Design and Development of a High-Altitude, In-Flight-Deployable Micro-UAV

by

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B.S. Aerospace Engineering, 2010
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Submitted to the Department of Aeronautics and Astronautics on May 24, 2012 in partial fulfillment of the requirements for the degree of Master of Science in Aeronautics and Astronautics at the Massachusetts Institute of Technology May 2012

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Abstract

A micro-UAV (μUAV) system was developed to provide maximum endurance for a small atmospheric sensing payload. The system, composed of a μUAV and protective case, folds and fits into a MJU-10/B flare cartridge (7.1” × 2.4” × 1.9”) and is designed to be ejected in-flight from altitudes up to 30,000 ft at 300 G, to open and unfold in freefall, and to autonomous fly, sense, and transmit data for up to 45 minutes at maximum altitude. The μUAV has a wingspan of 11.8”, a length of 6.6”, and a mass of 220 grams. Guided by first-principles, a series of design studies are conducted to maximize the airframe performance. The μUAV is refined through computational analysis, prototyping, and a multi-phase testing program involving wind tunnel, structural shock, and deployment tests. A series of airfoils was developed for the low Reynolds numbers in which the wings operate (between 30,000 and 80,000) and for manufacturing considerations. Detailed design of aircraft components is presented with a discussion of small-scale composites manufacturing processes. Folding and control mechanisms were developed to actuate control surfaces on a swinging wing. The resulting design carefully balances low Reynolds number aerodynamic effects, small-scale composite structures, and manufacturing capabilities in a configuration that offers unprecedented endurance (for aircraft of this size and altitude) in a widely-compatible package with mission-reconfigurable payload.
Acknowledgements

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I would also like to acknowledge the contributions by the members of the fall 2010 16.82 and spring 2011 16.821 classes for their input.

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# Nomenclature

## Letters

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<tr>
<td>$AR$</td>
<td>aspect ratio, $b^2/S$</td>
</tr>
<tr>
<td>$b$</td>
<td>wingspan</td>
</tr>
<tr>
<td>$c$</td>
<td>chord</td>
</tr>
<tr>
<td>$C_d$</td>
<td>2-D profile drag coefficient</td>
</tr>
<tr>
<td>$C_D$</td>
<td>3-D drag coefficient</td>
</tr>
<tr>
<td>$CDA$</td>
<td>equivalent drag area; $CDA = S_{ref}C_D$ at $C_D = 1.0$</td>
</tr>
<tr>
<td>$C_l$</td>
<td>2-D lift coefficient</td>
</tr>
<tr>
<td>$C_l/C_d$</td>
<td>2-D airfoil lift-to-drag ratio</td>
</tr>
<tr>
<td>$C_L$</td>
<td>3-D lift coefficient</td>
</tr>
<tr>
<td>$C_m$</td>
<td>pitching moment referenced to the chord of the main lifting wing, positive nose-up</td>
</tr>
<tr>
<td>$D$</td>
<td>drag</td>
</tr>
<tr>
<td>$D^*$</td>
<td>displacement thickness</td>
</tr>
<tr>
<td>$\bar{E}$</td>
<td>specific energy of battery</td>
</tr>
<tr>
<td>$H$ or $H_k$</td>
<td>shape parameter</td>
</tr>
<tr>
<td>$L/D$</td>
<td>3-D lift-to-drag ratio</td>
</tr>
<tr>
<td>$N_{crit}$</td>
<td>critical boundary layer instability amplification ratio</td>
</tr>
<tr>
<td>$Re$</td>
<td>chord Reynolds number</td>
</tr>
<tr>
<td>$q$</td>
<td>dynamic pressure</td>
</tr>
<tr>
<td>$W$</td>
<td>weight</td>
</tr>
<tr>
<td>$V$</td>
<td>air speed</td>
</tr>
<tr>
<td>$S$</td>
<td>area of main lifting or front wing</td>
</tr>
<tr>
<td>$S_t$</td>
<td>area of tail or rear wing</td>
</tr>
<tr>
<td>$SM$</td>
<td>stability margin, as normalized to the front wing chord</td>
</tr>
<tr>
<td>$t$</td>
<td>time in seconds</td>
</tr>
<tr>
<td>$t/c$</td>
<td>airfoil thickness ratio</td>
</tr>
<tr>
<td>$x/c$</td>
<td>normalized airfoil chordwise coordinate</td>
</tr>
<tr>
<td>$XCG$</td>
<td>location of the center of gravity</td>
</tr>
<tr>
<td>$XNP$</td>
<td>location of the aircraft neutral point</td>
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**Abbreviations**

CG center of gravity
MFCG “maximum-forward CG”, the most forward-possible CG location with a given external airframe geometry, found by placing all components as forward as possible
MRCG “maximum-rearward CG”, the most rearward CG allowable given a set of stability constraints
OD outer diameter
SRDS sprung rotary drive system, used to control the elevons
UAV unmanned aerial vehicle
µUAV micro-unmanned aerial vehicle, the topic of this paper
#X aircraft #-scale size

**Units**

A amps
Ahr amp-hours
°C degrees Celsius
C battery charge or discharge rate, as a multiplier of 1/hr to the cell's capacity
ft feet
G acceleration as a multiplier of Earth's gravitational constant, \(9.81 \frac{m}{s^2}\)
in inches (”)
J joules
KTAS knots true air speed
lbf pounds force
m meters
mAh milliamp-hours
mm millimeters
N newtons
s seconds
V volts
Whr watt-hours

**Symbols**

\(\alpha\) angle of attack
\(\delta_e\) elevator deflection, positive trailing-edge down.
\(\eta_X\) energetic efficiency of component X
\(\theta\) momentum thickness
\(\rho\) air density
Chapter 1: Introduction

In the past 10 years, the development in lithium-polymer batteries and miniaturization of electronics hardware has led to the popularization of electric remote-controlled aircraft and the development of many small unmanned aerial vehicles (UAVs) for recreational, research, and military use. Small UAVs provide interesting and powerful capabilities at low costs, but as designers continue to design UAVs to ever smaller envelopes and Reynolds numbers, they move into a design regime in which traditional design assumptions break down, and great care must be taken to balance design elements across multiple disciplines to ensure the creation of an efficient system.

This thesis presents the design, development, production, and testing of a micro-UAV (henceforth referred to simply as the μUAV) system in just such a design space, characterized by the juncture of a highly-constrained envelope, low-Reynolds numbers, and small scale composite structures.

The μUAV is designed to provide persistent atmospheric sampling capability at high altitudes (up to 30,000 ft) and deploys from a countermeasure flare cartridge (less than 1.9" x 2.4" x 7.1"). The flare cartridge ejects the μUAV package with an acceleration of 300G. The μUAV deploys from its protective case mid-freefall and flies for up to approximately one hour before descent and disposal. This project started as a design problem in the fall 2010 Flight System Engineering class, for which the author was the teaching assistant. At the conclusion of the class, the design effort was continued as the work presented in this thesis.

The μUAV was developed through a first-principles-based configuration trade-space analysis combined with the construction and testing of many flight and manufacturing prototypes. The result is a design that achieves unprecedented performance by placing and using system components to mutual advantage. As best known to the author, this problem is novel and has never been resolved in the way described here within.

Much of the work presented in this document is the product of multiple iteration cycles. To create a semblance of order, the paper will forgo chronology in favor of thematic consistency.

Chapter 1 presents the mission background and an analysis and synthesis of detailed requirements.

Chapter 2 presents the main features of the μUAV, the protective case. Its purpose is to set the context for the following chapters, which describe the development and detailed design of the system.

Chapter 3 presents the exploration of the configuration trade space and the analysis that created the external lines of the aircraft wings and fuselage.
Chapter 4 presents aerodynamic analysis and validation for the design created in Chapter 3. Wind tunnel tests, the development of a set of airfoils for the μUAV, and a second series of wind tunnel tests for the revised airframe are presented.

Chapter 5 presents the detailed design of structures and mechanisms as well as construction techniques.

Chapter 6 presents the design of the protective case and an ejection simulation.

Chapter 7 presents the system integration and testing efforts and results.

Chapter 8 presents the conclusion.
1.1. Mission Overview

The μUAV is designed for a baseline mission that aims to study the atmosphere via multi-point real-time in-situ sampling. To accomplish this, a carrier aircraft ("mothership") ejects multiple μUAVs (via the MJU-10/B flare cartridge) which would stay aloft as long as possible while collecting data in a predefined flight path. The μUAV must also be able to station-keep the winds-aloft. The mission concept is shown in Figure 1 below.

The data collected is transmitted wirelessly in real-time to either the mothership or a receiving ground station. Multiple μUAVs are used to span a greater flight area, to provide longer persistence, and for simultaneous multipoint data collection. The μUAVs are expendable and not recovered after use.

The baseline payload for the aircraft is an atmospheric sensing package which will measure temperature, pressure, and relative humidity at the μUAV’s GPS-located position. The package weighs approximately 5 grams. Aside from the baseline mission, a secondary goal is to design the μUAV to be compatible with other, larger payloads.

Figure 1: μUAV high-altitude mission [1]
1.1.1. Concept of Operations

A concept of operations was developed that splits the atmospheric sampling mission into seven phases, as illustrated in Figure 2 and described below.

Phase 1: Ejection:
The μUAV package is ejected from the mothership by the MJU-10/B flare ejection charge. A switch on the case initiates the deployment timer which will open the case in Phase 3.

Phase 2: Freefall
The μUAV package is oriented vertically by a drag device which slows the package down to its terminal velocity that is designed to coincide with the μUAV's deployment speed as limited by the μUAV's structure.

Phase 3: Deployment
The deployment timer triggers a servomechanism which opens the case from the bottom. The case is spring-loaded and opens to 180° to maximize drag. The case, now shaped like a bucket against the air, pulls away from the lower-drag μUAV, which accelerates downward away from the case. A switch powers on the μUAV.
The μUAV may be tumbling due to the dynamic motion caused by interference with the case during opening. The μUAV’s aerodynamic surfaces deploy, and then the μUAV must dampen out any initial tumble, which leaves the μUAV pointing downwards.

Phase 4: Recovery
The μUAV pulls up from its nose-down orientation, generating the maximum load factor the wings will experience. Using the onboard IMU, the μUAV stabilizes and levels, at which point the motor is activated. The μUAV flies level until the GPS signal is acquired. If a preset maximum safety time has elapsed without GPS acquisition, the μUAV will enter a failsafe mode and enter a stall to drop to the ground.

Phase 5: Data Collection
When the μUAV’s GPS location is found, it then flies in its predetermined pattern. The μUAV samples data using its sensors and broadcasts its data to the receiving station on the ground or aboard the mothership via an omnidirectional antenna. In long-distance missions where the receiving station cannot receive data from all the μUAVs, the data may be “daisy-chain” relayed to the receiving station. If the μUAV loses track of its GPS location or contact from the receiving station, it will enter the failsafe mode.

Phase 6: Descent
Toward the end of the flight, when the battery voltage reaches its motor-cutoff voltage (approximately 2.5V per cell), the motor will be shut off and the μUAV will enter a glide. During this glide phase, the μUAV continues to sample data and transmit it back to the receiving station.

Phase 7: Landing
The μUAV lands and is not recovered after use.

1.1.2. Driving Requirements:
As described in Section 1.1.1 and Section 1.1.2 above, there are four main driving requirements for the μUAV design:

1) Endurance – the goal of the μUAV is to maximize flight endurance.
2) High Altitude – the μUAV must be capable of operating up to an altitude of 30,000 ft.
3) Deployment – the μUAV system must be compatible with the size and ejection forces of the MJU-10/B flare cartridge.
4) Stability – the μUAV must be able to go from an unknown orientation and rate of gyration into a stable flight configuration.
1.2. Requirement Details

1.2.1. High Altitude Environment

Because of the high altitude the uUAV is carried and operated at, it must be designed to operate in cold temperatures. As illustrated by the red circle in Figure 3, the uUAV system must operate at a minimum temperature of from -45°C at 30,000 ft.

![Temperature vs. Altitude (1976 Standard Atmosphere)](image)

Figure 3: Temperature vs. altitude (1976 Standard Atmosphere)

Because the uUAV must at least be able to station-keep in ambient wind conditions, operating at 30,000 ft also introduces a speed capability requirement. Figure 4 below shows the average NOAA data collected above Edwards AFB for December, the month with the highest average wind speed. For the uUAV to remain useful during windy days, it must be capable of a forward flight speed of at least the mean wind speed shown below.

![Wind speed vs. Altitude for Edwards AFB (NOAA data)](image)

Figure 4: Wind speed vs. Altitude for Edwards AFB [2]
A speed requirement of 58 knots (30 m/s) was selected for medium-cruise at 30,000 ft, where the standard air density is 0.459 kg/m³. This cruise speed makes the μUAV 14% faster than the average wind aloft, which allows the μUAV to recover from course deviations while staying within its efficient cruise range.

### 1.2.2. MJU-10/B Flare Cartridge

The μUAV will be stored and ejected from the MJU-10/B flare countermeasure cartridge, which externally measures approximately 2" × 2.5" × 8". This type of flare is commonly used on many types of military aircraft such as the C-17 Globemaster III and F-15 Eagle [3] as well as some research aircraft. A drawing and picture is shown in Figure 5 with the internal structure (not to scale) shown in Figure 6.

![Figure 5: MJU 10/B flare cartridge [4]](image)

![Figure 6: Schematic of flare cartridge system](image)

As shown in Figure 6 above, the system envelope for the μUAV (blue) is encased inside the shell (orange). The end is sealed with a cap (purple), crimped at the edges to keep it locked to the
shell. Ejection of the μUAV package is accomplished by the pyrotechnic cartridge (red) which pushes the μUAV package via the piston mechanism (green).

The forces experienced by the μUAV package during deployment are severe. The compressive force required to open the cap is at minimum 125 lbf, and the acceleration experienced during ejection is approximately 300 G, which yields an exit speed of approximately 55 mph.

The MJU-10/B flare cartridge also sets the maximum system envelope for the μUAV package, as shown in Figure 7 below. The maximum dimensions are 48mm × 62mm × 180mm with a 3.175mm corner fillet.

![Figure 7: Available system envelope](image)

1.2.3. Stability

The mid-air deployment creates a requirement that the aircraft must be passively stable in pitch and yaw so that it is capable of recovering from an initial tumble.
1.3. Requirements Compared to Other UAV Systems

A review was conducted to compare the system requirements of the proposed µUAV to other current small electric UAVs that are either in development or used in the field [5]. The results are plotted below in Figure 8. Data is collected from only publically-available specifications and articles, so this review is not expected to be complete.

![Survey of small electric UAVs: endurance, altitude range, vs weight](image)

Figure 8: Required performance vs. survey of current small UAVs

In Figure 8 above, each circle represents a UAV system and is plotted by weight and typical mission altitude. The size of the circle represents the endurance of the aircraft, and the line protruding from the dot spans the standard altitude range in which the aircraft can operate. The altitude range lines for the AeroVironment Nano Hummingbird and Lockheed Martin Desert Hawk 3 are obscured by their endurance circles.

As shown in Figure 8, the aircraft with the longest endurance, the AeroVironment (AV) Raven [6] and Puma [7][8] and the Lockheed Martin Desert Hawk 3 [9], typically fly at low altitudes to achieve their long endurance performance. Three aircraft, the AV Nano Hummingbird[10], AV Wasp III [11], and Aurora Flight Sciences Skate [12] are within the weight range specified for the µUAV. Two of these UAVs, the AV Switchblade [13] and BAE Systems Coyote [14], have folding tandem configurations. The small and folding UAVs have significantly less endurance than the Raven, Puma, and Desert Hawk.
The requirements for the µUAV system cover a traditionally-unexplored portion of the design space due to the combination of high altitude, small package size and weight, and maximum endurance requirements.
Chapter 2: \( \mu \)UAV System

This chapter describes the major design features of the \( \mu \)UAV to establish the framework for the analysis presented in later chapters.

2.1. Overview

The \( \mu \)UAV system is composed of two components – the aircraft and its protective case, shown together in Figure 9. The streamer attached to the back of the protective case is not shown.

![Figure 9: Aircraft and case stowed (left) and deployed (right)](image)

The case shown in Figure 10 protects the \( \mu \)UAV from the compressive burst-force and the high wind-speed forces at ejection. The rear end (hinge plate) part of the case also provides an attachment point for the 1.5-meter long drag streamer, which slows the \( \mu \)UAV to a safe speed before the wings are opened. The case and aircraft together are referred to as the “\( \mu \)UAV package.” The \( \mu \)UAV package is ejected from the flare cartridge with the \( \mu \)UAV pointed backwards, as shown in Figure 10.
The μUAV has a tandem-wing, pusher-propeller configuration. The external features are shown in Figure 11 below.

Both front and rear wings are of carbon-fiber composite construction and employ a series of custom airfoils designed for low-Reynolds number. Both wings are spring-loaded; the front wings fold backwards and the rear wings fold forwards the stowed configuration. The rear wing houses the elevon control surfaces inboard and has “finlet” vertical surfaces on the tips for lateral stability. To control the μUAV’s pitch, the elevons are deflected in the same direction, and to control its roll, the elevons are deflected in opposite directions. From the region just outboard of the elevons to the start of the finlets, the wing has a section of slight dihedral.

The fuselage is designed for low-drag and to place the CG (center of gravity) forward and is constructed from a Kevlar-composite shell. Two holes are placed near the nose to provide total-
energy and static sensing ports for the autopilot and atmospheric sensing payload. The section
of the fuselage above the front wings is flexible and acts as a fairing.

The internal features of the μUAV are shown in Figure 12.

As shown in Figure 12, the μUAV houses a large battery pack, placed toward the front of the
fuselage to place the CG forward. The batteries arranged in a staggered stack and encased in a
shock-dampening and insulating foam shell. Nine cells, arranged in a 3-series, 3-parallel
configuration provide a nominal capacity of 2100 mAh at 11.1 V. At the right-front corner of
the fuselage is the atmospheric sensing payload. To either side of the battery pack, the front
wing hinge and folding mechanisms are housed. At the rear of the battery pack, inside the foam
shell, is the electronic speed controller (ESC).

The autopilot board, which also houses the GPS, IMU, and communications radio, is mounted
at the rear-top of the fuselage. Below the autopilot board are the motor and propeller shaft, the
control mechanisms, and the rear wing deployment mechanism. The motor is a NeuMotors
Proton 12-30-4000, and the control mechanisms employ two Hitec HS-35HD servos. The
autopilot, motor, and servos are attached to a central assembly constructed from carbon fiber
called the “motor cage” which also reinforces the fuselage shell. The servos drive the elevons
through a variant of the “rotary drive system” or RDS control system used on some remote-
controlled gliders [15]. The switch that powers on the μUAV during deployment is located on
the bottom of the fuselage.

The μUAV weighs 220 grams and is 168 mm (6.6 in) in length with a wingspan of 301 mm (11.8
in).
2.2. Extended Payload

By reducing the number of battery cells from 9 to 6, it is possible to use a different packing configuration to greatly increase the available payload volume to approximately 32 mm × 32 mm × 43 mm.

![Payload volume](image)

Figure 13: Maximum-payload configuration

Reducing the battery size to 67% of the original size decreases the endurance to approximately 63% due to the increased losses in the power system. For this configuration to be viable, the payload needs to be dense enough to place the CG location at the same point as the standard mission.

2.3. Autopilot

The small dimensions of the µUAV fuselage necessitated the development of a custom autopilot system, as no commercial units could provide a processor, GPS, communications radio, and servo controls in a package sufficiently small to fit inside the vehicle while leaving sufficient room for other aircraft components. The hardware and software development is conducted by MIT Lincoln Laboratory and is not covered in detail in this thesis. The current version of the custom autopilot is shown in Figure 14.

![Top and Bottom](image)

Figure 14: MIT LL custom autopilot
2.4. Folding and Deployment

To fit in the protective case, the µUAV's wings fold 90 degrees and align with the fuselage. With the wings folded, the µUAV is referred to be in the "stowed configuration" and with the wings deployed, it is referred to be in the "flight configuration". Both wings are spring-loaded and, without a restraining force, automatically open to the flight configuration.

Stowed, mid-transition, and flight configurations are shown in Figure 15 below. When the aircraft is in the stowed configuration, the fairing on the top deforms and deflects upward so that the wings are stored underneath it.

![Figure 15: µUAV wing folding](image)

The front wings fold with some overlap of the trailing edges, as shown in Figure 16. This design feature is due to the fact that low-Reynolds aerodynamics is best negotiated with thin airfoils, and the trailing edges can be made to be flexible. When the front wings enter the stowed configuration, their trailing edges flex and locally stack, generating 10 mm of overlap. This is the maximum amount of overlap possible without adding significantly to the thickness of the wing stack.

![Figure 16: Front wing overlap and flexure](image)

Since the leading edges instead of the trailing edges of the rear wings meet together, the rear wings cannot have the same overlap feature; instead, the rear wings share a coaxial hinge and are stacked vertically, as shown in Figure 17. Figure 18 shows the swinging motion of the wings.
The right wing is colored blue and the left wing is colored red for clarity. The finlets fold in front of the nose in the stowed configuration.

![Figure 17: Rear wing stacking](image)

When the μUAV's wings are folded, the μUAV fits inside the protective case, sized to fit inside the flare cartridge. A side-view of the stowed μUAV inside the protective case is shown in Figure 19 below (some components hidden for clarity). The available vertical space is completely occupied.

![Figure 18: Rear wing folding](image)

![Figure 19: Stowed configuration, side projection](image)

The protective case is composed of three main physical groups: two sides and a backplate. Figure 20 illustrates the geometry of the case opening.

![Figure 20: Geometry of the case opening](image)
Shown in Figure 20, the backplate, which holds onto the tail end of the μUAV, is also attached to the streamer. The right and left sides are hinged to the backplate and are spring-loaded. The components inside the case are placed to avoid interfering with the μUAV during deployment.

A 3-view of the μUAV and the protective case is shown in the expanded 3-view in Figure 21 below.
Figure 21: Expanded 3-view of μUAV
Chapter 3: Sizing and Configuration Design

This chapter presents the design of the μUAV's wings, fuselage, and basic internal configuration. Designing a maximum-endurance folding aircraft requires the consideration of not only the equations that govern flight energy, power, and stability, but the interactions between the system envelope, the stowed geometry, and the flight geometry. The envelope sets a limit on maximum component size, and the swinging configuration couples the chordwise and spanwise geometries.

The dimensionally-constrained envelope is handled by a four-step process. First, the performance equations are analyzed to optimize the aircraft size and battery configuration for endurance. Second, a continuous configuration-space is mapped by relating the location of possible wing hinge points to the planform areas. Doing so allows determination of the relationship between wing performance and allowable CG location. Third, a general internal packing scheme is developed to place the CG toward the region of the design space that maximizes performance. Fourth, a series of fuselage candidates are studied to generate a space that links CG location, drag, and available wing hinge location. The final design is created from the intersection of the trends of the wing performance and fuselage performance spaces. This design best-combines wing performance and fuselage geometry into a configuration that yields the best total performance.

To derive the performance equations and to design the configuration, it is first necessary to consider the characteristics of electrical propulsion and their effects on aircraft geometry.
3.1. Electric Propulsion Design Considerations

Due to the small system size and cold-start requirement, an electrical propulsion system was selected for the μUAV. Electrical propulsion systems are composed of three main components arrange in Figure 22 below) – the battery, the electronic speed control (ESC), and the motor.

The battery supplies constant DC electrical power to the ESC, which reads the throttle setting provided by the control unit, either an RC receiver or autopilot, in servo encoding. The size of the ESC, motor, and propeller scale directly with the power the system has to output.

Using an electrical propulsion system has several consequences on the way the μUAV must be designed. First, unlike liquid fuel systems, the weight of an electrical propulsion system does not change as energy is depleted. This means that the aircraft must carry the weight of the entire energy system throughout the mission, and increasing energy capacity beyond a certain point becomes detrimental to endurance, as will be discussed in detail in Section 3.2. Another consequence of constant battery weight is that the batteries may be placed strategically to place the center of gravity (CG) of the aircraft, unlike in liquid-fueled aircraft where the fuel tanks need to be placed to avoid unfavorable CG shifts throughout the mission.

Second, the volumetric density of batteries is far less than traditional petroleum-based fuels even accounting for the efficiency of the energy delivery system. Therefore, the relation between fuselage drag to energy capacity is far stronger than in liquid-fueled aircraft.

Third, unlike a liquid fuel, battery voltage, capacity, and dimensions are quantized to specific units produced by battery manufacturers, so the design of the power system requires a discrete study of available batteries and physical configurations.

Fourth, a battery’s useful capacity decreases the faster it is discharged, or can be discharged. There are two components to this correlation. First, in operation, a reduction in useful capacity is due to the battery’s internal resistance – higher current through the battery will cause more energy to be lost as heat via Ohm’s law. Second, designing a higher-current battery reduces the
amount of energy it can hold as there is a trade-off between specific power and specific energy. This relationship can be summed in a Ragone plot, as shown in Figure 23 below.

![Ragone plot showing specific energy vs. specific power](image)

**Figure 23**: Ragone plot showing specific energy vs. specific power [16]

The curves in Figure 23 show that the trade-off between specific energy and specific power is monotonically negative. Using a battery with higher specific power will always entail losing specific energy. As practical examples, batteries created for remote-controlled airplanes or cars sacrifice energy density for higher power density, and batteries in smart phones tend toward the opposite, yielding batteries that perform poorly if drawn at high rates, but have high specific energies at low draw rates. As a result of the internal resistance and the Ragone trend, it is desirable to match the power output of the battery to the requirements of the vehicle, and to not design in any more power capacity as is necessary, such that the highest energy density can be achieved, yielding the greatest flight time.

Figure 23 also shows the reason lithium-ion (or polymer) batteries are the best choice for aircraft performance, as it has the best specific energy and specific power in the region of interest, above the 0.1h oblique line.

The Ener1 Korea SPB463048 lithium polymer battery was selected for the μUAV for its high volumetric and gravimetric energy density, at 391 Whr/L and 192 Whr/kg, and for its dimensions, which allows an efficient packing scheme. The nominal voltage and capacity are 3.7V and 700 mAh, respectively, yielding a nominal capacity of 2.59 Whr. The rated maximum draw rate is 2C, or 30 minutes. [17] A schematic of the battery is presented in Figure 24.
Battery testing was conducted to verify the cell's performance at room temperature. Using a battery tester, the cell was discharged at a constant draw rate of 0.5C to 2C and the voltage was measured until the voltage cutoff was reached at 2.8 V. Data from a typical cell is shown in Figure 25 below.

![Battery drain test results](image-url)

Figure 25: Battery drain test results
The data in Figure 25 shows that at a low draw rate (0.5 C), the battery performance matches the claimed specification, with an average voltage of 3.7 V and an energy capacity of 2.62 Whr. As the draw rate is increased, the energy-available drops linearly, as shown in Figure 26, which plots the energy available per cell against the drain rate in C.

![Energy available per cell vs drain rate](image1)

Figure 26: Energy available per cell vs. Drain rate in C

The energy-available can also be plotted against the discharge power, which produces Figure 27 below.

![Energy vs discharge power](image2)

Figure 27: Energy available vs. discharge power
3.2. Endurance-Performance Relations

In this section, the equations of flight forces and energy are analyzed to synthesize an equation that relates aircraft endurance to its geometric and aerodynamic qualities. By making reasonable assumptions about these qualities, insight is gained into the aircraft configuration design space, and the battery size is selected.

3.2.1. Endurance Equation Formulation

The endurance derivation starts with the governing equations of flight forces and energy. The dynamic pressure (Eq. 3.1), lift (Eq. 3.2) and drag (Eq. 3.3) equations for steady level flight are defined below.

\[ q = \frac{1}{2} \rho V^2 \]  \hspace{1cm} \text{Eq. 3.1}

\[ W = L = qSCL \]  \hspace{1cm} \text{Eq. 3.2}

\[ T = D = qSC_D \]  \hspace{1cm} \text{Eq. 3.3}

The drag equation (Eq. 3.3) is broken down into four main contributions – the CDA, or equivalent drag area of the fuselage, the area, \( S \), and drag coefficient, \( C_D \), of the main wing and tail, and the induced drag in producing lift. The expression for induced drag assumes that the front wing is providing nearly all the lift, and the induced drag of the tail is negligible in comparison. In a tandem configuration, the front wing is the main wing and the rear wing is the tail, even though both are producing lift. The resulting equation (Eq. 3.4) is shown below.

\[ D = q(CDA + SC_{dw} + StC_{dt}) + W^2 \]  \hspace{1cm} \text{Eq. 3.4}

By substituting the lift coefficient relation (Eq. 3.5) and definition of aspect ratio (Eq. 3.6) into the drag equation (Eq. 3.4), the induced drag is expressed as the result of the square of the span loading of the main lifting wing (Eq. 3.7).

\[ C_L = \frac{2W}{\rho V^2 S} \]  \hspace{1cm} \text{Eq. 3.5}

\[ AR = \frac{b^2}{S} \]  \hspace{1cm} \text{Eq. 3.6}

\[ D = q(CDA + SC_{dw} + StC_{dt}) + \frac{W^2}{q\pi b^2} \]  \hspace{1cm} \text{Eq. 3.7}

Using a rough scaling relationship, \( K_t \) (defined in Eq. 3.8), it is possible to approximate the relative drag sizes between the rear and front aerodynamic surfaces in Eq. 3.9. As approximations, \( K_t = 0.3 \) for a conventional aircraft, and \( K_t = 1 \) for a tandem-wing configuration.

\[ K_t = \frac{C_{d_t}S_t}{C_{d_w}S} \]  \hspace{1cm} \text{Eq. 3.8}
\[ D = q(CDA + SC_d(1 + K_t)) + \frac{w^2}{q_\text{eb}b^2} \]  
Eq. 3.9

The product of the flight drag with the speed is the power-required for steady level flight, as shown in Eq. 3.10.

\[ P_{\text{flight}} = DV = \left( q(CDA + SC_d(1 + K_t)) + \frac{w^2}{q_\text{eb}b^2} \right) V \]  
Eq. 3.10

The propulsive power can be related to the power required from the battery via power system efficiencies in Eq. 3.11. Setting the propulsive power equal to the flight power (Eq. 3.12), the battery power is related to the variables that compose flight power (Eq. 3.13).

\[ P_{\text{propulsive}} = \eta_{\text{ESC}}\eta_{\text{motor}}\eta_{\text{propeller}}P_{\text{battery}} \]  
Eq. 3.11

\[ P_{\text{propulsive}} = P_{\text{flight}} \]  
Eq. 3.12

\[ P_{\text{battery}} = \frac{1}{\eta_{\text{ESC}}\eta_{\text{motor}}\eta_{\text{prop}}} \left( q(CDA + SC_d(1 + K_t)) + \frac{w^2}{q_\text{eb}b^2} \right) V \]  
Eq. 3.13

The battery’s energy can be related to its weight by the battery’s specific energy, here denoted as \( \bar{E} \), shown in Eq. 3.14. When operating at a temperature that would reduce battery performance, it is necessary to reduce \( \bar{E} \) to account for the loss in capacity. At room temperature, a typical \( \bar{E} \) value for average lithium-polymer batteries is 16.5 Whr/N or 59,400 J/N.

\[ E_{\text{battery}} = W_{\text{battery}}\bar{E} \]  
Eq. 3.14

The available energy divided by draw rate is the endurance of the vehicle in Eq. 3.15:

\[ t_{\text{endurance}} = \frac{E_{\text{battery}}}{P_{\text{battery}}} = \frac{W_{\text{battery}}\bar{E}}{\eta_{\text{ESC}}\eta_{\text{motor}}\eta_{\text{prop}}\left( q(CDA + SC_d(1 + K_t)) + \frac{w^2}{q_\text{eb}b^2} \right) V} \]  
Eq. 3.15

To relate the weight of the battery with the weight of the aircraft, the weight of the aircraft is broken into four categories based on its role in the system: the payload, the power throughput components (motor, ESC, and propeller), the airframe, and the battery (Eq. 3.16). The autopilot board is considered part of the payload.

\[ W = W_{\text{payload}} + W_{\text{motor ESC prop}} + W_{\text{airframe}} + W_{\text{battery}} \]  
Eq. 3.16

By dividing each component weight by the total aircraft weight (Eq. 3.17), the aircraft is broken into weight fractions (Eq. 3.18). The weight of the battery (Eq. 3.19) and payload (Eq. 3.20) are thus related to the aircraft weight.

\[ \frac{W}{W} = 1 = \frac{W_{\text{payload}}}{W} + \frac{W_{\text{motor ESC prop}}}{W} + \frac{W_{\text{airframe}}}{W} + \frac{W_{\text{battery}}}{W} \]  
Eq. 3.17

\[ 1 = f_{\text{payload}} + f_{\text{motor ESC prop}} + f_{\text{airframe}} + f_{\text{battery}} \]  
Eq. 3.18

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\[ W_{\text{battery}} = f_{\text{battery}} W \quad \text{Eq. 3.19} \]

\[ W = W_{\text{payload}} / f_{\text{payload}} \quad \text{Eq. 3.20} \]

Weight fractions (Eq. 3.20 and Eq. 3.19) and the lift equation (Eq. 3.2) are substituted into Eq. 3.15, which is then inverted to group the variables. The result is Eq. 3.21, shown below. To maximize endurance, the expression on the right side of the equation should be minimized.

\[ \frac{1}{t_{\text{endurance}}} = \ldots \]

\[ \frac{V}{\eta_{\text{ESC}} \eta_{\text{motor}} \eta_{\text{propeller}}} \left( \frac{C_{dW} (1+K_t)}{C_L f_{\text{battery}}} + \frac{qCDA}{W_{\text{battery}}} + \frac{W_{\text{payload}}}{q b^2 \rho f_{\text{battery}} f_{\text{payload}}} \right) \quad \text{Eq. 3.21} \]

Eq. 3.21 reveals the relations between various aircraft characteristics and the effects they have on the aircraft performance.

The first grouping \( V / (\eta_{\text{ESC}} \eta_{\text{motor}} \eta_{\text{propeller}}) \) has units of 1/seconds, and the three terms inside the parentheses are nondimensional. This first group scales endurance to the characteristics of the power system. To maximize endurance, the \( \mu \text{UAV} \) should fly as slowly as the mission requirements allow and should use the most efficient propulsion system available. The \( \mu \text{UAV} \) should also maximize the specific energy of the battery.

The first term in the parentheses of Eq. 3.21 is effectively the endurance as related to the profile drag of the wings. Assuming the 2-D lift coefficient is roughly equal to the 3-D lift coefficient, the term can be rewritten as Eq. 3.22 below. This relationship shows that, to maximize endurance, the airfoil glide ratio at the cruise point should be maximized. This term also shows that ideally, the tail should be small and the battery weight fraction should be maximized.

\[ \frac{C_{dW} (1+K_t)}{C_L f_{\text{battery}}} \approx \left( \frac{C_l}{C_d} \right)^{-1} (1+K_t) \quad \text{Eq. 3.22} \]

The second parenthetical term of Eq. 3.21 describes the energetic loss due to the fuselage’s drag on endurance, normalized by the weight of the battery. All three variables, \( q \), \( CDA \), and \( W_{\text{battery}} \), and the way they interact must be considered such that adding more battery does not reduce performance through large fuselage drag and/or higher flight speed which can easily outweigh the increased battery energy.

The third parenthetical term of Eq. 3.21 describes the induced drag and is affected by the span of the main lifting surface squared. To maximize endurance, the span loading should be minimized. This term also serves the purpose of linking the aircraft size to the payload size.

While the first parenthetical term describes the wing performance and its relation to battery weight fraction and the second parenthetical term describes the ratio between fuselage drag and the contained-volume, it is only the third parenthetical term that links the size of the aircraft to the weight of the payload and thus sizes the aircraft for the mission.
3.2.2. **Endurance Maximization Baseline**

To quantitatively explore these relations, Eq. 3.21 is plotted (in Figure 29 and Figure 30) with the following assumptions and inputs. The following assumptions and equation parameters makes up the “baseline” case:

1) Because the mission goal is mainly high-altitude flight, \( \rho \) is set to be standard density at 30,000 ft.
2) As set by the mission requirements (Section 1.2.1), the aircraft flies at 30 m/s.
3) The mass of the payload (sensor package and autopilot combined) is assumed to be 20 grams.
4) The mass of the power delivery system (motor, ESC, and propeller) is assumed to be 25 grams.
5) The mass of the airframe (including servos and internal structure) is assumed to be 80 grams.
6) Each battery cell is assumed to weigh 13.5 grams (the weight of the Ener1 SPB463048 cell).
7) From a battery discharge test conducted at -20°C, it is found that the useful battery capacity drops to approximately 75% of the nominal specification [2]. The SPB463048 (with a nominal specific energy of 70,404 J/N) is therefore discounted to 52,803 J/N.
8) The efficiency of the ESC, \( \eta_{ESC} \), is assumed to be 90%. This is true for when throttle is above 50% [18].
9) The efficiency of the motor is \( \eta_{motor} = 75\% \) [19]
10) The efficiency of the propeller is assumed to be \( \eta_{prop} = 70\% \) [20].
11) The wingspan of the main lifting surface is 0.3 m.
12) The airfoil is assumed to perform with \( C_l/C_d = 30 \). This 2-D glide ratio is achieved at \( C_l = 1.0 \).
13) The aircraft configuration is assumed to be a tandem wing, resulting in \( K_t = 1 \).
14) A rough empirical fit to an initial set of wind tunnel tests for rectangular-cross-section fuselages at approximately 5° angle of attack was used to determine the relationship between $CDA$ and $W_{battery}$. The drag of the fuselage, rather than rising linearly with cross-sectional area, rises with the square due to the constrained fuselage length, which disallows a minimum-drag pressure recovery profile. Figure 28 below shows the vertical projection shapes for a family of fuselages, sized to contain the required number of batteries as indicated by color.

![Figure 28: Constrained fuselage battery accommodation](image)

This relation fitted to the wind tunnel tests is shown in Eq. 3.23 below with $CDA$ in $m^2$ and $W_{battery}$ in $N$.

$$CDA = 0.0003 + 0.00015 \times W_{battery}^2$$  \hspace{1cm} \text{Eq. 3.23}$$

The relationship generates a $C_D$ from 0.1 (zero batteries) to 0.2 (when the fuselage is filled with 15 cells) with reference to the cross-sectional area of the envelope (62 mm x 48 mm).

The analysis is conducted by varying the number of battery cells under the assumptions listed above. Initially, the cell configuration (series and parallel cell arrangement) is neglected. The results for this baseline case are presented below with endurance vs. battery weight fraction in Figure 29 and endurance vs. battery cell count in Figure 30. Because the number of battery cells is quantized, each node represents a possible solution with a whole number of battery cells.
Counterintuitively, endurance is not maximized through the maximization of battery capacity. As cell count is increased, the fuselage drag and power-required is increased above the capacity added by the battery, resulting in an overall reduction of flight time. Figure 30 shows that under the baseline assumptions, a 12-cell configuration produces the system-optimum point, yielding a flight time of 59 minutes. To improve the accuracy of the model, however, two more refinements can be made.
3.2.3. Refining Results for Maximum Wing Area

This model can be refined by considering a limit on the available main wing area, $S$, due to the MJU-10/B form factor that the baseline assumptions do not account for. The maximum front wing area available (due to the height of the available envelope and the height of components inside) is two panels, each having a span of $165\ mm$ and a chord of $35\ mm$. This geometry yields a total wing area of $11,550\ mm^2$. The maximum aircraft weight for which the baseline assumptions listed in Section 3.2.2 are valid is calculated using the lift equation shown in Eq. 3.24 below.

$$W = 0.5 \left(0.459 \frac{kg}{m^3}\right) \left(30 \frac{m}{s}\right)^2 (11550 mm^2)(1.0) = 2.39N = 243 grams$$ Eq. 3.24

Above this weight limit, the aircraft must either cruise at a higher $V$, a higher $C_l$ above peak $C_l/C_d$, or do a combination of both. For the low-Re regime, maintaining $C_l/C_d$ is very difficult above a $C_l$ of 1.0 due to the problem of laminar separation on the upper surface. Therefore, for this study, the speed of the aircraft was increased to meet $L = W$ with limited wing area.

The maximum $SC_L$ (rounded up to $12,000\ mm^2$ to account for some carryover lift by the fuselage) is then incorporated into the model such that the airspeed is calculated either by the minimum mission speed requirement ($30\ m/s$) or the speed necessary to generate lift equal to weight at the maximum $SC_L$ - whichever one is larger. Thus, speed is defined formally via Eq. 3.25 below.

$$V = \max (30\ m/s, \left(\frac{2W}{\rho \cdot 12,000\ mm^2}\right)^{0.5})$$ Eq. 3.25

Figure 31 plots the $30\ m/s$ line in magenta, the $SC_L = 12,000\ mm^2$ line in black, and the resulting speed value highlighted in green. Below a cell count of 10, the configuration at maximum $SC_L$ is able to reduce $S$ and increase speed to meet the minimum speed requirement. Above a battery size of 10 cells, however, the aircraft must cruise faster than minimum speed to generate enough lift to equal weight.
The excess speed required of configurations with 10 cells or more reduces the endurance and pushes the optimum aircraft design point toward a smaller battery, as Figure 32 and Figure 33 below illustrate.
Figure 33: Endurance vs. battery cell count, limited by SCL

Figure 32 and Figure 33 show that the maximum-endurance configuration drops from a 12-cell battery pack to a 10-cell battery pack.

3.2.4. Refining results for motor efficiency considerations

To refine the predictions made by this model for the effect of the battery configuration, the efficiency of the motor is corrected for its dependence on voltage. Figure 34 below shows that peak efficiency for the Proton 12-30-4000 brushless inrunner motor drops from 79% at 11.1 V to 73% at 7.2V.

Figure 34: Efficiency trends for Proton 12-30-4000 motor [19]
The 11.1V and 7.2V voltages are important because these are the possible voltages the battery pack can have. Nominally, 3 lithium-polymer cells in series generates 11.1V and 2 cells in series generates 7.4V. Because the cell count is quantized, and battery packs are made with parallel groups of cells in series, only cell counts divisible by 2 or 3 can be achieved. A plot of system endurance vs. battery cell count is shown below in Figure 35. The solid nodes on the plot are the achievable configurations.

![Figure 35: Endurance time vs. cell count and voltage](image)

The analysis presented in Figure 35 shows that the optimum cell count of 10 becomes suboptimal since the battery configuration yields a lower motor efficiency. Cell counts divisible by 3 yield approximately 8% more endurance. As a result of the motor's efficiency dependence on the battery cell number, a 9-cell configuration (highlighted by the orange circle in Figure 35) was selected. The total aircraft weight is estimated to be 241.5 grams, composed of 121.5 grams of batteries, 80 grams of airframe, 20 grams of sensors, and 25 grams of motor, ESC, and propeller.
3.3. Wing Planform Trade Analysis

The analysis presented in this section links the wing performance to two geometric parameters - the location of the front wing hinge and the most-rearward-allowable CG location. This analysis is accomplished with some basic assumptions about the geometric and aerodynamic characteristics of the wings and basic aircraft stability relations. Once the wing performance trends have been quantitatively mapped, the internal layout and fuselage design can be addressed in Section 3.4 and Section 3.5.

The wing folding geometries for the front and rear wings are shown in Figure 36 and Figure 37 respectively. In Figure 36 below, the front wings are translucent green and pivot about the front wing hinge (red), located at $x_{\text{hinge}}$. In the center, the darker green denotes the region where the front wings overlap. The area covered by the grey bounding box is allocated to the protective case, reducing the total available area for the $\mu$UAV to 170mm $\times$ 60mm.

Figure 36: Front wing folding geometry

Figure 36 illustrates how the location of the hinge in a swinging-tandem-wing airframe configuration couples the dimensions of the lengthwise envelope to the span of the front wing. A larger $x_{\text{hinge}}$ value reduces the available span of the front wing, which must terminate at the end of length allocated to the $\mu$UAV. The $x_{\text{hinge}}$ parameter is also important to the fuselage shape as will be elaborated in Section 3.5.
The rear wing folding geometry is not highly coupled with the fuselage shape, though to have a tapered tail for low drag, the hinge location must be placed on the aircraft centerline. The hinge of the rear wing must be placed such that the trailing edge cannot overlap the propeller plane. As shown in Figure 37, the rear wing extends the full length to the nose so that the vertical finlet surface can be attached at the tip.

![Figure 37: Rear wing folding geometry](image)

Under a set of reasonable assumptions and parameters, it is possible to link wing aerodynamic performance and allowable CG location to the location of the front wing hinge alone in a continuous configuration space. The assumptions, relation formulations, and rationales are listed below. The longitudinal variables are referenced to the front wall of the space allocated to the µUAV.

Seven assumptions and parameters about the wing planforms are used in this analysis.

1) It is assumed that both wings are rectangular.
2) The total effective lifting area, $S_{CL}$, is set to be 12,000 mm$^2$. Assuming the aircraft flies at a $C_L = 0.6$ with reference to the total (front and rear) wing area, this yields a total wing area of 20,000 mm$^2$ which is distributed between the front and rear wings.
3) It is assumed that the chord of the front wing is 35 mm since this is the maximum chord the wings can have without needing to stack vertically. This setting was found to maximize the L/D of the wings and affords the largest possible Reynolds number to the front wing so that its higher $C_L$ can be achieved most efficiently. Because the chord $Re$ is on the order of $10^4$, 

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significantly reducing the \( Re \) will exacerbate nonlinearities in the lift curve slope and drag caused by laminar separation and laminar separation bubbles.

4) The wing span is maximized for both front and rear wings. This is done to minimize the induced drag of the aircraft as \( 1/t_{endurance} \) varies as \( W/b^2 \) as derived in Section 3.2.1.

5) The rear wing is placed as far back as possible such that the wing trailing edge, when deployed, is aligned with the end of the fuselage. This feature places the neutral point, and therefore, CG, as far back as possible for a given front and rear wing size distribution.

6) The \( x_{wall offset} \) distance on the front wing pin is set to be 5 mm. This is necessary for the wing to have a feature that allows it to come to rest at a stop point.

There are seven stability assumptions, as enumerated below.

1) The aircraft must be statically stable, as established in Section 2.3. A 15\% static margin was specified with respect to the front wing chord, as defined in Eq. 3.26 below:

\[
SM = \frac{x_{NP} - x_{CG}}{c_{front}} = 0.15 \quad \text{Eq. 3.26}
\]

2) The fuselage behaves like a wing in its contribution to the aerodynamic center.

3) All surfaces have aerodynamic centers at 25\% chord.

4) The wing interaction effects can be neglected in the calculation of the neutral point.

5) It is assumed that the pitching moments are negligible in their contribution to the neutral point compared to the lift effects.

6) The 3D lift-curve slope is approximated using Eq. 3.27 below: [21]

\[
C_{LA} = C_{LA} \left( \frac{AR}{AR + \left( \frac{2 \times AR + 4}{AR + 2} \right)} \right) \quad \text{Eq. 3.27}
\]

7) The neutral point is calculated as the lift-curve-slope weighted areas of the two wings and fuselage as determined by Eq. 3.28 below. The contribution from each surface originates at the surface's aerodynamic center.

\[
x_{NP} = \frac{x_{AC_{fuse}} S_{fuse} C_{L_{fuse}} + x_{AC_{front}} S_{front} C_{L_{front}} + x_{AC_{rear}} S_{rear} C_{L_{rear}}}{x_{AC_{fuse}} S_{fuse} C_{L_{fuse}} + x_{AC_{front}} S_{front} C_{L_{front}} + x_{AC_{rear}} S_{rear} C_{L_{rear}}} \quad \text{Eq. 3.28}
\]

There are two drag assumptions used to calculate the L/D performance.

1) To account for Reynolds effects on the skin friction drag of the wings, a simple drag model is used (shown in Eq. 3.29) to approximate the profile drags of the airfoils.

\[
C_d = 2 \times \frac{1.328}{\sqrt{Re}} + 0.025 \times C_l^2 \quad \text{Eq. 3.29}
\]

2) It is assumed that the induced drag produced by the two wings can be calculated separately without considering interaction effects.
By asserting the maximum-span, the maximum-front-chord, and the total wing area conditions on the wings, the location of the front wing hinge uniquely determines both front and rear planforms. The planforms are then analyzed to determine the neutral point location, the allowable CG location, both wing $C_L$s, the Reynolds number, and L/D of the airframe (minus the higher-order 3D interaction and fuselage effects).

A continuous linear space of $x_{hinge}$ values was sampled from 5 mm to 100 mm, and using the geometry and equations defined above, the relations presented in the following figures are produced. The right side of Figure 38 below presents a plot of the location of the aircraft neutral point and the CG location (with 15% static margin) as a function of the location of the front wing hinge. Three points, labeled A, B, and C, are highlighted on the graph, and their corresponding configurations displayed on the left. The red and blue dots show the locations of the neutral point and the maximum-rearward CG (MRCG) location that produce at least 15% static margin, respectively.

Configuration A places the front wing hinges as forward as possible, yielding a design that maximizes the front wing area. Therefore, the neutral point is close to the nose (58 mm) and a more-forward CG (52.5 mm) is required to keep the aircraft stable. Configuration C places the front wing hinge rearward, resulting in a much smaller front wing and a larger rear wing, which places the neutral point at 107 mm and the CG is allowed to be more rearward, at 101.5 mm. Configuration B exists somewhere between A and C.

![Figure 38: NP, CG, and wing hinge relationship configuration study](image-url)
Since the stability conditions, and therefore front and rear \( C_L \)s, for all configurations can be calculated based on the geometry, it is then possible to calculate the drag coefficients and therefore L/D performance of the wings and link this information to the allowable location of the center of gravity. These results are shown below in Figure 39.

![Wing L/D vs CG location](image)

**Figure 39: Configuration study results - wing L/D vs CG location**

Figure 39 shows the predicted L/D of the aircraft wings as a function of the allowable max-rearward CG location. Toward the right side of the horizontal axis, the L/D drops significantly due to the increased induced drag of the front wing, which when placed rearward, must carry a large loading with a small span. Because the relationship goes as the span loading squared, the penalty for a rearward CG is severe. By placing the CG toward the nose, the neutral point can follow, which permits the front wing to grow maximally, yielding lower induced drag and better aerodynamic efficiency.

Overall, this analysis shows that maximum wing performance is achieved by placing the center of gravity as far forward as possible.
3.4. Internal Arrangement

From the results calculated in Section 3.3, it is clear that higher performance can be achieved by placing the CG as far forward as possible. This is accomplished by moving the densest components forward. Figure 40 presents the packing densities of the components, defined as the mass of the component divided by the useful space it occupies (e.g., gaps between components of the circuit board are included in its volume calculation).

![Packing density of aircraft components](image)

Figure 40: Packing densities of μUAV components

The approximate layout of components inside the μUAV is shown below in Figure 41.

![Layout of major μUAV components](image)

Figure 41: Layout of major μUAV components

The driving feature of the layout is the propeller. To maximize the available diameter of the propeller without resorting to a folding mechanism, it must be placed in the middle of the μUAV system envelope. The motor, which has to be in-line with the propeller, is pushed forward as far as possible, clearing the way on the top for the autopilot board, and on the bottom for the rear wing folding mechanism. The motor drives the propeller via a drive shaft. The servos are
placed on either side of the motor. The battery stack is arranged in a stack with cells staggered to conform to the nose shape of the fuselage. A 9-cell stack is shown in Figure 42 below.

![Staggered battery pack](image)

**Figure 42: Staggered battery pack**

The batteries are glued to each other and encased in a protective and insulating foam shell as will be elaborated in Section 5.5.1. The asymmetric pack configuration allows the atmospheric sensor board to be placed at the front-right.
3.5. Fuselage Design, Testing, and Selection

While the wings reach maximum L/D at the most forward hinge and CG placement, fuselage performance as a function of the hinge geometry must be considered carefully so that the total system optimum can be found. Since the weight of the system is already defined via the analysis presented in Section 3.2, the goal of this section is to minimize the total drag.

For example, building a fuselage that accommodates the hinge located at the maximum-forward location while accommodating the battery pack produces a flat, sharp-cornered fuselage nose region, as illustrated in Figure 43. The wing hinge is located 5 mm from the forward limit and is labeled with the red A. The square nose shape is necessary to allow the wings to swing above the height of the battery pack. This nose feature results in high drag via early flow transition and vortex shedding along the edge.

![Figure 43: Fuselage with wing hinge placed at best wing L/D location](image)

3.5.1. Fuselage Design and Design Matching

To maximize the total airframe performance, a balanced fuselage shape must be developed that allows a forward CG (and therefore high wing L/D via Figure 39) without incurring a profile drag penalty that outweighs the wing performance benefit. Because the drag for a non-asymmetric 3D body is difficult to predict, a discrete study was performed. Five candidate fuselages (named F0 through F4) were produced which placed the wing hinge at a range of locations. Each fuselage was then configured with the packing scheme defined in Section 3.4, and the maximum-forward CG (MFCG) was calculated.

Prototypes of these fuselages were constructed and tested in the wind tunnel to determine their CDA, which allowed the total aircraft performance to be calculated. Based on the CG and wing
hinge location, the wing performance was matched to the fuselage candidates, and a final design was selected.

Geometrically, the fuselages were designed as the intersections of vertical and horizontal projections necessary to hold the internal components. The vertical projections of all candidate fuselages are the same due to the width of the batteries and the space required of the atmospheric sensing payload, shown in Figure 44.

![Irreducible width](image)

**Figure 44: Fuselage nose packing vertical projection**

The horizontal projections, shown in Figure 45, were designed to investigate a range of front wing hinge locations.

![Family of fuselage shapes](image)

**Figure 45: Family of fuselage shapes**

The fuselage shapes were cambered for three reasons. First, this allows the fuselages to produce lift so that the lift distribution is smoother across the front wing. Second, a downturned nose allows the fuselage to be tolerant to high angle of attack situations without leading to massive flow separation on the top side; this helps the aircraft stay stable during the high-G recovery maneuver during the transition phase between deployment and flight. Third, the fuselage end must provide a mounting point at both the middle (to providing a mounting location for the propeller shaft, as discussed in Section 3.4) and the bottom, to provide the mounting location for the rear wing; this results in a low tail.
For each of these fuselage shapes, the MFCG is calculated by placing all the components as forward as possible. Figure 46 shows the fuselage candidates, the locations of the front wing hinges (the red lines), and the MFCG locations marked by the \( \bullet \) symbol. The staggered battery layout is also shown for each candidate.

\[
\begin{align*}
F0 & \quad x_{\text{hinge}}=20.4\,\text{mm} \\
& \quad \text{MFCG}=59.4\,\text{mm} \\
F1 & \quad x_{\text{hinge}}=25.3\,\text{mm} \\
& \quad \text{MFCG}=61.3\,\text{mm} \\
F2 & \quad x_{\text{hinge}}=30.3\,\text{mm} \\
& \quad \text{MFCG}=63.4\,\text{mm} \\
F3 & \quad x_{\text{hinge}}=35.2\,\text{mm} \\
& \quad \text{MFCG}=66.2\,\text{mm} \\
F4 & \quad x_{\text{hinge}}=39.1\,\text{mm} \\
& \quad \text{MFCG}=68.1\,\text{mm} \\
\end{align*}
\]

Figure 46: Fuselage family with wing hinge, battery pack, and CG

As Figure 46 shows, the batteries in the fuselage are shifted fore and aft to fit the fuselage side-profiles, resulting in a range of MFCG locations. The area above the battery pack (split by the blue line) is used to store the front wings, but the hinge must be placed approximately 10 mm behind the tip of the split section to provide a stoppage location for the wing.
There are now two constraints necessary to match the fuselage to a wing. First, the wing's \( x_{hinge} \) location must be behind the fuselage's available \( x_{hinge} \) (shown in red in Figure 46). Second, the fuselage's MFCG must be in front of the wings' MRCG. Both of these conditions need to be met to couple the fuselage's drag to the wing's drag curves.

Fortunately, the coupling is simplified because the fuselage's MFCG is always forward of the wing MRCG for a given \( x_{hinge} \). In Figure 47 below, the fuselage MFCG vs. \( x_{hinge} \) alongside the MRCG vs. \( x_{hinge} \) derived in the stability analysis from Section 3.3.

![Figure 47: NP and allowable CG; fuselage MFCG vs. wing hinge location](image)

Figure 47 shows MFCG values as the black line, and the location of the neutral point in red, and MRCG in blue. Because the black line is always below (closer to the fuselage nose) the blue line, all fuselage candidates meet the stability conditions that their wing hinge locations require. Therefore, the wing hinge location can be used as the coupling variable between the fuselage performance and the wing performance.

### 3.5.2. Fuselage drag test and selection of final design

The five candidate fuselage shapes were constructed from foam using a hot wire CNC foam cutter and tested in a 1 ft × 1 ft open wind tunnel at a length-\( Re \) of 200,000. Figure 48 shows the experimental setup.
The fuselage drag was measured with a load cell for 0° and 10° angle of attack to bracket the range of fuselage drag that may be seen in operation. The resulting data is presented in Figure 49 below.
The wind tunnel tests presented in Figure 49 show that the F2 fuselage creates the least drag at both 0° and 10° angles of attack. As the fuselage nose shape deviates from the F2 shape, the drag rises monotonically in both directions.

To obtain an estimate of the total airframe drag performance, the fuselage drag is then compiled with wing performance at the corresponding wing hinge location to determine the drag of the entire system. Both 0° fuselage and 10° fuselage test results are presented to bracket the expected in-flight fuselage contribution, though the wing performance is not a function of the angle of attack in this estimation. The results are shown in Figure 50.

As Figure 50 shows, the F2 fuselage, coupled with a wing that hinges at 30.3 mm yields the airframe with the highest L/D for both fuselage drag values. The F0 and F1 fuselages produce more drag than is reduced from the wings by the wingspan and therefore are suboptimal solutions. The F3 and F4 fuselages not only produce more drag, but the wing configurations compatible with these fuselages are less efficient, leading to aircraft configurations that perform far less well than the F2 solution.
3.6. Resulting Airframe

The airframe resulting from the analysis presented in Section 3.5.2 is presented in Figure 51 below with all dimensions in millimeters.

As Figure 51 shows, the fuselage is 158 mm long, has a width of 55 mm, and a height of 40 mm. The front wing is mounted at $\alpha = 0^\circ$ and has a chord of 35 mm and a span of 301 mm. The rear wing is mounted with a $-1.88^\circ$ incidence and has chord of 30 mm and a span of 294 mm. The elevons are 7 mm in chord and are located from 7.5 mm to 77.5 mm in the spanwise direction. The total projected wing area is $19,355 \text{ mm}^2$, counting the effective wing area engulfed within the fuselage.
Both wings are rectangular in their vertical projections. The rear wing is designed with approximately 9° of dihedral starting at 91 mm from the centerline. This dihedral section allows the wing to fold neatly around the fuselage and not intrude outside of the vertical envelope due to the mounting incidence. Figure 52 shows the rear wing curvature in the stowed configuration. The dihedral sections also contribute positively toward spiral stability.

![Figure 52: Stowed fuselage - side view](image)

To provide longitudinal stability, the μUAV requires vertical surfaces. Because the span of the rear wing was maximized, the vertical aerodynamic surfaces are placed at the tips of the rear wing such that they fold around the nose of the fuselage in the stowed configuration. Placing the vertical surfaces at the wing tips provides the necessary vertical area without having to introduce an additional mechanism, reducing both complexity and weight.

The trailing edge tips of the finlets are designed to tolerate small strikes against the leading edges of the front wings that may occur during deployment. Figure 53 below shows the wings mid-deployment.

![Figure 53: Finlets sized for deployment strikes](image)

The curvature and spacing allows the wings to deflect 7 mm closer on the right side (9 mm on the left side) without locking the wings together. Instead, the front wing will glance off the rear due to the curved feature.
3.7. Predicted Performance

To predict the performance of the μUAV in detail, the aircraft was modeled in an aerodynamics program called XFLR5, written by Andre Deperrois [22]. The program was chosen for its ability to apply Re-dependent airfoil properties, which is very important in the low-Re regime where drag is very sensitive to Re. Therefore, the performance plotted cannot be simply reduced to the lift and drag coefficients, but must take into account the speed and air kinematic viscosity at which these coefficients are calculated.

The first airfoils used on the μUAV were designed by Dr. Mark Drela. The front wing airfoil was the ASV02 (shown in Figure 54) and the rear wing airfoil was the ASV10 (shown in Figure 55).

![Figure 54: ASV02 front wing airfoil](image1)

![Figure 55: ASV10 rear wing airfoil](image2)

These airfoils were designed to operate in a low Reynolds number range, between 30,000 and 100,000. The thin trailing edge allows the wings to be flexible so that stack together, as previously discussed in Section 2.4. A set of XFOIL polars for the two airfoils at Re = 30,000 and Re = 80,000 are shown in Figure 56 below. The front wing at 30,000 ft operating near stall has Re ≈ 30,000 and at sea level and maximum dash speed (C_L = 0.2) operates at Re ≈ 80,000.
In Figure 56 above, the red polars shows the performance of the ASV02 airfoil and the green polars show the performance of the ASV10 airfoil. The dashed lines are polars calculated for $Re = 30,000$ and the solid lines show the polar for $Re = 80,000$. The right side of the plot shows the lift curves for both airfoils which both have small but noticeable discontinuities between $\alpha = 2$ and $\alpha = 4$. This feature corresponds to laminar flow separation on the upper surface, as can be seen in the increase in drag in the $C_l$ vs. $C_d$ plot on the left. At $\alpha \approx 4$, the upper boundary layer transitions from laminar to turbulent flow and is thus able to maintain flow attachment, and yields an increase in $C_l$.

An analysis was conducted in XFLR5 by setting the lift equal to the weight at the altitude limits, sea level and 30,000 ft. A screenshot of the model in XFLR5 is presented in Figure 57 below.
Because the drag due to non-axisymmetric bodies are difficult to predict, the analysis was conducted without the fuselage, and an average F2 fuselage drag from wind tunnel testing was added after the XFLR5 simulation was performed. The resulting predicted aircraft glide ratio (L/D) vs. cruise speed for constant lift is shown below in Figure 58.

![UAV L/D predictions with ASV airfoils](image)

**Figure 58: XFLR5-predicted μUAV L/D Performance**

The performance plot in Figure 58 shows that the μUAV is expected to have a maximum L/D of approximately 8 and L/D is expected to fall nearly linearly with increasing speed. As the flight altitude is increased, the glide ratio is expected to fall slightly due to the decreased $\text{Re}$ from an increase in the air's kinematic viscosity at high altitudes. The fuselage drag approximately halves the glide ratio performance of the wings.

A simulation code was written by Russell Stratton to predict the endurance and speed range of the μUAV. This code interpolates from the L/D curves presented in Figure 58 and calculates endurance assuming the batteries can provide 75% of their nominal capacity due to cold operating conditions. Figure 59 below presents the predicted endurance and Figure 60 presents the operating speeds against the wind speed.
As Figure 59 shows, the endurance of the µUAV is predicted to be approximately 43 minutes at 30,000 feet and increases nearly linearly to approximately 75 minutes at sea level. The speed at altitude also meets mission requirements, as shown in Figure 60. At 30,000 ft, the µUAV’s most efficient cruise is around 60 knots.
The performance of the µUAV is plotted against other small electric UAVs in Figure 61 below.

![Small electric UAV performance: endurance, altitude range, vs weight](image)

**Figure 61: Comparison of small electric UAVs by altitude, weight, and endurance**

As Figure 61 shows, the µUAV provides a unique combination of altitude, weight, and endurance.
Chapter 4: Aerodynamic Testing and Refinement

To move from theoretical and computation design to prototyping and manufacturing, it was necessary to validate the aerodynamic design of the μUAV. To quantify the performance and to determine the stability characteristics, a series of engineering prototypes were constructed and tested. This chapter will cover the prototypes constructed to perform flight testing, wind tunnel tests of the aircraft, revisions to the design resulting from the wind tunnel tests, and a detailed stability analysis of the refined aircraft.

4.1.1. Flight Prototypes

To test the handling properties of the μUAV in-flight, two non-deploying prototypes were constructed.

The first model, built at double-scale (2X) was constructed from Expanded Polypropylene (EPP) foam, balsa, and carbon fiber was flight-tested via remote-control. The size was doubled to slow the flight characteristics and make the aircraft more visible to the pilot. The 2X aircraft is shown in Figure 62.

![Double-scale flight model](image)

While onboard attitude logging was unsuccessful, it can be reported that the aircraft was found to be controllable and stable in all modes except a mild spiral divergence, which was not difficult for the pilot to control.

After the successful test flight of the 2X model, a full-scale (1X) model was constructed to test handling qualities with moments of inertia that are closer to the final aircraft. The 1X fuselage was constructed from fiberglass laid-up on a foam-cut core. The front wing was constructed from carbon fiber laid-up on a mold made from tooling wax, and the rear wing was constructed from carbon fiber laid-up on an aluminum mold. A picture of this aircraft is provided in Figure 63.
The 1X aircraft was also flight-tested via remote-control and was found to have similar lateral stability characteristics to the 2X model. It was noted in flight testing that the 1X model was very sensitive in pitch despite the same longitudinal stability margin as was implemented in the 2X flight model.
4.2. Wind Tunnel Testing

To measure the aerodynamic performance of the μUAV, the 1X aircraft presented in Section 4.1.1 was tested in the MIT Wright Brothers wind tunnel to measure the performance of the full-scale prototype aircraft. All wind tunnel tests were unpressurized (airflow was at atmospheric density and pressure). The air flow speed was incremented to measure the μUAV's performance at different $Re$.

4.2.1. Custom free-pivot wind tunnel balance

Because the forces on the μUAV were smaller than the available wind tunnel balance could resolve (drag of the vehicle is predicted to be less than an ounce), a custom wind tunnel balance (depiction shown in Figure 64 below) was designed and constructed by one of the undergraduate researchers, Sam Range.

![Free-pivot wind tunnel balance diagram](image)

**Figure 64: Free-pivot wind tunnel balance diagram**

The lift-drag balance shown in Figure 64 is composed of a drag cell mounted on a lift cell. The μUAV is attached via a horizontal wire to a blade that pushes on the drag cell. The μUAV was mounted via a wire that pierces the CG location, and the μUAV's attitude was controlled by the elevon control surfaces. The drag cell is supported and counterbalanced by a parallelogram which pushes on a second cell, which determines the lift. There is a vibration-damping layer between the bottom of the parallelogram and the lift cell. A Mylar fairing is wrapped around the assembly in an attempt to reduce vibration.

Using this wire-suspension method, the μUAV floated freely in pitch, and a remote control was used to adjust the model's attitude. This scheme allowed trim drag to be incorporated
automatically into the total performance data. Another consequence of the wire-suspension method is that the μUAV only trimmed at attitudes possible in flight. A photograph of the experimental setup is shown in Figure 65.

Figure 65: μUAV being tested in the wind tunnel

By using a remote control, the μUAV's attitude is adjusted at small increments using the trim function. The lift and drag data were measured, and due to the wire-suspension setup (shown in Figure 66), \( C_m = 0 \) for all data points.

Figure 66: 1X μUAV suspended by wire in wind tunnel

The results and aerodynamic diagnosis is discussed in the following section.
4.2.2. Wind tunnel test results

The data from the initial wind tunnel test (plotted as $L/D$ vs. $C_L$) is shown alongside XFLR5-predicted data in Figure 67. The experimental data is shown with data points and dashed lines while the predicted performance is shown with solid lines. The data points were collected by incrementing the trim position on the 1X model. The lift coefficients are referenced to the total projected area of the front and rear wings, 19,355 mm².

![Figure 67: Measured performance of the μUAV](image)

As Figure 67 shows, the model did not perform nearly as well as predicted. First, the $L/D$ reached only 67% of the predicted value. Second, the $C_{L_{\text{max}}}$ was approximately 85% of the prediction. The third and most significant issue can be seen from the lack of data points between $C_L = 0.3$ and $C_L = 0.55$ - the aircraft could not trim in the middle of its $C_L$ range.

4.2.1. Low-Reynolds airfoil nonlinear behavior

It was hypothesized that the gap in the $C_L$ range was caused by nonlinear $C_{L}(\alpha)$ behavior of the front wing due to fabrication inaccuracy of the wax mold, which caused the airfoil have a very blunt leading edge. To investigate this possibility, the profile of the 1X front wing was scanned, digitized, and compared to the ASV02 airfoil (shown in Figure 68 below). The 1X as-built profile is shown in red and the ASV02 is shown in blue.
The 1X wing airfoil shown in Figure 68 has a much larger leading edge radius and an upper surface that is everywhere slightly offset from the ASV02 design. To analyze the discontinuity in lift-curve slope, a side-by-side analysis of the two airfoils was conducted in XFOIL at $Re = 30,000$. The results are presented in Figure 69 below.

In Figure 69, the blue polar represents the ASV02 airfoil performance and the red polar presents the performance of the 1X wing airfoil. The right plot shows that the 1X wing airfoil has a much larger discontinuity in $C_l$ and a slightly lower $C_{l_{\text{max}}}$ than the designed ASV02. The boundary
layer plots, displayed below the two polars, show that at point A (the bottom node of the discontinuity), the airfoil's upper surface is fully laminar and is fully-separated after approximately 50% chord. As the airfoil's $\alpha$ is increased slightly to point B, the flow turbulence amplification becomes sufficient to cause transition in the laminar separated layer, resulting in a laminar separation bubble and a turbulent reattachment, which is able to maintain flow attachment, leading to a much higher $C_l$.

Because of the wire-suspension scheme, it was impossible to measure the pitching moment, but a qualitative illustration is provided in Figure 70 that shows the nature of the trim problem introduced by the nonlinear lift-curve mechanism described above.

![Figure 70: Qualitative effect of nonlinear lift curve slopes to trim and performance](image)

To be stable and trimmed, an aircraft must exist in a configuration where $C_m = 0$ and $\frac{dC_m}{d\alpha} < 0$. As Figure 70 shows, the $C_m(\alpha)$ curve exhibits nonlinear behavior, which makes a large portion of its performance inaccessible.
To elaborate, at low elevator deflections, the $C_m = 0$ line is only intersected by the dotted segment of the $C_m(\alpha)$ curve. This means that the aircraft is stable and trimmed with the front wing having completely laminar flow on the upper surface. This trim state is considered the “laminar trim” state and is colored green in Figure 70.

As the elevator deflection is increased, the aircraft’s trim state moves into the “biphasic trim” state denoted by the red curve on the $C_m$ vs. $\alpha$ plot. In this state, there are three intersections of the $C_m = 0$ line and the $C_m(\alpha)$ curve – the one at the lowest $\alpha$ at which the flow over the front wing is laminar, a medium-$\alpha$ one where the flow is unstable and characterized by an unstable turbulent reattachment region, and a high-$\alpha$ intersection point at which the flow over the wing forms a complete and stable laminar separation bubble and the airfoil operates with a stable region of turbulent flow over the upper surface. In this biphasic trim state, the aircraft is trimmed and stable in two distinct angles of attack, but cannot be stable between them, as $\frac{dc_m}{d\alpha} > 0$, as shown by the positively-inclined dashed portion of the $C_m(\alpha)$ line. Unfortunately, as the vertical lines show, this unstable region also coincides with an $\alpha$ range that yield high $L/D$ performance, rendering that portion of the performance space completely inaccessible.

By increasing the elevator deflection slightly, the $C_m(\alpha)$ curve moves out of the red region and into the blue region, here called the “turbulent trim” state where the only intersection between $C_m(\alpha)$ and $C_m = 0$ is the solid part of the curve which denotes stable, attached turbulent flow over the front wing.

The qualitative result of the non-monotonic $C_m(\alpha)$ curve is a slight “stickiness” at both trimmed angles of attack, high sensitivity to small perturbations near the biphasic state, and a sharp pitching motion as the elevator deflection causes the aircraft to “hop” from one stable state to another.
4.3. Airfoil Refinement for Manufacturing Limits

By examining the ASV02 airfoil and the front wing mold used to create the 1X aircraft, it was discovered that the leading edge region was constructed inaccurately due to machining tolerances. While the tolerance could be improved by switching to aluminum as the tooling material, it was discovered that with the available machining capabilities of the milling machines, it would be mathematically impossible to accurately fabricate both the ASV02 and the ASV10's leading edge regions perfectly with the minimum-usable ball mill size (diameter of 1/32”). Figure 71 illustrates the problem at the tip of the ASV10 airfoil.

![Figure 71: ASV10 leading edge mill size incompatibility](image)

In Figure 71, two 1/32” ball mill positions are shown by the blue and orange translucent circles with their respective tangency locations pointed by the arrows. The two locations are cross-tangent, meaning that the centers of the circles and the tangency locations switch their order as they follow the curvature of the airfoil line. Mathematically, this means that the circles cannot be wholly contained inside the airfoil shape and therefore cannot make the mold lines. The red crescent shows the area that the ball mills will be unable to reach, and is approximately 0.7 % of the chord long, which at the leading edge may lead to poor airfoil performance. While the effect is smaller with the front wing, both airfoils exhibit this machining size problem.

To rectify this situation, a new set of airfoils were produced for the μUAV, designed not only for low-Re performance, but also for manufacturability with the available tools.
4.3.1. Front Wing Airfoils

Using the ASV02 as a starting point, the TLR2 (TLR stands for “Thin Low-Reynolds”) (comparison presented in Figure 72) was developed with the goals of having a leading edge radius large enough for the 1/32” ball mill limit, stability in the lift-curve slope, and better high-alpha performance.

As Figure 72 shows, the TLR2 has a very similar overall shape and retains a similar thin rear section which allows for the aft section to flex and overlap. The TLR2 is slightly thicker (t/c is 4.8% instead of 4.3%) with the additional thickness added around x/c = 20%. The curvature at the nose was designed around a 1/32” ball mill, which exactly fits the curvature of the airfoil on the upper surface at the leading edge, as shown in Figure 73.

In Figure 73, the ball mill circle (black) is exactly tangent to the TLR2 airfoil curvature (blue). While it was impossible to maintain airfoil performance if the lower surface was also made to be tangent to this ball mill size, the lower surface is never in danger of catastrophic laminar separation and is therefore less sensitive to surface inaccuracies.

The lift-curve slope consistency at low Re was improved with the TLR2 airfoil, as Figure 74 below shows.
Figure 74 shows (on the right) the lift-curve slope for the TLR2 airfoil (in blue) and the ASV02 airfoil (in red). The TLR2’s lift-curve slope is slightly more consistent and the transition from the laminar regime to the turbulent regime is more gradual, taking from $\alpha = 2.5^\circ$ to $\alpha = 4^\circ$ to transition between the two, whereas the ASV02 jumps between the two regimes between $\alpha = 4^\circ$ and $\alpha = 4.25^\circ$.

At first glance of the $C_l-C_d$ polar (left side of Figure 74), the increased drag seems to make the TLR2 a slightly less attractive option. A deeper investigation into the boundary layer results shows, however, that the TLR2 handling is expected to be better and more consistent at high $C_l$ due to a significantly healthier boundary layer (lower displacement, momentum thickness, and shape parameter).

The upper surface displacement thickness ($D^*$) and momentum thickness ($\theta$) for both airfoils vs. $x/c$ at $Re = 30,000$, $\alpha = 7.5^\circ$, and $N_{crit} = 9$ are shown in Figure 75. At the bottom of Figure 75, the boundary layer is plotted for the TLR2 (green) and ASV02 (red) showing that the overall boundary layer thickness is less for the TLR2.
Figure 75: TLR2 vs. ASV02 boundary layer performance at Re=30,000, Ncrit=9

The TLR2 also has lower maximum boundary layer shape parameter (H) and slightly lower H throughout the recovery region (x/c from 0.5 to 1.0) at low Re, as shown in Figure 76.

Figure 76: Kinematic shape parameter for ASV02 and TLR2 airfoils, Re=30,000, alpha=7.5
The improvements in handling are also present at higher $Re$; a comparison of XFOIL polars for $Re = 80,000$ is shown in Figure 77 below.

The TLR2 airfoil yields a slightly higher $C_{i\text{max}}$, has less of a drag penalty due to laminar separation bubbles, and maintains a healthier turbulent boundary layer above $C_l = 1.0$.

The improved boundary layer health is accomplished by the adjustments of the pressure distribution such that the buildup of the amplification ratio starts as early upstream as possible and has enough distance to build up to $N_{\text{crit}}$ before reaching the trailing edge and minimize the size of the laminar separation bubble. The amplification ratio vs. $x/c$ is plotted for $Re = 30,000$ and $Re = 80,000$ at $\alpha = 4^\circ$ in Figure 78 below.
In Figure 78, the amplification ratio plot for $\alpha = 4.0^\circ$ is presented. In the upper plot, the blue curve shows the boundary layer amplification ratio at $Re = 30,000$ and the red curve shows the amplification ratio at $Re = 80,000$. These colors also correspond to the boundary layer thicknesses shown graphically below the plot.

For the reasons listed above, the TLR2 replaced the ASV02 as the airfoil used on the front wing of the $\mu$UAV.

### 4.3.2. Rear Wing Airfoils

Two rear wing airfoils were produced to replace the ASV10 - the TLR15 for the main section and TLR16 for the finlets. A separate airfoil is used on the finlet to prevent negative (toe-out) stall, which would greatly reduce $\mu$UAV’s ability to recover from a tumble.
As Figure 79 shows, the TLR15, designed to replace the ASV10 in the center section, is almost twice as thick as the ASV10, with a \( t/c \) of 6.67% instead of the ASV10’s 3.50%. The added thickness allows the TLR15 to have a stiffer trailing edge, making the elevon control surfaces more structurally stable. The leading edge radii of the TLR15 and TLR16 are both much larger than the ASV10’s, allowing them to be produced accurately by the 1/32” ball mill.

A series of XFOIL polars for \( Re = 30,000 \) and \( Re = 80,000 \) comparing the TLR15 and ASV10 are shown in Figure 80 and Figure 81 below.

![Figure 80: TLR15 and ASV10 performance at \( Re = 30,000 \), \( N_{crit} = 9 \)](image1)

![Figure 81: TLR15 and ASV10 performance at \( Re = 80,000 \), \( N_{crit} = 9 \)](image2)
As Figure 80 and Figure 81 show, the TLR15 generally produces slightly more drag than the ASV10, but also provides a slightly broader performance range.

The TLR16 airfoil was designed to have a wide range of operating $\alpha$ to avoid stalling the finlets during a tumble or gust-disturbance. The useful range of $C_t$ goes from $C_{i_{\text{min}}} = -0.7$ to $C_{i_{\text{max}}} = 0.7$.

Figure 82: TLR16 performance at Re=30,000 (dashed) and Re=80,000 (solid)
4.4. Performance of New Airfoils

A new 1X vehicle (henceforth referred to as the 1Xb) was constructed with a front wing using the TLR2 airfoil. The wing mold was milled from aluminum to improve airfoil definition. The fuselage and rear wing construction techniques were unchanged from the first model.

The wind tunnel testing showed that the biphasic stability problem described in Section 4.2.2 was entirely mitigated by the new front wing airfoil, which more smoothly transitions between the laminar and turbulent flow regimes. While the \( C_l(\alpha) \) curve is not perfectly linear, it is monotonic and the slope is much more stable, as shown in Figure 83.

![Figure 83: CL vs control position for the 1Xb in the wind tunnel](image)

As Figure 83 shows, the \( \mu \)UAV's lift-curve slope is more linear and the aircraft did not "hop" between its two flight regimes. The 78-mph run was discontinued at trim position 6 because the aircraft produced enough lift to lift the drag cell portion of the wind tunnel balance off the load cell, saturating the balance. Interestingly, for the same trim position, the aircraft trims at different \( C_l \) at different speeds. This irregularity is believed to be caused by the use of the inaccurately-molded rear wings.

The experimental and XFLR5-predicted lift-to-drag ratio is plotted against lift coefficient in Figure 84. To account for fuselage drag, the \( C_{DA} \) from the fuselage test was linearly interpolated and included in the \( L/D \) calculation.
Figure 84: Wind tunnel results of the 1Xb prototype

Figure 84 shows that the 1Xb prototype’s predicted performance matches well with the predicted \( L/D \) polar, although generally tended to be approximately 10% lower, which can be attributed to interference drag and the drag due to the folding geometry and control mechanisms of the rear wing. The peak \( L/D \), from this data, is considered to be 10.
4.5. Detailed Stability and Control Analysis

To determine the stability properties of the μUAV, the aircraft was modeled in XFLR5 and analyzed using vortex lattice method (VLM). The weight of the μUAV, the CG location, and moments of inertia were calculated using a detailed SolidWorks model with measured component weights. Because the fuselage is cambered and non-axisymmetric, it is expected to provide some lift and is therefore modeled as a low-aspect-ratio wing. This modeling method is expected to predict the pitching moment contribution from the fuselage element [23].

In this type of analysis in XFLR5, the aerodynamic surfaces are modeled as a lattice of horseshoe vortices aligned with the wing camberline [24]. At the edges of the control surfaces, the vortex panel is split and the control surface panels are deflected by the control input. This deflection can be seen visually at the trailing edge of the rear wing in Figure 85.

![Figure 85: XFLR5 model of μUAV showing an elevator-up configuration](image)

4.5.1. Establishment of Regime

Because VLM is most accurate in the linear regime (and is untrustworthy outside of it), it is first necessary to establish the range of flight attitudes in which the aircraft is behaving linearly and the results are valid. To do this, a range of elevator deflections ($\delta_e$) was investigated under the mixed-viscous-inviscid mode in XFLR5. The purpose of the run is to establish a range of $\delta_e$ where no aerodynamic surfaces are operating in a stalled $\alpha$ and $Re$ regime.

In the simulation, $\delta_e$ and $\alpha$ are set at fixed values. The vortex-lattice system is then solved for a $C_t$ and $C_m$. The $C_L$ along with wing area, weight, and air density determines the speed of the aircraft operating at the current $\delta_e$ and $\alpha$ pair. Based on the speed, kinematic viscosity, and geometry, the $Re$ of each surface is calculated, and therefore drag data can be interpolated from the XFOIL polar library pre-computed for each airfoil. If the vortex-lattice-generated $C_t$ of a
surface is too high and is outside of the interpolation region, the analysis omits that calculation point since that part of the μUAV has stalled. In this fashion, the analysis only retains operational points at which the surfaces are operating in their non-stalled regimes. This process is repeated for $\delta_e$ from $4^\circ$ to $-8^\circ$ and $\alpha$ between $0^\circ$ and $15^\circ$ intervals. The sign convention of $\delta_e$ is such that the angle is positive trailing-edge-down.

The relationship between $C_m$, $\alpha$, and $\delta_e$ as predicted by the mixed-viscous-inviscid method is shown in Figure 86 below. $C_m$ is referenced to the front wing’s real area and chord ($8750\text{mm}^2$ and $35\text{mm}$, respectively).

![Figure 86: Cm-alpha-elevator curves for the μUAV](image)

As Figure 86 shows, the μUAV is trimmed and stable for a $\delta_e$ range between $-8.0^\circ$ and $2.0^\circ$. The purple line showing $\delta_e = -8.0^\circ$ does not cross the $C_m = 0$ line, and therefore is unable to trim in steady flight without partial stall. For all $\delta_e$, the aircraft started experiencing stall at $\alpha = 12^\circ$ and therefore no data is available beyond that $\alpha$.

This range of $\delta_e$ yields speed predictions that corresponds well with the desired speed range. Figure 87 presents the predicted $C_m$ vs. $V$ at $30,000$ ft.
Figure 87 above shows that the μUAV’s cruise speed range is between 52 knots and 80 knots, meeting the 58 knot speed specification set by the mission requirements.

### 4.5.2. Stability Results

Having set the valid range of $\delta_e$, a stability eigenvalue analysis was conducted using this vortex lattice model. The model was analyzed with a sweep of elevator angles from $\delta_e = 2.0^\circ$ to $\delta_e = -6.0^\circ$ with four altitude levels from sea level to 30,000 ft to ensure stability over the entire flight regime. For each point presented in the stability analysis below, the aircraft is trimmed in level flight, so speed varies with $\delta_e$ and altitude. The results of the stability analysis are presented as longitudinal mode eigenvalues (shown in Figure 88) and lateral mode eigenvalues (shown in Figure 89).
As Figure 88 shows, the μUAV has a well-damped short-period and stable phugoid modes. The short-period is predicted to have an oscillation frequency of between 5.4 Hz (at sea level, $\delta_e = 2.0^\circ$) and 3.7 Hz (at 30,000 ft, $\delta_e = -6.0^\circ$). The phugoid result magnitudes are not very meaningful since the model is purely inviscid and the total drag needs to be taken into account to get a real estimate of phugoid damping.
Figure 89: Lateral eigenvalues of μUAV with varying altitude and elevator deflection

Figure 89 shows the aircraft has a well-damped roll subsidence and a fast, stable Dutch roll mode, with a frequency of between 2.2 Hz ($\delta_e = -6.0^\circ$) and 3.0 Hz ($\delta_e = 2.0^\circ$). The Dutch roll mode is insensitive to altitude. The only unstable mode the aircraft is predicted to have is the spiral mode, which is predicted to have an eigenvalue of between 0.03 and 0.18. This maximum value of 0.18 occurs at sea level and $\delta_e = -6.0^\circ$, as Figure 90 shows. Luckily, the spiral tendency drops off sharply at all altitudes with increasing speed.
The predicted time-to-double-perturbation of the maximum-spiral-amplification point is 3.8 seconds, while the time-to-double takes 20.6 seconds to occur at the minimum-spiral amplification point. Both these times are well-within the response times of the autopilot, however, and the aircraft would not usually fly near stall, so the maximum-spiral-amplification poses little operational hazard to the aircraft. Based on this analysis, the airframe configuration and stability characteristics were deemed satisfactory, and the configuration design was finalized.
Chapter 5: μUAV Detailed Design and Construction

This chapter focuses on the detailed design and construction techniques of the production aircraft configuration as finalized in Chapter 4. Because of the μUAV’s size, many of its parts are constructed with as few as one or two layers of thin composite fabric and are limited by the availability of material sizes. As a result, special care was to be taken to design the components and lay-up processes for manufacturability.

All aircraft components were constructed using a molded wet layup process on aluminum molds built by Sam Range. The epoxy system used was the MGS L-285 epoxy resin and MGS 285 hardener, chosen for its low viscosity and high cure strength achieved with high-temperature curing [25]. High-temperature cure is accomplished with the use of a thermocouple-controlled hot plate which heats the mold directly, providing even heating for the lay-up. A combination of unidirectional and woven fabrics are used in the construction of the μUAV; all fabrics are plain-weave.

5.1. Fuselage Shell

To avoid creating a Faraday cage which would block the communications signals from the antenna, the fuselage is constructed from Kevlar due to the material’s nonconductive nature. Kevlar was selected over fiberglass for higher strength-to-weight ratio. The fuselage is composed of three main components – two structural shell halves and a flexible fairing, as shown in Figure 91.

![Figure 91: Fuselage shell features](image-url)
As Figure 91 shows, the two primary shell halves are split at an incline from the leading edge to the propeller shaft exit point. This split line allows the fuselage to be simply-projected on both sides, resulting in a 1-dimensionally-protruded geometry that can be constructed on a 3-axis CNC. The front wing fairing is split from the top shell by the front wing split plane (highlighted in a translucent blue) in which the front wings swing. The fairing and two shell pieces together make up the exterior shape of the F2 fuselage.

5.1.1. Shells

The two large shell pieces carry the primary loads of the fuselage. The front wings are attached to the top shell and all other system components (propulsion, controls, electronics) are attached to the bottom shell.

Figure 91 shows that, in the top shell, there are two vertical columns used to support the front wing’s hinges. The integration of the front wing structure and the loads transmitted to the fuselage shell are shown in the cross-section diagram below (Figure 92).

![Figure 92: Front wing – fuselage integration showing forces acting on the fuselage shell](image)

The front wing structures (colored blue in Figure 92) transmits the root bending moment into the fuselage via the support columns (orange) which are lined with carbon fiber to bear the rotation of the wings. The bending moment puts the fuselage shell in compression at the top and in tension on the bottom. The battery pack helps to provide support for the compressive forces on the fuselage shell.

The acceleration ejection forces on the μUAV are transferred into the fuselage through two foam blocks (red) that are placed to either side of the finlets. These foam blocks push on the corners of the fuselage shell (magenta), as shown in Figure 93.
By placing the heaviest components of the μUAV toward the nose and ejecting the aircraft backwards, the structure required to bear the acceleration load of the internal components is minimized. Because the fuselage’s shape is domed at the supported region, the fuselage shell exhibits “eggshell” strength, which resists deformation and allows the fuselage shell to support the acceleration load of the battery pack during acceleration, which is approximately 86 lbf.

The motor cage assembly (which contains the autopilot, motor, and servos) is supported both by the lower shell walls and floor. The acceleration force is transferred to this assembly by the alignment of Kevlar fibers that run longitudinally. The primary fabric fiber directions for the top and bottom shells are shown in Figure 94.
Because Kevlar has poor compressive strength, three of the four layers are aligned with fiber directions as shown in Figure 94. On the top shell, the fabric is aligned to provide compressive strength for the wing bending moment, and on the bottom, the fibers are aligned longitudinally to support the acceleration load and tension load. The fourth layer is aligned at 45° to this primary layer.

The Kevlar layers are laid-up on a double-sided mold, as depicted in Figure 95.

To mold the fuselage shell for the μUAV, a double-sided mold with a silicone plug/liner was developed. The Kevlar fabric is wetted with epoxy and laid into the inside of the mold base shown in Figure 95. A silicone plug is then inserted to press the fabric down onto the mold base. The plug is approximately 10 mm thick, which allows it to be deformed and placed inside the mold without disturbing the fabric layers. The top side of the aluminum mold, named the “boat” is then pushed down into the silicone plug and is tightened onto the mold base with bolts at either side. The silicone plug distributes the compressive force evenly, compressing the composite fabric to approximately 560 psi. Excess epoxy flows out of the mold between the boat and the base and, along with the fabric overrun fabric, is discarded. Figure 96 shows a photograph of the top shell mold.
Also shown in Figure 96, two steel pins protrude upwards from the mold base; these pins are used to create the front wing pin mounting columns. Carbon fiber sleeves are slipped over these pins after the Kevlar is laid in the mold base. The silicone plug is then inserted, and the top half of the mold is inserted and tightened down to provide pressure.

After the part cures, it is removed from the mold and the steel pins are removed from the part. A top shell produced from the mold is shown in Figure 97. The dark band that is seen on the outside of the shell is the carbon fiber sleeve column on the aircraft’s left side, visible due to the translucency of the Kevlar.
5.1.2. Front wing fairing

The fairing (shown in Figure 98 serves two purposes – first, it continues the smooth lines of the fuselage once it has been broken by the swing-plane of the front wings. Second, it provides a mounting location for a stopping feature for the front wings.

As shown in Figure 98, a flat “wing-stop” feature, constructed from carbon fiber, is molded at the front. Aside from the wing stop, the fairing is constructed from 2 layers of Kevlar. For minimum drag, the fairing must seal against the fuselage. Therefore, the mold for the fairing does not follow the external lines of the fuselage; instead, the mold shape is warped downwards (such that the trailing edge of the fairing is 3mm lower than the split plane of the wing), creating a shape that pre-loads the fairing when it is glued to the fuselage shell.
5.2. Front Wings

The front wings of the μUAV are each 128.5 mm by 35 mm and constructed from a Rohacell® -carbon fiber composite structure. The geometric features of the (right) wing are shown in Figure 99 and Figure 100 below; the left wing is identical but mirrored.

As Figure 99 shows, the wing is a straight-extruded section of the TLR2 airfoil with a hole near the root in which the wing mount pin is attached. There is a shallow groove that extends from the hole for the placement of the “torsion bundle”. The trailing edge is thin and flexible and made of a single layer of 3.5-oz/yd² carbon fiber fabric. Figure 100 shows the underside of the wing. The “filleted pin boss” (4.4 mm tall) is used to transfer the wing bending and lift loads to the pin, and a “root support boss” is used to spread the forces chordwise at the root.
5.2.1. **Hinge Design**

The front wing of the μUAV is designed with a compact hinge system; a diagram is shown in Figure 101.

![Figure 101: Wing deployment mechanism](image)

In Figure 101, the wing is glued to the wing pin (highlighted in translucent blue). This pin carries the wing root bending load. Because most of the bending strength in a rod is provided by the walls, it was possible to use a hollow tube and store the torsion spring (called the "torsion bundle" inside). At the top, the torsion bundle is glued into the groove in the wing. At the bottom, the bundle is secured to the fuselage's wing mounting column structure. The bundle therefore twists inside the wing pin and supports the lift forces of the μUAV. The wing is stopped at the flight configuration by a stoppage feature on the fairing.

A bundle was used instead of a single torsion rod due to the high angle of deflection required of the wing, the length available to the spring, and the yield stress of spring steel. To have some zero-deflection restraining force, the angle of deflection required of the torsion spring is approximately 100° and the torsion rod must be less than 30 mm long. The yield stress of steel limits the radius of a steel torsion rod to approximately 0.15 mm. Any rods thicker than this size would yield and be unable to deflect the angle necessary. While a single strand of 0.15 mm-radius steel wire is too weak to actuate the wing, a set of multiple wires, aligned in parallel, allows the spring torque to be multiplied such that a reliable deployment can be achieved.
With these considerations, a 0.083" OD wing pin was selected. This hollow wing pin houses eight pieces of 0.012"-diameter music that form the torsion bundle. A cross-section of the wing mechanism showing integration with the fuselage is shown in Figure 102 below.

5.2.2. Construction

The wings are constructed using a two-piece aluminum mold as shown in Figure 103.

As shown in Figure 103, the mold has six alignment pins (not pictured) and a mandrel which produces the hole for the wing pin. The jackscrews are used to compress and open the mold. The layers of the wing layup are shown in Figure 104.
Figure 104: Wing layup layers

Figure 104 shows a cross-section view of the arrangement of the layers of composite that form the front wing. In the center of the wing, a Rohacell® core (grey) is used due to its compressive rigidity and low density. The core is sanded to a triangular profile before compression in the mold. To either side of this core is carbon fiber tow (red) which also wraps to form the wing boss, as will be discussed in detail in Section 5.2.2.1. Outside of this layer is a layer of unidirectional 3.0 oz/yd² carbon fiber (orange) which forms the spar caps of the wing with fibers running spanwise.

The outside, a layer of 3.5-oz/yd² carbon fiber cloth is used. The fabric of this outer layer is aligned such that the strands are at ±45° to the spanwise axis. From the leading edge to approximately 70% x/c, the core, tow, and spar caps are fully-enclosed by the skin. The ±45° fiber alignment gives the wing very high torsional rigidity. Behind 70% x/c, the fabric layers taper down to a single layer of carbon fiber fabric. The ±45° fiber alignment allows the trailing edge to be flexible. The cloth layer is stopped just short of the leading edge, which is discussed in Section 5.2.2.2.
5.2.2.1. Wing boss layup

For the construction of the wing boss, the carbon fiber strands have to be designed carefully to achieve the desired strength of critical features. Since the layup has strength in the fiber direction and very little strength across it, it is necessary to run fibers along the span into the boss to transfer the bending moment into the wing pin. Ideally, as shown in Figure 105, the fibers (red) would run along the wing and continue into the boss.

![Figure 105: Conceptual fiber direction for wing boss strength](image)

To provide this type of fiber arrangement, the following wing boss layup scheme (shown in Figure 106) was developed.

![Figure 106: Carbon fiber wrapping scheme for front wing fillet boss](image)

Figure 106 shows the wrapping scheme used to create the wing boss. First, the polished steel mandrel is inserted into an epoxy-wetted 0.1”-diameter carbon fiber sleeve (illustrated in green). The sleeve fibers align diagonally, as indicated by the cross-hatch lines. To bring more fibers
down into the boss, a bundle of wetted unidirectional carbon fiber (red) is laid up parallel against the mandrel, as indicated by the red lines.

The fibers in the sleeve and unidirectional fibers alone, however, are not strong enough to withstand the wing root bending moment - without fibers that provide hoop-strength to the boss, the boss fails by splitting of the epoxy matrix between the carbon strands. To supply the hoop strength, the unidirectional fibers are then used to wrap the parallel section of the unidirectional fibers and sleeve. This finished group is then inserted into the mold to create the wing boss feature. The unidirectional carbon is extended into the wing to attach to the spar caps.

5.2.2.2. Leading edge geometry

Another problem that arose with constructing carbon fiber on a small scale is non-negligible fabric thickness. As Figure 107 shows, the 3.5 oz/yd² fabric (0.007” thick) used for the wing skin is very large compared to the leading edge feature of the airfoil. If a gap between the two sides of the mold were left for cloth overrun, the airfoil leading edge region would be produced inaccurately, which may lead to poor performance.

Using a thinner fabric would reduce the problem, but this has other detrimental effects. The thinnest available carbon fabric, a 2.0-oz/yd² (0.005” thick) cloth was tested, but this fabric was too weak and floppy to use by itself and introduced voids and pinholes in the rear section of the wing due to the sparsity of the fibers.

The method developed to handle the thickness of the carbon fabric was not to leave a gap for the fiber overrun at all. Instead, the molds are brought to 0 gap thickness at the leading edge, as shown on the left side of Figure 108 (below).
The wing skins are stopped short of the leading edge (as shown in Figure 104), and a bead of "carbon flox" mixture is laid down at the leading edge.

The carbon flox mixture is created by chopping unidirectional carbon fiber to approximately 1-mm lengths and mixing it with epoxy to create a gelatinous suspension of fibers. Because it is a suspension rather than an ordered fabric structure, the carbon flox is able to flow out between the top and bottom molds. When cured, the epoxy and carbon form a harder matrix than the epoxy alone. This material provides small-length strength transfer so the torsional rigidity of the wing's enclosed cross-sectional area can be maintained.

At the rear of the airfoil, a 1-layer-thick gap was left between the two sides of the mold, as shown on the right side of Figure 108. The one layer of 3.5-oz carbon fabric runs off and is cut off and discarded after de-molding.

5.2.2.3. Resulting wing

Figure 109 shows the wing produced by the molding method (upper) and overlaid with the TLR2 airfoil (lower). The airfoil reproduction is very accurate, with average thickness deviance of less than 0.001".
Figure 110 below shows the two front wings stacked together at the rear end of the μUAV. As the arrow points out, the single layer of carbon fiber that forms the rear 30% of the airfoil is able to deflect, allowing the two wings to overlap.

![Deflected Trailing Edge](image)

**Figure 110:** Stacked front wings showing trailing edge deflection
5.3. Rear wing

The rear wing has a chord of 30 mm and length of approximately 148 mm from the centerline to the tip. The right and left wings are nearly symmetric with the left one being longer by approximately 2 mm for folding purposes. The main features are shown in Figure 111 below.

As shown in Figure 111, the wing root is attached to a flat mounting plate. The mounting plate, 15 mm wide by 30 mm long, has a hole in the center for mounting to the wing turning mechanism. At the back of this plate, a stop feature is built to stop the rear wing as it swings backwards during deployment. The elevon, positioned adjacent to the mounting plate, is 7 mm in chord and 70 mm in span. The airfoil section used from the root to the dihedral break is the TLR15, which at the dihedral break lofts into the TLR16, which is used for the finlet. The dihedral added by the break is approximately 9°.

5.3.1. Deployment mechanism

All deployment mechanism components were designed based on standard-sized steel tubes and stock springs while still being strong enough to withstand flight loads.

The deployment mechanism for the rear wings is mounted inside the fuselage, under the propeller shaft, as shown in Figure 112. The spring shown in the features is 8.3 mm tall.
The mechanism works by pushing the wings against each other, swinging them backwards. Both wings rotate freely from the fuselage until the wing-side stop feature (shown in blue in Figure 113) hits the fuselage-side stop point (shown in red). At that point, the hitting wing is stopped and the other wing continues to swing backward until it, too, hits the stop feature. In this way, the two wings "clamp" to the fuselage-side stop feature.

Figure 113: Wing-fuselage stop features

An exploded view of the rear wing deployment mechanism is shown in Figure 114.
As shown in Figure 114, the rear wing mechanism is composed of a stack of components. These components form three concentric layers. The innermost layer is the left wing subassembly (colored red for clarity) which is composed of the smallest tube ("tube1") and the left wing. The tube is glued to the wing via the hole in the mounting plate.

The second layer is the right wing subassembly, which is composed of tube2, a flange, and the right wing, all colored green in Figure 114. The flange and right wing are glued to tube2 such that they vertically capture the third concentric layer, composed of the bushing (grey), the elevon "guideplate", and the fuselage shell.

The bottom of the torsion spring (magenta) is then attached to the flange of the right wing subassembly, and the top of the spring is attached to tube1 of the left wing subassembly. This point of the spring both rotates and supports the left wing subassembly vertically.

Figure 115 shows the deployment mechanism collapsed with the same colors (the 3rd concentric layer is shown as blue in this figure). As shown in Figure 115, the left wing subassembly (in red) is the tallest, going from the very top (where it is attached to the torsion spring) to the bottom, where it is attached to the left wing. The right wing subassembly is shown in green and captures the blue section (the bushing, fuselage shell, and elevon guideplate).
5.3.2. Elevon Control Mechanism

A Sprung Rotary-Drive System (SRDS) was designed for the μUAV’s elevon control surfaces which takes inspiration from the rotary drive system (RDS) used in discus-launch gliders [15].

Figure 116 below shows the SRDS system. A 0.012"-diameter piano wire spring (red) is glued at the elevon hinge line (one tip to the wing, the other to the elevon) to provide a force that deflects the elevon upwards. The center of the spring is unsupported.
A control "finger", composed of a 1/16" diameter wire, is shown in Figure 116 by the black arrow. The finger is servo-actuated in torsion by a Hitec HS-35HD servo as shown in Figure 117. The finger pushes the elevon down by sliding across the trailing edge with its circular wall such that the tip of the finger never touches the elevon. This control design minimizes the friction between surface and actuation mechanism and prevents gouging of the control surface by the tip of the finger. Prior to deployment, the elevon guideplate keeps the elevons at zero deflection so they do not jam against the control fingers.

![Control fingers](image)

**Figure 117: SRDS servo drive**

Three elevon deflections are shown in Figure 118 with the finger contact point circled in green.

![Max upward deflection](image)
![Neutral](image)
![Max downward deflection](image)

**Figure 118: Control surface deflection system**
As shown in Figure 118, as the finger slides on the elevon, the contact point moves in the spanwise direction. Due to the different heights of the right and left wings and the nonlinear nature of this mechanism, the control mapping from servo angle to control surface angle is nonlinear, as Figure 119 shows. This nonlinearity in control mapping is taken into account while programming the autopilot system.

![Elevon deflection angle vs servo actuation angle](image)

**Figure 119: Elevon deflection vs. servo actuation angle**

Figure 119 also shows that the maximum upward deflections on the right and left side are not equal. On the left side, due to the lowered wing, the elevon is capable of a maximum of 67° of upward-deflection whereas the right side is capable of 35°. Both elevons are capable of being deflected downwards to -17°.

### 5.3.3. Construction

The rear wings of the µUAV are constructed using nearly the same method as the front wings (as Figure 120 shows) with three major differences.

![Layup layers of rear wing](image)

**Figure 120: Layup layers of rear wing**
First, since the attachment feature is a mounting plate instead of a wing pin boss, no carbon wrapping scheme is necessary for mounting, and no tow needs to be integrated into the layers of the wing lay-up. Second, to achieve a rigid elevon, the elevon is made from two layers of 3.5-oz/yd\(^2\) carbon fiber fabric instead of one layer. Third, to make a "live" hinge, a single layer of 1.7-oz/yd\(^2\) Kevlar is laid-up on the upper surface. This layer is stretched in the spanwise direction such the fibers are not aligned perpendicular to each other, but rather, at approximately 60°, as shown by Figure 121. Biasing the fibers of the Kevlar layer softens the hinge.

![Figure 121: Rear wing prototype with showing embedded Kevlar hinge](image)

The elevon is never detached in assembly. To make the elevon, chordwise cuts are made at the ends, then a cut is made on the underside of the wing using a utility knife until the control surface can be cracked and bent, at which point the magenta area (shown in Figure 120) is sanded out. The result is an elevon that is rigidly-attached to the wing but is still pliable enough to be used with the SRDS system.

Figure 122 below shows the guideplate, which is constructed from layers of 3.5-oz/yd\(^2\) carbon fiber fabric. Figure 122 also shows the wing tubes and the flange component of the right wing subassembly. The full rear wing assembly (with an older version of rear wings without the Kevlar live hinge) is shown in Figure 123.
Figure 122: Elevon guideplate and wing hinge tubes

Figure 123: Full rear wing assembly
5.4. Motor Cage

The support structure for the motor, servos, and autopilot board is called the “motor cage”. The cage holds the motor in the middle, the servos to its sides, and the autopilot board on the top, as shown in Figure 124. The motor cage also acts as the mid-ship structural bulkhead to support the fuselage shells.

![Motor cage with attached components](image1)

Figure 124: Motor cage with attached components

The motor cage must support the acceleration loads of these components during ejection, as shown in Figure 125 below.

![Motor cage acceleration loads](image2)

Figure 125: Motor cage acceleration loads
The components held by the motor cage total approximately 40 grams, which at an ejection force of 300 G is a load force of 117.7 N, or 26 lbf. To support this load efficiently, the motor cage is designed with fiber orientations as shown in Figure 126.

![Figure 126: Motor cage fiber directions](image)

Due to the acceleration load in the -X direction (shown by the axes in Figure 126), the forward bulkhead is put in compression and the rear bulkhead is put in tension. To support these loads, the fibers are aligned in the Z direction and attach to the floor of the lower shell of the μUAV and to the side walls. The two webs that connect the two bulkheads (pointed with red arrows) support the acceleration load in shear. Accordingly, the fibers on the webs are aligned ±45° in the X-Z plane.

To create this structure, a 5-piece mold is created with three center sections and two large walls, illustrated in Figure 127.
As shown in Figure 127 above, the motor cage is composed of five pieces of 3.5-oz/yd² carbon fiber fabric. The aluminum blocks are shown as clear (for fiber clarity) and white. The colored dashed lines at the top of each block show the wrapping design of the piece of carbon fabric with the corresponding color below. The wrapping diagram is summarized at the top-left corner. The cross-hatches describe the alignment of fibers on each piece of fabric.

The two outer pieces (colored green and grey in the figure) provide the fibers aligned with the Y and Z axes. The center piece (colored red) is laid up with fibers running at ±45° to the Z axis, wrapped twice around the center block. To either side, two pieces (blue) are wrapped with the same fiber orientation. The red and blue fabric components compose the web features of the motor cage.

The motor cage is molded and then post-processed to generate the mounting and clearance geometry for the motor, servos, and autopilot board. A prototype of the motor cage (missing the mounting points for the autopilot) is shown in Figure 128. This prototype has a mass of 2 grams.
Figure 128: Constructed prototype motor cage
5.5. Power System

This section focuses on the design of the battery pack, the design of the propulsion system, and an estimate of endurance performance. Through the effort in manufacturing and structural design, the aircraft’s weight estimate has reduced from 240 grains to 220 grams; the updated value is used for the propulsion analysis below.

5.5.1. Battery pack

Lithium polymer batteries suffer both power and capacity degradation at low temperatures through the reduction of cell voltage. A discharge capacity vs. discharge temperature plot for a lithium-polymer battery illustrates this degradation in Figure 129 below. Figure 129 also shows that lithium polymer cells can tolerate higher operating temperatures without losing significant performance.

![Figure 129: Discharge capacity vs. discharge temperature for a lithium polymer battery [26]](image)

Lithium-polymer batteries also exhibit self-heating, a result of the current drawn and the internal resistance of the battery. A cold battery, if discharged at high current, will increase its temperature and therefore increase the cell voltage. If constant power is required from the battery (as would be the case for a cruising aircraft), an increase in voltage leads to a decrease in the current required, which produces an automatic feed-back loop to control the temperature of the battery. The self-heating process is therefore stable and self-regulating, and although overall capacity is still reduced, the available capacity is not as low as Figure 129 first appears. Figure 130 below shows the increased-voltage effect of self-heating of an un-insulated lithium-polymer battery at 0.3 C.
To deal with the cold temperatures at altitude, the strategy used for the μUAV was to design a battery system that would maximize the temperature gained due to the internal heating effect. To do this, a foam protective shell is designed to encase the batteries and the ESC together, as shown in Figure 131. On the left side of the figure, a cross-sectional view is shown with the foam colored blue, the batteries grey, and the ESC red. An isometric view is shown to the right.

This foam shell provides thermal insulation to the batteries, allowing the heat generated by self-heating to build up and increase battery temperature. A thin sheet of aluminum that is thermal-pasted onto the ESC and bent to contact the batteries allows the ESC to use the batteries as a heat sink, thereby warming the batteries with the waste heat from the ESC. These two combined methods are designed to maximize the battery’s temperature. As there is little penalty for high temperature, this scheme provides a way to maximize battery performance at all altitudes.
The battery pack is created by placing the batteries and ESC in a mold, into which the foam mixture is poured and then hardened. The molded pack conforms to the interior lines of the μUAV shell.

![Figure 132: Mold diagram of foam shell](image)

While the foam shell will maximize the battery’s temperature, it does not guarantee that the battery pack will have sufficient temperature and power to sustain flight at the start of the mission. Unfortunately, battery performance data for temperatures below -20° are unavailable and perhaps untested for this battery cell. Further testing is needed to characterize the battery’s performance at these temperatures and whether the battery pack design can warm the cells quickly enough to perform the mission. If the insulating shell and self-heating methods are insufficient, it may be necessary to add heaters to the flare cartridges or limit the cold-soak time at high altitude.
5.5.1. Power system and performance

The motor and ESC were selected using a discrete study of available units and the propeller was sized using JavaProp [28], a blade element theory-based propeller design and analysis program. An image of motor, propeller shaft, and propeller is shown below in Figure 133 below. For this flight model, a GWS-3030 propeller was used as the custom propeller has not yet been constructed.

![Image of motor, propeller shaft, and propeller](image)

Figure 133: Constructed propulsion system

The NeuMotors Proton 012-30-4000 motor selected for the μUAV for its high efficiency, packing compatibility, RPM-range, and consistent build quality. The kV is tested to be 4375 [19], which at the battery's 11.1V gives it a maximum unloaded RPM of 48,600. The motor weighs 15 grams - while not the lightest motor the μUAV could use, it makes up for the weight by having a higher efficiency, which peaks at approximately 79% at 3 amps.

The ESC selected was the EXCEED Volcano 6A as it is the smallest and lightest ESC that could provide the full range of power the motor could draw in flight. It also provides sufficient power from the 5-volt BEC (battery eliminator circuit) to run the autopilot and control surfaces.

The propeller's maximum diameter is limited to the size of the envelope available in the flare cartridge, as shown in Figure 134. The maximum propeller diameter is 66 mm.
A propeller for the μUAV is designed and analyzed in JavaProp. The propeller geometry and performance are calculated using the minimum-induced-loss method under the following assumptions:

1) The aircraft is cruising at 30,000 ft and 30 m/s.
2) The airframe has a mass of 220 grams and an $L/D$ of 10.
3) The propeller rotational speed is assumed to be is 36,400 RPM, 75% of the motor's maximum speed.
4) The ARA-D 6% thick airfoil is used.

A plot of the propeller shape from JavaProp is shown below in Figure 135.

![Diagram of propeller shape](image)

**Figure 135: JavaProp-generated propeller**

The propeller is predicted to have an efficiency of 75% and a shaft power draw of 8.65 W.

From this calculation, it is found that the motor needs to draw approximately 1.15 amps at an efficiency of 68%, which puts the operating point at the red circle shown in Figure 136.
Combined, the propeller and motor are 51% efficient. Assuming an ESC efficiency of 90% (no ESC data test is available), and an assumed power-drawn by the autopilot of 3 W this yields a power-draw from the batteries of 17.1 W, or 1.9 W per cell.

Applying this power discharge rate to the data collected during cell testing, it is found that the cell is able to deliver approximately 98% of its maximum energy at this draw rate.

Dividing the total onboard energy by 17.1 W yields a maximum endurance of 80 minutes for warm-battery flight.
5.6. Assembled Aircraft

Figure 138 and Figure 139 below show the constructed aircraft in its stowed and deployed configurations, respectively.

As shown in Figure 138, the front wings are stowed beneath the front wing fairing and the rear finlets are stowed at the nose.

Figure 139 shows a prototype of the µUAV in its deployed state with the elevons at maximum upward deflection. A GWS-3030 propeller was used with this prototype.
Chapter 6: Protective Case and Deployment Sequence

This chapter presents the design and construction of the protective case and the ejection trajectory simulation. Much of the design and prototyping of the protective case was conducted by undergraduate researcher Jonathan Allen.

6.1. Case design

The case was designed to protect the µUAV from the ejection loads while keeping clear of the deployment so the µUAV does not get stuck. An overview of the case design is shown in Figure 140 below, showing the major components: the two sides, electronics and mechanisms, a spring-loaded backplate, and protective foam dispersed throughout the box.

![Figure 140: Major protective case components](image)

At ejection, the MJU-10/B uses a pyrotechnic charge to push on the µUAV package. The package is compressed against the cap, which has a minimum burst-force of 125 lbf; this compressive load must be withstood by the package. As the cap opens, the pressure from the pyrotechnic cartridge accelerates the µUAV package at approximately 300 G for 0.01 seconds, ejecting it at approximately 55 mph.

To deal with these forces, the case is designed to direct all the forces around the fuselage (illustrated in Figure 141) via a shell made from 8 layers of 5.7-oz/yd² fiberglass. Fiberglass is used instead of Kevlar since the forces are compressive, and Kevlar's compressive strength is only approximately 10% of its tensile strength [29], making fiberglass a more suitable option. Fiberglass is also preferable to carbon fiber in this application since its greater flexibility makes it less like to fracture while busting the end-cap of the flare cartridge.
The external shell of the case supports the entire compressive force of the ejection. The internal holding features of the case (shown in Figure 142) are designed such that the μUAV fuselage is never loaded by the 125-lbf compressive force.

As Figure 142 shows, the nose of the μUAV is supported by two foam "noseblocks" and the rear is supported by a thin aluminum flange on top and bottom. If the compressive force deforms the case shells, the flange deflects as to not induce any loads on the μUAV fuselage.
The noseblocks dampen the 300-G acceleration by providing a crumple zone that supports the μUAV with the strong, domed section of the fuselage. The foam selected for the noseblocks has a compressive yield strength of 60 psi, which yields under the μUAV's acceleration force of 146 lbf and is crumpled to approximately 50% of its original size.

6.2. Case opening mechanism

The protective case's opening mechanism is shown below in Figure 143.

![Figure 143: Protective case opening mechanisms](image)

The opening is triggered by the microcontroller which actuates the Dymond D47 servo to pull on two carbon fiber rods. The rods slide within sleeves that are affixed to the fiberglass shell. The tips of these rods act like deadbolts for the "tongue" which is mounted on the right side of the case (highlighted in blue in Figure 144 below). When the carbon rods are retracted, the tongue is freed, and the springs attached at the rear of the protective case force the case open.

![Figure 144: Right side locking tongue](image)
The end caps of the protective case (shown in Figure 145 below) are milled from solid bars of aluminum. The slots in the foreground are cut into the block to allow the tongue to be captured by the two carbon rods. In the back (not visible from this angle), there are holes for 0.125” alignment pins that lock the two end caps together when the protective case is closed.

Figure 145: Milled case end cap

Figure 146 below shows a finished protective case open, and Figure 147 shows the case closed around a µUAV, forming a complete µUAV package.

Figure 146: Fully-assembled protective case, open
Figure 147: μUAV inside protective case
### 6.3. Launch Simulation

A simulation program was written in MATLAB to predict the trajectory and speed profile of the µUAV package after its ejection from the mothership. The simulation applies drag estimates and basic kinematics and integrates them to determine the location of the package. Using the speed profile, a case-opening delay is selected.

A mothership airspeed of 250 KTAS in the positive-\(x\) direction was used for all cases, and the altitude of the mothership is referenced as \(y = 0\). Due to the low-\(Re\) of the case, a \(C_D = 1.0\) assumption was used with reference to the frontal area of the protective case. The drag streamer with dimensions 1.5 m \(\times\) 62 mm, is used with a \(C_D\) of 0.08 [30]. The mass of the µUAV package is estimated at 390 grams. To compare the motion of the µUAV package at different altitudes, multiple air densities are investigated, from sea level to 30,000 ft, spaced at 5,000-ft increments.

The predicted geometric flight path for the µUAV package is shown in Figure 148.

![Figure 148: Predicted µUAV flight path](image)

In Figure 148, the horizontal axis is the direction along the flight vector of the mothership while the vertical axis shows how far the µUAV package travels in relation to its launch altitude. At sea level air density, the µUAV package falls approximately 220 meters in altitude in 10 seconds whereas at 30,000 ft, the µUAV package falls approximately 325 meters. This result is expected since higher air density results in higher drag which slows the descent of the µUAV package at lower altitudes.
Figure 149 below shows the predicted total speed vs. time for various altitudes. “Total speed” accounts for the speed imparted to the μUAV package by the mothership. Again, the terminal velocity of the package is slowest at sea level and rises with altitude.

![Speed vs time](image)

**Figure 149: Predicted total speed vs. time**

To determine the optimum time of deployment, the speed-vs.-time data can be collapsed to a terminal speed ratio vs. time plot, shown in Figure 150. The terminal speed ratio is defined as the speed at any point in time divided by the terminal velocity as calculated by equating drag to weight.
As Figure 150 shows, the terminal speed ratios for all altitudes drops to approximately 1 at \( t = 4 \). Waiting beyond this point in time does not further reduce the \( \mu \text{UAV} \) package’s speed. Therefore, \( t = 4 \) was selected as the time after ejection at which to open the case and deploy the \( \mu \text{UAV} \).
Chapter 7: Integration and Testing

As discussed in Section 1.1.1, the μUAV and protective case must survive the ejection forces and deploy in midair. The μUAV must then recover from the deployment into a stable flight configuration and carry out its mission under autopilot guidance. The development of the μUAV system was split into three main tracks – the μUAV airframe (the flight vehicle exclusive of autopilot), the autopilot, and the protective case. An Integration and Testing Program (ITP) was developed to evaluate the ability of the system to perform the required mission events and to provide a framework to integrate the three tracks together into a fully-working system. An overview of the ITP is shown in Figure 151.

As shown in Figure 151, the ITP is composed of two major technology merge points (M1 and M2) and subsequent Testing Phases (TP1 and TP2). The merge point M1 is the integration of the airframe and protective case. TP1 is composed of two tests, a shock test and a deployment-into-flight test. These tests evaluate the ability of the airframe and protective case to survive the ejection, to deploy in midair, and to recover into steady level flight from the deployed configuration.

The autopilot was developed on a separate schedule track, and therefore has not been sufficiently developed to reach the M2 point, and the tests in TP2 will be discussed in the Future

This chapter will focus on TP1. TP2 will be described in the Future Work section of this chapter.
7.1. Shock Test

The purpose of the shock test is to evaluate the survivability of the µUAV package to the ejection forces. A µUAV package was ejected from an MJU-10/B flare magazine at MIT Lincoln Laboratory with the same pyrotechnic charge as would be used in flight. The µUAV used in the test was a structurally-functional, folding airframe with batteries, motor, servos, and mock-ups of the payload and autopilot boards. The protective case was fully-functional with a working opening mechanism, although the microcontroller board was replaced with an AR6100 receiver to provide a remote-release capability. The test was performed with all components at room temperature as composite structure performance is not expected to vary significantly at low temperature.

Six frames from high-speed footage taken of the ejection are shown in Figure 152 below. The µUAV package was caught by a net (not shown).

![Figure 152: µUAV package ejection shock test](image)

As shown in Figure 152, the µUAV package exits the flare magazine cleanly.

After the ejection, a deployment was attempted to verify the survival of the deployment mechanisms. The µUAV package was suspended by a wire to best-mimic a free-body drop, and
the protective case was opened, at which point the \( \mu \text{UAV} \) deployed successfully. Six frames from a video recording of the deployment are shown in Figure 153.

![Figure 153: In-lab deployment after ejection shock](image)

Figure 153 shows a series of frames taken from the high-speed footage of \( \mu \text{UAV} \) deployment after the shock test. Between frames 4, 5, and 6, wing bounce is visible – both the front and rear wings hit their stops and bounce at high frequency (approximately 10 Hz) briefly until the motion is damped out. Minor damage was sustained by the battery pack, which was punctured by the payload board; to mitigate this problem, the protective foam case around the battery pack was developed.
7.2. Deployment Into Flight

An aerial deployment test was conducted to demonstrate the deployment functionality of the aircraft and protective case in freefall and to investigate the possibility of aircraft-case interference. To accomplish this, a μUAV package was dropped from a platform suspended from a helium balloon. A diagram showing the drop platform is presented in Figure 154 below.

![Diagram showing the drop platform](image)

**Figure 154: μUAV aerial deployment test platform**

A reduced-weight glider version of the μUAV was used for this test to slow the flight dynamics of the aircraft. The μUAV was packaged inside a protective case, to which a streamer was attached. The package was then held below the drop platform by a radio-controlled servo mechanism. Two guy-wires, controlled by ground crew, were used to stabilize and control the location of the balloon.

The pilot controlled three receivers from the radio transmitter – one receiver on the balloon drop platform used activate the release mechanism, a second one in the protective case to trigger opening and deployment, and a third in the μUAV to control the recovery and flight after deployment. A delay time of approximately 2 seconds was used between the release of the package from the drop platform to the opening of the protective case, which allowed the μUAV package to reach 80% of its terminal velocity.

A video-frame composite of the drop and subsequent flight of the glider is shown in Figure 155 below.
In Figure 155, the orange line traces the location of the μUAV, and the white lines link the location of the μUAV to the location of the case at the same moment in time. The deployment was delayed from the release by 1.9 seconds. $t = 0$ is referenced from the moment at which the case opened. By $t = 0.3$, the μUAV's wings have come to rest in the flight configuration and the μUAV has damped out the deployment-induced perturbations, resulting in a stable nose-down flight trajectory. As the white lines show, after the case opens, the μUAV accelerates downwards away from the case as designed.

At $t = 0.7$, the pilot initiated the recovery maneuver, which resulted in the μUAV entering a level flight attitude. At $t = 1.6$, the pilot controlled the μUAV to enter the downwind leg of a left-handed landing pattern. At approximately $t = 5$ (μUAV crossing left-to-right in the foreground of the figure as it flies the base leg of the pattern), the case and streamer re-enter the frame. The final leg of the landing pattern is not shown in the figure.

From this test, it is observed that the μUAV is able to deploy cleanly from the protective case, and that the perturbations from the deployment are small enough that the μUAV is able to
dampen them out quickly. The µUAV flew in a stable nose-down trajectory to accelerate away from the protective case and was able to easily recover into level flight. The SRDS control mechanism was able to actuate the control surfaces effectively, and the µUAV handled predictably enough that the pilot was able to bring it into a clean and controlled landing.
7.3. Summary and Future of Integration and Testing

As discussed above, the ITP is partially-complete at time of writing. The ITP in detail is shown in Figure 156.

Test to evaluate the performance of the airframe and case alone are complete. The wind tunnel tests described in Chapter 4 have verified the aircraft’s aerodynamic performance, and the bench-top tests of the airframe and case were used to develop the structures described in Chapter 5 and Chapter 6.

The tests performed in TP1 have demonstrated the ability of the μUAV package to survive the ejection shock forces, to deploy in midair, and to fly controllably with the SRDS control system. Work is currently underway at the MIT Lincoln Laboratory to develop the hardware and software of the autopilot, which will then be integrated with the airframe and case at the M2 point.

The TP2 tests will evaluate the performance of the full system (μUAV airframe with autopilot and protective case). The first is a shock test which will determine if the airframe, case, and autopilot can survive the ejection, as the shock of ejection may damage the autopilot’s MEMS components such as the gyroscope. If the shock test is successful, the next step would be a low-
altitude deployment-into-flight test, similar to the test carried out in TP1, except controlled by the autopilot instead of the human remote-control pilot. When these two tests are complete, full flight tests (ejection from an aircraft-mounted flare system, deployment in freefall, and autopilot-controlled flight) are planned, first at low altitude, and then at high altitude.
Chapter 8: Conclusion

In this thesis, a novel, highly-constrained micro unmanned aerial vehicle (μUAV) design problem was investigated. The mission requirements place the aircraft in a previously-unexplored portion of the design space, characterized by high-altitude, small-envelope, low Reynolds numbers, small-scale manufacturing limits, and small-scale composite structures.

The μUAV was designed to meet its mission requirements through careful sizing and configuration trade-space analysis, prototyping, and multidisciplinary considerations in the design of many of its components. To meet the challenges of small-scale low-Reynolds aerodynamics, it was found that the airfoil design had to simultaneously consider aerodynamics and manufacturability limits. To design molding and manufacturing processes, it was found that it is necessary to take into consideration not only the desired geometry, but the fabric thickness and fiber density of even the lightest composites available. The result from this work is an aircraft that delivers unprecedented capability and performance in a previously-unexplored design regime.

Through this research, several innovative designs arose, including the small and efficient wing hinge mechanisms, methods to construct small-scale composite wings while accounting for fabric thickness, the Sprung Rotary Drive System which engages and controls a swinging wing, and a molded and insulating battery pack that recycles waste energy as a method to maximize battery performance at cold temperatures.

An integration and testing program was implemented and tests have begun to verify the performance of the various components. The μUAV system has proven to be successful in initial wind tunnel, shock, and deployment tests. Development on the μUAV is expected to continue and will focus on the characterization of battery pack performance at low temperatures and the programming, integration, and testing of the autopilot system. The integration and testing program will lead to low-altitude and then high-altitude full-system tests in which the μUAV package will be launched by the flare dispenser, deploy in midair, and enter autopilot-controlled flight in a simulated mission.
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Works Cited


