimating the inflow velocities. The overall RMS errors in velocity predictions were 0.48, 0.408, and 0.897 m/sec for wind speeds of 6, 10, and 15 m/sec respectively. The LEFS algorithm also performs well for the C_n estimations, as displayed in the results.

(a) Velocity estimation

(b) Normal force coefficient

Figure 4.8: Case - 1: Wind speed = 6 m/sec

(a) Velocity estimation

(b) Normal force coefficient

Figure 4.9: Case - 2: Wind speed = 10 m/sec

(a) Velocity estimation

(b) Normal force coefficient

Figure 4.10: Case - 3: Wind speed = 15 m/sec

4.3 Chapter 4 Summary

The effectiveness of the LEFS algorithm was assessed by applying it to wind-tunnel data obtained from different experimental test campaigns. The results were presented for the estimation of the aerodynamic data from just the five pressure tap measurements near the leading edge of the airfoil. The air-data quantities estimated from the LEFS algorithm were compared with the actual values measured by instrumentation in the wind-tunnel experiments. The look-up tables were generated using only the simulation data extracted from the XFOIL code. It was seen that the algorithm functions excellently in the pre-stall regime for both rotating and non-rotating airfoils. Larger errors at and beyond stall were seen in the α and C_l calculations partially because of the flow separation after stall and partly because of the discrepancy between the experimental and XFOIL simulation data.

Chapter 5

A Minimally-Intrusive Approach for Measuring Leading-Edge Pressures

5.1 Pressure Sensing Technologies

The focus of this research has been primarily on developing the methodology to determine the flowfield characteristics from the measured pressure values. But the applicability of the LEFS algorithm is entirely dependent upon the accurate measurement of the surface pressure data. In this regard, it is worth briefly reviewing the traditional techniques adopted and the currently-developing sensor technologies available to make such measurements.

Conventionally, pressure orifices are drilled on the surface of the wing/blade section all along the chord to capture the pressure distribution with sufficient resolution. These holes are pneumatically connected to multi-point pressure scanning modules, which are housed inside the blade. The advantage of this design is that the orifices are flush on the surface and accurate measurements can be made without disturbing the flowfield. But the greatest drawback for this setup is the cost involved in the installation and the complexity in fabricating the blade with such extensive built-in instrumentation. The pressure taps also tend to easily get clogged and the sensors can be damaged due to harsh environmental conditions. The replacement of any

damaged units becomes a very daunting task because of the inaccessibility of the components and may even necessitate the complete tear-down of the blade structure.

Alternatively, pressure measurements using Micro-Electro-Mechanical Systems, or MEMS, technology offer greater advantages and can effectively resolve the issues associated with conventional pressure taps. These miniaturized transducers have a diaphragm or membrane that deflects on application of pressure. The deflection or vibration can be measured by employing different types of strain-sensing techniques. Piezoresistive sensors make use of the piezoelectric effect to measure the pressure differential across a thin membrane, while capacitive sensors detect the change in capacitance between the membrane and an underlying electrode. These transducers can be fabricated using micro-manufacturing techniques and the size of these micro-sensors favor their application on the surface of the wing/blade section without causing any flowfield perturbations. Numerous innovative research efforts are being undertaken by the MEMS community to evolve this into a more mature sensing technology. Mohamed et al. [25] presents an excellent review of these emerging sensors that derive inspiration from biological systems for application in Micro Air Vehicles (MAVs).

Even with all the progress, the objective of developing an integrated device, complete with sensing units and wireless transmission capability, is still a vision for the future. The desire for such a capability provided the motivation to devise the concept of a leading-edge sleeve that can work with currently available commercial sensing technology.

5.2 Leading-Edge Sleeve

The objective of the sleeve concept is to develop a self-contained minimally-intrusive leading edge-device that can be externally mounted on a blade section and houses miniaturized sensors along with the necessary modules for wireless transmission. The outside of the sleeve will have pressure taps with channels leading to the sensors embedded inside the sleeve. Another possibility is to have integral diaphragms on the surface of the sleeve covering a sealed cavity. The inside of the diaphragm will have micro-sensing devices to measure the deflection of the

diaphragm in response to external pressure. Such a design will avoid troubles due to clogged pressure taps and tubing. The sleeve can be removed and replaced easily. Such a concept provides the capability to effortlessly mount the sleeve on any existing blade without requiring access or modifications to the inside of the blade.

In this research, only preliminary experiments were conducted at the NC State subsonic wind tunnel to characterize the intrusiveness and effectiveness of the leading-edge sleeve concept in terms of the aerodynamic behavior without venturing into the MEMS technology and the integration of sensors on the surface of the sleeve.

5.2.1 Design and Fabrication

The primary requirement of the leading-edge sleeve is to closely mimic the aerodynamic behavior of the leading edge of the base airfoil on which the sleeve is designed to be mounted. The best possible solution to accomplish this objective is by considering a scaled version of the base airfoil profile and use that to generate the sleeve that can externally slide onto the leading edge of original airfoil. The size of the sleeve may vary depending upon the region of interest in the vicinity of the leading edge. Numerous design iterations were required to reach a final form that met the size specifications for the pressure taps and also was closely representative of the base airfoil shape.

The final working design was made by scaling the base airfoil profile 1.5 times about a point at 20% of the chord. The region of interest for this research extends upto the first 6% of the chord. Aft of this point, the sleeve is smoothly splined back onto the base airfoil. A total of 11 pressure ports were accommodated on the upper and lower surface, with one at the leading edge.

The design was first created using Solidworks and is shown in Figure 5.1. The sleeve design had pressure taps of 2 mm diameter at 1.2%, 2.2%, 3.4%, 4.45% and 6% chord on both the upper and the lower surface of the airfoil. It envelopes the upper surface region until 30.24% chord and extends until 26.25% of the chord on the lower surface, covering a total chord-wise

(a) Isometric view

(b) Side view

Figure 5.1: Leading-edge sleeve model in Soliworks.

length of 9.5 cm and spanwise distance of 5 cm. The leading edge of the sleeve extends 9.37 mm into the flow and the sleeve thickness is restricted to a maximum of 6 mm on the upper surface and 5 mm on the lower surface.

The model was fabricated using a professional-level 3D printer, Stratasys uPrint SE Plus. This 3D printer uses Fused Deposition Modeling (FDM) Technology to build models and functional prototypes with ABSplus thermoplastic. The model was printed using a soluble support material which was dissolved after the process. The uprint SE prints with a minimum layer thickness of 0.330 mm. The final fabricated model, shown in Figure 5.2, was fixed with tubula-

(a)

(b)

Figure 5.2: 3D printed leading-edge sleeve model.

tions that could be connected to the sensors placed outside the wind tunnel.

5.3 Wind-Tunnel Testing

Wind-tunnel experiments were conducted with the leading-edge sleeve mounted on a base airfoil model with the primary objective of characterizing the intrusiveness of the sleeve. The pressure

measurements were acquired from the sleeve pressure ports for comparison with the airfoil model ports to study the deviation in the aerodynamic behavior induced due to the presence of the sleeve.

5.3.1 NCSU Subsonic Wind Tunnel

The NCSU subsonic wind tunnel is a closed-circuit tunnel with a test section that is 0.81 m high, 1.14 m wide and 1.17 m long. Upstream of the test section is a settling chamber consisting of an aluminum honeycomb screen followed by two stainless steel anti-turbulence screens. The turbulence levels in the tunnel have been determined to be less than 0.33% [26]. The contraction section is composed of four sides of identical curvature and the test section walls diverge slightly to allow for boundary layer growth. The test section walls are made of Plexiglas hinged at the top for access and visibility. And in order to ventilate the tunnel to room pressure, a breather is located downstream of the test section.

The tunnel fan is equipped with blades capable of variable pitch 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 720 Pa [27]. The maximum tunnel speed results in a Reynolds number of approximately 0.8 million for a 12-inch chord airfoil model.

5.3.2 Airfoil Setup

The airfoil model used to mount the leading edge sleeve is a 12%-thick low-Reynolds-number airfoil designed at NC State for wind tunnel experiments. The model is composed of 9 identical inter-changeable sections of chord of 30.5 cm and a span of 9 cm. The sections were assembled together to span the entire test section from the floor of the tunnel to the ceiling. This configuration simulates a 2-D airfoil by eliminating any tip interference in the test. Each of the section was fabricated using stereolithography and has pressure ports on the surface along the entire chord length. There are 44 pressure taps in total along the entire surface of the airfoil, 22 on the upper surface and 22 on the lower surface. These taps are able to communicate the

Figure 5.3: Leading-edge sleeve mounted on the airfoil model in the wind tunnel.

instantaneous pressure values to the measurement system via urethane pressure tubing that runs from 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.

The leading-edge sleeve was mounted on the airfoil model in the neighbourhood of the middle section at a distance of 6.25 cm from the airfoil pressure taps which are connected to

Figure 5.4: Leading-edge sleeve mounted on the airfoil model in the wind tunnel.

the pressure measurement system. Figure 5.3 shows the sleeve attached to the airfoil in the wind tunnel test section and the pressure tubes connecting the sleeve ports to the data acquisition system.

The airfoil model was mounted on a vertical support through the floor of the wind-tunnel test section. An existing LabView program controlling the beta angle of the support was used to control the angle of attack of the airfoil model by a lab computer through a standard nine-pin serial port.

The pressure sleeve was designed with 11 pressure ports but only 5 were tested because of the limitations on the number of sensors available in the pressure scanning modules. The measurements from the airfoil pressure ports were taken simultaneously with sleeve pressure ports to ensure exactly the same free-stream conditions for both configurations and to prevent the introduction of any variations in the flowfield arising due to separate test runs.

Figure 5.5: Pressure ports used for measurements on the airfoil model and on the leading-edge sleeve.

5.3.3 Results

Wind-tunnel testing was conducted to compare the pressures on the surface of the airfoil model with the pressures sensed by the leading-edge sleeve. The results obtained from both these cases are co-plotted in Figures 5.6-5.9 for a range of angles of attack until stall. It is evident from these plots that the surface pressures on the airfoil near the leading edge closely match those from the sleeve pressure ports on both the upper and lower surfaces for small angles of attack. At greater than 4 degrees, larger variations exist on the upper surface, while the lower surface of the sleeve closely follows the aerodynamic behaviour of the base airfoil. These major deviations on the upper surface are attributed to the suction peak that develops on the upper surface for positive angles of attack. Similar behavior is observed for the two different velocity runs conducted.

The lower surface is the region of interest in these runs because for positive angles of attack the stagnation-point moves on the lower surface from the leading edge. The positive pressures sensed by the sleeve ports on the lower surface have greater resemblance to the pressures seen by the airfoil and this result is significant because the LEFS algorithm requires only the input pressures in this region of interest. These results provide confidence that the sleeve can be effectively integrated with the LEFS algorithm to track positive pressure points in the

neighborhood of the maximum-pressure location and use this information in the algorithm.

(a) $\alpha = 0$ deg.

(b) $\alpha = 2$ deg.

(c) $\alpha = 4$ deg.

(d) $\alpha = 6$ deg.

Figure 5.6: Pressure comparison for $Re = 402091$ and $\alpha = 0, 2, 4$, and 6 degrees.

(a) $\alpha = 8$ deg.

(b) $\alpha = 10$ deg.

(c) $\alpha = 12$ deg.

(d) $\alpha = 14$ deg.

Figure 5.7: Pressure comparison for Re = 402091 and $\alpha = 8, 10, 12$, and 14 degrees.

(a) $\alpha = 0$ deg.

(b) $\alpha = 2$ deg.

(c) $\alpha = 4$ deg.

(d) $\alpha = 6$ deg.

Figure 5.8: Pressure comparison for Re = 569510 and $\alpha = 0, 2, 4$, and 6 degrees.

(a) $\alpha = 8$ deg.

(b) $\alpha = 10$ deg.

(c) $\alpha = 12$ deg.

(d) $\alpha = 14$ deg.

Figure 5.9: Pressure comparison for $Re = 569510$ and $\alpha = 8, 10, 12$, and 14 degrees.

5.4 Chapter 5 Summary

An overview of the current pressure-sensing techniques was presented along with the idea of the leading-edge sleeve concept. The design and fabrication procedure used to fabricate the leading-edge sleeve for a given base airfoil profile was discussed in this chapter. Wind-tunnel tests were conducted in the NCSU subsonic wind tunnel to evaluate the feasibility of the sleeve concept

and results were presented comparing the pressure measurements from the airfoil pressure ports and the sleeve pressure ports. The results showed acceptable matching in the pressures sensed by the sleeve with those measured on the airfoil model in the vicinity of the stagnation-point, for the sequence of angles of attack tested. This makes the sleeve suitable for the LEFS algorithm as an effective alternative in the near future until more suitable non-intrusive and self-sustaining surface pressure sensing technologies with transmission capabilities are available.

Chapter 6

Conclusions

This thesis presents the research effort undertaken to address the issue of aerodynamic parameter estimation of an airfoil. The knowledge of the air data quantities plays a crucial role in the flight assessment and performance evaluation. The benefits of greater aerodynamic control in different applications have motivated numerous research efforts in this direction. The conventional approaches adopted in the identification of the flowfield characteristics from the pressure measurements involve either direct parameter calibration techniques, lookup table approaches, or development of simplified aerodynamic models to represent the actual behavior. The research presented in this thesis follows the third approach but introduces a novel methodology towards the development of the aerodynamic model.

The leading-edge flow sensing (LEFS) technique developed in this research uses the fact that the leading edge of any airfoil (symmetric or cambered) can be defined using a parabolic profile. This approximation allows for the aerodynamic model to be derived from the solution of flow past a parabola. The inviscid equations of flow over a parabola, given by Van Dyke, are used as the foundation for building the aerodynamic model. The LEFS algorithm integrates the simplified model with a switching logic to make the method applicable to the real viscous flows. Pressures sensed at five discrete locations are given as inputs to the LEFS algorithm, which employs iterative nonlinear regression techniques to solve for the required aerodynamic

parameters. The algorithm returns the local airspeed and the parameter a' as direct outputs, where the value of a' is explicitly related to the stagnation-point location. It also has strong correlation with the angle of attack. Hence, a lookup table approach discussed in this work is used to map the angle of attack and lift coefficient with a' using simulated data gathered from XFOIL. These tables generated using computation data are then integrated in the algorithm and employed to determine the air data on airfoils and blade sections from real wind-tunnel test data.

The functionality of the LEFS algorithm is evaluated by testing it using experimental data gathered from wind-tunnel investigations conducted at Ohio State University. The results are presented which compare the estimated aerodynamic quantities with the actual values measured during the experimentation. The inflow velocity, which is a direct output of the algorithm, is determined for the S809 airfoil with an absolute error of 0.243 m/sec within the interval of $-12^\circ < \alpha < 12^\circ$. The angle-of-attack estimation has an absolute error of 0.689° within this specified pre-stall regime. The LEFS algorithm takes advantage of the aerodynamic behavior exhibited by airfoils when the flow is attached. This feature renders the algorithm less effective in the post-stall cases when the flow is separated. The results for the lift coefficient show greater matching within the pre-stall regime. A modified version of the LEFS algorithm (LEFS-mod) is also assessed, which integrates a stagnation-point relationship with α derived by an approach proposed by Brand [23]. This modification enabled direct estimation of alpha without the need of the lookup tables. The results for α determination from both algorithms were compared, and it was observed the LEFS performs better with an absolute error of 0.689° compared to the LEFS-mod that results in an absolute error of 1.1014° for the S809 airfoil at Reynolds number of 0.75 million. Similar behavior in the performance was observed for the other Reynolds number cases and are presented in this thesis.

Apart from these results, the LEFS algorithm was also applied to the pressure measurements obtained from the surface of a rotating wind turbine blade. The experiment was a part of the UAE test campaign conducted by NREL. Discrete pressure measurements taken near the

leading edge were used in this research to test the algorithm for a rotating blade, where 3D flow aspects come into play. The results are presented for the velocity estimation and the normal-force predictions in this thesis. The estimated freestream velocity and section normal-force coefficient agree well with those observed in the experiments. Future work in this direction involves the validation of the algorithm for more airfoil cases and extension of the methodology for the analysis of unsteady flows.

Additionally, the leading-edge sleeve concept introduced in this research shows promising results as a self-contained pressure-sensing unit that can be externally mounted on any existing wind/blade section. A prototype was designed and tested in the NCSU subsonic wind tunnel to characterize the intrusiveness of the sleeve. The pressures sensed by sleeve mounted on the airfoil model were compared with the pressure measured by the surface taps on the actual airfoil model. The results show good agreement between them in the region of interest for the LEFS algorithm. This favorable agreement makes the sleeve a potentially viable device that is well suited to work in synergy with the LEFS algorithm. Since the current research only established the feasibility of such a concept, further work is required to evolve the sleeve concept as a self-sufficient device capable of measuring pressures and determining the aerodynamic parameters in real time.

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APPENDIX

Appendix A

Wind Tunnel Results

The experimental data from Ohio State Wind Tunnel was obtained for two more airfoils and the leading-edge algorithm was applied to estimate the aerodynamic parameters. The airfoils considered were NACA4415 and LS0417. Local in-flow velocity, angle of attack and the lift coefficient were predicted for different Reynolds number cases and the results are presented in the following sections.

A.1 NACA 4415

Figure A.1: Parabola defined for the NACA 4415 airfoil

Figure A.2: Velocity estimation for different Reynolds numbers

(a) Angle of attack estimation from LEFS and LEFS-mod

(b) Lift coefficient estimation

Figure A.3: Case - 1: Re = 0.75 million

(a) Angle of attack estimation from LEFS and LEFS-mod

(b) Lift coefficient estimation

Figure A.4: Case - 2: Re = 1 million

(a) Angle of attack estimation from LEFS and LEFS-mod

(b) Lift coefficient estimation

Figure A.5: Case - 3: Re = 1.25 million

(a) Angle of attack estimation from LEFS and LEFS-mod

(b) Lift coefficient estimation

Figure A.6: Case - 4: Re = 1.5 million

A.2 LS 0417

Figure A.7: Parabola defined for the LS0417 airfoil

Figure A.8: Velocity estimation for different Reynolds numbers

(a) Angle of attack estimation from LEFS and LEFS-mod

(b) Lift coefficient estimation

Figure A.9: Case - 1: Re = 0.75 million

(a) Angle of attack estimation from LEFS and LEFS-mod

(b) Lift coefficient estimation

Figure A.10: Case - 2: Re = 1 million

(a) Angle of attack estimation from LEFS and LEFS-mod

(b) Lift coefficient estimation

Figure A.11: Case - 3: Re = 1.25 million

(a) Angle of attack estimation from LEFS and LEFS-mod

(b) Lift coefficient estimation

Figure A.12: Case - 4: Re = 1.5 million