Spaceport America Cup IREC

2018-2019 Internal Critical Design Review

Rev A

Last Edited By Samuel Williams

1 Introduction

1.1 Document Description

The following document is River City Rocketry's Critical Design Review (CDR) for the launch vehicle, payload, and axillary systems that will compete in the 2019 Spaceport America Cup's Intercollegiate Rocket Engineering Competition (IREC). The document serves to act as a check for the team's designs, as well as documenting why the team made design decisions and how they did so. The document will not be scored or graded by any outside entity but will serve as a reference for future team members. For any future River City Rocketry team member reading this, the 2019 team hopes that the following document will be of use to you, and we wish you the best of luck in your current endeavors.

1.2 2019 Endeavors

The 2019 team is developing team built solid rocket motors, develop a liquid rocket engine, develop a reefed main parachute, an autonomous fixed wing aircraft capable of being deployed from a rocket, and a vehicle/payload telemetry and tracking system. The payload and solid rocket motor will compete in the 2019 Spaceport America Cup, while the liquid engine will serve as a development platform that the team will build on in the future. An overview of competition deadlines is shown in Figure 1.

DATE	ACTION(S)
1700 16 NOV 2018	Entry Application due per Section 2.6.1 of the *IREC Rules & Requirements Document* https://www.dropbox.com/request/d1bZulrnbMlanLzoLQx7
1700 3 DEC 2018	Acceptance Announcement per Section 2.6.1 of the *IREC Rules & Requirements Document*
1700 25 JAN 2019	Submit 1st Progress Update per Section 2.6.1 of the *IREC Rules & Requirements Document* https://www.dropbox.com/request/C50eSdxUpbib0U9NtFZK
1700 15 FEB 2019	$200 Entry Deposit fee is due using the **(TBR)_payment link provided on the IREC Fees webpage of the ESRA website: http://www.soundingrocket.org/irec-team-fees.html**
1700 8 MAR 2019	Submit 2ndProgress Update per Section 2.6.1 of the *IREC Rules & Requirements Document* https://www.dropbox.com/request/4phqwFFbfntxXAxqyE5l
1700 29 APR 2019	The $200 Entry Deposit fee is no longer refundable if paid after this date, or if a team who have paid announce their intention to withdraw after this date.
	This is the on-time payment deadline for the $500 Rocket Fee. This fee increases to $700 for late-payment. The Rocket Fee will be paid using the **(TBR)_payment link provided on the IREC Fees webpage of the ESRA website http://www.soundingrocket.org/irec-team-fees.html**

DATE	ACTION(S)
	This is the on-time payment deadline for the $50 individual Rocketeer Fee. This fee increases to $70 for late-payment. Each participant will pay their individual Rocketeer Fee using a (TBR)_online ticketing agent. http://www.soundingrocket.org/irec-team-fees.html
1700 17 MAY 2019	Submit 3rd Progress Update per Section 2.6.1 of the *IREC Rules & Requirements Document* https://www.dropbox.com/request/yUqWRfNO4mZMZf5AMPic
	Submit Project Technical Report per Section 2.6.2 of the *IREC Rules & Requirements Document* https://www.dropbox.com/request/gKwrhu6vn1y16QTv6rav
	Submit Poster Session Materials per Section 2.6.3 of the *IREC Rules & Requirements Document* https://www.dropbox.com/request/wXNlo3WrL10H4wTCYbJV
	Submit Podium Session Materials per Section 2.6.4 of the *IREC Rules & Requirements Document* *Slide Deck:* https://www.dropbox.com/request/JkLPyQPyhHPBrfpXOTlt *Extended Abstract:* https://www.dropbox.com/request/YGTXAlERhBefXAOSITQR
	Submit School Participation Letter per Section 2.6.5.1 of the *IREC Rules & Requirements Document* https://www.dropbox.com/request/JefGetTCj0jaw4RDWlqG
	Submit Spaceport America Cup Waiver and Release Of Liability Formper Section 2.6.5.3 of the *IREC Rules & Requirements Document* https://www.spaceportamericacup.com/2018-spaceport-america-cup-waiver.html
ASAP (Before Event)	Submit School Proof of Insurance (if available) per Section 2.6.5.2 of the *IREC Rules & Requirements Document (Note: Due to late updates simply make sure this is delivered prior to the competition)* https://www.dropbox.com/request/SYadsFIRzezvlq4MMRTk

Figure 1: IREC deliverables schedule

1.3 Team Organization

The 2018-19 team is the first River City Rocketry team to compete in the Spaceport America Cup. The transition to this new competition required some restructuring from previous team organization. The new competition resulted in a creation of a solid propulsion, liquid propulsion, and telemetry sub-team. The new competition also saw the removal of the variable drag system which allowed for an increase in resources allocated to telemetry and payload. The recovery and vehicle sub-teams somewhat combined, so that recovery sub-team members could contribute and own vehicle related projects. The 2019 team organization is shown below in Figure 2.

Figure 2: Team Organization chart.

2 Mission Overview

For the 2019 Spaceport America Cup IREC, River City Rocketry aims to fly a vehicle to 10,000ft above ground level (AGL) using a student researched and developed (SRAD) solid rocket motor, deploy an autonomous fixed wing UAV that will fly to predetermined targets and relay live video feed back to a ground station, track and recover all vehicle sections, and recover under a reefed main parachute.

2.1 Vehicle Summary

The launch vehicle will consist of four sections: booster, recovery bay, payload bay, and nose cone. The launch vehicle will be 156 inches in length, 6.25 inches in diameter, and have a dry mass of 62 lbs.

2.2 Payload Summary

River City Rocketry has chosen to design and build a payload capable of detecting ground targets for this year's Space Dynamics Laboratory Payload Challenge. Trade studies of various designs to complete this challenge were evaluated. The design with the highest probability for success was determined to be a deployable fixed-wing unmanned aerial vehicle (UAV). The UAV will deploy from the launch vehicle, autonomously navigate to ground targets via GPS waypoints, stream live video to a ground station, and return to land at the launch site. A mass breakdown of the entire payload is shown below in Table 1.

Payload Bay w/ UAV (lbs)	Payload Bay w/o UAV (lbs)	UAV (lbs)
9.12	4.2	4.92

Table 1: Payload summary.

2.3 Telemetry Summary

To track the flight of the launch vehicle and payload UAV in real time, the team has proposed a telemetry system which will send launch data to a custom ground station webserver. The telemetry system will provide real-time status of the launch vehicle and payload UAV to a range of at least 10,000 feet, and, in the event of vehicle loss, data leading up to the loss would be preserved. Telemetry will also assist with recovery of the vehicle post-launch with the utilization of direction-finding and GPS tracking systems.

Frequencies of 862-915 MHz will be used for telemetry systems in the launch vehicle. Payload telemetry will be transmitting at a frequency of 433 MHz, which falls within rules laid out in FCC Part 15.231 for license-free usage.

2.4 Propulsion Summary

3 Changes Made Since PDR

Sub-team	Change	Justification/Impact
Launch Vehicle	Fin Material Change from Aluminum to Carbon Fiber	Sturdier, lighter
Launch Vehicle	Nose Cone shape change from Elliptical to LV Haack	Better performance in Transonic flight
Payload	Replaced actively controlled camera with statically mounted camera	1) Lack of space in stowed configuration. 2) Added complexity of managing and routing electrical components and wires for rotation. 3) Actively controlled system is not required to satisfy mission requirements.
Payload	Motor was changed from Black power up 32 to an E-Flite 620KV motor	E-Flite motor is more readily available and meets both motor performance requirements.
Payload	Reduced battery size from 8000mAh to 2800mAh	Overly conservative assumptions were made in PDR battery sizing calculations. This change also reduces payload mass.
Payload	Changed wing alignment from level to offset during flight	Aligned wings require a complicated and impractical deployment mechanism.
Payload	Wing deployment mechanism bracket and clevises changed from 6061-T6 to 7075-T6 aluminum.	7075-T6 has greater strength
Payload	Switched from custom made JST connectors to premade connectors	Premade connectors make a firmer connector and are more practical
Payload	Wooden wing closeouts were designed into the wing	Increased torsional stiffness and simpler manufacturing
Telemetry	GPS module change – S1315F to Copernicus II	Breakout GPS module improves system integration

		while meeting previous performance requirements
Telemetry	Using refurbished laptop as ground station computer	Cost reduction; laptop meets the same requirements as model that was to be purchased.
Telemetry	Payload telemetry frequency – 900 MHz to 433 MHz	Ensures separation of frequencies transmitted by all telemetry modules.
Telemetry	Addition – Adafruit Feather M0 Bluefruit LE	Utilizing Bluetooth to elevate the receiver modules and antennas for better reception

4 Vehicle Criteria

4.1 Mission Overview

The launch vehicle's mission is to ascend to an altitude of approximately 10,000 feet above ground level, deploy all recovery hardware, deploy the UAV payload at a predetermined altitude during descent, and land safely on the launch field causing no damage to itself or spectators. The launch vehicle will be propelled by a SRAD Ammonium Perchlorate Composite solid rocket motor.

4.2 Vehicle Design Overview

4.2.1 General Design

The launch vehicle utilizes three fins and consists of four sections: nosecone, payload bay, recovery bay, and booster, as shown in Figure 3 and Figure 4.

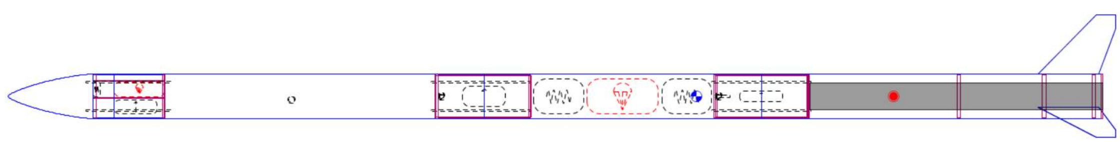


Figure 3: Launch Vehicle Layout in OpenRocket

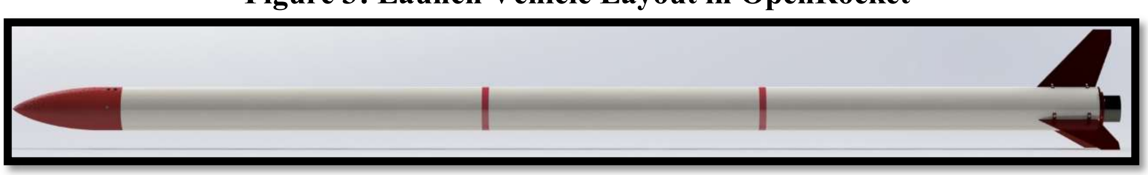
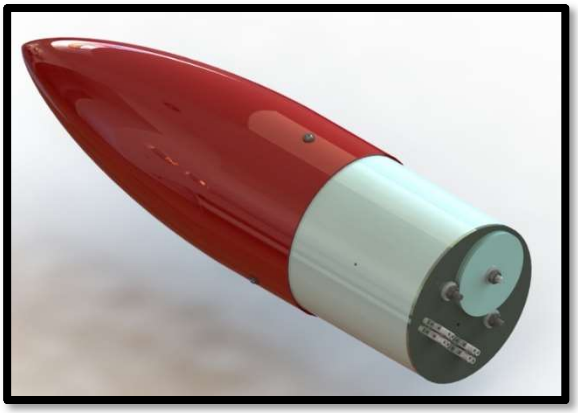


Figure 4: Rendering of the Assembled Launch Vehicle

All airframe sections are made of 4-5 layers of QISO quasi-isotropic triaxial weave carbon fiber fabric imbedded with Aeropoxy Laminating Epoxy. The carbon fiber fabric will be wound around a 6-inch diameter mandrel and allowed to set, resulting in a wall thickness of approximately 0.125 inches.

4.2.1.1 Nose Cone



4.2.1.2 Payload Bay

The payload bay has a total airframe length of 52 inches. The payload bay will house the payload, payload coupler, and payload deployment hardware.

4.2.1.3 Recovery Bay

The recovery bay has a total airframe length of 39 inches. The recovery bay will house the drogue parachute, the reefed main parachute, and the payload deployment hardware.

4.2.1.4 Booster

The booster, shown in Figure 5, Figure 6, and Figure 7, has a total airframe length of 48 inches and will house the propulsion system, centering rings, thrust plate, and fins.

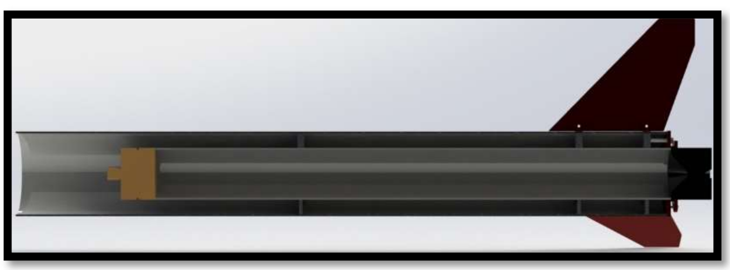


Figure 5: Booster Cross – Section

Figure 6: Booster Exploded View

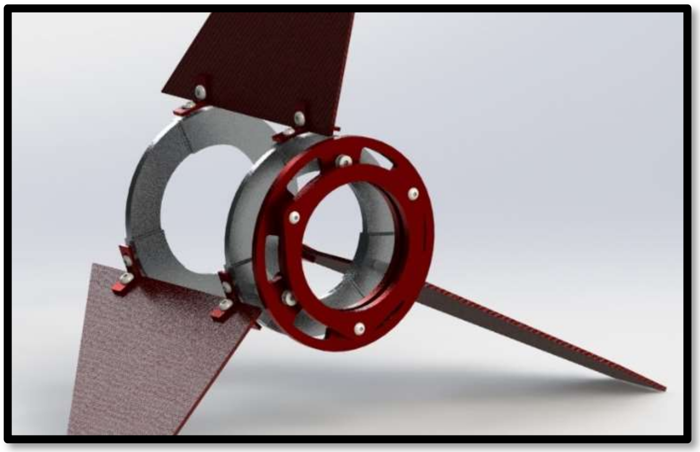


Figure 7: Booster Internal Assembly

4.2.1.4.1 Fins

The fins, shown in Figure 8, are made of 0.125-inch-thick quasi-isotropic carbon fiber sheet.

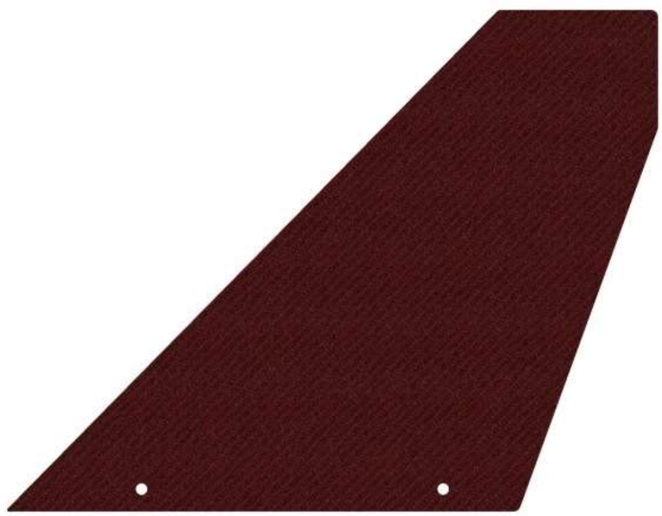


Figure 8: Rendering of Fins

The fin shape was optimized for flight stability and material usage using the OpenRocket Fin Optimization tool. The flat section at the aft of the fin was added to allow a sturdy platform for the rocket to stand on when in the vertical position off the launch pad. Fillets were added to the corners of the fins to aid in material durability. The fin dimensions are shown in Figure 9.

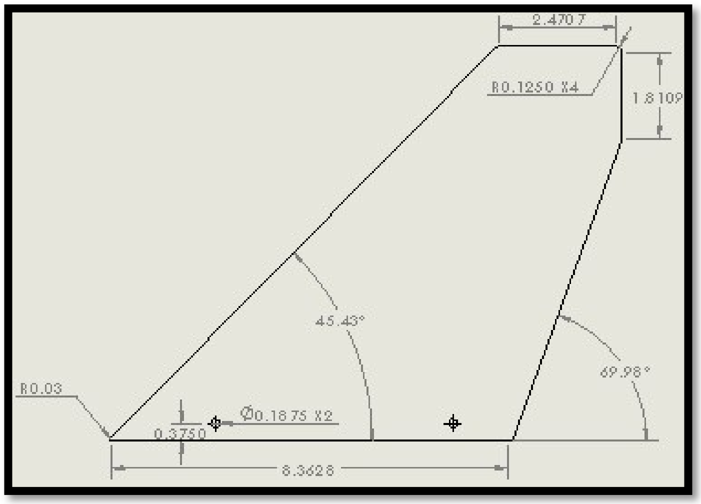


Figure 9: Dimensional Drawing of Fins

Using AeroFinSim fin flutter analysis, it was determined that the launch vehicle would need to reach a velocity of 1,456 ft/s for carbon fiber fins of this shape to experience fin flutter, giving it a factor of safety of approximately 1.5.

The fins are each attached to the airframe and centering rings via four custom machined L brackets shown in Figure 10.

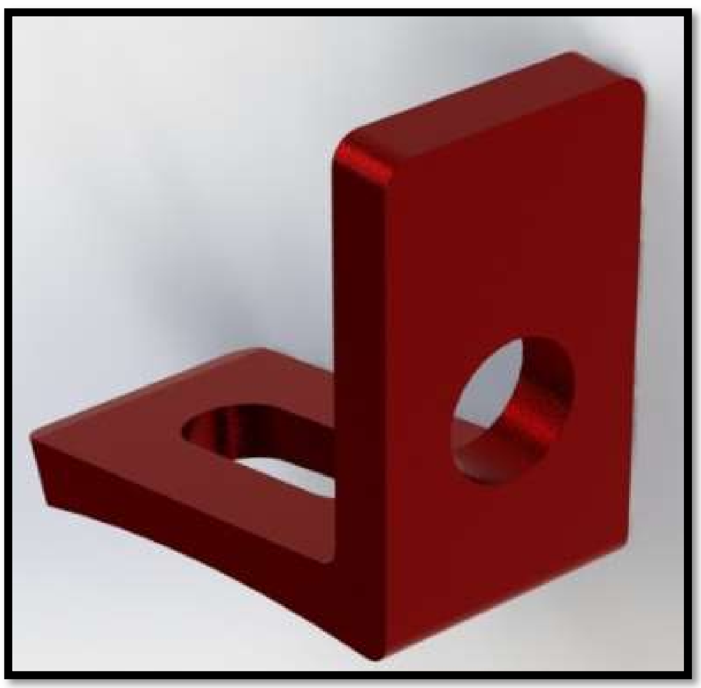


Figure 10: Fin Mounting Bracket

The fin mounting brackets are CNC machined out of 7075-T6 aluminum stock. One outside face of the bracket is flat to interface with the fins while the other is curved to match the outer radius of the airframe. Both inside faces of the fin mounting brackets are flat to allow for positive surface contact with fastener heads. The dimensions of the fin mounting brackets are shown in Figure 11.

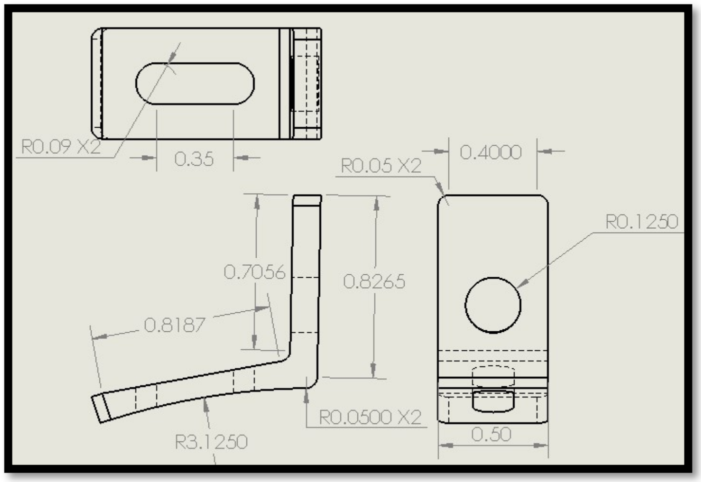


Figure 11: Dimensional Drawing of Fin Mounting Brackets

The fins attach to the mounting brackets via one zinc plated steel binding barrel and screw per pair of opposing brackets. The barrel and screw fasteners were chosen because they provide a consistent clamping force and a symmetrical surface area on either side of the fins. The fin-bracket interface is shown in Figure 12.

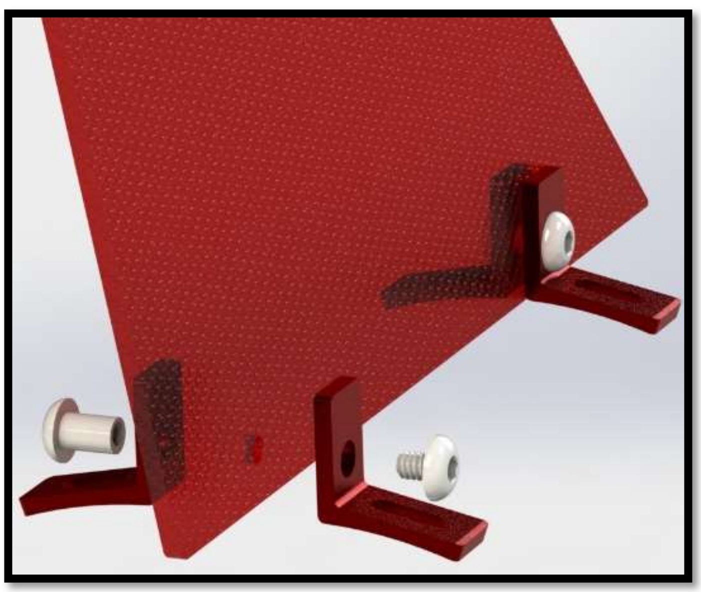


Figure 12: Fin - Mounting Bracket Interface, Exploded and Normal View

The fin mounting brackets attach to through the airframe into the middle and aft centering rings via one zinc plated steel button head cap screw per bracket. The button head cap screw will pass through the airframe and be fastened to a corresponding tapped hole in the centering ring. The bracket-airframe-centering ring interface is shown in Figure 13.

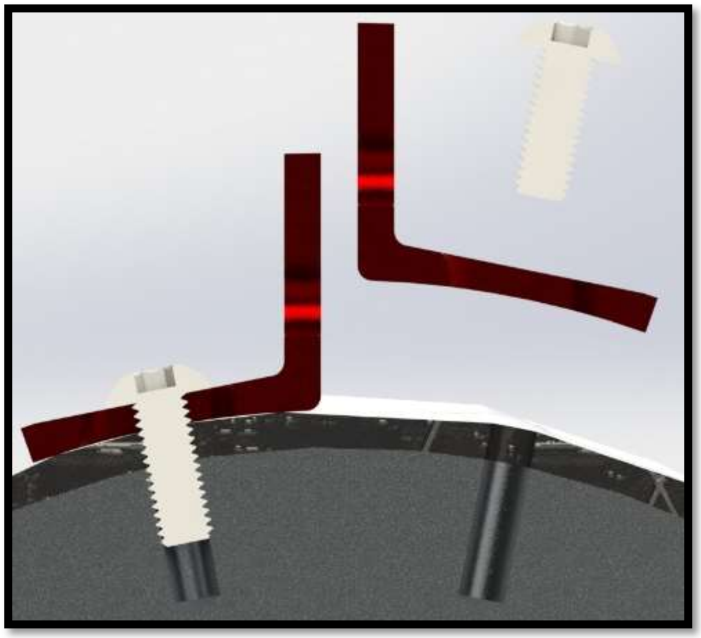


Figure 13:Section View of Bracket-Airframe-Centering Ring Interface with Exploded View
The fins were designed to be easily removable and replaceable, which serves multiple purposes. The removable fins allow for the replacement of damaged fins, removal of fins during transport, and the changing of fin shape without having to reconstruct the booster.

4.2.1.4.2 Centering Rings

The three centering rings, shown in Figure 14, are CNC machined from 0.75-inch-thick 7075-T6 aluminum plate.

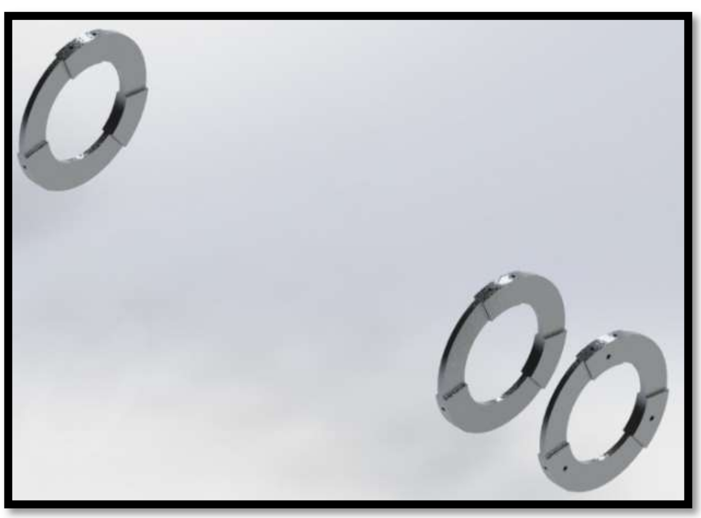


Figure 14: Centering Rings
The centering rings were designed to be held in place by six 10-32 zinc plated alloy steel button head cap screws attached radially through the airframe. To ensure that the threaded holes in the

centering rings did not fail before the button head cap screws, the minimum thread engagement for like materials, L_e, was calculated using the equation

$$L_e = \frac{2 \times A_t}{0.5\pi(D - 0.64952p)} \quad \textbf{(1)}$$

where A_t is the tensile stress area, D is the major diameter of the screw, and p is 1 / number of threads per inch (n). The tensile stress area was calculated using the equation

$$A_t = \frac{\pi}{4}(D - 0.938194p)^2. \quad \textbf{(2)}$$

The tensile stress area was calculated to be $0.0203\ in^2$, and the minimum thread engagement was calculated to be $0.152\ in$. Since the centering rings and button head cap screws have different tensile strengths, the length of engagement had to be increased to prevent the threads on the centering rings from stripping. The actual length of engagement, L_{ea}, was calculated using the equation

$$L_{ea} = J \times L_e \quad \textbf{(3)}$$

where J is the thread engagement ratio. The thread engagement ratio was calculated using the equation

$$J = \frac{T_s}{T_h} \quad \textbf{(4)}$$

where T_s is the tensile strength of the screw material and T_h is the tensile strength of the threaded hole material. The thread engagement ratio was calculated to be 1.446, and the actual minimum thread engagement was calculated to be $0.220\ in$. The centering rings were designed to have a thread engagement of 0.5 inches, giving the thread engagement a factor of safety of 2.27.

The centering rings were designed to be removable, replaceable, and as light as possible. The removable centering rings allows for replacement of damaged components and the ability to accommodate motors of different diameters without having to reconstruct the booster. The dimensions of the centering rings are shown in Figure 15.

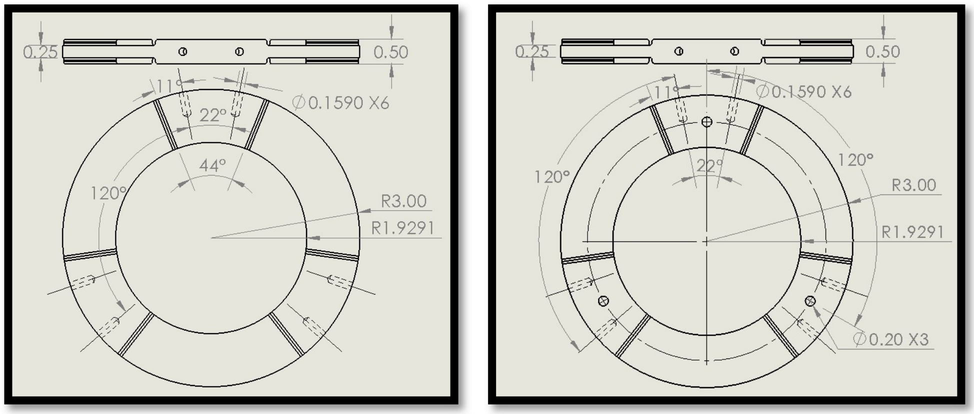


Figure 15: Fore/Middle (left) and Aft (right) Centering Ring Dimensional Drawings

The centering rings were designed so that weight could be reduced as much as possible while retaining adequate wall thickness for fasteners. In between each set of fasteners in the centering rings, the thickness of the part was reduced from 0.5 inches to 0.25 inches to reduce weight while retaining enough structure to support and center the motor.

4.2.1.4.3 Thrust Plate and Motor Retainer

The launch vehicle utilizes a thrust plate, shown in Figure 16 and Figure 17 , to transfer the thrust from the motor directly to the base of the booster airframe. The outer diameter of the thrust plate is 0.125 inches larger than the outer diameter of the airframe to ensure positive contact. A 0.125-inch wide lip is machined into the forward face of the thrust plate to fit against the inside diameter of the airframe to prevent position sliding. To transfer the thrust, the aft lip of the motor casing mates against the aft face of the thrust plate, as shown in Figure 18: Motor-Thrust Plate Interface Cross Section. To secure the thrust plate to the launch vehicle, three one inch long 10-32 alloy steel shoulder screws are fastened through the thrust plate into the aft centering ring, lightly compressing the thrust plate against the base of the airframe.

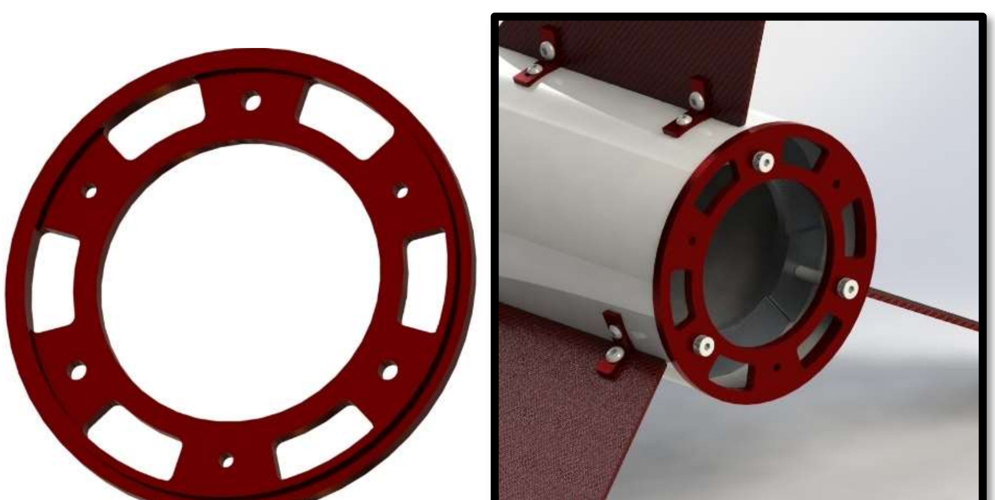


Figure 16: Thrust Plate (left) and Thrust Plate-Airframe Interface (right)

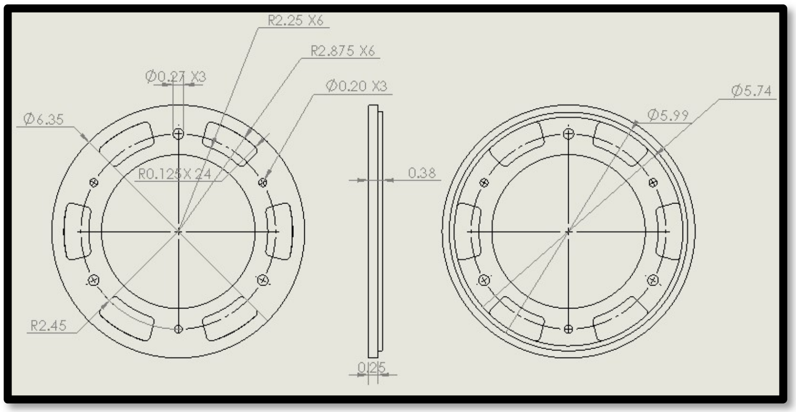


Figure 17: Motor Retainer Dimensional Drawing

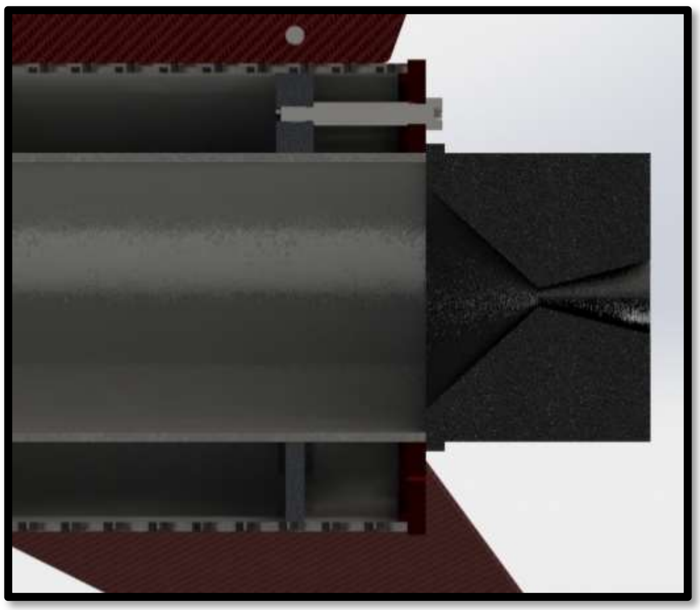


Figure 18: Motor-Thrust Plate Interface Cross Section

The thrust plate is CNC machined from a 0.5-inch thick 6061-T6 aluminum plate. Six one inch thick 30-degree arcs were cut away from the thrust plate for weight reduction. A finite element analysis (FEA) was conducted on the thrust plate to simulate launch conditions. A 1000 lb. force was applied within a circle offset 0.071 in. from the inside diameter of the thrust plate to simulate the motor thrust applied to the area covered by the lip of the motor casing. Stress and displacement results from the FEA are shown in Figure 19: Thrust Plate Finite Element Analysis Stress Plot and Figure 20 respectively.

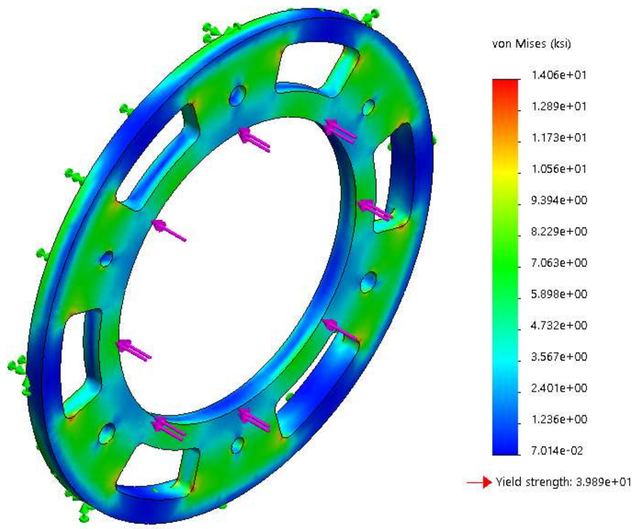


Figure 19: Thrust Plate Finite Element Analysis Stress Plot

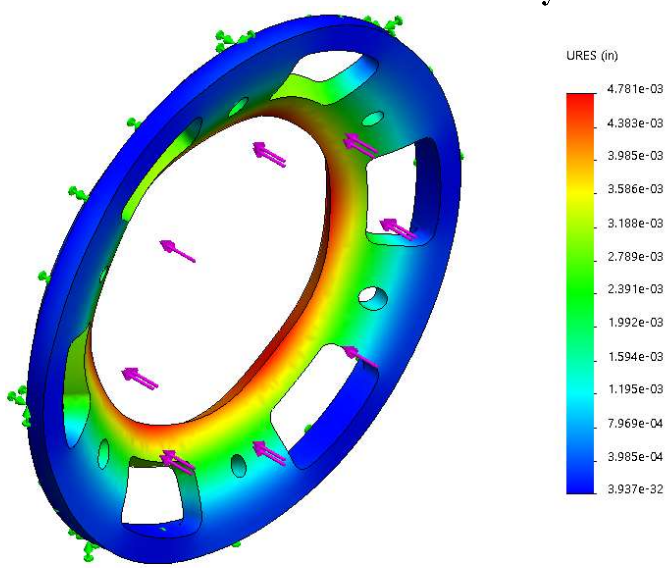


Figure 20: Thrust Plate Finite Element Analysis Displacement Plot

The results of the FEA revealed that the thrust plate would experience a maximum stress of 14.06 ksi, with the yield strength of 6061-T6 aluminum being 39.89 ksi., giving the strength of the thrust plate a factor of safety of 2.84. The maximum displacement that the thrust plate would experience was revealed to be 0.00478 in., which is considerably less distance than the nearest component fore of the motor casing.

The launch vehicle utilizes a motor retainer to ensure the motor remains held in position throughout the flight. The motor retainer is constructed of a 0.25in. thick 6061-T6 aluminum plate and was designed to be machined in a single pass by a CNC waterjet. The motor retainer is attached to the thrust plate via three 10-32 alloy steel button head cap screws in threaded holes spaced 120 degrees apart and offset from the thrust plate attachment holes by 60 degrees. The motor retainer is shown in Figure 21 and Figure 22.

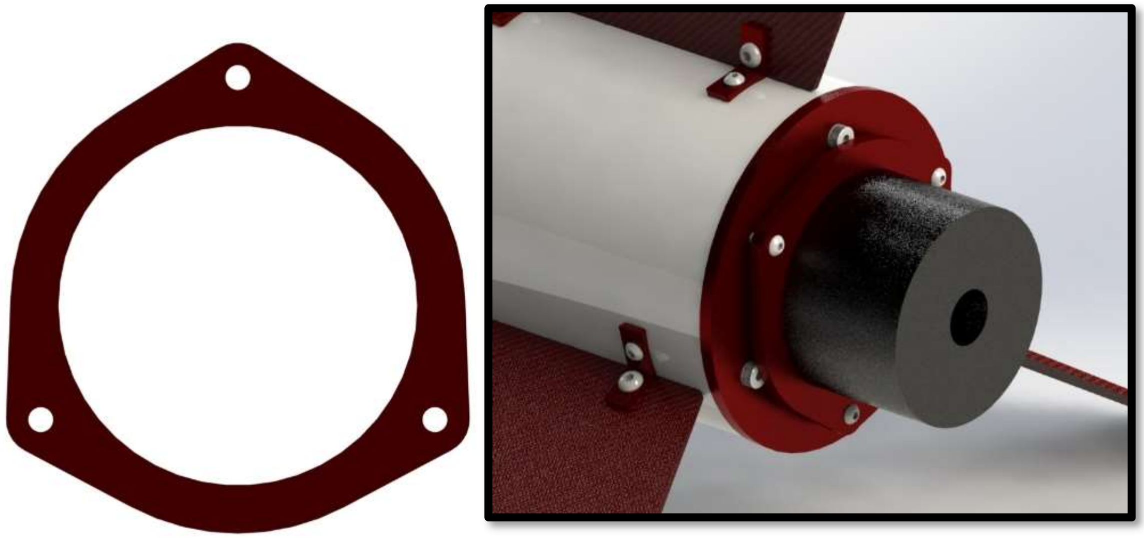


Figure 21: Motor Retainer (left) and Motor Retainer-Booster Interface

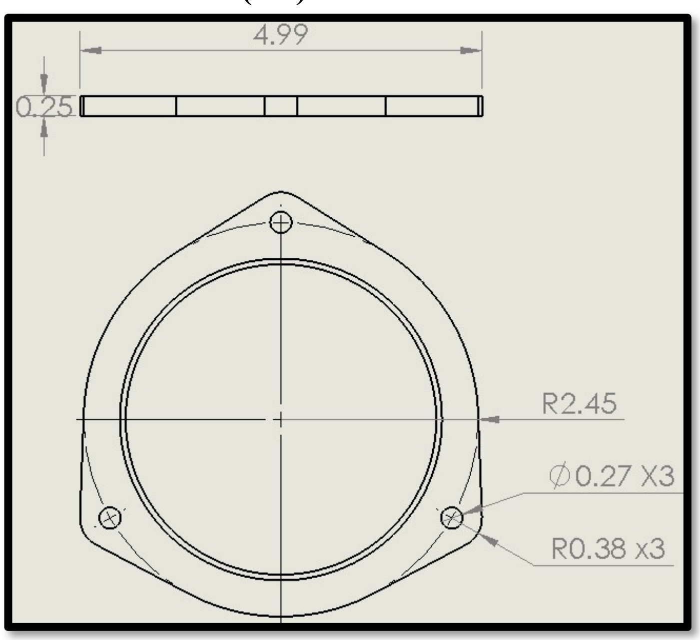


Figure 22: Dimensional Drawing of Motor Retainer

The motor retainer must be able to support the empty motor hardware during recovery deployment and descent, so FEA was conducted, shown in Figure 23, to determine the maximum stress it would endure.

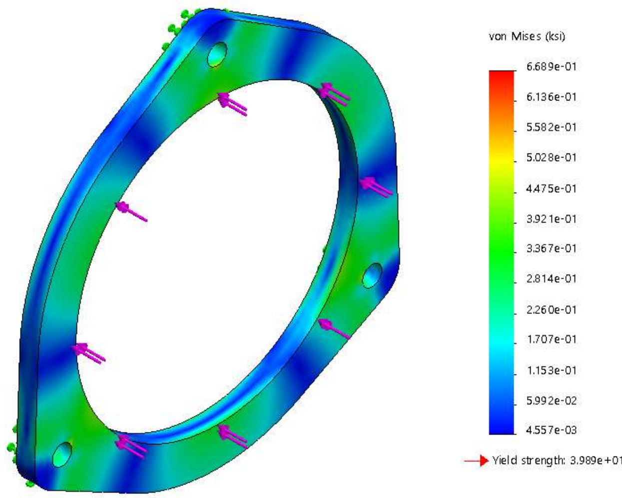


Figure 23: Motor Retainer Finite Element Analysis Stress Plot

By setting the button head cap screw positions as fixed points and providing a force of 26 lbs., equivalent to the mass of the empty motor hardware multiplied by the maximum acceleration during descent, to a circular area offset from the inside diameter by 0.071in., it was determined that the maximum stress that the motor retainer would endure was 0.6689 ksi. Since the yield strength of 6061-T6 aluminum is 39.989 ksi., the motor retainer has a factor of safety of 59.8.

4.2.2 Flight Simulations and Predictions

Flight simulations for development of the launch vehicle were performed using the software OpenRocket. In the absence of custom motor test data, two commercial Level Three motors, a Cesaroni N-1975 and N-2220, were selected to estimate launch vehicle performance. Simulation settings are explained in Table 2: Flight Simulation Conditions

Launch Site Location:	Spaceport America 33°N, 107°W
Launch Site Altitude:	4500 ft. AGL
Wind Speed:	5 mph
Wind Direction:	90°, due East
Launch Rail Length:	17 ft.
Launch Angle:	0°

Table 2: Flight Simulation Conditions

The average launch vehicle performance is shown in Table 3.

Average Stability at Rail Exit:	2.72
Average Maximum Velocity:	943 ft/s
Average Apogee:	10,898 ft.

Table 3: Average Simulated Launch Vehicle Performance

IREC requirements state that the stability of the rocket should remain between 2.0 and 6.0 between launch rail exit and apogee. The simulations determined that the launch vehicle's stability would remain within standards for the entirety of flight.

4.2.3 Schedule

The schedule for the remainder of the season is as follows:

Event	Date	Description
Airframe Manufacturing	02/02/2019	Complete manufacturing of all airframe sections.
Machined Parts Manufacturing	03/02/2019	Complete manufacturing of all machined components.
Small Scale Motor Test Flight	03/16/2019	Successfully complete a commercial Level 2 motor flight.
Small Scale Motor Test Flight	04/13/2019	Successfully complete a commercial Level 2 motor flight.
Competition Motor Test Flight	05/11/2019	Successfully complete a competition motor test flight.

Table 4: Launch Vehicle Season Schedule

4.2.4 Telemetry

4.2.4.1 Launch Vehicle Telemetry Design Overview

Launch Vehicle telemetry will transmit vehicle parameters to a ground station for real-time data tracking. The payload/booster section of the launch vehicle will contain the telemetry data logging system, as well as sensors to calculate vehicle status. Key vehicle parameters to be logged will include but are not limited to: altitude, three-axes velocity, temperature, GPS coordinates, and launch vehicle apogee separation. The nose cone section of the launch vehicle will contain a separate, smaller telemetry system primarily used for tracking purposes. Each of these systems will be radio transmitting data to the ground station.

4.2.4.2 Launch Vehicle Telemetry Hardware

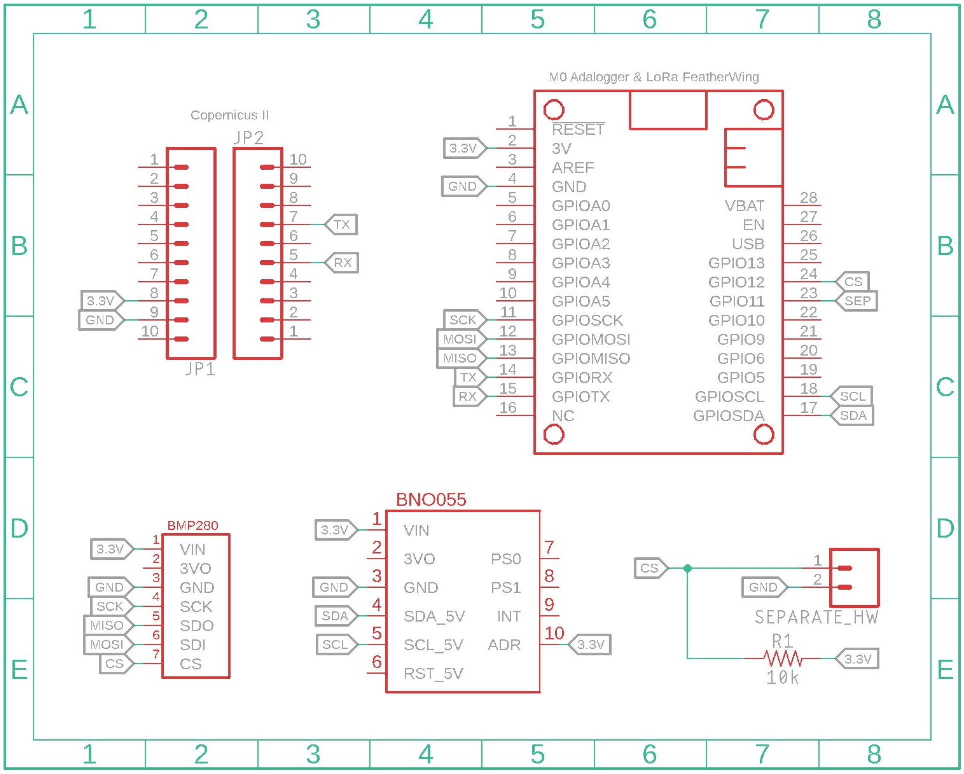


Figure 24: Telemetry Data Transmission System – Schematic

Launch vehicle telemetry will utilize a system of controllers and sensors to transmit real-time vehicle status to ground station. To do so, each system must be fully developed with compatible devices that can communicate data back to the controller to be packed for transmission as an RF signal to ground station. Figure 24 depicts the schematic of devices used for data collection and transmission, as well as the connections made with each component.

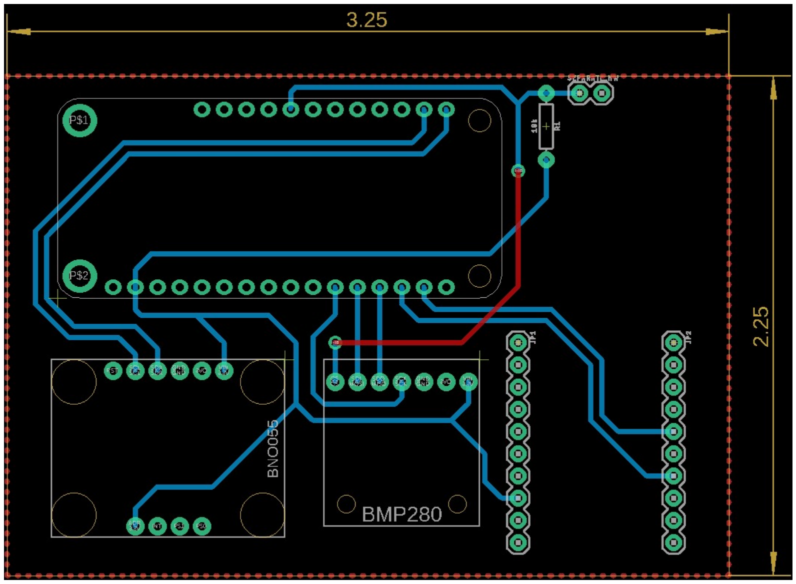


Figure 25: Telemetry Data Transmission System – Board Design

Figure 25 depicts the physical model of the telemetry system's design. Each component of the telemetry system is outlined below in full detail.

The launch vehicle devices will be operated/powered by microcontroller systems. To both log and transmit the vehicle status, two modules, the Adafruit Feather M0 Adalogger and the Adafruit RFM95W LoRa Radio FeatherWing, will be utilized in unison.

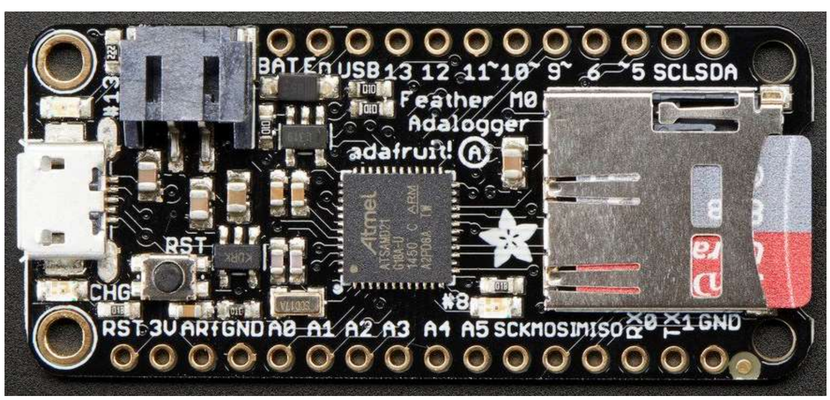


Figure 26: Adafruit Feather M0 Adalogger Microcontroller

The Adafruit Feather M0 Adalogger microcontroller has an on-board micro SD-slot to capture data, on-board battery power connector, and processor clocked at 48MHz. The microcontroller also contains 256 KB of FLASH memory and 32 KB of RAM. The microcontroller uses 3.3V logic. The board has 20 GPIO pins, including Serial Peripheral Interface (SPI), Universal Asynchronous Receiver-Transmitter (UART), and Inter-Integrated Circuit (I2C) communication. The board also has a JST power connector with a built-in 100 mA LiPo charger. These features included everything required to communicate with the sensor modules, radio module, and GPS module utilized with the data transmission system, as well as performing calculations at speeds necessary for the telemetry system.

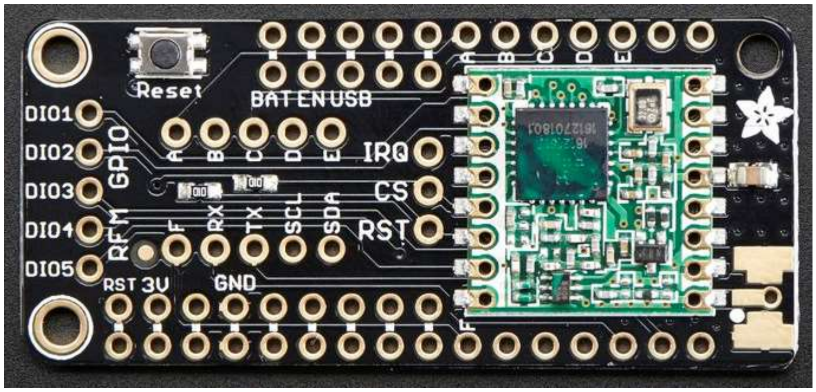


Figure 27: Adafruit RFM95W LoRa Radio FeatherWing – 900 MHz

The Adafruit RFM95W FeatherWing connects seamlessly with the M0 Adalogger, clearing up space for other modules. The FeatherWing uses SPI communication. The board uses USA license-free Industrial, Scientific, and Medical (ISM) bands. The module has an output of +20 dBm or 100 mW constant RF signal, as well as a receiving sensitivity of -148 dBm. Radio data packets can transmit up to a length of 256 Bytes. These features fit requirements needed to transmit data at a range of at least 10,000 feet to the ground station.

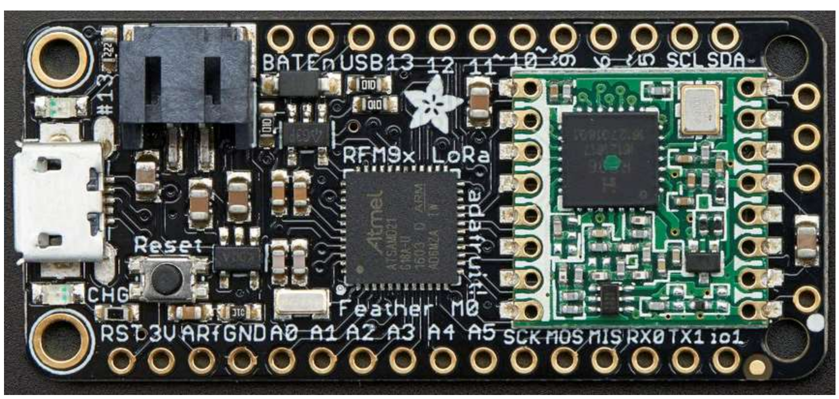


Figure 28: Adafruit Feather M0 with RFM95 LoRa Radio – 900 MHz

To receive the transmission signals, two separate Adafruit Feather M0 RFM95 microcontrollers, one for 862 MHz reception and one for 915 MHz reception, will be connected to the ground station computer. This module has the same processor and nearly identical hardware features as the M0 Adalogger, while also containing the same RFM95 transceiver module as the LoRa FeatherWing. This module fits our requirements of receiving launch vehicle transmission signals.

Figure 29: Adafruit BMP280 – Barometric Pressure and Temperature Sensor

The Adafruit BMP280 pressure sensor records altitude within ± 1 hPa accuracy, and temperature within $\pm 1^{\circ}$C accuracy. The sensor uses SPI communication. The board is 3.3V logic capable, a necessity with the selected 3.3V logic microcontrollers. To accurately measure the altitude, the sea level pressure of the current day and location will be input into the data transmission system from the ground station controls.

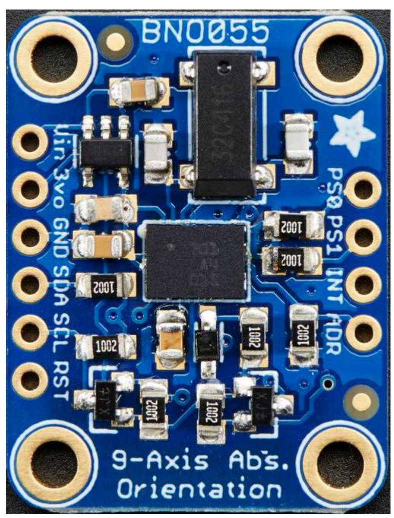


Figure 30: Adafruit BNO055 – 9DOF Absolute Orientation IMU

To determine the velocity of the launch vehicle along the vehicle thrust vector, the telemetry system will use a BNO055 Inertial Measurement Unit (IMU), combined with Kalman Filtering and BMP280 altitude data. Velocity of the launch vehicle can be determined from the integral of linear acceleration data and derivative of altitude data. Once the acceleration and altitude values have been calculated, the controller will estimate the rocket's velocity using the basis of the kinematic equations listed below.

$$v^2 = v_0^2 + 2a\Delta x$$

Equation 5: Kinematic Equation #1

$$\boldsymbol{v = v_0 + a\Delta t}$$

Equation 6: Kinematic Equation #2

Where v is the vehicle's state velocity, v_0 is the vehicle's past state velocity, a is the vehicle's vertical acceleration, x is the vehicle altitude readings, and t is the time at which sensor values have been read.

The BNO055 has an initial operation mode called 'fusion mode'. In this fusion mode, the maximum acceleration range is +4g, too low for the forces faced during flight. The device must be set to a 'non-fusion mode' to increase the acceleration range to +16g. The IMU will be set to accelerometer-only mode, in which the telemetry system will receive raw 3-axes acceleration values. To determine the acceleration along the vehicle thrust vector, the dot product of the 3-axes raw acceleration with 3-axes gravity will be calculated, then the gravity constant divided from the result.

Figure 31: SparkFun Copernicus II Breakout (12-Channel)

Copernicus II is a 12-channel GPS module on a breakout board. The module has a high sensitivity of -144 dBm for acquiring a GPS-lock. The GPS updates at a rate of 1 Hz. In particular, the breakout board GPS was chosen for flexibility during integration and testing. The GPS is not good enough for altitude or velocity measurements, as it is not accurate enough or fast enough. Therefore, the primary purpose of the GPS is locating the vehicle after launch. With this consideration, the GPS will only be logged after apogee separation.

4.2.4.3 Launch Vehicle Separation Detection

To observe vehicle separation at apogee without direct sight of the launch vehicle, telemetry systems will power a vehicle separation detection device. The device will report its state within the data packets sent to ground station. All launch vehicle separations will be observed by a unique detection device.

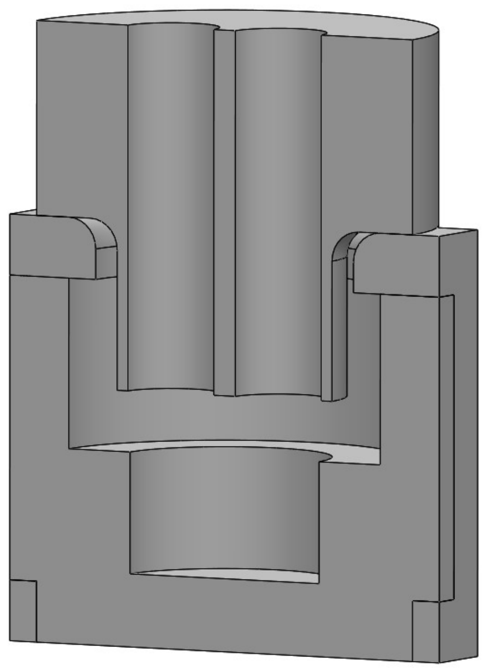


Figure 32: Telemetry Separation Detection Hardware (Cross-Section)

Telemetry separation detection hardware consists of a male and female block. The female block houses a spring and metal plate. The male block houses two wires, a common ground and microcontroller input pin, that will run back to the telemetry board. To detect launch vehicle separation, a microcontroller pin will read inputs from a pull-up resistor circuit.

When the detection hardware circuit is closed (i.e. when the male block is compressing the spring within the female block), the input pin will be pulled to ground and the system will report that there is no separation. When the detection hardware circuit is open, the input pin will be pulled to +3.3V through at 10 kΩ resistor and the system will report that separation has occurred. The circuit schematic will be setup as shown below.

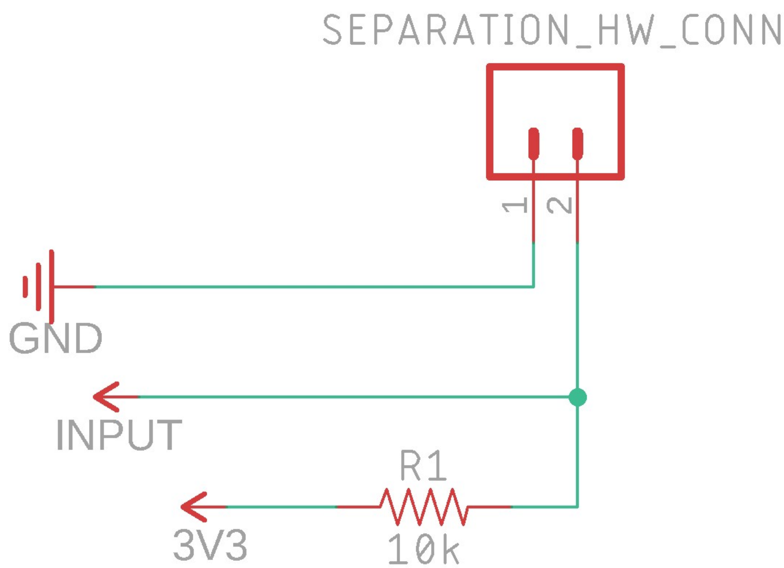


Figure 33: Telemetry Separation Detection Schematic

4.2.4.4 Telemetry Power Requirements

Launch vehicle systems will require a remote power supply for an extended duration. Vehicle launch data needs to be recorded and the vehicle components need to be recovered before power is fully consumed. Below tables depict the current load requirements that will be placed on the microcontrollers and the Lithium Ion Polymer (or LiPo) battery ratings required for each telemetry system.

Device	Max Current Consumption [mA]
Feather M0 Adalogger	50
FeatherWing LoRa 900MHz Radio	120
BMP280 Sensor	1.12
BNO055 Sensor	12.3
Copernicus II GPS Module	44
Total Electronics Current	227.42
With Factor of Safety of Two	454.84

Table 5: Power Consumption of Data Transmission Telemetry System

The Feather M0 Adalogger voltage regulator has a max continuous output of 500 mA. The microcontroller will successfully handle the current load of the above system with a current rating of ~450 mA (including a factor of safety of two).

Device	Max Current Consumption [mA]
Feather M0 LoRa 900MHz Radio	120
Copernicus II GPS Module	44
Total Electronics Current	166
With Factor of Safety of Two	332

Table 6: Power Consumption of Nose Cone Recovery Telemetry System

Similar to the telemetry data transmission system, the nose cone recovery section will have a max current output requirement of 500 mA. This the above system, the microcontroller will handle the current load of ~330 mA (including a factory of safety of two).

Additionally, to reduce current consumption before and after vehicle launch, certain systems will be placed in standby until commands are received from ground station or certain parameters are met.

Current [mA with FOS of 2]	Battery Rating (mAh)	Operation Time (hours)
~450	1200	2.67
	2500	5.56

Table 7: LiPo Battery Estimated Operation Times for Telemetry Electronics

LiPo options of 1200 mAh and 2500 mAh for powering telemetry systems have little difference in physical dimensions and weight. Therefore, the batteries chosen will supply 2500 mAh, capable of 5+ hours at full power consumption (including the factor of safety rating). The battery weighs ~52 grams and has physical dimensions: 2” x 2.55” x 0.30”.

4.2.4.5 Launch Vehicle Antenna Specifications

Since the carbon fiber body tube of the launch vehicle is non-RF-transparent, the data transmission system will require an antenna that can be placed on the external surface of the body tube. Since an antenna added to the outside surface of the body tube would increase the vehicle’s drag, the antenna chosen must have dimensions small enough to avoid any interference with the vehicle’s flight. The antenna must also operate within the frequency band of 860-928 MHz.

Figure 34: Leaf Antenna – 470-862 MHz

A leaf antenna used for capturing UHF and VHF signal frequencies will satisfy the requirements listed above. The antenna has a flexible membrane, perfect for wrapping around the body tube of the vehicle. The antenna has a minimal thickness that won't disturb the drag of the vehicle. It has a frequency range that complies with the telemetry radio modules. The antenna has an f-type (RF coaxial) connection, which will be converted to SMA connections at the telemetry board.

The nose cone is made of a 3D printed material that is RF-transparent. Therefore, the nosecone telemetry system will use a ½ wave 915 MHz antenna.

The expected RF signal range from each transmitter is three miles. The leaf antenna was recently tested at multiple ranges of three miles, two miles, and ½ miles. The leaf antenna did not transmit any signals at the three-mile range or two-mile range. At the ½ mile range, the signals were received with great signal strength, but there were also long stretches of no signals being received.

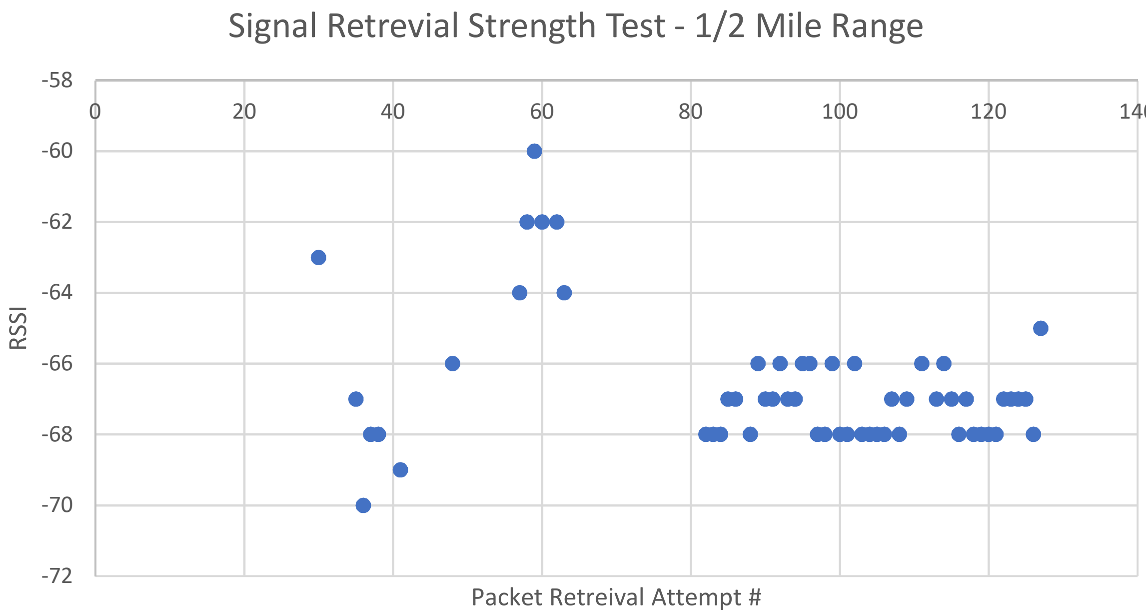


Figure 35: Signal Strength vs. Packet Retrevial Attempts – ½ Mile Range

The above chart data of the ½ mile range test revealed errors with the radio update rates. Further testing will be conducted to ensure data rates do not interfere with data packet retrieval.

4.2.4.6 Telemetry Operation Specifications

The following pseudo code lays out the operations of the telemetry data transmission system:

```
INITIAL SETUP:
     WHILE (no "start" signal from receiver) THEN
          DELAY
     IF (a sensor, radio module, or SD card fails to initialize)
     THEN
          Send a diagnostic signal to the ground station (or blink
          red LED if radio fails initialization)
          STOP
     WHILE (no "start data transmissions" signal from receiver)
     THEN
          DELAY


LOOP:
     IF (vehicle has separated) THEN
          Long delay between data logs
     ELSE
          Short delay between data logs
```

```
        Acquire sensor data and send radio packet to receiver
        Save data to SD file
```

The following pseudo code lays out the operations of the ground station data receiver system:

```
    INITIAL SETUP:
        IF (radio module fails to initialize) THEN
            Send a diagnostic signal to the ground station
            STOP
        WHILE (no “ready to start” signal from transmitter) THEN
            DELAY
        WHILE (no “start telemetry” signal from ground station) THEN
            DELAY
        Send “start data transmissions” signal to transmitter

    LOOP:
        IF (RF signal is found) THEN
            IF (signal has no errors) THEN
                Send data to ground station
```

4.2.4.7 Ground Station Overview

The telemetry ground control system will be used to receive, log, and visualize real-time data from vehicle launches and payload deployments. This system should be simplistic in design so that it can be set up at launches with relative ease and can be developed upon by future generations of engineers. Furthermore, the ground station needs to be capable of displaying data from the launch efficiently while simultaneously making such data accessible by those within range of the ground controls network.

This ground system communicates with four main systems: the data collection system’s receiver module, the nose cone telemetry system, the payload’s telemetry module, and the payload’s first-person view (FPV) video transmitter module. Ground Station needs to send and receive data packets from these sources via radio signal transmission and to have the data from the three radio streams accessible in real time.

4.2.4.8 Ground to Launch Vehicle Communication

The ground-to-vehicle connection relies on a pair of Adafruit Feather RFM95 radio microcontrollers for telemetry communication. The microcontroller located in the booster section of the vehicle will be the main piece of hardware communicating with the ground station computer (GCOM). This telemetry subsystem must be capable of not only sending data but also receiving commands from GCOM.

The telemetry team proposes elevating the ground-to-vehicle receiver antenna, as opposed to having the receiver sit right next to GCOM at ground level. This should allow for a clearer and stronger connection between the vehicle and the ground station. In order to bridge the increase of distance between the receiver and the computer, the receiver will also be utilizing a Bluetooth system to send the packets from the receiver to the GCOM. This also means that the GCOM computer must have Bluetooth capabilities.

4.2.4.9 Ground Station to Client Communication

The ground-to-client connection allows 3rd parties to view the data from the launch in real time. By configuring the ground station into a web server, people on the same local network as the ground station will be able to view launch data from the vehicle and flight data from the payload in real-time. The ground station is to act as a receiver of telemetry data from the vehicle and payload, sending the collected data to the webserver to visualize the data and make it easily accessible to those with close range of the GCOM.

The GCOM uses a Linux Debian operating system. This operating system was chosen because it supports the ground control software and has simplistic web server configuration capabilities. GCOM uses Apache HTTP server, which allows computers on the same network to view a webpage when the IP address of GCOM is put into a web browser. This will only work on a local network, and only when GCOM is powered up. On top of the computer being a web-server, GCOM will host a database that will store telemetry data from the vehicle and payload. The ground station essentially acts as a redundant measure of data preservation, as well as making the data easily accessible by the webserver for visualization.

Once the database and schema have been created, the webpage which displays all the data must still be designed. For this, either C# of HTML/CSS/JS will be utilized to set the layout and query the database. One design that has been drafted allows clients to switch between viewing data from the vehicle and viewing data from the payload, as can be seen below. These windows should also provide real-time data as well as the overall averages and the initial values of the vehicle right off the rail. See below for a draft of the UI.

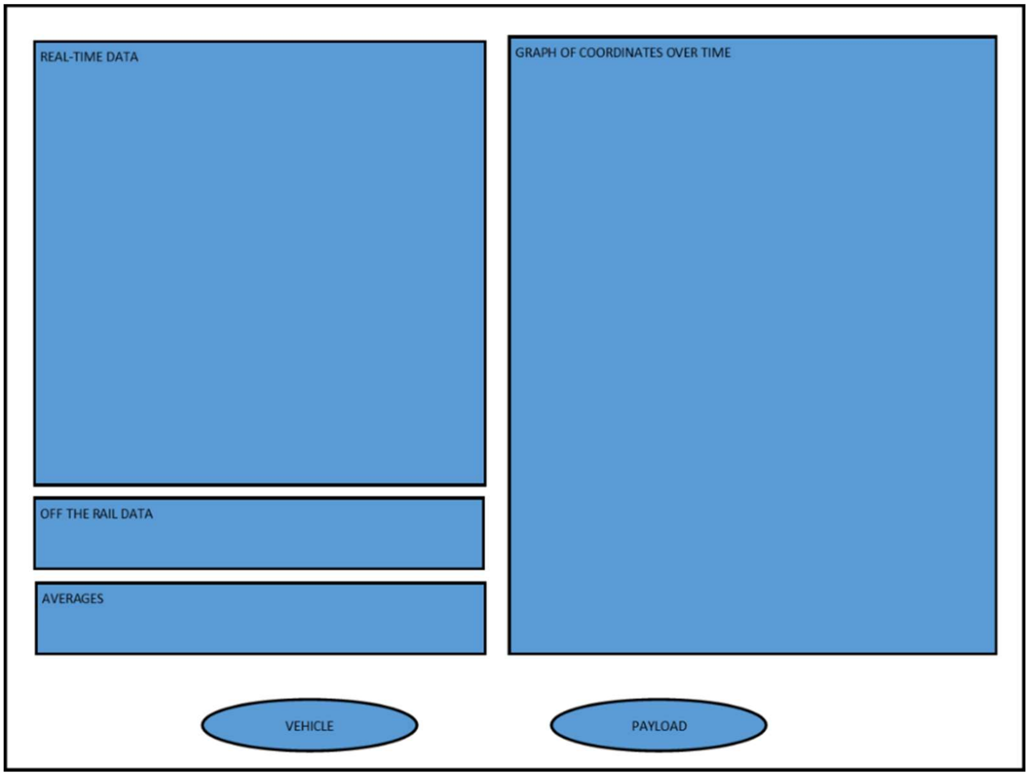


Figure 36: Ground Station UI

4.2.4.10 Telemetry Direction-Finding Overview

Direction-finding will be utilized to track down sections of the vehicle after launch. The direction-finding system will operate separately from the rest of the telemetry systems. The system will be blind to the specifics of the radio technology used except for the transmitting frequency.

The setup consists of a "log amplifier" and a low-noise RF amplifier. The log amplifier outputs a voltage based on the decibels of radio frequency input it receives through the connected antenna, between 1 MHz and 8 GHz. Because of the wide frequency range, there is a great amount of noise present in the output. By connecting the output of the log amplifier to speakers, it is possible to pick up the output of the telemetry radio as an audible periodic clicking (about once per second). It diminishes in perceived loudness when the antenna was rotated away from the radio or if they were a further distance apart. Qualitative analysis is not practical for this setup due to the large amount of noise in the output.

To make long distance detection possible, the input to the log detector will need frequency filtering. Currently under experimentation are passive RC filters, which will help reduce the background noise that makes detection difficult. We will be consulting electrical engineering professors for advice on the design of our radio frequency filters.

4.2.4.11 Radio Transmission Calculations

The log detector has an "input slope" of -25 mV / dB, meaning the output varies by 25mW for every dB increase of radio power received, and it can receive anywhere from -65dBm to +5dBm.

The low-noise amplifier (which may be used as a pre-amplification stage for the log detector) has a gain of 60dB at 500Mhz and a noise figure of 2dB at 300Mhz.

$$P_R = \frac{P_T G_T G_R \lambda^2}{(4\pi R)^2}$$

Equation 7: Power transmission equation between antennas

Where P_R is the power received, P_T is the power transmitted, G_T is the transmitter gain, G_R is the receiver gain, λ is the wavelength, and R is the distance between the transmitter and the receiver. Using 3dB for both gains, 100mW for transmitter power, 900MHz, the receiver gets –42 dBm at 100 meters, -61dBm at 1000 meters, and –82dBm at 10,000 meters. For a frequency of 433Mhz, these values are –35.6dBm, -55dbM, and –75dBm, respectively. This indicates the log amplifier should be able to detect the signal from 1000 meters, and at greater distances using the low noise amplifier.

This is assuming no interfering noise. Even if we can filter the frequency perfectly, electronic noise may make detection at these extremely low levels impossible. Consultation with Dr. Faul will help us understand what can be done to improve the performance of our system further.

4.2.4.12 Launch Vehicle Integration

The below checklist covers a rough guideline of telemetry system procedures prior to, during, and after launch. Each section provides brief descriptions of steps to follow, as well as spaces for notes and confirmation signatures by team members.

To be checked and signed by Telemetry Lead and team member when all steps are completed, indicating that Telemetry Team is **PREPARED FOR LAUNCH.**
1.____ 2. ____

PRIOR TO LEAVING FOR LAUNCH SITE

Equipment to Pack:

1. Soldering Iron 2. MicroSD Cards 3. MicroSD Adapter Cards 4. LiPo Batteries 5. Telemetry Boards 6. Ground Station Computer 7. USB to Micro-USB cords 8. GPS Antennas 9. Yagi Antennas 10. RFM95W Radio Antennas	11. Tiny Screwdriver Set 12. Wire Cutters/Strippers 13. Glue 14. Tape 15. Needle-Nosed Pliers 16. Extra Wires 17. Antenna connectors (N-type, F-type, SMA, RP-SMA, uFL) 18.	19. 20. 21. 22. 23. 24. 25. 26. 27. 28.

Telemetry Systems Packing: All steps must be confirmed by the Telemetry Lead

	1. Ensure code on microcontrollers is up-to-date (including redundant systems)
	2. Place telemetry boards in a safe location to be transported
	3. Ensure all antenna connectors that will be needed are packed
	4. Fully charge LiPo batteries
	5.
	6.
	7.

To be checked and signed by Telemetry Lead and team member when all **PRIOR TO LEAVING FOR LAUNCH SITE** steps are completed, indicating that Telemetry Team is **PREPARED TO DEPART FOR LAUNCH.**	
1.____ 2. ____	
LAUNCH DAY PROCEDURES	
Telemetry Systems Preparation and Integration: **All steps must be confirmed by the Telemetry Lead**	
	1. Insert microSD card into Adalogger on data transmission board
	2. Ensure all antenna connections are solid
	3. Ensure boards/modules/batteries are secure in sled
	4. Fully plug in batteries to telemetry boards
	5. Connect wires for separation hardware, screw down wires until tight, give wires a test pull
	6. Calibrate BNO055 IMU
	7. Set sea level pressure for current location and day
	8. Test ground station-to-receiver connection
	9. Begin data transmissions as vehicle is taken to launch pad
	10.
	11.

Troubleshooting:	
NOTE:	If a problem does not present a trivial solution immediately, swap the faulted system with its backup and re-run through vehicle integration
	1. Read diagnostic code received through ground station
	2. If no diagnostic code, check for blinking red LEDs on microcontrollers (signifies radio module fault)
	3. Ensure all connections on boards are secure
	4. If the telemetry board shows no power received (no LEDs on), swap LiPo batteries
	5. Cycle power to controllers to perform a software reset
	6.
Post Flight Inspection:	
	1. Track last GPS reading from both nose cone and payload/booster section with Individual direction-finding systems
	2. Disconnect batteries from systems as soon as possible
	3. Retrieve microSD card
	4. Examine telemetry hardware for any signs of stress or other types of damage
	5. Power down Ground Station when both telemetry and payload telemetry are completed
	6.

5 Propulsion Criteria

5.1 Propulsion Overview

River City Rocketry will be using a “Solid Composite Rocket Motor” to propel the vehicle to an altitude of 10,000 ft AGL. A solid rocket motor is the most common form of rocket propulsion used in the high-power rocketry hobby. Large orbital rockets such as the Space Shuttle used a similar propellant in the side boosters, while liquid propulsion methods are normally used for most modern orbital class rockets. A diagram of an example solid rocket motor is shown in Figure 37.

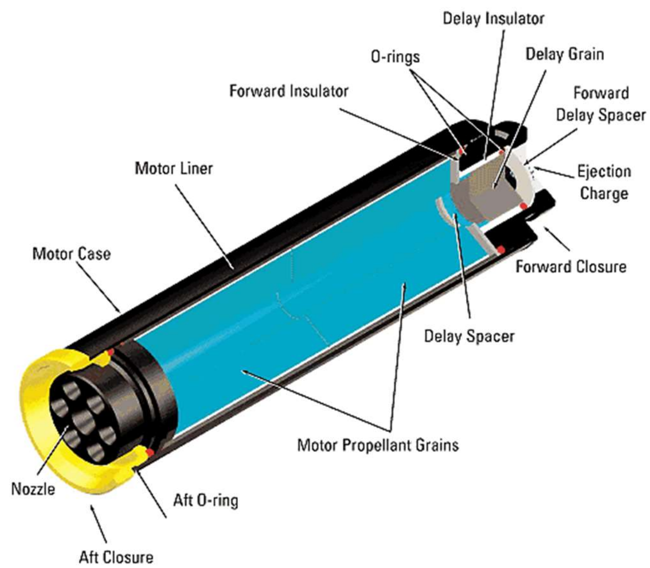


Figure 37: Sample solid rocket motor cut-away.

Solid rocket motors on a small scale are inert enough to be sold at hobby stores around the country with no certifications needed for the buyer. The small motors available for purchase at the local hobby store have similar chemical composition to the propellant discussed later in this document.

5.1.1 System Overview

River City Rocketry chose to build and mix its own solid rocket motor for the 2019 Spaceport America Cup. The goal of the solid rocket motor is to successfully ignite and then propel the rocket to as close to 10,000 feet AGL as possible. River City Rocketry has sought the experience and expertise of Darryl Hankes, a rocketry hobbyist, United States Navy veteran, and mentor to River City Rocketry since 2011, to initiate the team’s solid rocket motor development program. Mr. Hankes has extensive knowledge and experience with the production of solid rocket motors as he has been producing his own motors for over 10 years. Mr. Hankes has designed and built high power rockets more complex than what the team is attempting, such as the one shown below which consisted of two stages, with 4 motors producing 2,241lb of thrust, and flying to an altitude of 15,000ft.

Figure 38: Team Rocketry Mentor Darryl Hankes.

Figure 39: Darryl Hankes built rocket just after motor ignition.

Mr. Hankes provided the team with a list of numerous potential solid fuel mixtures, containing a variety of chemicals. These mixtures have all been produced and developed by experienced rocketeers and tested extensively outside of the University of Louisville. The team has decided to use a previously developed recipe to accelerate our development of our capabilities of producing a successful solid fuel mixture.

Since this was the team’s first attempt to produce a solid rocket motor, the team chose to use an recipe without metal powder additives. Many solid fuel mixes have metals added as burn rate modifiers, but the decision was made not to include metals in the fuel mixture. Metals, such as aluminum, in solid motors are added as a powder and can increase the performance of the motor. Despite the performance gains, the use metal powder introduces additional complexity and safety concerns, as the powder could be inhaled and cause adverse health effects. For this reason, the team decided to omit any powdered metals from the solid fuel mixture.

The team also chose to use a high solid fuel mixture (greater than 80% of the ingredients by weight are in the solid state of matter when mixed). High solid mixtures packed rather than poured into casting tubes. They have the consistency of clay or wet sand and can be easily picked up and molded by hand (wearing proper safety gloves), prior to insertion into the casting tube. A BATES geometry, which consists of a cylindrical bore hole through which the hot gases can exit the motor, was chosen for the motor grain geometry because the bore hole is the easiest to manufacture as it can be drilled into the grain or molded around a mandrel during casting.

Table 8 shows the fuel mixture RCR will use for its solid motor.

Chemical	**Amount (% by weight)**
Ammonium perchlorate	83.5
Carbon black	0.5
Hydroxyl-terminated Polybutadiene (HTBP)	11
2-Ethylhexyl acrylate (EHA)	2.7
Rubidium	1.7
Tepanol	0.6
Silicone Oil	1 drop per 1.1 lbs. propellent
Total	**100**

Table 8: Solid Fuel Mixture.

The recipe is given in percent weight, because the exact amount is subject to change depending on the final weight of the vehicle and the performance of the motor. All of which is unknown until tests can be run. Anticipated amounts can be found in the solid motor SOP in section 5.2.

Ammonium perchlorate is the oxidizer in the mixture. Carbon black stains the ingredients black and acts as an opacifier to reduce radiation heat transfer. Lighter color propellant can absorb UV light as the motor burns, thus increasing radiation heat transfer, and increasing the overall temperature of the hot gases within the motor. Without an opacifier, fuel would burn faster than intended which can cause over pressurization and destruction of the motor. HTBP acts as a binding agent as well as fuel source. EHA is an additional binding agent which simply binds the fuel particles together to form more of a solid. Rubidium is the curative and tepanol is an additional bonding agent.

5.2 Solid SOP

Procedure For:	Solid Propulsion Standard Operating Procedures
Date Created/Created By:	02/1/2019 Chase Renner and Samuel Williams
Location:	1960 Arthur Street, Louisville, Kentucky, 40208

Notes: Operation of solid rocket propulsion system is not allowed to be performed alone. Four individuals are required to perform this procedure. Additional personnel may be present during the procedure.

Revision: A

Last revised by Samuel Williams

Changes and correction since last revision

- Removed step to drill center bore, added step to use mandrel instead
- Specified container class to store AP composite motors
- Corrected several typos regarding the specific amounts of chemicals
- Updated steps to include references to relevant sections
- Changed required people from 2 to 4
- Added requirement for two 10 lb ABC fire extinguishers
- Changed wording to refer to mixing location
- Clarified that a stand mixer will be used, not a handheld
- Added mineral spirits to cleaning procedure
- Specified that the mixer will be grounded
- In section 0, changed “certified” to verify
- Added instructions to print most up to date of section 4
- Added emergency chemical burn and eyewash station to preparation section
- Added limit to materials to be transported to mixing location
- Added recommendation for double axel for long term transportation (our trailer has a double axel)
- Section 9 has been edited for clarity
- Section 10 and 11 have been rewritten to be relevant regardless of mixing location
- Updated waste disposal procedures. Subject to change based on DEHS requirements
- Added revision sections and revision labelling

Section 0 – Potential Hazards and Mitigation

Hazard	**Mitigation**
Propellants are flammable.	No smoking or open flames permitted in the vicinity of the propellants. Grounding straps will be worn to prevent static electricity buildup. Fire retardant sand to be kept on hand at all times. Mixer shall be grounded.
Catastrophic failure of motor	System will be operated remotely, specifically at a minimum distance of 1000 ft for rocket engines with an impulse range of 10,240 N-s to 20,480 N-s in accordance with NFPA 1127.
Failure of motor casing	Motor casings will be bought commercially, and total pressure and impulse shall not exceed rated limits
Improper mixing of propellants	Propellants shall be visually inspected for discoloration or damage prior to use.
Accidental inhalation of mixture gases.	All propellants will be handled in a well-ventilated area where risk of accidental inhalation will be low.
Accidental contact with chemicals	Proper PPE is to be worn throughout procedure. If exposed to chemicals, wash affected area with soap and water and notify safety officer.
Structural failure of rocket engine test stand.	Test stand will be structurally tested prior to rocket engine testing to ensure integrity of the structure. In addition, the test stand was designed in accordance with ASME guidelines.
Improper storage of propellants.	Propellants shall be stored in a locked facility with limited access only to members of the team who have been verified by the safety officer.
Vegetation fire from exhaust.	Testing site shall be kept clear of any flammable debris.
Improper waste disposal	Spills of propellants and propellant waste shall be destroyed by burning in a manner acceptable to the authority having jurisdiction in accordance with NFPA 1125.

Section 1 – PPE to be used

- ☐ Cotton/organic fiber clothing/lab coat (no nylon/synthetic/other fibers which cause static buildup)
- ☐ Safety goggles
- ☐ Full face shield
- ☐ Nitrile gloves
- ☐ ESD grounding wrist strap. Must be bonded to large metal object with path to ground.
- ☐ Long pants
- ☐ Close-toed shoes
- ☐ Reflective vests for greater visibility

Section 2 – Materials/equipment required

- ☐ 5 gallons of Sand (or other fire retardant)
- ☐ Two 10 lb ABC fire extinguishers
- ☐ Sacrificial surface (cardboard, drop cloth)
- ☐ Paper towels
- ☐ Chemical ingredients
    - Ammonium perchlorate (oxidizer)
    - Binder system constituents (see section 4 for detailed recipe)
        - Monomer
        - Bonding Agent
        - Plasticizer
        - Curative
- ☐ Paper cups
- ☐ Non-sparking mixing tools such as wood or nylon
- ☐ Planetary Stand Mixer
- ☐ Digital Scale
- ☐ Scoop
- ☐ Non-static plastic bowl
- ☐ Weigh boats
- ☐ Sifting Screen
- ☐ Measuring cup
- ☐ Casting tube
- ☐ Casting plug
- ☐ Tamping tool
- ☐ Emergency chemical burn and eyewash station

NOTE 1: No procedures are to be performed alone. Four people are recommended to be nearby and aware of procedures being conducted.

NOTE 2: Ingredients used in fabrication of composite solid propellant composite solid propellants themselves are export controlled. US Citizens only may handle each. Ingredients and propellant articles must be secured by lock when not in use and/or unattended.

NOTE 3: At time of writing, the largest anticipated batch size is 7 kg or 15 lbs.

NOTE 4: Mixing AP Composite Motors will produce Ammonia gas, a non-toxic respiratory irritant. Mixing should be conducted in well ventilated area. Review SDS for each material for detailed hazard assessment.

Section 3 – Preparation

- ☐ Adjust recipe in section 4 as required and print recipe
- ☐ Wash hands with soap and water.
- ☐ Don PPE (see section 1).
    - PPE must be during entire procedure
    - Do not remove PPE until entire mixing procedure has ended, all equipment is cleaned, and all waste has been safely disposed of per NFPA 1125 and DEHS recommendation, with exception of section 10.
- ☐ Gather materials (see section 2).
- ☐ Prepare mixing and packing stations.
- ☐ Apply sacrificial surface to packing station to prevent contamination with propellant.
- ☐ Prepare mold equipment.

Section 4 – Recipe "Mr. Clean" 7 kg 12,996 N-sec

"Mr. Clean"	12996 N-sec	Weight (grams) =	7000
Chemical	Type	%wt	Total Weight (grams)
HTPB	Monomer	11.0%	770
EHA	Plasticizer	2.7%	189
Tepanol	Bonding Agent	0.6%	42
Silicone Oil	Penetrant	14	drops
AP	Oxidizer	83.5%	5845
Carbon Black	Opacifier	0.5%	35
Rubidium	Curative	1.7%	119
Sum		100.0%	7000

Chemical	Type	Weight of Propellant (g)	Weight of Adhesive (g)
HTPB	Monomer	770	385
Rubidium	Curative	119	59.5
Sum		886	444.5

Section 5 – Mixing and Casting Procedure

This is a general mixing and casting procedure that may change based on the amounts mixed, slight modifications to recipe, etc. Please refer to section 4 for specific recipe.

- ☐ Prepare for mixing (see section 3)
- ☐ Verbally inform personnel working in the vicinity of the propellant of the propellent mixing activity
- ☐ Use butcher paper or cardboard / other to prevent leaving propellant residue on work surfaces.
- ☐ Weigh out required quantities of binder, monomer, bonding agent, and plasticizer. See section 4.
- ☐ Place syringe of HX-878 tepanol bonding agent/curative (if used) in warm water bath in a cup to lower viscosity.
- ☐ Set mixer to lowest setting, add, in order (see section 4):
    - Monomer
    - Plasticizer
    - Tepanol

    - Silicone oil
- [ ] Mix for 10 minutes
- [ ] Sieve AP to remove chunks if necessary
- [ ] Add:
    - Oxidizer
    - Carbon Black
- [ ] Mix for 1 hour
    - Every 15 minutes, stop to scrape ingredients from side
- [ ] Prepare casting tube adhesive
    - Mix monomer and curative with wooden stirrer in separate container (Section 4)
    - Evenly apply a thin coat of the adhesive on inside of casting tubes
    - Secure wax paper to one end of casting tube using masking tape
- [ ] Add curative to main mixture
- [ ] Mix for 15 minutes at lowest setting
- [ ] Manually shape propellant into rough spheres
- [ ] Mount mandrel to casting tube
- [ ] Tightly pack into casting tubes one sphere at a time
    - Tamp to remove air pockets using tamping tool.
- [ ] Store in approved container for 24 hours to cure (see section 7)
- [ ] Clean all equipment and work surfaces with mineral spirits
- [ ] Remove mandrel from cured casting tube
- [ ] Load grains into reloadable motor kit
- [ ] Dispose of all waste in approved containers and manners (see section 6)

Section 6 – Waste Disposal Procedure

- Until disposal, waste is to be stored in loosely covered flammable waste metallic container. DO NOT tightly seal any flammable waste container.
- Excess propellants shall be destroyed by burning in a manner acceptable to the authority having jurisdiction in accordance with NFPA 1125. In the case that DEHS prefers to collect the waste themselves, they must be scheduled to pick up the waste the same day.
- Waste should be disposed of the same day as mixing, do not store flammable waste for extended periods of time.

Section 7 – Propellant Storage

- AP composite motors must be stored in a Type II OSHA and NFPA approved fire safety storage cabinet.
    - Label propellant grains with the recipe used, the date they were mixed, and the name of all team members who participated in the mixing and casting procedure.
- Finished propellant grains should be stored as follows:
    - Away from radiators and other sources of heat.
    - In a well-protected, well-ventilated area.
    - In a dry area (not on earth/ground).
    - Away from salt, corrosives, chemicals, and fumes.
    - Out of exits and egress routes.
    - Protected from physical damage from striking or falling objects.
    - Protected from public tampering (i.e. secured).
    - Out of direct sunlight (i.e. less than 120 degrees F).
- Store chemicals of the same hazard class in the same area (i.e. poisons/highly toxic, flammables, inert, corrosives, oxidizers, and cryogenic gases).
- Storage areas must be identified with the materials they contain.
- Oxidizers and fuels must be separated by at least 20 feet, or a noncombustible wall at least 5 feet high with at least a half-hour fire rating.
- Oxidizers and flammables must be kept at least 20 feet from all sources of ignition.
- No more than 500 lb of the combined weight of the propellant, delay, and ejection composition shall be permitted at any one time in the mixing room or process building in accordance with NFPA 1125.

- ☐ In process storage of less than 100 lb of uncast propellant shall be permitted in covered containers in the mix room.

Section 8 – Propellant Transport

- ☐ Propellant storage must meet the Department of Transportation's requirements for labeling, marking, and placarding in compliance with DOT Chart 15.
- ☐ A suitable vehicle must be used to transport the propellant.
- ☐ In the case that a trailer is being used for long distance transportation, a double axel trailer is preferred.
- ☐ Inspect the storage case and propellant for existing damage prior to attempting transport.
- ☐ Do not smoke during transport.
- ☐ Take a direct route to the new location and do not make any intermediate stops along the way. Avoid heavy traffic routes.
- ☐ Remove the propellant from the vehicle as soon as you have reached your destination. Place it in proper storage.
- ☐ No more than 3 times the amount of constituent chemicals in a recipe shall be transported at a time, whichever is less. See section 4 for detailed amounts.

Section 9 – Propellant Static Firing Procedure

Please note that the static firing procedure is identical to the flight procedure. Before any static firing procedure, the appropriate fire department should be notified.

- ☐ Don safety glasses
- ☐ Load propellant into motor casing
- ☐ Verbally notify all nearby that motor is now loaded
- ☐ Mount loaded casing into test stand
- ☐ Insert ignitor into motor bore hole. Insert fully into the bore hole, ensuring ignitor physically touches top most propellant grain
- ☐ Remove safety key from ignition system
- ☐ Connect leads of ignitor to ignition electronics
- ☐ Return to safe distance. For the planned impulse class of motor all personnel must be minimum 1000 ft away from test stand/rocket per NFPA 1127 standards. For other sizes please refer to NFPA 1127.
- ☐ Alert those nearby of launch, verbally and with air horn

- Once launch area is cleared, perform countdown, and ignite
- Following completion of ignition sequence, disconnect all ignition electronics
- In accordance with NAR HPR Safety Code, wait a minimum of 60 seconds following any firing or misfire before approaching motor.
    - Warning: Following a successful firing, motor casing will be hot to the touch.

Section 10 – SUPPLEMENTAL PROCEDURE: Contact of uncast propellant with skin

- Remove PPE and wash affected area with warm water and soap or chemical burn and eyewash fluid.

Section 11 – SUPPLEMENTAL PROCEDURE: Occurrence of fire

- If fire occurs in area where others are working, evacuate area.
- Locate nearby bucket of sand or other flame retardant.
- Apply fire retardant sand to base of fire.
- If the fire is not extinguished:
    - Evacuate the area.
    - Pull fire alarm (if applicable)
    - Contact the fire department (911).
    - Notify faculty advisor (Yongsheng Lian) and safety officer (Taylor Hsieh).

Approved By:

Yongsheng Lian (PI)	Date
Samuel Williams	Date
Chase Renner	Date
Taylor Hsieh	Date

Team members trained on and authorized to use this procedure (print name and sign):

Name (Print)	Name (Sign)	Date
Name (Print)	Name (Sign)	Date
Name (Print)	Name (Sign)	Date
Name (Print)	Name (Sign)	Date
Name (Print)	Name (Sign)	Date

Add additional signatures as necessary.

5.3 Hardware

To accelerate development and simplify the design stage, all motor hardware will be purchased from commercial vendors. Selection of motor hardware was based on a similarly sized commercially available and estimated vehicle weight. The team anticipated that the motor would need to produce 12,980 N·s of impulse, requiring 15.46 pounds of propellant. The density of the solid motor mix (estimated at 0.0543 lb/in^3) means that the vehicle will need to be 98mm diameter motor to accommodate the estimated quantity of fuel in an acceptable length to diameter ratio. The casing was selected to accommodate six standard size grains. If fewer grains are needed, spacers may be used within the same casing to reduce cost. Table 9 displays all solid motor hardware. Motor hardware is subject to change after static fire testing.

Item	Quantity	Description
98mm casting tube	6	Cardboard tubing that the solid fuel adheres to
98mm phenolic tube	1	Phenolic lining that surrounds grains in motor
Cesaroni Pro98-6 grain case	1	Aluminum casing that houses the liners and grains; outer layer
Grain spacer O-ring	5	Provides space between adjacent grains
98mm nozzle w/ .734" throat	1	Motor nozzle. Throat diameter is less than bore hole diameter
Forward O-ring	1	O-ring at the top of motor casing
Aft O-ring	1	O-ring at the rear of motor casing
Seal disk O-ring	1	O-ring between casing and forward closure
Forward closure	1	Seals the forward end of the motor
Aft closure	1	Seals the aft end of the motor
Retaining ring	1	Ensures all aft hardware stays in place

Table 9: Solid motor hardware.

According to the book *Experimental Composite Propellant* by Terry W. McCreary, Ph.D., the throat of a nozzle of a solid motor should be ¾ the diameter of the bore diameter. The reference also states that a 98mm BATES motor should have around a 1" diameter bore hole. The team chose a commercially available graphite nozzle with a .734" throat diameter. The rocket nozzle the team is using is shown in Figure 40.

Figure 40: Rocket motor nozzle.

5.4 Testing

5.4.1 Testing Overview

The team has constructed a solid motor test stand based on plans purchased from Aerotech. Aerotech is a leading manufacturer of solid rocket motors. The team has a Test Stand Data Pack that contains drawings of the Aerotech test stand. The test stand has a load cell that the rocket pushes against when fired. The load cell measures and records the force the motor applies, which can be used to generate a thrust curve for a given motor and solid fuel mix. This thrust data is then used in the design of the overall rocket using the OpenRocket simulation software. Figure 41: shows the solid motor test stand. Figure 42 shows the Test Stand Data Pack.

Figure 41: Solid motor test stand with 75mm motor.

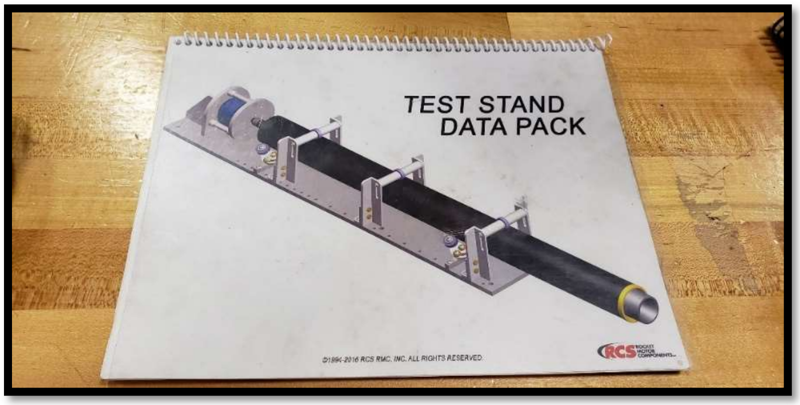


Figure 42: Test Stand Data Pack.

Developing and test firing the solid rocket motor is broken up in to three distinct phases. The first phase involved researching the basics of solid motor design, the principles behind the different mixtures of fuel, and the proper safety equipment and techniques. The second stage will involve mixing and firing a small-scale motor. The third phase will be mixing and firing full scale motors as the team hones their techniques to build a motor that can propel the vehicle to reach the desired 10,000 feet AGL. The team has already completed the first phase and is waiting on approval from UofL's Risk Management and Department of Environmental Health and Safety in order to begin the second phase. The schedule for the second and third phases is in Table 10:.

Event	Date	Description
Mix small batch	TBD (waiting on approval)	Meet with Darryl Hankes to mix first propellant batch. Make enough fuel to fill 75mm motor
Test fire 75mm mixture	1 week later	Static ground test fire of the 75mm motor with first fuel batch
First full-scale static fire	2 weeks later	Mix and static fire first 98mm fuel batch
Second full-scale static fire	2 weeks later	Mix and static fire second 98mm fuel batch
Third and final full-scale static fire	2 weeks later	Mix and static fire third 98mm fuel batch

Table 10: Solid motor testing schedule.

Static fire testing will occur at a concrete pad located at 1561 Hooper Station Rd, Shelbyville KY. The pad will be shared with the liquid engine, but tests will not be conducted concurrently. As both stands are mobile, they will not be on the pad at the same time to isolate possible failures and damage to other components.

Solid rocket motor static testing procedures can be found in the accompanying standard operating procedures in section 5.2.

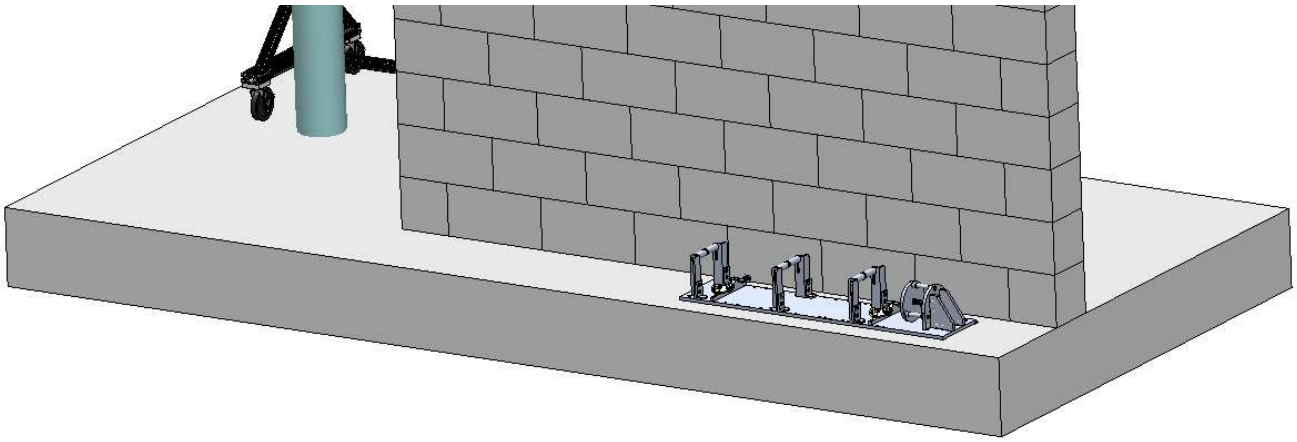


Figure 43: Engine and motor test pad

The following software programs will be used to assist in gathering and analyzing the data: ProPEP 3 (ProPep), BurnSim, and OpenRocket.

- ProPep allows the user to enter the amounts and types of chemicals used in a solid fuel mix. It then generates two values about the fuel: the characteristic velocity (c*) and the specific impulse (ISP). The c* and ISP values are needed to use the next program, BurnSim.

- BurnSim takes the c*, the ISP, and the density of the fuel mix, as well as information about grain geometry, number of grains, bore hole diameter, and nozzle throat diameter to generate a thrust curve. The thrust curve is then uploaded to the final program, OpenRocket.

- OpenRocket simulates a flight by combining the thrust curve of the motor with the current model of the vehicle to give an estimated apogee. Flight conditions such as temperature, wind speed, and launch location (to determine altitude above sea level) are all variables OpenRocket uses to estimate apogee. OpenRocket has been used extensively in previous River City Rocketry projects.

5.5 Safety

5.5.1 Solid Rocket Motor Testing Risk Assessment Matrix

Operating and firing a solid rocket motor can pose numerous risks. Table 35 lists possible hazards and how the team will mitigate risks associated with static fire and flight operation.

Hazard	Possible Causes	Mitigation Approach	Risk Level After Mitigation
Uncontrolled fire surrounding test location	CATO causes scattering of burning propellant	Area around concrete pad is cleared of tall vegetation	Low
	Exhaust flame catches vegetation on fire	Static fire will be conducted on concrete pad instead of on earth.	Low
	Rocket motor breaks free from test stand and catches fire to surrounding area	The test stand is rated with a factor of safety greater than 2 against a 1500 lb load, anticipated maximum thrust is 1000 lb.	Low
Fragmentation of motor casing components from explosion during test fire	Cracks in propellant	The team will visually inspect all motor grains for evidence of cracks or de-bonding from the liner walls.	Medium
	Debonding of propellant from liner wall	During the casting of the motor grain, the liner will be coated in an adhesive to promote adhesion to liner wall. The team shall inspect the motor for damage from transportation to the test site.	Low
	Gaps between propellant sections and/or nozzle cause increase in burn rate and over pressurization of motor casing	O-rings will be placed between propellant sections and other motor components to absorb small impact energies	Medium
	Motor case unable to contain operating pressure or pressure spikes	Tightly secure all motor closures and double check that hardware is in place.	Medium

		RCR is purchasing a commercial motor casing from an established manufacturer, Cesaroni. In addition, the project lead and one captain must sign off to ensure motor is properly secured according to Aerotech or Cesaroni specifications.	Low
Fragmentation of motor components from explosion during flight	Cracks in propellant	The team will visually inspect all motor grains for evidence of cracks or de-bonding from the liner walls.	Medium
	Debonding of propellant from liner wall	During the casting of the motor grain, the liner will be coated in an adhesive to promote adhesion to liner wall. The team shall inspect the motor for damage from transportation to the test site.	Low
	Gaps between propellant sections and/or nozzle cause increase in burn rate and over pressurization of motor casing	O-rings will be placed between propellant sections and other motor components to absorb small impact energies	Medium

	Motor case unable to contain operating pressure or pressure spikes	Tightly secure all motor closures and double check that hardware is in place. RCR is purchasing a commercial motor casing from an established manufacturer, Cesaroni. In addition, the project lead and one captain must sign off to ensure motor is properly secured according to Aerotech or Cesaroni specifications.	Medium
	Cracks in propellant	The team will visually inspect all motor grains for evidence of cracks or de-bonding from the liner walls.	Medium
	E-match not inserted into grain core properly	Disconnect all launch control electronics from igniter. Wait a minimum of 60 seconds to ensure motor does not ignite, per National Association of Rocketry's High Power Rocketry Safety Code.	High
Failure to ignite motor	Insufficient contact between E-match and propellant grains	Only essential, fully trained, and certified personnel go to pad to inspect E-match.	High

	Igniter was installed prematurely	Proper procedure and necessary checklists will be followed precisely during motor test operations. This includes not installing the igniter into the motor until the motor is mounted on the launch pad.	
Motor ignition occurs prematurely	Ignition sequence was started prematurely	Proper procedure and necessary checklists will be followed precisely during motor test operations. This includes sounding of an alarm prior to ignition and ensuring no personnel are near the test pad.	Low
	Improper personnel were given command of launch control system	Launch controls will be unpowered until all personnel are minimum 1000ft away and it is time to fire the motor. In addition, only trained and certified personnel will be given command of the launch control system. If any non-certified team member violates said rule, it will result in immediate dismissal from the team. All team members wishing to attend the test will be required to attend a Safety Briefing the night before. Failure to do so will result in member being barred from attending the event.	Low

Table 11: Hazards of testing and flying solid motor.

6 Recovery Criteria

6.1 Mission Overview

The launch vehicle will ascend to an altitude of approximately 10,000 ft. AGL and activate a black powder (BP) charge, separating the vehicle at the midsection and deploying a drogue parachute that tethers both sections together. At a TBD altitude optimal for payload mission success, the main parachute is deployed in a 80% reefed state and will decelerate the launch vehicle. The nose cone will be jettisoned via BP charge and will eject a self-supporting parachute from the Nose Cone Recovery System (NCRS). At optimal payload release altitude, the payload ARRD (Advanced Retention and Release Device) is activated and the payload is released from the airframe, opening a programming parachute that will bring the payload to specified conditions for release. After conducting preflight checks, the payload will release the programming parachute and begin nominal flight. The launch vehicle will un-reef the main parachute and descend to a landing at optimal velocity.

Event	Altitude	Phase	Description
1	Apogee (~10,000 ft.)	Launch Vehicle Separation and Drogue Event	Launch vehicle separates at the midsection and deploys a drogue parachute that tethers the booster and payload bay together.
2	3000 ft.	Main Deployment in Reefed State	Main parachute is deployed in a reefed state at an altitude that is optimal for payload mission success.
3	2200 ft.	Nose Cone Ejection and Parachute Deployment	Nose cone is jettisoned from the launch vehicle and deploys an independent parachute from the NCRS.
4	2000 ft.	Payload Release and Programming Parachute Deployment	ARRD is activated and payload is released from the launch vehicle. The payload deploys a programming parachute and conducts preflight checks
5	1500 ft.	Programming Parachute Release	Payload completes preflight checks and releases the programming parachute. Small masses attached to the shroud lines cause the programming parachute to invert and descend independently.
6	800 ft.	Main Parachute Un-reefed State	Launch vehicle un-reefs main parachute and descends to a landing at nominal ground hit velocity of 11.11 ft/s. Landing will occur within a 2-mile radius of the launch site.

Table 12: Parachute deployment sequence of events

The recovery sequence of events is outlined below in Figure 44: Recovery sequence of eventsFigure 44.

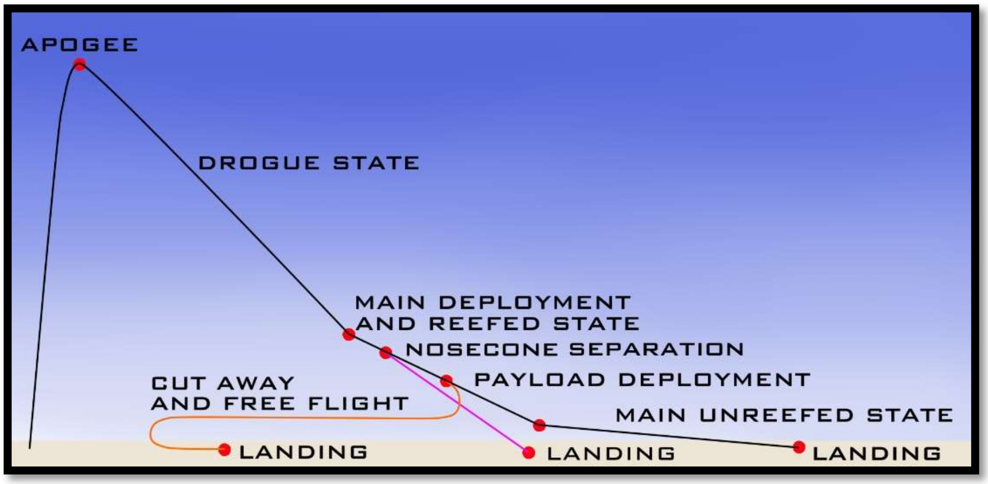


Figure 44: Recovery sequence of events

The Mission Elapsed Time (MET) was calculated using OpenRocket flight simulations and decent velocities of various sections therefore each deployed segment will have a separate MET. For the sake of the main segments of the launch vehicle, the flight of the payload will not be considered, only the landing time under parachute. These times are shown below in Table 13.

MET	353.00		
section	**Nosecone**	**Payload**	**Rocket**
ascent	24		
drogue	68		
reefed	164	170	211
main	88	80	49

Table 13: Mission Elapsed Time

6.2 Drift Calculations

The drift calculations using the above times and a constant horizontal wind speed are shown below in Table 14.

Windspeed		**Section**		
MPH	*ft/s*	*Nosecone*	*Payload*	*Rocket*
5	*7.3*	2348	2338	2410
10	*14.7*	4695	4676	4820
15	*22.0*	7043	7014	7231
20	*29.3*	9390	9353	9641

Table 14: Drift values

The above drift values for the worst-case conditions (main vehicle segments in 20 mph winds) are visualized below over the approximated launch site in Figure 45.

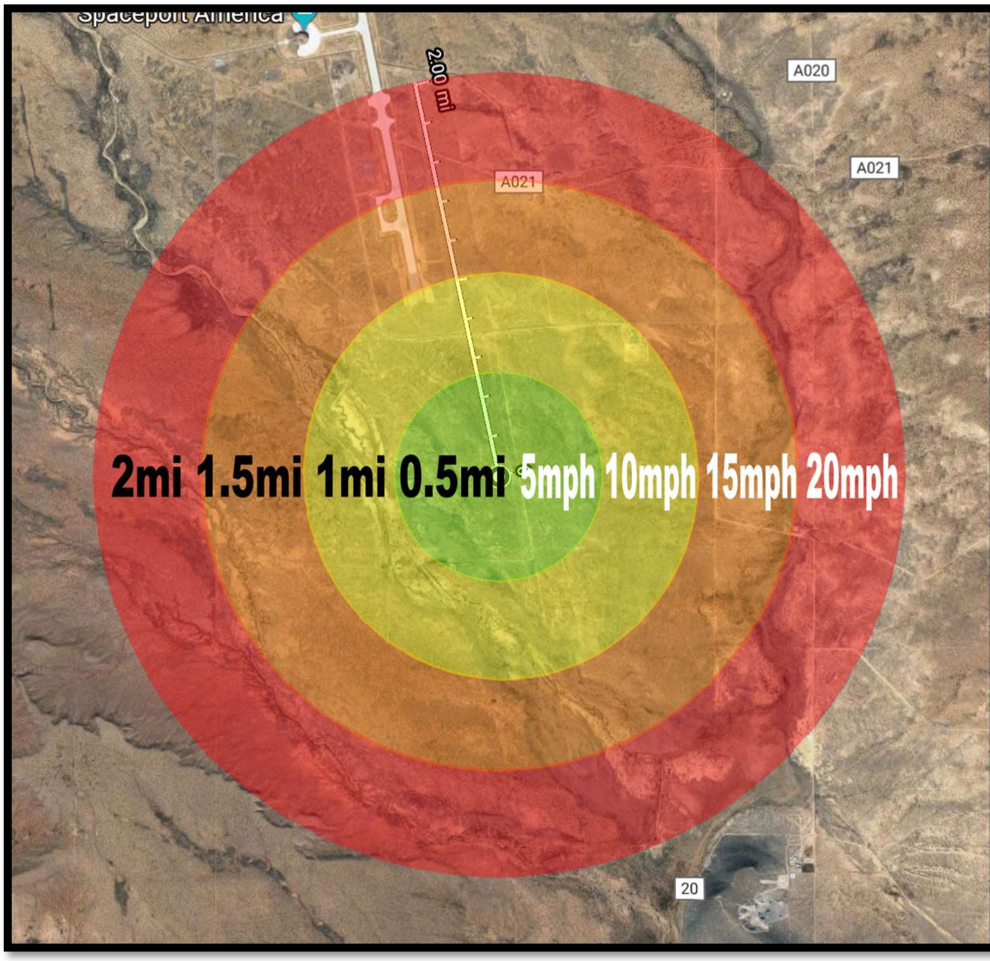


Figure 45: drift visualization

Opening Forces were calculated for the ringsail main parachute and toroidal secondary parachutes to ensure the rigging materials would not break during deployment. Using the weights of the launch vehicle, payload and nosecone, opening force was found using

$$F_x = (1/2)(C_D S)_P \rho V^2 C_x X_1$$

Where Fx is the opening force of the main parachute, $(C_D S)_P$ is the drag coefficient proportional to the surface area of the parachute, ρ is the density of air at ground level, V is terminal velocity at opening, C_x is the opening force coefficient, and X_1 is the opening force reduction factor. C_x and X_1 are scaleable constants reliant on the design of the parachute. Using data from text resources and previous team flights, we can obtain an X_1 and a C_x for a ringsail and toroidal parachute. The values obtained are shown below in Table 15.

Opening Force Calculator (Fx calc)	Newtons	lbs-f	G-force	m/s^2	ft/s^2
nosecone	20.87	4.69	0.94	9.19	30.16
payload	23.16	5.21	0.68	6.64	21.77

rocket	82.94	18.64	0.43	4.25	13.95
reefed parachute	631.98	142.07	2.55	25.02	82.08

Table 15: Opening force from equation

These values were then solved for using a deceleration method where the pre-opening and post-opening speeds were divided by a measured or estimated time. This acceleration was then multiplied by the masses of the objects to get the forces.

Opening Force Calculator (delta V)	Newtons	lbs-f	G-force	m/s^2	ft/s^2
nosecone	15.25	3.43	0.69	6.72	22.04
payload	130.26	29.31	0.68	6.68	21.92
rocket	77.46	17.43	0.41	3.97	13.03
reefed parachute	564.55	127.02	2.28	22.35	73.33

Table 16: Opening force from velocity change

6.3 Rigging Hardware

The shock cord chosen for the launch vehicle is a 9/16 in. tubular nylon cord rated for 1200 lbs which will be connected to the vehicle using 5/16 in. zinc plated steel quick-links rated for 1000 lbs. with a bowline hitch knot on each end of the cord. The bowline hitch was chosen for its self-tightening characteristics. The lengths of each shock cord was calculated by multiplying the length of the attached section by four as a rule of thumb. These lengths are shown below in Table 17.

Segment	Connections	Cord Length plus Knots (in)
Nosecone	Nosecone - Parachute	60
Recovery Bay	Payload Section - Drogue	208
	Drogue - ARRD	277
	ARRD - Bag	28
	Bag - Main	276
	Main - Booster	277
	Reef Line (Main Parachute)	375
Payload	Payload - Parachute	56

Table 17: Shock Cord Lengths

The drogue parachute shock cord, the deployment bag tether for the main parachute, and the reefing line will be directly tied to the attachment point of an Advanced Retention and Release Device (ARRD) and will not utilize a quick-link. The ARRD serves to connect a point in the rigging to the launch vehicle's bulk plate until a piston acted upon by a black powder charge contained within the body releases the shackle, moving the tension to the next hard point in the rigging. This will be used to release the drogue to pull out the main deployment bag as well as to release the reefing line that will constrict the main parachute. An ARRD is shown below in **Figure 24: ARRD assembly**

Most all of the other hardpoints used will be zinc plated steel 3" U bolts rated for 1075 lbs. shown below in Figure 18 .

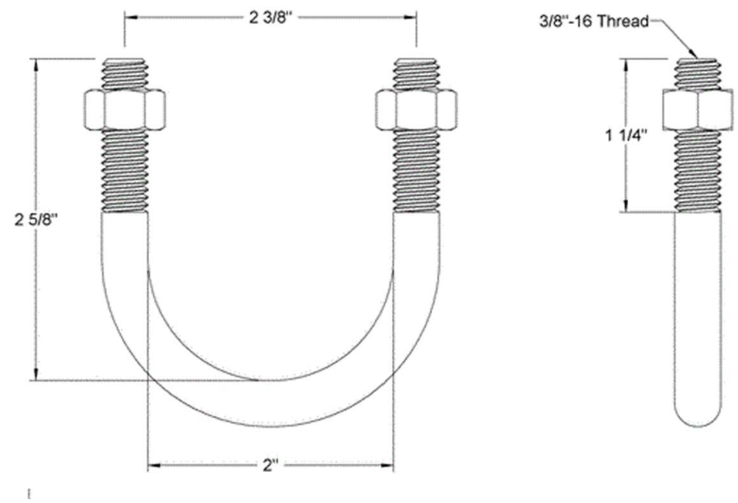
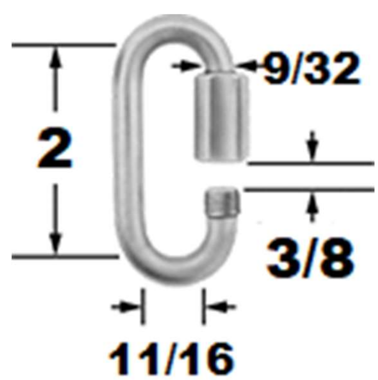


Figure 18: U bolt

These will be attached to the bulkplate with a large washer and multiple nuts for better thread engagement. The riggings system will utilize quick links made of zinc plated steel rated for 1000 lbs. in order to eliminate the action of tying knots deep within the airframe. They will serve as a connection point between the shock cord and the U bolt. A dimensioned drawing of one of these quick links is shown below in Figure 19 .



46.

Figure 24: ARRD assembly

Most all of the other hardpoints used will be zinc plated steel 3" U bolts rated for 1075 lbs. shown below in Figure 18 .

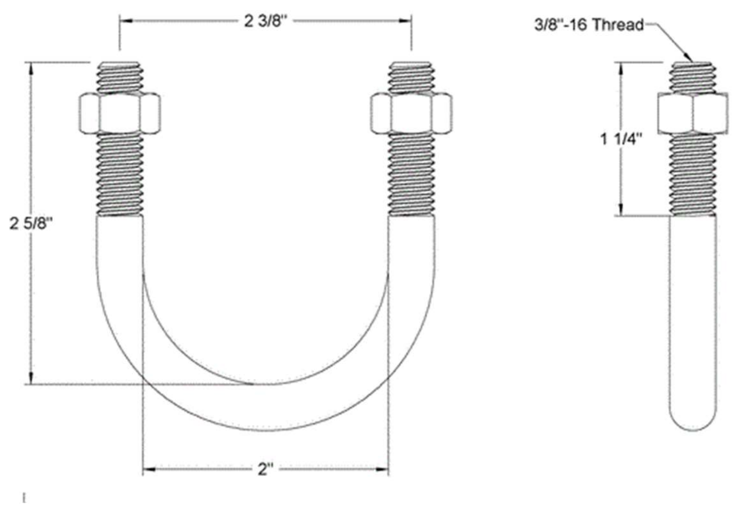
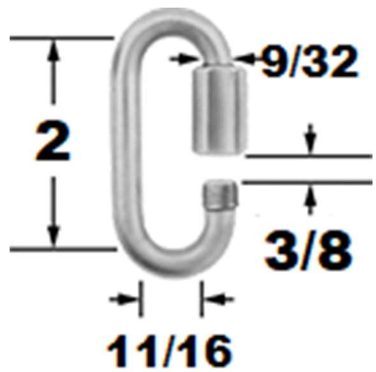


Figure 18: U bolt

These will be attached to the bulkplate with a large washer and multiple nuts for better thread engagement. The riggings system will utilize quick links made of zinc plated steel rated for 1000 lbs. in order to eliminate the action of tying knots deep within the airframe. They will serve as a connection point between the shock cord and the U bolt. A dimensioned drawing of one of these quick links is shown below in Figure 19 .



46Figure 19: Quick link

A custom deployment bag will be produced for the main parachute to protect it from pyrotechnic energetics, snagging, or tangling during the deployment process. The deployment bag will be constructed out of fire-retardant spray-treated canvas material with an inner lining of the same ripstop nylon as the parachute material. The nylon is a low friction material resulting in a smoother deployment. The bag will feature shroud line loops along the outside of the bag to ensure an orderly unfurling of the lines before the canopy is released. The bag will also feature a pouch to retain a shock cord that will keep the payload and booster segments attached during main descent. All secondary parachutes and drogue parachute will be wrapped in a fire retardant Nomex fabric square for simplicity.

6.4 Electronics

For all separation and parachute deployment events, redundant altimeters will be used. The altimeters will be programmed to light a pyrotechnic charge via an electric match that will initiate recovery events. Redundant altimeters will be used for each recovery event as a backup to the first altimeter in the case of a malfunction. To prevent over pressurization of the airframe/parachute mortar, the redundant events will be programmed 1 second apart from each other. Two types of altimeters will be used for the recovery events.

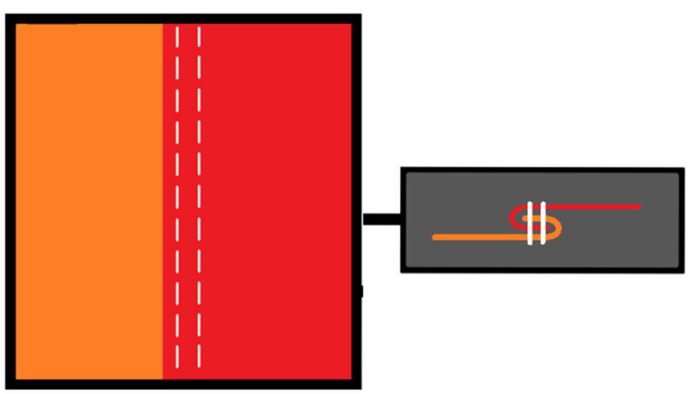
RRC3 “Sport” Altimeters will be used for the drogue and main parachute – as 3 events are needed. These altimeters allow for 2 events and one auxiliary event. The first event will separate the recovery section from the payload with a BP charge (which will pressurize the recovery bay) at apogee to release the drogue parachute. The next event will release the reefed main parachute using the ARRD. The third event will release the reefing line to unreef the main parachute using the ARRD. The RRC3 Altimeters will be stored in the booster/recovery coupler.

StratoLoggerCF Altimeters will be used for the separation event of the nosecone from the payload section and the deployment of the nosecone parachute. The first event will activate the BP charge to pressurize the vehicle cabin to separate the nosecone. The second event will activate the BP charge in the parachute mortar to deploy the nosecone parachute so the nosecone can be recovered independently. The redundant StratoLoggerCF altimeters will be housed in the nosecone on a 3D printed sled.

6.5 Design

All manufactured parachutes will feature zero porosity ripstop nylon which will be CNC laser cut or water jet into panels while pinned to a wooden board. Cutting out the panels in this manner has two primary benefits. First, it adds a level of precision, speed and repeatability that cannot be matched by cutting the panels out manually. Secondly, as nylon is a thermoplastic made from synthetic polymers, it can be melted to fuse edges and retain strength; using a laser to cut out the panels therefore also serves to cauterize the panel edges. To use the Universal Laser System (ULS), a board of eighth inch plywood is cut to the exact dimensions of the ULS. The ripstop is then stretched out across the board and pinned into place using thumbtacks as can be seen in.

Ripstop nylon was chosen due to its exceptionally low porosity, high tensile strength and low weight to surface area ratio. ripstop nylon is named accordingly for the crosshatched reinforcing threads that are woven into the fabric. For all ripstop stitches, a single straight stich is used due to the lesser number of holes it punctures into the fabric. Panels are sewn together in a 0.5 in. wide clasping style which maximizes the amount of grip the two panels have when inflated and stretched and reduces the chances of the seams being undone. This is shown below in. All stiches were done using Anefil Bonded Nylon Fire Resistant thread.



and shroud lines made from 1/8" diameter round nylon cord rated for 400 lbs. the main parachute features 10 active lines, making the tensile strength 4000 lbs when combined. The parachute will not have much eccentric loading if at all due to even tension being a requirement for proper inflation.

6.5.1 Drogue

The drogue parachute will serve to carry the entire vehicle from its apogee to 3000 ft. at 70 mph. The design chosen was a custom "SkyAngle™" of 34.2 inches in diameter. The SkyAngle™ parachute has a single circular top with three triangular panels extending downwards which can be manufactured relatively quickly and with minimal asymmetry. This will help to reduce the amount of rotation compared to the instability past drogue parachutes have exhibited. The shroud lines are sewn into the bottom three tips of the triangles and extend up to the center. The SkyAngle™ parachute will be manufactured by laser cutting or water jetting zero porosity ripstop nylon panels and sewing them together with nylon thread. The diameter of the drogue was calculated using the surface area in equation ## below.

$$V_e = \sqrt{\frac{2mg}{C_D S_O \rho}}$$

where m is the mass of the launch vehicle, g is acceleration due to gravity, S_O is the surface area, C_D is the coefficient of drag, ρ is the density of air at the launch site (4,000 ft above sea level), and V_e is the terminal velocity. We chose a velocity of 70 mph to produce a parachute having slightly more of a surface area than that of the launch vehicle if on its side. This prevents any kind of backsliding to occur if the fins were to create lift. The dimensions for the drogue parachute are shown below in Figure 47.

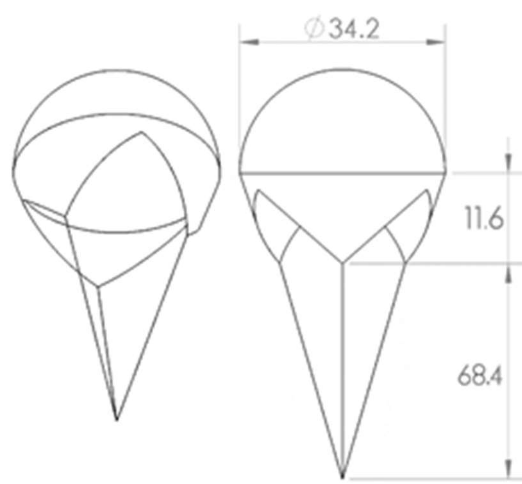


Figure 47: Drogue parachute dimensions

6.5.2 Main

This season, the nature of payload deployment warranted a more controlled decent of the launch vehicle. In order to achieve this decent earlier than main deployment, a reefed main parachute will be deployed at 3000 ft. and unreefed at 800 ft. This will decrease the falling speed from 70 mph to 15 mph and then to 11 mph. The main parachute will be a custom ring-sail elliptical parachute with two rings at the edges and a large inner panel with one central vent hole. These rings create slots in the parachute near the skirt where air can enter from the "outside-in" to inflate the parachute when reefed. In the full inflated state, the parachute will bring the remaining launch vehicle segments to a ground hit velocity of 11.11 mph or a KE at landing of 101 joules to meet NASAs standards of a safe landing per the USLI (our previous competition) handbook. This was the determining factor in the size and was used to calculate it. The following equation was used

$$D_o = \sqrt{\frac{4m_v m_s g}{\pi E C_D \rho}}$$

Where D_o is the effective diameter, m_v & m_s are the mass of the total vehicle and the mass of the heaviest segment, respectively, E is the kinetic energy at landing, C_D is the coefficient of drag of 0.9, g is the gravitational constant, and ρ is the air density at ground level. Next, the shroud line length was determined by multiplying the diameter of the parachute by two. The schematics of the parachute are shown below in Figure 48.

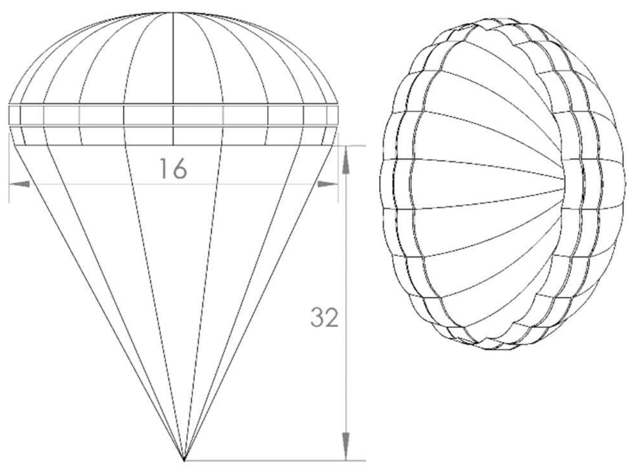


Figure 48: Main parachute dimensions

The main parachute will feature ten lines with ten more mid panel lines to connect the bands to the main panel. This will mitigate any inverting or flipping seen in the bands during decent. This parachute will also be reefed by a passive system. Metal O-rings will be attached to each of the shroud line attachment points and a flat nylon shock cord rated for 1400 lbs will be run through each of the rings. This cord will be cut to a specific length that will keep the parachute reefed at 80% of its fully inflated state while also reaching the bulkplate where the parachute is attached. It will be connected to an ARRD which will release the line to let the parachute fully open at 800 ft.

6.5.3 Nosecone

The nosecone will separate from the payload to allow the payload to exit the launch vehicle. Once the nosecone separates from the payload section at 2200 ft. it will be recovered independently by a 36 in. toroidal parachute. Toroidal parachutes are modified elliptical parachutes where the center vent hole is pulled lower into the canopy by a centerline to increase the coefficient of drag. The diameter of this parachute was chosen so that the kinetic energy of the nosecone upon landing would be less than the recommended 101 Joules. This parachute will be purchased (not manufactured in house). The parachute will be stored within the nosecone in a mortar tube with a pressure sealed cap. After the nose cone separates, the altimeter will set off a second BP charge which will expel the parachute and cap. The cap will be retained via a small tether to an eyebolt affixed to the all thread protruding from the nosecone's bulkplate. This is shown below in Figure 49: ***Nose Cone Recovery System***

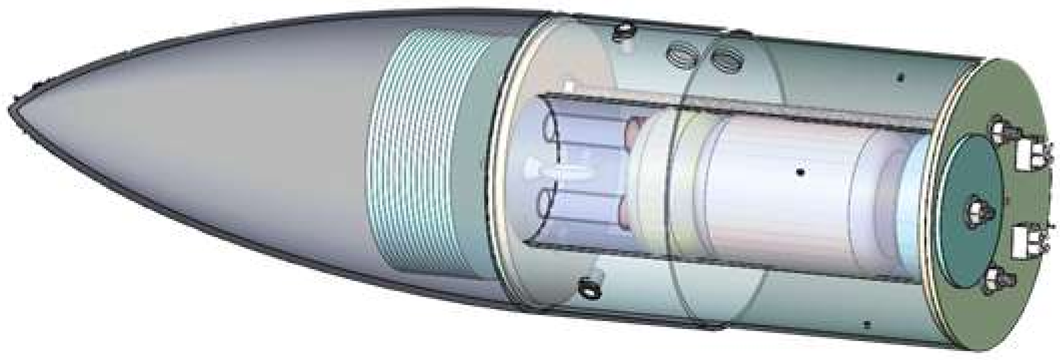


Figure 49: Nose Cone Recovery System

7 Payload

7.1 Overview

7.1.1 Mission Overview

This year's experimental payload mission is to design and build a payload capable of detecting ground targets. This mission will be performed with the construction and operation of a deployable fixed-wing unmanned aerial vehicle (UAV). The UAV will be retained in the airframe prior to deployment, and at a determined altitude the payload section will separate. At separation, the UAV will deploy under parachute. Following successful deployment, pre-flight performance checks will be manually triggered to determine if safe flight is possible per requirements. Upon the telemetry ground station's receipt of successful pre-flight checks, the UAV will be manually triggered to separate from parachute and begin autonomous function. It will autonomously navigate to ground targets via GPS waypoints, while streaming live video to the ground station, and complete the mission by return for landing at the launch site.

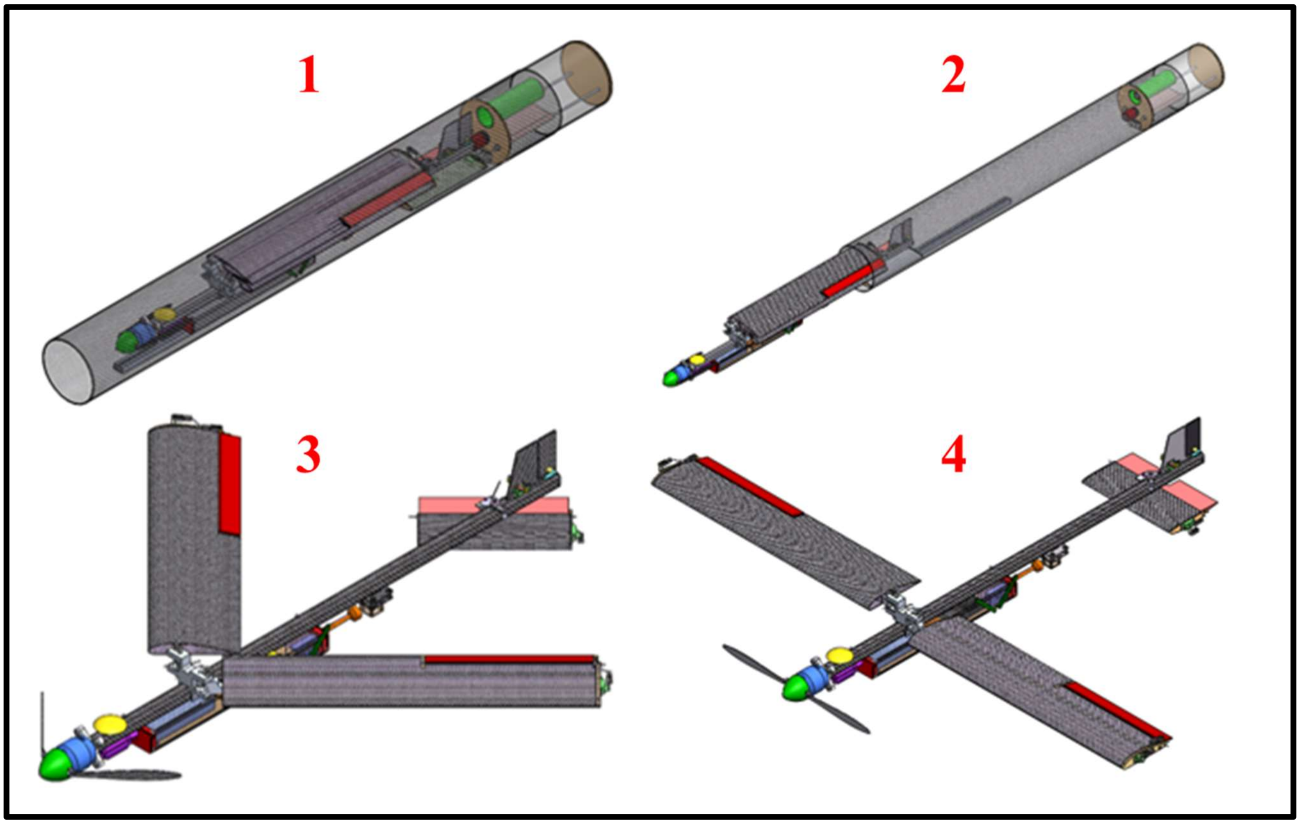


Figure 50. Payload deploymment sequence.

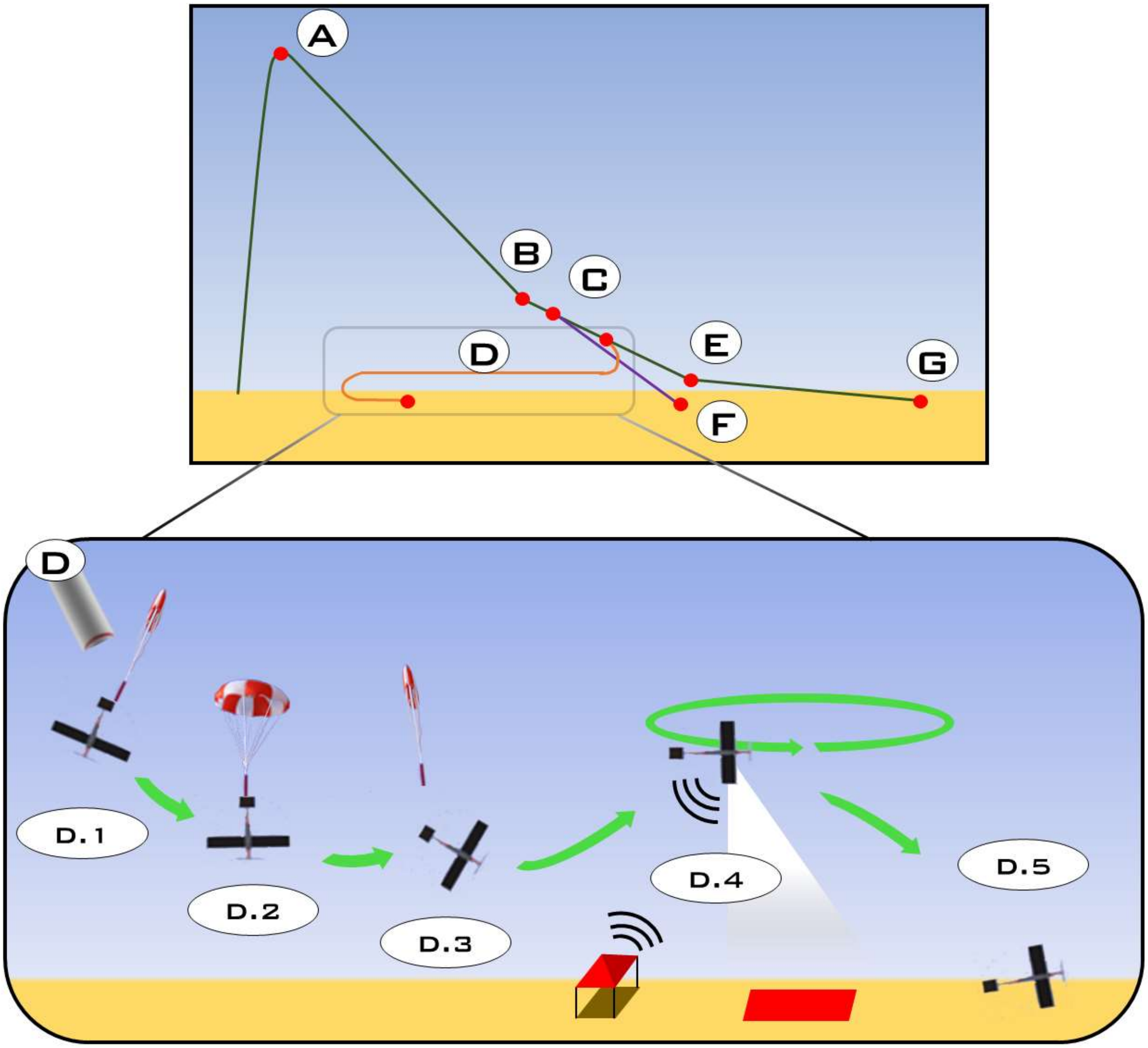


Figure 51. Payload mission overview diagram.

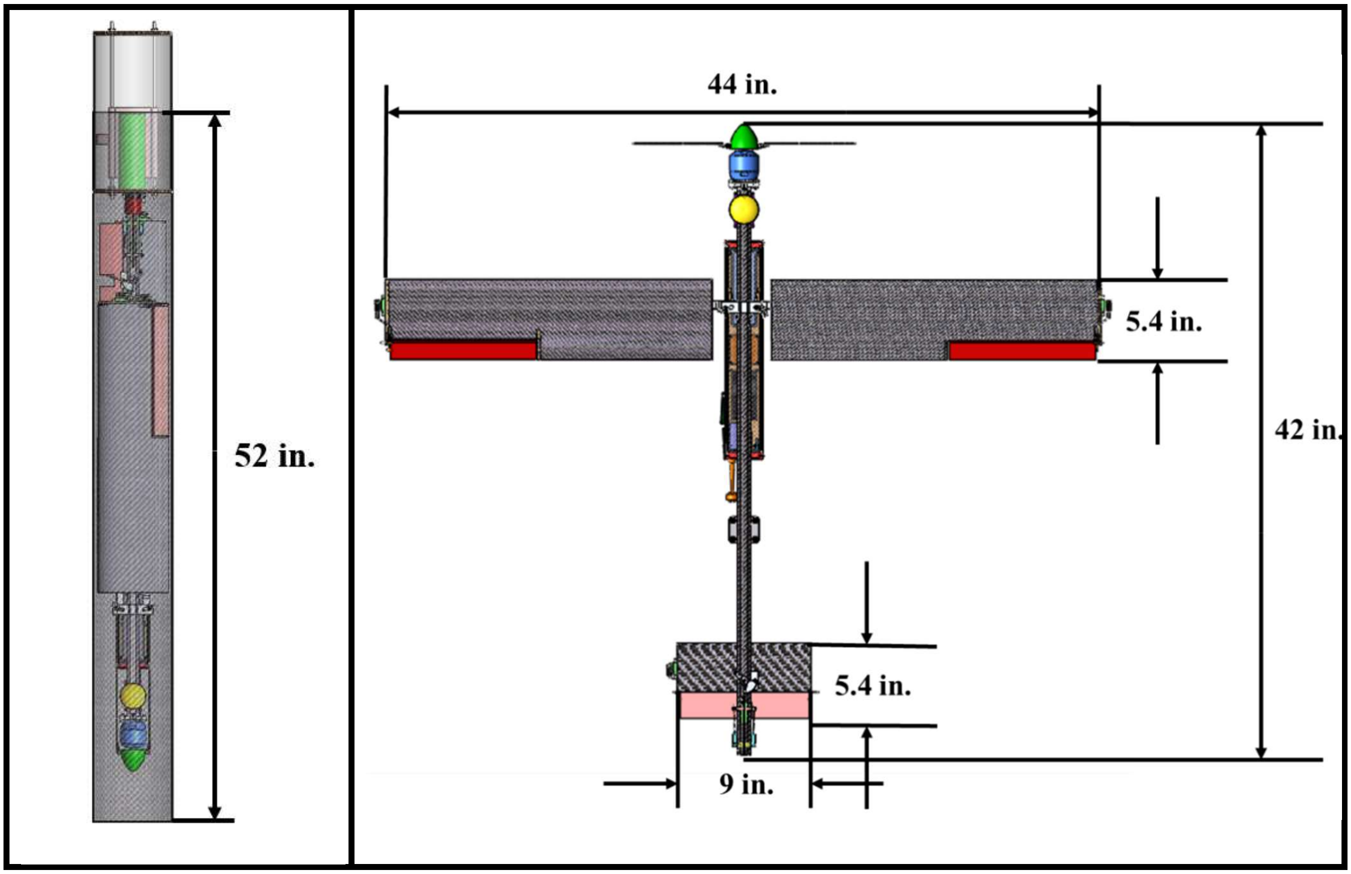
Mission Overview Event	Event Name	Event Description
A	Apogee	Payload remains stowed inside launch vehicle.
B	Main Deployment and Reefed State	Payload remains stowed inside launch vehicle.
C	Nosecone Separation	Payload remains stowed inside launch vehicle. However, front

		of the payload is now exposed to open air.
D	Payload Deployment	Triggered by a PerfectFlite Stratologger CF. An ARRD attached to the payload bay AFT bulkplate releases the payload from the launch vehicle. The payload deployment altitude will be determined by test XX .
D.1	Payload Deployment Under Parachute	Once separated from the launch vehicle, the UAV will fall under its independent parachute. Actuation and locking of all deployment mechanisms occurs during this event. The UAV is now fully in the flight configuration.
D.2	Pre-flight Checks Completion	Preflight checks will be conducted on each of the UAV’s 4 control surfaces. Full actuation of each control surface will occur to ensure all systems are ready for flight. After checks are completed, another signal will be sent to the UAV
D.3	Manual Parachute Release	Once the “go” signal is received from the UAV’s preflight checks, a manual detachment signal will be sent from the ground station to the UAV. This detachment signal will trigger an ARRD to release the payload for self-sustaining flight.
D.4	Free Flight and Target Detection	Payload cutaway from airframe, flight computer begins attempts to calculate level flight path.
D.5	Payload Skid Landing	After all waypoints have been achieved, the UAV will navigate to the final landing waypoints to conduct an autonomous skid landing.

Table 18. Payload mission overview.

7.1.2 Payload Design Overview

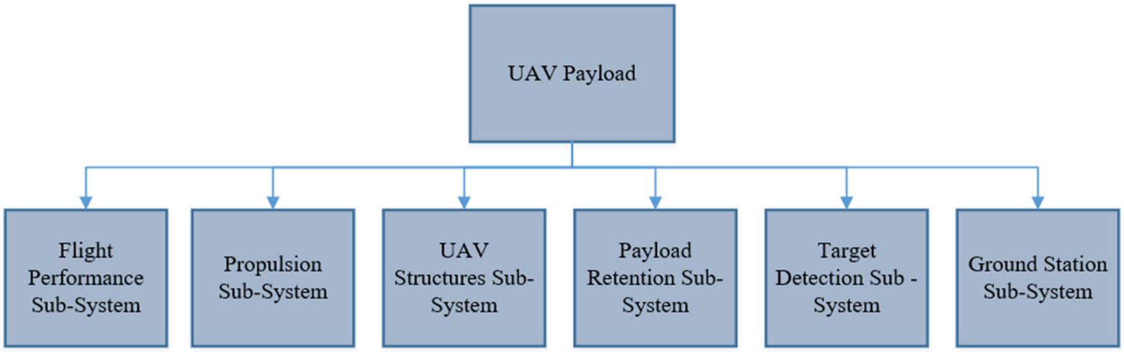
The payload's construction mainly consists of carbon fiber, foam, and 3D printed components. The payload's two mechanisms will be constructed out of aluminum. The payload's major dimensions are shown below.



Dimension	Value (in.)
Payload Bay Length	52
UAV Length	42
Wingspan	44
Chord	5.4

Table . UAV overall dimensions.

7.1.3 Payload Subsystem Overview



Payload Subsystem	Subsystem Overview
Flight Performance Subsystem	This consists of any component whose function directly effects the payload's flight characteristics. Flight controllers, wings, stabilizers, and all control surfaces fall within this sub-system.
Propulsion Subsystem	For a fixed-wing aircraft to sustain flight a reliable method of thrust production is required. The propulsion sub-system focuses on all components whose function is to provide thrust. This includes the battery, ESC, motor, and propeller.
Structures Subsystem	The structures sub-system includes any component responsible for structurally supporting the payload during captive carry and flight.
Payload Retainment Subsystem	Systems responsible for integrating, housing, and deploying the payload from the launch vehicle are considered the payload retention sub-system.
Target Surveillance Subsystem	The camera sub-system's main function is to detect ground targets through an on-board camera. It is also responsible for providing a live video feed from the camera to a ground station monitor.
Ground Station Subsystem	Components whose main function is to transmit and receive telemetry data. It also represents the controller for the UAV and its functions for deployment.

7.1.4 Payload Requirements

Requirements 1, 2 and 3 stated in the IREC Rules and Requirements serve as the baseline for all the payload's team derived requirements. Team derived requirements are also formed around the SDL payload objective which is stated in Requirement 4, which states that the payload must accomplish a relevant function and provide useful learning opportunities. From these high-level requirements set by Spaceport America Cup and the SDL, team derived requirements were created for each payload sub-system. These requirements are outlined in Figure 52**.**

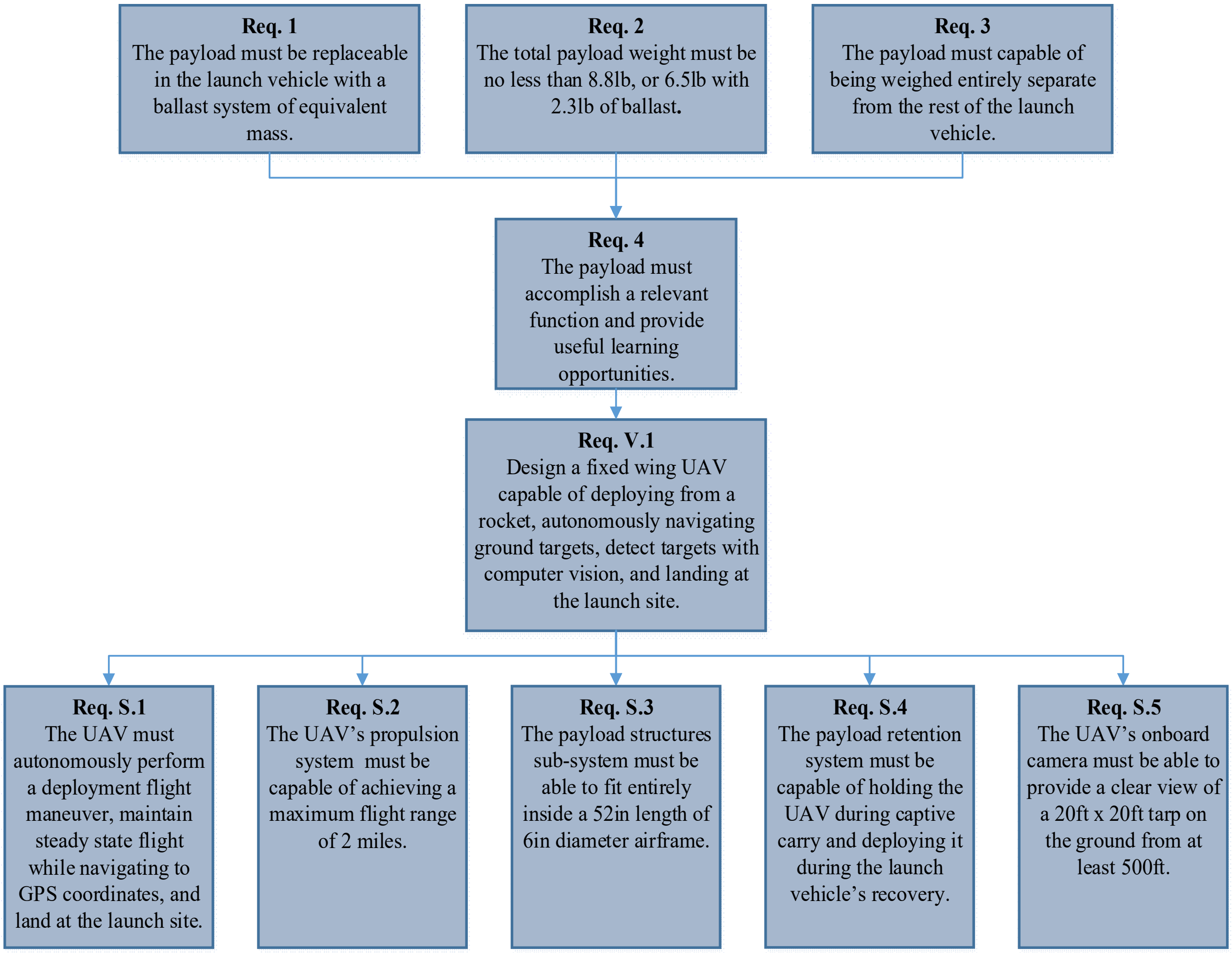


Figure 52: Payload system requirements flow diagram.

The payload system level requirements and team derived requirements, along with the corresponding methods of verification are shown in Table 3 below.

Requirement Number	Requirement	Method of Verification <u>(Test, Inspection, Demonstration, or Analysis)</u>
1	The payload must be replaceable in the launch vehicle with a ballast system of equivalent mass.	<u>Demonstration</u> During the launch vehicle's first test flight, a ballast will replace the payload to demonstrate that this requirement has been fulfilled.

2	The total payload weight must be no less than 8.8lb, or 6.5lb with 2.3lb of ballast.	Inspection A calibrated scale will be used to weight the payload.
3	The payload must be capable of being weighed entirely separate from the rest of the launch vehicle.	Analysis The payload will be designed to be completely removeable from the launch vehicle.
4	The payload must accomplish a relevant function and provide useful learning opportunities	Demonstration Design documentation and execution of the payload mission will demonstrate the satisfaction of this requirement.
V.1	Design a fixed wing UAV capable of deploying from a rocket, autonomously navigating to ground targets, detecting targets with computer vision, and landing at the launch site.	Test A fully integrated flight test of the UAV will be done on 5/4/2019. A sub-scale testing campaign will also be held for each payload sub-system to verify that all UAV components can complete their mission requirements.
S.1	The UAV must be able to autonomously perform a deployment flight maneuver, maintain steady state flight while navigating to GPS coordinates, and land at the launch site.	Test A full-scale UAV prototype will be constructed before 12/20/18. This test platform will allow for six months of flight testing to be conducted before competition.
S.2	The propulsion system must be able to sustain flight for a maximum range of 2 miles.	Test The RCTP will be flown for 2 miles using an identical battery, motor, and propeller as the final payload configuration. This test will be conducted before 12/20/18.
S.3	The payload structure's sub-system must be able to fit entirely inside a 40in length of 6in diameter airframe.	Analysis All structural components will be designed to fold or stow inside the allotted amount of airframe. A fitment inspection will also will be conducted with the RCTP before 12/20/2018.

S.4	The Payload Retention System (PRS) must retain the payload throughout captive carry and allow the UAV to separate from the launch vehicle's payload bay during recovery.	Test A fully integrated flight test will be done on 5/4/2019. This flight test along with full-scale ground testing and component testing will be responsible for verifying this requirement.
S.5	Cameras onboard the UAV must be able identify a 20ft x 20ft tarp on the ground from at least 500ft.	Test A flight test on the RCTP will be done to ensure the camera can identify targets from a 500ft altitude. Full scale flight tests will also be conducted to further satisfy this requirement.

Table 19: Functional requirements.

Derivation of Requirement V.1.1.2

The 2-mile maximum range of the UAV was selected to provide margin between the worst-case scenario maximum drift of the payload bay, and the maximum flight distance of the UAV. Satisfaction of this requirement ensures the UAV will always be within range of the ground targets and landing site.

Derivation of Requirement V.1.1.3

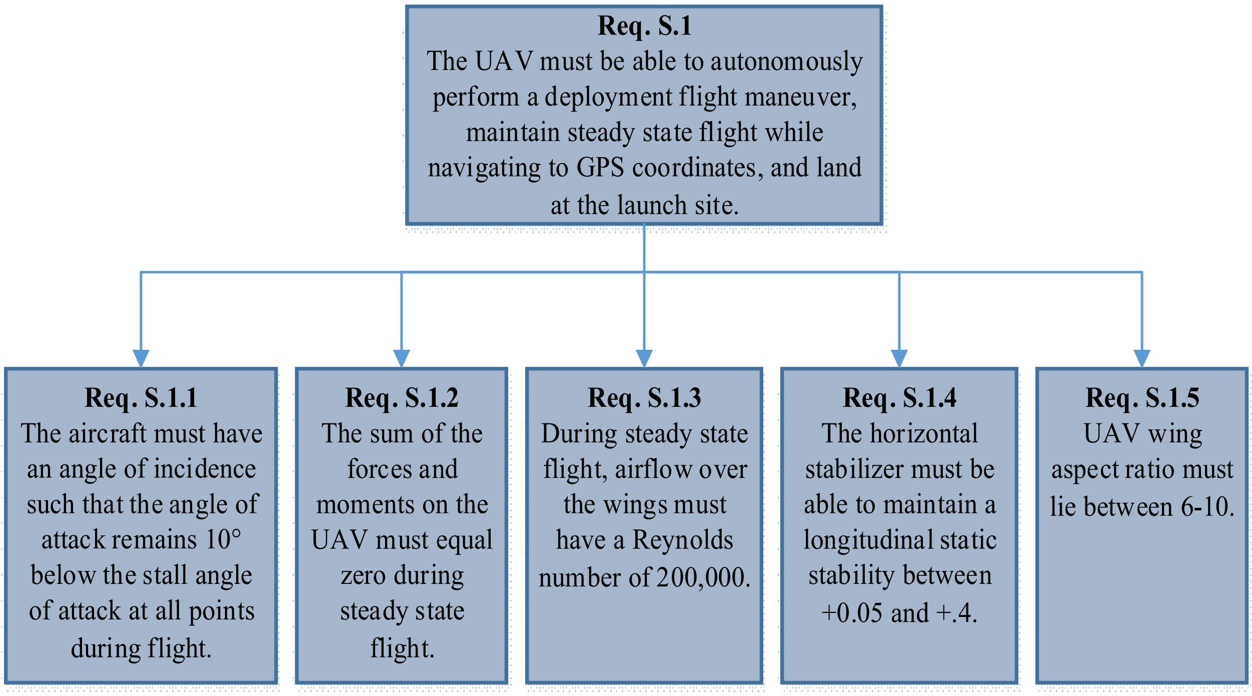
Airframe length requirements for the launch vehicle specify 52in of 6in diameter airframe is the maximum amount of space allotted for the payload. This 52in allows for 6in of coupler interference on both sides, leaving 40in of useable bay space.

Derivation of Requirement V.1.1.5

500ft will be the cruising altitude of the payload. Designing the camera system to detect targets at a cruising altitude simplifies the design of the flight controller's logic. This requirement eliminates the need to change altitude during the target detection phase of flight.

7.1.4.1 Flight Performance Sub-System Requirements

Flight performance is broken up into two main sub-systems, mechanical performance and controls performance. Mechanical performance includes the wings, stabilizers, control surfaces, any other structural component whose geometry or function drives the flight performance of the payload. Controls performance includes the flight controls hardware, flight software, and all electronic systems that enable the payload to operate autonomously.



Requirement Number	Requirement	Method of Verification <u>(Test, Inspection, Demonstration, or Analysis)</u>
S.1	The UAV must be able to autonomously perform a deployment flight maneuver, maintain steady state flight while navigating to GPS coordinates, and land at the launch site.	<u>Test</u> A full-scale UAV prototype will be constructed before 12/20/18. This test platform will allow for six months of flight testing to be conducted before competition.
S.1.1	The aircraft must have an angle of incidence such that the angle of attack remains 10° below the stall angle of attack at all points during flight.	<u>Analysis</u> The UAV will be designed to cruise at 10° below the stall angle of attack.
S.1.2	The sum of the forces and moments on the UAV must equal zero during steady state flight.	<u>Analysis</u> A free body diagram containing all external forces acting on the UAV during cruise will be created.
S.1.3	During steady state flight, airflow over the wings must have a Reynolds number of 200,000.	<u>Analysis</u> Chord length and cruise velocity of the UAV will be calculated based on Reynolds number.

S.1.4	The horizontal stabilizer must be able to maintain a longitudinal static stability between +0.05 and +.4.	Analysis The UAV will be designed so the CG, CP and neutral point are located to be statically stable.
S.1.5	The wing aspect ratio must lie between 6-10.	Analysis Dimensions of the wing will only be chosen if they meet this requirement.

Derivation of Requirement S.1.1

A 10-degree angle between the UAV's cruise angle of attack and the critical angle was selected according NACA airfoil data. According to the NACA airfoil database most airfoils significantly lose lift beyond 10 degrees angle of attack. The requirement allows the UAV to fly with no angle of attack and still meet this requirement. A 10-degree margin was determined to be a conservative angle that ensures the UAV will not hit its stall angle of attack despite any flight anomalies.

Derivation of Requirements S.1.2

This requirement was implemented to ensure that the UAV is statically stable about all degrees of freedom. Fulfilling this requirement also ensures that the propulsion system produces a thrust equal to the expected drag. In depth analysis of the free body diagram is covered in the flight performance technical design section.

Derivation of Requirement S.1.3

The NACA airfoil database provides test data for all its airfoils at set Reynolds numbers of 50K, 100K, 200K, 500K, and 1 million. All flight performance calculations are done assuming a Reynolds number of 200K at the lifting surfaces. This ensures the calculated properties closely match the listed properties in the NACA database. The lift calculated from multiple airfoils at a Reynolds number of 200K all satisfied Req. S.1.6. Multiple airfoils were evaluated at both a Reynolds number of 100K and 200K. No airfoils evaluated at a Reynolds number of 100K generated enough lift at cruise to satisfy requirement S.1.2, while those evaluated at 200K did.

Derivation of Requirement S.1.4

According to an MIT report detailing aircraft longitudinal pitch stability, a Static Margin (SM) of +0.15-+0.40 provides a statically stable aircraft. The range allows for accommodation for the limited axial bay length in the airframe. The SM range allows for a range of locations for the CG of the aircraft and restricts the placement of the electronics bay. From Figure 53Figure 53 it is noted that a static margin of +0.15 to +0.40 is considered strongly stable.

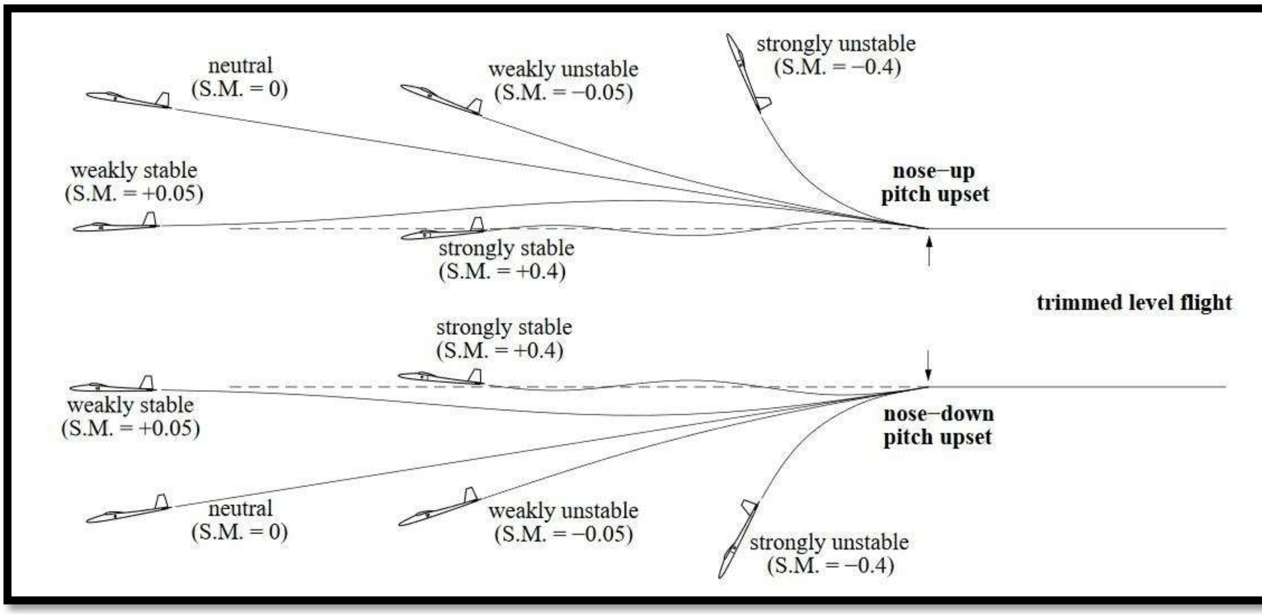


Figure 53. Static margin and stability graphic.

Derivation of Requirement S.1.5

The range of desirable aspect ratios from 6-10 was selected based on empirical data from multiple aircraft design handbooks. This span covers the "Moderate -speed sport" to "Low-speed trainer" power band. In Figure 54, a correlation table from *Lennon's RC Model Aircraft Design* [1] is shown.

TABLE 1

Model type	Power loading oz./cid 2-stroke	Wing loading oz./sq. ft.	Aspect ratio
High-speed, highly maneuverable	200-250	22 to 26	4 to 6
Moderate-speed sport	250-300	16 to 22	6 to 8
Low-speed trainer	300 and up	12 to 16	8 to 10
Slope gliders	—	12 to 14	8 to 10
Soaring gliders	—	8 to 12	10 to 15

Figure 54. Common model aircraft aspect ratios.

The wings are attached to the fuselage by an actuating mechanism. High aspect ratios above 10 create large bending moments at the wings root and at the actuating mechanism. High aspect ratio wings also take up more axial bay length in the stowed configuration. Low aspect ratios below 6 are also undesirable since the UAV's mission does not require a highly maneuverable vehicle.

7.1.4.2 Propulsion Sub-System Requirements

Team derived sub-system requirements were created for the propulsion sub-system. These requirements ensure that the UAV can satisfy high level requirements XXXX and XXX. A flow diagram showing the requirements is shown below.

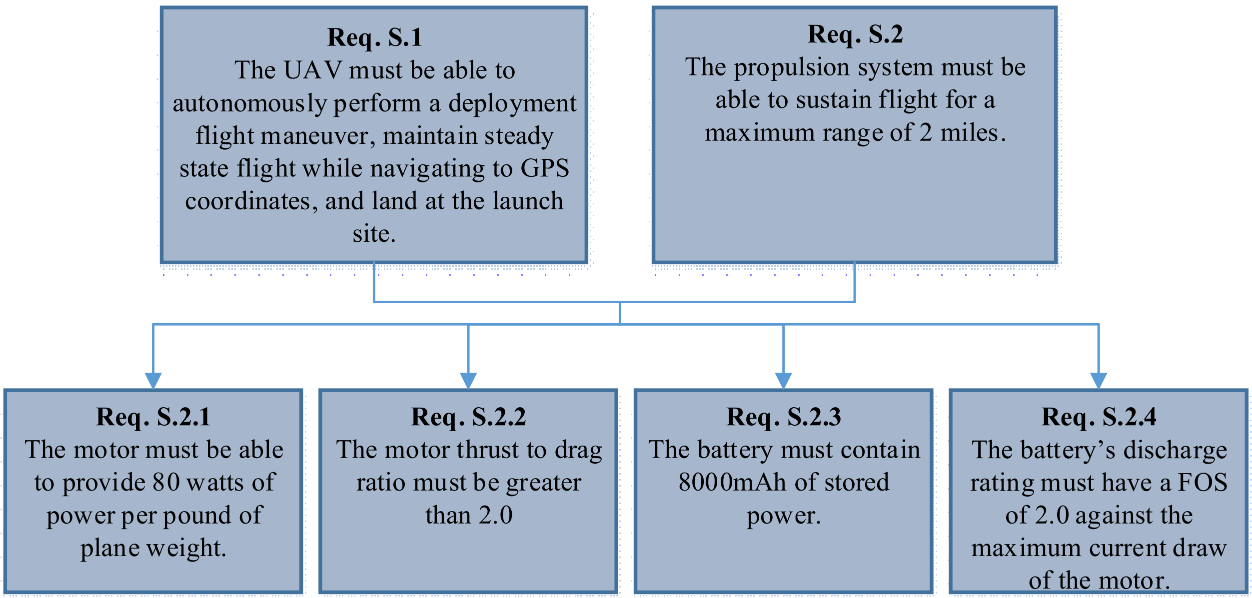


Table 20 below describes each requirement and its method of verification.

Requirement Number	Requirement	Method of Verification (Test, Inspection, Demonstration, or Analysis)
S.2	The battery powering the UAV motor must be able to supply the required current and voltage for a flight of 2 miles.	Test The RCTP will be flown for 2 miles using an identical battery, motor, and propeller as the final payload configuration. This test will be conducted before 12/20/18.
S.2.1	The motor must be able to provide 80 watts of power per pound of plane weight.	Test Full scale flight tests will be conducted to verify the motors power output. Component tests will also be done to measure the motor's thrust.

S.2.2	The motor thrust to drag ratio must be greater than 2.	<u>Testing</u> A benchtop test rig will be constructed to ensure the selected motor/propeller configuration provides the required thrust.
S.2.3	The battery must contain 8000mAh of stored power.	<u>Inspection</u> The battery will be inspected to make sure all technical specifications meet this requirement.
S.2.4	The battery's discharge rating must have a FOS of 2.0 against the maximum current draw of the motor.	<u>Testing</u> Component testing will be done to ensure the motor will not pull too many amps.

Table 20. Propulsion system requirements.

Derivation of Requirement S.2.1

A minimum power output of the UAV's motor was determined using the Watts per pound rule. Figure 55 illustrates common power consumption for model RC airplanes.

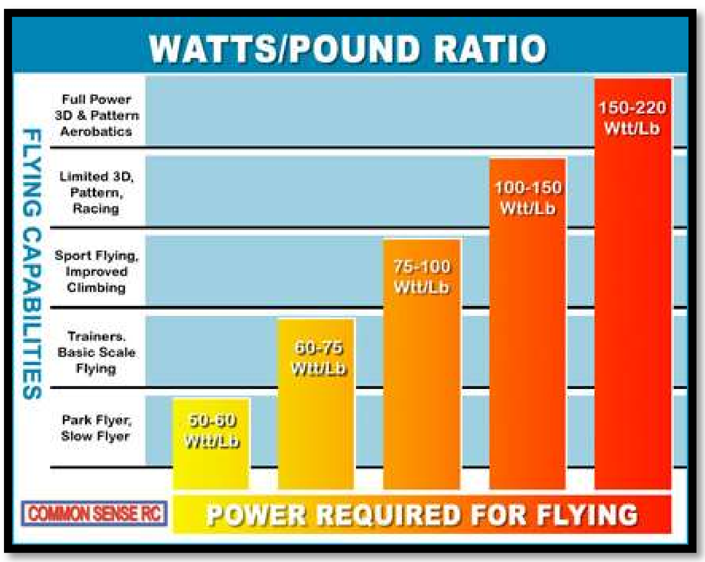
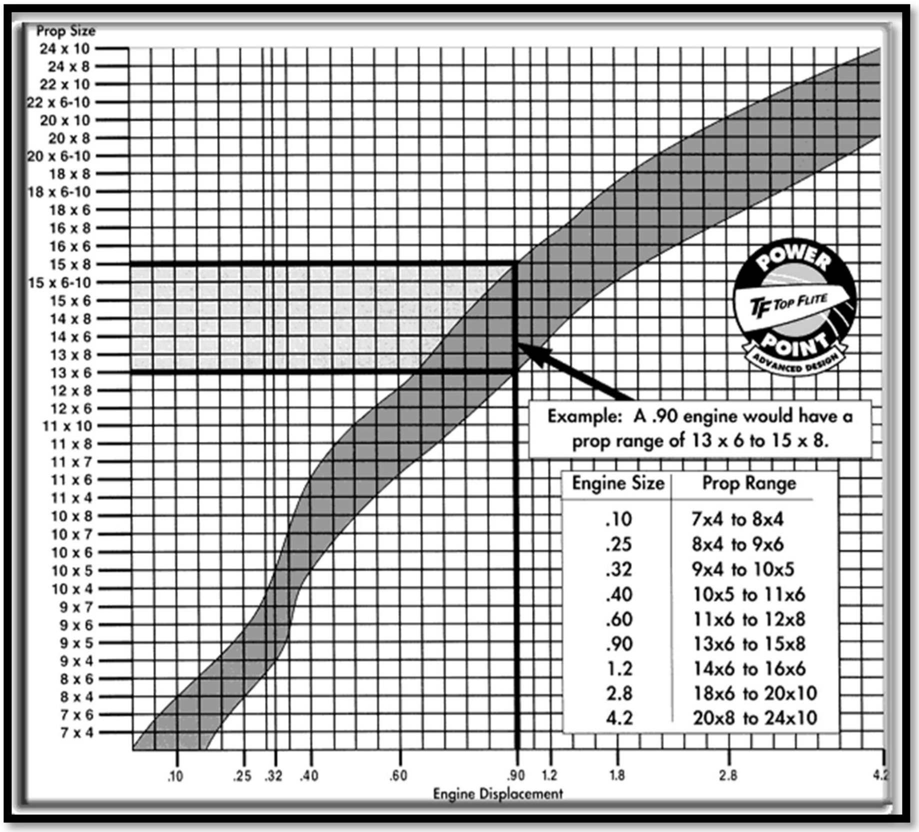


Figure 55. Watts per pound rule.

The payload is designed to be a surveillance aircraft and does not require the power to conduct acrobatic flight envelopes. Based on the payloads mission a watt per pound rating of 80 was selected. Once this rating was selected, the required motor output for cruise was calculated using

$$P_R = P_{lb} W \qquad \textbf{(8)}$$

where P_{lb} is the watt per pound rating, and W is weight of the UAV in pounds. Evaluating (8 a required power output of the motor to sustain cruise at a 0 degree angle of attack was determined to be 420W.



Equation 9. Propeller sizing chart.

Derivation of Requirement S.2.3

The required battery life for a given subsystem and stage of flight is the product of current (in milliamps) and time (in hours) during which that current is supplied or:

$$B = It \tag{1}$$

Where B, I, and t, represent battery life, current and time respectively. The first part of the flight will be as the UAV is packed into the launch vehicle, at which point the motor will not be connected and will draw no current. The battery will need to power the flight control and telemetric electronics. The second part of flight is post deployment, when the UAV will also be powering servos for control surfaces, and the main motor (a Black Power Up 32 Sport 800Kv outrunner brushless motor).

Table 21 shows test data for the UAV electronics, specifically the Pixhawk Px4 flight controller (which also includes a barometer, magnetometer, accelerometer,

Component	Current Draw (mA)
Pixhawk Px4	175
GPS	55
Compass	5

Table 21. Test data for current use of flight control electronics

During flight the motor will pull a maximum of 50 Amps. Assuming maximum current during flight and a 2 mile flight at 47 mph (153 seconds) plus a thirty second overhead for deployment from the launch vehicle, Eq. 1 gives a battery life contribution of about 2550 mAh.

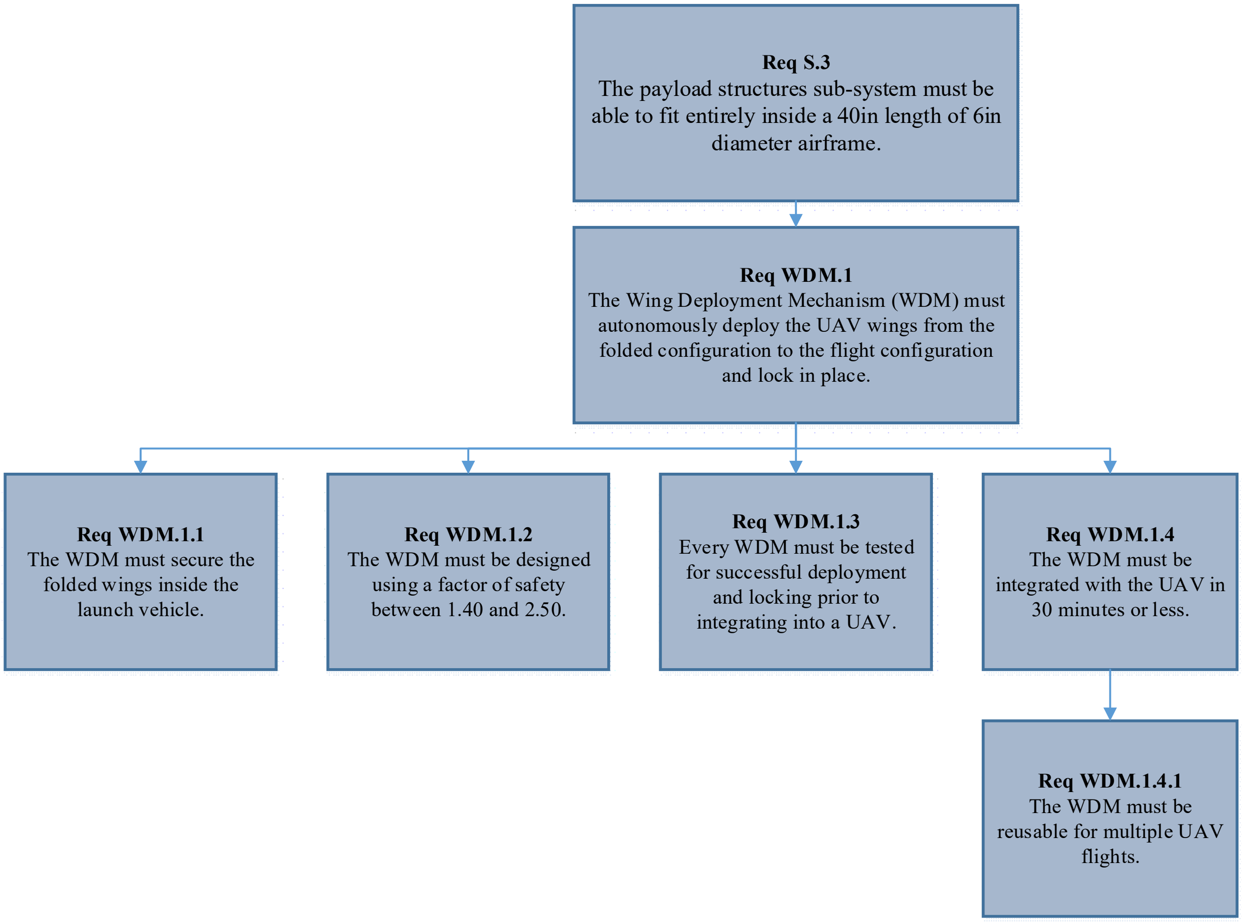
These produce a sum of 3255 mAh. Lastly the telemetry system onboard the UAV must be powered, and a conservative factor of .7 must be applied. An equation that modeled the overall battery life needed then would be:

$$\frac{(3255+\quad {}_{T})}{0.7} \tag{2}$$

where I_T is the supply current to the telemetry transceiver (in mA), multiplied by 3 hours. This component is also derived from Eq 1. Based on this equation, an estimated battery size is shown in Figure 25 below.

7.1.4.3 Wing Deployment Mechanism Requirements

These high-level functional requirements outline the WDM's objective to autonomously deploy the UAV wings from the folded configuration to the flight configuration and lock in place after detaching from the launch vehicle's payload bay.



Requirement Number	Requirement	Method of Verification
S.3	The payload structures sub-system must be able to fit entirely inside a 40in length of 6in diameter airframe.	<u>Test</u> The payload must complete a fitment check to ensure nominal fit.
WDM.1	The Wing Deployment Mechanism (WDM) must deploy the UAV wings from the folded configuration to the flight configuration and lock in place.	<u>Test</u> A testing campaign consisting of several component and full-scale tests to verify the WDM's locking reliability.
WDM.1.1	The WDM must allow the folded wings to fit inside the launch vehicle airframe.	<u>Demonstration</u> A fitment inspection will be conducted with the RCTP before 12/20/2018.
WDM.1.2	The WDM must be designed using a factor of safety between 1.40 and 2.50.	<u>Analysis</u> Hand calculations and finite element analysis will be performed to verify final WDM design.

WDM.1.3	Every WDM must be tested for successful deployment and locking prior to integrating into a UAV.	Test Deployed wings will be subjected to simulated flight conditions to test locking mechanism.
WDM.1.4	The WDM must be integrated with the UAV in 30 minutes or less.	Test A stopwatch will be used to measure the WDM integration time.
WDM.1.4.1	The WDM must be reusable for multiple deployments.	Test The WDM must allow the wings to return to folded configuration and must successfully redeploy to the flight configuration.

Table 22. WDM sub-system requirements.

Derivation of Requirement S.3
The WDM ensure the payload's main wings fit inside the payload bay airframe.

Derivation of Requirement WDM.1
The WDM must successfully deploy the wings from the folded configuration to flight configuration to allow the UAV to complete its mission. If the WDM is unsuccessful, the UAV will fail the mission.

Derivation of Requirement WDM.1.1
The UAV must be completely stowed in the launch vehicle's payload bay, therefore the WDM must be capable of stowing the wings inside the airframe.

Derivation of Requirement WDM.1.2
According to the NASA Structural design and test factors of safety for spaceflight hardware, 1.40 is the minimum required FOS against ultimate strength for a metallic fastener. To prevent excessive weight from oversizing fasteners, a maximum FOS of 2.50 was selected.

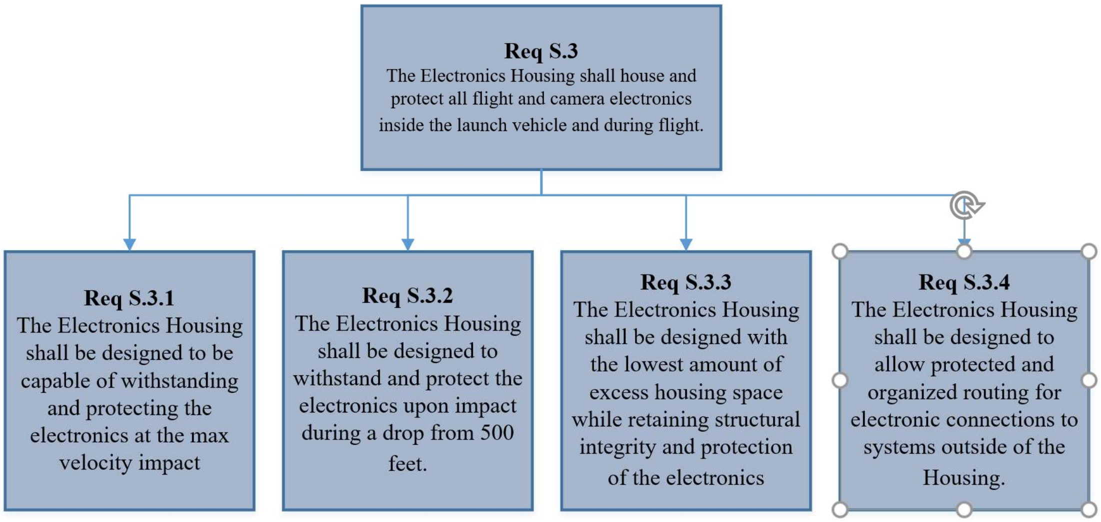
Derivation of Requirement WDM.1.3
The WDM must be tested to determine whether the WDM can rotate the wings from the stored configuration to the flight configuration and lock in place. The WDM tests must be completed prior to assembling any WDM's onto a flight vehicle. Unsuccessful WDM's cannot be assembled with the UAV until a successful test is completed.

Derivation of Requirement WDM.1.4
The WDM must be integrated with the UAV in 30 minutes or less to reduce UAV assembly time to simplify UAV assembly and reduce launch day assembly time.

Derivation of Requirement WDM.1.4.1
The WDM must be reusable for multiple UAV flights to avoid wasting time remaking components.

7.1.4.4 Electronics Housing Requirements

The following flowchart outlines the requirements for the electronics body.



These high-level functional requirements outline the electronics body's objective to house and protect all flight and camera electronics during flight and landing of the UAV.

Requirement Number	Requirement	Method of Verification
S.3	The Belly shall house and protect all flight and camera electronics inside the launch vehicle and during flight.	Inspection The Belly will be inspected after the test launches and the electronics and belly will be inspected for damage and have their operations tested.
S.3.1	The Belly shall be designed to withstand the max acceleration impact.	Test The Belly will be put through an impact test. This test will be repeated 3 times and the data will then be analyzed to see if the electronics would be affected.
S.3.2	The Belly shall be designed to withstand a drop from 500 feet.	Test The electronics body will be dropped from 500 feet. This will be repeated 3 times and then the data collected from each trial will be analyzed and will determine if the electronics would be affected.
S.3.3	The Belly shall be designed with the lowest amount of excess housing space while retaining structural integrity and protection of the electronics.	Analysis After meeting the other requirements and their successful verification, using knowledge of the system's

		design an engineering judgement will be made.
S.3.4	The Belly shall be designed to allow protected routing for electronic connections to systems outside of the Belly.	<u>Test</u> The Belly will be examined after test launch for moisture buildup, damage, and connections. The electronics will be tested to be operational.

Derivation of Requirement S.3.1
The impact test would determine the electronics body's structural integrity and its ability to protect the electronics inside. To determine the with stance of the max velocity impact, the electronics body would have to survive with little to no damage sustained from the impact.

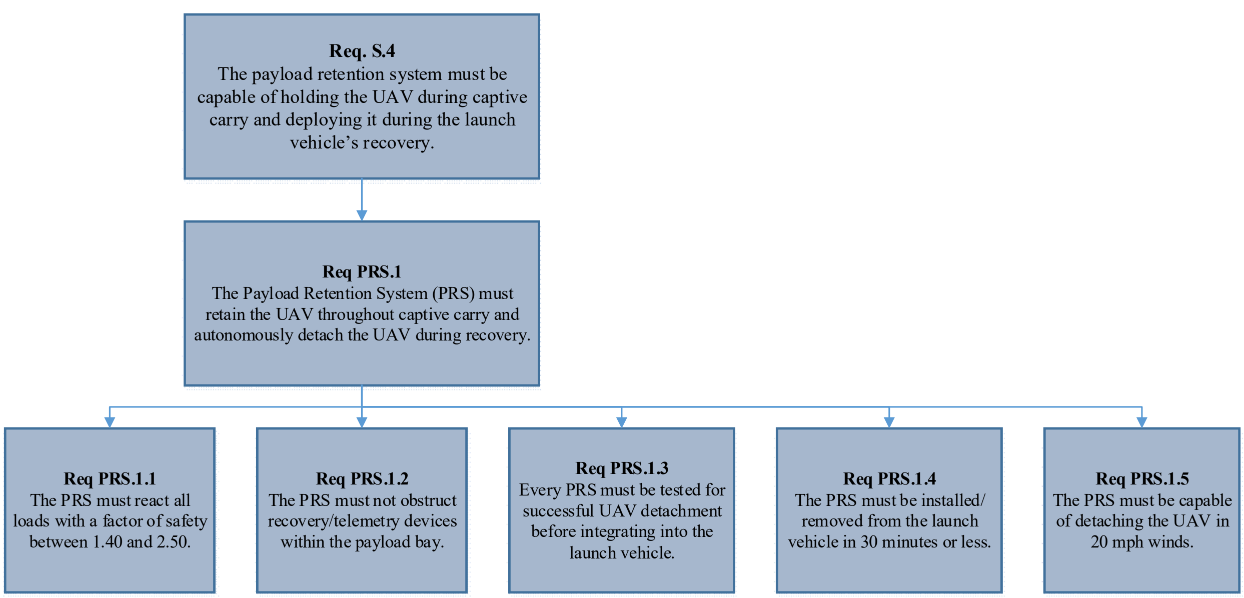
Derivation of Requirement S.3.2
The distance for the impact drop test is defined by the cruising altitude. To withstand a drop from 500 feet, the electronics body would have to survive with little to no damage sustained from the impact.

Derivation of Requirement S.3.3
To successfully meet the requirement of being designed with the lowest amount of excess housing space while still maintaining structural integrity and protection, the electronics body would have to meet the other requirements, and only then can it be analyzed for unnecessary excess space.

Derivation of Requirement S.3.4
To successfully meet the requirements of protecting the routing of the wiring, the wires being protected must be operational and undamaged after launch.

7.1.4.5 Payload Retainment Sub-System Requirements

The following flowchart outlines the mandatory PRS requirements that were set to ensure the payload is deployed at the correct altitude without sustaining any damage from captive carry.



Derivation of Requirement S.4
The payload must be deployed at a target altitude without sustaining damage during captive carry.

Derivation of Requirement PRS.1
The PRS must be capable of retaining the payload during captive carry, protect the payload from any hazards (launch thrust, apogee re-orientation, stage separation), and autonomously deploy the payload

Derivation of Requirement PRS.1.1
According to the NASA Structural design and test factors of safety for spaceflight hardware, 1.40 is the minimum required FOS against ultimate strength for a metallic fastener. To prevent oversizing fasteners and to conserve weight, a maximum FOS of 2.50 selected.

Derivation of Requirement PRS.1.2
The payload and PRS must not obstruct any telemetry or recovery hardware to ensure every sub-system operates successfully.

Derivation of Requirement PRS.1.3
The PRS must integrated into the payload bay and assembled with the payload to recreate a competition-flight payload assembly. The payload will be released using methods identical launch day operations to ensure the test is realistic to launch day conditions.

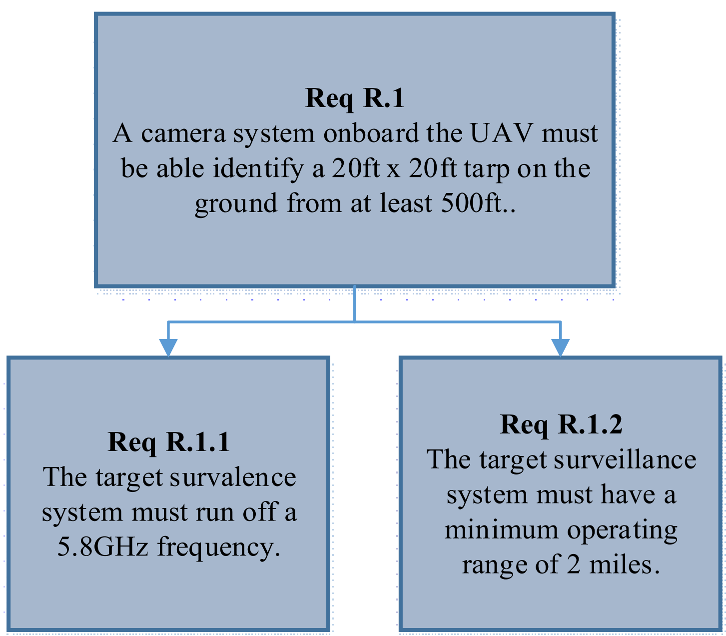
Derivation of Requirement PRS.1.4
A 30-minute installation time of the PRS prevents the payload from blocking the rest of the assembly of the launch vehicle and maximizes our probability of being ready to launch on the first day.

Derivation of Requirement PRS.1.5
A wind speed of 20mph was selected because it is the maximum wind speed NAR will allow for launches to take place. Higher wind speeds due to elevation were ignored since the UAV will be deploying at an altitude of 500ft.

Requirement Number	Requirement	Method of Verification
S.4	The payload retention system must be capable of holding the UAV during captive carry and deploying it during the launch vehicle's recovery.	<u>Test:</u> The UAV will be assembled into the PRS and payload bay and released to replicate a launch-day scenario.
PRS.1	The Payload Retainment System (PRS) must retain the UAV throughout captive carry and autonomously detach the UAV during the launch vehicles recovery.	<u>Test</u> Worst-case launch loads will be applied to verify the PRS's ability to retain the UAV. The PRS will receive input to detach the UAV.

PRS.1.1	The PRS must react all loads with a FOS between 1.40 and 2.50.	Analysis Stress calculations along with finite element models and will be utilized to verify final WDM design FOS.
PRS.1.2	The PRS must not obstruct recovery/telemetry devices within the payload bay	Test The fully-assembled payload bay will be tested for UAV retainment and detachment.
PRS.1.3	Every PRS must be tested for successful UAV detachment before integrating into the launch vehicle.	Test Each PRS that is built must be integrated into the payload bay and assembled with the payload. The payload must be released identically to launch day operations to ensure a realistic test.
PRS.1.4	The PRS must be installed/removed from the launch vehicle in 30 minutes or less.	Test A stopwatch will be used to measure the WDM integration time.
PRS.1.5	The PRS must be able to deploy the UAV safely in up to 20mph winds.	Test The PRS will be tested for detachment while loaded with a simulated 20 mph wind force.

7.1.4.6 Target Surveillance Sub-System Requirements



To ensure the highest quality of video footage, the gimbal on the UAV's camera must be able to clearly observe any objects on the ground. The Pan/Tilt System (PTS) will be capable of 180° of rotation in any direction while maintaining optimal conditions for recording in high definition. The flow chart below illustrates the PTS requirements.

Requirement Number	Requirement	Method of Verification <u>(Test, Inspection, Demonstration, or Analysis)</u>
S.5.1	The camera system must be able to identify a 20ft x 20ft tarp on the ground from at least 500ft.	<u>Test</u> A flight test on the target surveillance system will be done to ensure the camera can identify targets from a 500ft altitude. Full scale flight tests will also be conducted to further satisfy this requirement.
S.5.2	The target surveillance system must run off a 5.8ghz frequency.	<u>Inspection</u> The system will be tested before any test flights occur.
S.5.3	The target detection system must have a minimum operating range of 2 miles.	<u>Test</u> Range tests will be completed on the ground before any flight tests occur.

7.2 Flight Performance Subsystem

7.2.1 Airframe Design

7.2.1.1 Geometry Derivation

Definition of the UAV is centered around the geometry of the wings. This approach was taken since the wings are the most constrained component due to the airframe. Parameters of the payload such as thrust, weight, power density, and stability can be adjusted more easily than the wing geometry. Because of this the first analysis was conducted on the wings to determine a range of acceptable geometries.

A MATLAB tool was created to evaluate every wingspan and chord combination possible for our application. Next, parameters evaluated by the flight characteristics requirements such as the lift, velocity, aspect ratio, and airframe geometry constraints are plotted for every span and chord combination. To satisfy flight characteristics requirement S.1.3, a required velocity for the flow to have a Reynolds number of 200K was calculated for each span and chord combination. Required velocity was calculated using a rearranged version of Equation X shown below.

	$V = \frac{R_n \mu}{\rho C}$	

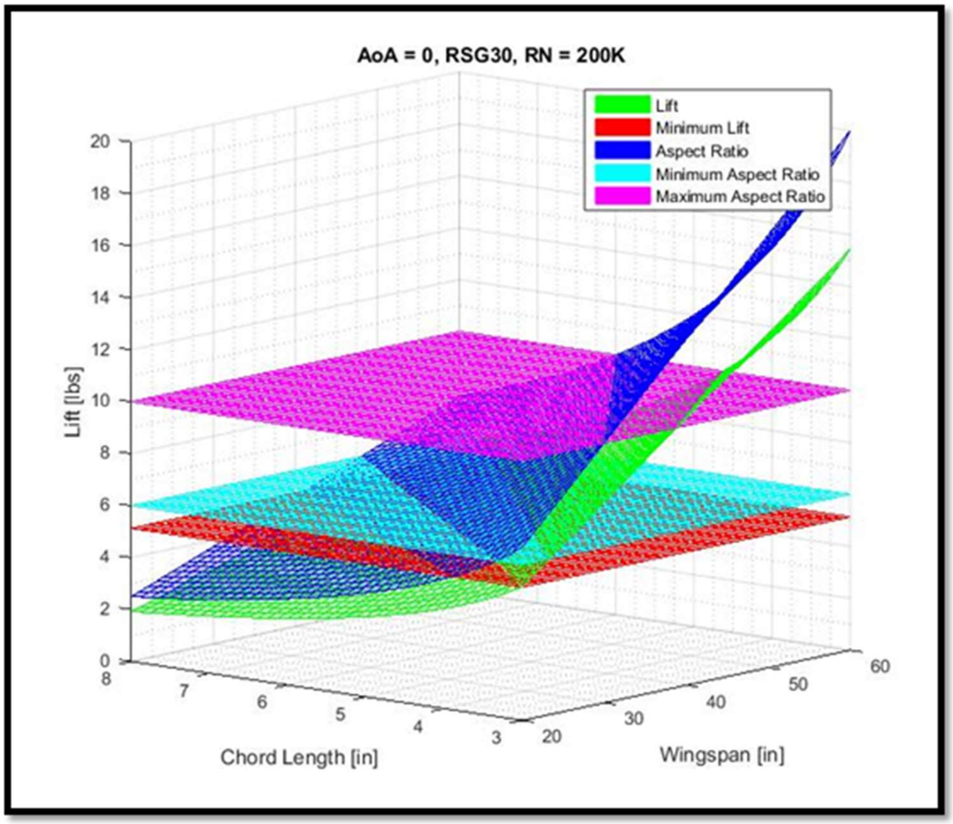
Here is the required velocity, is the Reynolds number (200K), is the density of air at the launch fields altitude, is the chord length of the main wing, and is the kinematic viscosity of air. This velocity is assumed to be the cruise velocity of the UAV and will be used in the rest of the analysis. Next the lift generated from the main wings was calculated for each span and chord combination using

	$L = \frac{1}{2}\rho V^2 C_L S$	

where is the projected area of the wing, and is the coefficient of lift at 0 degrees angle of attack. To ensure satisfaction of requirement S.1.5 the aspect ratio of each wing geometry was calculated with

	$AR = \frac{WS}{C}$	

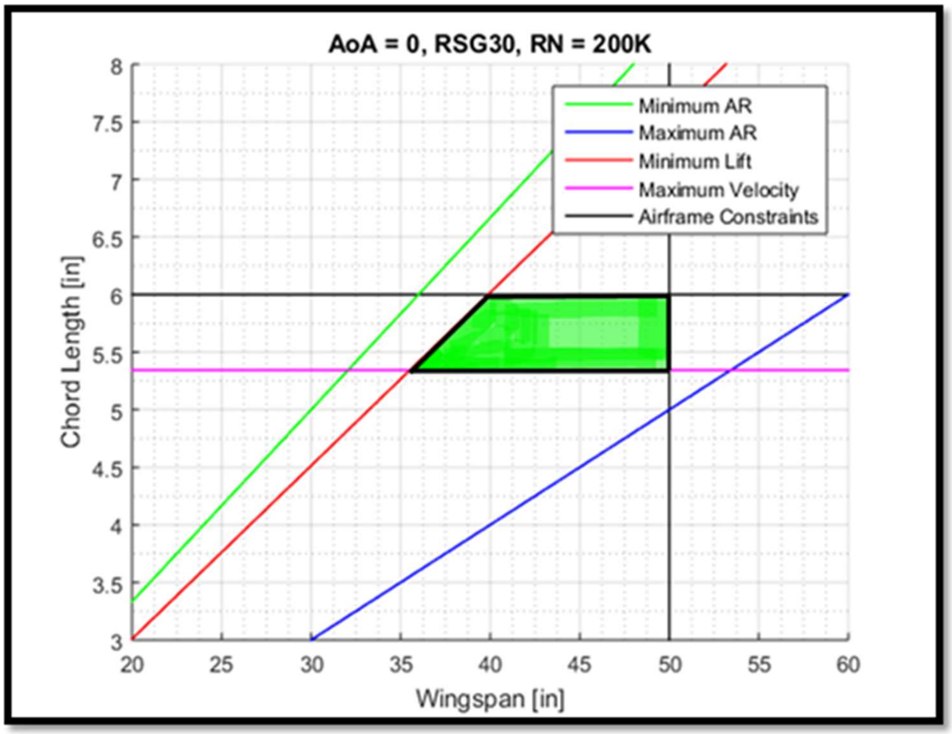
where is the wingspan. Once cruise velocity, lift and aspect ratio are determined at each point, a surface plot is created for all three parameters. This surface plot is shown below in Figure 17. Note that velocity is excluded in Figure 17 to prevent scaling.



Horizontal planes corresponding to each flight performance requirement is plotted in the same figure. A table outlining each requirement is shown below.

Constraint Parameter	Value	Units
Max. Aspect Ratio	10	Unitless
Min. Aspect Ratio	6	Unitless
Min. Lift	4.92	lbs
Max. Wingspan	50	in
Max. Chord	6	in
Max. Cruise velocity	50	mph

For example, a horizontal plane was plotted at 5.25 on the Z-axis. This represents the minimum lift requirement the wing geometry must create to sustain flight. Any combinations whose green surface plot is below the red requirement plane do not meet lift requirement S.1.2. The intersection line between each surface plot and its respective requirement plane is plotted as an X-Y projection. This projection can be seen below in



Lines corresponding to physical airframe constraints are added to the plot. Plotting these lines and the intersections of each requirement plane and its respective surface plot, an interior shape is formed. This shape contains all geometry combinations that meet the defined flight requirements. In Figure 18 this is illustrated as a green highlighted box. The center point of the bounding box was selected to be the wing's chord and span. The center point was selected because it can withstand the most geometry adjustment before falling out of requirement. Due to space constraints during the stowed configuration the chord has been reduced from 5.6in. to 5.40in.

With the wing dimensions established the total drag on the UAV was calculated using

$$C_D = \frac{CDA_0}{S} + C_d(C_L, R_n) + \frac{{C_L}^2}{\pi AR} \tag{10}$$

where is the drag coefficient of the entire UAV.   is the total area of the non-wing components, and is the drag coefficient listed in the NACA airfoil data. The table below displays the parameters used to calculate the total UAV drag. Replacing with in equation 2 results in a total vehicle drag of 2.5lbs at cruise.

Multiple plots were created to understand the UAV's takeoff velocity in various configurations.

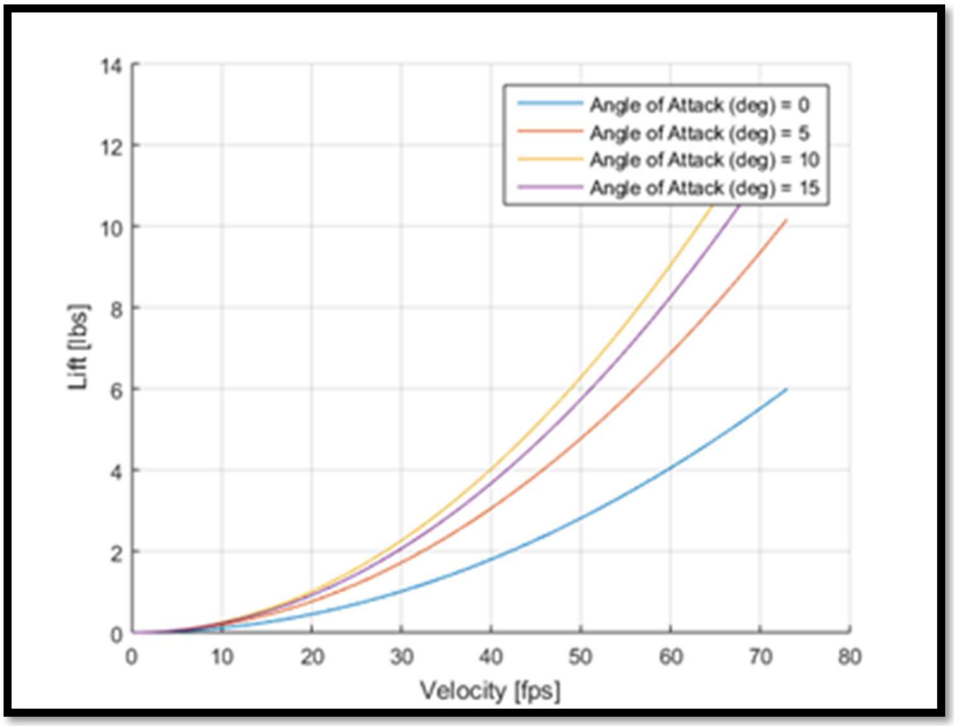


Figure 56. Lift versus velocity.

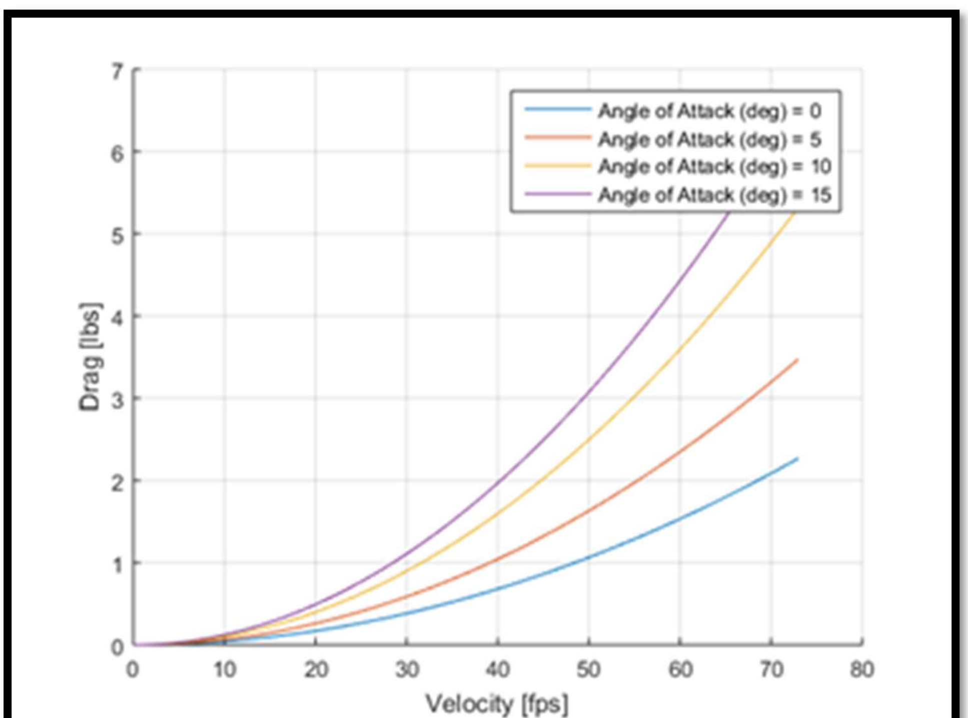


Figure 57. Drag versus velocity.

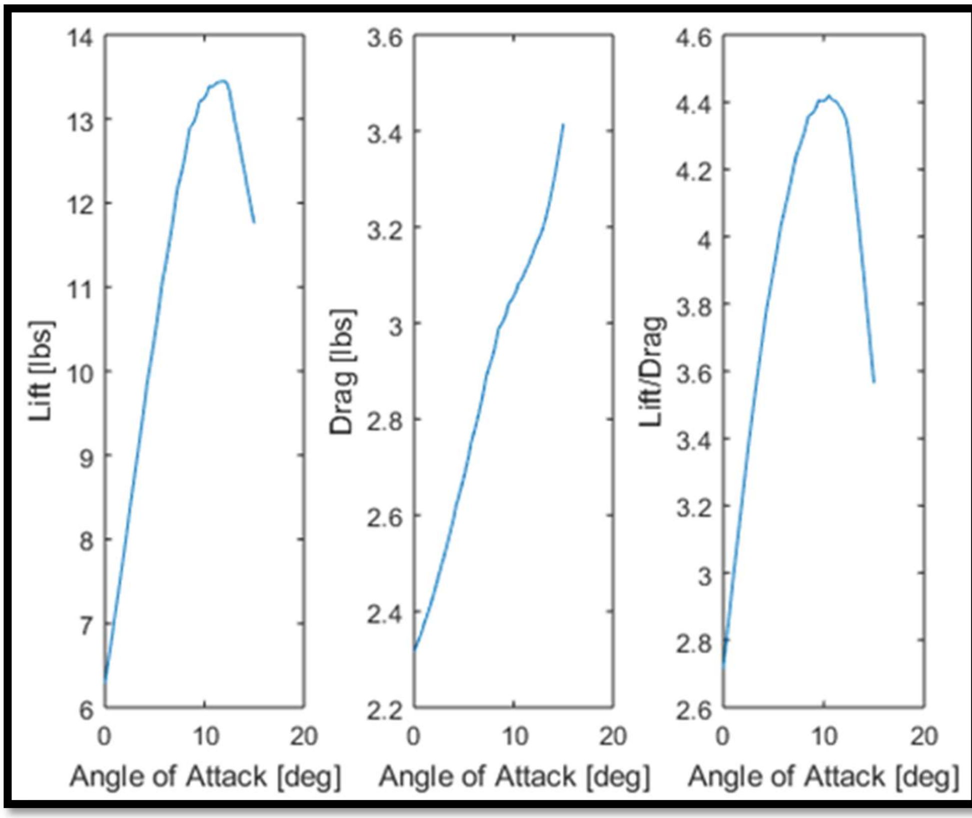


Figure 58. LIft, drag, and L/D ratio versus AoA.

From Figure 58 the most efficient angle of attack is determined to be approximately 10 degrees.

7.2.1.2 Static Margin

The static margin of an aircraft is defined as the distance between the aircraft's neutral point and center of gravity, normalized by the wing's mean aerodynamic chord. In his book "Model Aircraft Aerodynamics", Martin Simons calculates the neutral point as the distance behind the wing's leading edge that would make the aircraft's static stability zero. The neutral point h_n can be calculated using

$$h_n = h_0 + \eta_s V_s \left(\frac{a_s}{a_w}\right)\left(1 - \frac{\partial \varepsilon}{\partial \alpha}\right)$$

where h_0 is the aerodynamic center of the wing, η_s is the stabilizer efficiency, V_s is the horizontal stabilizer volume coefficient, a_s and a_w are the slopes of the lift coefficient vs angle of attack curves at cruise velocity for the horizontal stabilizer and wing respectively, and $\frac{\partial \varepsilon}{\partial \alpha}$ is the slope of the horizontal stabilizer downwash angle versus the angle of attack curve at cruise velocity.

The stabilizer efficiency can be conservatively estimated as 0.6, and the horizontal stabilizer downwash derivate is estimated as 0.415 using the average of typical limits of 0.33 to 0.5. The ratio of slopes of the lift coefficient curves are estimated at zero angle of attack using fit curves as shown in **Error! Reference source not found.** and **Error! Reference source not found.**.

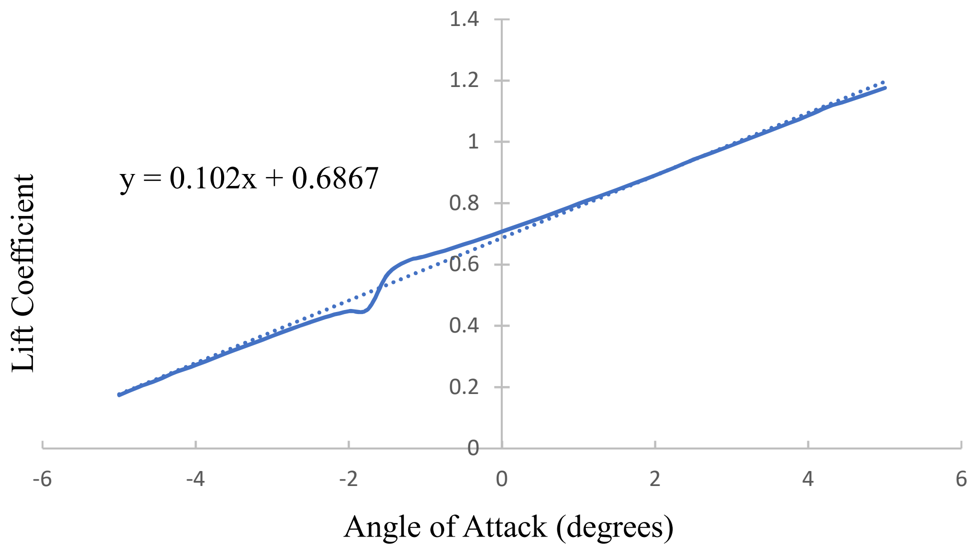


Figure 59. Wing lift coefficient vs. angle of attack.

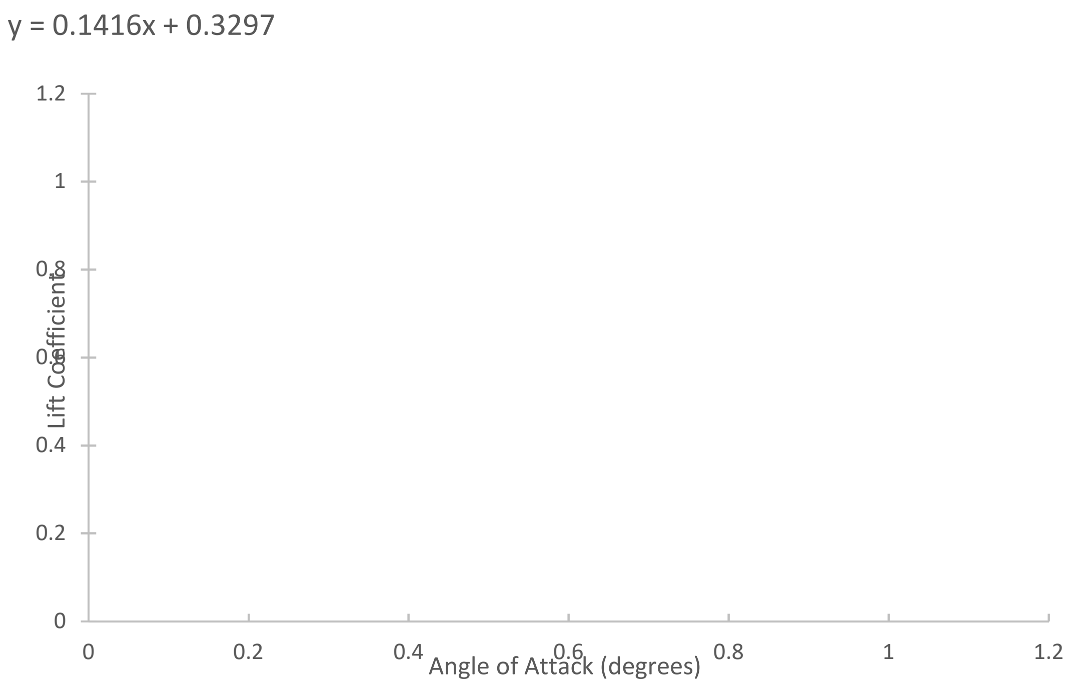


Figure 60. Horizontal stabilizer lift coefficient vs. angle of attack.

The horizontal stabilizer coefficient V_s is a number useful for sizing the horizontal stabilizer and ensuring sufficient pitch stability and can be defined as

$$V_h \equiv \frac{S_h l_h}{Sc}$$

where S_h is the area of the horizontal stabilizer, l_h is the moment arm of the horizontal stabilizer about the aircraft center of gravity, S is the wing area, and c is the wing mean aerodynamic chord. Typical horizontal stabilizer volume coefficients are in the range of 0.30 and 0.60 according to L. Pazmany's book, "Light Airplane Design".

Using these considerations, the aircraft's CG limits can be calculated as shown in **Error! Reference source not found.**.

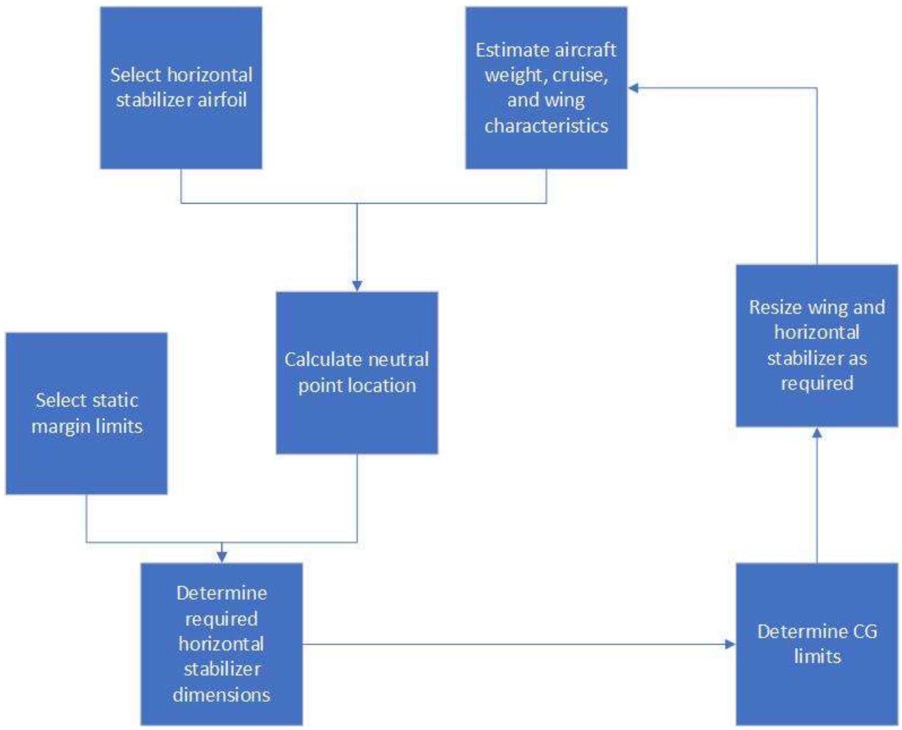


Figure 61. CG limit calculation process.

The final CG limit calculations for a static margin between 15% and 40% are shown in **Error! Reference source not found.**.

CG min	1.81	in aft from wing LE
CG max	0.46	in aft from wing LE

Table 23. CG limits.

The forward and aft CG limits correspond to 8.5% to 33.5% of the mean aerodynamic chord of the wing respectively. These agree with industry standard CG limits as shown in **Error! Reference source not found.**.

Aircraft	Estimated, % *mac*	Actual, % *mac*
B737 (forward/aft)	0.0/68.0	12.0/30.0
B767 (forward/aft)	-4.0/67.0	11.0/32.0
DC10 (forward/aft)	-7.0/46.0	8.0/18.0
B747 (forward/aft)	4.0/63.0	13.0/33.0

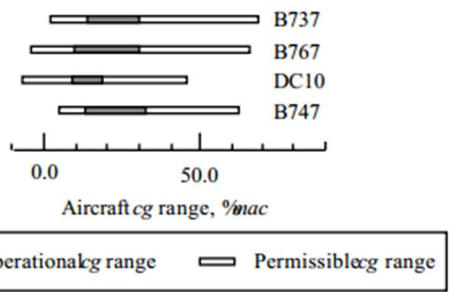


Figure 62. Standard mass produced aircraft CG limits.

7.2.1.3 Aileron Sizing

The ailerons of the aircraft were sized to be able to meet the desired roll rate with actuation of the control servos in a minimum time, while avoiding sudden accelerations. The process for sizing the ailerons is illustrated in **Error! Reference source not found.**.

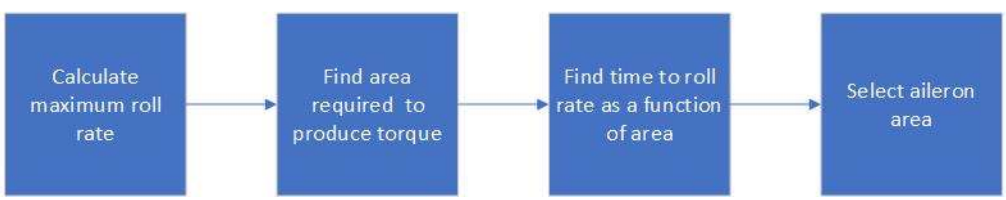


Figure 63. Aileron design flow chart.

In the book “Aircraft Preliminary Design Handbook”, the desired maximum roll rate p is be described as

$$\frac{pb}{2V} > k$$

where b is the wing span, V is the cruise speed, and k is a constant used to describe the desired acceleration of the aircraft. The constant k takes the value of 0.09 for aerobatic aircraft, and 0.07 for cargo/surveillance aircraft. The constant is taken to be 0.07 for our purposes.

Assuming steady flow, the force F produced by on a control surface of area A can be described using

$$F = \int_{cs} V\rho\left(\vec{V} \bullet \vec{n}\right) dA$$

where ρ is the air density, V is the velocity of the incoming air relative to the wing, n is the unit normal vector to the plane of the actuated control surface, and A is the area of the control surface.

The time to maximum roll rate as a function of net aileron area is shown in **Error! Reference source not found.**.

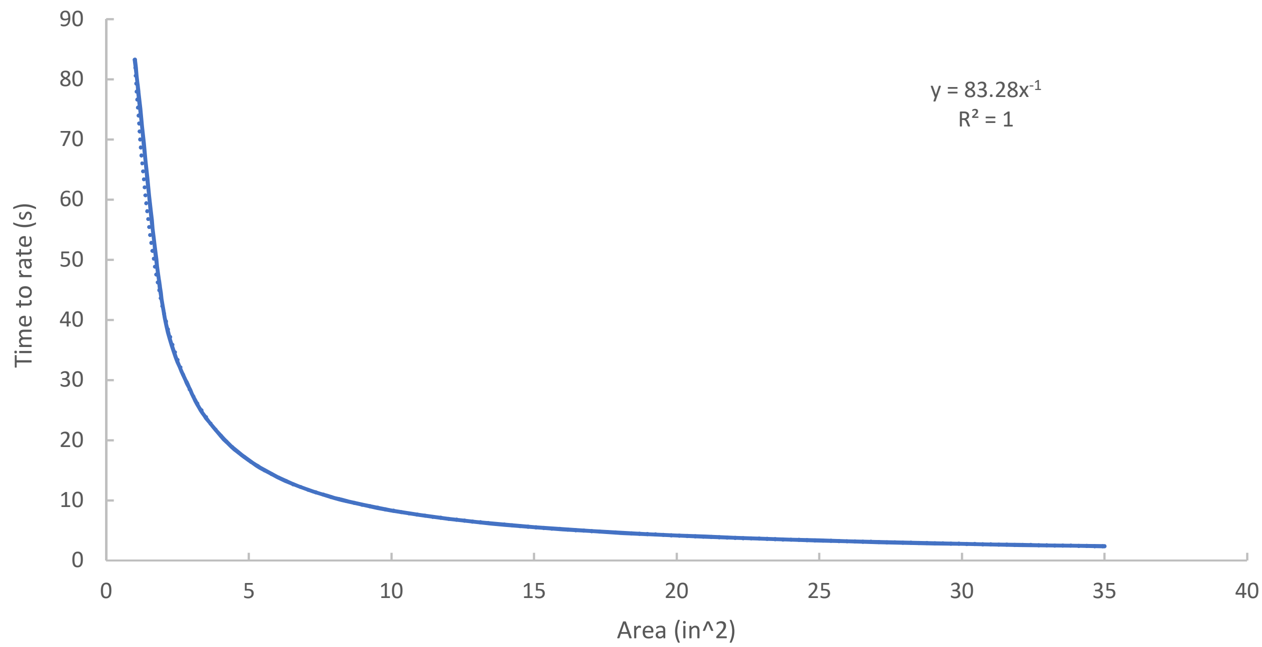


Figure 64. Time to maximum roll rate vs. total aileron area.

The function has a power law characteristic so diminishing returns are given after around 4 square inches of area. The final selection for the aileron size is shown in Table 24.

Time to rate (s)	Net aileron area (in^2)	Individual aileron area (in^2)
3.0	27.8	13.9

Table 24. Aileron design dimensions.

Similar to the horizontal stabilizer coefficient described earlier, a vertical stabilizer coefficient can be useful in sizing the vertical stabilizer. The formula used for calculating the vertical stabilizer volume coefficient V_v is

$$V_v \equiv \frac{S_v l_v}{Sb}$$

where S_v is the planform area of the vertical stabilizer, l_v is the moment arm of the vertical stabilizer, S is the wing planform area, and b is the wing span.

7.2.1.4 Control Surface Servo Sizing

Once aileron dimensions were selected, torque criteria could be determined for the servos. An online calculator from the Minnesota Scale RC Club was used to determine the required torque output of the servos. Entering the aileron parameters, a required torque of 4.161 oz-in is required by the servo. This calculation was only done for the aileron since it will require the greatest force to actuate.

Measurement	US Standard		Metric	
Max Speed	50	mph	80.465	kph
Control Surface Chord (front to back)	1.3	inches	33.02	mm
Control Surface Length	10	inches	254	mm
Control Surface Max Deflection (from center)	45	degrees		
Servo Max Deflection (from center)	45	degrees		
Calculate Servo Torque				
Servo Torque required	4.161	oz-in	2.994	Ncm
Control Deflection at max torque	45	degrees		

Figure 65. Servo torque calculator output.

In addition to providing the required torque for actuation, the servos were also required to have an analog feedback potentiometer built in. This feedback line allows for a pre-flight check to be conducted on the UAV before it is cutaway. This is discussed in detail in the flight controls section. The Batan B2122 metal gear micro servo was selected to control each of the UAV's four control surfaces. This servo is shown below in Figure 66.

Figure 66. Batan analog feedback servo.

A table outlining the specifications of the servo is shown below

Parameter	Value
Max. output torque (oz-in)	25
Degrees of rotation (deg)	120
Weight (grams)	15.81

Figure 67. Servo specifications.

The UAV uses four Adafruit metal gear servos that have a built-in potentiometer. The Power Distribution Board has eight slots for servos, we use the first four for servos, and the eighth one is the BEC connection that powers the servos. Through Mission Planner we labeled each servo as a control surface. Servo 1 is the left aileron, Servo 2 is the right aileron, Servo 3 is the elevator, and Servo 4 is the rudder. Through Mission Planner we were able to adjust the midpoint trims, which is the place where the servo goes to when there is no control input from the Taranis. We also adjusted the endpoint trims, which is the maximum and minimum amount the servo moves for each control surface. We had to also manually adjust the servo for each control surface so we could have the maximum amount of movement. Through this process we achieved the correct amount of degrees of movement for each control surface.

The servos are powered by the Venom Battery through the Power Distribution Board. The servos are also actuated by input from the Pixhawk received from its Ground Station subsystem. Actuations are performed both in response to manual and autonomous inputs. Autonomous inputs come from the flight path of the Pixhawk. Manual inputs come from the Ground Station subsystem.**

7.2.2 Flight Controls

A schematic of the UAV's electronics is shown below.

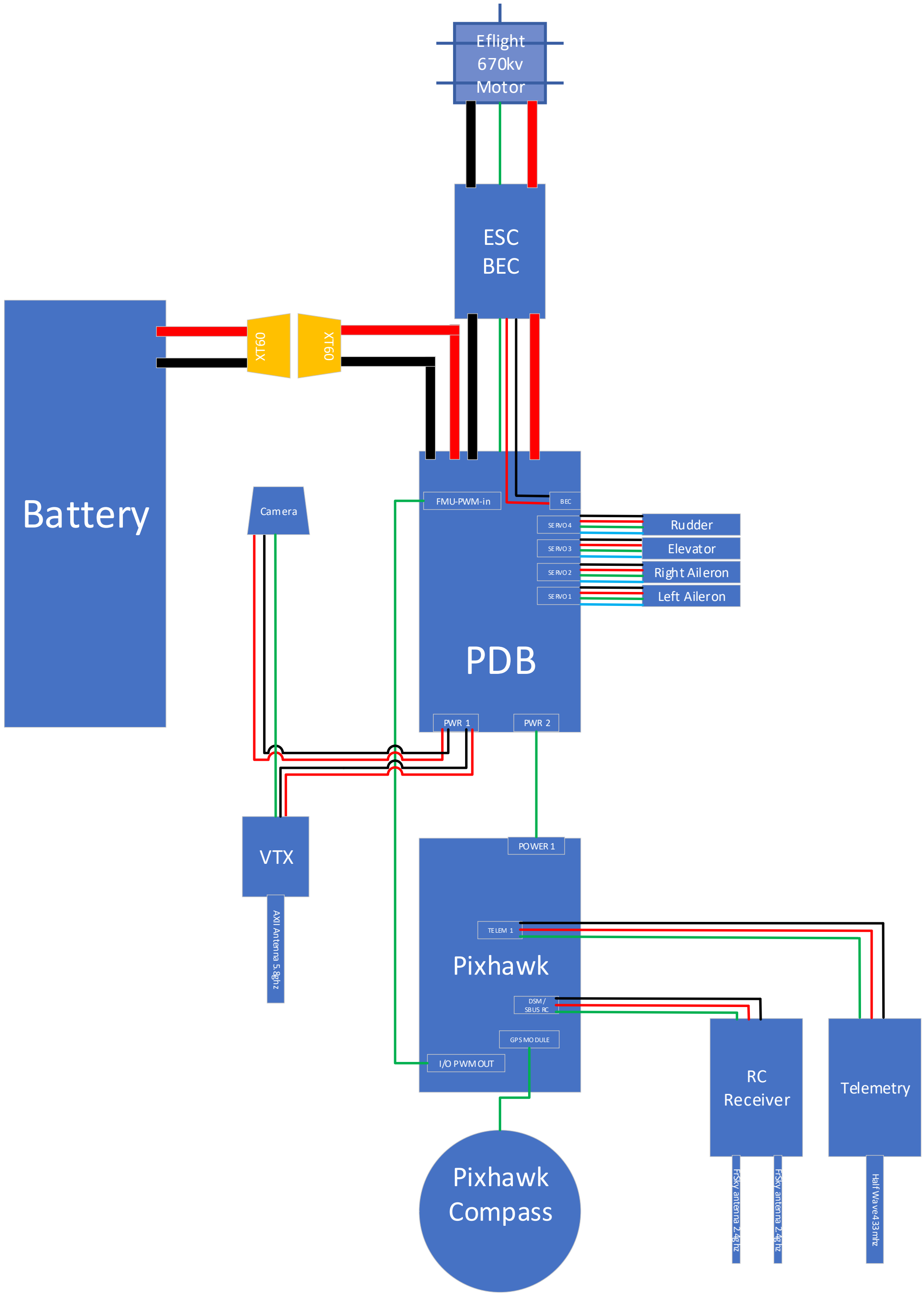


Figure 68. UAV controls schematic.

The flight controller we selected is the Holybro Pixhawk F4. In combination with the ardupilot firmware, we have the ability for manual flight, autonomous flight, and semi-autonomous flight. Autonomous and semi-autonomous flight is guided in part by the Pixhawk Compass, a GPS component.

Figure 69. Pixhawk flight controller.

The Power Distribution Board is part of the Pixhawk and solely powers all electronics on the UAV. After verification of compatibility, we connected all components of the UAV and successfully had them function through the ardupilot firmware. The Pixhawk and Power Distribution Board are connected to each other through the FMU-PWM-in on the Pixhawk, and the I/O PWM OUT on the Power Distribution Board.

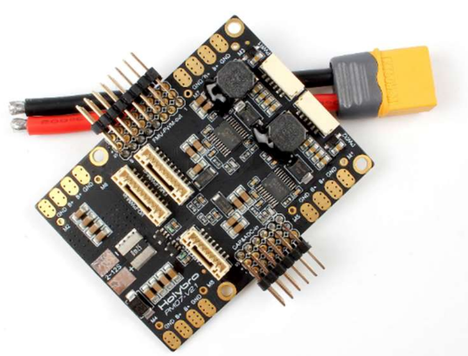


Figure 70. Power distruibution board.

7.2.2.1 Pre-Flight Checklist

Once deployment from the launch vehicle has occurred, the UAV will conduct an autonomous preflight check to verify that all control surfaces are capable of performing propulsive flight. This is a test of all Batan B2122 servos which each control an individual control surface. The test is to collect the maximum and minimum position voltage readings gathered from the potentiometer line of each servo. This data will represent the outermost positions of the servos and serve as indication of their full actuation. These values will be gathered using a Multimeter.

The lowest values at each boundary will establish a lower limit for the maximum and minimum servo positions, and the highest values at each boundary will establish the upper limit.

After the lower and upper limit of each threshold is established, these thresholds will be used to programmatically apply the actuation test to the Pre-flight Checklist. This will allow an onboard microcontroller to perform the same voltage tests over an analog line, and will confirm all control surfaces full actuation.

Figure 71 shows an overview of how the microcontroller will interact with the servos to perform the Servo Motor Actuation Test.

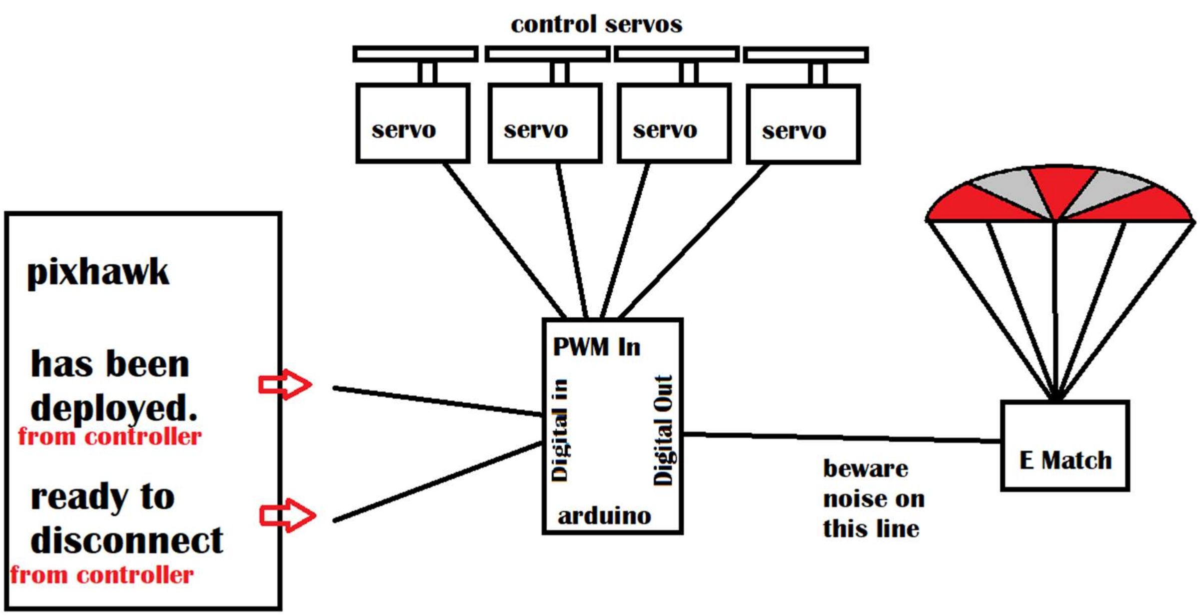


Figure 71. Pre flight check schematic.

The Flight Performance subsystem electrical controls power and operate the control surfaces and Propulsion motor.

[elaborate all of the following: mention arduino mini. Mini will be the microcontroller to perform the tests above & below, and after manual trigger from ground controller to pixhawk to mini, the mini will send the ematch (???) signal for the parachute disconnect]

7.2.2.1.1 Pre-flight Checklist

Checklist Number	Requirement to Check	Method of Verification (Testing, Demonstration, Inspection, Analysis)
1.	Confirm telemetry and video connection.	Visual confirmation of 90%+ Mission Planner connection and visual confirmation of video
2.	Begin pre-flight arduino polling	
3.	Actuate Servo Motors to full limits.	
4.	Verify success of arduino actuation verification.	
5.	Prop motor pulse test.	
6.	Transition to autonomous mode.	Verify prop spinup from auto?
7.	User confirms chute release.	

7.3 Propulsion Subsystem

7.3.1 Propulsion Overview

The propulsion sub-system contains three major components. The UAV motor, battery, and ESC. The location of these components and the propulsion sub-system is highlighted in blue below.

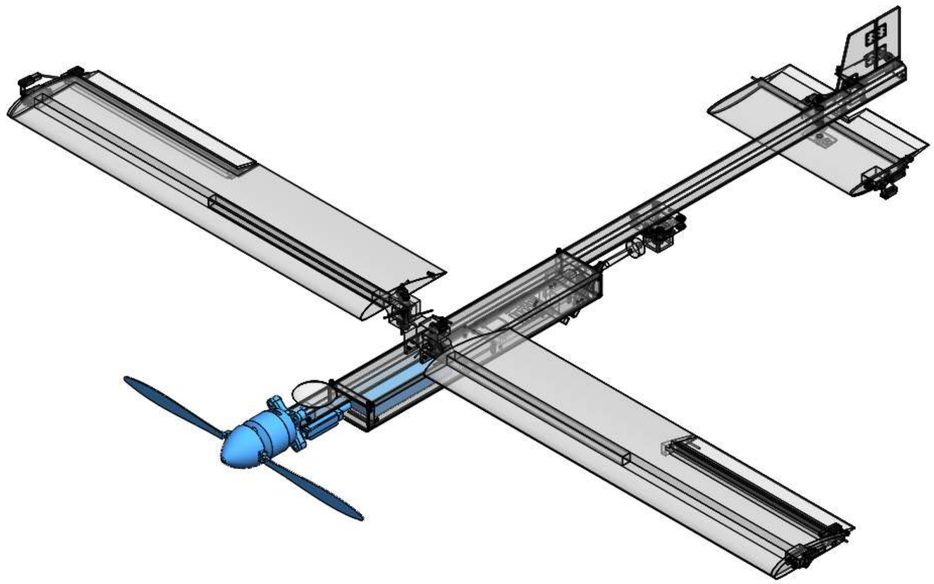


Figure 72. Propulsion Subsystem highlighted.

7.3.2 Propulsion Electrical Components

7.3.2.1 Propulsion Sizing Calculations

The UAV's motor was selected based on the requirement of 2:1 thrust/drag ratio, and it had to produce 80 watts of power per pound of UAV weight. The entire drag on the UAV is 2.2lbs. This calculation is explained in section XXXX. From the overall drag of the UAV, we can determine the required thrust from the UAV to be 5.5lbs. Table 25 splays data listed on HorizonHobby.com.

ESC	Castle Creations Phoenix 60
Propeller	ACP 13x8
Battery	Sanyo RC-3000HV 14-Cells
Flying weight w/ battery	6.7 [lb]
Current	44 [A]
Voltage	13.4 [V]
Power Output	590 [Watts]
Input Watts/Pound	89

Table 25. Eflite 46 test parameters.

The prototype UAV propulsion system is shown below.

7.3.2.1.1 60 A Pro SB Brushless ESC / E-flite 46 Power Brushless Outrunner Motor

The motor we are using is the E-flite Park 400 Brushless Outrunner 740kv motor. The reason we selected this motor instead of the original selection from PDR was because it was more available and from a more trusted brand. The motor and ESC combination still fit the requirements for the UAV, and are compatible with each other. We have not completed force tests on the motor yet, but through Mission Planner we were able to see that when the motor was at maximum throttle, it consistently pulled around 35 amps. This data let us know that there was no possibility of the ESC failing as it is rated for 60 amps. This also helped us re-size the battery further as we were able to calculate that we would have more than the required flight time with the current battery.

The ESC chosen is the E-flite 60A Pro Brushless ESC which also has a built in 5v BEC to power the servos. This BEC connection runs directly into the Power Distribution Board, connecting the ESC to the flight controller as well as powering the servos. The modifications made to the ESC included shortening the 14AWG connectors that go to the motor and to the Power Distribution Board. This was to allow the ESC to be moved up more to the front of the UAV to make room for the battery and other electronics in the back. The ESC is mounted just behind the motor and is attached via velcro and an RJX strap.

7.3.2.1.2 3600mAh 4s 30C Venom Battery

Figure 73: Venom battery.

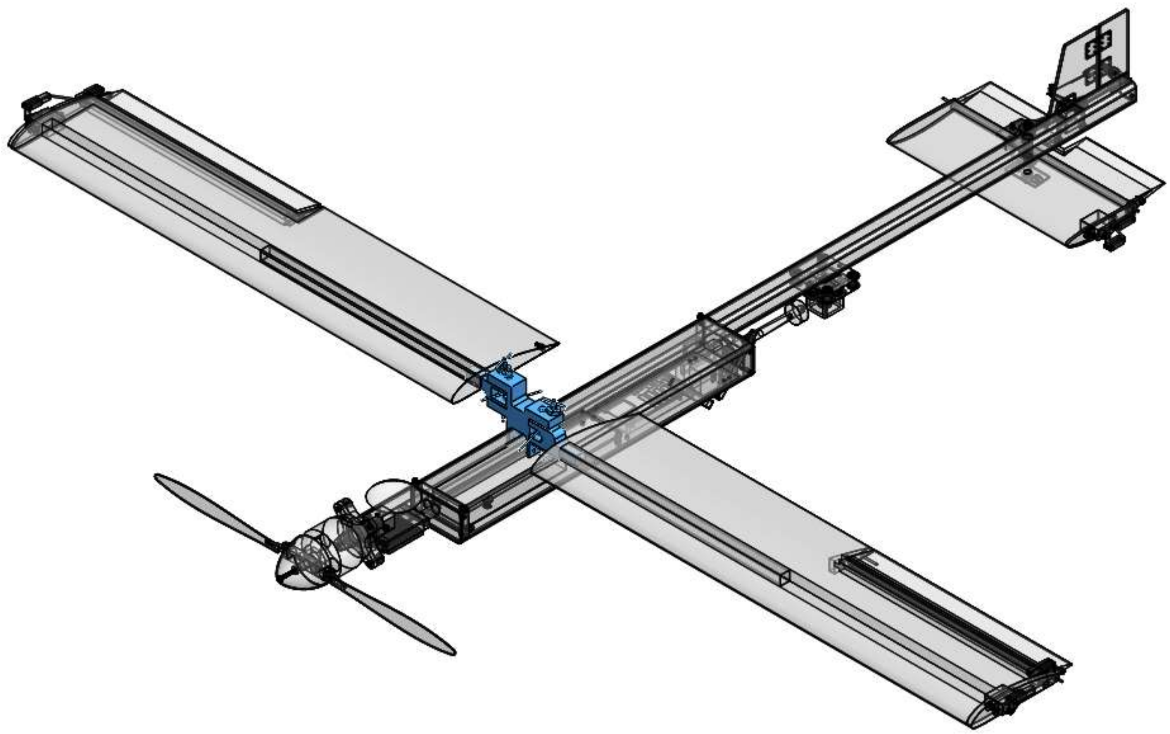
The power supply selected for the UAV and its onboard electronics is the Venom 3600mah 4s 30C LiPo battery. This battery is significantly smaller than what was last calculated. The previous calculation was based on a longer flight time, requiring a greater storage and weight. The battery is mounted underneath the ESC and attached via velcro and an RJX strap. The balance lead for the battery is tucked in-between the 14AWG wire to keep it from moving during flight. We found that the 30C discharge rate of the battery is sufficient to power all electronics onboard the UAV, and power the motor sufficiently for the full flight time.

7.3.3 Mechanical Components

[adapter]

7.4 Structures Subsystem

7.4.1 Wing Deployment Mechanism (WDM)



7.4.1.1 Wing Deployment Mechanism

The Wing Deployment Mechanism (WDM) is responsible for rotating the wings from the stowed configuration to the flight configuration and locking in place. Changes from PDR include: replacing the extension/compression spring combination with a torsion spring to reduce the mechanism complexity and ensure the mechanism is successful, as well as improving the clevis and bracket geometry to improve stress results. The WDM assembly is shown in **Error! Reference source not found.**.

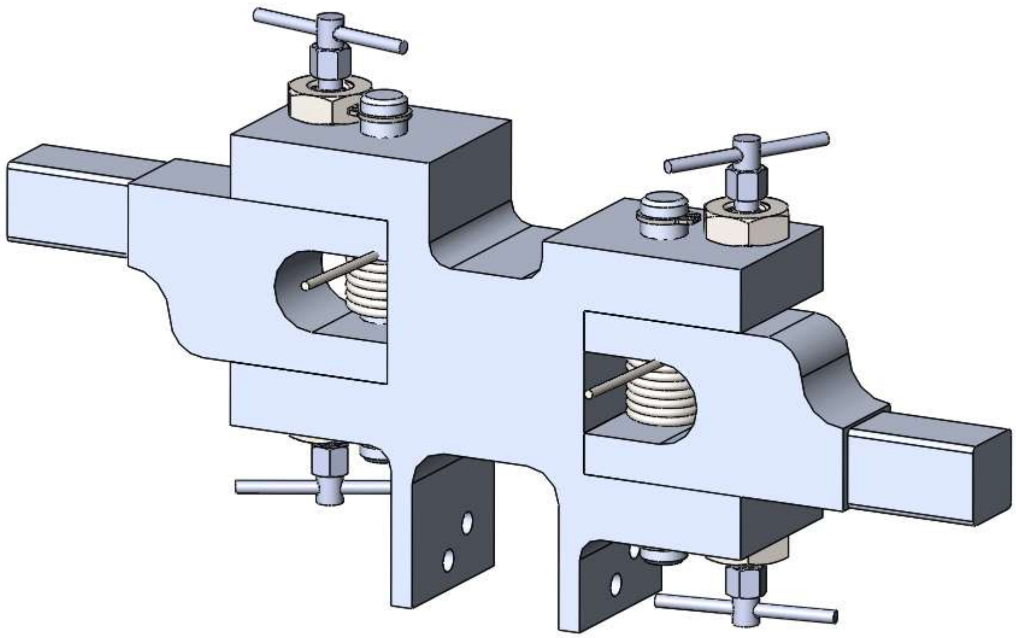


Figure 74: WDM Assembly

7.4.1.1.1 Clevises

The clevises are responsible for connecting the wings to the central bracket. Each clevis has a ¼" through hole to align the dowel pin to the central bracket. The wing spar will be pressed onto the clevis plug and riveted using two 1/8" blind rivets on the bottom face. The spring-loaded locking pins will engage the AFT face of each clevis to lock the wings in place during flight. The clevis will be machined out of 7075-T6 aluminum. The left and right wing clevises are shown in **Error! Reference source not found.** and **Error! Reference source not found.**.

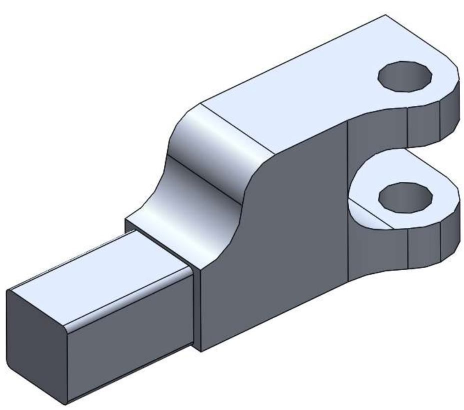


Figure 75: WDM left wing clevis

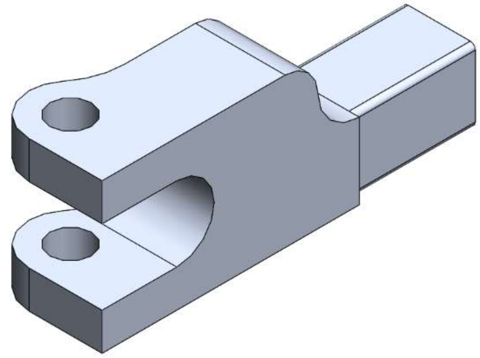


Figure 76: WDM right wing clevis

The clevises were analyzed for a dynamic loading scenario where the rapid deceleration of the payload due to the opening force of the parachute creates an impact load on the hard stop feature where the clevis strikes the central bracket. To calculate an equivalent static force that is representative of the dynamic impact force, the static deflection and tangential velocity of the wing must be calculated.

To calculate the static deflection, the wing and clevis were treated as a rigid body cantilever beam. The cantilever beam was fixed at the pin and a uniform distributed load was applied across the beam. The static deflection was calculated using

$$\delta_{st} = \frac{Wx^2}{24LEI}(x^2 + 6L^2 - 4Lx)$$

where W is the weight of the wing and clevis, x is the distance from the pin center to the CG of the wing (later referred to as h_{CG} below), L is the length of the entire wing and clevis, E is the elastic modulus of the carbon fiber spar, and I is the area moment of inertia of the spar.

The total energy in the wing was calculated using

$$PE_{spring} + PE_{wing} = KE_{wing}$$

where PE_{spring} is the potential energy in wing due to the compressed torsion spring, PE_{wing} is the potential energy in the wing due to the opening force acceleration, and KE_{wing} is the kinetic energy in the wing. Furthermore,

$$PE_{spring} = \frac{1}{2}k\theta^2$$

$$PE_{wing} = m_w a_p h_{CG}$$

$$KE_{wing} = \frac{1}{2}m_w {v_{CG}}^2$$

where k is the torsion spring constant, θ is the torsion spring angle of rotation, m_w is the mass of the wing and clevis, a_p is the acceleration of the plane due to the opening force, h_{CG} is the height (distance) of the CG from the pin center (previously referred to as x above), and v_{CG} is the tangential velocity of the wing at the CG. From these three equations, the wing tangential velocity at the CG (v_{CG}) can be calculated.

Using the static deflection and tangential velocity of the wing, the equivalent static force was calculated using

$$F_e = W\left(1 + \sqrt{1 + \frac{v^2}{g\delta_{st}}}\right)$$

where W is the weight of the wing and clevis, v is the tangential velocity of the wing at the CG, g is the acceleration due to gravity, and δ_{st} is the static deflection.

A finite element analysis (FEA) static study was setup to model the impact force acting on the clevis. The clevis upper face and impact faces were fixed with roller supports and the pin holes were fixed as hinge supports. A remote force equal to the equivalent static force was applied at the CG of the wing. The FEA stress results are shown in **Error! Reference source not found.**.

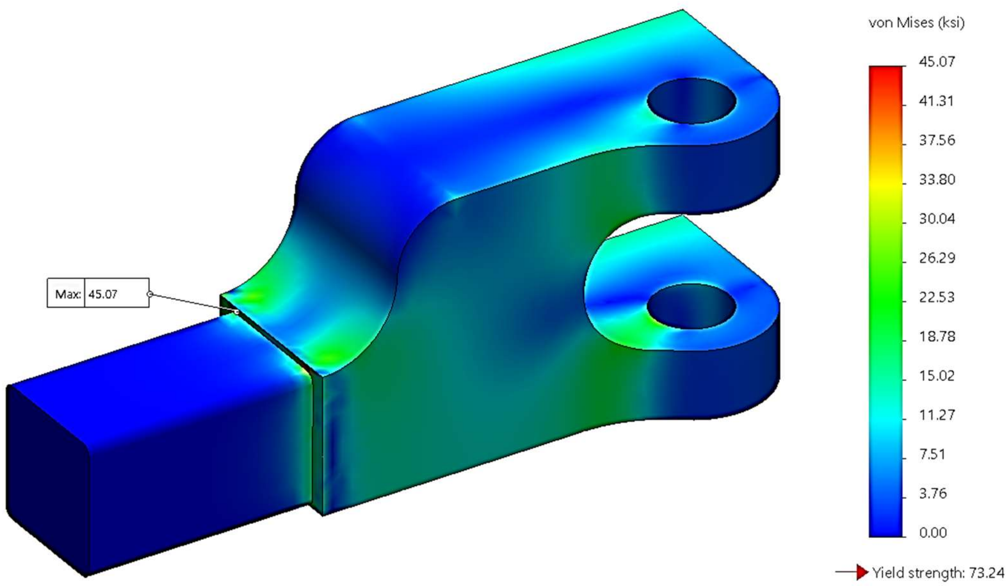


Figure 77: WDM clevis impact force FEA stress results.

The pin reaction forces were calculated using the FEA model and the impact force was calculated using the free body diagram in **Error! Reference source not found.**.

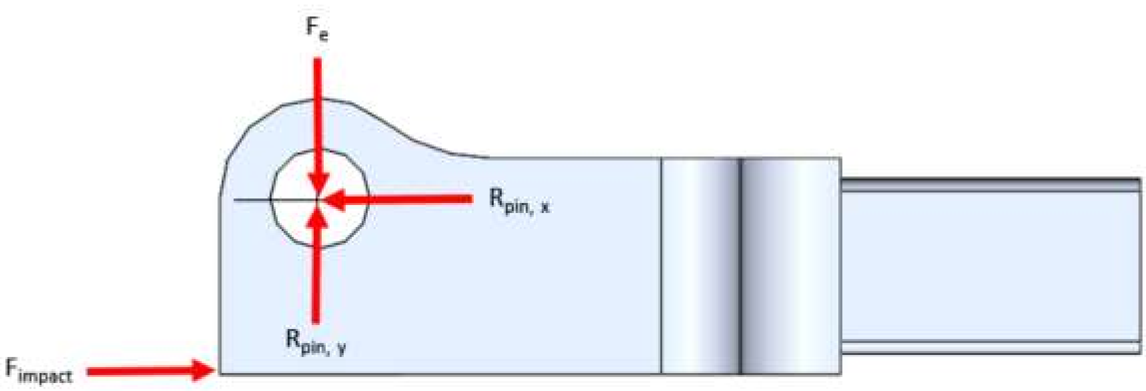


Figure 78: WDM clevis FBD.

To verify the FEA results, the pin shear-out stress, bearing stress, and normal stress was calculated using

$$\tau = \frac{R_{pin}}{A_{shear}}$$

$$\sigma_{brg} = \frac{R_{pin}}{A_{brg}}$$

$$\sigma_n = \frac{F_{impact}}{A_n}$$

where R_{pin} is the pin reaction force, F_{impact} is the impact force on the clevis/bracket, A_{shear} is the shear area, A_{brg} is the bearing area, A_n is the area normal to the equivalent force acting on the clevis.

The dynamic loading stress results are shown in **Error! Reference source not found.**.

FEA Stress Results		Pin Shear Out Stress		Pin Bearing Stress		Clevis Normal Stress	
F_e (lbf)	29.28	R_{pin} (lbf)	697.40	R_{pin} (lbf)	697.40	F_{impact} (lbf)	760.20
x_{CG} (in)	13.14	A_{shear} (in^2)	0.117	A_{brg} (in^2)	0.063	A (in^2)	0.094
σ_{max} (ksi)	45.07	τ (ksi)	5.98	σ_{brg} (ksi)	11.16	σ (ksi)	8.11
η_y	1.62	$\eta_{s, y}$	4.52	η_y	4.39	η_y	4.93

Table 26: WDM clevis dynamic loading stress results.

Based on the results of the FEA model and hand calculations, it is reasonable to conclude this clevis geometry meets the factor of safety requirements ($\eta_y > 1.4$). The geometry will be optimized for weight while maintaining the factor of safety requirement throughout the remainder of this season.

7.4.1.1.2 Central Bracket

The central bracket is responsible for connecting the wings to the payload's central spine. The bracket has two ¼" through holes to locate the dowel pins and two 5/16"-18 tapped through holes to install the threaded inserts for the locking pins. The bracket is riveted to the central spine using four 1/8" blind rivets located on the lower sides of the bracket. The bracket will be machined out of 7075-T6 aluminum. The central bracket is shown in **Error! Reference source not found.**.

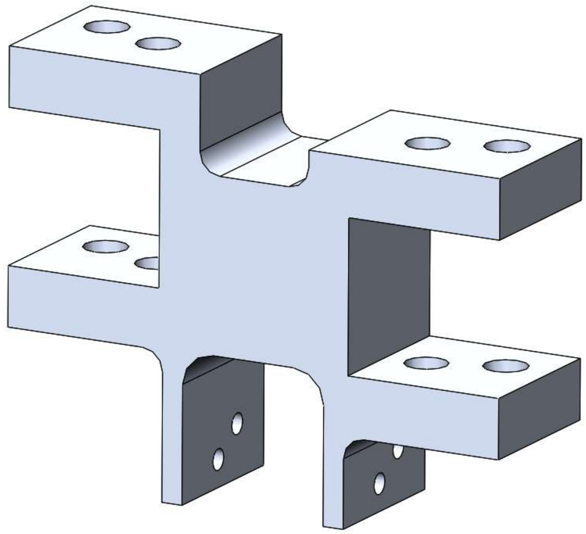


Figure 79: WDM Central Bracket

Two finite element analysis (FEA) models were created to analyze the bracket against the dynamic impact loading of the clevis and the static loading due to lift. During both models, the bracket was fixed at the base to represent the riveted connection between the bracket and carbon fiber spine and the dowel pin holes were treated as hinge supports. For the dynamic model, an impact force was applied across the impact area from the clevis and the pin reaction force was applied on the pin holes. The impact force was derived from the clevis analysis. The dynamic loading FEA results are shown in **Error! Reference source not found.**.

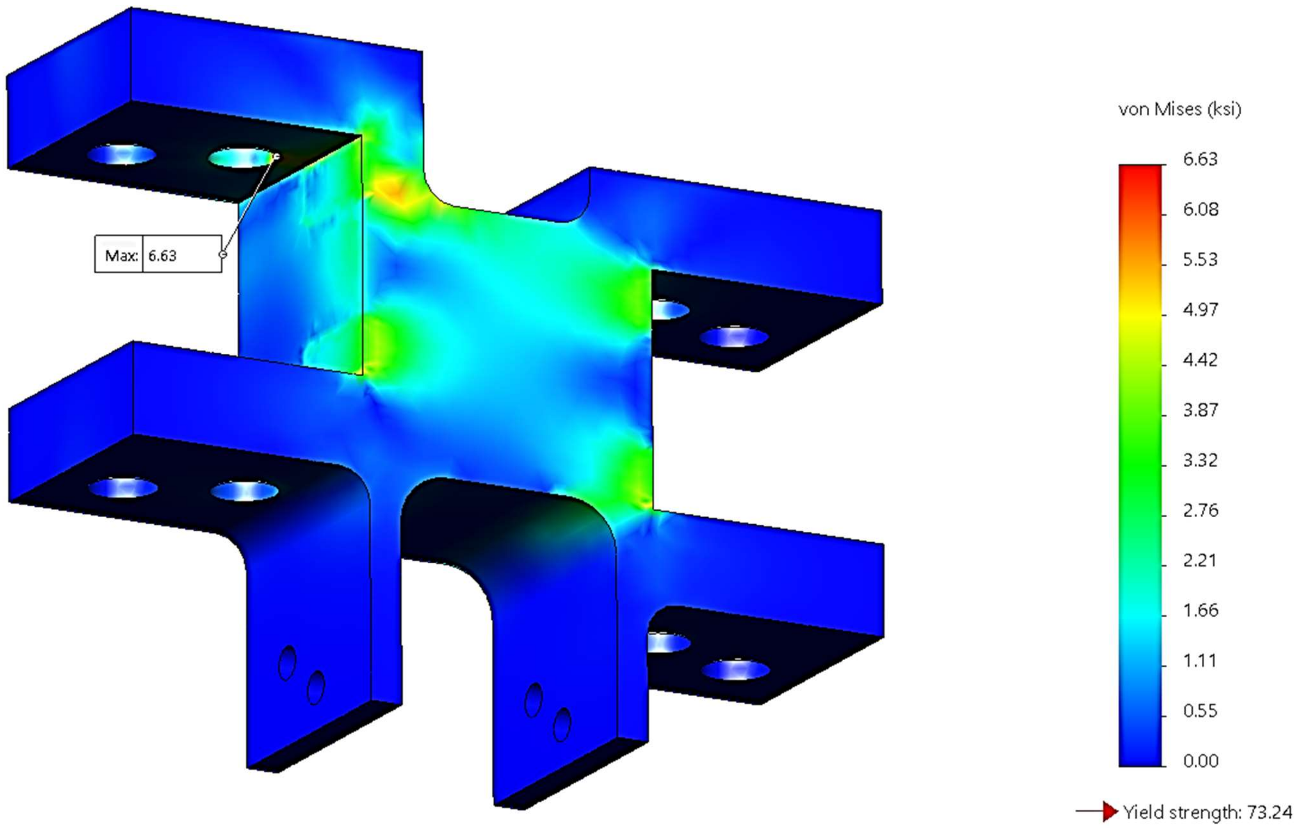


Figure 80: WDM central bracket dynamic impact loading FEA results.

The static loading model maintained the same fixtures as the dynamic model, however a remote load equal to the lift on each wing was applied to the dowel pin holes. The static model FEA results are shown in

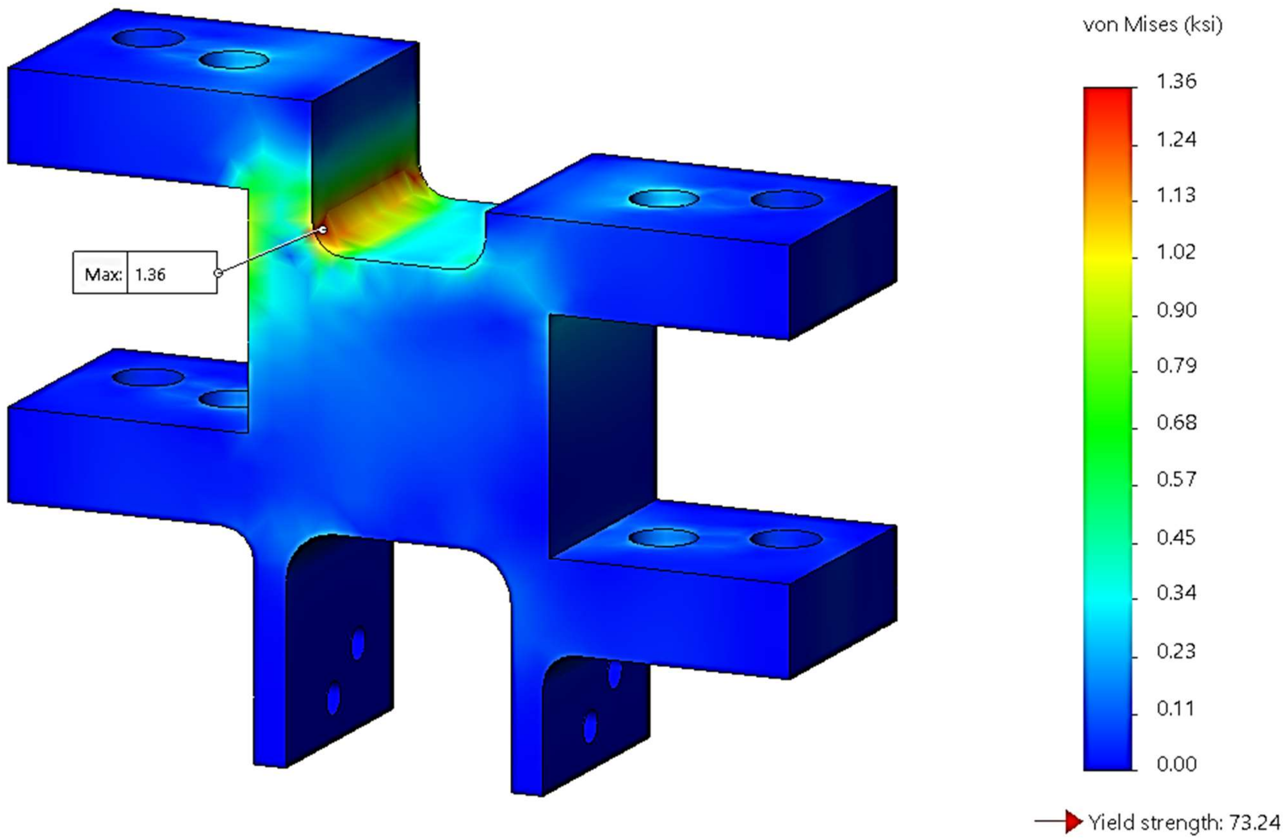


Figure 81: WDM central bracket static lift loading FEA results.

Error! Reference source not found. illustrates the max stress results for the dynamic and static FEA results.

	Dynamic impact loading	Static lift loading
σ_{max} (ksi)	6.63	1.36
η_y	11.01	53.68

Table 27: WDM central bracket stress results.

Since both loading scenarios result in factors of safety above the requirement, the central bracket geometry is considered acceptable. The geometry will continue to be optimized for weight while maintaining the factor of safety requirement throughout the remainder of the season.

7.4.1.1.3 Dowel Pins

The dowel pins are used to hinge the clevises to the central bracket and allow the wings to rotate. Two steel retaining rings will be pressed onto each pin to prevent the pin from slipping out of the bracket. The pins will be machined from a ¼” diameter 6061-T6 aluminum rod. The dowel pin is shown in **Error! Reference source not found.**.

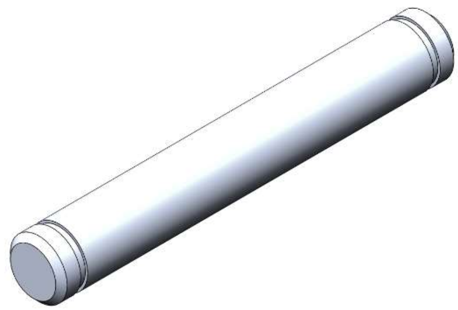


Figure 82: WDM dowel pin

The dowel pin was sized to withstand the bending and shear loading caused by the impact force.

7.4.1.1.4 External Retaining Rings

Four steel external retaining rings are used to secure the dowel pins in the central bracket and clevises. The retaining rings are pressed onto the notches once the pin is located on the bracket and clevis. The retaining rings can be purchased from the McMaster-Carr online catalog.

7.4.1.1.4.1 Music Wire Torsion Springs

The torsion springs are responsible for initiating the rotation of the wings from the stowed configuration to the flight configuration. Each torsion spring is housed on a dowel pin with one leg pressing against the central bracket and one leg pressing against the AFT face of the clevis. The torsion springs can be purchased from the McMaster-Carr online catalog.

7.4.1.1.4.2 Corrosion-Resistant Retractable Spring-Loaded Locking Pins

Four spring-loaded locking pins are responsible for locking the wings in the flight configuration. Two locking pins are located on the top face of the central bracket and two locking pins are located on the bottom face of the central bracket. The wing clevises were designed to compress the locking pins until the wings rotate to the flight configuration. When the wings are fully rotated, the locking pins will extend outward and engage with the AFT face of the clevises, preventing the wings from rotating inwards. These specific locking pins were selected for their small profile and simple locking mechanism. The locking pins have four threads with thread locker to prevent the pins from backing out during flight. The locking pins can be purchased from the McMaster-Carr online catalog.

7.4.1.1.4.3 Stainless Steel Key Locking Threaded Insert

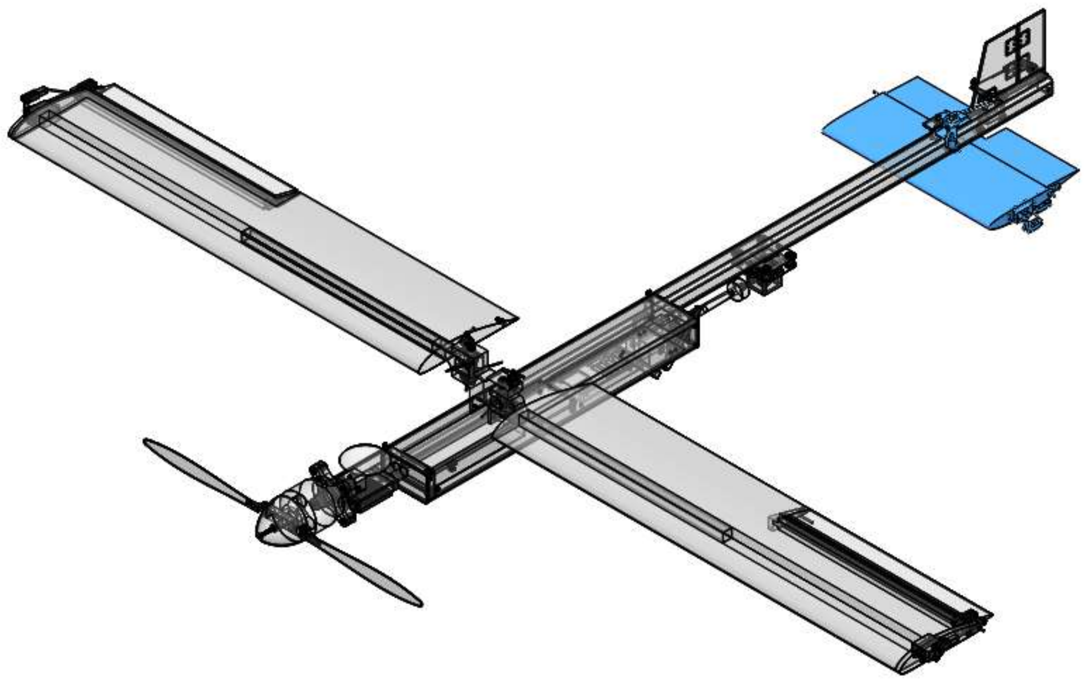
Four threaded inserts are used to fasten the locking pins to the central bracket. Threaded inserts were selected to avoid shearing the central bracket and ensure the locking pins are stay fastened during flight. The threaded inserts can be purchased from the McMaster-Carr online catalog.

7.4.1.1.4.4 Grade 5 Thin Hex Nuts

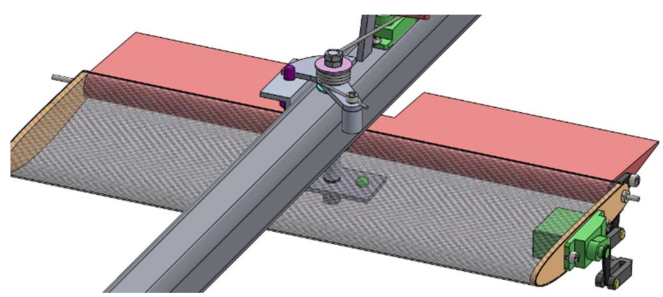
Four Grade 5 steel thin hex nuts are used to fasten the locking pins to the central bracket and prevent the pins from backing out. Thin nuts were selected over regular nuts for their small profile; regular hex nuts could not be used with these specific locking pins. The thin hex nuts can be purchased from the McMaster-Carr online catalog.

7.4.2 Stabilizer Deployment Mechanism (SDM)

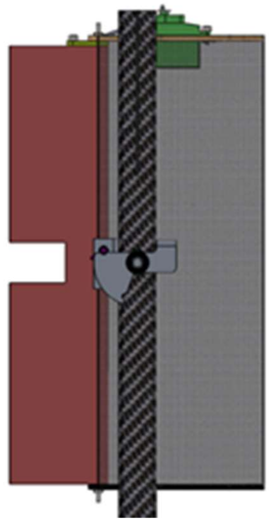
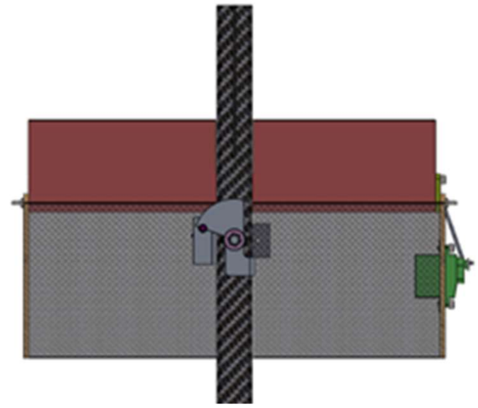
SDM Design Overview



The function of the SDM is to stow the horizontal stabilizer inside the launch vehicles airframe. The SDM rotates the entire stabilizer, elevator, and elevator mechanism from the stowed position to the flight position. Once in the deployed position, a 316 SS retractable spring plunger will lock behind the mechanism to secure the stabilizer into place. Identical plungers are used on the WDM.



Shown below are the stowed and flight configuration of the SDM.

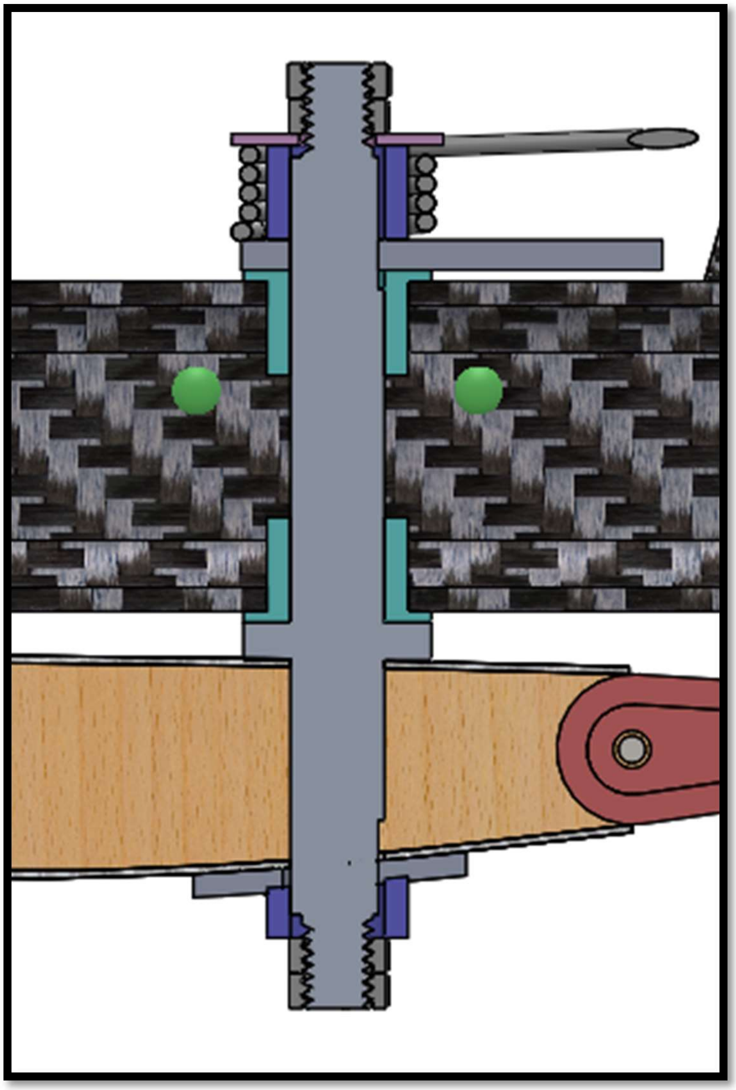
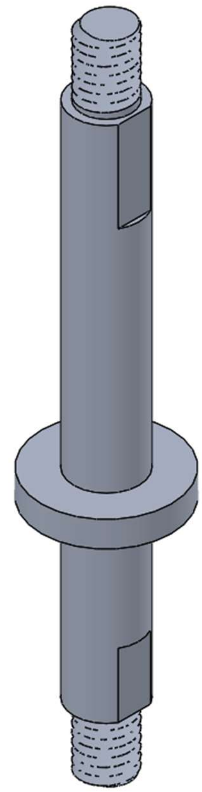
 

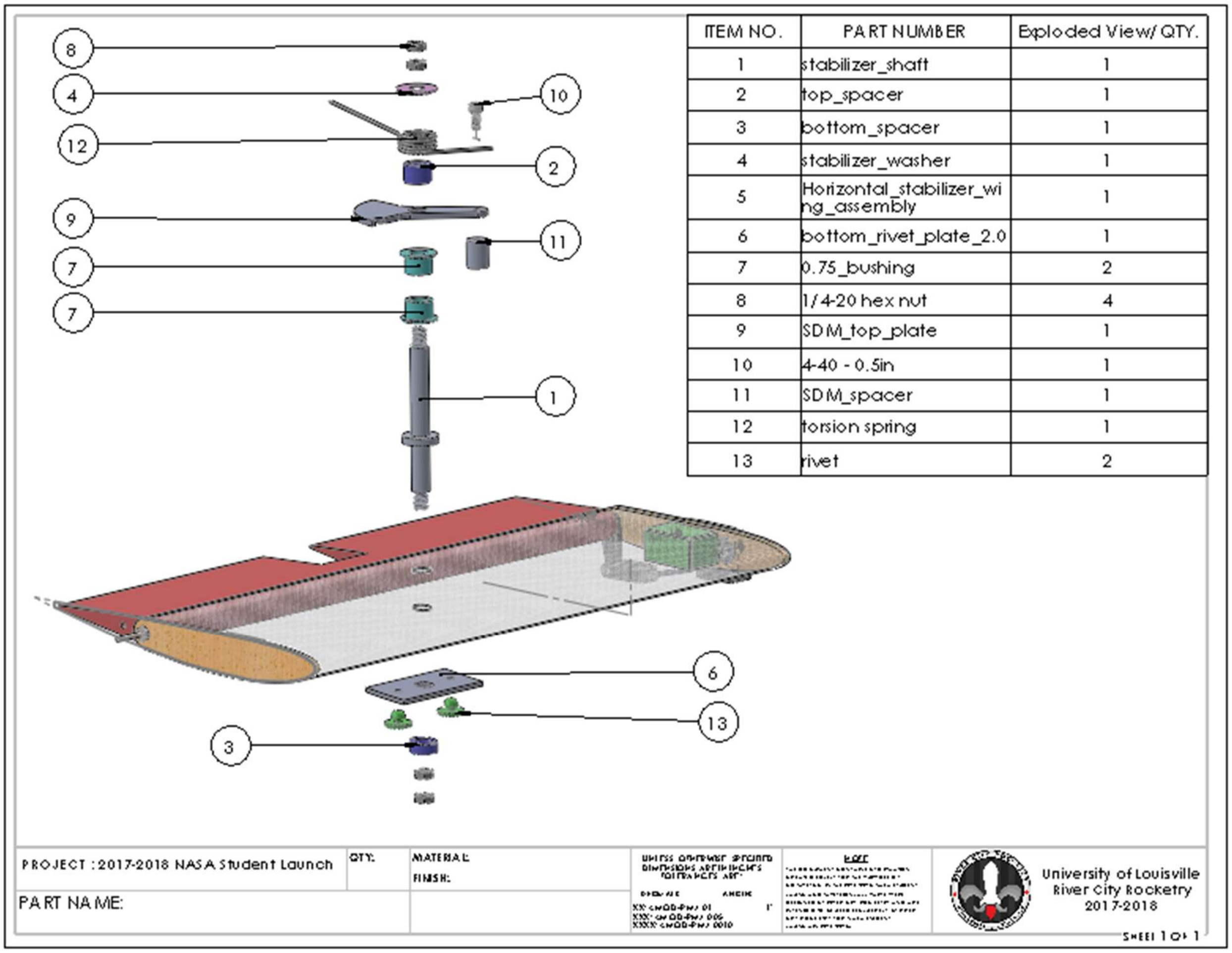
SDM Design

The SDM was analyzed against two scenarios. The first was to be able to deploy the stabilizer assuming all drag is acting on one side of the stabilizer. The second requirement was for the SDM to deploy in 0.25 seconds.

The mechanism is driven by a 270 degree torsion, left hand torsion spring.

Parameter	Value	Units
Stowed torque	0.5	in-lb
Flight torque	0.33	in-lb
Shaft diameter	0.21	in



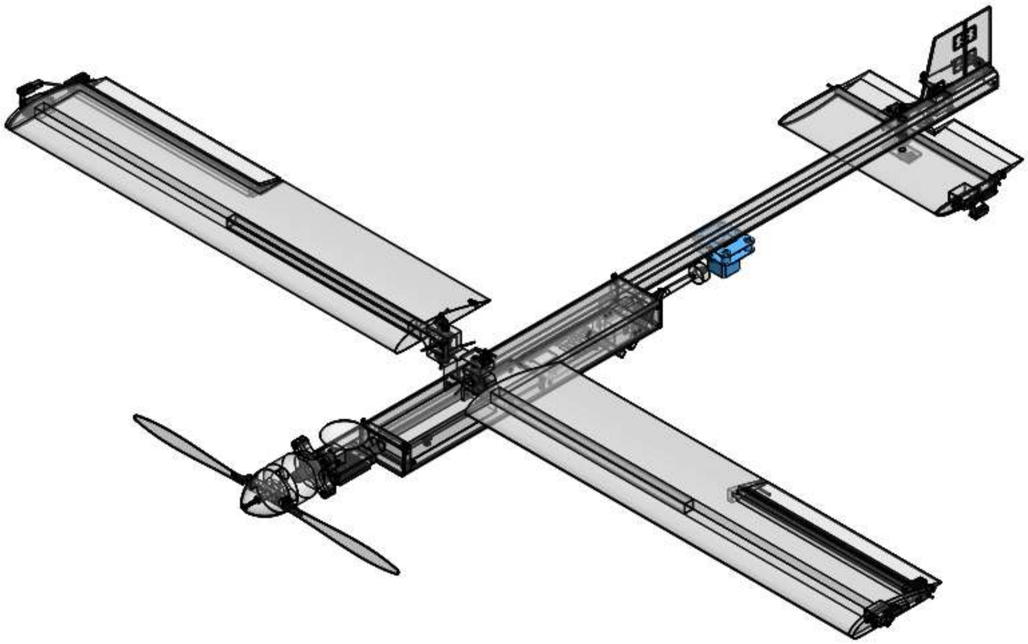
ITEM NO.	PART NUMBER	Exploded View/ QTY.
1	stabilizer_shaft	1
2	top_spacer	1
3	bottom_spacer	1
4	stabilizer_washer	1
5	Horizontal_stabilizer_wing_assembly	1
6	bottom_rivet_plate_2.0	1
7	0.75_bushing	2
8	1/4-20 hex nut	4
9	SDM_top_plate	1
10	4-40 - 0.5in	1
11	SDM_spacer	1
12	torsion spring	1
13	rivet	2

In order to stow the horizontal stabilizer inside the launch vehicles airframe

7.4.3 Controls Surface Mechanisms

7.5 Payload Retainment Subsystem

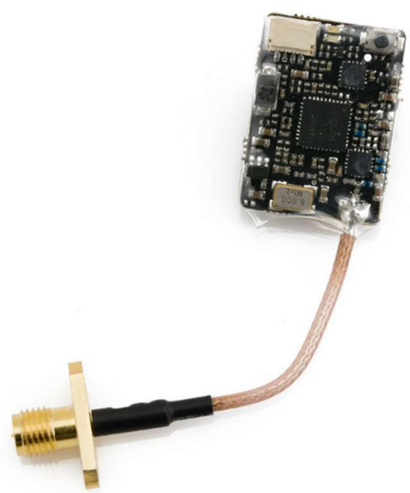
7.6 Target Surveillance Subsystem



7.6.1 Target Detection Electrical Components

The electrical components of the Target Detection Subsystem receive power from the 3600mAh Venom Battery (S 7.2.2.5 **INSERT REFERENCE**). The components operate in conjunction with the Telemetry Ground Station (S 7.7.1.3 INSERT) via a 5.8GHz broadcast from a video transmitter. This video transmitter is on-board the UAV electronics.

7.6.1.1 Micro Runcam Swift 2 Camera / Unify Pro Video Transmitter 5.8GHz



The Target Detection Subsystem includes the Micro Runcam Swift 2, the TBS Unify Pro v3 Video Transmitter (VTX), and the Lumenier AXII antenna.

The Runcam camera lens is 2.1mm thick, with 150° of view. The camera was tested by demonstration on the prototype UAV. This required mounting directly to the 3D printed WDM central bracket (S 7.XXX). The camera's signal wire (which wire?) sends data directly to the VTX, and has no physical connection to the Pixhawk controller nor any other RX or TX components.

The VTX transmits data over the AXII antenna via a 5.8GHz signal. The VTX is powered from the same connection as the camera (powered through connection to camera board?). It has an SMA pigtail connector that connects to the AXII antenna. *The antenna is mounted outside of the electronics box, but the VTX is not mounted and floats freely inside the electronics box (will this be the electronic sled design in the final build?).* This is because it transmits at 800mW and can overheat (mount with a heatsink to a physical sled?) if it is covered or mounted. This also allows cable slack in the event of tension on the SMA pigtail or antenna.

The AXII antenna allows transmission from the VTX to the Ground

7.7 Telemetry Subsystem

7.7.1 Telemetry Electrical Components

7.7.1.1 Onboard Telemetry Systems

Figure 83: HolyBro 100 mW Transceiver – 433 MHz

The telemetry onboard the UAV is the Holybro 100mW transceiver, with an identical one of these transceivers on the ground station. Both units transmit at 433mhz frequency and have a max output power of 100mW. Payload telemetry frequency falls within rules laid out in FCC Part 15.231 for license-free usage of short-burst 433 MHz frequencies. The transceivers can easily communicate with each other and give us control of the UAV's autonomous flight functions, parameter tree, connection with the Taranis radio, servo configuration, and all other essential functions the UAV needs.

In the PDR document no antennas were selected for the ground station or UAV, so we later selected a 10dbi Yagi antennas for the ground station, and a flexible halfway antenna for the UAV. The antenna on the UAV is mounted directly out of the electronics box and is not touching other antennas as this would cause interference and possible connection loss with Mission Planner.Flight Performance and Propulsion subsystems

7.7.1.2 L9R RC Receiver 2.4GHz

The FrSky receiver selected is the L9R long range receiver. This receiver runs on a 2.4ghz frequency and has less than 20ms response time, allowing us to manually fly the UAV. At the bottom of the receiver is the sbus connection, this connection goes directly to the Pixhawk F4 which has an sbus slot specifically made for a receiver of this type. In Mission Planner we were able to set the trims for the QX7 Taranis to get the most accurate control from it. The receiver has two antennas running out the back of the module, these were mounted at 45 degree angles to gives us the best polarization no matter what orientation the UAV is flying in. These antennas were mounted away from all other antennas as close proximity with one could interrupt the signal and cause a failsafe.

7.7.1.3 Ground Station Equipment

The purpose of connecting payload telemetry to the ground station is to receive real-time telemetry data while also sending flight controls and commands to the payload. The ground control software that is being used by the payload is Mission Planner. Because of the software's technical support and documentation for the payload's flight controller, the Pixhawk v4, configuring the radio telemetry modules is very simple; plug one of the telemetry transceivers (coupled with an antenna) into the flight controllers port specifically configured for radio telemetry modules and connection will be made automatically. The 2nd payload telemetry transceiver is connected to the GCOM via a USB connection. See **Error! Reference source not found.** and **Error! Reference source not found.** for an example of Mission Planner's Graphical User Interface (GUI) layout.

Figure 84: Mission Planner GUI layout

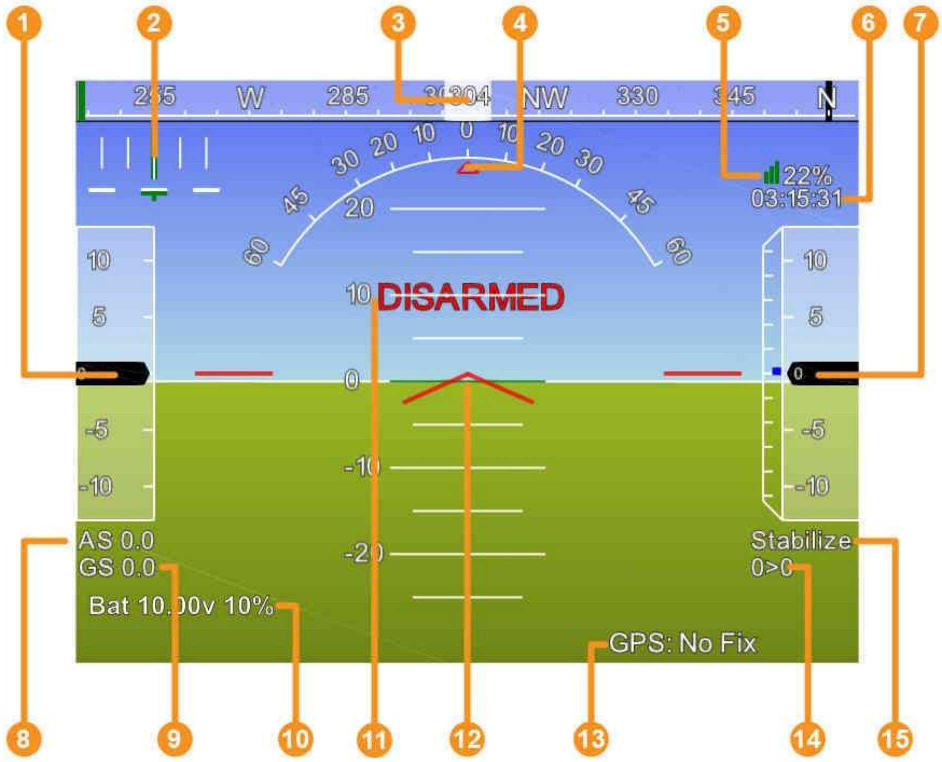


Figure 85: Mission Planner – Heads-Up Display (HUD) View

1. **Air speed (Ground speed if no airspeed sensor is fitted)**
2. **Cross-track error and turn rate (T)**
3. **Heading direction**
4. **Bank angle**
5. **Telemetry connection link quality (averaged % of good packets)**
6. **GPS time**
7. **Altitude (blue bar is rate of climb)**
8. **Air speed**
9. **Ground speed**
10. **Battery status**
11. **Artificial Horizon**
12. **Aircraft Attitude**
13. **GPS Status**
14. **Current Waypoint Number > Distance to Waypoint**
15. **Current Flight Mode**

For the FPV video modules, the ground control will be utilizing a diversity receiver operating at a frequency of 5.8 GHz. This means that the receiver will have two antennae at the same frequency and will automatically use the antenna with the stronger connection. The system will use two 5.8 GHz patch antennae. The payload team's range tests on the FPV module resulted in a maximum range of roughly three miles.

7.7.1.4 Payload Telemetry Testing

7.7.2 Electrical Testing

7.7.2.1

Electronics in the UAV must be able to sense orientation, velocity, and altitude during flight.	Test A demonstration of the Pixhawk PX4's sensing capabilities will initially be shown by its data transmittance to a ground station.
The vehicle shall be able to communicate drone data and have its parameters set via a ground control station must be shown.	Demonstration An ability to set parameters via a ground control station must be demonstrated.

7.8 Payload Prototype

Prototype Overview

A prototype of the UAV was constructed to serve as a test platform. The prototype features the same dimensions, wings, stabilizer, rudder, and control surfaces as the final payload design. Several of the features on the prototype that differ from the actual UAV is the implementation of a tail dragger style landing gear, which does not exist on the actual UAV. The prototype also does not feature and folding mechanisms on the wings or stabilizer. Another benefit of assembling a prototype was that it allowed for manufacturing techniques to be practiced and iterated. Wing manufacturing was a process that was targeted with this process.

7.8.1 Wing Manufacturing

The payload's wings were intended to be manufactured as closely to the final flight vehicle to practice correct manufacturing techniques and ensure the flight vehicle wings were identical to the design. The manufacturing process can be broken into six steps: hot wiring foam, carbon fiber sheet layup, carbon fiber filament layup, carbon fiber wrapping, curing, and trimming.

7.8.1.1 Hot wiring foam

The first step to manufacturing the payload wings is to cut the foam stock using a hot wire. The hot wire was fastened to a wooden jig and tightened as tight as possible. The hot wire was then connected to a power source where a voltage ranging between 15-20V was applied. Once the foam stock was cut, two wooden airfoil templates were nailed to each side of the stock to guide the hot wire across the foam. The foam wing and wooden template is shown in Figure 86.

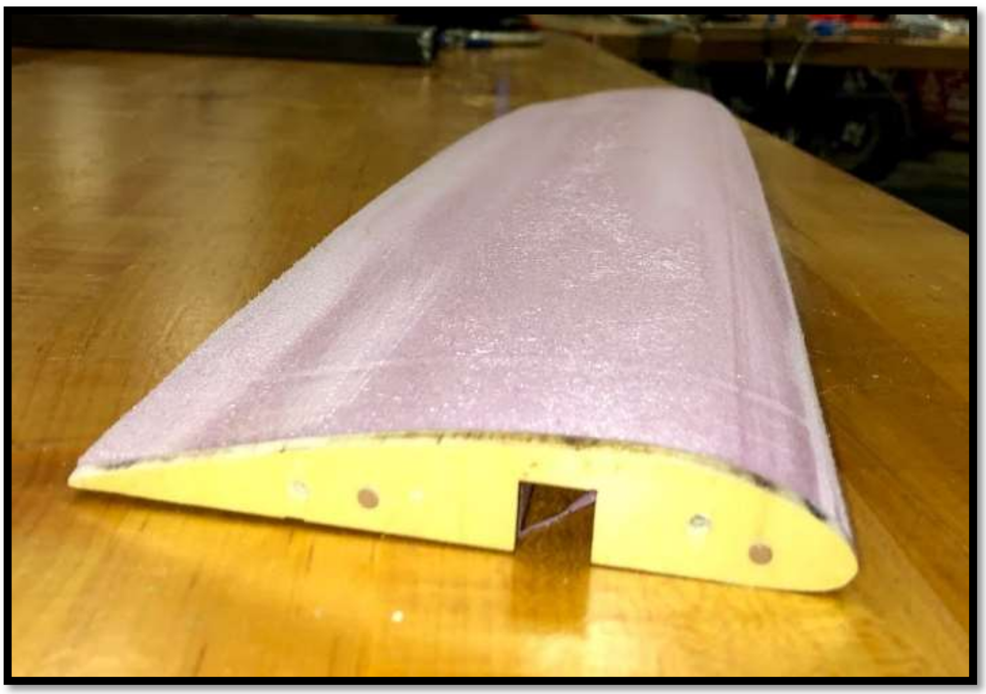


Figure 86: Foam wing and wooden template.

7.8.1.2 Carbon fiber sheet wrapping

The second step in the wing manufacturing process was the carbon fiber sheet layup. First, one layer of dry A&P Bimex carbon fiber sheet was cut using a box cutter. Next, two-part Aeropoxy PH3660 resin and hardener were mixed and lathered onto the dry carbon fiber sheet. The epoxy was carefully scrapped across the carbon fiber until the sheet was fully saturated. All excess epoxy was scraped off to minimize weight. Figure 87 illustrates the carbon fiber sheet layup process for the wings.

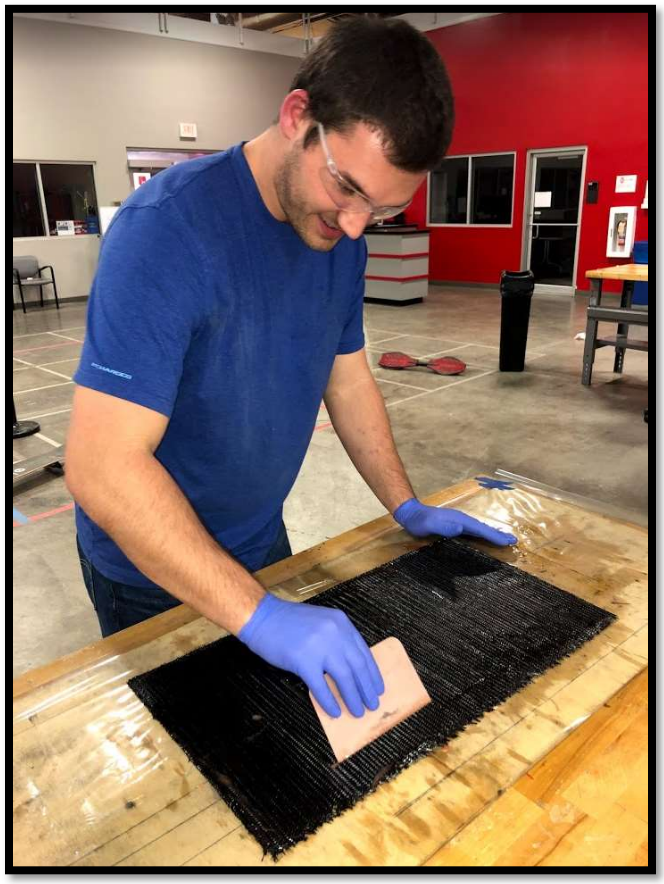


Figure 87: Wing carbon fiber layup.

7.8.1.3 Carbon fiber filament layup

The next step in the wing manufacturing process is laying up the carbon fiber filaments. Once the carbon fiber sheet was saturated with epoxy, carbon fiber filaments were cut to approximately 24" and laid along the long axis of the wing to improve the wing's strength against bending. The filaments were laid on the wing and covered with a light layer of epoxy. The carbon fiber filaments can be seen in Figure 88 below.

Figure 88: Carbon fiber filaments layed on carbon fiber sheet.

7.8.1.4 Carbon fiber wrapping

The fourth step in manufacturing the wings is wrapping each wing in the carbon fiber sheet/filament layup. During this process, it was critical to spread the carbon fiber layup evenly across the foam wing to prevent wrinkles from forming. Once an even layer of carbon fiber was pressed onto the wing, a putty scraper was used to spread the epoxy across the wing, moving from leading edge to trailing edge.

7.8.1.5 Curing

The fifth step in manufacturing the wings is curing the carbon fiber onto the foam wing. Once the carbon fiber was laid onto the foam wing, a thin layer of vacuum bag was wrapped around the wing for a smooth, glossy surface finish. The negative shell that was left from cutting the foam stock was then stacked on top of the wing and weighed down with 24” steel tubing to evenly compress the wing. The wing was left to cure for 24 hours. Figure 89 illustrates the carbon fiber wing curing under weight.

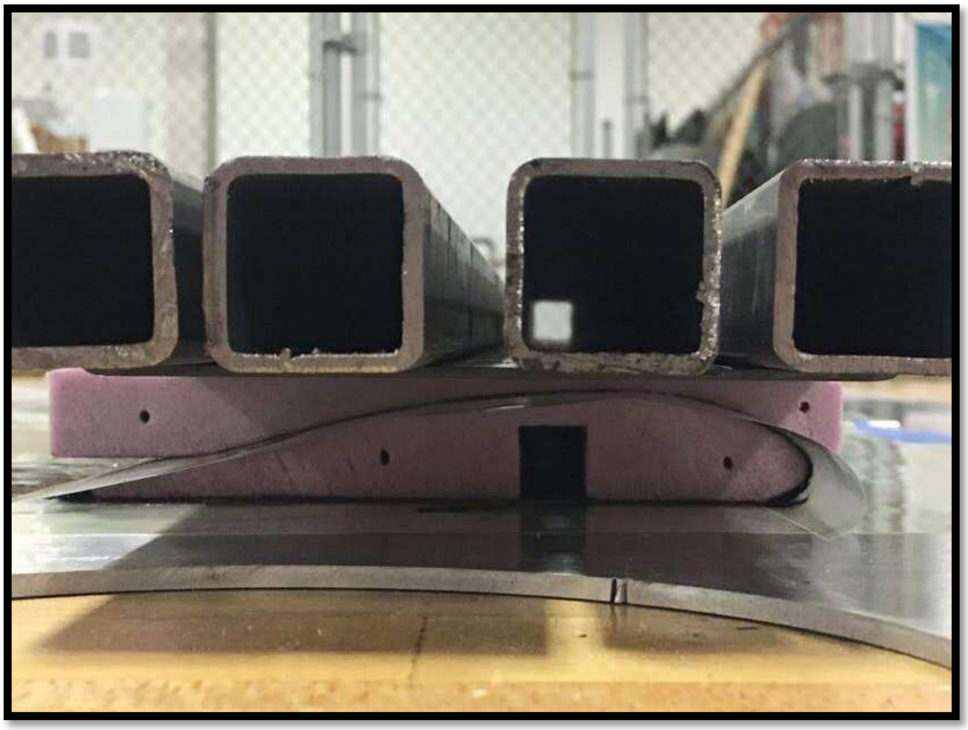


Figure 89: Carbon fiber wing curing under weight.

7.8.1.6 Trimming

The final step in manufacturing the wing was trimming the wing. Once the carbon fiber was cured, the weights and negative mold were removed. The wing was then cut to proper dimensions (22.00" length, 5.40" chord) using a miter saw with a . Lastly, the aileron was cutout using a Dremel. Figure 90 illustrates the final carbon fiber wing before the wooden supports and control surfaces were installed.

Figure 90: Carbon fiber wing without wooden supports and control surfaces.

Control Surface Assembly

Once the foam core composite wing was cured and trimmed. Wooden ribs and closeouts were assembled into the wing.

Driven Design Changes

The table below lists several design changes to the final UAV design that were driven by the assembly of the prototype.

7.8.2 Liquid Propulsion Test Plans

The following illustrates the completed tests which will be performed to guarantee the integrity of the designed liquid propulsion system. The following tests have been broken up into the following test campaigns:

- Injector Test Campaign
- [Sub-system #2 name] Test Campaign
- [Sub-system #3 name] Test Campaign

7.8.2.1 Injector Test Campaign

The Injector Test Campaign consists of several sub-tests which will prove the integrity of the design. The testing campaign for the injector assembly will be used to verify the structural integrity and performance of the design. A summary of the sub-tests in this campaign are shown below in Table 28.

Test	Test Description	Requirements Verified	Status
Hydrostatic Proof Test	The injector will be statically pressurized to the proof pressure (150% of nominal inlet pressure).	RE.1.2.4	Incomplete
Cold Flow Test	Pressurized flow testing with deionized water will be the performed with the injector to verify pressure drop, flow distribution, fluid atomization, and structural integrity of the design.	RE.1.2.1 RE.1.2.2 RE.1.2.3 RE.1.2.6	Incomplete
Hot Fire Pulse	A short duration hot fire will be performed for the liquid engine to verify combustion back pressure and hard start overpressures are under functional limits of the engine.		Incomplete
Full Duration Hot Fire	A full duration hot fire will be performed for the liquid engine to verify full functional of the development engine. Sub-objectives include verifying combustion heat transfer to injector, ablative liner characteristics, and injector flow properties.		Incomplete

Table 28: Injector sub-tests, test descriptions, requirements verified, and status.

7.8.2.1.1 Hydrostatic Proof Test

The injector will be statically pressurized to the proof pressure (150% of nominal inlet pressure).

Items to be tested

- Inlet Pressure
- Leakage

Pass/Fail Criteria

The following table describes the requirements that are being tested and the specifications to be able to accept tested requirement as a passed requirement.

ID #	Requirement	Pass/Fail Criteria	Results Summary
RE.1.2.1	The injector shall properly distribute the uniform flow of mass within the combustion chamber.	Video will record flow distribution during testing and pass/fail will be assigned qualitatively.	Test scheduled for [2/03/19]
RE.1.2.2	The injector shall not limit flow of fuel and oxidizer propellants into the combustion chamber.	Pressure measurements will be used to ensure significant back pressure does not build up upstream of the injector, thus indicating choked flow. Pass/fail will be determined quantitatively after analyzing pressure transducer data.	Test scheduled for [2/03/19]
RE.1.2.3	The injector shall drop the inlet pressure by 30% of the chamber pressure.	Pressure drop across injector will be determined by analyzing pressure transducer data after cold flow tests and hot-fires.	Test scheduled for [2/10/19]
RE.1.2.4	The injector shall maintain positive margins of safety on yield and ultimate strength of during worst-case combined loading with factors of safety of 1.1 and 1.4, respectively.	The injector will be inspected for any signs of detrimental deformation after completion of testing.	Test scheduled for [2/02/19]
RE.1.2.6	The injector shall inject propellant into the combustion chamber at a pressure of 1,000 psia.	Combustion pressure during hot-fire tests will be measured indirectly by analyzing strain gauge data on the combustion chamber.	Test scheduled for [2/02/19]

Table 29: Pass/fail criteria for all requirements tested.

Pre-Test

Equipment

- Pressure Transducers: Measure hydrostatic pressure inside fluid system/injector.
- DAQ system: Measure pressure data during testing.

Setup

Proof Pressure Setup

Deionized water will be loaded into the propellant tanks and pressurized using the nitrogen gas pressurant. A proof pressure test plate part will be installed on the outlet of the injector to maintain constant pressure during pressurization.

1) Load deionized water into propellant tanks.
2) Verify all fluid system connections are tight.
3) Turn on DAQ system and verify pressure data is being collected correctly.
4) Open manifold isolation valves.
5) Increase pressure regulator to desired proof pressure.
6) Hold pressure for a minimum of 5 min.
7) Verify no major leaks occurred.

Safety Notes

- Stand being concrete wall during system pressurization.
- Do not exceed a pressurization rate of 500 psig/min.

Procedure

The procedure for this test can be found in the [hyperlink to corresponding section] section.

Results

This test has not been conducted yet. It will be completed by 2/02/19.

7.8.2.1.2 Cold Flow Test

[Description of test objective.].

Items to be tested

- [Test parameter #1].
- [Test parameter #2].

Pass/Fail Criteria

The following table describes the requirements that are being tested and the specifications to be able to accept tested requirement as a passed requirement.

ID #	Requirement	Pass/Fail Criteria	Results Summary
RE.1.2.1	The injector shall properly distribute the uniform flow of mass within the combustion chamber.	Video will record flow distribution during testing and pass/fail will be assigned qualitatively.	Test scheduled for [2/03/19]

RE.1.2.2	The injector shall not limit flow of fuel and oxidizer propellants into the combustion chamber.	Pressure measurements will be used to ensure significant back pressure does not build up upstream of the injector, thus indicating choked flow. Pass/fail will be determined quantitatively after analyzing pressure transducer data.	Test scheduled for [2/03/19]
RE.1.2.3	The injector shall drop the inlet pressure by 30% of the chamber pressure.	Pressure drop across injector will be determined by analyzing pressure transducer data after cold flow tests and hot-fires.	Test scheduled for [2/10/19]
RE.1.2.4	The injector shall maintain positive margins of safety on yield and ultimate strength of during worst-case combined loading with factors of safety of 1.1 and 1.4, respectively.	The injector will be inspected for any signs of detrimental deformation after completion of testing.	Test scheduled for [2/02/19]
RE.1.2.6	The injector shall inject propellant into the combustion chamber at a pressure of 1,000 psia.	Combustion pressure during hot-fire tests will be measured indirectly by analyzing strain gauge data on the combustion chamber.	Test scheduled for [2/02/19]

Table 30: Pass/fail criteria for all requirements tested.

Pre-Test

Equipment

- [Equipment #1]: [Description of equipment #1 and its purpose during testing]
- [Equipment #2]: [Description of equipment #2 and its purpose during testing]

Setup

[Setup name] Setup

[High level description of test setup (how equipment will be used, where it be performed, etc.)]

8) [Test setup step #1]
9) [Test setup step #2]

[Setup name] Setup (if applicable)

[High level description of test setup (how equipment will be used, where it be performed, etc.)]

1) [Test setup step #1]

2) [Test setup step #2]

Safety Notes
[State any notable safety consideration that need to be adhered to during testing]

Procedure
The procedure for this test can be found in the [hyperlink to corresponding section] section.

Results
This test has not been conducted yet. It will be completed by [M/DD/YY].

7.8.2.1.3 Hot Fire Pulse Test

[Description of test objective.].

Items to be tested

- [Test parameter #1].
- [Test parameter #2].

Pass/Fail Criteria
The following table describes the requirements that are being tested and the specifications to be able to accept tested requirement as a passed requirement.

ID #	Requirement	Pass/Fail Criteria	Results Summary
RE.1.2.1	The injector shall properly distribute the uniform flow of mass within the combustion chamber.	Video will record flow distribution during testing and pass/fail will be assigned qualitatively.	Test scheduled for [2/03/19]
RE.1.2.2	The injector shall not limit flow of fuel and oxidizer propellants into the combustion chamber.	Pressure measurements will be used to ensure significant back pressure does not build up upstream of the injector, thus indicating choked flow. Pass/fail will be determined quantitatively after analyzing pressure transducer data.	Test scheduled for [2/03/19]
RE.1.2.3	The injector shall drop the inlet pressure by 30% of the chamber pressure.	Pressure drop across injector will be determined by analyzing pressure transducer data after cold flow tests and hot-fires.	Test scheduled for [2/10/19]
RE.1.2.4	The injector shall maintain positive margins of safety on	The injector will be inspected for any signs	Test scheduled for [2/02/19]

	yield and ultimate strength of during worst-case combined loading with factors of safety of 1.1 and 1.4, respectively.	of detrimental deformation after completion of testing.	
RE.1.2.6	The injector shall inject propellant into the combustion chamber at a pressure of 1,000 psia.	Combustion pressure during hot-fire tests will be measured indirectly by analyzing strain gauge data on the combustion chamber.	Test scheduled for [2/02/19]

Table 31: Pass/fail criteria for all requirements tested.

Pre-Test

Equipment

- [Equipment #1]: [Description of equipment #1 and its purpose during testing]
- [Equipment #2]: [Description of equipment #2 and its purpose during testing]

Setup

[Setup name] Setup

[High level description of test setup (how equipment will be used, where it be performed, etc.)]

10) [Test setup step #1]
11) [Test setup step #2]

[Setup name] Setup (if applicable)

[High level description of test setup (how equipment will be used, where it be performed, etc.)]

3) [Test setup step #1]
4) [Test setup step #2]

Safety Notes

[State any notable safety consideration that need to be adhered to during testing]

Procedure

The procedure for this test can be found in the [hyperlink to corresponding section] section.

Results

This test has not been conducted yet. It will be completed by [M/DD/YY].

7.8.2.1.4 Hot Fire Full Duration Test

[Description of test objective.].

Items to be tested

- [Test parameter #1].
- [Test parameter #2].

Pass/Fail Criteria

The following table describes the requirements that are being tested and the specifications to be able to accept tested requirement as a passed requirement.

ID #	Requirement	Pass/Fail Criteria	Results Summary
RE.1.2.1	The injector shall properly distribute the uniform flow of mass within the combustion chamber.	Video will record flow distribution during testing and pass/fail will be assigned qualitatively.	Test scheduled for [2/03/19]
RE.1.2.2	The injector shall not limit flow of fuel and oxidizer propellants into the combustion chamber.	Pressure measurements will be used to ensure significant back pressure does not build up upstream of the injector, thus indicating choked flow. Pass/fail will be determined quantitatively after analyzing pressure transducer data.	Test scheduled for [2/03/19]
RE.1.2.3	The injector shall drop the inlet pressure by 30% of the chamber pressure.	Pressure drop across injector will be determined by analyzing pressure transducer data after cold flow tests and hot-fires.	Test scheduled for [2/10/19]
RE.1.2.4	The injector shall maintain positive margins of safety on yield and ultimate strength of during worst-case combined loading with factors of safety of 1.1 and 1.4, respectively.	The injector will be inspected for any signs of detrimental deformation after completion of testing.	Test scheduled for [2/02/19]
RE.1.2.6	The injector shall inject propellant into the combustion chamber at a pressure of 1,000 psia.	Combustion pressure during hot-fire tests will be measured indirectly by analyzing strain gauge data on the combustion chamber.	Test scheduled for [2/02/19]

Table 32: Pass/fail criteria for all requirements tested.

Pre-Test

Equipment

- [Equipment #1]: [Description of equipment #1 and its purpose during testing]
- [Equipment #2]: [Description of equipment #2 and its purpose during testing]

Setup

[Setup name] Setup

[High level description of test setup (how equipment will be used, where it be performed, etc.)]

12) [Test setup step #1]

13) [Test setup step #2]

[Setup name] Setup (if applicable)
[High level description of test setup (how equipment will be used, where it be performed, etc.)]

5) [Test setup step #1]
6) [Test setup step #2]

Safety Notes
[State any notable safety consideration that need to be adhered to during testing]

Procedure
The procedure for this test can be found in the [hyperlink to corresponding section] section.

Results
This test has not been conducted yet. It will be completed by [M/DD/YY].

8 Safety

8.1 Introduction

Maintaining a safe work environment and safe work practices is essential for the safety and health of the team. This is a means of protecting all personnel from harm as well as the rocket and any other team equipment from damage. The following sections detail general safety guidelines that are set forth by the team as well as the Engineering Garage and the University of Louisville.

8.2 Engineering Garage Rules and Regulations

8.2.1 Participation Release Form

The University requires that Mike Miller, the Garage manager, have all people working in the Garage to sign a U of L participation release form. Mike will keep the originals in his office and a binder will be kept in the main area of the garage for the Safety Officer or team lead to ensure that all prospective or current team members comply.

8.2.2 Garage Access

Once the Participation Release Form has been completed, the team member will be granted access to the building with their University of Louisville Student ID called a Cardinal Card. Students that have not completed the form may enter the garage but are not allowed to assist with or complete any work for the team. The Safety Officer is responsible for ensuring that each member has completed this form.

8.2.3 Engineering Garage Tables

The Engineering Garage consists of 6 clusters of 4 tables each that students use for Engineering Fundamentals classes during the spring semesters. These tables are available for River City Rocketry to conduct work on, but it must be done so in a careful manner. The tables should never be damaged by a River City Rocketry team member in any way. When working with epoxy or other messy substances, one must place a large sheet of cardboard (available next to FSAE's cage) on top of the table to protect it. Additionally, if a table were to become messy, one should never use acetone to clean off the table as it will damage the table’s surface. If extremely messy or potentially damaging work must be done, the work benches near the loading dock may be used.

8.2.4 Machine Cage

If the team member needs access to the garage Machine Cage he or she must pass an online safety quiz as required by the Garage. Following completion, the team member will notify Mike Miller and receive access through the Cardinal Card. Prior to using the equipment in the Machine Cage, the team member must be mentored by an experienced member and certified by Mike Miller. CNC classes are offered by request. Members are only allowed to operate machines that they have been certified on doing otherwise is grounds to be barred from the Garage. Safety glasses, long pants, and close toed shoes are required to be worn whenever in the Machine Cage. Two people must be in the Machine Cage any time when equipment is turned on. In case there is an emergency, the lab partner will be able to engage an emergency shutoff. There are Job Safety and Sequence Instruction cards at each machine that will remain available for review at any time. The safety manuals for each machine are available from Mike Miller upon request. If a team member has any issues or concerns with a piece of equipment, place an “Equipment needs repair” sign on the machine and contact Mike Miller. If the machine makes an unusual noise or acts in any suspicious manner, you must stop the machine and inform the Engineering Garage manager, Mike Miller, immediately.

8.2.5 Electronics’ Work Bench

Miller must certify any team member who wishes to use the Electronics’ Work Bench. The soldering irons tips must always be tinned to prevent damage. Caution must be exercised when using the soldering tips as they can reach up to 750 degrees Fahrenheit. Similarly, the heat gun approaches 480 degrees Fahrenheit and require the same caution. Power supplies must be used properly to avoid shorting the equipment leading to hazards as severe as explosions.

8.3 Manufacturing and Assembly Rules and Regulations

8.3.1 Attire

All team member must wear appropriate attire when working in the Engineering Garage. While this may differ depending on the particular job at hand, below are general guidelines to apply to most projects.

- No loose clothing or jewelry that could be caught in moving or revolving machinery or tools should be worn. This includes lanyards, wired earbuds, and jackets with strings.
- Long hair should be tied back to prevent tangling in moving machinery.
- Jewelry or other metal objects should not be worn while working with energized electrical circuits/equipment.
- Long sleeves and long pants must be worn when sanding, cutting, or manufacturing carbon fiber or fiberglass.

8.3.2 Tool Training

Team members must be trained on any tools used such as drills, Dremels, soldering equipment, etc. Those not trained should not attempt to learn on their own and should instead seek out an experienced team member who can mentor them.

8.3.3 Workspace

All work spaces should be kept clean and orderly. A clean-as-you-go mentality should be used to prevent a cluttered workspace and misplaced tools. All flammable materials should be returned to the flammable cabinet when they are no longer in use. The flammable cabinet is to remain closed all other times. The following materials cannot be disposed of in a regular trash can and require separate waste bins that are available in the Engineering Garage.

- Oily waste
- Metal or wooden sharps
- Solvent contaminated rags or paper towels
- Aerosol cans
- Batteries of any kind

Cardboard should be placed on the workbenches prior to protect the surface of the benches. Any dust or metal shavings shall be vacuumed, and all tools and parts should be accounted for and put away. By using Foreign Object Elimination (FOE) techniques, Foreign Object Debris (FOD) can be avoided. FOD can be particularly dangerous in the electrical components that are Foreign Object Sensitive (FOS), causing shorts, failures, or the charging of undesired components.

You must not enter the Engineering Garage under the influence of drugs or alcohol, specifically over the counter drugs that include warnings against operating machinery. You must not consume alcohol within 8 hours of entering the Engineering Garage.

8.3.4 Emergency Equipment

The Engineering Garage is equipped with the following emergency equipment

- Seven mounted fire extinguishers
- Four first aid and eyewash stations are available with single use eyewash kits
- Automated external defibrillator
- Emergency burn kit
- Bloodborne pathogens protection kit

Each team member is personally responsible for knowing where each is located as shown in Figure 91.

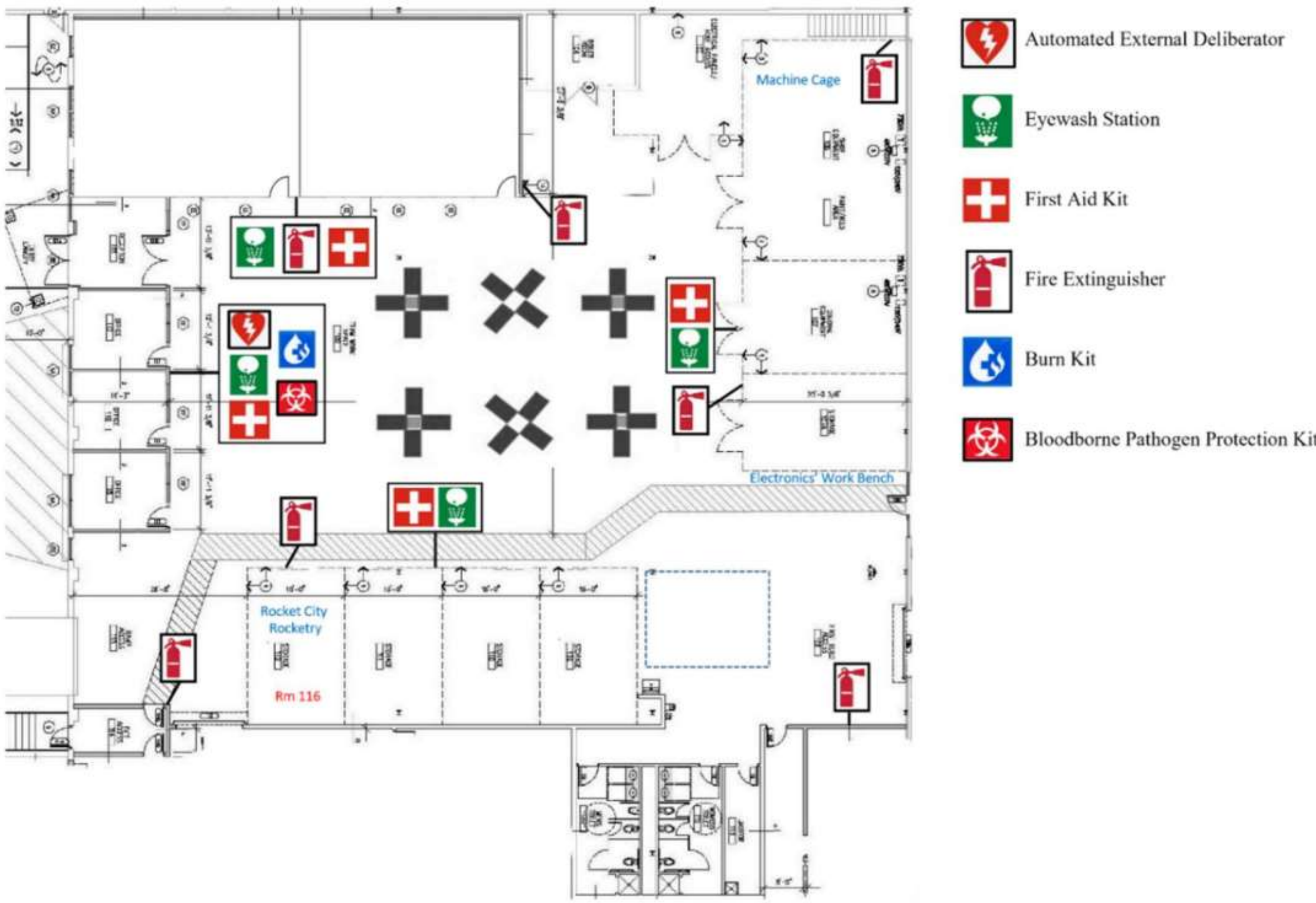


Figure 91: Engineering Garage emergency equipment locations.

8.4 Personal Protection Equipment (PPE)

8.4.1 Hazardous Tasks

All lab user shall assess any hazardous tasks and take the proper precautions. Material Safety Data Sheets (MSDS) and operator manuals shall be consulted as necessary. Appropriate Personal Protective Equipment (PPE) should be worn and is required when performing hazardous tasks.

8.4.2 Safety Glasses

Safety glasses shall be worn when performing tasks where flying foreign debris or splashed chemicals could get in one's face. Safety glasses will always be available in the cage and near the first aid kits in the Garage. Children must wear safety glasses during educational event activities where foreign debris or objects could damage the student's eye.

8.4.3 Hearing Protection

Hearing protection should be worn when working with heavy power tools and during any other loud operations as required. Ear plugs are available at the first aid kits in the Garage and in dispensers. All other nearby personnel in the garage are to be warned beforehand before work is to begin.

8.4.4 Protective Gloves

Different types of gloves shall be worn to protect hands during various operations:

- Cut-resistant gloves should be worn when working with cutting instruments to mitigate the risks of severe cuts from tooling. These can be at least ANSI 2011 cut level 3 and constructed of either HPPE based material or Kevlar.
- Nitrile gloves should be worn while working with materials such as airframe epoxy, carbon fiber dust, paint, or lubricants that could irritate the skin, by toxic, or be difficult to wash off.
- Thermal gloves should be worn while working with heat elements or heated components.

8.4.5 Face Masks

Different types of masks shall be worn during various tasks to mitigate various safety hazards. MSDS should be consulted to determine if the use of a mask is necessary.

- Dust or particulate masks shall be worn when working with materials such as carbon fiber or fiberglass that generate dust particles, particularly while sanding
- When working with chemicals producing toxic fumes, a vapor mask should be worn. The MSDS should be consulted prior to working with any chemicals to see if a vapor mask is necessary. It is important to make sure that the vapor mask provides the appropriate filtration for the chemical used.
- A full-face shield should be worn when performing any operation that produces sparks as a secondary measure in addition to safety glasses.

8.4.6 Task Specific Protective Equipment

Additional protective equipment will be worn depending on the task on hand, based on industry standards. Examples include:

- Welding, which would require the necessary tinted facemask, coat, gloves, footwear, and other required PPE.
- Using of harsh chemicals or materials, particularly for solid or liquid propulsion, requiring chemical resistant PPE.

8.5 Tool Specific Safety

It is important to be familiar with any tool specific safety information for personal safety and to ensure that the rocket is not damaged by improper tool use. It is particularly important to know these guidelines to keep the equipment in good condition. The team heavily uses the Garage equipment that other engineering teams depend on as well.

8.5.1 Universal Laser System

The Garage has a universal laser system. The following are the safety rules and regulations for the laser set forth by the Garage:

- Do not run a material without knowing what it is.

- Turn on compressed air and filtration system/dust collector (foot switch will activate dust collector).
- Always follow the step by step read me file on laser's PC.
- Complete job.
- Make sure the Read Me First Document PDF is open on the Desk Top so that it is the first thing the next user sees.

8.5.2 SawStop Table Saw

The Garage has a SawStop table saw. The team utilizes this machine to cut sheets of materials into smaller manageable pieces. The following are the safety rules and regulations for the SawStop table saw set forth by the Garage:

- Inspect blade and guard before using.
- Set fence to desired location.
- Hit Foot Switch to turn dust collection on.
- Turn on machine (Note: Cut non-conductive materials only). If you are using a Formica or other plastic that may build up electrostatic charge, see shop manager to have that material tested before you use the saw. Failure to do so will trip the brake and damage the brake cartridge and blade).
- Slide the material against fence and push toward blade, holding material down on table while cutting.
- Push material toward saw blade, (Note: If possible, use push T-slide or finger board tools while pushing material, keeping hands away from blade at all times).
- After desired cut is made, turn off table saw.
- Wait for the saw blade to stop rotating, then remove material.
- Clean area and equipment.

8.5.3 Jet Vertical Band Saw

The Garage has a vertical band saw. The following are the safety rules and regulations for the vertical band saw as outlined by the Garage:

- Adjust saw guard to the height for object being cut.
- Turn on machine.
- Slide work to blade to cut insuring hands are well away from blade.
- Push workpiece against cutting blade. If possible use T-slide to push material against blade. Never get close to the blade.
- Let the saw do the work. Excess pressure will damage the blade.
- When desired cut is made, turn off machine.
- Remove workpiece.
- Clean area and equipment.

8.5.4 Bench Grinder

The Garage has a bench grinder. The following are the safety rules and regulations for the bench grinder set forth by the Garage:

- Adjust plastic shield if needed to protect against flying sparks from metal.
- Turn on machine.

- Hold workpiece against grinding wheel, insuring hands are well away from grinding wheel. (Use pliers or vise grips if working on a small piece).
- Do not let workpiece bind in the between the grinding wheel and grinder housing that could cause it to catch and turn work piece into projectile if not clamped securely.
- When desired sanding is complete, remove workpiece.
- Turn off machine.
- Clean area and equipment.

8.5.5 Jet 55 Ton Shop Press

The Garage has a 55 ton shop press. The following are the safety rules and regulations for the 55 ton shop press set forth by the Garage:

- Do not bend or shear material larger than 30" 20-gauge mild steel. Failure to comply may cause serious injury and/or damage to the machine.
- See attached pages for basic operation (attached pages will be located directly to the left of the shop press.
- Clean area and equipment.

8.5.6 Jet Horizontal Band Saw

The Garage has a horizontal band saw. The following are the safety rules and regulations for the horizontal band saw as set by the Garage:

- Adjust lever on hydraulic cylinder to hold blade away from work area.
- Clamp workpiece in place securely.
- Keep hands away from blade and turn on machine.
- Let the weight of the saw do the work. Do not ad pressure to saw.
- When desired cut is made, turn off machine, if the machine did not turn off automatically.
- Remove workpiece.
- Clean area and equipment.

8.5.7 Sand Blaster

The Garage has a sand blaster. The following are the safety rules and regulations for the sand blaster as set forth by the Garage:

- Open side door.
- Place work piece on blaster table.
- Close side door.
- Turn on machine and light.
- Put hands in black long gloves.
- Hold part and blaster want.
- Depress foot switch.
- Move wand over part to blast.
- Turn off machine.
- Remove workpiece.
- Clean area and equipment.

8.5.8 Clark 3-axis Manual Mill

The Garage has a 3-axis manual mill. The following are the safety rules and regulations for the 3-axis manual mill as set by the Garage:

- Place work in mill vise.
- Load tool in mill head.
- Return all tools and chuck key to the tool tray beside the machine.
- Machine work as needed.
- Turn off machine.
- Remove workpiece.
- Clean area and equipment.

8.5.9 4' x 8' SHOPBOT

The Garage has a 4' x 8' SHOPBOT. The following are the safety rules and regulations for the SHOPBOT as defined by the Garage:

- Turn on machine
- Program machine.
- Set work holding for job.
- Select and set tools for job.
- Check program in machine.
- Dry run program
- Hit foot switch to turn dust collector on.
- Run program.
- After program is finished move SHOPBOT head away from part and shut off machine.
- Retrieve your part from the work area.
- Clean area and equipment

8.5.10 Tormach CNC 3-axis mill

The Garage has a CNC 3–axis mill. The following are the safety rules and regulations for the CNC 3-axis mill set forth by the Garage:

- Turn on machine.
- Program Machine.
- Set work holding for job.
- Select and set tools for job.
- Check program in machine.
- Always hit Emergency Stop before putting your hand or head in the enclosure.
- Never have someone operate the controls with your hand or head in the enclosure.
- Dry run program.
- Run program.
- Clean area and equipment once coolant has dried.

8.5.11 Jet Drill Press

The Garage has a drill press. The following are the safety rules and regulations for the drill press as outlined by the Garage:

- Secure drill bit with chuck key (key can be found in a clip on the back-right side of the drill).
- Remove key from chuck and place back in the clip on the drill press.
- Adjust worktable to the proper vertical and horizontal position so the drill bit will reach the workpiece and drill through the center hole of this worktable without hitting the worktable.
- Place workpiece on drill press worktable or clap in vise to secure material. Never hold small workpieces with your hands. Clamp them with C-Clamps or U-Clamps.
- Turn on machine and lower spindle to drill hole.
- Once hole is drilled, turn off machine.
- Remove workpiece.
- Clean area and equipment.

8.5.12 Manual Lathe

The Garage has a manual lathe. The following are the safety rules and regulations for the lathe set forth by the Garage:

- The chuck key is on the front right attached to a magnet.
- Place workpiece in lathe using the chuck key.
- Always remove chuck key and return it back to its home on the magnet on the right front, not in the tray. Never ever leave it in the chuck or it will become a missile.
- Load tool in tool holder.
- Machine workpiece as needed.
- Turn off machine.
- Remove workpiece.
- Clean area and equipment.

8.6 NRA/TRA Procedures

8.6.1 NAR Safety Code

Table 33 describes each component of the High-Power Safety Code, as provided by NAR, and how the team will comply with each component. The only exception to this rule is section 3, motors, because of the team's research and development into solid motors and liquid rocket engines. The team's safety procedure is noted in Section 8.8.2. The minimum distance table is also included below in Table 34.

NAR Code	Team Compliance
1. **Certification.** I will only fly high power rockets or possess high power rocket motors that are within the scope of my user certification and required licensing.	Only Matthew Cosgrove, the Launch Vehicle Lead, Darryl Hankes, the team mentor, or certified team members are permitted to pack or handle the rocket motors.

2. **Materials.** I will use only lightweight materials such as paper, wood, rubber, plastic, fiberglass, or when necessary ductile metal, for the construction of my rocket.	The Vehicle and Payload sub-teams will select appropriate materials for the rocket while considering structure and weight.
3. **Motors.** I will use only certified, commercially made rocket motors, and will not tamper with these motors or use them for any purposes except those recommended by the manufacturer. I will not allow smoking, open flames, nor heat sources within 25 feet of these motors.	The motors will be purchased from Chris' Rocket Supplies and will be stored and handled only by certified members. The entire team will understand and agree to the motor safety portion of this safety manual.
4. **Ignition System.** I will launch my rockets with an electrical launch system, and with electrical motor igniters that are installed in the motor only after my rocket is at the launch pad or in a designated prepping area. My launch system will have a safety interlock that is in series with the launch switch that is not installed until my rocket is ready for launch, and will use a launch switch that returns to the "off" position when released. The function of onboard energetics and firing circuits will be inhibited except when my rocket is in the launching position.	All launches will be conducted at NAR/TRA certified events. The Range Safety Officer will have the final say over any safety issues. There will be arming switches for the altimeters that will inhibit premature activation of firing circuits that will not be armed before the rocket is on the launch pad. These arming switches may include screw switches, key switches, or pull pins.
5. **Misfires.** If my rocket does not launch when I press the button of my electrical launch system, I will remove the launcher's safety interlock or disconnect its battery, and will wait 60 seconds after the last launch attempt before allowing anyone to approach the rocket.	The safety officer will remind the team of this rule prior to the 5-second countdown. The safety captain will communicate any precautions given by the Range Safety Officer the day of the launch.
6. **Launch Safety.** I will use a 5-second countdown before launch. I will ensure that a means is available to warn participants and spectators in the event of a problem. I will ensure that no person is closer to the launch pad than allowed by the accompanying Minimum Distance Table. When arming onboard energetics and firing circuits I will ensure that no person is at the pad except safety personnel and those required for arming and disarming operations. I will check the stability of my rocket before flight and will not fly it if it cannot be determined to be stable. When conducting a simultaneous launch of more than one high power rocket I	The safety officer will sound an air horn prior to the 5-second countdown to ensure spectator awareness. The team will comply with this rule and any other rules given by the Range Safety Officer the day of the launch.

will observe the additional requirements of NFPA 1127.7.	
7. **Launcher.** I will launch my rocket from a stable device that provides rigid guidance until the rocket has attained a speed that ensures a stable flight, and that is pointed to within 20 degrees of vertical. If the wind speed exceeds 5 miles per hour I will use a launcher length that permits the rocket to attain a safe velocity before separation from the launcher. I will use a blast deflector to prevent the motor's exhaust from hitting the ground. I will ensure that dry grass is cleared around each launch pad in accordance with the accompanying Minimum Distance table, and will increase this distance by a factor of 1.5 and clear that area of all combustible material if the rocket motor being launched uses titanium sponge in the propellant.	The team will comply with this rule by launching out of the same rails provided by NAR at competition. The team prefers to launch in surface winds less than 4 times the exit rail velocity or less than 20 miles per hour to ensure the stability of the rocket
8. **Size.** My rocket will not contain any combination of motors that total more than 40,960 N-sec (9,208 pound-seconds) of total impulse. My rocket will not weigh more at liftoff than one-third of the certified average thrust of the high power rocket motor(s) intended to be ignited at launch.	The team will comply to this rule when designing the rocket and selecting an appropriate motor.
9. **Flight Safety.** I will not launch my rocket at targets, into clouds, near airplanes, nor on trajectories that take it directly over the heads of spectators or beyond the boundaries of the launch site, and will not put any flammable or explosive payload in my rocket. I will not launch my rockets if wind speeds exceed 20 miles per hour. I will comply with Federal Aviation Administration airspace regulations when flying, and will ensure that my rocket will not exceed any applicable altitude limit in effect at that launch site.	A wind gauge and weather predictions will be used to make a weather assessment will be conducted prior to launch. Appropriate FAA waivers and adequate notice will be in place before the launch occurs. The team will comply with this and any determination made by the Range Safety Officer on the day of the launch.
10. **Launch Site.** I will launch my rocket outdoors, in an open area where trees, power lines, occupied buildings, and persons not involved in the launch do not present a hazard, and that is at least as large on its smallest dimension as one-half of the maximum altitude to which rockets are allowed to be flown at that site or 1,500 feet,	All team launches will be at NAR/TRA certified events. The Range Safety Officer will have the final say over any rocketry safety issues.

whichever is greater, or 1,000 feet for rockets with a combined total impulse of less than 160 N-sec, a total liftoff weight of less than 1,500 grams, and a maximum expected altitude of less than 610 meters (2,000 feet).	
11. **Launcher Location.** My launcher will be 1,500 feet from any occupied building or from any public highway on which traffic flow exceeds 10 vehicles per hour, not including traffic flow related to the launch. It will also be no closer than the appropriate Minimum Personnel Distance from the accompanying table from any boundary of the launch site.	The team will comply with this rule and any determination the Range Safety Officer makes on launch day.
12. **Recovery System.** I will use a recovery system such as a parachute in my rocket so that all parts of my rocket return safely and undamaged and can be flown again, and I will use only flame-resistant or fireproof recovery system wadding in my rocket.	The Recovery team will be responsible for designing, constructing, and testing a safe recovery system for the rocket. A clear recovery checklist will be followed on the launch day to ensure that all critical steps in preparing and packing the recovery components are completed.
13. **Recovery Safety.** I will not attempt to recover my rocket from power lines, tall trees, or other dangerous places, fly it under conditions where it is likely to recover in spectator areas or outside the launch site, nor attempt to catch it as it approaches the ground.	The team will comply with this rule and any determination the Range Safety Officer makes on launch day. If necessary, professionals will be contacted for rocket retrieval.

Table 33. NAR regulations and team compliance.

Installed Total Impulse (Newton-Seconds)	**Equivalent High Power Motor Type**	**Minimum Diameter of Cleared Area (ft.)**	**Minimum Personnel Distance (ft.)**	**Minimum Personnel Distance (Complex Rocket) (ft.)**
0-320.00	H or smaller	50	100	200
320.01-640.00	I	50	100	200
640.01-1,280.00	J	50	100	200
1,280.01-2,2560.00	K	75	200	300
2,560.01-5,120.00	L	100	300	500
5,120.01-10,240.00	M	125	500	1,000

10,240.01-20,480.00	N	125	1,000	1,500
20,480.01-40,960.00	O	125	1,500	2,000
Note: A complex rocket is one that is multi-staged or that is propelled by two or more rocket motors.				

Table 34. NAR minimum distance table.

8.7 Local, State, and Federal Law Compliance

All team members are required to review and acknowledge the following regulations regarding unmanned rocket launches and motor handling

Federal Aviation Regulations 14 CFR, Subchapter F, Part 101, Subpart C: https://www.ecfr.gov/cgi-bin/text-idx?c=ecfr&tpl=/ecfrbrowse/Title14/14tab_02.tpl

Code of Federal Regulation 27 Part 55: Commerce in Explosives; and fire prevention: https://www.federalregister.gov/documents/2003/01/24/03-1657/reorganization-of-title-27-code-of-federal-regulations

NFPA 1127 "Code for High Power Rocket Motors": http://www.nfpa.org/codes-and-standards/all-codes-and-standards/list-of-codes-and-standards/detail?code=1127

Federal Explosives Laws and Regulations:

http://www.atf.gov/files/publications/download/p/atf-p-5400-7.pdf

8.8 Launch Safety

8.8.1 Launch Day Briefing

Prior to each launch a briefing will be held to review potential hazards and accident avoidance strategies. Attendance will be mandatory for team members to attend launches. To prevent accidents, thorough safety checklists will be created prior to launch day for each subsystem, the overall assembly, and the launch pad procedures. Throughout preparations, it will be the responsibility of the team captains to confirm that each of the necessary tasks for a successful launch are completed. This will be verified by having two team members sign off on each step as they completed it, holding them accountable for that portion of the assembly.

8.8.2 Motor Safety

Darryl Hankes, the team mentor, who has obtained his Level 3 TRA certification, has agreed to assist the team in all things related to motor storage, purchasing, and development. Darryl and team member Eric Lewis, who obtained his Level 2 certification in July 2018, are the only team members permitted to assist in the purchasing and assembly of commercially available solid rocket motors. If at any time, another member of the team acquires the appropriate certification, they will be added to the list of people permitted to handle the team's motors and the Safety Manual will be updated. By having obtained at minimum a Level 2 certification, the individual has demonstrated that he or she understands the safety guidelines regarding motors. Any certified member of the team that handles or stores the team's motors is responsible for following the appropriate measures. The motors for both test and competition launches will be transported by car to the launch site in a suitable container.

8.9 Educational Engagement Safety

General

- Children should never work unsupervised.
- A child is never to handle any size rocket motor on their own.
- Safety should always be encouraged when teaching young students about rockets and during construction.
- Personal Protective Equipment that a child, or any other participant in an educational event may need will be provided.

Rocket Construction

- Horseplay in the classroom is not to be tolerated.
- Students are not permitted to use exacto knives during builds. These operations must be performed by an adult.
- Students are not to use the hot glue guns without approval from their parent or teacher and must be informed on how to safely use a hot glue gun.
- Children must work supervised in order to ensure proper assembly of a safe rocket to launch.

Rocket Launch

- Prior to any launch, students are to be briefed on safety procedures. Any students not following these measures will not be able to participate in the remaining launches for the day.
- Students and educators standing near the launch pad are to be wearing safety glasses at all times.
- The secondary key for the launch mechanism must be removed when loading rockets onto the launch pad.
- Students must always remain a designated safe distance from the launch pad .
- The secondary key is not to be inserted until all launch pads are clear and the safety officer has given the okay.
- Students are not to launch until they have been given permission by the safety officer. Notification to observing students and a countdown will occur to signify this event.
- Should a rocket not launch on the first attempt, students and educators shall wait 30 seconds to ensure that it was a misfire. The secondary key shall be removed before approaching the launch pad to avoid and accidental launch.
- Students are never to catch a falling rocket or component of a rocket.

8.10 System Level Risk Assessment Analysis

In accordance with the IREC Rules & Requirements document, River City Rocketry has conducted a risk assessment on all projects deemed to generate significant risk to the safety of team personnel, spectators, and the environment. This section summarizes risk and reliability concepts along with analyzing failure modes and mitigation techniques associated with each system.

8.10.1 Vehicle Risk Assessment Analysis

Hazard	Probable Causes	Risk Level and Rationale	Mitigation Approach	Risk of Injury After Mitigation
Rocket deviates from nominal path shortly after rail exit	Incorrect design results in unstable vehicle	Low – design analysis and simulations documented previously have demonstrated that the vehicle design is stable.	Internal design reviews and simulations will be conducted to ensure a stable design.	Low
	Motor misfires resulting in lower exit rail speed and lower stability at rail exit	Medium – as this is the team's first attempt at developing solid rocket motors, there is an elevated risk of a misfire.	The team will be using a tested propellant recipe instead of developing a new one from scratch. The recipe has been flown several times by the team's mentor and has demonstrated reliability. Static test fires will also be conducted to ensure reliability.	Low

8.10.2 Solid Rocket Motor Risk Assessment Matrix

Operating and firing a solid rocket motor poses a variety of risks that must be mitigated. Table 35 lists the possible hazards with static fire testing and how the team will mitigate risk. There is only one additional risk specific to solid motors while the vehicle is flying (CATO during flight) as opposed to on the test stand, and that hazard is covered in Table 35 as well.

Hazard	Possible Causes	Mitigation Approach	Risk Level After Mitigation
Uncontrolled fire surrounding test location	CATO causes scattering of burning propellant	Wet sand will be scattered around the testing location to mitigate the chance of a fire spreading. Fire extinguishers and the local fire department will be on site.	Low

	Exhaust flame catches vegetation on fire	The concrete pad has been custom designed and built to mitigate the chance of the exhaust flume reaching the surrounding vegetation. In addition, the surrounding vegetation will either be removed or covered in wet sand.	Low
	Rocket motor breaks free from test stand and catches fire to surrounding area	The test stand was built from a proven design used in the rocket motor industry. Rather than try to design our own, the team chose to use a design used by industry leading rocket motor manufacturer Aerotech.	Low
Fragmentation of motor casing components from explosion during test fire	Cracks in propellant	Propellant mixture is a previously developed and tested recipe known to provide stable operation. The team will visually inspect all motor grains for cracks and de-bonding from the liner walls.	Medium
	Debonding of propellant from liner wall	During the mixing process, the liner will be coated in an adhesive to promote adhesion to liner wall. The team shall inspect the motor for damage from transportation to the test site.	Low

	Gaps between propellant sections and/or nozzle cause increase in burn rate and over pressurization of motor casing	River City Rocketry is not attempting to develop custom motor hardware or casings and will only use previously developed designs purchased from industry leaders. These motor casings are constructed from ductile (non-fragmenting) materials so in the event of a CATO, energy is dissipated via deforming the material rather than being converted to kinetic energy in the form of shrapnel.	Medium
	Motor case unable to contain operating pressure or pressure spikes	Tightly secure all motor closures and double check that hardware is in place. All team captains and project lead must sign off on whether or not motor components are properly secured according to Aerotech or Cesaroni specifications.	Low
	Motor end closures fail to hold		Low
Fragmentation of motor components from explosion during flight	Motor grain dislodges from liner during flight causing over pressurization of combustion chamber	Adhesive will be used during the mixing process to promote proper adhesion of motor grains to casing. In addition, during all test flights, personnel will be a minimum of 500ft from launch site.	Low

	Cracks in propellant	Propellant mixture is a previously developed and tested recipe known to provide stable operation. The team will visually inspect all motor grains for cracks and de-bonding from the liner walls.	Medium
	Debonding of propellant from liner wall	During the mixing process, the liner will be coated in an adhesive to promote adhesion to liner wall. The team shall inspect the motor for damage from transportation to the test site.	Low
	Gaps between propellant sections and/or nozzle cause increase in burn rate and over pressurization of motor casing	River City Rocketry is not attempting to develop custom motor hardware or casings and will only use previously developed designs purchased from industry leaders. These motor casings are constructed from ductile (non-fragmenting) materials so in the event of a CATO, energy is dissipated via deforming the material rather than being converted to kinetic energy in the form of shrapnel.	Medium

	Motor case unable to contain nominal operating pressure	Tightly secure all motor closures and double check that hardware is in place. All team captains and project lead must sign off on whether or not motor components are properly secured according to Aerotech or Cesaroni specifications.	Low
	Motor end closures fail to hold pressure from hot gasses cause motor exhaust to escape casing, setting fire to rocket.	Flame resistant components will be placed above motor to prevent spreading of flames to rest of vehicle, including critical recovery electronics.	Low
Failure to ignite motor	E-match not inserted into grain core properly	Disconnect all launch control electronics from igniter. Wait a minimum of 60 seconds to ensure motor does not ignite. This is in coherence with the National Association of Rocketry's High Power Rocketry Safety Code.	Low
	Insufficient contact between E-match and propellant grains	Only essential, fully trained, and certified personnel go to pad to inspect E-match.	
Motor ignition occurs prematurely	Igniter was installed prematurely	Proper procedure and necessary checklists will be followed precisely during motor test operations. This includes not installing the igniter into the motor until absolutely necessary.	Low

	Ignition sequence was started prematurely	Proper procedure and necessary checklists will be followed precisely during motor test operations. This includes sounding of an alarm prior to ignition and ensuring no personnel is near the test pad.	Low
	Improper personnel were given command of launch control system	Launch controls will be un-powered until it is time to fire the motor. In addition, only trained and certified personnel will begiven command of the launch control system. If any non-certified team member violates said rule, it will result in immediate dismissal from the team. All team members wishing to attend the test will be required to attend a Safety Briefing the night before. Failure to do so will result in member being barred from attending the event.	Low

Table 35: Hazards of testing and flying solid motor.

8.10.3 Liquid Engine Risk Assessment Analysis

Hazard	Probable Causes	Risk Level and Rationale	Mitigation Approach	Risk of Injury After Mitigation
Rocket deviates from nominal path shortly after rail exit	Incorrect design results in unstable vehicle	Low – design analysis and simulations documented previously have demonstrated that	Internal design reviews and simulations will be conducted to ensure a stable design.	Low

		the vehicle design is stable.		
	Motor misfires resulting in lower exit rail speed and lower stability at rail exit	Medium – as this is the team's first attempt at developing solid rocket motors, there is an elevated risk of a misfire.	The team will be using a tested propellant recipe instead of developing a new one from scratch. The recipe has been flown several times by the team's mentor and has demonstrated reliability. Static test fires will also be conducted to ensure reliability.	Low

8.10.4 Recovery Risk Assessment Analysis

Hazard	Probable Causes	Risk Level and Rationale	Mitigation Approach	Risk of Injury After Mitigation

8.10.5 Payload Risk Assessment Analysis

Hazard	Probable Causes	Risk Level and Rationale	Mitigation Approach	Risk of Injury After Mitigation

8.10.6 Telemetry Risk Assessment Analysis

Hazard	Probable Causes	Risk Level and Rationale	Mitigation Approach	Risk of Injury After Mitigation

8.11 Personal Hazard Analysis

9 Financials

9.1 Budget

The team has attempted to maintain close to the original budget described in PDR. The team budget was broken into sub-teams and travel. Since PDR, each sub-team has maintained well under budget, except for the liquid rocket engine. The cause of the higher than anticipated costs are largely due to the complexity of the project and not fully understanding what the team would need to purchase at the time of PDR. The original PDR budget and the total team expenses so far are shown in Table 36.

Category	PDR Budget	Expenses So Far
General Team	$0	$464

Vehicle, Propulsion, Recovery	$4,331	$2,176
Payload	$4,115	$1,652
Telemetry	$2,600	$716
Liquid Rocket Engine	$12,917	$16,196
Travel	$12,050	$0
Total	$34,813	$21,204

Table 36: PDR budget compared to expenses so far.

While much of the team will likely remain under budget or close to budget, the higher liquid engine costs and the addition of travel costs will likely push the project over budget.

9.2 Income

The team has had a successful year of fundraising and has secured funding from multiple sources.

9.2.1 Funderbolt

In October 2018, the team began a crowd-funding campaign using the Funderbolt service. Funderbolt allowed the team to create a webpage, shown in Figure 92**Error! Reference source not found.**.

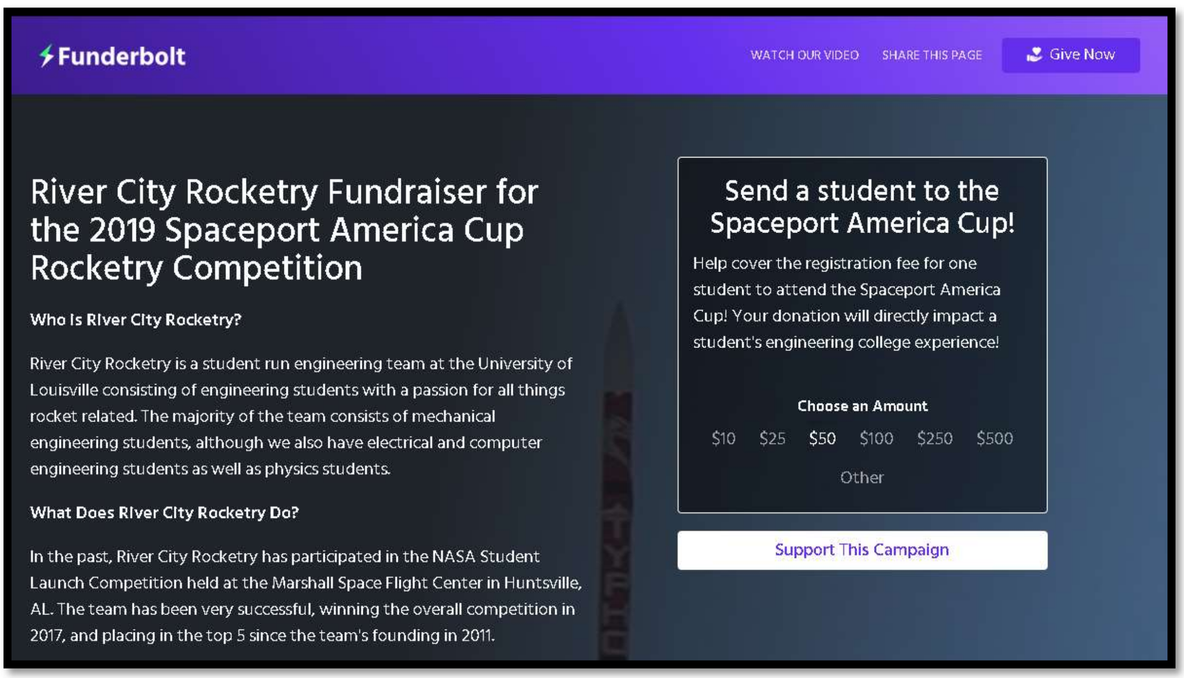


Figure 92: Funderbolt webpage.

The team members could use the service to send emails to their contacts that would allow people to donate a preset or custom amount of money to the team. When a potential donor would receive an email and click the provided link, it would take them to the team member's "advocate page" in which the member could provide a bio and advocate for the team and why the team needed their donations.

The team also received their own personal consultant with Funderbolt that helped set deadlines to send out emails, fundraising goals, etc. The downside of Funderbolt was that they charged 10% of our total funds raised to pay for the website and consulting. The team set a goal of $8,000, which was suggested by Funderbolt and somewhat arbitrary. The team ended up raising just over $6,000 on the Funderbolt platform, plus an additional $1400 via check donations. The team purposefully avoided Funderbolt with larger donations to avoid the 10% charge.

From a managerial standpoint, the Funderbolt service was well worth the 10% fee as it would've been much harder to set our own deadlines and goals and keep up with what each team member had done. Funderbolt allowed the captains to track how many emails each team member had sent, how many contacts they had, and how much they had raised. The team roster with the aforementioned info is shown in Figure 93.

Name	Advocate Page?	Contacts	Emails Scheduled	Emails Sent	Dollars Raised	Donors Inspired	Thank Yous Sent	Actions
Matthew Meier	Yes	22	0	29	$750	6	6	Preview Page
Jarett Coyle	Yes	6	0	3	$600	5	9	Preview Page
Justin Johnson	Yes	44	0	43	$570	9	0	Preview Page
Chase Renner	Yes	17	0	18	$395	9	0	Preview Page
Kaylee Norvell	Yes	24	0	38	$355	5	6	Preview Page
Matthew Cosgrove	Yes	22	0	49	$335	6	6	Preview Page
Eric Lewis	Yes	8	0	6	$300	2	0	Preview Page
Alexander Cooper	Yes	20	0	17	$275	4	3	Preview Page
Samuel Williams	Yes	31	0	30	$225	3	3	Preview Page
Paulo Ribenboim	Yes	37	0	85	$175	4	0	Preview Page

Showing 1 to 10 of 29 entries

Previous 1 2 3 Next

Figure 93: Funderbolt team roster.

9.2.2 NASA Kentucky

In August, NASA Kentucky released the 2018-2019 NASA Space Grant Consortium request for proposals. NASA KY offered Team Fellowships that could receive up to $15,000 if selected for funding. The team has received funding from the NASA Space Grant in the past, and used the previous year's proposal for reference when writing the new one. Dr. Lian was helpful in writing the proposal as he knows the Associate Director at NASA KY, Jacob Owen. On the first business day of 2019, NASA KY announced that we had been selected to receive funding from the space grant. The successful proposal is stored on the team SharePoint for future teams to reference.

9.2.3 Dr. Clinton Kelly

A longtime supporter of River City Rocketry, Dr. Kelly has continued and increased his support this year by committing to a donation totaling $25,000. Dr. Kelly has been invited to design reviews and maintains his position on the Speed School Industrial Board of Advisors. This season, the team captains have not communicated much with Dr. Kelly, only having met with him once for about an hour. Dr. Lian maintains a good relationship with Dr. Kelly and future teams should make sure this is still the case.

9.2.4 Speed School

Dean Collins donated $5,000 to River City Rocketry without any action from the team. Heather Mann facilitated transfer of funds to the correct account and should be the person future teams contact if looking for funding from Speed School.

9.2.5 Mechanical Engineering Department

The team captains met with Dr. Kevin Murphy, the department chair of the mechanical engineering department, and asked that the department donate to River City Rocketry. Dr. Murphy committed "at least $1,000." Dr. Murphy does not respond to emails asking him for money so future teams should walk into is office if you want money.

9.2.6 Raytheon MathMovesU

As the team has done in the past, we participated in an outreach event sponsored by Raytheon in October. The event was largely put on by River City Rocketry, as Speed School handled the logistics but River City Rocketry handled the activities entirely. Future teams should participate in this event but should take safety more seriously and prepare for it by having cones or caution tape to prevent children from being too close to model rockets or running after rockets that have landed. Additionally, future teams should print a megaphone and air horn to the event as speaking to the 50ish kids participating was difficult on the launch field, and a ballistic model rocket almost landed on someone and we had no way to communicate that they needed to be heads up at that moment. While Raytheon was the sponsor of the event, team captains did not communicate to any Raytheon employee and instead communicated to James (Ben) Craig of Speed School. Ben handled ordering supplies and setting up other logistics. Ben also facilitated getting $1,000 donated to River City Rocketry for running the event. This money was technically from Speed School but really it came from Raytheon as they sponsored the entire event and Speed School gave us a cut.

9.2.7 Summary

A summary of all funds raised this season is shown in Table 37.

Source	**Amount**
Dr. Kelly	$25,000
NASA KY Space Grant	$15,000
Speed School	$5,000
Mechanical Engineering Department	$1,000
Raytheon MathMovesU	$1,000
Funderbolt/Donations	$7,115
Total	**$54,115**

Table 37: Team income summary.

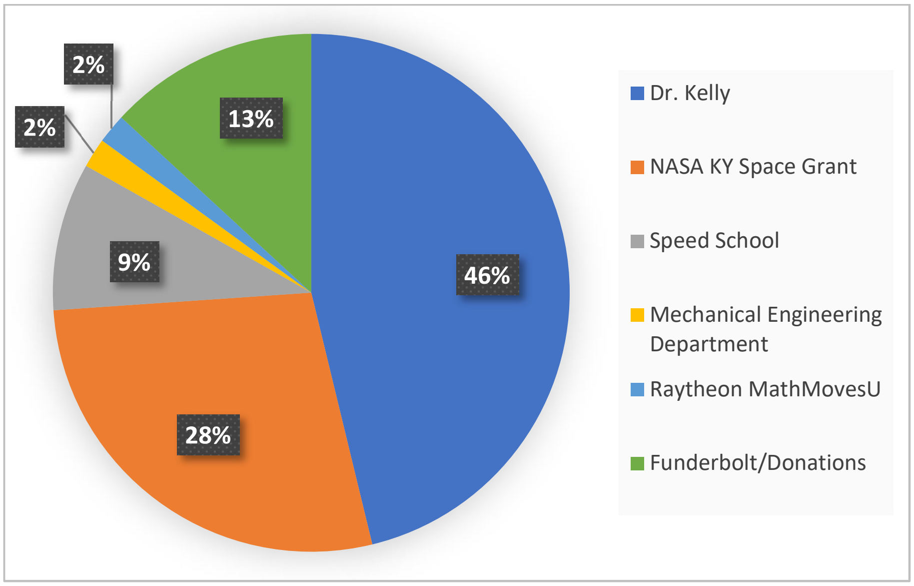


Figure 94: Team income by percentage.