ABSTRACT

CHRISTOPHER WILLIAM MCAVOY. Analytical and Experimental Approaches to Airfoil-Aircraft Design Integration. (Under the direction of Dr. Ashok Gopalarathnam.)

The aerodynamic characteristics of the wing airfoil are critical to achieving desired aircraft performance. However, even with all of the advances in airfoil and aircraft design, there remains little guidance on how to tailor an airfoil to suit a particular aircraft. Typically a trial-and-error approach is used to select the most-suitable airfoil. An airfoil thus selected is optimized for only a narrow range of flight conditions. Some form of geometry change is needed to adapt the airfoil for other flight conditions and it is desirable to automate this geometry change to avoid an increase in pilot workload. To make progress in these important aeronautical needs, the research described in this thesis is the result of seeking answers to two questions: (1) how does one efficiently tailor an airfoil to suit an aircraft? and (2) how can an airfoil be adapted for a wide range of flight conditions without increased pilot workload?

The first part of the thesis presents a two-pronged approach to tailoring an airfoil for an aircraft: (1) an approach in which aircraft performance simulations are used to study the effects of airfoil changes and to guide the airfoil design and (2) an analytical approach to determine expressions that provide guidance in sizing and locating the airfoil low-drag range. The analytical study shows that there is an ideal value for the lift coefficient for the lower corner of the airfoil low-drag range when the airfoil is tailored for aircraft level-flight maximum speed. Likewise there is an ideal value for the lift coefficient for the upper corner of the low-drag range when the airfoil is tailored for maximizing the aircraft range. These ideal locations are functions of the amount of laminar flow on the upper and lower surfaces of the airfoil and also depend on the geometry, drag, and power characteristics of the aircraft. Comparison of the results from the two approaches for a hypothetical general aviation aircraft are presented to validate the expressions derived in the analytical approach.

The second part of the thesis examines the use of a small trailing-edge flap, often referred to as a “cruise flap,” that can be used to extend the low-drag range
of a natural-laminar-flow airfoil. Automation of such a cruise flap is likely to result in improved aircraft performance over a large speed range without an increase in the pilot work load. An approach for the automation is presented here using two pressure-based schemes for determining the optimum flap angle for any given airfoil lift coefficient. The schemes use the pressure difference between two pressure sensors on the airfoil surface close to the leading edge. In each of the schemes, for a given lift coefficient this nondimensionalized pressure difference is brought to a predetermined target value by deflecting the flap. It is shown that the drag polar is then shifted to bracket the given lift coefficient. This non-dimensional pressure difference can, therefore, be used to determine and set the optimum flap angle for a specified lift coefficient. The two schemes differ in the method used for the nondimensionalization. The effectiveness of the two schemes are verified using computational and wind-tunnel results for two NASA laminar flow airfoils. To further validate the effectiveness of the two schemes in an automatic flap system, a closed-loop control system is developed and demonstrated for an airfoil in a wind tunnel. The control system uses a continuously-running Newton iteration to adjust the airfoil angle of attack and flap deflection. Finally, the aircraft performance-simulation approach developed in the first part of the thesis is used to analyze the potential aircraft performance benefits of an automatic cruise flap system while addressing trim drag considerations.
Analytical and Experimental Approaches to Airfoil-Aircraft Design Integration

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

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To my fiancé,

Kristen,

whose endless love and support continue to aid in the pursuit of all my goals.
BIOGRAPHY

Christopher William McAvoy was born in Johnson City, NY on April 4, 1978, to Bernard and Donna McAvoy. He was raised in Poughquag, NY with his six brothers and two sisters. He graduated from Arlington High School in 1996 and left New York with his older brother to study at North Carolina State University. He received a BS in Aerospace Engineering on May 20, 2000.

His graduate career began in the fall of 2000 when he was awarded a graduate teaching assistantship in the Mechanical and Aerospace Engineering Department at North Carolina State University. As part of his teaching assistantship responsibilities, he assisted with the senior aircraft design course for two semesters as well as an aircraft structures course for one semester. During the fall of that year he also met Dr. Ashok Gopalarathnam, whose interests in applied aerodynamics prompted Christopher to choose to work under his direction for the following two years. Under the guidance of Dr. Gopalarathnam, Christopher worked on a variety of research projects geared towards airfoil design, culminating in the publication of one article to appear in the *Journal of Aircraft*, one article in review for the same journal, and two AIAA conference presentations. After graduation, Christopher plans on studying intellectual property law at the Boston University School of Law.
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Nomenclature

\( AR \) wing aspect ratio

\( ac \) aerodynamic center

\( b \) wing span

\( c \) airfoil chord length

\( C_D \) aircraft or wing drag coefficient based on \( S_w \)

\( C_d \) airfoil drag coefficient based on the chord

\( CG \) aircraft center of gravity

\( C_L \) aircraft or wing lift coefficient based on \( S_w \)

\( C_l \) airfoil lift coefficient based on the chord

\( C_{l\text{ideal}} \) airfoil \( C_l \) at which the stagnation point is located at the leading edge of the airfoil in thin airfoil theory

\( C_M \) aircraft pitching moment coefficient about the center of gravity

\( C_m \) airfoil pitching-moment coefficient about the quarter-chord location

\( C_p \) pressure coefficient

\( \Delta C_p \) difference in leading-edge pressures nondimensionalized by dynamic pressure
\(\Delta C'_p\) difference in leading-edge pressures nondimensionalized by the absolute value of \((p_u-p_l)\)

\(\delta_e\) aircraft elevator angle

\(\delta_f\) trailing-edge flap angle, downward deflection is positive

\(e\) Oswald’s efficiency factor

\(T_l\) longitudinal distance from wing a.c. to tail a.c.

\(M\) Mach number

\(p\) pressure

\(P_{av}\) power available

\(P_{req}\) power required

\(q\) dynamic pressure

\(R\) aircraft range

\(R/C\) aircraft rate of climb

\(Re\) Reynolds number

\(S\) area

\(sfc\) specific fuel consumption

\(sm\) static margin

\(V_h\) horizontal tail volume ratio

\(V\) aircraft velocity

\(W\) aircraft weight
\(W_e\)  aircraft weight without fuel

\(W_f\)  aircraft weight with fuel

\(x\)  x-coordinate

\(y\)  y-coordinate

\(\alpha\)  angle of attack

\(\beta\)  \(dC_l/dC_d\) for the low-drag line

\(\eta_p\)  propeller efficiency

\(\rho\)  density

**Subscripts**

\(cg\)  aircraft center of gravity

\(f\)  fuselage and other components of aircraft except wing

\(i\)  induced

\(\infty\)  refers to freestream condition

\(l\)  location near the mid chord on the lower surface

\(ll\)  location near the leading edge on the lower surface

\(lu\)  location near the leading edge on the upper surface

\(max\)  maximum

\(min\)  minimum

\(0\)  refers to the stagnation-point condition
$p$ profile

$t$ horizontal tail

$u$ location near the mid chord on the upper surface

$w$ wing

**Superscripts**

0 denotes $C_d$ intercept of the low-drag line

$low$ lower corner of the airfoil low-drag range

$up$ upper corner of the airfoil low-drag range
Chapter 1

Introduction

1.1 Introduction to Airfoil Design and Optimization Techniques

In the past, aircraft designers would often choose airfoils for a particular application out of catalogs of previously designed and wind tunnel-tested airfoils. This type of design technique usually offers the advantage of providing extensive experimental results and thus few surprises in performance. However, with today’s sufficiently refined and proven analysis tools, an airfoil designer can be fairly confident that the predicted performance can be achieved in flight. Also with the many design variables and performance requirements specific to a particular aircraft, it is often hard to find an airfoil from a catalog that happens to coincide well with the goals of the desired wing section. As a result, it is now common to specifically design airfoils tailored to the particular application.

There are two basic methods for designing airfoils today. The first “direct” method stems from the approach of taking an airfoil from a catalog and iteratively adjusting the geometry until the desired performance is obtained. Unfortunately, it is not easy to predict how small changes in geometry will affect the overall airfoil performance. The “inverse” method allows the designer to prescribe a desired
velocity or pressure distribution and, using various conformal mapping techniques and numerical methods, obtain the appropriate geometry. Today’s inverse airfoil design programs\textsuperscript{1–4} even allow the designer to specify the velocity distributions at more than one condition, while also specifying such geometric constraints as maximum thickness ratio and pitching moment.

Despite all of these advances in airfoil design techniques, there is still surprisingly little research on how an airfoil should be tailored for a particular aircraft and mission. While modern inverse methods now make it possible to design for specific performance requirements, few tools exist that help guide the designer as to what airfoil performance characteristics are required to optimize the performance of a given aircraft. Such guidelines may also result in a more integrated approach to aircraft and airfoil design.

Even with the development of such airfoil design aids, a designer will still have to weigh trade-offs in performance characteristics. For example, designing an airfoil for maximum velocity may result in reduced $C_{l_{\text{max}}}$ or increased drag at lower velocities. Eventually the designer must compromise and select the airfoil that best meets the overall aircraft flight requirements. Such a compromise may not always be possible, however, and one may need to look into a variable-geometry airfoil that changes its shape for the particular flight condition. Currently this technique is most commonly used for high-lift conditions using devices such as flaps and slats. However, future advances in materials and actuation techniques may enable modification of the entire airfoil geometry in an attempt to optimize the airfoil performance throughout the entire flight envelope. Another less common example is a small trailing-edge flap, often referred to as a “cruise flap,” that can be used to extend the low-drag range of a natural-laminar-flow airfoil. One problem with all of these systems lies in the fact that the flap, slat, spoiler, or other geometry-modifying device must be actuated by the pilot at the appropriate
time in the flight of the aircraft. Because of this increase in pilot workload, geometry changes are usually limited to very specific flight conditions like take-off and landing.

### 1.2 Research Objectives

The overall objective of the research was to develop approaches for airfoil-aircraft design integration and for automation of variable geometry airfoils. The focus of the first section of the research was to develop a method for the integration of airfoil and aircraft design. For this method to be useful, it should take into account the aircraft geometry, power characteristics, and aerodynamic properties as well as trim effects and induced drag. The approach should help put trade-offs in airfoil characteristics in proper perspective with the effects on aircraft performance for the particular aircraft in question. An objective of the current research was also to explore the possibility of deriving simple equations that may guide the airfoil designer in the design of the best airfoil for a particular aircraft.

The second part of the research explored the use of what is commonly known as a “cruise flap” to effectively alter the geometry to maintain minimum airfoil profile drag over a wide range of aircraft speeds. Although these trailing-edge flaps have been used with success in several aircraft designs in the past, they are not widely in use primarily because of the increase in pilot workload required to constantly deflect the flap the correct amount for optimum performance. The objective of the current work is to develop the foundations of an automatic flap system that optimizes airfoil performance over a wide range of flight conditions. Such a system, or a similar system, may also have applications to segmented flaps, lift-distribution control, and wing morphing systems.
1.3 Outline of Thesis

The airfoil-aircraft design integration techniques are presented in Chapter 2. The chapter begins with a summary of past research in this area and then goes on to present a numerical method of simulating the performance of an aircraft in order to study the effect of changes to the airfoil characteristics. An analytical approach to airfoil-aircraft design integration is then presented and validated using the simulation approach. Chapter 3 presents the development of an automated trailing-edge flap for airfoil drag reduction over a wide range of lift-coefficients. This section includes a study of “cruise flaps” and how they work, followed by the development of pressure schemes that can be used to determine the optimum flap location during flight. The hypothesized schemes are then validated from numerical analyses and wind tunnel experiments on two well known NLF airfoils designed with cruise flaps. An algorithm for a simple logic-based closed-loop controller is then developed and tested in the wind tunnel. Finally, the benefits of such an automated flap system are analysed using the performance simulation approach developed in Chapter 2. Lastly, Chapter 4 presents the conclusions of the research as well as suggestions for continued work in these areas.
Chapter 2

Airfoil-Aircraft Design Integration

2.1 Background on Airfoil-Aircraft Matching

With advances in rapid, interactive inverse design methods for airfoils\textsuperscript{1–4} and robust analysis techniques,\textsuperscript{1,2,5} it is now possible to custom design a family of airfoils to suit a particular application. Two recent studies\textsuperscript{6,7} have demonstrated the suitability of an inverse approach for designing airfoils with systematic variations in the airfoil performance characteristics. While an aircraft designer greatly benefits by having such a family of airfoils tailored for the aircraft being designed, there is also a need for an approach to select the most suitable airfoil(s) from among the candidates available. Also of benefit would be an approach that can guide further airfoil refinement efforts to better suit the aircraft application. Even with all of the advances in airfoil and aircraft design, however, there remains surprisingly little guidance in the literature on how to tailor an airfoil to suit a particular aircraft.

For example, it is well-known that airfoils with larger extents of laminar flow and lower camber tend to be more suitable for high-speed performance. It is not known, however, as to whether or not there exists an optimum combination of
these airfoil characteristics. Currently, only a few references seem to exist that attempt to integrate airfoil and aircraft design. The paper by Maughmer and Somers describes the development of a figure of merit for airfoil/aircraft design integration to aid in the preliminary design efforts of an aircraft. The article by Kroo, although primarily devoted to the effects of trim drag and tail sizing, incorporates the effect of wing airfoil drag in the aircraft performance predictions.

In the current work, two approaches have been presented and compared for tailoring an airfoil to suit a given aircraft. The first approach involves designing a family of airfoils using the method described in Ref. 6, although airfoils from any other method or catalog could also be used. The predicted lift, drag and pitching moment characteristics for these airfoils are then used as inputs to a multiple lifting-surface vortex lattice code to compute the viscous and induced drag of a trimmed wing-tail configuration. These drag predictions, along with the engine power characteristics and estimates of the drag for the fuselage and other components, are then used in an aircraft performance prediction code. The outputs include the level-flight maximum speed and variations of performance parameters such as climb rate and range with flight speed for the aircraft. Thus the effect of airfoil characteristics on the different performance parameters of a particular airplane are obtained. This approach has the advantage that changes in the drag due to trim effects associated with the changes in the wing airfoil pitching moment are taken into consideration. To tailor an airfoil for a particular aircraft using this method, however, could require a considerable amount of trial and error.

The second approach involves derivation of analytical expressions for the ideal locations of the lower and upper corners of the low-drag range (drag bucket) of an airfoil to suit a particular aircraft. In this approach, the low-drag ranges of a family of airfoils with a prescribed amount of laminar flow have been described
by an equation for a straight line in the airfoil $C_d$-$C_l$ polar plot. The ideal $C_l$ values for the upper and lower corners of the low-drag range are then determined by computing the variation of different aircraft performance parameters along this straight line.

The following section presents the relevant information on a hypothetical general-aviation aircraft that has been used in the subsequent sections of this chapter. The next section describes the aircraft performance-simulation approach and its use in guiding the airfoil design for the wing. A section on the development of the analytical approach to size the airfoil low-drag range of the polar is then presented. Finally, the results of the analytical approach are validated by comparison with the results obtained from the performance-simulation approach with and without trim effects.

### 2.2 Aircraft Specifications

This section presents the relevant details of a hypothetical general-aviation aircraft used in the rest of this chapter. The aircraft considered is a conventional, aft-tail configuration with a constant-speed propeller driven by a piston engine. Figure 2.1 shows the planview of the wing and tail geometry.

Table 2.1 presents the relevant specifications for the aircraft. As shown, an equivalent parasite drag area ($C_{Df}S_f$) has been assumed for the fuselage and all the components of the airplane except the wing. The propeller efficiency has been assumed to be a constant. While this assumption may not be true for the entire speed range of the airplane even with a constant-speed propeller, the assumption makes the available power independent of the speed and is therefore useful for identifying the effects of airfoil changes. It must be mentioned, however, that both of the approaches discussed in this chapter can readily accommodate a non-
constant propeller efficiency. The static margin has been assumed to be 10% of the wing mean aerodynamic chord for the flight speeds where the wing airfoil operates in the low-drag region of the polar.

2.3 Aircraft Performance Simulation to Study Effect of Airfoil Changes

This section describes the approach of using aircraft performance simulations to study the effect of changes to the airfoil characteristics. For the purpose of describing the approach, two example airfoils have been designed using the methodology described in Ref. 6 to have the same $C_l$ for the lower corner of the low-drag range, but with different amounts of laminar flow. In this section, the performance of the aircraft for these two airfoil choices are compared and discussed in detail. The geometries and inviscid velocity distributions for the two airfoils are shown in Fig. 2.2. Airfoil X has laminar flow extending to 40% chord on the upper and lower surfaces, and airfoil Y has laminar flow to 60% chord. The predicted
Table 2.1: Assumed geometry, drag, and power characteristics for the hypothetical general aviation airplane.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Gross Weight (W)</td>
<td>7116.8 N (1600 lbf)</td>
</tr>
<tr>
<td>Wing area, reference area ($S_w$)</td>
<td>10 $m^2$ (107.6 sq.ft.)</td>
</tr>
<tr>
<td>Wing aspect ratio (AR)</td>
<td>10</td>
</tr>
<tr>
<td>Equivalent parasite drag area of airplane minus wing ($C_{Df}S_f$)</td>
<td>0.18 $m^2$ (2.24 sq.ft.)</td>
</tr>
<tr>
<td>Rated engine power ($P_{av}$)</td>
<td>74.63 kW (100 hp)</td>
</tr>
<tr>
<td>Specific fuel consumption ($sfc$)</td>
<td>10.7 N/s/W (0.5 lbf/h/hp)</td>
</tr>
<tr>
<td>Propeller efficiency ($\eta_p$)</td>
<td>85%, constant</td>
</tr>
<tr>
<td>Fuel volume</td>
<td>85.1 liters (22.5 U.S. gallons)</td>
</tr>
<tr>
<td>Tail area ($S_t$)</td>
<td>2.56 $m^2$ (27.56 sq.ft.)</td>
</tr>
<tr>
<td>Wing-to-tail moment arm ($\bar{l}_t$)</td>
<td>2.45 $m$ (8.04 ft.)</td>
</tr>
<tr>
<td>Tail volume ratio ($\bar{V}_h$)</td>
<td>0.63</td>
</tr>
<tr>
<td>Static margin (sm)</td>
<td>10% mac</td>
</tr>
<tr>
<td>Aircraft C.G. location</td>
<td>44% mac</td>
</tr>
</tbody>
</table>

performance for these two airfoils, obtained using XFOIL, are shown in Fig. 2.3 for $Re\sqrt{C_l}$ of 2 million and $M\sqrt{C_l}$ of 0.1. The use of constant values for $Re\sqrt{C_l}$ and $M\sqrt{C_l}$ for the analysis of these and all of the other airfoils in this study ensures that the changes in $Re$ and $M$ with $C_l$ due to change in the flight velocity are automatically taken into consideration. These relationships for the “reduced” $Re$ and $M$ can be derived from $L \approx W$ considerations for an airplane in steady rectilinear flight.

Comparing the drag polars for the two airfoils, it is seen that while airfoil $Y$ has a lower $C_{d_{\text{min}}}$ than airfoil $X$ resulting from greater extents of laminar flow, it also has a smaller $C_l$ range over which the low drag is achieved (i.e. smaller drag bucket). More specifically, although airfoil $Y$ has lower drag below a $C_l$ of 0.65, above this $C_l$ this airfoil has significantly greater $C_d$ than airfoil $X$. Examining this result, it is clear that while airfoil $Y$ will result in a higher cruise performance than $X$, it can also suffer from reduced climb performance. It is not clear, however, whether the benefit from the better cruise performance is worth the loss in climb
performance.

As a first step in understanding the effect on aircraft performance, it is useful to consider the total aircraft drag polars (aircraft $C_D$ vs. $C_L$) shown in Figure 2.4 for the two airfoils. The contribution from the $C_D$ for the aircraft minus the wing (labeled 1) is considered to be independent of the choice of the wing airfoil. Furthermore, because the two airfoils $X$ and $Y$ have nearly the same $C_m$, the trimmed aircraft $C_{D_i}$ (labeled 2) is also considered to be independent of the airfoil choice. This plot shows the well-known dominance of $C_{D_i}$ at high lift coefficients, and in doing so, puts in proper perspective the higher $C_d$ of airfoil $Y$ at these high values of $C_L$. The difference in $C_D$ due to change in the airfoil section at a $C_L$ of 1.0 is now approximately 4%, instead of the 20% difference in $C_d$ when only the section drag polars (shown in Fig. 2.3) are considered.

The aircraft $C_{D_i}$ variation was obtained from a trim analysis of the wing-tail combination shown in Fig. 2.1. This analysis was performed using Wings, a vortex-lattice code that can handle multiple lifting surfaces. The code uses a single chordwise row of lattices and has the capability to read in the XFOIL $\alpha$-
Figure 2.3: Performance of airfoils X and Y predicted using XFOIL.

$C_l$-$C_d$-$C_m$ polar output files for the airfoils used for the lifting surfaces. Thus, the analysis method can use the airfoil drag polars and pitching moment curves for several sections along the wing span in computing the drag of the wing-tail configuration. In the current analysis, the horizontal tail incidence is adjusted to trim the aircraft, so that $C_{M cg} = 0$. In other words, for each point on the $C_{D_i}$ curve in Fig. 2.4, the drag contributions associated with the trim considerations have been included.

Next, the total aircraft drag as a function of airspeed for the two airfoils is considered. Figure 2.5 shows the drag buildup for the two cases. As expected, it is seen that the induced drag is dominant at low speeds and is small at the high-speed end. At the high speeds, however, the fuselage parasite drag followed by the wing profile drag are the largest contributions to the overall drag. While these variations in induced, parasite, and profile drag are by no means new, the plot does put the drag changes due to airfoils in proper context. Another useful piece of information is the cross-over point for the two drag curves. From Fig. 2.3, it was seen that although airfoil Y has better performance below a $C_l$ of 0.65, it
Figure 2.4: Drag polar for the entire aircraft when using the two airfoils X and Y.

has higher $C_d$ for $C_l$ values from 0.65 to the $C_{l_{max}}$ of approximately 1.6. Because of the non-linear relationship between $C_L$ and the aircraft flight speed $V$, however, this $C_l$ range of 0.65 to 1.6 is squeezed into a small $V$ range from approximately 60 mph to 90 mph, whereas the $C_l$ range of 0.18 to 0.65 is magnified to a large $V$ range from 90 mph to 170 mph. This relationship can be better understood by examining the non-linear $C_L$ scale presented at the top of Fig. 2.5. As a consequence of this non-linear relationship between $C_L$ and $V$, there is a large velocity range over which better aircraft performance is achieved when airfoil Y is selected for the wing.

The power required for level flight is computed in the next step of the process to understand the effects of airfoil changes. Figure 2.6 shows the power required for level flight, $P_{req}$ as a function of the flight speed. Also shown is the full-throttle power-available curve based on the assumed engine and propeller characteristics.

Figure 2.7 shows the variations of aircraft rate of climb and range with airspeed for the two airfoil choices X and Y. The full-throttle rate of climb is computed
Figure 2.5: Drag variation for the entire aircraft when using the two airfoils X and Y.

Figure 2.6: Variation of the power required for level flight for the entire aircraft with the two airfoils, with contributions from the zero-lift drag of the aircraft minus wing (labeled 1), the induced drag (labeled 2), and the profile drag of the wing (labeled 3).
Figure 2.7: Variation of aircraft rate of climb and range with airspeed for the airfoil choices \( X \) and \( Y \).

from the difference between the \( \eta \mu P_{av} \) and the \( P_{req} \) curves in Fig. 2.6. It is seen that airfoil \( X \) has a better maximum \( R/C \) than airfoil \( Y \), because this flight condition occurs at low values of \( V \). The maximum level-flight speed at full throttle is the speed at which the \( R/C \) is zero. From the curves in Fig. 2.7, the increase in level-flight maximum speed is computed to be approximately 2 mph as a result of the longer runs of laminar flow when using airfoil \( Y \) instead of airfoil \( X \). This increase in \( V_{max} \) due to the increase in laminar flow is dependent on the specific aircraft characteristics. For example, the increase in \( V_{max} \) will be larger for an aircraft with a greater wing area and smaller fuselage equivalent parasite drag area.

Comparison of the range predictions shows that although airfoil \( X \) results in a greater maximum range, the best-range flight speed for airfoil \( X \) is less than that for airfoil \( Y \). Most general aviation aircraft, however, cruise at a speed that is greater than the speed for best range.\(^{14,15} \) With this consideration, there is a nearly 20-mile (approximately 5% at 125 mph) improvement in range for airfoil
Y at all flight speeds from 100 mph to 165 mph.

The performance-simulation approach described in this section can be a useful tool for putting the effects of changes in airfoil characteristics into proper perspective with the overall aircraft performance. In addition, this approach allows the inclusion of trim effects in computing the aircraft performance. While this type of performance simulation can be helpful for selecting the best airfoil out of a group of candidate airfoils, it does not give sufficient guidance on how an airfoil should be designed to optimize one or more performance parameters for a given aircraft. Tailoring an airfoil to suit an aircraft with this approach can be a tedious trial-and-error process. To improve the understanding of the airfoil-aircraft connection and to arrive at analytical methods to guide the airfoil design process, the following section presents the second approach that involves derivation of analytical expressions relating airfoil drag polar characteristics to aircraft performance.

2.4 Analytical Approach to Airfoil-Aircraft Matching

This section presents the second approach that involves deriving analytical expressions that can be used to tailor an airfoil to suit a particular aircraft. Two performance parameters are considered: level-flight maximum speed and maximum range. For this section, a family of 13 airfoils have been designed, all having 14% maximum thickness-to-chord ratio and the same amount of laminar flow (50%c on the upper and lower surfaces) but with different $C_l$ values for the lower corner of the low-drag range (drag bucket). This family has been designed using PROFOIL and the airfoil inverse design variables discussed in Ref. 6. More specifically, the airfoils in the family were designed by varying the design $C_l$ for the lower surface. As discussed in Ref. 6, this design $C_l$ in turn determines the
$C_l$ for the lower corner of the low-drag range, referred to in the rest of this chapter as $C_l^{\text{low}}$. The resulting airfoil shapes have systematic changes in the camber, although camber was not explicitly specified in the airfoil design.

The 13 airfoils have been labeled A–M, with airfoil A having the smallest $C_l^{\text{low}}$ of 0.1, and airfoil M having the largest $C_l^{\text{low}}$ of 0.83. In other words, airfoil A has the least camber and airfoil M has the largest camber. Figure 2.8 shows the geometries and inviscid velocity distributions for three of the 13 airfoils, B, H, and L. Figure 2.9 shows the predicted performance of the airfoils at $Re\sqrt{C_l} = 2$ million. It can be seen that the $C_d$ vs. $C_l$ variation for each airfoil is linear in the low-drag region and can be described by Eq. 2.1 for a straight line, referred to as the “low-drag line.”

$$C_d = C_d^0 + \frac{C_l}{\beta}$$

In this equation, $C_d^0$ is the $C_d$-intercept of the low-drag line on which the low-drag regions of all the polars lie, and $\beta$ is the the slope of the low-drag line $dC_l/dC_d$. In other words, the drag buckets of the different airfoils all lie on the low-drag line described by Eq. 2.1. With this description, it is possible to look for variations in the aircraft performance parameters as a function of $C_l$, by fixing the amount of laminar flow for a family of airfoils and therefore specifying $C_d^0$ and $\beta$. For the family of airfoils considered in Figs. 2.8 and 2.9, the $C_d^0$ is 0.0035 and $\beta$ is 467.

In this section, the approach is first adopted to determine the optimum value of $C_l^{\text{low}}$ for selecting the most suitable airfoil for an aircraft designed for maximizing the aircraft level-flight maximum speed $V_{\text{max}}$. Then, the variation of range with variation in the operating $C_l$ is studied for points on the airfoil drag polar that lie on the low-drag line — this study provides guidelines for tailoring an airfoil for maximizing the range of an aircraft. The aircraft for these studies is assumed to
Figure 2.8: Inviscid velocity distributions for airfoils $B$, $H$, and $L$.

Figure 2.9: Performance of airfoils $B$, $H$, and $L$ predicted using XFOIL.
have the specifications listed earlier in Table 2.1. It is realized that the changes in airfoil pitching moment coefficient for the family of airfoils under consideration will result in changes in the drag associated with trimming the aircraft. However, in order to derive simple expressions to help guide the tailoring of an airfoil to suit an aircraft, the trim effects on drag are ignored. To isolate the effects of the airfoil changes, the results in the following subsections pertain to the tail-off condition, and thus do not include the effects of the changes in the drag associated with trim. The consequences of ignoring the trim effects are discussed later in the section on the validation of the analytical approach.

### 2.4.1 Maximum Level Flight Speed

For tailoring an airfoil to suit an aircraft designed to have as high a $V_{\text{max}}$ as possible, it is necessary to select the airfoil based on the $C_{l}^{\text{low}}$ and the $C_{d\text{min}}$. From Fig. 2.9, it can be seen that for a family of airfoils with a prescribed amount of laminar flow, the $C_{d\text{min}}$ is directly related to the choice of $C_{l}^{\text{low}}$, as they both lie on the low-drag line. For a given laminar-flow extent, therefore, the selection of the airfoil for the $V_{\text{max}}$ condition needs to be made by choosing the optimum value of $C_{l}^{\text{low}}$, from among the points that form the low-drag line. Furthermore, because the power required $P_{\text{req}}$ equals $\eta P_{\text{av}}$, it is instructive to examine the variation of the $P_{\text{req}}$ for various points on the low-drag line. In this subsection, this variation of $P_{\text{req}}$ along the low-drag line is studied to arrive at an analytical approach to choose the value of the $C_{l}^{\text{low}}$ for the $V_{\text{max}}$ flight condition.

Assuming that the wing profile drag coefficient $C_{D\text{pw}}$ is equal to the airfoil $C_{d}$, the wing drag coefficient can be expressed as:

$$C_{D\text{pw}} = C_{d}^{0} + \frac{C_{l}}{\beta} \tag{2.2}$$
For the tail-off case, the aircraft $C_L$ is equal to the wing $C_{Lw}$, and $C_{Lw}$ is in turn taken as the average value of the airfoil $C_l$ over the entire wing span. As a result, $C_L = C_l$ for the expressions derived in the analytical approach. The aircraft drag coefficient $C_D$ can be obtained from Eq. 2.2 by adding the fuselage and induced drag contributions. The expression for this aircraft $C_D$ is presented in Eq. 2.3 in terms of $C_d^0$ and $\beta$.

$$C_D = C_d^0 + \frac{C_L}{\beta} + \frac{C_{Df}S_f}{S_w} + \frac{C_L^2}{\pi eAR} \quad (2.3)$$

The resulting aircraft drag and power required for level flight are presented in Eqs. 2.4 and 2.5.

$$D = \frac{1}{2} \rho V^2 (C_d^0 S_w + C_{Df} S_f) + \frac{W}{\beta} + \frac{2W^2}{\pi \delta^2 \rho V^2} \quad (2.4)$$

$$P_{req} = \frac{1}{2} \rho V^3 (C_d^0 S_w + C_{Df} S_f) + \frac{W}{\beta} V + \frac{2W^2}{\pi b^2 \rho V} \quad (2.5)$$

Knowing that at the $V_{max}$ flight condition, $P_{req} = \eta_p P_{av}$, it is possible to solve for the value of $V$ at which the $P_{req}$ in Eq. 2.5 equals $\eta_p P_{av}$. This value of $V$ will be the $V_{max}$ for an airplane that has a hypothetical airfoil drag polar that is the low-drag line itself. However, because the $C_l^{low}$ for the most suitable airfoil lies on this low-drag line, this value of $V$ is the $V_{max}$ for the airplane using the most suitable airfoil. From this $V_{max}$, the wing $C_{Lw}$ and hence the $C_l^{low}$ can be calculated by equating the lift and the aircraft weight.

The resulting expression is presented in Eq. 2.6 by setting $P_{req} = \eta_p P_{av}$ and $V = V_{max}$ in Eq. 2.5.

$$AV_{max}^3 + BV_{max} + C \frac{1}{V_{max}} - \eta_p P_{av} = 0 \quad (2.6)$$
where,

\[
A = \left( C_d^0 S_w + C_{Df} S_f \right) \frac{\rho}{2} \quad (2.7)
\]

\[
B = \left( \frac{W}{\beta} \right) \quad (2.8)
\]

\[
C = \left( \frac{2W^2}{\pi b^2 \epsilon \rho} \right) \quad (2.9)
\]

This value of \( V_{max} \) and the corresponding \( C_{l_{low}} \) represent the most suitable airfoil for the \( V_{max} \) flight condition from among the candidates that share the same value of \( C_d^0 \) and \( \beta \). If the extent of laminar flow is varied to generate several such families with different values of \( C_d^0 \), then it is possible to get the locus of the \( C_{l_{low}}-C_{d_{min}} \) points for these families. This locus defines the optimum placement for the lower corner of the low-drag range for any airfoil from among the families considered.

Figure 2.10 shows this locus for the optimum placement of \( C_{l_{low}} \) along with the polar for the most suitable airfoil \( D \) for the \( V_{max} \) flight condition from among the family of airfoils \( A-M \). It can be seen that if an airfoil is chosen with a \( C_{l_{low}} \) that is above this locus (such as the airfoil \( H \) or \( L \) in Fig. 2.9), then the airplane has potential for increase in \( V_{max} \) by selection of a wing airfoil with less camber. On the other hand, if an airfoil is chosen with the \( C_{l_{low}} \) that is lower than this locus line (such as the airfoil \( B \) in Fig. 2.9), then the airfoil has too low a camber for the aircraft, and the portion of the low-drag range below the locus line is not utilized except in a dive. The drawback is that the high-\( C_l \) performance is compromised and the airplane has an unnecessarily high stall speed.

It is to be noted that these analytical expressions for tailoring an airfoil for the \( V_{max} \) condition can also be used to instead tailor the airfoil for the velocity corresponding to level-flight speed at a cruise power setting. In this case, the \( P_{av} \)
would have to be replaced by the cruise power setting $P_{\text{cruise}}$. The ideal airfoil for this cruise-power condition would have a $C_{l}^{\text{low}}$ that is higher than the $C_{l}^{\text{low}}$ obtained for the $V_{\text{max}}$ condition.

### 2.4.2 Range

For tailoring an airfoil to maximize the range of an aircraft, it is necessary to make the airfoil selection based on the $C_l$ for the upper corner of the low-drag range, or $C_{l}^{\text{up}}$. Knowing that this upper corner also lies on the low-drag line defined by $C_{d}^0$ and $\beta$ for the family of airfoils with a prescribed amount of laminar flow, it is useful to examine the variation of aircraft range with $C_l$ for points on the low-drag line.

The well-known Bréguet range equation, shown in Eq. 2.10 for constant values of $\eta_p$ and $sfc$, clearly illustrates that range for an aircraft is maximized at the aircraft lift coefficient corresponding to the aircraft maximum (L/D) or minimum $C_{D}/C_{L}$ condition.
Thus the ideal value of $C_l^{up}$ for an airfoil tailored to maximize the aircraft range can be determined by finding the $C_l$ on the low-drag line that results in minimum aircraft $C_D/C_L$. From the expression for the aircraft $C_D$ as a function of $C_L$ along the low-drag line in Eq. 2.3, the expression for $C_D/C_L$ can be determined. With the earlier assumption of $C_L = C_l$, this expression is presented in Eq. 2.11.

$$\frac{C_D}{C_L} = \frac{C_d^0}{C_l} + \frac{1}{\beta} + \frac{C_{Df}S_f}{C_lS_w} + \frac{C_l}{\pi e AR}$$

(2.11)

The resulting ideal $C_l$ for maximizing the aircraft range can be obtained by taking the derivative of Eq. 2.11 with respect to $C_l$ and setting it to zero. The resulting expression for the $C_l$ is shown in Eq. 2.12.

$$C_l = \sqrt{\frac{\pi e AR}{C_l^0 + \frac{C_{Df}S_f}{S_w}}}$$

(2.12)

For the hypothetical aircraft used here, this equation provides a lift coefficient of approximately 0.78. As done earlier, if the value of $C_d^0$ is varied to generate several families of airfoils with different specifications for the extent of laminar flow, then the locus of points that form the ideal values of the $C_l^{up}$ for each family can be constructed. This locus is shown in Fig. 2.11 along with the polar for the most suitable airfoil $G$ from among the family of 13 airfoils $A$–$M$ that have 50% $c$ laminar flow on the upper and lower surfaces and a maximum thickness of 14% $c$. For the aircraft under consideration, the choice of an airfoil with the $C_l^{up}$ that is below this locus (such as the airfoil $B$ in Fig. 2.9) will mean that the range can be improved by the use of an airfoil with higher camber. On the other hand, selection of an airfoil with the $C_l^{up}$ that is above this locus (such as the airfoil $L$ in Fig. 2.9)
Figure 2.11: Ideal locations for $C_l^{up}$ corresponding to maximum range along with the drag polar for the most suitable airfoil $G$ and the low-drag line.

will not improve the range beyond that obtained with the use of the most suitable airfoil $G$, but will only result in a decrease in the low-$C_l$ performance.

It must be mentioned that the assumption of constant $\eta_p$ and $sfc$ may not be true in general. This assumption was made to illustrate the analytical approach and to obtain a simple closed-form expression for the ideal $C_l^{up}$. In a more detailed analysis, it may still be possible to use this approach if the variations in $\eta_p$ and $sfc$ could be approximated as analytical functions that could then be incorporated in Eqs. 2.10–2.12 to obtain a more accurate estimate of the ideal airfoil $C_l^{up}$.

2.4.3 Trim Considerations

The need to operate an aircraft in a trimmed condition results in three sources of drag: (1) induced drag of the wing-tail system, (2) parasite drag of the tail, and (3) increased wing profile drag due to higher $C_{Lw}$ required with tail download. The performance-simulation approach described in the earlier section for studying the effects of airfoil changes includes all of these three sources of drag. Owing to
the difficulty in integrating these sources of drag in the analytical expressions, the analytical approach in this section did not consider these sources of drag and were derived for the tail-off condition.

In the following section, the results from the analytical study are compared with the results from the performance-simulation approach, both with and without trim considerations.

2.5 Validation

It can be seen from the analytical expressions derived in the previous section that there are distinct ideal locations to place the upper and lower corners of the low-drag range when tailoring an airfoil for level-flight maximum speed and maximum range respectively. These ideal locations are compared with the predictions from the performance-simulation approach for the family of 13 airfoils A–M, three of which were previously shown in Fig. 2.9. These airfoils all have the same amount of laminar flow, but have different locations for the low-drag range.

Figure 2.12 shows the predicted variations from the performance-simulation approach for $V_{\text{max}}$ with change in the airfoil $C_{l_{\text{low}}}$. The predictions are shown for both the trimmed and the tail-off cases. The figure also shows the ideal value of $C_{l_{\text{low}}}$ of 0.2 from the analytical approach for an airfoil tailored for the $V_{\text{max}}$ flight condition. It is seen that as the $C_{l_{\text{low}}}$ is decreased from 0.6 to 0.2 by the use of airfoils with lower camber, there is an increase in $V_{\text{max}}$ for both the trimmed and tail-off cases. When the $C_{l_{\text{low}}}$ is decreased below the ideal value of 0.2, however, the further increase in $V_{\text{max}}$ is significantly reduced. This distinct slope change in the $V_{\text{max}}$ vs. $C_{l_{\text{low}}}$ curve validates the analytical expression derived for the $V_{\text{max}}$ flight condition, and demonstrates that airfoil $D$ is the ideal airfoil for the $V_{\text{max}}$ flight condition.
Figure 2.12: $V_{\text{max}}$ as a function of $C_{l}^{\text{low}}$ for the family of airfoils A–M.

It must be mentioned that if the low-drag regions for the airfoils A–M had lined up perfectly with the assumed low-drag line, then the $V_{\text{max}}$ vs. $C_{l}^{\text{low}}$ curve for the tail-off case in Fig. 2.12 would have shown an increase in $V_{\text{max}}$ as the $C_{l}^{\text{low}}$ was decreased from 0.6 to the ideal value of 0.2, and an exactly zero increase in $V_{\text{max}}$ for any decrease in $C_{l}^{\text{low}}$ below this ideal value. However, the results from the performance-simulation approach do not show the zero increase in $V_{\text{max}}$ for $C_{l}^{\text{low}} < 0.2$, but only a distinct reduction in the $V_{\text{max}}$ increase. This deviation from the expectations is because of the small deviations in the low-drag regions of the airfoils from the assumed low-drag line. In particular, the airfoils A–C have slightly lower $C_{d}$ than that for the low-drag line for $C_{l}$ ranging from 0.1 to 0.2.

To more clearly illustrate the effect of the airfoil $C_{l}^{\text{low}}$ on the aircraft $V_{\text{max}}$, Fig. 2.13 shows the rate-of-climb curves from the performance-simulation approach for the airfoils C–F, for which the $C_{l}^{\text{low}}$ values lie in the vicinity of the ideal value of 0.2. These curves have been plotted for the full-power condition. For each of the four airfoils, the point corresponding to the lower corner of the low-drag range
Figure 2.13: Rate of climb curves for airfoils $C$–$F$ illustrating the suitability of airfoil $D$ for the $V_{max}$ condition.

is also marked. It is seen that the lower corner for the ideal airfoil $D$ occurs almost exactly at the velocity that results in zero rate of climb, i.e. at $V_{max}$. For airfoils $E$ and $F$, however, the lower corners are at velocities less than the $V_{max}$ achieved. For this reason, these airfoils have too high a camber. Airfoil $C$ on the other hand, has the lower corner at a velocity that corresponds to a dive condition. A portion of the low-drag region for this airfoil cannot be used in level flight. The airfoil $C$, therefore, has too low a camber for the airplane under consideration. Thus, the results in Fig. 2.13 further demonstrate that airfoil $D$ is the ideal airfoil for the $V_{max}$ condition.

Figure 2.14 shows the variation in maximum range as a function of $C_l^{up}$, as predicted by the performance-simulation approach for both the trimmed and the tail-off cases. The predictions for both the cases show that there is a limit for the $C_l$ for the upper corner beyond which there is no improvement in the range. This limit is predicted to be approximately 0.81 for both the cases, and this limit agrees
Figure 2.14: Maximum range as a function of $C_{l_{up}}$ for the family of airfoils $A–M$.

well with the value of 0.78 predicted by the analytical expression in Eq. 2.12.

2.6 Discussion of Results

With modern inverse airfoil design techniques, it has been possible to design airfoils that have very specific lift, drag and moment characteristics. However, little guidance was available for tailoring an airfoil to suit a particular application. Toward this objective, two approaches have been presented in this chapter: (1) an aircraft performance-simulation approach and (2) an analytical approach. In the performance-simulation approach, the changes in airfoil characteristics are used as inputs to an aircraft performance code to predict the resulting changes to the aircraft performance. In the analytical approach, the low-drag portion of the airfoil drag polar is represented by an equation. This equation is used to search for ideal locations of the lower and upper corners of the low-drag range for aircraft designed for level-flight maximum speed and maximum range.
The analytical study shows that there is a distinct ideal value for the lift coefficient of the lower corner of the airfoil low-drag range when the airfoil is tailored for the level-flight maximum speed condition of an aircraft. Likewise, there is a distinct ideal value for the lift coefficient for the upper corner of the airfoil low-drag range when the airfoil is optimized for the maximum range condition of an aircraft. These ideal values for the upper and lower corners of the low-drag range are dependent not only on airfoil parameters such as the extents of laminar flow on the upper and lower surfaces, but also on the aircraft parameters such as the drag characteristics of the fuselage and other components, the available power from the engine, and the propeller efficiency. By using the performance-simulation approach for a family of airfoils with the same amount of laminar flow but with systematic changes in the camber, the analytical expressions for the ideal locations have been validated.

The results of the study and the analytical expressions derived in this chapter provide important criteria for positioning the low-drag range of the airfoil drag polar. In particular, these expressions provide guidelines for tailoring an airfoil to suit an aircraft and can be used to avoid the tedium associated with exploring numerous airfoils in an effort to find the best airfoil for a given aircraft. Although this chapter considers only propeller-driven piston-engine aircraft, the approaches can be extended for use in jet-powered airplanes. Additionally, although the analytical expressions have been developed for airfoils with well-defined low-drag ranges that can be defined by a straight line, the approach can be extended to other types of airfoils (such as low Reynolds number airfoils) by description of the trends in variation of the low-drag range using an appropriate analytical expression. These approaches are likely to be useful to both airfoil and aircraft designers for tailoring an airfoil to suit a particular aircraft. In addition, the expressions developed in this chapter are suitable for use in an “inner loop” within an aircraft
sizing or multidisciplinary optimization study for selecting the most appropriate airfoil from a family of airfoils.
Chapter 3

Automated Trailing-Edge Flap for Airfoil Drag Reduction Over a Wide Lift Range

The methods and expressions developed in Chapter 2 can be valuable in guiding the designer in the selection or design of the most appropriate airfoil for a given aircraft. However, a designer is still required to choose between improved performance in one area of the flight envelope at the expense of decreased performance in another. For example, Fig. 2.3 shows an example of an airfoil designed to have lower profile drag with increasing amounts of laminar flow. However, one of the consequences is that the width of the low-drag region becomes smaller. As a result the designer has to decide whether the increased high-speed performance is worth the decreased range, maximum rate-of-climb, and overall low-speed performance, shown in Fig. 2.7. This problem of narrow drag buckets becomes even more obvious is for NLF airfoils used at high Reynolds numbers. This problem can be solved if the shape of the airfoil could be changed during flight based on the flight conditions in order to optimize the performance for the flight condition under consideration. With rapid advancements in smart materials, it is conceivable that the entire airfoil shape could be altered on future aircraft. Currently, however, this is done using various forms of flaps. In particular, an airfoil designed for use
with what is commonly known as a “cruise flap” can have low drag by maximizing the extents of laminar flow over a wide range of lift coefficients.

3.1 Introduction to Cruise Flaps

It is well known that the deflection of a small trailing-edge flap, often referred to as a “cruise flap,” shifts the low-drag region of the airfoil drag polar. Positive, or downward deflection of the trailing-edge flap causes the laminar drag bucket to shift to higher lift coefficients, while negative flap deflections causes a shift to lower lift coefficients. The desired shift in the drag bucket is accomplished when, for a given lift coefficient, the flap is deflected to the appropriate angle at which the leading-edge stagnation point is brought to the optimal position resulting in favorable pressure gradients on both the upper and lower surfaces of the airfoil. Consequently, extended laminar flow and low airfoil profile drag are achieved over a wide range of lift coefficients, and hence wide range of aircraft speeds.

Originally conceived by Pfenninger around 1947, cruise flaps have been used to good advantage in the design of many natural-laminar-flow (NLF) airfoils and these flaps have been widely used on high-performance sailplanes for several years. Cruise flaps also enable the use of airfoils with extended amounts of laminar flow. With increasing extents of laminar flow, the decrease in airfoil drag is accompanied by a reduction in the width of the low-drag range (or drag bucket). When cruise flaps are used, the drag bucket is effectively widened and the high performance of the airplane is not restricted to a small range of flight speeds. In spite of the advantages, cruise flaps have not gained popularity for routine use on general aviation and other commercial aircraft. An important reason is believed to be the increase in pilot workload that accompanies the traditional
installation of cruise flaps. Current use of cruise flaps requires the pilot to continuously monitor the airspeed and adjust the flap to the optimum location using some form of a look-up table. Another problem with the current cruise flaps is that, unless they are to be extensive, the look-up tables connecting airspeed and flap angle are valid for only one flight condition; i.e. one airplane weight, load factor, and center-of-gravity (CG) position. An automation of the cruise flap with a closed-loop control system would provide the benefits without any increase in the pilot workload. In addition, such an automated system would enable the use of cruise flaps on autonomous and uninhabited aerial vehicles (UAVs), where low drag is desirable over a wide range of flight conditions from loiter to dash.

A key ingredient in an automated cruise flap system is a scheme to determine the optimum flap deflection for any given lift coefficient. It would be desirable that the scheme be independent of changes in weight, load factor, and CG position. For this reason, it is preferable to avoid a system that sets the flap angle based on the aircraft lift coefficient computed from the measured airspeed. The objective of the current work has been to develop a simple approach for determining the optimum flap deflection angle for a given airfoil lift coefficient. In the first part of this chapter, a concept is presented where the difference between two pressure measurements on the airfoil surface is nondimensionalized and used to determine the best flap deflection angle. The concept is presented and demonstrated first using a computational analysis of a generic laminar flow airfoil, along with two possible schemes for the implementation. The schemes are then also shown to be valid for the NASA NLF(1)-0215F and the NASA NLF(1)-0414F airfoils using both computational and experimental results. These two particular NASA airfoils have been chosen because they have both been designed and wind-tunnel tested with cruise flaps and extensive surface pressure data is available in the NASA reports. In the second part of this chapter, a closed-loop control system
is developed and demonstrated in a wind tunnel. The control system is shown to be effective at maintaining the airfoil $C_l$ and $\Delta C_p$ by adjusting the airfoil $\alpha$ and flap angle. Finally, in the last part of this chapter, a section is presented on the integration of automated cruise flaps on aircraft, including an analysis of potential aircraft performance benefits using a hypothetical long-range UAV with a customized flapped NLF airfoil as an example.

### 3.2 Flow Sensing Approaches for Flap Automation

In this section, a generic natural laminar flow (NLF) airfoil with a trailing-edge flap is analyzed using XFOIL and two flow sensing approaches for flap automation are presented. The first approach involves sensing of the stagnation point, and the second approach involves the use of pressure sensing at the leading edge. The drag polars and pressure distributions from XFOIL are examined to first study the variation of the leading-edge stagnation point location with airfoil lift coefficient and flap angle. Two pressure-based schemes are then proposed for determining the best flap angle for a given lift coefficient. In each scheme, the non-dimensional difference between the static pressures measured at two locations on the airfoil surface is required to be held at a constant value by adjusting the flap angle. The two schemes differ in the pressure value used for the nondimensionalization.

The example NLF airfoil used in the following discussion is analyzed at a constant “reduced Reynolds number” (constant $Re \sqrt{C_l}$) to simulate operation on an aircraft wing by accounting for changes in Reynolds number due to changes in flight velocity. The reduced Reynolds number chosen is typical of several light general-aviation aircraft.
Figure 3.1: Geometry of the example NLF airfoil with a 20% chord flap.

3.2.1 Stagnation-Point Location Sensing

Stagnation-Point Relationship

In order to understand how a cruise flap extends the low drag region of the airfoil drag polar, the role of the leading-edge stagnation point is first examined. Figure 3.1 shows the geometry of the NLF airfoil with a 20%-chord cruise flap. The predicted performance of the airfoil from XFOIL analyses is shown in Fig. 3.2 for flap deflections of -10, -5, 0, 5, and 10 deg flap angles. It can be readily seen that deflection of the flap from -10 to 10 deg results in low drag over a wide \( C_l \) range from 0.1 to 1.2. On the other hand, if no flap deflection is used, the low airfoil drag is restricted to a much smaller \( C_l \) range from 0.3 to 0.8. This benefit is the reason for integration of cruise flaps in the design of many NLF airfoils including the the NASA NLF(1)-0215F,\(^{22}\) NASA NLF(1)-0414F,\(^{19,20,23}\) Wortmann,\(^{24}\) and Drela\(^{25}\) sailplane airfoils. More recently, aerodynamic methods have also been developed for inverse design of airfoils with cruise flaps.\(^{28}\)

A key to automating the setting of the cruise flap can be deduced by examining the variation of the leading-edge stagnation point with \( C_l \) for different flap settings. This plot is shown in Figure 3.3 along with the points on each graph corresponding to the corners of the low-drag range for each flap deflection. The figure shows that irrespective of the flap setting, there is a small desirable region for the leading-edge stagnation point, shown in Fig. 3.4, that always results in low airfoil drag by promoting laminar flow on both surfaces of the airfoil.

It is interesting to note that this observation can be deduced even from thin airfoil theory. From thin airfoil theory, it is possible to compute the \( C_{l\text{ideal}} \) corre-
Flap = 10 deg
Flap =  5 deg
Flap =  0 deg
Flap = −5 deg
Flap =−10 deg

Figure 3.2: Predicted performance of the NLF airfoil with flap deflections of -10, -5, 0, 5, and 10 degrees.

...corresponding to the ideal angle of attack at which the stagnation point is exactly at the leading edge of the thin airfoil (see Appendix B). At this condition, there are no singularities (or suction peaks) on either the upper or lower surface. This condition corresponds closely to the $C_l$ for the middle of the low-drag range. When a thin airfoil is analyzed with a 20%-chord trailing-edge flap, it can be shown that the $C_{l\text{ideal}}$ increases by 0.28 for every 10 degrees of positive flap deflection. This result is close to the predicted shift in the low-drag range seen from the XFOIL analyses in Fig. 3.2.

**Hot-Film Arrays**

Thus if the stagnation point could be detected in real time, a system could be created that deflects the cruise flap to maintain the stagnation point in the optimum region. While it is possible to detect the location of the stagnation point using surface hot-film arrays, this thesis focuses on use of simpler surface pressure
measurements for indirectly detecting the effects of the stagnation point location.

3.2.2 Surface Pressure Sensing

In an attempt to find a simpler pressure-based method of determining the optimum flap deflection, the pressure distributions for the airfoil with zero flap angle are examined in Fig. 3.5 at three conditions: (1) when the airfoil $C_l$ is within the low-drag range (inside the drag bucket), (2) when the airfoil $C_l$ is less than the $C_l$ for the lower corner of the drag bucket (below drag bucket), and (3) when the airfoil $C_l$ is greater than the $C_l$ for the upper corner of the drag bucket (above drag bucket). It can be seen that when the airfoil is operating within the drag bucket, the pressure gradients are favorable on both the upper and lower surfaces for the first 50% of the chord. On the other hand, when the airfoil is operating outside the drag bucket, there is a leading-edge suction peak either on the upper surface (when the airfoil is above the drag bucket) or on the lower surface (when the
Figure 3.4: Geometry of the airfoil with inset showing desired range for stagnation point location.

Figure 3.5: Airfoil pressure distributions at three operating conditions.

airfoil is below the drag bucket). These observations provide clues for exploring the use of surface pressure measurements in automatically setting the flap angle for the airfoil to always operate within the drag bucket.

Figure 3.6 shows the pressure distributions of the example NLF airfoil, when operating within the drag bucket, at flap deflections of -10, -5, 0, 5, and 10 degrees. From the figure, it is seen that independent of the flap angle, the pressure distributions in the vicinity of the leading edge are almost identical when the airfoil is operating in the low-drag range. This result is not surprising considering
that the stagnation-point location is nearly independent of the flap angle, as seen from Figs. 3.3 and 3.4, when the airfoil is operating within the low-drag range. Examining the plots in Figs. 3.5 and 3.6, it was hypothesized that the difference in pressure coefficients measured from two pressure orifices close to the leading edge could provide a measure of whether the airfoil is operating within the drag bucket irrespective of the flap angle.

**Scheme 1**

To verify this hypothesis, the XFOIL results for the NLF airfoil were used to determine the pressure-coefficient difference between the 5% location on the upper surface and the 5% location on the lower surface. Figure 3.7 shows how this $\Delta C_p$ varies with $C_l$ for the different flap deflections, where $\Delta C_p$ refers to the $C_p$ difference between the 5% location on the upper surface and the 5% location on the lower surface. It is clearly seen that for any flap angle, there is a region of values of $\Delta C_p$ that, when achieved, ensures operation of the airfoil in the low

![Pressure distributions for different flap deflections with the airfoil operating within the drag bucket.](image)
Figure 3.7: Variation of $\Delta C_p$ with $C_l$ for pressure orifices at the 5\%$c$ locations on the upper and lower surfaces.

drag region of the polar. Figure 3.8 shows a similar variation in $\Delta C_p$ with $C_l$ for pressure measurements made at the 2\%-chord locations on the upper and lower surfaces. Comparison of the results in Figs. 3.7 and 3.8 shows that while the method is not highly sensitive to the exact location of the pressure taps, it is desirable to measure the pressures close to the airfoil leading edge to obtain a larger variation in $\Delta C_p$ for a given change in airfoil $C_l$.

Although there is a possibility that the use of pressure orifices in the vicinity of the leading edge may cause premature transition, there is evidence that a small-diameter pressure orifice in a favorable pressure gradient is less likely to cause premature transition than one in an adverse pressure gradient.\textsuperscript{33} Because the pressure orifices located in the vicinity of the leading edge will be in regions of favorable pressure gradient when the airfoil operates within the low-drag range, it is believed that these pressure orifices are unlikely to cause premature transition for the operating conditions of interest.

It must be remembered, however, that the difference in the measured surface
pressure \((p_{lu} - p_{ll})\) needs to be nondimensionalized to determine the \(\Delta C_p\). As seen from Eq. 3.1, this nondimensionalization can be readily done by dividing the pressure difference by the dynamic pressure measured by the aircraft pitot-static system.

\[
\Delta C_p = \frac{p_{lu} - p_{ll}}{q_{\infty}} = \frac{p_{lu} - p_{ll}}{p_0 - p_{\infty}} = C_{p_{lu}} - C_{p_{ll}} \quad (3.1)
\]

While this scheme for nondimensionalization is sufficient, it has some drawbacks: (1) it is dependent on a pitot-static pressure measurement that may be physically located far from the airfoil resulting in pressure lag errors, (2) the measured static pressure usually has position errors that vary with the aircraft flight condition and are often dependent on whether or not the flaps are deployed, (3) the dynamic pressure measured by a pitot-static system that is located far away from the airfoil may not accurately reflect that experienced by the airfoil if the airplane has significant angular velocity. Moreover a scheme that can determine the
optimum flap location without having to depend on an independent measurement of the dynamic pressure may have the advantage that different airfoil sections on the lifting surfaces of an aircraft may be able to independently control the respective flap angles using a self-contained pressure-based sensing system. For these reasons, it is desirable to arrive at an alternate scheme for automating the cruise flap setting.

**Scheme 2**

To avoid the drawbacks associated with the use of aircraft dynamic pressure for the nondimensionalization, several alternate schemes were investigated. An alternate scheme is presented that involves a substitute for the dynamic pressure used in the nondimensionalization in the first scheme described earlier in this section. This second scheme involves the use of the absolute value of the difference between pressure measurements made near the mid-chord locations on the upper and lower surfaces of the airfoil. In other words, an alternate nondimensionalization is used to define a $\Delta C'_p$ where the denominator is the absolute value of $(p_u - p_l)$, as shown in Eq. 3.2. The drawback with this second scheme is that the $\Delta C'_p$ tends towards infinity when $p_u$ nearly equals $p_l$. This condition may occur on an aircraft wing when using negative flap angles and when the airfoil lift coefficient is negative or close to zero. For most normal flight conditions and typical airfoils, this drawback is not a source for concern.

$$\Delta C'_p = \frac{p_{tu} - p_{tl}}{|p_u - p_l|} = \frac{C'_{pu} - C'_{pl}}{|C_{pu} - C_{pl}|}$$  \hspace{1cm} (3.2)

In this paper, the locations for the upper- and lower-surface pressure measurements $p_u$ and $p_l$ are chosen to be at the downstream ends of the design extents of laminar flow on the upper and lower surfaces for the airfoil under consideration.
Figure 3.9: Variation of $\Delta C'_p$ with $C_l$ for pressure orifices at the 2% $c$ locations on the upper and lower surfaces.

This choice of locations for $p_u$ and $p_l$ has been made to reduce the possibility of premature transition due to the pressure orifices and resulting loss in laminar flow. For the generic NLF airfoil used in this section, these locations correspond to 50% chord on both the upper and lower surfaces.

Figure 3.9 shows the variation in $\Delta C'_p$ with $C_l$ for the different flap deflections. As seen from this figure, although the curves are less linear than those seen in Fig. 3.8, there is a distinct range of $\Delta C'_p$ values for which the airfoil is always in the drag bucket regardless of the flap angle.

In the following sections, these two schemes — the first scheme in which dynamic pressure is used for the nondimensionalization and the second scheme in which $|p_u - p_l|$ is used for the nondimensionalization — are both validated using both computations and wind-tunnel data for the NASA NLF(1)-0215 and the NASA NLF(1)-0414 airfoils. The geometries and inviscid $C_p$ distributions for these two airfoils are shown in Fig. 3.10. These two particular airfoils have
been chosen because they have both been designed and wind-tunnel tested with cruise flaps. Moreover, extensive surface pressure data is available in the NASA reports.\textsuperscript{19,22} In all the cases in the rest of this paper, the leading-edge pressure differences ($p_{lu} - p_{ll}$) are the differences in pressure at the upper-surface $2\% c$ and the lower-surface $2\% c$ locations.

### 3.3 Numerical and Experimental Validation of Pressure Schemes

#### 3.3.1 NLF(1)-0215F Airfoil

The NASA NLF(1)-0215F airfoil was tested in the wind tunnel\textsuperscript{22} with a $0.25c$ trailing-edge flap, with the hinge located at $x/c = 0.75$ and $y/c = 0.0328$. Measured lift, drag, and moment data along with pressure distributions from surface-pressure orifices are documented in Ref. 22 for flap angles of -10, 0, and 10 degrees at a Reynolds number of six million. The verification of the two pressure-based
schemes for the NLF(1)-0215F airfoil has been presented in this section using computational results from XFOIL analyses and using the wind tunnel results of Ref. 22.

Figures 3.11(a) and 3.11(b) show the predicted performance for this airfoil from XFOIL analyses and the wind tunnel results respectively for a Reynolds number of 6 million. As expected, the deflection of the flap results in an extended $C_l$ range over which low-drag is achieved. The predicted variation in $\Delta C_p$, from the first scheme where the dynamic pressure is used for the nondimensionalization, is plotted against $C_l$ for flap angles -10 deg to 10 deg in Fig. 3.12(a) from XFOIL.
analyses and in Fig. 3.12(b) from the experiment data. When using the second scheme where the nondimensionalization is done using the absolute value of the mid-chord pressure difference ($p_u - p_l$), the predicted variation in $\Delta C'_p$ is shown in Fig. 3.13(a) from XFOIL analyses and in Fig. 3.13(b) from the wind-tunnel results. For this airfoil, the $p_u$ and $p_l$ correspond to surface pressures at the 0.4$c$ location on the upper surface and the 0.6$c$ location on the lower surface respectively.

Although the results from the XFOIL predictions and the NASA experiments do not agree perfectly with each other, it is clear that the trends compare well. More importantly, the computational and experimental results both demonstrate...
that either of the two schemes could be used to set the optimum flap angle for a given $C_l$ by achieving a target value either for the $\Delta C_p$ or for the $\Delta C'_p$.

### 3.3.2 NLF(1)-0414F Airfoil

The NLF(1)-0414F airfoil was tested in the wind tunnel with a 0.125$c$ trailing-edge flap. Measured lift, drag, and moment data along with pressure distributions from surface-pressure orifices are documented in Ref. 19 for flap angles of -10, -5, 0, 5, and 10 degrees at a Reynolds number of 10 million. As was done in the previous section, the two pressure-based schemes are verified using both computational
analyses using XFOIL and the results from the NASA wind-tunnel tests.\textsuperscript{19}

The predicted and experimental drag polars for this airfoil are shown in Figs. 3.14(a) and 3.14(b). When compared to the two airfoils in the earlier sections, the NLF(1)-0414F is seen to have a considerably narrower low-drag range. This small width of the low-drag range can be attributed to the larger extents of laminar flow as well as the higher Reynolds number. As a result of the smaller low-drag range, an automated cruise flap system will be of significant benefit when used with this airfoil. For the same reason, this airfoil is a good test case for the two pressure-based schemes in this paper. Figures 3.15(a) and 3.15(b) show the effectiveness...
of the first scheme in plots of the $\Delta C_p$ from computational results and from the wind-tunnel data. The effectiveness of the second scheme is shown in Figs. 3.16(a) and 3.16(b), where the $\Delta C'_p$ from computational results and wind-tunnel tests are plotted for the different flap angles. For this airfoil, the $p_u$ and $p_l$ correspond to surface pressures at the 0.6$c$ location on the upper surface and the 0.6$c$ location on the lower surface respectively. From the Figs. 3.15 and 3.16, it is seen that both the computational results and the experimental data verify the effectiveness of the two pressure-based schemes for flap-angle automation.
Figure 3.16: Predicted and experimental results for the $\Delta C_p'$ for the NLF(1)-0414F airfoil from: a) XFOIL, b) wind tunnel experiments.
3.4 Demonstration of an Automatic Trailing-Edge Flap in a Wind Tunnel

In the preceding sections, it was shown that irrespective of the airfoil $C_l$, if the nondimensionalized pressure difference $\Delta C_p$ or $\Delta C'_p$ was maintained at a predetermined target value by deflecting the flap, then the resulting airfoil $C_d$ would be close to the minimum profile drag possible. In other words, by deflecting the flap to the optimum angle, the drag bucket is shifted to bracket the desired $C_l$.

Having validated the effectiveness of the pressure-based schemes in determining the optimum flap angle, an investigation was carried out to develop and test a simple closed-loop system in a low-speed wind tunnel at the North Carolina State University. The focus of the investigation was to develop and test a simple closed-loop algorithm to measure the nondimensional pressure difference on an airfoil and use that in near real time to deflect the cruise flap to the optimum angle so that the pressure difference is made to coincide with a target prescription. The following sections describe the experimental setup and present the characteristics of the airfoil used. The closed-loop control algorithm is then described, followed by system response plots that demonstrate the effectiveness of the closed-loop control system.

3.4.1 Experimental Setup

NCSU Subsonic Wind Tunnel

The NCSU Subsonic Wind Tunnel shown in Fig. 3.17 is a closed-circuit tunnel with a 0.81$m$ high, 1.14$m$ wide, and 1.17$m$ long test section. Upstream of the test section, just before the contraction section, is a settling chamber consisting of an aluminum honeycomb screen followed by two stainless steel anti-turbulence
screen. Turbulence levels have been determined to be less than 0.33%. The contraction section is composed of four sides of identical curvature and the test section walls diverge slightly to allow for boundary layer growth. Two of the sides of the test section are made of Plexiglas and hinged at the top for easy access and visibility. A breather located downstream of the test section ventilates the tunnel to room pressure.

The tunnel fan is equipped with variable pitch blades allowing the velocity in the test section to be continuously varied up to a maximum speed of approximately 40m/s at a dynamic pressure of 720Pa. This maximum tunnel speed results in a maximum chord Reynolds number of approximately 0.8 million for a 12 inch chord airfoil model. It is recognized that this Reynolds number is less than the chord Reynolds numbers of typical NLF airfoils; however, the purpose of the wind-tunnel tests was to demonstrate the closed-loop system where the cruise flap is continuously adjusted to maintain the $\Delta C_p$ close to a desired target value. For this purpose the experimental setup is well suited even though the chord $Re$ is low.
Figure 3.18 shows the geometry and inviscid velocity distribution of the low-speed airfoil designed for the wind-tunnel model. The airfoil has a 12-inch chord and has been designed for a Reynolds number of approximately 0.6 million. The XFOIL results for the airfoil with a 20%-chord trailing-edge flap are shown in Fig. 3.19. Unlike the higher-$Re$ NLF airfoils examined in the earlier sections, the polars for this airfoil do not have distinct drag buckets. This behavior is typical of airfoils designed for this $Re$ regime, where the transition ramp is designed to extend over a large chordwise extent to mitigate the adverse effect of laminar separation bubbles.

The pressure-tapped airfoil model was fabricated at NC State by the author. The fiberglass skins were made using plug-and-mold type construction techniques. The following paragraphs briefly describe the construction techniques. The plug is constructed by first cutting out the airfoil core shape from a block of Polystyrene “blue foam.” This is accomplished by running heated music wire around Formica templates attached to the ends of the blue-foam block. The foam
is then sheeted with balsa wood and balsa leading and trailing edges are glued on to the plug. Finally, the plug is sheeted with a thin layer of fiberglass and then primed and sanded to a Class A finish. Figure 3.20 shows a photograph of the cross section of the plug.

Once the plug is finished, the molds are made by laying fiberglass around the plug. Parting planes are set up at the leading and trailing edges of the airfoil. Figure 3.21 shows the plug and mold after one half of the mold has been removed. Fiberglass is then laid up in the molds to make the “skins” of the airfoil and wood ribs were fabricated to provide increased stiffness as well as to provide mounts for flap hinges, a servo motor, and a 1 inch diameter aluminum tube designed to run through the quarter-chord axis of the airfoil (Fig. 3.22).

A commercial Scanivalve system available in the wind tunnel lab was used to measure the surface pressures. The Scanivalve system consists of two transducers and is capable of measuring the pressures from up to 96 pressure taps. A total of
Figure 3.20: Cross section of the plug showing the different layers in the foam sandwich construction technique.

74 pressure taps were used to obtain detailed pressure distributions for this airfoil; 38 taps on the upper surface, 36 on the lower surface, and a total of 11 of these on the flap. Figure 3.23 shows the locations of the pressure taps around the airfoil and shows how they are unequally spaced so as to obtain more detailed pressure data in areas of greater interest and curvature.

The pressure taps were staggered in the spanwise direction in order to limit the possibility of an orifice affecting the measurements of taps downstream. Earlier research has shown that a line of orifices placed directly behind one another is more likely to cause premature transition of the flow. As a result the taps were staggered at a 20-degree angle to the freestream direction. Twenty degrees was chosen as a more than sufficient angle based on research that has shown that for a similar size orifice located in flow of a similar Reynolds number, the area of most influence downstream of the orifice is limited to a turbulent wedge extending outwards from the transitioning element with a half-angle of roughly 10 degrees.

Bulged, 0.040-inch outside diameter, 0.55-inch long, steel tubulations were used for the pressure taps in the airfoil model. Each orifice was drilled perpendicular to the local airfoil surface angle and has a diameter of about 0.030 inches. Most of the Vinyl pressure tubing is housed in the aluminum tube that runs
through the quarter-chord of model in order to protect it as the airfoil pitches (Fig. 3.24). These tubes were then connected to the Scanivalve system.

**Flap Actuation System**

A JR-8101 Ultra Precision servo was installed in the airfoil model to actuate the 20% chord trailing-edge flap (see Fig. 3.22). An RS-232 serial servo controller made by BasicX was used to communicate with the servo from through a standard serial port (see Fig. 3.28). The flap was then calibrated and a LabView program was written to actuate the flap using the lab PC.

**Airfoil Pitch Control**

Figure 3.25 shows how the airfoil model was placed vertically in the test section with minimal gap between the model and the tunnel floor and ceiling. The existing sting motor (Fig. 3.26) located below the tunnel floor was used to control the airfoil
angle of attack. In order to use the sting in this manner, an aluminum fixture was made to fit over the top of the sting mechanism and to attach to the 1-inch diameter tubing that runs through the quarter chord of the airfoil model.

A bearing shown in Fig. 3.27 was attached to the top of the tunnel to limit vibrations of the model and to ensure smooth angle-of-attack changes to the airfoil. With this setup, an existing LabView program that controls the beta angle of the sting could now be used to control the angle of attack of an airfoil model mounted vertically in the wind tunnel. The sting motor was controlled by the lab computer through a standard nine-pin serial port.

**Pressure Measurement System**

The surface pressure measurements from the pressure taps on the model along with the tunnel dynamic pressure were used to compute the airfoil $C_l$, the $\Delta C_p$, and the $\Delta C'_p$. For the initial set of measurements in the following section, the
Scanivalve system was used. For the demonstration of the closed-loop control system described later, a miniature Electronic Pressure Scanner (ESP) module was used (Fig. 3.28). The switch to the ESP module was made to increase the speed of the pressure measurements in order for them to be used in a control system. This ESP module is equipped with an array of 32 silicon piezoresistive transducers. No drag data was obtained in the current investigation because of the limited wind-tunnel time available and also because drag was not essential for the development of a closed-loop control system that uses pressure measurements to set the trailing-edge flap.

3.4.2 Measured Aerodynamic Characteristics

The Scanivalve system was used to acquire surface pressures for angles of attack from -6 to 10 deg and flap angles of -10, -5, 0, 5, and 10 deg. The surface pressures were reduced to pressure coefficients using the tunnel dynamic pressure and were then integrated to compute the airfoil $C_l$. The complete data set can be found in Appendix A. Figure 3.29 shows a representative $C_p$ distribution from the measurements and a comparison with the $C_p$ distribution as predicted by XFOIL. The agreement is seen to be good.

The measured pressures were also used to study the variations of the nondi-
Figure 3.24: Photograph showing the plastic tubing running from the tubulations through the aluminum tubing. The tubing in the flap runs up in the flap and crosses over into the main airfoil section at the top.

Dimensionalized pressure differences from the two schemes. Figure 3.30 shows how $\Delta C_p$ from the first scheme varies with $C_l$ for this airfoil. The taps used in the calculation of $\Delta C_p$ for $p_u$ and $p_l$ are located at 4.5%$c$ on the upper surface and 5%$c$ on the lower surface, respectively. Figure 3.31 shows the variation of $\Delta C'_p$ from the second scheme with the airfoil $C_l$. The locations for the measurements of $p_u$ and $p_l$ were chosen to be at 50%$c$ on the upper surface and 51%$c$ on the lower surface respectively. Because drag measurements were not made, the lower and upper corners of the low-drag range are not marked in these figures. The trends in these curves, however, are similar to those observed earlier for the two NLF airfoils.

Finally, it should also be mentioned that tap numbers 48 and 52, located at 13% and 25% of the chord, respectively, on the lower surface, were omitted because it is believed that the Vinyl pressure tubings are either partially blocked or torn.
Additionally, examination of the pressure distribution shows the effects of blockage in the test section at the higher angles of attack. No corrections were made to the data, however, due to the fact that the primary focus of the effort was to develop a closed-loop control system and not to measure the absolute performance of the airfoil.

### 3.4.3 Control Algorithm

The objective of the closed-loop control system is to maintain the nondimensional pressure difference close to the target value prescribed. This objective is achieved by deflecting the flap. It is important to note, however, that the objective has to be achieved at a specified airfoil $C_l$, and not at a specified $\alpha$. In order to satisfy the prescriptions on both the $\Delta C'_p$ (or $\Delta C''_p$ for scheme 2) and the airfoil $C_l$, the control system must adjust the values of the airfoil $\alpha$ and the flap angle $\delta_f$. 
Figure 3.26: Photograph showing how the sting was used to pitch the airfoil.
Figure 3.27: Photograph showing the bearing at the top of the tunnel and how the pressure tubing for the 74 pressure taps were run through the bearing.
Figure 3.28: Photograph showing some of the data acquisition and control equipment used for the wind tunnel experiments.
This problem for the airfoil in a wind tunnel has similarity with the cruise-flap control problem for an aircraft. For a UAV, for example, the problem may be posed as having to maintain a desired airspeed (V) as measured by the pitot-static system and a desired $\Delta C_p$. In the aircraft case, the variables adjusted by the control system would likely be the elevator angle $\delta_e$ and the cruise flap angle $\delta_f$.

As a starting point for illustrating the control algorithm used in the current approach, the linearized equations for the $C_l$ and $\Delta C_p$ in terms of the two variables $\alpha$ and $\delta_f$ are considered in Eqs. 3.3 and 3.4.

\begin{equation}
C_l = (C_l)_{\alpha=0, \delta_f=0} + \frac{\partial C_l}{\partial \alpha} \alpha + \frac{\partial C_l}{\partial \delta_f} \delta_f \tag{3.3}
\end{equation}

\begin{equation}
\Delta C_p = (\Delta C_p)_{\alpha=0, \delta_f=0} + \frac{\partial \Delta C_p}{\partial \alpha} \alpha + \frac{\partial \Delta C_p}{\partial \delta_f} \delta_f \tag{3.4}
\end{equation}

The algorithm developed in the current approach for the control uses a continuously-running Newton iteration that monitors the differences between the desired and
Figure 3.30: Experimental results for the $\Delta C_p$ for the wind tunnel model airfoil.

Figure 3.31: Experimental results for the $\Delta C'_p$ for the wind tunnel model.

the current values of the $C_l$ and $\Delta C_p$. These differences form the residual vector for the Newton iteration. Because of the nearly constant values of the partial derivatives in Eqs. 3.3 and 3.4, the gradients for the Newton iteration can be pre-computed from the initial database of pressures for the airfoil. In other words, the Jacobian matrix remains unchanged. Using the Jacobian and the residual vector, the two-dimensional Newton iteration shown in Eq. 3.5 predicts the changes required to the $\alpha$ and the $\delta_f$ in order to satisfy the prescriptions. The Jacobian matrix, residual vector and the correction vector are shown in Eqs. 3.6, 3.7, and 3.8.
\[ J \cdot \delta x = -F \]  
\[ (3.5) \]

\[ J = \begin{pmatrix} \frac{\partial C_l}{\partial x} & \frac{\partial C_l}{\partial \delta_f} \\ \frac{\partial \Delta C_p}{\partial x} & \frac{\partial \Delta C_p}{\partial \delta_f} \end{pmatrix} \]  
\[ (3.6) \]

\[ F = \begin{cases} C_l_{\text{current}} - C_l_{\text{desired}} \\ \Delta C_p_{\text{current}} - \Delta C_p_{\text{desired}} \end{cases} \]  
\[ (3.7) \]

\[ \delta x = \begin{cases} \alpha_{\text{next}} - \alpha_{\text{current}} \\ \delta f_{\text{next}} - \delta f_{\text{current}} \end{cases} \]  
\[ (3.8) \]

The flow chart for each time step in the continuous iteration is shown in Fig. 3.32. It is to be noted that this iteration procedure can be used with either the pressure scheme 1 or the pressure scheme 2. For the pressure scheme 2, the \( \Delta C_p \) in the preceding equations is replaced with \( \Delta C'_p \).

**Program**

For the wind tunnel tests, this algorithm was programmed into a LabView code. LabView is not ideal for use in a closed-loop system if speed is an issue. For the wind tunnel tests, the speed of the response was not critical to the demonstration of pressure-based flap control, and the ease with which LabView can be programmed to interact with various data acquisition equipment and the user friendly graphical interfaces, made it an attractive software choice for the experiment.
Adjust $\alpha$ and $\delta_f$

Predict the change in $\alpha$ and $\delta_f$ by solving Eq. 3.5

Determine $F$ from Eq. 3.7

Measure pressures and compute $\Delta C_p$ and $C_l$

Read user inputs for desired $C_l$ and $\Delta C_p$

Adjust $\alpha$ and $\delta_f$

Figure 3.32: Flowchart illustrating one time step in the continuous Newton iteration process.

Figure 3.33 displays the graphical user interface (GUI) created for the system. The top window on the screen contains the knobs that adjust the desired $C_l$ and $\Delta C_p$, and a switch to turn the automatic cruise flap system on and off. These inputs can be changed at any time during the operation of the system. The operator can also use an optional window averaging technique and adjust the size of the window to reduce the sensitivity of the system to small fluctuations in pressure measurements. This was not a problem in the wind tunnel where the pressures remained very steady, and thus none of the results shown here use window averaging. On this GUI there are also several gages that show the current $\Delta C_p$, $\alpha$, and $\delta_f$. The graphs on the right plot the actual and desired $C_l$ and $\Delta C_p$ values in real time to monitor the performance of the automatic system. The other two windows are used for reference showing the current pressure distribution of the airfoil and the free stream conditions in the test section.
3.4.4 Effectiveness of the Control Algorithm

The algorithm for the closed-loop control described in the previous section was programmed in LabView. An ESP module was used to obtain the surface pressures from the airfoil model, and a USB cable was used for the data transmission between the ESP module and the lab PC. The switch from the Scanivalve system to the ESP module was made to increase the speed of the data acquisition. However, the limitation on the availability of only the USB cable for data transmission made the data acquisition slower than desirable. With the setup used, one time
To demonstrate the effectiveness of the control algorithm, the system was tested with the airfoil model in the wind tunnel. While the system was in operation, the values for the desired $C_l$ and $\Delta C_p$ were altered frequently. It was seen that the system was highly effective in predicting the required changes to $\alpha$ and $\delta_f$ in order to satisfy the specifications on the $C_l$ and $\Delta C_p$. In most cases, the system was successful in matching the specifications in just one time step. Figure 3.34 shows a sample window of operation of the closed-loop system for the pressure scheme 1. Figure 3.34(a) shows the plots of desired and actual values of $C_l$ and $\Delta C_p$ for each time step in the iteration. The effectiveness of the closed-loop con-
trol system is clearly seen. Figure 3.34(b) shows plots of the changes in $\alpha$ and $\delta_f$ required to achieve the variations in Fig. 3.34(a).

The labels on the plot in Fig. 3.34(a) represent points in time when the inputs to the system were altered. The closed-loop system was first switched on at point A. In a couple of time steps, the specifications to the $C_l$ and $\Delta C_p$ were achieved. At points B, C, and D, changes were made to the desired $C_l$ while keeping the desired $\Delta C_p$ unaltered. At points E and F, changes were made to the desired $\Delta C_p$ without altering the desired $C_l$. At point G, changes were made to desired values of both $C_l$ and $\Delta C_p$. In all these cases, the system was able to easily track the changes in the desired $C_l$ and $\Delta C_p$, demonstrating the effectiveness of the control system. In the vicinity of point H, the specification for the desired $C_l$ was changed frequently. In this case, an oscillatory behavior was seen for the $\Delta C_p$. The amplitude of the oscillation was small enough (approximately 0.5) that the airfoil is likely to continue operating within the low-drag range.

Figure 3.35 shows a sample window of operation of the closed-loop system for the pressure scheme 2. While the tracking performance is seen to be acceptable, it is below the performance seen in Fig. 3.34 for scheme 1. This performance degradation for scheme 2 is attributed to the nonlinearity in the curves seen in Fig. 3.31. As with the figure for scheme 1, labels in Fig. 3.31(a) represent instances when changes were made to the specifications. After the system is switched on at point A, changes were made to the desired $C_l$ at points B and C while maintaining the desired $\Delta C_p'$ unaltered. At point E, the desired $\Delta C_p'$ was changed without changing the desired $C_l$. At point D, both the specifications were changed. In all cases, the system was able to track the changes effectively.

It should also be addressed that in the wind tunnel, the pressure readings were always steady. This may not be true for all conditions encountered by an aircraft. On an aircraft, it may be necessary to use a windowed average of the
Figure 3.35: System responses of the automatic cruise flap system in the wind tunnel using pressure scheme 2.

data acquired during the past several time steps instead of the current values in Eq. 3.7. Details of such a technique need further development in future research efforts.

Overall, the closed-loop system performed very well and it was shown that the airfoil configuration (i.e. $\delta_f$ and $\alpha$) could be accurately and automatically controlled to achieve desired changes in $C_l$ while maintaining a constant $\Delta C_p$ with such an automation approach and an appropriately designed airfoil, low profile drag can be achieved over a wide range of lift coefficients.
3.5 Applications

3.5.1 Implementation of the Schemes on Aircraft

The development of the simple pressure-based schemes described in this chapter is believed to be an important step in the automation of cruise flaps for achieving improved performance over a large speed range for subsonic aircraft. While the verification of the schemes have been presented for two-dimensional flow, the concept can be implemented on an aircraft wing. For application on an aircraft wing, a section of the wing should be chosen for which the section $C_l$ is close to the wing $C_L$. On this wing section, the difference between the pressures measured at the two leading-edge pressure sensors can be nondimensionalized and used to drive the cruise flap to the correct angle.

Integration of cruise flaps on wings with control surfaces and high-lift flaps may require special consideration. For many aircraft concepts, it may be possible to design the flaps and control surfaces on the wing to double as cruise flaps. Also, cruise flaps could be integrated in the design of Fowler flaps by either incorporating a smaller-chord cruise flap at the trailing edge of a larger chord Fowler flap as was done on the Cirrus VK30 experimental aircraft\cite{37} or by allowing contour change of the Fowler flaps using modern form-variable structures.\cite{38}

The pressure-based sensing approach described in this paper can also be used for stall warning, in a manner similar to that described in Ref. 39. The use for stall warning is possible because for a given flap angle, the $\Delta C_p$ and $\Delta C'_p$ curves have one-to-one relationships with the airfoil $C_l$ up to the onset of stall. Thus, if the $\Delta C_p$ or $\Delta C'_p$ is known along with the flap angle, the section $C_l$ can be determined. Additionally, the schemes described in this paper can be used with multiple, segmented trailing-edge flaps along the wing span, with the $\Delta C_p$ or $\Delta C'_p$ for each segment of the wing controlling the respective flap section. This
concept allows for the tailoring of not only the local section profile drag, but also the spanwise lift and \( C_l \) distributions. The concept, thus, paves the way for a section-\( C_l \) sensing method that can be used for in-flight minimization of the induced drag, tailoring of the spanwise stall behavior on the wing, and spanwise load redistribution during maneuvering conditions.

### 3.5.2 Aircraft Performance Benefit Study

Interesting observations can also be made by comparing the airfoil drag polars, \( C_l \) and \( C_m \) curves for an airfoil without cruise flaps and for one with an automated cruise flap system. To illustrate these differences, a flapped NLF airfoil has been designed\(^40\) for application on a hypothetical uninhabited aerial vehicle (UAV). The geometry and inviscid \( C_p \) distribution for this airfoil is shown in Fig. 3.36. The airfoil has been designed to support approximately 70\% laminar flow on both the upper and lower surfaces. A 0.15\( c \) cruise flap is used to extend the width of the low-drag range. In order to postpone the positive flap deflection at which the flow separates on the flap upper surface, the upper surface of the airfoil in the vicinity of the hinge has been specially contoured resulting in a depression in the inviscid \( C_p \) distribution at 0.85\( c \) on the upper surface at the zero-flap condition. Such a flap-region contour tailoring is frequently done on flapped sailplane airfoils such as the FX 78-K-161/20.\(^26\)

To examine the behavior of this airfoil with an automated cruise flap, it is assumed that a target value of the \( C'_p \) of -1.5 will be achieved by deflecting the flap to the required angle. Figure 3.37 shows the variation of \( \Delta C'_p \) obtained from XFOIL analyses plotted against \( C_l \) for different flap deflections, with the upper and lower corners of the low drag region marked for each case. Also shown is the target value of -1.5, which if maintained, will result in the airfoil operating within the low-drag range for the range of flap deflections considered. The minimum and
maximum angles for the cruise flap have been set to -5 and +10 deg respectively. At flap angles much larger than +10 deg, the flow separates on the upper surface of the flap resulting in an increase in the drag. Flap angles less than -5 deg do not provide benefits for the application considered.

The resulting drag polar of the airfoil with the automated cruise flap system is compared in Fig. 3.38 to the drag polar of the airfoil without a cruise flap. Unlike the results for the NASA NLF airfoils shown earlier, the polars for this airfoil have been plotted at a constant value of reduced Reynolds number $Re\sqrt{Cl}$ of 2 million. The use of a constant reduced $Re$ ensures that the changes in $Re$ with $Cl$ due to changes in the flight velocity are automatically taken into consideration. From Fig. 3.38, while the large increase in the $Cl$ range for low drag is clear, it is seen that for the automated-flap case, the change in $\alpha$ required for an increase in $Cl$ is a small negative value. This differs from the usual lift-curve slope of approximately $2\pi$ per radian for the airfoil without the cruise flap. Additionally, the airfoil $C_m$ about the quarter chord has a large variation with $\alpha$ when an automated cruise flap is used instead of a nearly constant $C_m$ for an airfoil without a cruise flap. The nearly constant $\alpha$ for a wide $Cl$ range for the automated-flap case may prove quite
beneficial because the fuselage and nacelles can be optimized for a small variation in the aircraft $\alpha$. The unusual variations in the $C_l$ and $C_m$ for the automated-flap case, however, do need to be analyzed in the context of how they may affect trim drag, and thus overall aircraft performance. In particular, it is not clear if the large changes in $C_m$ and any resulting trim drag will outweigh the profile drag reduction due to the cruise flaps. To address this issue, the characteristics of the NLF airfoil with and without the cruise flap have been used as inputs to the aircraft performance-simulation method described earlier in Chapter 2 and in Ref. 16 for a hypothetical UAV.

The hypothetical UAV for this illustration is assumed to have the characteristics listed in Table 3.1, and is assumed to have the wing-tail planform shown in Fig. 3.39. Because the primary purpose of the performance simulation is to compare the airfoil with and without a cruise flap, the propeller efficiency and specific fuel consumption for the powerplant have been assumed as constants for simplicity. The method, however, can readily accommodate non-constant values...
for these parameters. As described in Ref. 16, the method first computes a drag buildup in which the total aircraft drag polar is obtained by summing the airfoil $C_d$ shown in Fig. 3.38, a calculated aircraft $C_{Di}$, and an assumed constant fuselage parasite drag. The aircraft $C_{Di}$ variation was obtained from a trim analysis of the wing-tail combination shown in Fig. 3.39. This analysis was performed using *Wings*, a vortex-lattice code that can handle multiple lifting surfaces and has the capability to read XFOIL $\alpha$, $C_l$, $C_d$, and $C_m$ polar output files for the airfoils used for the lifting surfaces. In the current analysis, the horizontal tail incidence is adjusted to trim the aircraft, so that $C_{M_{cg}} = 0$. In other words, the
Table 3.1: Assumed geometry, drag, and power characteristics for the hypothetical UAV.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Gross Weight (W)</td>
<td>10230.9 N (2300 lbf)</td>
</tr>
<tr>
<td>Reference area ($S_{ref}$)</td>
<td>10.5 $m^2$ (113 sq.ft.)</td>
</tr>
<tr>
<td>Wing aspect ratio (AR)</td>
<td>21.4</td>
</tr>
<tr>
<td>Equivalent parasite drag area</td>
<td>0.12 $m^2$ (1.29 sq.ft.)</td>
</tr>
<tr>
<td>of airplane minus wing ($C_{Df}S_f$)</td>
<td></td>
</tr>
<tr>
<td>Rated engine power ($P_{av}$)</td>
<td>74.63 kW (100 hp)</td>
</tr>
<tr>
<td>Specific fuel consumption ($sfc$)</td>
<td>10.7 N/s/W (0.5 lbf/h/hp)</td>
</tr>
<tr>
<td>Propeller efficiency ($\eta_p$)</td>
<td>85%, constant</td>
</tr>
<tr>
<td>Fuel volume</td>
<td>380 liters (100 U.S. gallons)</td>
</tr>
<tr>
<td>Tail area ($S_t$)</td>
<td>3.06 $m^2$ (32.94 sq.ft.)</td>
</tr>
<tr>
<td>Tail moment arm ($l_t$)</td>
<td>2.475 m (8.12 ft.)</td>
</tr>
<tr>
<td>Tail volume ratio ($V_h$)</td>
<td>1.03</td>
</tr>
<tr>
<td>static margin (sm)</td>
<td>10 % mac</td>
</tr>
</tbody>
</table>

drag contributions associated with the trim considerations have been included.

Figure 3.40 shows the variations of aircraft rate of climb and range with airspeed for the NLF airfoil both with zero flap deflection and with an automatic cruise flap. Comparison of the curves shows that despite the trim effects resulting from the unusual airfoil pitching moment variations, the aircraft with the automated cruise flap performs considerably better over the entire velocity range except for the region of zero flap deflection where the performance is the same. The level-flight maximum speed at full throttle is the speed at which the rate of climb is zero. In this example, it is seen that the level flight maximum speed is increased by approximately 6.5 mph (3.7%) as the result of using the automatic cruise flap. Also, the maximum range is increased by 425 miles (7.4%) and the range at $V_{max}$ is increased by nearly 300 miles (11.5%). The maximum rate of climb remains approximately the same, but occurs at a higher velocity.

Although the aircraft performance benefits have been illustrated, the unusual variations in the $C_l$ and $C_m$ for the automated-flap case in Fig. 3.38 do need to
be revisited in the context of how they may affect the aircraft static and dynamic stability and control characteristics. For example, because of the change in the airfoil $C_m$ due to flap deflection and its effect on the aircraft trim, it may be necessary that the control of the cruise flap be coupled with the control of the elevator angle to maintain trimmed flight at a desired speed.

### 3.6 Discussion of Results

A trailing-edge cruise flap system, when automated, has the potential to result in low airfoil drag over a wide $C_l$ range. An important step toward the automation is a method to determine the correct flap angle, that will result in extended laminar flow on both the upper and lower surfaces of the airfoil at a given $C_l$. To arrive at such a method, this research draws on the well-known fact that a cruise flap results in a wide low-drag range by bringing the stagnation point to the small desirable region close to the leading edge. This small desirable region is quantified, and
flow sensing approaches for detecting the stagnation point are then discussed. An approach has been presented where the nondimensional pressure difference between two points on the airfoil surface close to the leading edge can be used to set the cruise flap at the correct angle. It is shown that, independent of the flap angle, there is a band of values for this nondimensional pressure difference that when achieved, will result in the airfoil operating within the low-drag range.

Two schemes have been presented for the nondimensionalization: (1) in which the dynamic pressure is used and (2) in which the pressure difference between the upper-surface and lower-surface locations near the mid-chord region of the airfoil is used for the nondimensionalization. The advantage of the first scheme is that only two pressure sensors are required, whereas the second scheme requires four pressure sensors. The first approach, however, has the disadvantage of being dependent on the dynamic pressure measured by the aircraft pitot-static system, whereas the second scheme results in a system that relies on measurements made entirely on the airfoil section under consideration. Computational and wind tunnel results for two NASA laminar flow airfoils are used to verify the effectiveness of the two schemes.

The study also presents a control algorithm for closed-loop control of a cruise flap. A continuously-running Newton iteration is used for this purpose. The control system was implemented for an airfoil in a wind tunnel, and was shown to be able to successfully track changes to the prescribed values of $C_l$ and $\Delta C_p$ by adjusting the airfoil $\alpha$ and $\delta_f$.

Issues associated with the implementation of cruise flaps on aircraft are then discussed. To address the issue of whether trim drag due to changes in airfoil $C_m$ with flap deflection outweigh the profile drag benefits of cruise flaps, the effects on the performance of an example UAV have been presented. For the example used, the results clearly demonstrate the benefits of an automated cruise flap.
The results presented in this chapter also show that the schemes can be used to determine the operating $C_l$ of an airfoil section in flight by knowing the value of the non-dimensional pressure difference. This capability results in the potential for further benefits such as stall warning and the use of segmented flaps along the span for tailoring the lift and $C_l$ distributions along the span, that in turn allows for the control of induced drag, the tailoring of spanwise stall behavior, and redistribution of the spanwise loads on a wing.
Chapter 4

Concluding Remarks

4.1 Summary of Results

The current research examines several areas of airfoil performance optimization. The design phase of the airfoil was first examined in order to develop tools to aid in the integration of the airfoil and aircraft design. The second part of the thesis looks into a method of optimizing the airfoil performance in flight using an automatic “camber-changing” flap.

The first part of the thesis presents a two-pronged approach to airfoil-aircraft design integration. First, an aircraft performance simulation approach is used to predict the effects of airfoil changes on various aircraft performance parameters. The performance-simulation approach uses a drag buildup for the aircraft that incorporates the airfoil lift, drag, and pitching moment characteristics obtained from XFOIL with a multiple lifting surface vortex-lattice code to compute the drag polar curve for the entire aircraft. Thus the performance estimates take into account the aircraft parasite drag and power characteristics, the profile drag of the lifting surfaces based on the airfoil analyses coupled with the wing lift distribution, and the induced drag of the specific wing-tail combination including the drag associated with trimming the aircraft. This approach is useful in examining the effect of airfoil changes on aircraft performance and puts the airfoil characteristics
in proper perspective with the total performance of the specific aircraft.

In the second approach, analytical equations have been developed to guide the designer in tailoring an airfoil for a particular aircraft. Specifically, two analytical expressions have been presented that allow computation of the desired locations of the upper and lower corners of the drag bucket if maximum range or velocity are to be optimized. These locations are functions of the desired laminar extents of the airfoil, $Re$, and the specific aircraft drag and power characteristics. These equations have been validated using the performance simulation approach showing good agreement even when trim effects are considered.

In the second part of the thesis, the groundwork for an automated cruise flap has been presented. The cruise flap is shown to shift the drag bucket of the airfoil by bringing the stagnation point to the desired region at the leading-edge. In doing this, large favorable pressure gradients are achieved on the airfoil, promoting large extents of laminar flow over a wide range of lift-coefficients. Thus an airfoil with large amounts of laminar flow can be made practical despite the narrow drag bucket for the airfoil with a zero-degree flap. The desired region of the leading-edge stagnation point has been quantified in the thesis. Pressure schemes are then presented for determining the optimum flap angle for a given lift coefficient. These schemes have been verified in the current research using both computational and experimental results for two NASA NLF airfoils showing that by holding a predetermined desired value of the pressure difference, minimum profile drag is achieved for the airfoil.

Experiments in the North Carolina State University Subsonic wind tunnel were used to demonstrate the closed-loop control of a trailing edge flap by specifying a desired value of the pressure difference for the two pressure-based automation schemes. The two schemes both performed well in tracking changes to the specified values of the pressure difference and lift coefficient. These results demonstrate
how the pressure relationships can be used to control the flap deflection in an automated cruise flap system and open the door to the possibilities for other applications.

4.2 Recommendations for Future Work

The analytical expression in Chapter 2 for the airfoil-aircraft design integration could form the basis for such analytical expressions for other applications where NLF airfoils are used. Equations like those derived in Chapter 2 may also be useful in multidisciplinary optimization codes.

The performance-simulation approach may be further useful if it is modified to work in the reverse direction. For example, once the airfoil designer runs through the method for a given airfoil and aircraft design, he could then click on the aircraft performance curves and drag them to create the desired aircraft performance characteristics. From these desired curves, the program would calculate the aircraft drag polar providing a goal that the airfoil designer can strive to achieve. Whether the changes be made to the wing, tail, fuselage, or airfoil, this tool would help the designer to quickly realize what is necessary to achieve the performance requirements and thus further integrate the airfoil and aircraft design processes.

The cruise flap study opens up a world of possibilities for future work and applications. For one, a detailed system study should be conducted for an automatic cruise flap system for an aircraft. The foundations have been laid down in this work. However, there still remain questions as to the implementation of the system on an aircraft. Such questions involve mainly the dynamics and control of the aircraft. To examine this, a more controls-oriented analysis should be conducted with the end result being the implementation of an automated cruise flap system on a test aircraft or UAV. This would also be a good experiment in
which to examine the effectiveness and practicality of the pressure based system in comparison to a hot-film based system.

The fact that in scheme 2, $\Delta C_p''$, is nondimensionalized by local surface pressures, makes it possible to have the system fully enclosed in the airfoil section under consideration. This advantage along with the fact that there is a one-to-one relationship between the pressure scheme and $C_l$ makes the system further applicable to segmented flaps that could be used in induced drag minimization and/or lift redistribution systems. The results of the research presented in this thesis thus not only provide novel approaches to the integration of the airfoil with the aircraft, but also provide a starting point for follow-up research that can be useful for the design of adaptive aircraft of the future.
Chapter 5

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Appendix A

Wind Tunnel Data from Closed-Loop Demonstration Experiments

This appendix presents the experimental data obtained in the North Carolina State University Subsonic Wind Tunnel. The first two tables list the as-designed upper and lower surface orifice locations respectively for the airfoil model. The z-coordinates represent the spanwise locations of the pressure taps and is the result of the 20-degree stagger used to prevent the upstream taps affecting the downstream measurements. The coordinates are listed for zero-degree flap deflection. A program was written to calculate locations of the taps on the flap for a given flap deflection. It should be noted that these locations have not been electronically verified and represent the design locations. Judging by the good comparison of pressure distributions from XFOIL and experimental results, it is assumed that these coordinates are fairly accurate.

The third table presents the average tunnel operating conditions that were held nearly constant throughout the initial airfoil tests. This is followed by the pressure distributions of the airfoil for angles of attack from -6 degrees to 10 degrees and flap angles of -10, -5, 0, 5, and 10 degrees. This database of pressures was used to further validate the pressure schemes and to arrive at approximate gradients for
use in the Jacobian matrix of the Newton iteration for the closed-loop system.
Table A.1: Model upper-surface orifice locations.  
Delta F = 0.0 deg

<table>
<thead>
<tr>
<th>Orifice</th>
<th>x/c</th>
<th>y/c</th>
<th>z/c</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>0.97055</td>
<td>0.00583</td>
<td>0.35289</td>
</tr>
<tr>
<td>2</td>
<td>0.94207</td>
<td>0.01026</td>
<td>0.34253</td>
</tr>
<tr>
<td>3</td>
<td>0.91180</td>
<td>0.01539</td>
<td>0.33152</td>
</tr>
<tr>
<td>4</td>
<td>0.88108</td>
<td>0.02099</td>
<td>0.32034</td>
</tr>
<tr>
<td>5</td>
<td>0.85017</td>
<td>0.02696</td>
<td>0.30910</td>
</tr>
<tr>
<td>6</td>
<td>0.77189</td>
<td>0.04299</td>
<td>0.28062</td>
</tr>
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<td>7</td>
<td>0.74037</td>
<td>0.04955</td>
<td>0.26916</td>
</tr>
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Table A.3: Average freestream flow properties for initial tunnel runs.

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<tr>
<td>Velocity</td>
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</table>
Figure A.1: Pressure coefficient distributions for the model airfoil with 10 degree flap deflection.

Figure A.2: Pressure coefficient distributions for the model airfoil with 10 degree flap deflection.
Figure A.3: Pressure coefficient distributions for the model airfoil with 10 degree flap deflection.

Figure A.4: Pressure coefficient distributions for the model airfoil with 10 degree flap deflection.
Figure A.5: Pressure coefficient distributions for the model airfoil with 10 degree flap deflection.

Figure A.6: Pressure coefficient distributions for the model airfoil with 10 degree flap deflection.
Figure A.7: Pressure coefficient distributions for the model airfoil with 5 degree flap deflection.

Figure A.8: Pressure coefficient distributions for the model airfoil with 5 degree flap deflection.
Figure A.9: Pressure coefficient distributions for the model airfoil with 5 degree flap deflection.

Figure A.10: Pressure coefficient distributions for the model airfoil with 5 degree flap deflection.
Figure A.11: Pressure coefficient distributions for the model airfoil with 5 degree flap deflection.

Figure A.12: Pressure coefficient distributions for the model airfoil with 5 degree flap deflection.
Figure A.13: Pressure coefficient distributions for the model airfoil with 0 degree flap deflection.

Figure A.14: Pressure coefficient distributions for the model airfoil with 0 degree flap deflection.
Figure A.15: Pressure coefficient distributions for the model airfoil with 0 degree flap deflection.

Figure A.16: Pressure coefficient distributions for the model airfoil with 0 degree flap deflection.
Figure A.17: Pressure coefficient distributions for the model airfoil with 0 degree flap deflection.

Figure A.18: Pressure coefficient distributions for the model airfoil with 0 degree flap deflection.
Figure A.19: Pressure coefficient distributions for the model airfoil with -5 degree flap deflection.

Figure A.20: Pressure coefficient distributions for the model airfoil with -5 degree flap deflection.
Figure A.21: Pressure coefficient distributions for the model airfoil with -5 degree flap deflection.

Figure A.22: Pressure coefficient distributions for the model airfoil with -5 degree flap deflection.
Figure A.23: Pressure coefficient distributions for the model airfoil with -5 degree flap deflection.

Figure A.24: Pressure coefficient distributions for the model airfoil with -5 degree flap deflection.
Figure A.25: Pressure coefficient distributions for the model airfoil with -10 degree flap deflection.

Figure A.26: Pressure coefficient distributions for the model airfoil with -10 degree flap deflection.
Figure A.27: Pressure coefficient distributions for the model airfoil with -10 degree flap deflection.

Figure A.28: Pressure coefficient distributions for the model airfoil with -10 degree flap deflection.
Figure A.29: Pressure coefficient distributions for the model airfoil with -10 degree flap deflection.

Figure A.30: Pressure coefficient distributions for the model airfoil with -10 degree flap deflection.
Appendix B

Thin Airfoil Theory Analysis

This appendix briefly presents the thin airfoil theory calculations used in the analysis of the stagnation-point relationship of the cruise flap.

It is well known that the total circulation, $\Gamma$, of an airfoil can be calculated from thin airfoil theory using Eq. B.1.

$$\Gamma = \int_0^c \gamma(x)dx$$ \hspace{1cm} (B.1)

Using the coordinate transformation shown in Eq. B.2 the total circulation becomes Eq. B.3.

$$x = \frac{c}{2}(1 - \cos \theta)$$ \hspace{1cm} (B.2)

$$\Gamma = \frac{c}{2} \int_0^\pi \gamma(\theta)\sin \theta d\theta$$ \hspace{1cm} (B.3)

Where $\gamma(\theta)$ can be presented as the Fourier series shown in Eq. B.4.

$$\gamma(\theta) = 2V_\infty [A_0 \frac{1 + \cos \theta}{\sin \theta} + \sum_{n=1}^{\infty} A_n \sin n\theta]$$ \hspace{1cm} (B.4)

As it turns out using thin airfoil theory for an airfoil with small trailing-edge flap
deflections, the Fourier coefficients can be calculated from Eqs. B.5 and B.6.

\[
A_0 = \alpha + \frac{1}{\pi} \int_{\theta_f}^{\pi} \delta_f d\theta = \alpha + \frac{\delta_f}{\pi} (\pi - \theta_f) \tag{B.5}
\]

\[
A_n = -\frac{2}{\pi} \int_{\theta_f}^{\pi} \delta_f \cos n\theta d\theta = \frac{2\delta_f}{\pi} \sin \frac{\theta_f}{n} \tag{B.6}
\]

Where \(\delta_f\) and \(\theta_f\) are the flap angle and flap hinge location, respectively, in radians.

It also turns out from thin airfoil theory that the approximate \(C_l\) of the airfoil can be calculated from only the first two coefficients using Eq. B.7.

\[
C_l = 2\pi A_0 + \pi A_1 \tag{B.7}
\]

Finally, to compute the \(C_{l\text{ideal}}\) corresponding to the angle-of-attack at which the stagnation point is located exactly at the leading edge of the thin airfoil, one must find the \(C_l\) where there are no singularities (or infinite suction peaks) on either the upper or lower surface. This \(C_{l\text{ideal}}\) corresponds to the \(C_l\) where the Fourier coefficient, \(A_0\), goes to zero. Thus the \(C_{l\text{ideal}}\) can be calculated using Eq. B.8.

\[
C_{l\text{ideal}} = \pi A_1 = 2\delta_f \sin \theta_f \tag{B.8}
\]