DESIGN OF A PASSIVE FLOW CONTROL DEVICE DERIVED FROM BIRD WING AERODYNAMICS

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Abstract

Flow separation is a constant problem in airfoil applications ranging from airplane wings to turbine blades. Active solutions such as pulsed blowing require more energy and complexity while passive solutions such as vortex generators often come at a cost of increased drag. Bird wings have a unique solution to the problem that involves the covert feathers on the top of their wings. The feathers pop up under separated conditions and create two small circulations zones that increase lift and decreases drag. In this thesis, an airfoil is designed and fabricated along with feather-like flaps that are attached to the surface of the wing. Particle image velocimetry (PIV) testing is used to verify the flap’s effect on the flow field and show enhanced streamline curvature in the separated region. A three-component strain gage force transducer is designed and fabricated to measure lift, drag, and pitching moment on the wing. Experimental results show an increase in lift by up to 6.4%, decrease in drag by as much as 13.2%, and decrease in pitching moment by 8.3% by adding a flap to the bare wing. Finally, three-dimensional effects are investigated by testing flaps with a different number of span-wise divisions.
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Chapter 1. Introduction

1.1 Problem Definition

Airfoils in applications ranging from wind turbines to aircraft depend on their ability to produce high lift with minimal drag. Flow separation is a phenomenon that occurs when an increase in local pressure gradients causes the boundary layer to detach from the airfoil. This condition changes the apparent airfoil shape leading to a decrease in lift and an increase in drag. It is therefore essential to reduce the extent of flow separation in order to maintain a stable and efficient airfoil design.

Birds offer a unique solution to the problem of reducing flow separation. In conditions where a bird’s wing is prone to separation, feathers on the top of the wing called “coverts” pop up passively as boundary layer flow begins to reverse. This feather displacement as seen in Figure 1 confine the separated flow and keeps it from spreading across the wing. This results in two small circulation zones that pull the free stream closer to the wing and retain more lift in the separated region. As a result, the bird can compensate for sudden localized pressure gradients and keep the entire flow from separating. This flow control technique could be preferable to other solutions to separation and is the focus of this project.
1.2 Methodology

A standard low Reynolds number airfoil was fabricated using a rapid prototyper and augmented with small plastic flaps. The airfoil was placed in the test section of a closed-loop wind tunnel and rotated to different angles of attack. Particle image velocimetry was used for flow visualization around the flap and a homemade three-component force sensor was designed and built to measure aerodynamic forces. The force sensor was used to measure the effects of the flaps on the characteristic lift, drag, and pitching moment curves of the airfoil as well as optimize the flap design.

1.3 Thesis Outline

This thesis is divided into five parts that outline the project as a whole. The Introduction provides a brief description of the project while the Background gives a more in-depth look at airfoil theory, current technologies, and applications. The Methods section describes the experimental set up including the
wind tunnel, particle image velocimetry system, force transducer, wing, and passive flap design. Data from testing is presented and analyzed in the Results and Discussion section followed by the Conclusion and Recommendations based on the most pertinent observations. Appendices and references can be found at the end of the thesis.

1.4 Objectives

It is the purpose of this thesis to mimic the described avian flow control technique by designing and fabricating feather-like flaps that reduce the effects of separation on a given airfoil. The flaps are tested to ensure enhanced aerodynamic performance under separated conditions and compared to results from other sources. The design is then optimized to find the best way to divide the flaps in the span-wise direction.

Chapter 2. Background

2.1 Airfoil Principles

2.1.1 Definitions and Nomenclature

At the simplest level, an airfoil is a structure or surface that is designed to extract certain reaction forces from moving fluids. An airfoil may be as simple as a flat plate or become increasingly complicated with various degrees of thickness and curvature. In order to describe more complicated designs, specific nomenclature has been adopted for airfoil parameters. The “leading edge” of the airfoil is the farthest point upstream, while the “trailing edge” is the farthest downstream. The “chord line” is the mean distance between the leading edge and the trailing edge. The “span” is the distance between the tip of the wing and the
base of the wing (perpendicular to the chord). While some airfoils are symmetric about their chord, others have different curvatures along the top and bottom. “Camber” is used to describe these curvatures and is defined by a curve halfway between the top and bottom surfaces (for a symmetric airfoil, the camber line is straight). Finally, the “angle of attack” of an airfoil may be defined as the angle of the chord line relative to the free stream air. Each of these terms can be seen in Figure 2.

![Airfoil nomenclature](image)

**Figure 2. Airfoil nomenclature**

2.1.2 Lift, Drag, and Pitching Moment

The reaction forces created by an airfoil are often divided into two main forces: “lift” and “drag.” Lift is the resultant force perpendicular to the incoming (or “free stream”) fluid flow, while drag is parallel to incoming fluid. Figure 3 shows these two forces with respect to the free stream flow.
The presence of drag on an airfoil is relatively easy to understand. It can be divided into two types: form drag and skin friction. Form drag is the force on an airfoil due to the pressures acting on its surface. As high-speed air approaches the front of the airfoil, it slows down and stops at the leading edge. This dead zone, also known as a stagnation point, creates a high-pressure area at the front of the airfoil. If the fluid at the back of the airfoil is at lower pressure this will result in a force parallel and opposite to the direction of the free stream flow. However, under normal flow conditions, skin friction is often the dominant form of drag. This force is caused by the frictional resistance of the viscous fluid moving past the wing.

Lift is more complicated. A flat plate may produce lift when placed at an angle to incoming fluid flow, while some cambered airfoils don’t require any angle at all. There are a variety of theories about what actually gives an airfoil lift, many of which are wrong. A classic explanation, and perhaps one of the more popular theories, is the “Equal Transit Time” fallacy. It asserts when fluid
particles hit the front of an airfoil, they divide up with some flowing over the top and others flowing along the bottom. Eventually, they meet again at the tail of the wing. Since the flow over the topside of an airfoil needs to travel farther than that along the bottom (airfoils are generally more curved on top than the bottom), it must move faster. Bernoulli’s Principle states that faster moving fluid has lower pressure than slower fluid, so there is a resultant force pointing upward with respect to the chord.

There is of course, no physical reason why air “needs” to meet back up at the tail of the airfoil. In fact if it did, the flow field would violate Newton’s Third Law. What’s more, Bernoulli’s Principle only applies to inviscid flow. In 1752, a French mathematician by the name of Jean le Rond d’Alembert proved that there was in fact no lift or drag on an airfoil in inviscid flow. Known as d’Alembert’s paradox, the idea seems to prohibit the use of Bernoulli’s principle in explaining the origin of lift.

The prevailing theory for the origins of lift focuses on flow field formation. For simplicity, consider the airfoil as a flat plate. When the airfoil is placed at an angle into free stream flow, a stagnation point develops at the front lower part of the plate. In inviscid flow, another stagnation point develops at the top of the trailing edge of the plate, as flow along the bottom curves around to the top. These flow characteristics can be seen in Figure 4A.
As mentioned before, stagnation points create regions of higher pressure. Since the airfoil has stagnation points on both the top and bottom of the airfoil, there is no net pressure differential and in turn no lift for the inviscid scenario.

However, if the flow is in fact viscous (as it is in real applications), the flow field changes. As the airfoil begins its motion, a shear layer is shed off the tail of the airfoil in an induced vortex. This is often called the “starting vortex” and can be seen in Figure 5.
This starting vortex creates a circulation, and the Kelvin circulation theorem states that the net circulation around a closed curve remains constant with time. Therefore, if we take a closed curve around the airfoil and the starting vortex, there must be another source of circulation to maintain a net value of zero. A circulation zone thus forms around the airfoil (Figure 4B and Figure 5) and pushes the upper stagnation point to the rear of the airfoil. This circulation effect is known as the Kutta condition and the altered flow field can be seen in Figure 4C. Since there is now only a stagnation point on the bottom of the airfoil, there is only one region of low pressure. This creates a pressure differential between the top and the bottom of the airfoil resulting in lift.

There is, in fact, something to be said for Bernoulli’s principle in all this. While it does not apply to the formation of the flow field in Figure 5 (as the Kutta condition requires viscosity), it can in fact be used to describe the lift induced by the free stream flow. In the free stream viscosity can be considered negligible, and since the free stream is traveling faster over the top of the airfoil due to the previously described flow field formation, Bernoulli’s Principle correctly describes a lower pressure region.

Perhaps the simplest description of lift in an established flow is using Euler’s equation. Euler’s equation describes the relationship between streamline curvature and pressure gradients (where $p$ is the fluid pressure, $n$ is a vector normal to the streamline, $\rho$ is the fluid density, $v$ is the fluid velocity, and $R$ is the distance from the center of curvature of the streamline):

$$\frac{dp}{dn} = \frac{\rho v^2}{R}$$

(1)
From Eq. 1 it is apparent that the pressure of the fluid in a given streamline is inversely proportional to the distance from the center curvature of that streamline. In the case of an airfoil, the streamlines curve around the top surface as seen in Figure 6:

![Figure 6. Streamlines around an airfoil](image)

Moving away from the upper surface means increasing the distance from the center of curvature of the streamlines above the airfoil. This means that pressure increases to ambient conditions farther away from the airfoil. Likewise, the pressure decreases from ambient conditions moving towards the surface of the airfoil. Since the pressure underneath the airfoil is more or less ambient (there is minimal streamline curvature), there is a net pressure difference that results in a lift force.

The distribution of lift and drag on an airfoil is far from uniform. This imbalance of forces will actually create a torque around the ¼ chord of the airfoil (depending on the airfoil this may be referred to as the aerodynamic center) referred to as the pitching moment. An example of pressure distributions contributing to this kind of torque is shown in Figure 7.
Pitching moments can be important in the stability of an aerodynamic structure. Planes, for instance, compensate for pitching moments on their wings using horizontal stabilizers in order to avoid dipping their noses during flight. Pitching moments can also lead to higher stresses on wings that can be a limiting factor in design.

2.1.3 Performance

The performance of an airfoil can be characterized by the lift, drag, and pitching moment it is subject to at different angles of attack. The forces are usually normalized into a lift coefficient $C_L$, drag coefficient $C_D$, and pitching moment coefficient $C_M$ as follows:
In the above equations, $L$ is the average lift force, $D$ is the average drag force, $M$ is the average pitching moment, $\rho$ is the fluid density, $v$ is the free stream velocity, and $A$ is the planform area of the wing (typically calculated as the chord $c$ multiplied by the span). Through wind tunnel testing or computational modeling, the lift coefficients and drag coefficients can be calculated. They are typically plotted over a range of angles of attack in as shown in Figure 8. This is often referred to as an “alpha sweep.”

\begin{align*}
C_L &= \frac{L}{\frac{1}{2} \rho v^2 A} \quad (2) \\
C_D &= \frac{D}{\frac{1}{2} \rho v^2 A} \quad (3) \\
C_M &= \frac{M}{\frac{1}{2} \rho v^2 c A} \quad (4)
\end{align*}

Figure 8. Typical alpha-sweep for a NACA 4412 airfoil
As angle of attack increases the lift will increase as well. The peak of the $C_L$ graph is the “stall angle” and signifies the angle of maximum lift for the airfoil. This peak occurs due to a phenomenon called separation, causing the airfoil to effectively change shape resulting in a decrease in lift and increase in drag.

2.1.4 Separation

Separation is a major limitation to the performance of airfoils and is therefore an important problem in aerospace research. As fluid moves over an airfoil, it wants to “stick” to the upper surface due to the Coanda effect. Since the surface is curved, the streamlines will curve too. In turn, the fluid traveling over the top of the wing speeds up and then eventually slows back down to the free-stream speed as it reaches the tail. This deceleration creates an adverse pressure gradient in the chord-wise direction that can be seen in Figure 9A.

![Figure 1](image)

**Figure 1.** A) Pressure distribution over the surface of an airfoil (where $s_1$ is closest to the leading edge), B) Reverse flow due to adverse pressure gradient (Chklovski n.d.)
When airfoils are subjected to higher angles of attack the streamlines are forced to curve more and more. This causes the adverse pressure gradient to increase in magnitude. If the gradient is large enough, it can cause the fluid in the boundary layer to slow down and reverse its direction all together (Figure 9B). At this point the boundary layer detaches from the wing resulting in a loss of lift and increase in pressure drag. This detrimental flow phenomenon is called separation and can be seen in Figure 10. The goal of this project is to control this separation so that airfoils can be used at higher angles of attack with minimal separation effects.

![Image](image.png)

**Figure 10. Flow Separation**

2.2 Current Technology

2.2.1 Active Flow Control

A variety of technologies exist that limit the effects of flow separation. Each approach can be defined as either “active” or “passive” depending on how it is implemented. Active techniques require some sort of actuation and often lead to
extra onboard electronics or moving parts while passive ones do not require any power.

One example of an active flow control technique is leading edge slots as seen in Figure 11. Leading edge slots are modifications to the front of an airfoil that can be opened and closed to allow air from the bottom of the wing to move to the top. This extra airflow mixes with the boundary layer on the surface of the wing and essentially increases its kinetic energy. This energizing effect enables the boundary layer flow to resist adverse pressure gradients and ultimately delay flow separation.

![Figure 11. Leading edge slot installed on a wing](image)

In a similar approach, synthetic jets may be installed in the airfoil and directed into the boundary layer. In the active version of this design, air can be blown periodically over the wing to energize the boundary layer and delay separation. By blowing intermittently, the technique takes advantage of instabilities in the flow field to trip the flow into turbulence and delay separation.
This process can be enabled by an active control system that monitors flow conditions and adjusts accordingly.

2.2.2 Passive Flow Control

Passive flow control is a simpler solution to reducing separation that does not require any power or actuation. This can be useful for wing designs that require minimal complexity and can make for a more reliable technology. Vortex generators, for example, are small geometric structures that are attached to the top of an airfoil. These attachments (as seen in Figure 12) induce vortices in the airflow over the wing that draw higher velocity air into the boundary layer. Similar to the leading edge slot, this energizes the boundary layer and delays separation of the fluid flow.

Figure 12. Vortex generators on an aircraft wing

Even with existing technologies, flow separation still largely limits the maneuverability of aircraft and the lift on a given airfoil. In addition, many of the current solutions increase lift at the cost of increased drag. One design derived from bird wing aerodynamics could offer a better solution to separation control.
2.2.3 Bio-inspired Flaps

Birds perform aerial maneuvers such as sharp turns and stall landings with exceptional agility. This ability is due in part to a long history of evolutionary optimization resulting in aerodynamic advantages. In particular, a passive flow control technique on the surface of bird wings provides an interesting solution to separation.

In high lift situations or gusty flight conditions, a bird’s wing is prone to separation. However, certain “covert” feathers on the top of the wing can help limit the effects of separation. The onset of separation causes reverse flow in the boundary layer to spread towards the leading edge. This reverse flow causes the covert feathers to pop up, stopping the separated flow from moving farther up the wing.

The phenomenon was first studied by the former German Aeronautical Establishment in the late 1930s. An article by Liebe (1979) acknowledged that separation leads to reverse flow that causes the covert feathers to pop up, “acting like a brake on the spreading of flow separation towards the leading edge”. The feathers act as an effective barrier to the reverse flow and keep it from moving across the wing.

The result is an area of separation divided into two regions: one below the feather and one above it. If separation is strong enough, this can create two circulation zones as seen in Figure 13.
The development of these circulation zones is essential to the covert feather’s aerodynamic effects. Instead of allowing a large region of circulation to form, the feathers limit it to two smaller zones. This new separation shape essentially creates a new airfoil surface for the free stream to follow. It is smaller and more curved than the separated zone without a feather and can in turn produce more lift. If the wing returns to a lower angle of attack and flow is no longer separated, the feathers will be pushed back down to their original position by the reattached boundary layer. This effect can be seen in flow visualization and was even demonstrated in studies of flow around bluff bodies. A visual of this effect on an airfoil can be seen in Figure 14.
This idea of using circulation zones to direct airflow is far from new. An example that has become popular with RC planes is the Kline-Fogleman airfoil. Discovered by Richard Kline and Floyd Fogleman in the 1960s, these airfoils incorporate a stepped design that creates small circulation zones along the profile of the wing (Figure 15). Each circulation zone essentially fills the stepped portions of the wing to allow for smooth airflow. The idea is that at different angles of attack, these circulation zones change in shape and size to enhance performance. The major downfall of the airfoils, however, was a low lift to drag ratio. With circulation zones placed in multiple locations along the airfoil, the increase in pressure drag made the enhanced lift effects irrelevant.
Figure 15. Example of a Kline-Fogleman airfoil

The beauty in the flow control by covert feathers is that these circulation zones are only present during separation. For most angles of attack, the wing acts as a smooth airfoil and has relatively low amounts of drag. Only when there is separation, and pressure drag is already present, these circulation zones are created and lift ameliorated.

The most extensive research on feather-like flow control devices has come from Schatz et al (2004) in a combined effort from the Technical University of Berlin and the German Aerospace Center. Using an HQ17 glider airfoil, the group fabricated movable flaps using flexible plastic and thin sheet metal. The group was able to show an increase in lift coefficient by more than 10% in wind tunnel testing at Reynolds numbers of $1 \cdot 10^6$ to $2 \cdot 10^6$. During experimentation, they found that the best results could be achieved by placing the flap 80% of the way along the chord. The flap was cut to be 12% of the chord length and allowed to open 57 degrees. The experimental results were verified using numerical simulation as shown in Figure 16 (with “single” and “triple” mass referring to numerical simulations of flaps with different weights).
Similar results were reported by Schlüter (2009) in water tunnel tests. At Reynolds number of $3 \cdot 4 \cdot 10^5$, improved lift after stall was reported for four separate airfoils. In one case, the lift was increased by 50% after stall by moving the flaps up the wing to 60% of the chord.

Wang and Schlüter (2012) went on to do a more extensive characterization of flaps that dealt with their size and location. Flaps were made using carbon-fiber strips taped to the airfoil surface and experiments were conducted at a Reynolds number of $4 \cdot 10^4$. Using a SD8020 wing section, it was shown that the best single-flap design covered 30% of the chord located 70% along the chord covering 80% of the span. The improved lift characteristics for this design can be seen in Figure 17.
2.4 Applications

A successful bio-inspired flap design that limits the effects of separation would be beneficial to a variety of airfoil applications. The lightweight, simple design could be an attractive solution in small aircraft applications such as Unmanned Aerial Vehicles (UAVs) and Micro Aerial Vehicles (MAVs). For these applications, a device that maintains lift at high angles of attack could help the stability of the aircraft in dynamic conditions or even aid in smoother landings.

Another area where feather-like flaps could be useful is wind energy. By improving the performance in separated regions, flaps like these on wind turbines could increase power generation over large ranges of wind speeds. With no required power supply or actuation they would be a simple addition to turbine
Chapter 3. Methods

Chapter 3 is divided into sections explaining the design and evaluation of each part of the experimental set up. First each measurement component is discussed including the rotary table, pitot tube traverse, particle image velocimetry system, and force transducer. The National Instruments LabVIEW program controlling each test run is described, followed by the design and characterization of the wind tunnel itself. The chapter ends with a discussion of the airfoil and passive flaps followed by an uncertainty analysis of the aerodynamic measurements.

3.1 Rotary Table

A DCI rotary table was used to hold the airfoil and rotate it to different angles of attack. The rotary table was turned using a Servo Systems stepper motor controlled by a Stepperboard BC2D15 motor controller. A picture of the system can be seen in Figure 18.
The table was calibrated by placing a digital level at its center and rotating it to different angles of attack. A certain amount of hysteresis was present if the motor needed to switch directions and this was compensated for in the accompanying National Instruments LabVIEW program. Once calibrated, the rotary table had an error of less than 0.1 degrees and was assumed to contribute negligible error to the overall system.

3.2 Pitot Tube Traverse

The uniformity of the free stream flow was measured using a pitot tube attached to two linear stages (Figure 19). The stages were controlled by a StepperBoard BC2D15 motor controller that traversed the pitot tube up and down.

Figure 18. Rotary table
the test section at three span-wise locations. The tip of the pitot tube was placed at approximately the same location as the leading edge of the airfoil used during testing. The pitot tube was connected to an Omega PX653 Differential Pressure Transmitter that was read using a National Instruments 9215 Analog Input Module. Five hundred samples were taken at 1kHz for each vertical location and averaged. The test section was swept six times so that each point on the graph contained an average of six 500-point velocity measurements.

![Figure 19. Pitot tube traverse set up](image)

The accuracy of the vertical position of the pitot tube was measured by comparing the heights from separate runs with a ruler. The linear stages were able to raise and lower the pitot tube in ¼ inch steps with an accuracy of +/- 0.031
inches. The repeatability of the system was tested by placing a piece of tape below the pitot tube at the lowest point of the test section. Through multiple sweeps the linear stages returned the pitot tube to the same location within +/- 0.031 inches.

The angle of the pitot tube with respect to the free stream flow had an uncertainty of 1-2 degrees. Experimental tests showed that this could lead to a velocity error of up to 0.2 m/s. This corresponds to a 0.3% error in percent of the maximum expected lift coefficient.

3.3 Particle Image Velocimetry (PIV) System

3.3.1 Overview

A Particle image velocimetry (PIV) system was used for flow visualization and uniformity measurements. PIV measures flow velocity by taking two successive pictures of the air over a specified time-step. The air can be “seeded” with fog or even small glass beads so that the camera can track particles as they move through the tunnel. By measuring the distance each particle moves between each frame, the software can then determine the velocity profile of the flow. A typical PIV system can be seen in Figure 20.
For this study, the airflow was seeded using a Rosco 1000 Fog Machine that injected evaporated aqueous glycol solution (“Fog Juice”) into the tunnel. The particles were illuminated by a New Wave Research Solo PIV laser that emitted a laser sheet parallel to the flow field. Images were captured using a TSI Powerview Plus camera with a Nikon AF NIKKOR 50mm lens. The camera was mounted to a Klinger goniometer that could be manually adjusted to different locations and angles relative to the wing surface. An image of the experimental set up can be seen in Figure 21.
The intensity and trigger mechanism of the laser were controlled by a New Wave Research Laser Power Supply while the timing between the laser pulses and camera was controlled by a TSI 610034 Synchronizer. TSI’s Insight 3G software was used to adjust parameters and process the image data.

3.3.2 Determination of Parameters

Once a flow speed is determined for a given experiment, the desired time step between picture frames must be calculated to allow for appropriate movement of particles. If fog particles move too much or too little between each frame, the tracking software loses accuracy and provides less usable data.

To calculate the appropriate time step, the laser sheet was aligned with a ruler placed on the surface of the wing. The camera was then focused on the ruler
and calibrated in Insight 3G. This calibration process gave the number of pixels per millimeter \( N \) for the specific camera location. Using this number and the fact that ideal particle movement was approximately 16 pixels between frames (1/4 of the 64 pixel interrogation region), the time step \( \Delta t \) (in seconds) between each frame could be calculated based on the free stream velocity \( U \):

\[
\Delta t = \frac{16 \text{pixels}}{N \frac{\text{pixels}}{\text{mm}}} \left( \frac{1}{1000} \frac{\text{mm}}{m} \right) \left( \frac{m}{s} \right) \quad (5)
\]

This parameter was typically between 110-15 µs (it varied depending upon the calibration image) and entered into Insight 3G along with the timing of the laser pulse and camera shutter position. The latter two parameters were visually adjusted to ensure the flow was illuminated and photographed in the correct order. An example of the timing scheme can be seen in Figure 22.

![Timing Diagram](image)

Figure 22. Example of Insight 3G timing scheme for PIV testing
3.3.3 Data Processing

Once captured, the image pairs were processed using the Insight 3G software to determine the velocity profiles of the flow. The particles were tracked by looking at small areas or “interrogation regions” of the images. To do this, the software divided the pictures into grids and employed an autocorrelation strategy using two-dimension Fast Fourier Transforms. The displacements of the particles were resolved into velocities by the software using the time step calculated.

The processor used a Recursive Nyquist grid engine and a Gaussian peak engine. The starting interrogation region was 128 x 128 pixels and ended in a 64 x 64 pixel interrogation region. The processor set the maximum displacement of particles to 30% of the interrogation region. In each test 30 images pairs were typically taken and averaged to find the flow velocity.

3.4 Force Sensor

3.4.1 Overview

The measurement of aerodynamic forces is essential for the validation of flow control devices. Therefore, it was necessary to construct a force balance to measure the lift and drag on an airfoil in the wind tunnel test section. Since the sensor needed to be able to measure small changes in lift, it needed to be accurate to within at least 2% of the maximum expected force. The limited budget available for the project ruled out high-precision six-component force sensors that are typically used in wind tunnel testing. A combination of single or dual axis force sensors was investigated but the price and resulting complexity of the experimental set-up made this design undesirable. A custom, three-component
force sensor was therefore built to measure the lift, drag, and pitching moment on a given airfoil.

A strain gage based transducer design was selected due to the static loading conditions as well as the desire for low cost and relatively small amounts of drift. In this type of transducer, a spring element typically machined out of metal is allowed to bend under an applied load. Strain gages attached to the element in certain places measure the corresponding strain by outputting changes in voltage levels. These voltages can be calibrated in such a way that any applied force or combination of forces can be measured.

Strain gages are typically wired in a Wheatstone bridge configuration as seen in Figure 23. A full bridge circuit maximizes sensitivity while removing temperature effects and was therefore chosen to measure each force component. The full bridge configuration requires four strain gages attached to the spring element in different locations. Two of the gages measure equal tensile strain while the other two gages measure equal and opposite compressive strain. The bridge is then wired so that compressive and tensile gages are adjacent to each other. In Figure 23, for instance, gages $R_1$ and $R_3$ could measure compressive strain while $R_2$ and $R_4$ measure tensile strain.
In a full bridge configuration, the voltage ratio is measured by the following equation (where $V_o$ is the output voltage, $V_{ex}$ is the excitation voltage, $G$ is the gage factor, and $\varepsilon$ is the magnitude of the strain being read by all four gages):

$$\frac{V_o}{V_{ex}} = -G\varepsilon \quad (6)$$

The airfoil to be used in testing was relatively small, so an internal force balance would have been too delicate to stand up to aerodynamic loads. The sensor was therefore designed to attach to the side of the airfoil but remain outside of the test section. The sensor was fixed to a rotary table that could be turned to different angles with respect to the free-stream. The final design is shown in Figure 24.
3.4.2 Spring Element

The spring element for a strain gage based transducer is one of the most important parts of the design. The goal of this component is to react to applied forces in each axis in such a way that strain is localized and magnified in a relatively uniform manner. This allows the attached strain gages to measure wider ranges of strain and in turn produce higher resolution force measurements.

The material for the spring element was chosen to be aluminum due to the low cost and machinability. In addition, aluminum is relatively flexible and would therefore be appropriate for the small aerodynamic forces expected during testing.
A variety of geometries were considered for the spring element based on anticipated loads and geometric constraints. One particularly important consideration was the fact that the sensor would be experiencing varying degrees of torque (or pitching moment) as the aerodynamic center of the airfoil changed with angles of attack. Therefore, to read accurate lift and drag forces, the sensor would need to either cancel out or compensate for the pitching moment. One design that is often used is the “triple beam design” shown in Figure 25.

![Figure 25. Triple-beam spring element design](image)

In this geometry, the strain is measured on the central beam with four strain gages (at the ‘T’ and ‘C’ locations in the diagram) arranged in a Wheatstone bridge. The upper and lower beams react to externally applied torques while the central beam is free to bend exclusively in the desired plane.

Unfortunately, the triple beam design proved to be ill suited for the expected aerodynamic loads because the strain on the center beam was too small relative to the strain on the rest of the spring element. This resulted in a design that would have low sensitivity (with strain ranges below 1 micro-strain) for an
appropriate factor of safety. The popular binocular-style spring element geometry was investigated as well, but given the relatively small aerodynamic forces involved, the specialized shape complicated the design more than it helped. The final design used a similar “double-beam” strain strategy with a simpler geometry and can be seen in Figure 26.

![Figure 26. Final spring element geometry and corresponding Wheatstone bridge configuration](image)

With the double-beam geometry shown above an applied force at the end causes both beams to bend in an ‘s’ shape. This creates equal and opposite tensile and compressive strains on opposite sides of the beams. Strain gages placed at locations \( R_1, R_2, R_3, R_4 \), can therefore be arranged in a Wheatstone bridge to maximize the voltage differential.

In order to measure both lift and drag forces, the design in Figure 26 needed to be modified to measure forces in two directions perpendicular to each other. The vertical and horizontal components of each of these strain directions could then be combined to calculate lift and drag forces respectively. This was accomplished by creating a “box” design with each side consisting of the double-beam geometry (Figure 27).
The spring element was designed in SolidWorks and FEA simulations showed a factor of safety of 6.1 under normal loading conditions. Under extreme loading conditions (such as with the tunnel at full speed) the factor of safety was determined to be 1.13. Strain gages were then placed on opposite sides of the box in two Wheatstone bridges to measure strains in two directions (referred to as the X and Y components) normal to one another.

While the double-beam design was capable of accurately measuring loads placed at its tip, off-axis loading that creates moments along the beam’s axis could introduce errors into the strain reading. This was especially apparent in the box design, where a torque on the end would cause one side of the box to see excess upward force while the opposite side would see excess downward force.
At first, it appeared that clever placement of the strain gages on the design could cancel these moments out. By placing the gages on opposite sides of the box as well as on opposite beams (as shown in Figure 27) the strain gages would read equal and opposite errors that would ultimately be canceled out in the Wheatstone bridge. However, preliminary tests demonstrated that forces applied at different-off axis locations would induce different voltage readings, showing that errors were still making their way into the readings.

Since the initial box design was unable to eliminate moment effects, a new approach was needed. Torsional strain gages (strain gages with the gage area angled at 45 degrees) were attached towards one end of each beam to measure the moments acting on the spring element. The orientation of the strain gages was such that two measured tensile strains while the other two measured compressive strains. This allowed for another Wheatstone bridge to be wired that could accurately measure external torques on the sensor (referred to as the M component). The moments could then be calibrated out later to reduce the error in the lift and drag measurements. The final, three-component box design can be seen in Figure 28.
The deflection of the spring element was also examined to ensure that the wing wouldn’t be changing by a significant angle under aerodynamic loads. The final deflection of the tip of the wing under loading was only 0.2 inches, which resulted in a change in angle of about 0.5 degrees. This results in negligible changes in aerodynamic force readings and could therefore be ignored as a source of error.

3.4.3 Strain Gage Selection and Attachment

Appropriate strain gages were selected based on geometric and material constraints. In particular, the gages needed to measure strain with transducer-quality precision while fitting on the spring element and matching the temperature characteristics of aluminum. A constantan grid alloy was also preferable due to its excellent linearity and solderability. Eight linear strain gages were required to
measure the X and Y forces while four shear and torque pattern strain gages were needed for the moments.

Omega’s SGT-1/350-SY13 Transducer Quality strain gages were chosen for the X and Y measurements. These gages had a 350-ohm resistance and a gage factor of 2.13 with a tolerance of +/- 0.35%. For the pitching moment, Omega’s SGK-SS3A-K35OU-PC23 and SGK-SS3B-K35OU-PC23 Karma Shear and Torque gages were chosen to measure compressive and tensile strains respectively. They had a 350-ohm resistance as well and a 2.04 gage factor with a tolerance of +/- 0.30%. Dimensions in millimeters of each gage are shown in Figure 29.

![Figure 29. Strain gage dimensions (in mm) for A) linear gages and B) torsional gages](image)

The gages were attached to the spring element as shown in Figure 28. All products used in the bonding process were obtained from Micro-Measurements. The surface of the spring element was sanded and prepped with M-Prep Conditioner A and M-Prep Neutralizer 5A. The gages were then attached to PCT-2M Installation Tape and bonded to the surface using 200 Catalyst-C and M-Bond 200 Adhesive.
The gages were soldered into Wheatstone bridges and connected to a National Instruments 9219 24-Bit Analogue Input Module. The module was installed in a National Instruments cDAQ-9172 CompactDAQ Chassis that could then be read and analyzed by a task created in National Instruments Measurement and Automation (MAX) software. This task could in turn be called from National Instruments LabVIEW programs to measure strain during wind tunnel testing.

3.4.4 Quality of Readings

Before the force transducer could be calibrated, the strain readings needed to be observed to ensure consistent, accurate results. The strain readings of the unloaded sensor were observed over a period of 10 minutes and a statistical analysis is summarized in Table 1:

<table>
<thead>
<tr>
<th>Statistical Measure</th>
<th>X Component</th>
<th>Y Component</th>
<th>Torsional Component</th>
</tr>
</thead>
<tbody>
<tr>
<td>Standard Deviation</td>
<td>6.85·10⁻⁸ strain</td>
<td>9.04·10⁻⁸ strain</td>
<td>5.50·10⁻⁸ strain</td>
</tr>
<tr>
<td>95% Confidence Interval</td>
<td>+/- 0.097 (% of mean)</td>
<td>+/- 0.015 (% of mean)</td>
<td>+/-0.023 (% of mean)</td>
</tr>
<tr>
<td>Drift per second</td>
<td>0.0017·10⁻⁷ (strain/s)</td>
<td>-0.0013·10⁻⁷ (strain/s)</td>
<td>-0.00027·10⁻⁷</td>
</tr>
<tr>
<td>Drift over 10 minutes</td>
<td>1.03 (% of max strain)</td>
<td>-0.79 (% of max strain)</td>
<td>-0.16 (% of max strain)</td>
</tr>
</tbody>
</table>

Table 1. Statistical analysis of unloaded force sensor

The standard deviation of the strain readings was extremely low, giving a favorable confidence interval. While there was some drift in the sensor readings, it was slow and relatively low compared to the maximum expected values. With
proper nulling and length of force reading time intervals the sensor could provide consistent readings.

The response of the sensor was also observed under loading. Strain readings were taken before and after aerodynamic loading to observe how long it took the readings to approach a steady state. The results for each force component can be seen in Figure 30.
While the X and M components of the sensor appeared to reach a steady state value within 5 seconds, the Y component exhibited some drift. It appeared...
that the sensor overshot the steady state value and then returned asymptotically. The behavior could have been caused by a poor strain gage attachment but the exact source of this drift is unclear. Measuring strain immediately upon loading would clearly introduce error, so it was necessary to only take data after a certain time under load. If the sensor was allowed to be loaded for 30 seconds before taking data, the strain in that region had a standard deviation of just $1.63 \cdot 10^{-7}$ (1.7% of the mean). This gave a 95% confidence interval of $\pm 0.16\%$ which appeared accurate enough for the measurement of aerodynamic forces.

Hysteresis was tested by sequentially loading and unloading the airfoil with both weights and aerodynamic forces. It was found that the sensor exhibited some error in force measurements between uploading and downloading. In order to avoid these effects, the sensor was unloaded before each new force measurement. This removed all significant hysteresis and resulted in extremely linear relationships (each with a correlation coefficient greater than 0.99) between applied load and strain. Examples of the loading relationships for each force component can be seen in Figure 31.
3.4.5 Calibration

For any strain gage force sensor calibration, the goal is to describe the relationship between the applied forces and strain gage voltage readings as accurately as possible. The final force sensor design needed to be capable of measuring three forces and therefore required a three-component calibration technique. Since loading on a given axis was capable of affecting the readings on the other two axes, the calibration method needed to account for this dependency or “crosstalk.”

A series of sample measurements were taken by subjecting the sensor to known combined loads. To account for the effects of drift and hysteresis as described in section 3.4.4, the following loading sequence was used to get repeatable results:
1. Wait 30 seconds with sensor unloaded.
2. Measure strain readings for 30 seconds and average to get null value.
3. Apply combined load.
4. Wait 30 seconds with sensor loaded.
5. Measure strain readings for 30 seconds and average to get loaded value.
6. Subtract null value from loaded value to get actual strain value.
7. Unload sensor.

The combined loads were chosen based on the minimum and maximum aerodynamic loads expected for the airfoil. When graphed, the relationships between the strain readings of different force components appeared both linear and repeatable with respect to each other. Examples of the strain gage loading sequences can be seen in Figure 32. The graph shows the linear relationships between the X and Y strains, where each slope corresponds to sequential loading at different locations of the airfoil.

![Example of Strain Relationships During Calibration](image)

**Figure 32. Examples of strain gage loading relationships**
The linearity and repeatability of the strain relationships meant that a calibration approach could then be chosen. If the applied loads are combined into a vector $H$ and the corresponding voltage ratios are organized into a vector $R$, the relationship between the two matrices can be described by a calibration matrix $C$. Different calibration models don’t always express this relationship in the same way and therefore have different ways of calculating the calibration matrix.

Three separate calibration models are described by Leung and Link (1999) and their accuracies are compared. First, second, and third order versions of each model are presented as well. Since each model is found to have similar accuracies, the following model was chosen based on the ease of taking calibration points and its success in other papers:

$$\{R\} = [C]\{H\}$$ (7)

Since the strain readings for a given loading sequence were linear, a linear calibration model of the form in Eq. 7 was chosen. To determine the coefficients of the calibration matrix, a least-squares regression model first developed by Ramaswamy et al (1987) was used. The method involves finding the minimum of the sum of the squares of the differences between the actual strain gage output and the calibrated outputs. In other words:

$$e_1 = \sum_{p=1}^{N} \left[ C_{13}H_{1,p} + C_{1,2}H_{2,p} + C_{1,3}H_{3,p} - R_{1,p}\right]^2$$ (8)

$$e_2 = \sum_{p=1}^{N} \left[ C_{23}H_{1,p} + C_{2,2}H_{2,p} + C_{2,3}H_{3,p} - R_{2,p}\right]^2$$ (9)

$$e_3 = \sum_{p=1}^{N} \left[ C_{33}H_{1,p} + C_{3,2}H_{2,p} + C_{3,3}H_{3,p} - R_{3,p}\right]^2$$ (10)
In Eq. 8-10, $e_1$, $e_2$, and $e_3$ are the sums of squares of the lift, drag, and pitching moment respectively. The term $N$ is the number of calibration points and $P$ is the summation index. The first subscript for each $H$ and $R$ corresponds to a specific component of the sensor. For example, the term $H_{3,p}$ corresponds to the voltage reading of the pitching moment component for the $P^{th}$ calibration point.

If Eq. 8-10 are partially differentiated and equated to zero, the resulting equations can be organized into the following relationship:

$$[E][C]^T = [A] \tag{11}$$

Each matrix in Eq. 11 can be defined as follows:

$$[C]^T = \begin{bmatrix} C_{1,1} & C_{2,1} & C_{3,1} \\ C_{1,2} & C_{2,2} & C_{3,2} \\ C_{1,3} & C_{2,3} & C_{3,3} \end{bmatrix} \tag{12}$$

$$[E] = \begin{bmatrix} \sum H_1H_1 & \sum H_1H_2 & \sum H_1H_3 \\ \sum H_2H_1 & \sum H_2H_2 & \sum H_2H_3 \\ \sum H_3H_1 & \sum H_3H_2 & \sum H_3H_3 \end{bmatrix} \tag{13}$$

$$[A] = \begin{bmatrix} \sum H_1R_1 & \sum H_1R_2 & \sum H_1R_3 \\ \sum H_2R_1 & \sum H_2R_2 & \sum H_2R_3 \\ \sum H_3R_1 & \sum H_3R_2 & \sum H_3R_3 \end{bmatrix} \tag{14}$$

Each summation in $E$ and $A$ is taken over the entire set of calibration points (the bounds and calibration point index have been dropped for simplicity). After calculating the $E$ and $A$ matrices, the calibration matrix $C$ can finally be calculated:

$$[C] = ([E]^{-1}[A])^T \tag{15}$$
An 80-point calibration was conducted by hanging weights of varying mass from the wing. The weights were hung from pins located a known distance from the sting so that known torques could be applied and measured. After each of the 80 forces was applied and the corresponding strains recorded, the calibration matrix \( \mathbf{C} \) could be calculated:

\[
\mathbf{C} = 10^{-4} \begin{bmatrix}
0.02789 & 0.0003357 & -0.04312 \\
-0.003311 & 0.02394 & -0.1419 \\
0.0004316 & -0.004217 & 0.7235
\end{bmatrix}
\]

(16)

3.4.6 Evaluation of Final Sensor

After calibration, the sensor was tested for accuracy using standard error calculations. Standard error is a common way of assessing calibration accuracy and is defined by Eq. 17:

\[
se_i = \sqrt{\frac{\sum_{p=1}^{N} (H_{i,p} - \tilde{H}_{i,p})^2}{N - f}}
\]

(17)

The standard error \( se_i \) for the \( i \)th force measuring component of the sensor is a function of the applied normal force \( H_{i,p} \), the resulting force estimated by the sensor \( \tilde{H}_{i,p} \), the number of points used in the calibration \( N \), and the number of degrees of freedom \( f \) (which is three in this case). The standard error could be calculated for each of the three force components and given as a percent of the maximum load in that direction.

The standard errors were first calculated on the day of the calibration. The sensor was then reevaluated approximately two months later to ensure the
calibration was still effective. On both occasions, the accuracy was well within the required accuracy of the sensor design. The calculated standard errors can be seen in Table 2.

<table>
<thead>
<tr>
<th>Force Component</th>
<th>Standard Error (% of maximum design load)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Day of Calibration (12/8/11)</td>
</tr>
<tr>
<td>X</td>
<td>0.25</td>
</tr>
<tr>
<td>Y</td>
<td>0.32</td>
</tr>
<tr>
<td>Torsional</td>
<td>0.99</td>
</tr>
</tbody>
</table>

Table 2. Standard errors for X, Y, and Torsional components

3.4.7 Aerodynamic Force Measurement

Aerodynamic forces were measured using the same strategy described in section 3.4.5. The strain values of the unloaded wing were measured in the tunnel for 30 seconds at 10 Hz (the fastest available rate using the “Best 60 Hz Rejection” mode for the NI 9219 module). This provided 300 samples that were then averaged to create a “null value.” The tunnel was then turned on, the force sensor readings were allowed 30 seconds to settle, and then 300 more strain values were taken under load over 30 seconds. These were averaged and then converted to a force once the null values were subtracted.

The lift forces were computed by taking the vertical components of the X and Y readings from the sensor (depending on what angle of attack it was at) while drag forces were computed from the horizontal components. Moments could be computed as is. The result was a lift, drag, and pitching moment reading for a given angle of attack, each of which was comprised of an average of 300 strain readings. Based on the uncertainty calculated by the standard error equation (Eq. 17), the sensor should only contribute an error (in % of the
maximum design load) of 0.4%, 0.45%, and 1.22% for the X, Y, and M components respectively.

3.5 Wind Tunnel

All experiments for this project were conducted in Tufts University’s closed-loop wind tunnel. In the following sections describe the design of the tunnel as well as an evaluation of the flow within the test section.

3.5.1 Design

Experiments were conducted in Tufts University’s closed-loop wind tunnel. The free stream air was accelerated by a Chicago Blower Wind Tunnel Fan powered by a three-phase, 3490 RPM Baldor motor. The motor speed controlled by a Baldor VS1SP Inverter Drive that could be operated locally or remotely.

The test section of the wind tunnel measured 14 inches x 14 inches in cross section and 40 inches in length. The free stream was capable of speeds up to 23 m/s. The tunnel had a contraction ratio of 6.6 and a resistance temperature detector (RTD) located at the beginning of the test section to provide instantaneous temperature readings. A picture of the tunnel can be seen in Figure 33.
3.5.2 Characterization of the Tunnel

The particle image velocimetry (PIV) system was used to measure turbulence intensity in free-stream flow. Turbulence intensity $I$ is defined as the ratio between the root-mean square fluctuations ($u'$) and the mean velocity of the flow ($U$):

$$I = \frac{u'}{U} \quad (18)$$

Taking an average of 1800 image pairs at 23 m/s, the turbulence intensity was calculated to be in the range of 2%. The limited resolution of the PIV testing made this number more of an estimate than an exact percentage. While a
A turbulence intensity of less than 1% is desirable for a laminar flow test section, 2% was acceptable for our purposes.

The velocity distribution in the test section can be seen in Figure 34. The confidence interval of each point was +/- 0.026 m/s. The span-wise plots matched within this uncertainty, and if the air speed is assumed to follow a normal distribution, the mean free stream velocity was calculated to be 12.8 +/- 0.0063 m/s. It is important to note that the best-fit line shows a gradient of 0.005 m/s/in with a correlation coefficient of 0.75. While it seems there is certainly some kind of gradient, the velocity only changes by 0.07 m/s across the test section and falls within the uncertainty of the velocity measurements.

![Pitot Tube Velocity Profile](image)

**Figure 34. Measured test section velocity distribution**

The accuracy of the pitot tube pressure measurements was verified by measuring the pressure drop through the contraction section. This measurement
was then compared to u-tube manometer readings at the same locations. The results matched seemed to match exactly, though a 2.5% uncertainty was allowed due to the low resolution of the manometer.

To further verify the velocity distribution in the test section, PIV testing was conducted at the same location as the pitot probe. The velocity profile is shown in Figure 35 and has a mean value of 13.12 +/- 0.094 m/s.

![PIV Velocity Profile](image)

**Figure 35. PIV measurements of free stream velocity**

The mean of the velocity profile differs from the pitot tube measurements by 2.5%. Since the accuracy of the pitot tube sensor was confirmed by the manometer readings, the pitot tube measured mean was used in calculating lift, drag, and moment coefficients. Both free stream measurements vary by less than 1% of the mean. This uniformity is acceptable for this experiment, though the
differences between the two measurement techniques may need to be investigated if more quantitative PIV data was required.

3.6 National Instruments LabVIEW Program for Measuring Aerodynamic Forces

A National Instruments LabVIEW VI was written to measure the lift and drag on the airfoil at different angles of attack. The program was interfaced with the rotary table, force sensor, wind tunnel motor, and temperature sensor so that each alpha sweep could be run completely autonomously.

The program communicated with the rotary table via the StepperBoard motor controller using Virtual Instrument Standard Architecture (VISA) commands. Strain readings sampled by a National Instruments Measurements and Automation (MAX) task that took readings through the National Instruments 9219 module. The temperature was also read by a MAX task using the National Instruments 9217 module. The wind tunnel was turned on and off using ActiveX controls that were integrated with the Baldor motor controller. Figure 36 gives an abstract representation of this program and its components.
For each angle of attack the program automatically measured strain values in accordance with the loading strategy described in section 3.4.5. After a 30 second wait, the tunnel was turned on for 30 seconds to allow for the strain values to settle. Strain measurements were averaged over 30 seconds and converted to forces. The tunnel was then turned off and the wing was rotated to the next angle of attack. Once measurements were taken at angles up to 20 degrees the program saved them to a spreadsheet file.

3.7 Wing Design

3.7.1 Airfoil design and fabrication

Birds typically fly at low Reynolds number in the range of $10^5$ to $10^6$, though in gliding flight Reynolds numbers are typically closer to $10^4$ to $10^5$. Therefore, it was necessary to find a low Reynolds number, high lift airfoil that
could be augmented with feather-like attachments. After reviewing a variety of airfoils used for gliders and UAV applications, the Selig S1223 airfoil was chosen due to its similar shape to other bird wings as seen in Figure 37.

![Figure 37. S1223 airfoil compared to a Merganser and Seagull wing profile](image)

The chord length of the airfoil was chosen based on the size of wind tunnel test section. When measuring aerodynamic forces at relatively high angles of attack (such as during separation), it is important to avoid airfoil sizes that create blockage in the free stream flow. This can lead to false pressure gradients and large errors in measured aerodynamic forces in a phenomenon called “solid blocking.” In order to avoid this, the chord length was chosen to be less than 70% of test section height. At these scales, wall effects on the distribution of lift may be neglected.
Coordinates for the S1223 airfoil were obtained from the University of Illinois at Urbana-Champaign Airfoil Coordinates Database and imported into SolidWorks. The airfoil was scaled according to blockage calculations and designed to be detachable into two sections. A section of the starboard side of the airfoil was removed and replaced by a machined aluminum attachment. This allowed the airfoil to be attached to the three-component force sensor using a set screw. The trailing edge of the airfoil needed to be rounded off due to limitations in the resolution of the rapid prototyper. The final chord length was therefore 5.19 inches with a span of 13.75 inches. The airfoil design can be seen in Figure 38.

![Figure 38. Detachable S1223 airfoil design](image)

The wing was fabricated using a Stratasys Fortus 360mc 3D Production System. The resolution of the printing process produced small ridges along the surface of the airfoil, so it was necessary to sand the surface extensively to get
appropriate smoothness. Small discontinuities on the airfoil surface were filled in with Bondo Body Filler and sanded smooth. The surface was sanded with successively finer-grained sand paper and spray-painted with Rust-Oleum matte black spray paint. The paint allowed for an even smoother finish while providing a minimally reflective surface for PIV testing.

3.7.2 Flap design and fabrication

Feather-like flaps were fabricated using 0.003 inch-thick plastic that was spray painted black to minimize laser reflection during PIV testing. The plastic was chosen because it was lightweight enough to pop up in the separated regime and flexible enough to create some pocketing effects. The dimensions of the flaps are shown in Figure 39 and were based on the optimal design found by Schatz et al (2004). Note that the flap is divided into three sections. The flap was taped to the wing using Scotch Painter’s Tape for ease of modification and removal. The trailing edge of the flap was attached 1 inch from the back of the airfoil (approximately 80% along the chord).

![Figure 39. Final flap parameters (all dimensions in inches)](image)

The holes in the flap were created to equalize the pressure differential between the upstream and downstream side of the flap. For a non-porous flap
located towards the back of a wing, the pressure acting on the surface of the flap is less than that underneath due the adverse pressure gradient at that location of the airfoil. This flow condition can be seen in Figure 40, with the flap shown in red and the pressure on the top and bottom of the flap signified by P1 and P2 respectively.

![Diagram of pressure gradients around a non-porous aerodynamic flap](image)

**Figure 40. Pressure gradients around a non-porous aerodynamic flap**

The difference in pressures will cause the flap to stand up under normal flow conditions and act as a spoiler, increasing drag and decreasing lift. By adding holes to the flaps, high-pressure air from the underside is allowed to travel to the low pressure area on top and equalize the pressure differential. This characteristic can be seen on actual bird wings, as the feathers have a certain amount of transmissivity that allows air to flow through under a pressure gradient.
3.8 Overall Errors

With the combination of so many experimental components, it was important to be able to estimate the overall error to be expected when measuring aerodynamic forces. The error was calculated based on each component’s effect on measurements and was given in percentage of the maximum expected design load.

The angle errors from the rotary table and wing deflection were shown to be negligible. The standard error from the force transducer was a maximum of 1.3%. The velocity estimates from the pitot tube, given the uncertainty in probe angle and confidence interval of measurements, result in a 0.4% error. The maximum overall error for the system was therefore estimated to be 1.7% of the maximum load.

Chapter 4. Results and Discussion

4.1 Flow Visualization

The passive flap was taped to the airfoil as described in section 3.7.2. The free stream velocity was set to 10 m/s giving a Reynolds number of 85,000. To observe the effects of the flaps on separation, the airfoil was tested at angles of attacks ranging from 8 degrees through 20 degrees.

At each angle of attack the airfoil was tested without a flap, with the flap taped down, and with the flap allowed to pop up. These three configurations were chosen to ensure that simple attachment of flaps to the wing (as opposed to their passive “pocketing” effects) wasn’t altering the flow field significantly. The camera was aligned at the flap location to capture the flow field at and around the
flap. For each configuration, 10 image pairs were taken at each angle of attack and averaged. Figure 41 shows the PIV results for angles of attack 7 degrees through 12 degrees (with velocities colored coded by magnitude).
Figure 41. Examples of PIV results at different angles of attack (with approximate flap location shown in white)

In the above images, the separated region can be clearly identified in blue. The velocities are slow and create circulation regions at the trailing edge of the wing. The flow outside the separation region can be seen in green and has a magnitude ranging between 10 and 14 m/s. This is close to the free stream velocity and is higher in some cases due to the acceleration of the air over the wing surface.

The effect of the flaps is apparent by comparing the flow fields of each configuration. When the flap is held down the flow field looks nearly identical to that of the bare wing. However, when the flap is allowed to pop up due to the reverse flow, it creates another distinct circulation region upstream of it. This flow field is comparable to that described by Knacke (2003) as seen in Figure 13 and appears to have an effect on the flow outside of the separation region.

For the “flap up” configurations, the flow hugs the wing and curves downward significantly more than that of the “flap down” or “no flap” configurations. It appears that the flaps are indeed creating low pressure regions that draw the air down and increase curvature of the streamlines. A comparison of the flow field curvatures for each angle of attack (traced along the transition between the green and blue flow areas of the flow fields) can be seen in Figure 42.
9 Degrees

- No Feathers
- Feathers Down
- Feathers Up

10 Degrees

- No Feathers
- Feathers Down
- Feathers Up

11 Degrees

- No Feathers
- Feathers Down
- Feathers Up

Figure 42. Comparison of PIV free stream flow fields at different angles of attack

A larger curvature results in higher pressure gradients, suggesting the flaps are creating more lift than the bare wing at these angles of attack.

4.2 Lift, Drag, and Pitching Moment Data vs. Bare Wing

The lift curve of the bare S1223 airfoil was measured first to validate the force sensor’s measurement of aerodynamic forces. The wing was placed in the wind tunnel at a Reynolds Number of $10^5$ and rotated to angles of attack of 0
through 20 degrees. Five data points were taken at each angle of attack as described in section 3.4.7 and averaged. The lift curve was then compared to a various other sources as shown in Figure 43.

![Graph showing lift coefficient comparison](image)

Figure 43. Measured S1223 lift curve compared to other sources

The measured lift coefficients matched those from other sources reasonably well. There is some discrepancy in the magnitudes but separation appears to occur at an angle of attack between 6 and 8 degrees for each curve. The shape of the curves is also very similar, with a steep drop at separation followed by a slow climb back up.

The differences in magnitudes between each curve could have a variety of causes. The Airfoil Investigation Database (2012) data, for example, is computational therefore is not susceptible to variables such as tunnel effects or inconsistencies in flow. In addition, the surface roughness of airfoils at such low
Reynolds numbers can have large effects on aerodynamic performance. The different surface conditions between the fabricated S1223 airfoil and those made by Eldwin and Shaw (2002) or UIUC Applied Aerodynamics Group (2001) could certainly make a difference. Finally, the accuracy of the model S1223 profile may also play a role. With such a sharp trailing edge, the profile may need to be modified depending on the limitations of the material or fabrication strategy. The S1223 airfoil designed for these experiments required the removal of approximately 0.35 inches from the tail to meet the resolution of the rapid prototyper. This type of modification could have an effect on flow field as well as the effective angle of attack of the wing.

With all these variables, the measured lift coefficients nonetheless deviated by 20% at the most from other experimental data. The accuracy of the aerodynamic measurements was therefore deemed acceptable for comparing the lift of different flap configurations.

The final flap configuration was positioned on the wing and divided into three parts. Lift, drag, and moment coefficients were measured with the flaps held down and the flaps allowed to pop up on their own. The resulting curves, with error bars indicating the 95% confidence interval, are shown in Figure 44.
Effects of Flap on Lift Coefficient

(A)

Effects of Flap on Drag Coefficient

(B)
The lift curve shows the airfoil with a free flap doing visibly better than the other two configurations after separation. The shape agrees with research by others such as Wang and Schlüter (2012) as seen in Figure 17. The lift coefficient is improved by as much 6.4% in the separated region and remains high until an angle of attack of 20 degrees is reached. The drag in the separated region is also reduced as seen in Figure 44B with improvements of as much as 13.2% in some cases. The full savings of the feather-like flaps are illustrated in Figure 45, where the lift coefficient vs. drag coefficient is clearly favorable in the separated region.
Figure 45. $C_L$ vs. $C_D$ for different flap configurations

The pitching moment also shows substantial improvement in the separated region. A maximum decrease of 8.3% can be seen and it remains low even through an angle of attack of 20 degrees.

It is important to note that the attachment of flaps to the airfoil at angles of attack prior to separation seems detrimental to both lift and drag. This is almost certainly due to the way the flaps alter the surface as they are taped down. They aren’t perfectly flat (especially after being blown about at high angles of attack) and introduce a non-uniformity to airfoil profile. This can cause minor separation on the wing downstream of the flaps (which was observed in some cases with tufts flow visualization) leading to less lift and more drag.
4.3 Flap Optimization.

Once the flaps were shown to provide certain aerodynamic advantages, the effects of dividing them up were investigated. While the original design calls for a division of three flaps across the span, more or less span-wise divisions could result in better performance.

Maintaining all other original dimensions, a single flap was tested in the separated region along with flaps divided into 5, 10, and 20 smaller flaps. Five data points were taken for each angle of attack and averaged. The results are shown in Figure 46 along with error bars indicating the 95% confidence interval. Note the bare wing lift curve is shown for reference.

![Lift Coefficients for Different Feather Configurations](image)

**Figure 46. Lift coefficients of flaps divided in the span-wise direction**

While the flap-augmented wing clearly does better than the bare wing in the separated region, it is unclear which does better. The confidence intervals
make it impossible to pick an optimized design since the means themselves only vary by a few percent between configurations.

One would expect there to be measurable differences between the single flap design and multiple flap design. A look at the feather distribution of a seagull, for instance, shows nearly twenty feathers across the span of the wing. On a bird, these feathers act in pockets in response to separation at different points on the wing.

A single flap, on the other hand, pops up across the entire airfoil even if there is only a small region of separation. This affects attached flow as well as separated flow, acting as a spoiler in some regions and a feather-like flap in others. Dividing the flap up allows portions of the flap to pop up only when they need to and maintain maximum lift across the entire span. This difference in flap response was observed in wind tunnel testing and can be seen in Figure 47.
Figure 47. A) Single flap configuration at 20 degrees B) 10 flap configuration at 20 degrees
One reason why these two responses, as different as they are, could produce similar lift forces could be because of the choice of material for the flaps. The thin plastic was flexible and could bend even in the single flap configuration. While the flap does pop up over a large percentage of the span, parts are able to stay relatively low as seen in Figure 47A. It is possible that the single flap was therefore able to act in a somewhat localized way and achieve similar lift coefficient to the multiple flap designs.

Chapter 5. Conclusion and Recommendations

This study was focused on the design and optimization of a passive flow control device based on bird feather characteristics. The study aimed to investigate the aerodynamic effects of placing feather-like flaps on a low Reynolds number airfoil and show an optimum way of dividing them in the span-wise direction. A closed-loop wind tunnel was used to apply aerodynamic loads to an S1223 airfoil rotated at different angles of attack. A novel three-component force sensor was designed and fabricated to measure aerodynamic forces while PIV data was taken for flow visualization. Passive flaps fabricated from thin plastic were attached to the surface of the wing using tape. The lift, drag, and pitching moment data from the force sensor as well as the flow field from the PIV testing were compared for the bare wing and the wing with feather-like flaps. Experiments were done for a single flap as well as flaps divided into 5, 10, and 20 span-wise parts.

Attaching flaps to the airfoil showed an increase in lift by as much as 6.4%, decrease in drag by 13.2%, and a decrease in pitching moment by 8.3% in
the separated region. The alpha sweeps had similar shapes to those from previous works and appeared to exhibit similar flow fields described by Knacke (2003) in the PIV results. There were, however, some detrimental results observed at the angles of attack before separation. A slight decrease in lift and increase in drag suggest that the flaps disrupted attached airflow over the surface of the wing.

Though this is almost certainly a problem with this flap design, creating a small recessed space in the airfoil to attach the flap could easily solve this problem. That way the flap itself wouldn’t change the airfoil surface significantly and lift at low angles of attack would be maintained. Ideally, the flaps would also be slightly curved and tapered to match the surface of the airfoil. A more resilient flap material, such as the carbon-fiber strips used by Wang and Schlüter (2012), could also help limit these detrimental effects by minimizing prolonged deformation caused by the reverse flow.

A comparison of different divisions of flaps showed that configurations did similarly well when divided into 5, 10, and 20 parts. While the resolution of the force sensor made it impossible to determine an optimal design, it was interesting to note how close the designs were in lift coefficient improvement. To improve the force sensor, some of the drift reported in section 3.4.4 could possibly be removed by reattaching strain gages and resoldering connections. In addition, a more accurate placement of strain gages could help the accuracy in some cases. If all else fails, the spring element of the force sensor could be redesigned to induce higher strains at each strain gage location.
The flap optimization could also be improved by simply taking more data. A more representative population could help decrease the error bars and provide a better idea of how different number of flaps affect the lift and drag. With more time this could have provided more convincing results.

The results shown in this study as well as those conducted by other aerodynamic researchers create a strong argument that this technology should be studied further. If the design of these passive flaps could be optimized further and fabricated in a robust way, they could be useful in a variety of industrial settings. Even augmenting aerodynamic performance by only a few percent could lead to significant savings for large-scale applications.
References


