THOMAS BARRIER, ADRON. Airfoil Flow Characteristics in a Pure Surge Environment at Constant Incidence. (Under the direction of Dr. Kenneth Granlund.)

Presented is a parameter study of a NACA 0018 Airfoil at constant incidence undergoing chordwise sinusoidal variations. The work was conducted in the NC State Unsteady Fluid Mechanics lab in the NC State Water Tunnel. The study looked to uncover aerodynamic relations between surge amplitude and frequency and vortex interactions in a highly unsteady environment. Airfoil lift and pitching moment are quantified using force data and related to dye visualization phenomena. A practice of Reynolds number normalization is address based on experimental results. A new method using quasi-steady lift is also presented.
Airfoil Flow Characteristics in a Pure Surge Environment at Constant Incidence

by
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A thesis submitted to the Graduate Faculty of
North Carolina State University
in partial fulfillment of the
requirements for the Degree of
Master of Science

Aerospace Engineering

Raleigh, North Carolina

2018

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DEDICATION

To my Dad and Grandpapa for instilling a love of flight at a young age.
BIOGRAPHY

Adron Barrier grew up in Mount Pleasant North Carolina where he played soccer, did woodworking and enjoyed being outdoors. After high school, he attended NC State to pursue an undergraduate degree in Aerospace Engineering. In 2016, he graduated with his bachelor’s degree and choose to pursue a Masters in Aerospace Engineering. After college, Adron plans to begin a career at Virgin Galactic in California.
ACKNOWLEDGEMENTS

I would like to thank my friends and family for supporting me, pushing me, and believing in me over the past six years. I would also like to thank my adviser for asking me about woodworking during my senior year. I never thought such a simple question would change my life. Finally, I would like to thank my wife for her support and patience through the entire process.
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Chapter 1

Introduction

1.1 Motivation

For many years, aerodynamic principles have been studied to understand the fundamentals that cause aerodynamic phenomena to occur and how these phenomena can be replicated in both low order methods as well as experimental verification. With a basic understanding of a phenomenon, these aerodynamic principles can be engineered to create a practical and beneficial product for a consumer that advances both aerodynamic understanding and application of aerospace knowledge. One of the earliest aerodynamic principles was the understanding of fluid physics over a flat plate and a symmetrical blended body, known commonly as an airfoil. Airfoils, and the study of airfoil dynamics, is essential to the basic understanding of aerodynamics. Airfoil aerodynamics and kinematics allow for a variety of engineering systems and devices such as aircraft, wind turbines, and helicopters to function effectively and serve an intended purpose. Fixed wing aircraft use forward freestream air provided by aircraft momentum to produce lift. Unlike fixed wing aircraft, rotorcraft use rotating airfoil elements, or blades, to produce a vertical force due to the blades angular velocity. This allows the aircraft to produce lift with no direct forward motion, giving rotorcraft a special characteristic when compared to fixed wing aircraft. The rotating blades on a rotorcraft control all motions of the vehicle about the 3 main axes. Attitude control (pitching, yawing, and rolling) is done by varying the pitch angle of the blade during the cyclic motion to create a asymmetry of force along the rotors main axes, causing the aircraft to pitch, yaw, or roll. The subsequent moment, along with lift produced by the main rotor, causes the aircraft to move in the specified direction. During flight, rotor RPM is typically held constant, allowing blade angle of attack to control the overall lift enacted on the aircraft. Because of this, the blades see cyclic motion over a variety of frequencies, Reynolds numbers, pitch and plunge oscillations, and surges in relative velocity.
Unsteady aerodynamics has fundamental challenges that make the study of individual airfoil elements difficult. The aerodynamics of a full-scale helicopter rotor system are very complex due to airfoil blade element pitching, plunging, stream-wise velocity fluctuation, rotating blades, compressibility effects, blade vortex interaction and perturbations, wake instabilities and vehicle interference. Each interaction, although not unique to helicopter aerodynamic regimes, contains a unique time constant due to the oscillatory frequencies present in the rotor system and complicates analysis and predictive methods [2]. To build a basis of knowledge around a complex system, a basic understanding of the underlying flow physics must be known and iterated to grow the knowledge basis of these flow interactions. The understanding of unsteady aerodynamics using low order methods is a relatively new research area that requires verification using experimental methods such as this parameter study.

This research will look to study the effects and flow characteristics of a thick airfoil in a sinusoidal freestream velocity at constant incidence. The focus is to discover what happens in the flow field on and around the suction surface as the airfoil experiences a sinusoidal velocity variation. During the motion, separation location and behavior will be studied as well as the shift between leading and trailing edge separated flow regimes. Theoretical equations by Greenberg [3] describing lift and pitching moment are studied and compared to experimental results. This research study also looks to develop an analysis method for high velocity amplitude ratio surging that correlates mean and fluctuating components of such motions when the overall lift profile is non-sinusoidal. A comparison between quasi-steady Reynolds number corrective method will be investigated and compared to standard corrective methods as described in Section 1.2.1.

1.2 Background

1.2.1 Theoretical Treatment

As mentioned in Section 1.1, airfoil kinematics can be broken down into three main components: longitudinal surging, plunging, and pitching. The linearized models that were studied in this report began using the bases of potential flow theory. Following this assumption, along with the Kutta Condition, the theories were built using Thin Airfoil Theory (TAT). Under TAT, airfoil potential flow geometries can be modeled using a summation of vortexes across a chord line as well as a freestream velocity. Vortex strength and placement along the lifting line can be varied to mimic chord thickness and camber. It is commonly understood that for a 2-D airfoil geometry with attached flow, $C_{l_{\alpha}} = 2\pi/\text{rad}$. As shown, $L'$ can be varied by increasing $\Gamma$ or $V_{\infty}$. Circulation of the foil can be increased by increasing camber or angle of attack.
TAT assumes steady state conditions and ignores time domain based motions of a airfoil such as surging, pitching or plunging. For a pure surge environment, the relative velocity \( U(t) \) seen by the airfoil of chord 'c' can be described by:

\[
U(t) = U_\infty (1 + \lambda \sin(\omega t))
\]

where \( \lambda \) is the velocity amplitude ratio and \( \omega \) is the angular frequency of the motion. When rearranged and substituted, the physical amplitude of the sinusoidal motion \( A \) can be normalized as shown:

\[
k_v = \frac{\omega U_\infty c}{2U_\infty}
\]

\[
\lambda = \frac{2Ak}{c}
\]

These two parameters, \( k \) & \( \lambda \), dictate the overall motion profile, and thus, the relative velocity seen by the airfoil and will be the subject of this parameter study.

The first attempt to modify TAT for unsteady aerodynamics was conducted by von Karman and Sears [4]. Introducing small oscillations, all vortex pairs were summed using the circulation of the vortex as well as the distance from the airfoil quarter chord to create equations that predicted lift during a time dependent "sharp" gust. The analysis assumed all vortices lied within the chord line of the airfoil. With circulation and distance denoted, the momentum of the fluid from a given reference frame could be calculated. The rate of change of this momentum determined the magnitude of lift created by the airfoil and could be broken down into 3
components: apparent mass contribution, quasi steady lift, and lift produced by the vorticity of the wake [4]. Theodorsen studied the effects of unsteady pitching and plunging motion on a small aileron, looking at small perturbations in the velocity at high frequencies typically found during aircraft flutter scenarios. Theodorsen’s work produced velocity potentials of an airfoil motion that assumed attached flow and low angle of attack. The solution neglected freestream velocity variation, but laid the foundation for following solutions and predictive methods [5]. Theodorsen’s theory was modified by Isaacs [6] to include a solution that incorporated a sinusoidal freestream element, a similarity needed to model rotorcraft blade elements. The theory investigated two phenomena that occur in a sinusoidal freestream: wake vortices that shed due to varying lift, as well as the pressure differentials that occur due to air accelerating over the rotor. The main contribution was a two-dimensional solution that calculated lift using a constant velocity plus a sinusoidal component that encompassed a cyclic action and ignored startup conditions. The solution, although universal, was difficult for the understanding of basic aerodynamic principles and contained many mathematical terms that complicated the solution.

Greenberg [3], of which this study is compared, created a simplification of Isaacs’ model for sinusoidal velocity variations that ignored higher order terms of the solution. Assuming a continuous vortex sheet being shed from the trailing edge, Greenberg demonstrated that the assumption of a sinusoidal wake correlated well with Isaacs’ original results. The equations, as described by Greenberg, quantified lift and pitching moment of a thin airfoil and includes all three dynamic motions: surging, pitching and plunging:

\[
L = -\frac{\pi \rho c^2}{4} \left[ \dot{h} + v\dot{\beta} + \dot{v}(\alpha + \beta) + \frac{ac\dot{\beta}}{2} \right]
\]

\[-\pi \rho v c \left\{ v_0 \alpha + \lambda v_0 \alpha C(K_v)e^{i\omega_v t}\left[ \frac{c}{2} \left( \frac{1}{2} - a \right) \dot{\beta} + v_0 \beta \right] C(k\beta) \right. \]

\[+ \dot{h}C(k_h) + \lambda v_0 \beta C(k_{v+\beta})e^{i\omega t} \}
\]

\[
M_a = -\frac{\pi \rho c^2}{4} \left[ \frac{vc}{2} \left( \frac{1}{2} - a \right) \dot{\beta} - \frac{vca}{2} (\alpha + \beta) - \frac{ca\dot{h}}{2} + \frac{c^2}{4} \left( \frac{1}{8} + a^2 \dot{\beta} \right) + \frac{\pi \rho v c^2}{2} \right]
\]

\[+ \frac{\pi \rho v c^2}{2} \left( a - \frac{1}{2} \right) \left\{ v_0 \alpha + \lambda v_0 \alpha C(k_v)e^{i\omega_v t}\left[ \frac{c}{2} \left( \frac{1}{2} - a \right) \dot{\beta} + v_0 \beta \right] C(k\beta) \right. \]

\[+ \dot{h}C(k_h) + \lambda v_0 \beta C(k_{v+\beta})e^{i\omega t} \}
\]

where \( a \) is the pivot point located on the airfoil, and therefore the location at which the moment is calculated. The Bessel function \( C_k \) is defined as \( C(k) = F_n + iG_n \) [5]

Under Greenberg, mid-chord is defined as \( a = 0 \) with leading and trailing edge normal-
ized between -1 and 1, positive aft. This parameter study only investigates streamwise velocity fluctuation, therefore, pitching and surging terms are omitted ($\beta = h = 0$). Van der Wall and Leishman [7] reviewed the limitations and assumptions of Greenberg to determine where the equations broke down and no longer followed a modern Euler code. Greenberg’s equation, a simplification of Isaacs’ equation, follows Theodorsen’s classical theory in terms of structure when looking at pitching and plunging. Because of the high frequency simplification making the wake sinusoidal, Greenberg’s simplified theory is questionable with high freestream oscillation amplitudes ($\lambda$) and differs from Issacs’ theory when $\lambda > 0.4$. This assumption is similar to neglecting the flow oscillation amplitude for induced velocities. For Greenberg’s equations, van der Wall also suggested that the magnitudes and phase angles do not correlate when compared to other methods and the Eulerian code despite basic behavior being well represented. Simplifications and studies by van der Wall [7] collapse Eq.1.7,1.8 into lift and moment variations based on the static mean lift and moment under TAT. Each equation can be broken up into circulatory and non-circulatory components. Non-circulatory components of lift and moment production are only present in unsteady motions and are often referred to as the apparent mass of a body in a flow. Non-circulatory components are independent of a freestream velocity and can be generated by moving a test article inside a fluid with no freestream velocity, causing a force to be generated on the body. This force, can be modeled in 2D as a volume of fluid surrounding the object as it moves through a fluid. For the purpose of this study, only the circulatory component of lift and moment is desired since the component is governed and modeled by TAT. Simplifying using approximations as described above, Greenberg’s equations, as noted by van der Wall, are as follows:

\[
\frac{L_{nc}}{L_0} = \frac{k}{2} [\lambda \cos(\omega t)] + [\lambda(1 + F(k))] \sin(\omega t) \quad (1.9)
\]

\[
\frac{L_c}{L_0} = \left[1 + \frac{\lambda^2}{2} F(k)\right] + [\lambda G(k) \cos(\omega t)] - \frac{\lambda}{2} [\lambda F(k)] \cos(2\omega t) + \frac{\lambda}{2} [\lambda G(k)] \sin(2\omega t) \quad (1.10)
\]

\[
\frac{M_{nc}}{M_0} = -k\lambda a \cos(\omega t) \quad (1.11)
\]

\[
\frac{M_c}{M_0} = (1 + 2a) \frac{L_c}{L_0} \quad (1.12)
\]

A quantification done by Granlund et al. [8] uses a cylindrical volume to describe the apparent mass of a surging airfoil test article in relation to standard lift production of the same test article in a constant freestream. [8]. Using Eq.1.4 to determine $a(t)$

\[
U(t) = U_\infty [1 + \lambda \sin(\omega t)] \quad \frac{du}{dt} = U_\infty \lambda \omega = U_\infty \lambda \frac{2U_\infty k}{c} \quad (1.13)
\]
\[
\frac{L_{nc}}{L_0} = \frac{\sin(\theta)\rho \pi (c^2/4)(du/dt)}{2\pi \sin(\theta)(1/2)\rho U_\infty^2} = \frac{c(du/dt)}{4U_\infty^2} = \frac{2c\lambda k U_\infty^2}{4U_\infty^2 c} = \frac{\lambda k}{2} < 0.06
\] (1.14)

Using the peak values in this parameter study, \( \lambda = 0.5, k = 0.25 \), apparent mass effects account for 6\% of the overall force. This contribution is negligible when looking at lift and allows raw lift data to be effected by circulatory lift.

\[
\frac{L(t)}{L_0} = \frac{L_{nc}}{L_0} + \frac{L_c}{L_0} \rightarrow, \quad \frac{L(t)}{L_0} = \frac{L_c}{L_0}
\] (1.15)

For pitching moment, a similar process can create a relation for apparent mass contribution using Eq.1.11 & 1.12

\[
\frac{M_{nc}}{M_c} = \frac{M_{nc}}{M_0} \frac{M_0}{M_c}
\] (1.16)

It is important to note that both pitching moment equations require a selection of pitch axis \( a \) for moment calculations. In this study, the pitching of the test article was taken about the \( c/4 \) position, producing a normalized pitch location of \( a = -0.5 \). This substitution causes \( M_c/M_0 = 0 \), causing Eq. 1.16 to asymptote to \( \infty \). To cancel apparent mass contributions, the mid-chord, \( a = 0 \), was studied. When inserted in equation Eq.1.11, causes \( M_{nc}/M_0 = 0 \), producing no pitching moment. Therefore, the bounds between \( c/2 \& c/4 \) lie between 0 and \( \infty \).

Table 1.1: \( M_{nc}/M_c \) Pitch Location Comparison

<table>
<thead>
<tr>
<th>( a )</th>
<th>( c/4 )</th>
<th>( c/2 )</th>
</tr>
</thead>
<tbody>
<tr>
<td>( M_{nc} )</td>
<td>-0.50</td>
<td>0</td>
</tr>
<tr>
<td>( M_c )</td>
<td>0</td>
<td>( \frac{L_c}{L_0} )</td>
</tr>
<tr>
<td>( M_{nc}/M_0 )</td>
<td>( \neq 0 )</td>
<td>0</td>
</tr>
<tr>
<td>( M_{nc}/M_c )</td>
<td>( \infty )</td>
<td>0</td>
</tr>
</tbody>
</table>

This indicates that apparent mass contributions from the airfoil motion cannot be neglected and must be considered when analyzing for pitching moment created by circulatory effects. Because of high \( \lambda \) values desired in this study, the noncirculatory effects, as predicted in Eq.1.16 must be subtracted from raw measurement data to accurately depict circulatory pitching mo-
ment.

\[ \frac{M_c}{M_q} = \frac{M(t)}{M_0} - \frac{M_{sc}}{M_0} \quad (1.17) \]

To understand the unsteady effects present during a sinusoidal motion, a normalization of quasi-steady lift can be used. Quasi-steady theory assumes no unsteady effects such as vortex shedding on or around the test article, and is solely dependent on the instantaneous dynamic pressure. In a physical sense, quasi-steady lift can be seen as the unsteady value when \( k \to 0 \). Because \( \rho \) is constant in this study, the quasi-steady predictions are solely based off of velocity fluctuations. This special case causes the Theodorsen function, \( C(k) \), to have values \( F(k) = 1 \) & \( G(k) = 0 \). Using Eq. 1.10 and substituting, the following is obtained:

\[ \frac{L_{qs}}{L_0} = (1 + \lambda \sin(\omega t))^2 \quad (1.18) \]

Using this relation, experimental and theoretical lift can be manipulated and expressed in the quasi-steady normalized form:

\[ \frac{L(t)}{L_{qs}} = \frac{L(t)}{L_0} \left( \frac{1}{1 + \lambda \sin(\omega t)} \right)^2 \quad (1.20) \]

Both experimental and theoretical quasi-steady moment calculations will also be compared. Because of \( c/4 \) pitching coordinate, theoretical quasi-steady pitching moment will collapse to 0, following TAT predictions. As shown in Eq. 1.17, noncirculatory effects will be deducted based on Greenberg’s predictions:

\[ \frac{M_c}{M_{qs}} = \frac{M_c}{M_0} \left( \frac{1}{1 + \lambda \sin(\omega t)} \right)^2 \quad (1.21) \]

where \( M_c/M_{qs} \) is denoted by Eq. 1.17. Quasi-steady comparisons allow for insight into where in the motion unsteady effects occur. If no unsteady effects were present in the motion, \( L/L_{qs} \) would equal 1 throughout the entire motion. Any deviation from 1, either positive or negative, indicates unsteady effects when compared only to the change in dynamic pressure due to surging.

The final normalization that will be done involves quantifying the fluctuating lift present during the motion. As demonstrated by Granlund et al. [9], lift production is the sum of a fluctuating and mean component of the motion and can be broken down as such:

\[ \frac{L(t)}{L_0} = \frac{\overline{L(t)}}{L_0} + \left( \frac{L(t)}{L_0} \right) \quad (1.22) \]

As show in Eq.1.22, \( \overline{L(t)}/L_0 \) is the mean lift value over the entire surge motion and not the mean
at a steady freestream velocity. \((L(t)/L_0)’\) is the fluctuating component while \(L(t)/L_0\) is the raw lift generated by the motion \((measured\ lift\ over\ cycle)\). For \(\lambda \leq 0.1\), as shown in Granlund et al. [9], a sinusoidal assumption allows for the fluctuating components to be extracted due to a small \(\lambda\) assumption in which \(\bar{L}(t) = L_0\). Because \(\lambda > 0.1\) in this study, this extraction must be modified to account for large amplitude motions.

![Figure 1.2: Greenberg Normalized Lift Profile with Normalized Velocity Comparison, \(\lambda = 0.1, 0.3, 0.5\)](image)

In Fig.1.2, normalized velocity variation \((dashed)\) is plotted against Greenberg’s normalized lift predictions \((solid)\). Along with phase offset, the retreating portion of the motion, \(t/T > 0.50\), varies significantly in amplitude except for the \(\lambda = 0.1\) case. Because quasi-steady accounts for \(\lambda\) variations in the denominator as shown in Eq. 1.20, velocity amplitude can be collapsed to expose fluctuating components:

\[
\frac{L(t)}{L_{qs}} = \frac{\bar{L}(t)}{L_{qs}} + \left(\frac{L(t)}{L_{qs}}\right)’ \tag{1.23}
\]

Rearranging Eq. 1.23, the fluctuating component can be extracted:

\[
\left(\frac{L(t)}{L_{qs}}\right)’ = \frac{L(t)}{L_{qs}} - \frac{\bar{L}(t)}{L_{qs}} \tag{1.24}
\]

To compare the fluctuating magnitude on a case by case basis, the peak-to-peak value of
\( (L(t)/L_{qs})' \) curves will be taken for theoretical and experimental cases. The goal is to quantify the fluctuation changes over both \( k \) and \( \lambda \). Theoretical values [3], [7] will be compared to experimental data sets using normalizations shown above. All data sets will be plotted against normalized time based on the period of motion. The time scale \( t(s) \) will be from 0 to 1 with 1 corresponding to the 360\(^{th} \) phase angle \( \phi \) of the motion.

\[
\frac{L(t)}{L_0} \text{ vs. } t \\
\frac{M(t)}{M_0} \text{ vs. } t \\
\frac{L(t)}{L_{qs}} \text{ vs. } t \\
\frac{M(t)}{M_{qs}} \text{ vs. } t \\
\left( \frac{L(t)}{L_{qs}} \right)' \text{ vs. } t
\]

1.2.2 Experimental Facilities

One of the biggest challenges of validating theories aforementioned is a facility in which parameters can be studied in a controlled environment and replicated. In unsteady motion, the unsteady motion can be done by gusting the freestream velocity inflow or by moving the test article in a constant freestream. Hakkinen and Richardson [10] used the MIT Aeroelastic and Structures Research Laboratory for turbulence measurements by oscillating a vertically oriented NACA 0012 airfoil at 10Hz, upstream of the test section. The results from this test showed that analysis of gusting was possible, but the results were inconclusive due to scattering of the measurements when measuring the sinusoidal freestream, therefore corrections were needed. Buell [11], of NASA Ames Research Center, constructed a gust generator that used both longitudinal and latitudinal vanes to create span wise and chord wise gusting flows depending on test article orientation. The vanes were constant-chord NACA 0015 airfoil sections and pitched about the quarter chord. Depending on the desired gusting response, vanes could be substituted and geared to vary frequency and amplitude of the gust which was dependent on the angle of attack of the blades. Although the gusts were generated and varied using the prescribed method, Buell was unable to limit the gust direction and noted that oscillating vanes would not be suitable for synthesizing two or three dimensional gusts. Ham, Bauer & Lawrence [12] created a gust generator similar to Buell for studying the gust response of a model rotor propeller using
the Wright Brothers Tunnel at MIT. For their testing, a circulation-controlled airfoil (CCA) with 20% chord thickness was used which incorporated air injectors that delayed boundary layer separation. Through the method of active blowing over the gusting vanes, consistent results comparing theoretical gusting values was achieved. JA Miller & A.A Fejer [13] modified a closed circuit wind tunnel at IIT Chicago using rotating shutter vanes placed downstream of the test section. This allowed for a frequency range of $4 < f < 125\text{Hz}$ and velocity amplitudes from $0.0755 < \lambda < 0.667$ and modified the entire test section freestream velocity. Following the success of gusting flows using oscillating vanes, many others created similar test sections, each with a limitation as described by Greenblatt [14]. Grandlund [8] used a free surface water tunnel to achieve reversed flow conditions with $\lambda > 1$.

One of the most recent innovations that has been developed in unsteady aerodynamics is the creation, testing and implementation of a unsteady wind tunnel at Israel Institute of Technology (Technion). The Technion wind tunnel is a open return wind tunnel powered by a 75 kW blower with 13 louvers downstream of the variable length test section. The wind tunnel plenum is wheel mounted, allowing for segments to be removed or added before entering the 8 : 1 contraction. The contraction feeds into a variable length test section that contains three 1004 mm x 610 mm x 720 mm modules that allow for varying tunnel bandwidth, resonance, and damping. The louver system used to vary flow conditions lie at the end of the test section and vary the cross sectional exit area, varying head loss through the tunnel and creates a pressure gradient through the test section. The innovation behind this tunnel is the high oscillation amplitudes that can be achieved under this method. Typical gusting tunnels can only produce $0.1 < \lambda < 0.3$ while Technion has the capability of $\lambda = 0.5$ [14]. For mimicking gusting flows, this is an advantage due to the nature of a gust the occurs in real aerodynamic situations. When testing varying freestream velocity profiles, this experimental method may show discontinuities when compared to a dynamic test article. Along with comparison to theoretical predictive methods and flow structures, another goal is to validate and compare gusting flow experimental methods to dynamic test article methods.

1.2.3 Experimental Studies

Some of the first research into large freestream oscillations came from J.A. Miller and A.A Fejer [13] using the previously mentioned IIT Chicago Wind Tunnel. The study focused on transitional Reynolds number of a NACA 0012 through freestream oscillations and concluded that the transitional Reynolds number is influenced only by the amplitude of the motion, not the frequency. The location at which this transition occurs however is influenced only by the frequency. Fejer also continued this work with Obremski [15] in uncovering a burst Reynolds number in which two distinct bursts occur during a periodic motion. The first is considered
a startup burst that occurs upstream of the test article as well as a convective phase that consists of continuous vortex shedding at the trailing and leading edges. In both experiments, the $0 < \lambda < 0.29$ respectively.

Granlund et al [16] investigated the similarities of gusting and dynamic test articles in a constant freestream by studying a NACA 0009 airfoil at $\lambda = 0.1$ and high frequency responses, $0.1 < k < 3$. For varying freestream oscillations, The Illinois Institute of Technologys Andrew Fejere Unsteady Flow Wind Tunnel was used. The tunnel displays a similar setup as Greenblatt [14] using shuttered vanes beyond the test section to create the unsteady airflow inside the test section. Due to tunnel limitations, measurements were taken from reduced frequencies of $0.1 < k < 1.5$. For the oscillating test article in constant freestream, the U.S. Air Force Research Laboratory Water Tunnel was used. Because of test section length, $k$ could be varied between $0.1 < k < 3$. At $Re = 57k$, the experiments performed show no discernible difference between oscillating the flow or oscillating the test article when the incidence angle is below $10^\circ$ if the experimental results are corrected for the corresponding buoyancy or model inertia present during testing. The other main contribution from this study was the validation of Greenberg’s [3] equations under the condition that flow was attached to the airfoil. As the incidence angle increased, the theory began to break down in both amplitude and phase. As the angle of attack increased, the forces were more dependent on oscillation frequency. This research hopes to repeat Granlund et al. [16] experiment in a similar fashion by using higher oscillation amplitudes from the Technion Wind Tunnel and the NC State Water Tunnel.

Much of the research surrounding unsteady surging has dealt with low oscillation amplitudes as well as thin airfoils ($y/c < 0.12$). Surge amplitude studies are typically limited by physical testing limitations in either experimental facilities or computing power. Due to the constraints presented by theory, many researches have chosen to use thin airfoils to validate results due to the basis for which the theories were created. Departure from these assumptions leaves voids in theoretical methods and possible pitfalls. Because of structure requirements that are necessary for rotor operations, either helicopters or wind turbines, airfoils and subsequent wings contain higher thickness ratios. Because of this, it is important to study the effects of thicker airfoils such as a NACA 0018 or custom airfoil to see how the behavior varies from theoretical requirements and predictions.

Greenblatt [17] investigated a NACA 0018 & 0012 airfoil and demonstrated that at 50% stream wise oscillations, overall lift histories for dynamic motions compared well in both testing methods, attributing many of the differences in the data to test-section blockage, models support configurations, flow quality, and Reynolds number effects. It is important to note that reduced frequency values for this study were relatively low ($k < 0.1$). Strangfeld [18] and others also utilized a gusting wind tunnel with a tripped boundary layer and concluded that unsteady predictions at velocity amplitudes near 0.5 hold well with theory. Without a tripped boundary
layer, a mid-chord separation occurs and grows as the airfoil retreats in the sinusoidal motion. During the motion, there is a significant undershoot in lift production which is due to the separation bubble shedding. Trailing edge vortex wake interaction was also studied to determine effects of the wake on lift and compared to theoretical results. Strangfeld et al. [19] showed strong correlations for a NACA 0018 airfoil when compared to Isaacs’ equations [6] for a pure surge environment at attached flow conditions.

1.3 Research Goal

This research will look to study the effects and flow characteristics of a NACA 0018 airfoil in a sinusoidal freestream velocity at high velocity amplitude ratios at constant incidence. Using force measurements and dye tracking visualization, a correlation between leading and trailing vortex shedding separation and airfoil performance metrics can be studied. Theoretical equations by Greenberg [3] & van der Wall [7] will be compared to experimental results at attached, partially detached, or separated conditions. This research study also looks to develop an analysis method for high velocity amplitude ratio surging that correlates mean and fluctuating components of such motions when the overall lift profile is non-sinusoidal.
Chapter 2

Experimental Setup & Validation

2.1 Experimental Facility

2.1.1 North Carolina State University Free Surface Water Tunnel

Experimental testing was conducted in the North Carolina State University Free Surface Water Tunnel. The water tunnel is a closed circuit free surface water tunnel that is constructed of fiberglass foam composite panels and all glass test section. The tunnel uses a 3 sided 4.5:1 contraction which feeds into a 32" wide x 24" tall x 96" long all glass test section. Flow quality is conditioned through 2 plastic honeycomb screens and three turbulence screens constructed of metallic mesh and stainless steel. The tunnel is powered by a 10 HP EM3774T Baldor Electric motor that uses a belt pulley system to drive a 24" brass marine propeller. Performance data of the tunnel was done using Particle Image Velocimetry (PIV). The tunnel has a freestream velocity range of $0.15 < U_\infty < 0.97 \text{m/s}$ and turbulence intensity between 0.2-0.8% depending on the test section location and tunnel freestream speed [1]. When operating, the tunnel typically holds 3500 gallons of water, but is varied accordingly to keep a smooth free surface during testing. As tunnel speed is increased, the free surface height decreases, and additional water must be added. During testing, the free surface level was held constant through observation to limit surface waves due to sloshing.
2.1.2 Linear Motion Gantry

For dynamic motion of the airfoil, a H2W rail stage gantry was used. The gantry is powered by 2 DRS-080-08-013-01 linear stages. Each stage uses a RLS LM 13 linear magnetic encoder system that allows for 1 micron movements and is powered by a Xenus XTL Digital Servo Drive. The motors are BLDM-D08 cog-free brushless linear motors and ride along a slide bearing rail. The gantry has a stroke of 83.5”, a peak force capacity of 200 lbs., and a continuous capacity of 66 lbs. [20]. The upper gantry contains 2 cross member rails that span the width of the test section and allow for attachment points for the test article and subsequent measurement systems. Because of the all glass test section floor, optical access it present throughout the entire motion. For dye tracking visualization, a H2W dual rail stage was used and mounted beneath the all glass test section along the tunnel center-line. The lower stage has a stroke of 80.5” and force peak of 30 lbs. which allows for dynamic tracking of the test article at high $\lambda$ values.
All motor components of the transverse system are controlled by a Galil 4080 DMC controller. The controller accepts encoder inputs up to 22 MHz and has position update rates up to 32 kHz [20]. To control the motor system, Galil Suite software was used. For sinusoidal oscillations, a code was written that creates incremental position steps over a given time increment. For the parameter study, the cycle time of one motion ranged from 2.3 sec to 30.8 sec. Therefore, a time difference of 4 ms was chosen for position calculation and encoder movement input. This allowed for 625 position steps to played over the quickest motion of $k=0.25$. This allowed for all 360 phase angles to be determined accurately during the motion. It is important to note that the internal clock of the controller is accurate down to 1 ms and is also based on binary, and therefore slight position error occurs over 1 cycle. Compounded over 25 cycles, the overall position error can be noted. Over 1 cycle of motion, on average, it was determined that the gantry would drift by 1-3mm towards the downstream end of the test section. Although this drift is noticeable, the relative velocity change seen by the airfoil is negligible when compared to an exact motion. Because the airfoil operates on micron increments, the amplitude of the drift can be compared to the smallest amplitude conducted for this parameter study.

$$A_{\text{drift/cycle}} = 1000 \mu m, \quad A_{\text{min}} = 50,000 \mu m \quad \frac{A_{\text{drift}}}{A_{\text{min}}} \ll 0.1 \quad (2.1)$$
With position drift of 1000\(\mu m\) per cycle, it can be assumed for one motion profile, the peak values at \(t/T = 0.25 \& 0.75\), as denoted in position steps, is off by 500 \(\mu m\). This correlates to maximum velocity error of 0.99\% for the quickest motion. The difference in the velocity profile will not effect the overall trends and conclusions of this study. The controller was operated in contour mode, which operates using incremental encoder steps over a specified time period. To calculate position steps, a Galil Suite code was created that asked for parameters of the motion, \(U_\infty\), \(\lambda\), and \(k\), and calculates each relative position increment at every time step through one oscillation. A difference calculation is done between each relative position step, and fed into a matrix that stored each distance. The matrix is then read back during the motion, and cycled using a repetition loop. The number of repetitions of the loop determines the number of cycles the motion is conducted. Because the upper rail is constructed of 2 single rail stages, the C axis, indicated on the left side of the test section, was coded to become a slave axis to the B axis, the right side linear stage. The slave was based off of a gearing function as described by Galil. A gearing slave was chosen due to the inconsistency present in the end position of the upper stages. Both B and C axes were placed by hand, and therefore could not be aligned down to 1\(\mu m\). With gearing, the motor sends the same amount of power to the slave axis, and therefore causes it to follow the master stage, regardless of absolute position along the rail. For dye tracking, the camera axes (A) was set to run the same contour delta matrix as previously described. This allowed for the camera axes to follow the entire motion, or stay idle during testing.

2.1.3 Force Measurements

For this parameter study, lift \((L)\) and pitching moment values were desired to understand the performance of the airfoil quantitatively. The process by which forces were obtained involved a iterative process of new methods, corrections, and complete system overhauls. The 3 methods used, described below, were tested during the parameter study.

**Futek Iteration**  The forces desired in this parameter study \((L\&M_{c/4})\) can be collapsed into a idealized 2-D case. For a cantilevered beam type setup as described in this experiment, adverse moments caused by a resultant force on the airfoil enact large moments during operation. Because many load measuring devices such as strain gauge load cells have operation limits defined by all 3 forces and moments, it is desirable to eliminate the measuring of large moments by preventing rotation about in axis except the airfoil pitching moment line. Deriving a similar setup as Granlund [16],3 separate bi-axial loadcells could be used and places away from the rotational axes, allowing for resulting moments to be absorbed vertically into a non active measuring direction, allowing much higher Reynolds numbers and chord lengths to be accurately measured. Us-
ing Futek MBA-200 Bi-Axial Loadarms, a "T" configuration was made that placed 2 load directions along the $c/4$ line of the airfoil and a 3rd along the center-line of the airfoil as shown below:

As lift and drag were generated, the airfoil would absorb the adverse moments along the $z$-axis of the loadcell which contributed very little to the overload limit. For assembly, rectangular aluminum brackets were fitted to each loadcell to allow for mounting on a extruded aluminum rail frame, resembling a "T" shape. 2 "T" rail assemblies were created to connect the top and bottoms of the 3 supporting loadcells. The distance between the loadcells could be adjusted depending on the resolution desired on the force output. Due to the high load limits of the loadarms, each loadarm was place 8" from the center-line of the airfoil as shown as "x" and "y" in Fig 2.3a. Because this parameter study involved constant incidence, it was determined that the rotary axis pitch motor, shown in Fig.2.2, was not needed due to complications of mounting and size requirements of the aforementioned system. To adjust pitch angle, a 0.5" aluminum plate was fabricated with CNC milled holes that allowed for the airfoil to be pitched at $\alpha = 6^\circ$ and $20^\circ$ based on CAD drawings of the tunnel and subsequent mounting system. To obtain a minimal tip gap, the entire configuration was mounted on 3 vertical oriented bars that connected to both horizontal gantry rail, allowing for minor adjustments of the entire assembly. Each Futek was connecting to a National Instruments SCXI-1000 data acquisition box using 10-pin connector. Each loadcell was NIST calibrated and operated off of 10V DC power as prescribed by the SCXI system. Each direction of the arm was sampled at 1kHz and tared using a 1sec average based off incoming static signals. Although the potential setup was a success, trouble shooting occurred. Because of motion during testing, dynamic tares were
conducted on the gantry system. During this initial phase of testing, it was noted that the dynamic tare values had a strong correlation with the motion profile and did not follow the moving mass predictions. Initial investigating suggested that the signal wire, although shielded, was being disturbed by large current draw from the gantry motors. Since both cables were within close proximity of each other and laid in parallel, it was possible that a induced magnetic field was present in the signal wires. To validate this, the gantry was locked into position with the motors on, and force was applied parallel to the gantry track system. After many trials, it was determined that the magnetic interference was only induced 0.01 N on the load cells, which is significantly smaller than the resolution of the load cell and resembles electronic noise. By moving the upper gantry by hand with no motor power, it was determined that the induced loading on the system was positioned based. Because the load cells spanned both gantry cross members, it was concluded that the 2 upper gantry were not parallel. This issue was caused by uneven B and C axis rails as shown in Fig. 2.2. The uneven rails caused a torsion effect, causing the CG of the cantilevered weight to shift as well as inducing stress on the measurement system, producing false readings, as the motors traversed the test section. This problem was combated by leveling and shimming the main rail stages, and connecting the rear load cell to the same gantry support bar as load cells 1 and 2. With this, the tare value decreased drastically by nearly 1/8th the original tare value. To validate the measurement system, static load test were conducted on the wing in the -x-direction (drag). To do this, a pulley system was rigged and weight was hung from the mid-span of the airfoil. The goal was to determine if the system could accurately predict lift and pitching moment. Although the trend and differentials can be observed, the values recorded were inaccurate. The lift direction of the airfoil was also checked using a digital scale. Although the test induced many inaccuracies, the scale, overall, portrayed accurate results when used in a proper configuration. To try to correct the inaccuracies of the measurement system, additional collars were placed on the load cell mounting arrangements to prevent the lower T-configuration from inducing unwanted loads in either direction. The collars were precision machined and contained set screw that allowed no movement of the load cell on the rig. This addition did very little to correct the overall lift, drag and pitching moment measurements of the wing. The main issue with the tri-load cell setup was the connection method between the load cells and induced loading. In an ideal case, the moment generated by the wing would be absorbed in the positive z-direction of the load arms. However, through testing, it is seen that a load is introduced in both the X and Y directions. When doing a shear load on the entire gantry, a change can be seen, but the value also induced additional loading due to binding between the load arms. Although many attempts were made to try to reduce this binding, additional tightening of the connector bolts induced more binding and loading that was undesirable. Additional loading from the wing also exaggerated this issue. It should be noted that the Futek iteration could measure relative change when normalized by a mean
value such as $L_0$. Overall change in lift drag and pitching moment could be calculated over a motion, but raw lift values, used for calculations of $C_L, C_D, & C_M$, were extremely inaccurate despite this effect.

**ATI Gamma Iteration** The 2nd iteration of force data acquisition was done using a ATI-Gamma Loadcell. The ATI Gamma is 6 degree of freedom load balance that contains 2 sets of load flexures and is designed to measure strong axial loading and torque, typically seen in a propeller test stand. Under the US -30-100 calibration, the sensor displayed the following load limits and ranges:

<table>
<thead>
<tr>
<th>Calibration</th>
<th>$F_x, F_y$</th>
<th>$F_z$</th>
<th>$T_x, T_y$</th>
<th>$T_z$</th>
</tr>
</thead>
<tbody>
<tr>
<td>US-30-100</td>
<td>30 lbf</td>
<td>100 lbf</td>
<td>100 lbf</td>
<td>100 lbf</td>
</tr>
</tbody>
</table>

Sensing Ranges

- 1/160 lbf
- 1/80 lbf
- 1/80 lbf – in
- 1/80 lbf – in

Resolution

Modifications were made to the original mounting plate allowing the load cell to be affixed to the upper gantry. An additional plate was made to mound the airfoil to the Gamma. The same hole pattern was recycled as described earlier with the addition of $\alpha = 13^\circ$. The center $z$-axis of the load cell was placed directly above the $c/4$ mark of the airfoil to measure pitching moment. In the configuration, $+Y$ denoted $+\text{lift}$ while $-X$ denoted drag force. An ATI LabView interface was used and modified to perform data acquisition. Static tares were done using a 1000 sample average with a sampling rate of 1kHz. To ensure the loadcell accurately detected forces at the mid chord of the airfoil, a load test was conducted.
Results from the ATI Gamma load test as shown in Fig. 2.4 show a strong correlation between measured results and values measured by the Futek Loadarm along one axis. Because the Futek was NIST calibrated, it was used as the comparison and considered to be correct in measuring weight additions. With this success, the ATI Gamma conducted the entire parameter study except $\alpha = 20^\circ$. Because of the bending moment produced by resultant force, the Gamma was unable to capture the full range, but gave an accurate insight into what behavior would be seen when conducted abbreviated parameter study was conducted using the ATI Gamma loadcell. A drawback to the Gamma was an outdated calibration file from 2008. When compared in loading to the Futek sensors, the Gamma contained a slight drift at higher loading situations, causing concern about accuracy. Although inaccuracy’s occurred, a proof of concept was performed, allowing for future work to continue. It should also be noted that the ATI Gamma, when normalized, displays similar results to a Futek in terms of varying lift production over a cycle. A comparison case is shown below:
With successful test of the brief parameter study and strong correlation of data, it was determined that 6-DOF loadcell was successful in measuring pitching and moment force acting on the airfoil.

**ATI Delta Iteration** To alleviate the aforementioned drawbacks of the Gamma loadcell, a ATI Delta was purchased and NIST calibrated to the specifications described in Table 2.2

<table>
<thead>
<tr>
<th>Calibration</th>
<th>$F_x, F_y$</th>
<th>$F_z$</th>
<th>$T_x, T_y$</th>
<th>$T_z$</th>
</tr>
</thead>
<tbody>
<tr>
<td>SI-660-60</td>
<td>660N</td>
<td>1980N</td>
<td>60Nm</td>
<td>60Nm</td>
</tr>
</tbody>
</table>

Sensing Ranges

<table>
<thead>
<tr>
<th></th>
<th>$1/8N$</th>
<th>$1/4N$</th>
<th>$10/1333Nm$</th>
<th>$10/1333Nm$</th>
</tr>
</thead>
</table>

Resolution

The ATI Delta was mounted in a nearly similar fashion as the ATI Gamma. The existing plates were modified to accept the Delta as well as the addition of a slotted bolt sec-
tion that allowed for a $\alpha$ sweep of the airfoil for testing. This was done using a CNC mill and CAD software. The loadcell was connected to a ATI DAQ F/T Power Supply with an internal amplifier for the loadcell. Each loadcell setup was read into a Dell Optiplex 780 using a National Instruments NI-6280 A/D Converter with a 16-bit resolution. A loadcheck was conducted with entire assembly to ensure accurate readings of prescribed data as well as check wing tip deflection. To do this, the entire measurement setup was fixed horizontally and leveled using a digital level. A measuring bar was placed perpendicular to the wing to allow for displacement to be measured as the loading increased. The results are seen below:

<table>
<thead>
<tr>
<th>Sequence</th>
<th>Load (N)</th>
<th>Load Measured (N)</th>
<th>Deflection (mm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Tared/Bias</td>
<td>8.56</td>
<td>9.566</td>
<td>0.0</td>
</tr>
<tr>
<td>Weight 1 Added</td>
<td>31.80</td>
<td>34.06</td>
<td>2.0</td>
</tr>
<tr>
<td>Weight 2 Added</td>
<td>54.49</td>
<td>58.7</td>
<td>3.5</td>
</tr>
<tr>
<td>Weight 3 Added</td>
<td>76.51</td>
<td>83.03</td>
<td>5.5</td>
</tr>
<tr>
<td>Weight 4 Added</td>
<td>86.35</td>
<td>93.83</td>
<td>6.0</td>
</tr>
</tbody>
</table>

Because of the higher load limits available for the Delta loadcell, the 95% confidence interval in lift of 11.5N as prescribed by ATI. Therefore, many of the measurements conducted are within this confidence interval. With a resolution of $1/8N$, a self load check was conducted to determine the actual uncertainty present during testing. To do this, the wing was loaded at all angles of attack in a horizontal configuration. The test was conducted 10 times sequentially, similar to dynamic oscillations.

### 2.1.4 Test Article

The test article used for the parameter study was a NACA 0018 steel composite airfoil. The airfoil can be broken down into 3 main components: external fiberglass shell, internal support structure, and mounting hardware. The nature of the research requires a smooth airfoil with little imperfections. Because of this, it was important to have a clean leading edge and transition to the upper surface to prevent tripping of the boundary layer or distortion of the separation location over the airfoil. To create the surface, the airfoil exo-skeleton was cut into 2 large pieces with a seam located on the lower surface at $x/c = 10\%$ as denoted by the green line in Fig 2.6. The connection of the skins in this location would allow for minimal interference of separation location and overall behavior of the airfoil during testing. To create the 2 separate pieces, three molds were created.
Using SolidWorks, part files were created and uploaded to a CNC mill where the negatives were milled into layers of multi-density fiberboard (MDF). After milling, the MDF molds contained small fibers that had to be removed so a thin layer of epoxy could line the molds to allow fiberglass positives to be created. To create a smooth surface finish, the molds were sanded by hand using 200 & 300 grit sandpaper. A thin layer of West Systems Epoxy-Resin mix was layered onto each mold using a foam brush. The thin layer created a barrier between the MDF board and future fiberglass layers. After applying the epoxy layer, each mold was sanded with 400 grit sandpaper. The entire process was repeated 5 times to allow for a strong barrier that would not deform or tear when creating external fiberglass skins. The smooth leading edge mold was created by using 1 mold that contained the entire upper surface geometry, from trailing to leading edge. Mounted atop the full mold was a small quarter circle mold that encompassed the lower surface geometry from the leading edge to x/c=10%. This assembly was bolted together, creating a negative mold that could encompass the entire upper surface, leading edge, and 10% of the lower surface. The second external mold created the remaining lower surface.

The outer layers of the airfoil were created using 4 layers of 6 oz. fine weave fiberglass and West Systems epoxy. Partial Film #10, a water and alcohol based polyvinyl alcohol (PVA) was sprayed over the mold and acted as a release agent for the layups. The first attempt at laying up the large upper surface ended poorly due to fiberglass wrinkles occurring at the transition from upper to leading edge surfaces. This likely occurred due to an attempt to vacuum bag after the layup was complete. Due to the nearly negative draft angle, the peel ply and breather fell from the lower surface curve, causing an indentation to occur along the length of the part. During the second attempt, it was decided to not vacuum bag the mold to prevent this from occurring. Although vacuuming bagging is ideal to hold the shape of a mold, the small number of layers used in the layup allowed for the shape to not be compromised during curing.

The internal structure of the test article was constructed of stainless steel ribs and fiberglass rods. Using SolidWorks, 4 airfoil profiles with 3 circular slots for carbon fiber struts and dye tubes were cut out of 1 6061 stainless steel plate using wire-EDM. Using the wire EDM allowed for the airfoil cutouts to be highly accurate when compared to the specified airfoil profile.
The airfoil test article was designed to have a 26 span to allow for a 2 clearance from the free surface to the mounting hardware and subsequent load cell. The stainless steel ribs were interconnected using 2 carbon fiber rods with outer diameter of 0.5" & 0.375". These 2 rods attach the 4 inner steel ribs together using epoxy resin. To prevent scratching of the test section floor, a PVC cutout of airfoil profile was mounting on the end of the wing. After the internal structure was assembled, it was placed inside the large upper surface mold and attached using epoxy and many clamps. The surface was allowed to cure for 36 hours. For dye visualization, 4 ports were created on the airfoil. Port locations included the leading edge, 0.20c, 0.40c, and 0.60c along the upper surface. The ports consisted of 0.015 polyvinyl tubing that was flush mounted to the surface of the airfoil. The remaining 4" of tubing was run through the center hole of the airfoil ribs where is could be attached to dye extrusions devices. For mounting, a 0.5 aluminum plate was attached to the airfoil using epoxy resin and carbon fiber rods. The rods were inserted into the plate with epoxy resin and trimmed to allow for a flush mount to the airfoil. The plate contained sets of holes and reference points that allowed the airfoil to be pitched at $\alpha = 0^\circ, 6^\circ, 13^\circ, \text{and } 20^\circ$. Modifications to the original plate also included 4 slotted channels which allowed for the incremental pitching on the airfoil.

The completed geometry of the airfoil was validated using a laser scanner provided the NCSU Industrial Engineering Department. Geomagics software was used used to measure the accuracy of the airfoil down to 24 micro-meters. Due to the size limitation of the scanner arm, only a 12” span, ±6” from the center-line was evaluated. The software compares a digital scan of the wing to a SolidWorks model of the assembly. The two are combined in the software using a reference point, located at the base of the airfoil stand. The change between the 2 can be calculated, thus giving a position difference of the scanned surface from ideal. It is important to note that the reference line used for the comparison was only as accurate as the mounting plate in which the wing was connected.
As shown in Fig. 2.7a, the upper surface as a average 3D deviation of \( \pm 0.04" \) from the predictive value. For the lower surface, the average deviation is \( \pm 0.03" \). Both values are within 1/100th of the overall chord length. This indicates that the airfoil geometry is correct in manipulating a NACA 0018 and should behave as such.

2.1.5 Dynamic Dye Visualization

Flow visualization around the test article was done using dye injection from the upper surface. Rhodamine 6G, a powder in its normal state, was mixed with water and placed into a 100mL dye syringe and placed in a MX4048 Positive Displacement Syringe pump. For attached flow cases, dye was ejected from the leading edge dye port at a volumetric flow rate of 1.5mL/min. The flow rate was determined by trial and error testing in which dye concentration was high.
enough for viewing below the test section, but did not disturb the boundary layer or create synthetic jets, causing a disturbance of the flow field. For separated cases, a seen at $\alpha = 20^\circ$, dye was extruded at the leading edge, 0.40c and 0.60c dye port location. Because of heavily detached flow and additional ports, the volumetric flow rate on the syringe pump was increased to 4.5mL/min. Although the intent was to have 3 ports running simultaneously, differences in line length caused a variation between the 0.40c and 0.60c ports during testing. Even with this discontinuity, dye visualization was successful in capturing the relative flow field around the airfoil. For illumination, three-1W 520nm LED constant illumination LEDs were placed perpendicular to the upper surface of the airfoil. Because of high $\lambda$ values, the LEDs were mounted to the upper gantry and suspended on a 1”x1” wooden gantry. Wood was chosen for the support system to keep weight and inertial movement to a minimum. The diodes were connected in parallel and contained a X-driver circuit that allowed all diodes to be powered on simultaneously using a Techtronix MB-800 power supply. Power cables for the system were run through the signal side cable carrier and connected to the support system. The LEDs were aligned perpendicular to the upper surface by ensuring the returning beam, reflected from the opposite side wall and glass test section, was aligned with the original reflection of the beam. To focus the laser, the waist of the LED beam was adjusted to focus on the dye ports and hot-glued into place to prevent drift during testing. Horizontal alignment was done by aligning the laser sheet with the 4 dye ports along the airfoil. Between all 3 laser diodes, the overall laser sheet thickness was 3-5mm. Although this is thick for a scientific laser sheet, the use of dye injection does not require a ultra-thin laser sheet similar to PIV methods. Capturing of the dye flow visualization was done using a Photron AX200 Mini 1Mpx High Speed Camera. For tracking separation location and flow fields on and around the airfoil, a Nikon 45mm tilt shift lens with a Tiffen 77mm filter was used. This method is modeled from Granlund et al. [21]. A frame rate of $250$fps was chosen to allow for 1-2 images per phase angle during high k motions. The camera that was mounted vertically on the lower dual rail stage using mounting hardware and followed the exact motion of the upper stage gantry. Post processing of the images was done using Adobe Photoshop in which the images were inverted and contrast adjusted for viewing.

2.1.6 LabView Interface & Control

Force data was recorded into a LabView graphic interface based on a LabView processor created by ATI. The LabView control was modified to include a 5V external trigger that signaled the Galil motion controller to begin oscillations. The same control also initiated data recording into prenamed text files that would be used for data processing. The signal was recorded at 1kHz and read 100 samples into the GUI as well as the text file. After the motion of 25 cycles was complete, the recording continued for 10 seconds to obtain a constant freestream value that
would be used for later post processing. To check for external delay, the loadcell was pre-loaded with a 5 lb. weight in the positive x-direction. During a step function performed by the motor gantry, the loadcell produced a distinct peak. During the test, the trigger voltage, which initiates both recording and motion, was recorded using a LabView SCXI-1000 interface. The trigger test was conducted 5 times to check for phase delay in either motion start or data recording. It was concluded that the delay between motion start and force recording was negligible, averaging between 1-3ms. It should also be noted that the delay is less than the quickest phase angle present in the parameter study. It should also be noted that the phase delay recorded was also dependent on the hardware available for testing. Because of this, it would be a educated "guess" to determine a factor to shift the data by phase angle. Because of this, it was determined that no corrections would need to be made.

2.1.7 Error & Uncertainty Analysis

Uncertainty analysis was conducted using the Kline McClintock method. As stated previously, the ATI Delta used for the research contained at 95% interval of 11.5N. With a resolution of 1/8 N, a self prescribed load check was conducted on the loadcell with the test article attached to mimic testing scenerios. The wing was loaded along the centerspan with weight. The load was loaded and unloaded 3 separate times to mimic a cyclic motion.

Figure 2.8: ATI Delta Load Test
Although it difficult to see, the load test has error bars that peak at the max load case of 85N with a 0.7% uncertainty. This will be used for the max uncertainty of force measurements as well as used for subsequent calculations for $C_L$. The water tunnel freestream is controlled by the VFD and is controlled in Hz, therefore, it is assumed the VFD is accurate to $\pm 0.1$Hz. This corresponds to a $V_\infty$ change of 0.0016m/s. Water temperature was assumed to be $15^\circ C < T < 25^\circ C$. Span (b) and chord(c) where measured with calipers and could be measured to 1/1000 inch. However, because of a slight varying trailing edge, this was extended to 1/4 inch for span and 1/8inch for chord. The governing equation for error analysis in the relation between lift coefficient and raw measured lift:

$$C_L = \frac{L}{\frac{1}{2} \rho V^2 bc} \quad (2.2)$$

Table 2.4: Equations for uncertainty analysis

<table>
<thead>
<tr>
<th>Derivative</th>
<th>Equation</th>
<th>Value</th>
<th>Reference</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\frac{\partial C_L}{\partial L}$</td>
<td>$\frac{2}{\rho V^2 bc}$</td>
<td>0.0306</td>
<td>$0.595N$</td>
</tr>
<tr>
<td>$\frac{\partial C_L}{\partial \rho}$</td>
<td>$-\frac{2L}{\rho^2 V^2 bc}$</td>
<td>-0.0026</td>
<td>$5^\circ C$</td>
</tr>
<tr>
<td>$\frac{\partial C_L}{\partial V}$</td>
<td>$-\frac{4L}{bc p V^3}$</td>
<td>-6.062</td>
<td>$0.10 kg/m^3$</td>
</tr>
<tr>
<td>$\frac{\partial C_L}{\partial b}$</td>
<td>$-\frac{2L}{b^2 c p V^2}$</td>
<td>-4.263</td>
<td>$0.00635 m$</td>
</tr>
<tr>
<td>$\frac{\partial C_L}{\partial c}$</td>
<td>$-\frac{2L}{c^2 p V^2}$</td>
<td>-17.796</td>
<td>$0.003175$</td>
</tr>
</tbody>
</table>

$$\sigma [C_L] = \sqrt{\left(\frac{\partial C_L}{\partial L}\right)^2 \sigma (L)^2 + \left(\frac{\partial C_L}{\partial \rho}\right)^2 \sigma (\rho)^2 + \left(\frac{\partial C_L}{\partial V}\right)^2 \sigma (V)^2 + \left(\frac{\partial C_L}{\partial b}\right)^2 \sigma (b)^2 + \left(\frac{\partial C_L}{\partial c}\right)^2 \sigma (c)^2} \quad (2.3)$$

28
Table 2.5: Force and Lift Uncertainties

<table>
<thead>
<tr>
<th>Property</th>
<th>Units</th>
<th>Reference Value</th>
<th>Absolute Uncertainty</th>
<th>Relative Uncertainty</th>
</tr>
</thead>
<tbody>
<tr>
<td>Force ($L, D$)</td>
<td>N</td>
<td>85</td>
<td>±0.765</td>
<td>±1.79%</td>
</tr>
<tr>
<td>Water Density ($\rho$)</td>
<td>kg/m$^3$</td>
<td>999.10</td>
<td>±0.10</td>
<td>±0.00%</td>
</tr>
<tr>
<td>Airfoil Wet Span ($b$)</td>
<td>m</td>
<td>0.6096</td>
<td>±0.00635</td>
<td>±0.13%</td>
</tr>
<tr>
<td>Airfoil Chord ($c$)</td>
<td>m</td>
<td>0.14605</td>
<td>±0.003175</td>
<td>±0.02%</td>
</tr>
<tr>
<td>Freestream Velocity ($V_\infty$)</td>
<td>m/s</td>
<td>1.00</td>
<td>±0.00162</td>
<td>±0.02%</td>
</tr>
</tbody>
</table>

Using the values denoted in Table 2.5 and Eqt.2.3, $C_L$ uncertainty is 0.0676. This is well within the limits needed to validate airfoil performance.

2.2 Experiment Validation

This parameter study was the first study to use the NCSU Water Tunnel for its design goals and objectives. Although each subsystem had been validated as described in Sections 2.1, it was important to ensure that that the overall system behaved in a respectable manner when comparing to other airfoil performance databases. Because the system was untested, it was important to conduct basic experiment to understand the test article behavior and the reliability of the system. Although tunnel parameters have been quantified by Stewart [1], aerodynamic performance metrics were untested and required validation.

2.2.1 Angle of Attack ($\alpha$) Sweep

The first validation test conducted was a angle of attack sweep through positive incidence. The study allowed for airfoil performance metrics to be studied and plotted: $C_{Lvs.\alpha}$, $C_{mvs.\alpha}$, & $C_{Lvs.C_d}$. The test was conducted at $Re = 125,000$ and consisted of a sweep spanning the entire section length to ensure no significant differential in airfoil performance was noticed over a sweeping motion. Test locations included $x/X = 0.125, x/X = 0.500, & x/X = 0.875$. For angle of attack indication, paper markers, created in SolidWorks, were placed on the mounting hardware and aligned with the 0 position as indicated on the existing hardware. Each mark indicated a 1° increase and was adjusted using the slots as indicated in Sec.2.1.4. To account for any gantry deflection, a tare value was taken at $x/X = 0.875$. The gantry was then moved to the test location where a tare value was recorded for the loadcell. This tare value was then subtracted from subsequent calculations. For performance comparisons, XFOIL was evaluated at varying turbulence values. XFOIL [22] is an inviscid linear-vorticity panel method that uses a
two-equation lag integral method to determine laminar and turbulent boundary layers along a airfoil profile. A Newton iteration as well as a \( e^9 \) amplification formula allows for XFOIL to predict turbulent separation location as well as turbulent separation bubbles. As \( \alpha \) begins to reach towards \( \alpha_{stall} \) XFOIL predictions begin to break down as noted in Gopalarathnam [23]. It should also be noted that at low Reynolds numbers, as performed in this parameter study, XFOIL predictions are highly sensitive to turbulence intensity, as noted by a amplification factor, \( N_{crit} \).

Figure 2.9: \( C_L \) vs.\( \alpha \), Re = 125,000 with XFOIL Comparison

Figure 2.9 displays a strong trend between predicted values and experimental results. One key validation point in the figure is collapsing of all x-positions to one single performance line. With this validation, it can be concluded that airfoil performance is not dependent on relative position inside the test section, despite slight change in turbulence intensity changes as noted in Stewart [1]. For airfoil characteristic performance, it can also be seen that \( \alpha = 0 \) displays a negative lift value, indicating that the "0" position as set in the tunnel creates a model that creates slightly negative lift (-1N). Because of this, the indicated angles of attack behave slightly less than indicated on the support mount. The \( \alpha_{zt} = -0.75^\circ \). However, it should also be indicated that the first data points, \( \alpha = 0^\circ \) & \( 1^\circ \) display a slight dip in the curve when compared to higher \( \alpha \) values. Because of this, it is possible that the airfoil does not need a correction factor, but
should be checked with an accurate servo-pitch motor. Nevertheless, a correction of $\alpha = -0.75$ was done to shift the data to allow for $\alpha_{zl} = 0$.

Figure 2.10: $C_l$ vs. $\alpha$, correct for $\alpha_{zl} = -0.75$

Figure 2.11: NACA 0018 Performance Results for Parameter Study
The same test also concluded $C_m$ and $C_d$ results. Overall, both parameters follow similar trends as XFOIL, although slightly off from exact values. This is to be expected due to the simplifications done by XFOIL as well as the testing setup. It is also important to note that low-order tools such as XFOIL are subject to error when Reynolds number is less than 100k. In $C_m$ plot, a distinct dip in pitching moment is noted where the $C_m$ increases from negative to positive. This dip is noted in XFOIL predictions as well as work conducted by Greenblatt [14]. In aerodynamic testing, drag coefficient is difficult to measure. Because drag forces are typically lower in relation to lift in attached flow conditions, $C_d$ values are difficult to obtain, particularly with a large 6-DOF loadcell. However, $C_d$ values obtained through the sweep show a similar trend to XFOIL and are within the same order of magnitude in value. This indicates that all forces, lift, drag, and pitching moment, can be used for experimental validation when surged.

2.2.2 Reynolds Number Sweep

In high velocity amplitude research, $\lambda > 0.3$, it is standard to normalize unsteady lift by the instantaneous Reynolds number mean lift value. Although this practice will be evaluated in this research, a Reynolds number sweep was conducted on the test article. The test ranged from $30k < Re < 138k$ due to test section speed and airfoil chord length. The airfoil was set to $\alpha = 6^\circ$ and placed in the center of the test section. For comparative data, XFOIL was again used to have a metric on which to measure airfoil performance. To account for changing turbulence levels inside the test section, the critical amplification factor, $N_{crit}$, was changed for each XFOIL batch and denoted as $N_{crit} = Shift$. This was done because water tunnel turbulence levels increased as freestream velocity increased, which was the method used to measure raw $L_0$ values. It should be noted this method was only used to create a predicted value for $C_l$ for the test article and was not used for later interpolation.
Because of size and speed restrictions of the facility, the testing gamut of the unsteady range could not be tested statically. During a $\lambda = 0.5$ motion, the instantaneous Reynolds number ranges $62.5k < \text{Re} < 187.5k$. With a $\text{Re}_{\text{max,e}} = 138k$, values were interpolated based on theoretical trends provided by XFOIL. Despite varying peaks in the theoretical data, each simulation converges to a value between 0.55 - 0.61. Because the test article displayed a slightly higher $C_l$ value at $\text{Re} = 140k$, it was determined that $C_l$ would continue to decrease, similar to all XFOIL predictions, before leveling off slightly above XFOIL predictions. Similar to Fig.2.12, the raw lift values recorded during the Reynolds number sweep could be used to determine a Reynolds number normalization as discussed earlier. For $\text{Re} = 75k$, all motion ranges could be captured using the tunnels freestream velocity.
For Re = 125k, interpolations had to be made to extend the curve to fit the instantaneous Reynolds numbers present in the motion. $C_L$ values were extrapolated using XFOIL data provided by Fig. 2.12. For Re = 125k, this extrapolation and test was conducted 3 times for each angle of attack test. For Re = 75k, the same method was applied, but extrapolations were not needed since the overall Reynolds number present in the motion could be tested. As noted in Fig. 2.13, interpolation between each data point is linear. To prevent non-physical interpolations from being present, each Reynolds number in between measured data points was interpolated linearly. If a second order curve was used, this would have implied a constant $C_L$ value which is incorrect. This interpolation was implied in MATLAB and calculated for every Reynolds number present during the motion.
As shown in Fig. 2.14, as the airfoil move through a quasi-steady motion profile, $L_0$ does not vary at the same rate as the instaneous Reynolds number. Because it is a measured value that is linearly interpolated, the overall profile is not purely sinusoidal and is not symmetrical. This is the interpolated profile that will be used to compare Reynolds number normalization. This process was conducted for each $\alpha$ case.
Chapter 3

Effect of Velocity Fluctuation Amplitude at Constant Reduced Frequency

This parameter study in total investigated 150 test cases, varying Reynolds number, $\alpha$, $\lambda$, and $k$. The breakdown of results will be done by comparing each $\alpha$ case in Reynolds number and normalized parameters. Performance overviews will be done with force measurements and subsequent dye visualizations. After comparing airfoil performance, Reynolds number normalization will be presented and studied for each case. The final result will be the normalization of the fluctuating components as presented in Eq. 1.24.

3.0.1 $\alpha = 6^\circ$

$Re = 75k, \alpha = 6^\circ$ In steady flow, at $Re = 75k$ and $\alpha = 6^\circ$, the overall performance of the airfoil in nominal, with $C_l = 0.62$. Laminar flow is present over the airfoil up to $x/c = 0.4$ before transitioning to turbulent flow. Boundary layer thickness is thinner at the leading edge and grows until transitions occurs. Overall, the flow is attached over the entirety of the airfoil as shown in Fig. 3.1
For this parameter, $k = 0.1$, $0.1 < \lambda < 0.5$, as shown in Fig. 3.2, quasi-steady lift production follows Greenberg’s predictions and typically scales with increased $\lambda$. Pitching moment, although not comparable to Greenberg, displays a shift from positive to negative normalized pitching moment between, $0.25 < t/T < 0.75$, increasing in magnitude as $\lambda$ increases. Although trending is comparable to Greenberg, there is slight unsteady amplification of lift at $t/T = 0.75$. All values except $\lambda = 0.4$ have peak values that are higher than predicted. In pitching moment, we expect to see normalized pitching moment about $c/4$ to stay at 0 since it is based on TAT. Measured values show that pitching moment is non-zero and stays constant until $t/T = 0.25$. As the airfoil begins to slow, normalized pitching moment decreases in magnitude and scales with increasing $\lambda$. Because the airfoil displays a negative pitching moment at steady-state, there is a positive pitch up moment at $t/T = 0.50$. This would be indicative of separation growing from the trailing edge. This can be seen in Fig. 3.3, as time increases, separation region from the trailing edge, causing the separation bubble to grow significantly and reduce suction on the
upper surface. This reduction in upper surface suction causes a pitch up moment.

**Re = 125k, α = 6°** Steady flow dye visualization at Re = 125k, α = 6° shows a thinner laminar boundary layer up to 0.5c before transitioning to turbulent. At 125k, the flow structure is similar to 75k, but thinner incoming boundary layer and higher frequency motion of the turbulent boundary layer causes slight performance differentials which are highlighted in the
dynamic motions.

Figure 3.4: Re = 125k, \( \alpha = 6^\circ \), Steady Flow Profile

![Figure 3.4](image)

As shown in Fig.3.5, there is a large deviation from Greenberg’s prediction in lift at \( t/T = 0.25 \) when compared to experimental data. Although each \( \lambda \) increase trends well at \( t/T = 0.75 \), the forward surge motion shows significant unsteady lift attenuation when compared to theoretical predictions. The moment displays a much large parabolic moment production, but is positive for much of the surging motion, unlike Re= 75k. It is also interesting to note that at \( t/T = 0.75 \), each \( \lambda \) increase returns to a similar normalized point at minimum dynamic pressure.

Figure 3.5: Re = 125,000, \( \alpha = 6^\circ \), k = 0.1, 0.1 < \( \lambda < 0.5 \)

![Figure 3.5](image)
of the motion. To compare this case, the lift profile of each case is co-plotted along with dye visualization at $t/T = 0.75$.

![Graph showing lift profile comparison](image)

Figure 3.6: Quasi-steady lift, $\alpha = 6^\circ$, $k = 0.1$, $0.1 < \lambda < 0.5$

As shown in Fig. 3.6, at maximum dynamic pressure, a large differential between quasi-steady normalized circulatory lift is observed. Dye visualization at $t/T = 0.25$ offers differences at the performance of the airfoil at this instance:

![Dye visualization](image)

Figure 3.7: Dye injection at leading edge for $k=0.1$, $\lambda = 0.2$ for $\alpha=6^\circ$ at $t/T = 0.25$, $Re=75k$ (top) and $Re=125k$ (bottom)
For Re = 75k, a laminar separation bubble is present at t/T=0.25 as shown in Fig. 3.7. In this scenario, it appears that the laminar separation bubble does not attenuate lift but causes the airfoil to behave similar to thin airfoil, indicating that the bubble adds to the overall circulation of the airfoil due to a slight effective cambering of the symmetric airfoil. At Re=125k, a thinner laminar boundary layer transitions directly to a turbulent boundary layer at max dynamic pressure. This behavior would not indicate a loss of lift, however, because of the thick airfoil geometry, minimal trailing edge separation at the high Reynolds number can be contributed to decambering the airfoil, causing lift production to decrease as the airfoil passes through max instantaneous Reynolds number. Overall performance at Re=125k at 6° varies slightly with increased k and λ. In the forward surge, there is a deficiency in lift up to t/T = 0.5. As the airfoil retreats in the motion, t/T = 0.5, there is a strong correlation in lift when looking at normalized lift. A possible explanation for this is a difference in circulation as the airfoil retreats in the motions. Although flow reversal does not exist, it is possible that trailing edge vortexes in the retreating motion cause the normalized lift to be higher when compared to the forward surge. Apart from overall performance, another aspect of this study was to determine if quasi-steady Reynolds number correction could be use to better normalize raw lift data.

Fig. 3.8 demonstrates that at a fixed k, Reynolds number effects are dependent on the

Fig. 3.8: Reynolds Number normalization, α = 6°, k = 0.25, λ = 0.1 & 0.5
physical time domain that governs the overall motion. Lower amplitudes at \( k=0.25 \) allow the Reynolds number relation to collapse closer to theoretical predictions of Greenberg, indicating that the normalization can account for Reynolds number and unsteady effects if the overall acceleration of the motion is slow enough to approach a quasi-steady condition. As the physical time domain is shortened between Reynolds numbers, the normalization breaks down. As \( \lambda \) increases, there is unsteady effects that are not captured by this normalization. In Fig. 3.8, under this assumption, accounting for instantaneous Reynolds number change throughout the motion would allow only unsteady effects to be present in the resulting plot. However, for both Reynolds number regimes, Reynolds number effects play a role in the behavior of the airfoil but are not instantaneous. This effect can be seen in the boundary layer growth and decay as the airfoil moves through a profile. Although there is no separation bubble present at \( \text{Re} = 125k \), attached flow still varies in boundary layer thickness and separation, causing lift production to vary, but not instantaneously. A phase lag between instantaneous velocity and flow response is seen and mimics Greenberg’s derivation. Vortex shedding and boundary layer movement are not in phase, and therefore cannot simply be extracted as shown, even at fully attached flow.

![Figure 3.9: Dye Visualization for \( k=0.1, \alpha=6° \) for \( \lambda = 0.1, 0.3, 0.5 \) at \( t/T = 0.75, \text{Re}=75k \)(left) and \( \text{Re}=125k \)(right)](a) \( \lambda = 0.1 \)  
(b) \( \lambda = 0.1 \)  
(c) \( \lambda = 0.3 \)  
(d) \( \lambda = 0.3 \)  
(e) \( \lambda = 0.5 \)  
(f) \( \lambda = 0.5 \)

In Fig.3.9, we see that the behavior of the airfoil boundary layer is dependent on the overall time of motion of the airfoil. As \( \lambda \) increases at \( \text{Re} = 75k \), the separation bubble, shown on the left hand side, begins to grow in size before dispersing right before the startup surge. The bubble is essentially carried through the motion, due to a Reynolds number change as well
as unsteady effects caused by the boundary layer change. At Re =125k, the boundary layer thickens before transitioning to turbulent. Although this occurs at nearly all λ values, increased λ causes turbulent vortices to become larger as shown. We can show this be displaying where the airfoil reaches a instantaneous Reynolds number of 75k during a mean Re = 75k.

In Fig. 3.10, all 3 images capture the same instantaneous Reynolds number of 75k. However, there is a clear difference between boundary layer profile and subsequent wake. At t/T = 0, the airfoil has accelerated from minimum dynamic pressure to the mean Re = 75k. At this instance the boundary layer is attached for slightly longer that in the static case At t/T = 0.5, the airfoil has decelerated from maximum dynamic pressure to the mean Re =75k. Here the opposite effect can be seen. The boundary layer separates closer to the leading edge before transitioning to turbulent and reattaching.
3.0.2 $\alpha = 13^\circ$

\textbf{Re = 75k, $\alpha = 13^\circ$} At $\alpha = 13^\circ$ at Re = 75k, the airfoil has a steady state $C_l = 1.02$. In this configuration, the airfoil is on the verge of stalling and is nearing max $C_l$. Because of this, each section of the surging motion causes the airfoil to transition from attached flow to separated via the trailing edge. The steady state flow profile of the airfoil can be seen below in this configuration:

![Figure 3.11: Re = 75k, $\alpha = 13^\circ$, Steady Flow Profile](image)

As shown in Fig.3.11, the airfoil has a laminar boundary layer before separating instantaneously as the flow reaches the adverse pressure gradient on the airfoil. It is difficult to say whether the flow is reversed at all anywhere along the upper surface. At the trailing edge of the airfoil, there is a slight change in dye flow angle, but overall, it can be assumed that the airfoil is mostly attached in the static case. Turbulent boundary layer mixing above the airfoil causes the dye boundary layer to grow, but is not separated from the airfoil. The lift profile and moment profile of a motion are shown:
Figure 3.12: $Re = 75k$, $\alpha = 13^\circ$, $k = 0.1$, $0.1 < \lambda < 0.5$
(a) $t/T = 0.00$

(b) $t/T = 0.25$

(c) $t/T = 0.50$

(d) $t/T = 0.75$

Figure 3.13: $Re = 75k\alpha = 13^\circ, k = 0.10, \lambda = 0.5$
For this parameter, \(k = 0.1, 0.1 < \lambda < 0.5\), as shown in Fig. 3.12, quasi-steady lift production does not follow Greenberg’s predictions as well as 6°. Closer inspection has the measured values almost in opposite phase of predictive values. It is interesting that as \(\lambda\) increases, this shift in phase increases to nearly 180° out of phase and increases in magnitude. Quasi-steady moment values display a similarity to quasi-steady lift values with increasing magnitudes as \(\lambda\) increases. At 13°, the onset of separation causes the moments to stay constant through the \(\lambda\) sweep. Because the performance of the airfoil is dependent on physical time, the quickness of the motion dictates the performance of the airfoil. This physical time difference can be contributed to an advection lag of the boundary layer as the motion occurs. As shown however, this lag is not in phase with dynamic pressure. Because the mean values of pitching moment are nearly the same for all cases, we see a strong shift in lift and pitching moment as \(\lambda\) changes. Although it is not shown, the same effect occurs with increasing \(k\) at constant \(\lambda\). In Fig. 3.13, the time stamps provide insight into airfoil performance and the attenuation in lift shown in Fig. 3.12. At \(t/T = 0.25\), there is fully attached, turbulent flow that is causing lift to be higher than mean values due to the increase in velocity and overall flow structure. At \(t/T = 0.50\), separation begins to appear as the airfoil enters the retreating phases, causing large separation vortexes to occur on the upper surface. Unlike the 6° cases, the higher \(\alpha\) cases allow for the flow to fully advect away from the airfoil, causing a large attenuation in lift. At previously mentioned, this is dependent on a physical time scale which is dependent on \(\lambda\) and \(k\).

\[Re = 125k\] At the higher Reynolds of 125k, the laminar boundary layer present is thinner than the 75k case. However, the transition point is near the same \(x/c\) location as 75k case and transition to turbulent. The turbulent boundary layer has much smaller and faster vortices that are present inside the boundary layer, causing the over dye visualization profile to be much thinner and uniform when compared to the 75k case. Like the 75k case, the overall profile is considered attached although it is on the verge of beginning to stall with a \(C_l\) of 1.14 which is slightly higher than \(Re= 75k\) case.
As shown in Fig. 3.15, quasi-steady lift production is somewhat in phase of Greenberg’s prediction when compared to the Re = 75k case. For the forward surge motion, the agreement is consistent except for the $\lambda = 0.5$ case which has a similar trend but slightly lower values overall. At minimal dynamic pressure of the motion, there is also a dip in lift production which grows as $\lambda$ increases. This dip could be representative of a vortex being shed as the airfoil is losing circulation and lift. The moment profile also displays a constant value of $t_0/T = 0.75$ where a slight change is seen through all the motions.
Figure 3.16: Quasi-steady lift, $\alpha = 13^\circ$, $k = 0.1$, $0.1 < \lambda < 0.5$

Figure 3.17: Dye injection at leading edge for $k=0.1$, $\lambda = 0.5$ for $\alpha=13^\circ$ at $t/T = 0.25$
As shown in Fig. 3.17, both Reynolds numbers appear to have attached flow over the upper surface, but at Re = 75k, the flow is turbulent nearly instantaneously at max dynamic pressure while Re=125k appears to have brief laminar boundary layer phase. It is difficult to say what is exactly happening at the very beginning of the boundary layer, but is obvious that the overall thickness is thinner than Re=75k and produces a small wake off the trailing edge of the airfoil. Like $\alpha = 6^\circ$, a Reynolds number relation is presented to see if a static Reynolds sweep can be used to eliminate Reynolds number effects and only study unsteady effects.

As shown in Fig. 3.25, Reynolds number normalization based off static force measurements does not collapse raw measured lift into simple unsteady lift production. The reason for this is a time dependent delay on boundary layer movement as the airfoil moves through the oscillatory motion. As the acceleration of the motion increases, the profile begins to change and moves further from static behavior as shown below.
Figure 3.19: Dye Visualization for $k=0.1$, $\alpha=13^\circ$ for $\lambda = 0.1$, 0.3, 0.5 at $t/T = 0.75$, $Re=75k$ (left) and $Re=125k$ (right)

As shown in Fig. 3.19, trailing edge separation, particularly at $Re=125k$, is dependent on the overall acceleration of the motion, causing the retreating motion to see full separation from the upper surface. At $Re = 75k$, the flow is clearly turbulent, but tends to stay at or near the upper surface.
Like the 6° case, normalizing around instantaneous Reynolds number over the motion will not produce the theoretical values of Greenberg due to the means by which the normalization occurred. Static measurements cannot be translated to dynamic effects, even if put in a sinusoidal motion.
3.0.3 $\alpha = 20^\circ$

Re = 75k, $\alpha = 20^\circ$. At $\alpha = 20^\circ$, the airfoil is fully stalled in both Reynolds number cases and displays large amount of separation. Because of this, Greenberg’s theory should not match as it is based off attached flow theory and circulation shed from the trailing edge. In mean flow, the airfoil has flow structure as follows:

![Figure 3.21: Re = 75k, $\alpha = 20^\circ$, Steady Flow Profile](image)

As shown in Fig. 3.23, flow separation occurs immediately near the leading edge. The resulting flow exhibits a Kelvin-Helmholz [24] instability in which rolling oscillations are present in the shear layer. Dye injection from the leading edge as well as the 0.60c ports was used to visualize the wake on and around the upper surface. Because of varying dye line length, the dye also came out of the 0.40c, causing a slight differential in pictures. However, the overall wake structure can be seen.
As originally stated, quasi-steady lift was not expected to match theoretical predictions due to detached flow conditions, which is not predicted in theory. When looking at a sample, the quasi-steady lift seems to be out of phase of theory and negated in magnitude. For $\lambda = 0.1$, there are insignificant unsteady lift predictions when compared to $\lambda = 0.5$. At $\lambda = 0.5$, measured values are out of phase with Greenberg’s predictions, showing distinct peaks at $t/T = 0$ and $t/T = 0.75$. This snapshot of the dataset also displays an interesting trend with quasi-steady pitching moment. Although it is not 0, there is very little change between $0.1 < \lambda < 0.3$. It appears that normalized moment tends to hold constant in this configuration until higher $\lambda$ values are approached. Because the airfoil is also fully separated from the leading edge, varying velocity and physical time domain do little to affect the moment through the motion.

\( \text{Re} = 125k, \alpha = 20^\circ \) Because of higher flow speeds and mixing within the flow, \( \text{Re} = 125k \) is difficult to see, but the overall flow field can be understood. Like \( \text{Re} = 75k \), this configuration is fully stalled with large upper surface shedding. The frequency in which the shedding occurs is much faster than the 75k cases, but behaves in similar path and shape.
Figure 3.23: Re = 125k, $\alpha = 20^\circ$, Steady Flow Profile

Figure 3.24: Re = 125k, $\alpha = 20^\circ$, k = 0.1, 0.1 < $\lambda$ < 0.5

Similar to Re = 75k, Fig.3.24 shows a negated and out of phase trend when compared to Greenberg’s prediction, with increasing $\lambda$ causing magnitude increases at $t/T = 0.75$ & $t/T = 1.0$. In the retreating phase, it is possible that the airfoil is still shedding vortices from the upper
surface due to high angles of attack with this trend being exaggerated as the airfoils relative velocity decreases with increased acceleration. Unlike the previous angles of attack, the stalled airfoil causes the vortices to be so far detached from the airfoil that they do not increase lift production despite retreating into the wake where circulation would typically be higher.

Figure 3.25: Reynolds Number normalization, $\alpha = 20^\circ$, $k = 0.25$, $\lambda = 0.1$ & 0.5
Figure 3.26: Re = 75k, α = 20°, k = 0.25, λ = 0.5
At $\alpha = 20^\circ$, each flow structure is heavily dependent on unsteady effects although the effective Reynolds number at each time step is the same. At the start of the cycle at $t/T = 0$ following an acceleration from minimum dynamic pressure, the flow appears to attempt to attach to the upper surface. At $t/T = 0.50$, the separation grows following a deceleration from maximum dynamic pressure at $t/T 0.25$ the flow appears to be fully separated as the airflow ends the forward surge motions and begins to retreat. Both extreme cases vary in separation angle from the leading and correlate with measured lift values. The closer the separation angle is to the mean profile, normalized lift approaches a value near 1. Where this occurs however is not in phase with the effective Reynolds number. As displayed in Fig.3.25, although instantaneous Reynolds number will not collapse data sets to predictive values due a physical lagging of the boundary profile.

3.0.4 Normalization Method

The goal of the normalization technique was to create a method in which comparisons could be made between flow regimes of an airfoil (attached, separated, or semi-attached) and vary $\lambda$ and $k$ values. The concept would allow for proper scaling of certain components of the sinusoidal motion that would allow for certain run cases to be scaled to other run cases. In this report, the focus was on the fluctuating components and the method to reduce down these components. The process, as shown in Sect.1.2.1, used the peak-to-peak values from the quasi-steady lift parameter to attempt to create the relation. Upon using this method, the data, although collapsible, did not reflect the desired "behavior" in which data could be scaled. To attempt to correct this, the non-circulatory terms, originally ignored in the data presentation, was subtracted off by predictive values denoted by Greenberg’s equations. These values, aided in the "collapse" of the data sets, but were less than 10% of the total values. The results from the study are shown below:
(a) Re = 75k, $\alpha = 6'$

(b) Re = 75k, $\alpha = 13'$

(c) Re = 75k, $\alpha = 20'$

(d) Re = 125k, $\alpha = 6'$

(e) Re = 125k, $\alpha = 13'$

(f) Re = 125k, $\alpha = 20'$

Figure 3.27: Normalization Method of Data
In Fig. 3.27, the final normalization used to collapse all data as presented previously is displayed. In the plots, purple diamonds denote data provided by Strangfeld [19] using the Technion gusting wind tunnel at $\lambda = 0.5$. Although the fluctuating components of the quasi-steady terms were used, there was still a increase in peak-to-peak values with increasing $k$ and $\lambda$. This increasing trend relative to $\lambda$ can be seen in Fig. 3.27. Theoretical predictions provided by Greenberg do not fully collapse to a known value and still have a small, but noticeable trend. To try to collapse the data for better understanding, an additional factor of $10\lambda$ was used to bring the magnitudes of each data set closer to one another. This appeared to decrease the margin between $\lambda$ values, but it did not contribute to the increasing $k$ trend. Although numerical manipulation can be done, as done with the $10\lambda$, this is not recommended as there was no mathematical reasoning or assumptions being made. Because of this, either an additional factor or another method may need to be used to investigate high $\lambda$ surging. However, when looking at the data, trends of the experimental and theoretical data can still be made since both sets were manipulated using the same method. In Fig. 3.27, each Reynolds number regime trends well with Greenberg’s theory at $\alpha = 6^\circ$ except for an abnormality at $k = 0.05$, $\lambda = 0.1$. Both regimes have an increasing trend and follow a similar profile as Greenberg. It is important to note that under this method for $k = 0.00$, theoretical and experimental data should go to 0 since there are no unsteady effects when the airfoil is stationary. Although the theoretical data does this, the experimental data does not quite follow as well as it should. However, there is a decreasing trend, that if extrapolated further to lower $k$ values, would intersect around a 0 value. At $13^\circ$, we see a departure from both data sets from predictive values. At Re = 75k, there seems to be a decrease in fluctuating components peak value as $k$ increase from 0.05 until $k = 0.20$, where it begins to increase, indicating that the unsteadiness of the motion actually decreases at certain $k$ values lower than 0.25. It is possible that $13^\circ$, when flow is semi-attached and detached during the motion profile, certain reduced frequencies can actually help the airfoil behave closer to a steady state. Although there is still unsteadiness throughout the profile, the magnitude of this unsteadiness can be somewhat controlled by keeping $k$ at a certain value depending on a variety of flow parameters. At Re = 125k, we see a departure from Greenberg’s predictions around $k = 0.10$ and increasing up to $k = 0.25$. It appears that at Re = 125k, many of the experimental values begin to become constant as $k = 0.20$. This is interesting since Re = 75k, shows an unsteady term that is increasing rapidly at $k = 0.25$. There is clearly a difference in flow structures and behaviors as $k$ increase beyond $k = 0.20$. It would be interesting to see if each trend continued as $k$ increased beyond the measured data sets. As previously stated, $\alpha = 20^\circ$ is stalled and therefore should not display any resemblance to Greenberg’s theory. In both cases, there is very little correlation in quasi-steady fluctuating components from experimental to theoretical. Although trends can be seen in both cases, it is difficult to say physically what is happening as $k$ increases. At Re = 125k, it would appear that fluctuating components actually
converge down to a similar value near $k = 0.05$. This would indicate that when the flow is unsteady in steady flow, the unsteadiness can be increased by small movements of the airfoil. At high accelerations, it appears that the flow is nearly always unsteady, causing the peaks used in this normalization to be near each other. In a physical sense, the unsteady effects present in steady flow appear constantly during the unsteady motion, essentially making the mean flow profile that causes unsteady effects appear relatively small. A final result from the normalization in the agreement between gusting wind-tunnel testing and dynamic test articles. Although only one data point is used, it appears that the 2 methods behave nearly identical at attached flow conditions such as $\alpha = 6^\circ$. As $\alpha$ increases, it would appear that a differential between the two methods is observed and could propagate as flow becomes more unsteady and separation location varies depending on the experimental setup. This could be due to pressure gradients present in the wind tunnel causing flow to stay attached longer or further down the airfoil.
Chapter 4

Conclusions

4.0.1 Conclusions

In this study, a NACA 0018 airfoil was surged at constant incidence at varying Reynolds numbers, angles of attack and surge parameters. Force measurements were taken using a 6-DOF force balance inside a free surface water tunnel. Dye visualization was conducted using dye injection out of leading edge and 0.60c port locations. Reynolds number normalization conducted on all test parameters shows that Reynolds number normalization is not accurate for denoting unsteady effects during a pure surge motion. Instantaneous Reynolds number as described in the theoretical sense does not translate to experimental results due to boundary layer delay and thickness change over a motion. This change is based on a physical time and therefore cannot be implemented into a normalized time study. During a retreating motion, vortices from separation bubbles and trailing edge vortices change in frequency and size, changing the overall performance of the airfoil. Because of higher lift amplitude at $\lambda > 0.1$, fluctuating lift components can be extracted by subtracting out the mean quasi-steady motion from the original quasi-lift values. The peak-to-peak values can be partially collapsed by dividing by $10\lambda$ from the quasi-steady fluctuating component, but introduces non-physical aspects of the normalization that must be noted. Although this occurs, trends in fluctuating components can still be made about the flow structure at certain surge parameters. Physical performance of the airfoil was also documented and compared to a shuttered wind tunnel. Although only 2 points were extracted, initial indications show that at attached flow, both experimental methods produce similar normalized results. As $\alpha$ increases beyond fully attached flow, there could be a departure in results due to pressure gradients, Reynolds number effectiveness, and general testing differences, causing the flow to behave differently.
4.0.2 Future Work

With a well established testing method in the water tunnel, future work for this data set involves the normalization method as presented in Fig. 3.27. As previously mentioned, currently there are non-physical parameters added to the normalization that have no mathematical justification. Because of this, a new method may need to be presented in order to collapse the data into a cohesive structure. The main focus of the future work would be to collapse the theoretical values, allowing for experimental results to be compared to both the theoretical values and other experimental cases. Other future works will also include expanded the motion to surge, pitch, and plunge as well as thin and thick airfoil as varying Reynolds numbers.
REFERENCES


APPENDICES
Appendix A

Reynolds Number $= 75k$ Database

A.0.1 $\alpha = 6^\circ$, $\text{Re} = 75k$, $0.05 \leq k \leq 0.25$, $0.1 \leq \lambda \leq 0.5$ Parameter Sweeps
Figure A.1: $Re = 75,000$, $\alpha = 6^\circ$, $0.05 < k < 0.25$, $\lambda = 0.1$
Figure A.2: \( \text{Re} = 75,000, \alpha = 6^\circ, 0.05 < k < 0.25, \lambda = 0.2 \)
Figure A.3: $Re = 75,000$, $\alpha = 6^\circ$, $0.05 < k < 0.25$, $\lambda = 0.3$
Figure A.4: $Re = 75,000$, $\alpha = 6^\circ$, $0.05 < k < 0.25$, $\lambda = 0.4$
Figure A.5: $Re = 75,000$, $\alpha = 6^\circ$, $0.05 < k < 0.25$, $\lambda = 0.5$
Figure A.6: Re = 75,000, α = 6°, k = 0.05 , 0.1 < λ < 0.5
Figure A.7: $Re = 75,000$, $\alpha = 6^\circ$, $k = 0.10$, $0.1 < \lambda < 0.5$
Figure A.8: $Re = 75,000$, $\alpha = 6^\circ$, $k = 0.15$, $0.1 < \lambda < 0.5$
Figure A.9: $Re = 75,000$, $\alpha = 6^\circ$, $k = 0.20$, $0.1 < \lambda < 0.5$
Figure A.10: \( \text{Re} = 75,000, \alpha = 6^\circ, k = 0.25, 0.1 < \lambda < 0.5 \)
A.0.2 $\alpha = 13^\circ$, Re = 75k, $0.05 \leq k \leq 0.25$, $0.1 \leq \lambda \leq 0.5$ Parameter Sweeps
Figure A.11: Re = 75,000, α = 13°, 0.05 < k < 0.25, λ = 0.1
Figure A.12: Re = 75,000, $\alpha = 13^\circ$, $0.05 < k < 0.25$, $\lambda = 0.2$
Figure A.13: Re = 75,000, $\alpha = 13^\circ$, $0.05 < k < 0.25$, $\lambda = 0.3$
Figure A.14: $Re = 75,000$, $\alpha = 13^\circ$, $0.05 < k < 0.25$, $\lambda = 0.4$
Figure A.15: Re = 75,000, α = 13°, 0.05 < k < 0.25 , λ = 0.5
Figure A.16: Re = 75,000, $\alpha = 13^\circ$, $k = 0.05$, $0.1 < \lambda < 0.5$
Figure A.17: $Re = 75,000$, $\alpha = 13^\circ$, $k = 0.10$, $0.1 < \lambda < 0.5$
Figure A.18: Re = 75,000, α = 13°, k = 0.15 , 0.1 < λ < 0.5
Figure A.19: Re = 75,000, $\alpha = 13^\circ$, $k = 0.20$, $0.1 < \lambda < 0.5$
Figure A.20: Re = 75,000, $\alpha = 13^\circ$, $k = 0.25$, $0.1 < \lambda < 0.5$
A.0.3 $\alpha = 20^\circ$, Re = 75k, $0.05 \leq k \leq 0.25$, $0.1 \leq \lambda \leq 0.5$ Parameter Sweeps
Figure A.21: \( Re = 75,000, \alpha = 20^\circ, 0.05 < k < 0.25, \lambda = 0.1 \)
Figure A.22: \( Re = 75,000, \alpha = 20^\circ, 0.05 < k < 0.25, \lambda = 0.2 \)
Figure A.23: $Re = 75,000$, $\alpha = 20^\circ$, $0.05 < k < 0.25$, $\lambda = 0.3$
Figure A.24: Re = 75,000, α = 20°, 0.05 < k < 0.25, λ = 0.4
Figure A.25: Re = 75,000, α = 20°, 0.05 < k < 0.25 , λ = 0.5
Figure A.26: $Re = 75,000$, $\alpha = 20^\circ$, $k = 0.05$, $0.1 < \lambda < 0.5$
Figure A.27: Re = 75,000, $\alpha = 20^\circ$, $k = 0.10$, $0.1 < \lambda < 0.5$
Figure A.28: $Re = 75,000$, $\alpha = 20^\circ$, $k = 0.15$, $0.1 < \lambda < 0.5$
Figure A.29: $Re = 75,000$, $\alpha = 20^\circ$, $k = 0.20$, $0.1 < \lambda < 0.5$
Figure A.30: $\text{Re} = 75,000, \alpha = 20^\circ, k = 0.25, 0.1 < \lambda < 0.5$
Appendix B

Reynolds Number = 125k Database

B.0.1 $\alpha = 6^\circ$, $Re = 125k$, $0.05 \leq k \leq 0.25$, $0.1 \leq \lambda \leq 0.5$ Parameter Sweeps
(a) Normalized Lift

(b) Normalized Pitching Moment

(c) Quasi-Steady Lift

(d) Quasi-Steady Pitching Moment

(e) Normalized Fluctuating Lift

Figure B.1: Re = 125,000, $\alpha = 6^\circ$, $0.05 < k < 0.25$, $\lambda = 0.1$
Figure B.2: Re = 125,000, $\alpha = 6^\circ$, $0.05 < k < 0.25$, $\lambda = 0.2$
Figure B.3: Re = 125,000, α = 6°, 0.05 < k < 0.25, λ = 0.3
Figure B.4: \( \text{Re} = 125,000, \ \alpha = 6^\circ, \ 0.05 < k < 0.25, \ \lambda = 0.4 \)
Figure B.5: $Re = 125,000$, $\alpha = 6^\circ$, $0.05 < k < 0.25$, $\lambda = 0.5$
Figure B.6: $Re = 125,000$, $\alpha = 6^\circ$, $k = 0.05$, $0.1 < \lambda < 0.5$
Figure B.7: Re = 125,000, α = 6°, k = 0.10 , 0.1 < λ < 0.5
Figure B.8: Re = 125,000, $\alpha = 6^\circ$, $k = 0.15$, $0.1 < \lambda < 0.5$
Figure B.9: $Re = 125,000$, $\alpha = 6^\circ$, $k = 0.20$, $0.1 < \lambda < 0.5$
Figure B.10: $Re = 125,000$, $\alpha = 6^\circ$, $k = 0.25$, $0.1 < \lambda < 0.5$
B.0.2 \( \alpha = 13^\circ, \text{ Re } = 125k, 0.05 \leq k \leq 0.25, 0.1 \leq \lambda \leq 0.5 \) Parameter Sweeps
Figure B.11: Re = 125,000, $\alpha = 13^\circ$, $0.05 < k < 0.25$, $\lambda = 0.1$
Figure B.12: Re = 125,000, $\alpha = 13^\circ$, $0.05 < k < 0.25$, $\lambda = 0.2$
Figure B.13: Re = 125,000, \( \alpha = 13^\circ \), 0.05 < \( k \) < 0.25, \( \lambda = 0.3 \)
(a) Normalized Lift

(b) Normalized Pitching Moment

(c) Quasi-Steady Lift

(d) Quasi-Steady Pitching Moment

(e) Normalized Fluctuating Lift

Figure B.14: $Re = 125,000$, $\alpha = 13^\circ$, $0.05 < k < 0.25$, $\lambda = 0.4$
Figure B.15: $Re = 125,000$, $\alpha = 13^\circ$, $0.05 < k < 0.25$, $\lambda = 0.5$
Figure B.16: Re = 125,000, $\alpha = 13^\circ$, $k = 0.05$, $0.1 < \lambda < 0.5$
Figure B.17: $Re = 125,000$, $\alpha = 13^\circ$, $k = 0.10$, $0.1 < \lambda < 0.5$
Figure B.18: \( \text{Re} = 125,000, \, \alpha = 13^\circ, \, k = 0.15, \, 0.1 < \lambda < 0.5 \)
Figure B.19: Re = 125,000, α = 13°, k = 0.20 , 0.1 < λ < 0.5
Figure B.20: $Re = 125,000$, $\alpha = 13^\circ$, $k = 0.25$, $0.1 < \lambda < 0.5$
\( B.0.3 \quad \alpha = 20^\circ, \; \text{Re} = 125k, \; 0.05 \leq k \leq 0.25, \; 0.1 \leq \lambda \leq 0.5 \) Parameter Sweeps
Figure B.21: $Re = 125,000$, $\alpha = 20^\circ$, $0.05 < k < 0.25$, $\lambda = 0.1$
Figure B.22: Re = 125,000, $\alpha = 20^\circ$, $0.05 < k < 0.25$, $\lambda = 0.2$
Figure B.23: Re = 125,000, $\alpha = 20^\circ$, $0.05 < k < 0.25$, $\lambda = 0.3$
Figure B.24: \( \text{Re} = 125,000, \ \alpha = 20^\circ, \ 0.05 < k < 0.25, \ \lambda = 0.4 \)
Figure B.25: $Re = 125,000, \alpha = 20^\circ, 0.05 < k < 0.25$, $\lambda = 0.5$
(a) Normalized Lift

(b) Normalized Pitching Moment

(c) Quasi-Steady Lift

(d) Quasi-Steady Pitching Moment

(e) Normalized Fluctuating Lift

Figure B.26: Re = 125,000, $\alpha = 20^\circ$, $k = 0.05$, $0.1 < \lambda < 0.5$
Figure B.27: Re = 125,000, α = 20°, k = 0.10, 0.1 < λ < 0.5
Figure B.28: Re = 125,000, $\alpha = 20^\circ$, $k = 0.15$, $0.1 < \lambda < 0.5$
Figure B.29: $Re = 125,000$, $\alpha = 20^\circ$, $k = 0.20$, $0.1 < \lambda < 0.5$
Figure B.30: $Re = 125,000$, $\alpha = 20^\circ$, $k = 0.25$, $0.1 < \lambda < 0.5$