Rose Rocketry

Project Silverstein
Critical Design Review

Rose-Hulman Institute of Technology

January 6, 2022

5500 Wabash Ave, Terre Haute, IN 47803

Table of Acronyms

Acronym	Definition
3D	Three Dimensional
ADC	Analog-To-Digital Converter
AGL	Above Ground Level
APCP	Ammonium Perchlorate Composite Propellant
BIC	Branam Innovation Center
CAD	Computer Aided Design
CDR	Critical Design Review
CG	Center of Gravity
COTS	Commercial Off-The-Shelf
CP	Center of Pressure
FAA	Federal Aviation Administration
FIRST	For Inspiration and Recognition of Science and Technology
FMEA	Failure Modes and Effects Analysis
FRC	FIRST Robotics Competition
FRR	Flight Readiness Review
GPIO	General-Purpose Input/Output
GPS	Global Positioning System
GUI	Graphical User Interface
HPR	High Powered Rocketry
I2C	Inter-Integrated Circuit
IMU	Inertial Measurement Unit
KIC	Kremer Innovation Center
LRR	Launch Readiness Review
NAR	National Association of Rocketry
NASA	National Aeronautics and Space Administration
OTFR	One Time Funding Request
PDR	Preliminary Design Review
PLAR	Post-Launch Assessment Review
PPE	Personal Protection Equipment
RF	Radio Frequency
RHIT	Rose-Hulman Institute of Technology
RR-SL	Rose Rocketry - Student Launch
RSO	Range Safety Officer
SDR	Software Defined Radio
SGA	Student Government Association
SL	Student Launch
SPI	Serial Peripheral Interface
STEM	Science Technology Engineering and Math
TRA	Tripoli Rocket Association
USB	Universal Serial Bus
USLI	University Student Launch Initiative

1. Summary of CDR Report

1.1 Team Summary

Table 1.1: Team Summary and Mentor Contact Information

Team Name	Rose Rocketry - Student Launch (RR-SL)
Mailing Address	5500 Wabash Ave, Terre Haute, IN 47803
Mentor Name	Gary Kawabata
Mentor Contact	rocketguy9914@gmail.com
Mentor Certifications	NAR 89092; TRA 3019; level 3
NAR/TRA Sections	Indiana Rocketry Group Tripoli #132 NAR Section #711
Hours Spent on CDR	900
Primary Location and Date	SL Launch Field at Bragg Farm Toney, Alabama April 23, 2022
Secondary Location and Date	Indiana Rocketry Club Field Pence, Indiana April 9, 2022

1.2 Launch Vehicle Summary

Table 1.2: Launch Vehicle Summary

Official Target Apogee	5000 ft.
Final Motor Choice	Cesaroni Technology Inc. L2375WT-P
Recovery System	Main: SkyAngle Cert3 XXL Drogue: SkyAngle Cert3 Drogue
Rail Size	12' 1515 Rail

Table 1.3: Vehicle Size and Mass Summary

Vehicle Length	162 in.
Vehicle Airframe Nominal Diameter	5.5 in.
Vehicle Wet Mass	48.1 lbm
Vehicle Dry Mass	38.9 lbm

Vehicle Subsystem	Mass (lbm)	Length (in)
Payload	6.08	30
Recovery	14.62	57.25
Altitude Assurance	13.39	42.75
Booster	14.01	32

1.3 Payload Summary

1.3.1 Payload Title

This year's payload will be named "The RHIT Stuff," inspired by Tom Wolfe's *The Right Stuff*.

1.3.2 Payload Experiment

The payload experiment's goal is to autonomously locate the rocket. The objective is to be robust enough for interplanetary travel, thus we will use two methods with minimal required hardware. The RF system uses directional transmissions from the ground station to determine the position of the rocket. The IMU System uses two accelerometers to continuously measure acceleration starting from a reference position. All computation will be done using a flight computer. Both techniques will be used to calculate a most probable flight path. Additionally, GPS and a separate altimeter will be used and will have their data transmitted to the ground station.

2. Changes Made Since PDR Report

2.1 Vehicle Criteria

Changes to the vehicle sizing have been made since the PDR report. The final vehicle design has a reduced diameter of 5.5in. from the initial 6in. The overall length of the final design is larger at 13.5ft. from the initial 12ft. Although a 12ft. vehicle length was the predicted maximum, additional length was afforded using exact launch vehicle mass and thrust parameters in verifying rail exit velocity.
The airframe design has changed from the leading design alternative described in the PDR report in order to accomodate altitude assurance hardware. The final design uses a hybrid of cylindrical monocoque and semi-monocoque airframe. The altitude assurance module will be 3D printed, which presents a potential weak point in the structure of the airframe. Because of this, in the final design, internal structural elements join the monocoque airframe sections above and below the Altitude Assurance Module, while using the printed section as a stiffening web. As a result, the final design incorporates both monocoque and semi-monocoque airframe sections.

The final vehicle recovery design has changed since the PDR report. The main parachute selected is a SkyAngle Cert3 XXL parachute, from the initial SkyAngle Cert3 L. The final drogue parachute selected is a SkyAngle Cert3 Drogue parachute, from the initial Rocketman 7ft. Pro Experimental Drogue Parachute.

2.2 Payload Criteria

The changes in the payload from the PDR include a shock-mount, selection of a specific method of location with the RF system, and the inclusion of the altimeter and GPS transmitter. We determined these changes were necessary after extensive analysis, research, and feedback from the Preliminary Design Review. For example, the shock mount was included to ensure that the payload IMU stays within the acceleration that it can measure. For the RF system, we simplified the method of locating for the distance to a simple calculation of the power received by the antenna on the rocket. After consulting with faculty with experience in the field, it was determined the other options proposed were not feasible to implement. The ground station components of the RF system necessary have also been specified.

2.3 Project Plan

Since the PDR, the team has secured additional sources of project funding. The team was granted a One Time Funding Request (OTFR) from our Student Government Association (SGA) for $12,000. This funding has been used to purchase team startup tools, such as 3D printers, screw drivers, nut drivers, tape measures, in addition to consumable resources to manufacture the subscale launch vehicle. The team also received an anonymous $1000 donation. This funding is being reserved as a source of emergency funding should any budget issues occur throughout the season.

The team has also updated the project timeline with additional deadlines and launch dates. Indiana Rocketry Incorporated has graciously been working with the team to organize launch dates in addition to their scheduled monthly launches. These additional launch dates allow the team more dates for testing and project deadlines have been added to make use of these additional launches.

2.4 Safety Officer

Due to unforeseen circumstances, the team's previous Safety Officer, Donald Hau, will be unable to fulfill the duties of the position. A newly elected Safety Officer, Ben Graham, will now take over Donald's responsibilities. Ben has multiple years of experience in high-powered rocketry as a hobbyist, and has recently acquired his NAR Level 2 High Power Certification.

3. Vehicle Criteria

3.1 Mission Statement and Mission Success Criteria

The objective of Project Silverstein is to design and fabricate a launch vehicle and payload that will ascend to a target apogee with high confidence and report its landing location to our ground station. This is to be done to industry standards of reliability and will support the development of a team history of successful mission execution.

A successful mission for 2021-22 NASA USLI meets all of the following criteria:

- All members abide by all safety regulations put into effect.
- The launch vehicle is launched on a safe, stable, and predictable trajectory.
- The payload can robustly locate the launch vehicle upon descent.
- The launch vehicle is recovered in a state suitable for reuse.

3.2 Overview of Vehicle Systems

Following the requirements outlined in the 2022 USLI handbook, the launch vehicle was divided into subsystems to perform system-level design. These subsystems have been selected to present individual objectives and provide a focus for an exploration of the potential vehicle design space. Table 3.1 shows a summary of these vehicle systems.

Table 3.1: Vehicle Systems Overview

Vehicle Subsystem	Objective
Airframe	Provide sufficient structural housing for the vehicle components considering spatial and mass constraints.
Aerodynamics	Support the vehicle in safe ascent considering constraints imposed on the propulsion system.
Deployment	Allow for separation of the vehicle to eject payload and/or recovery subsystems.
Altitude Assurance	Support the launch vehicle in achieving an apogee as close to the target as possible.
Propulsion	Provides the necessary thrust for the rocket to reach a desirable range of apogees while considering structural load to the vehicle.
Recovery	Assure that the rocket returns safely to the ground without sustaining significant damage to itself and the payload during descent.

3.3 Vehicle Systems Final Alternatives and Justifications

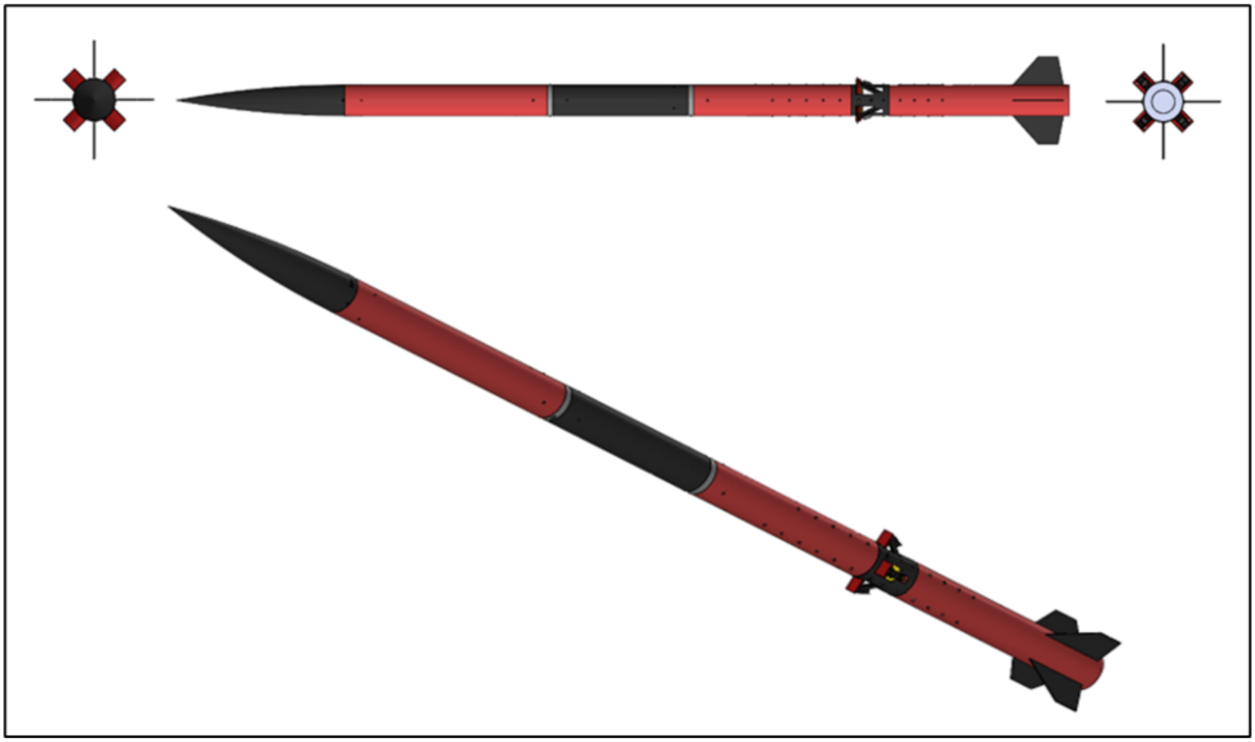


Figure 3.1 Full-Scale CAD Model: Three View

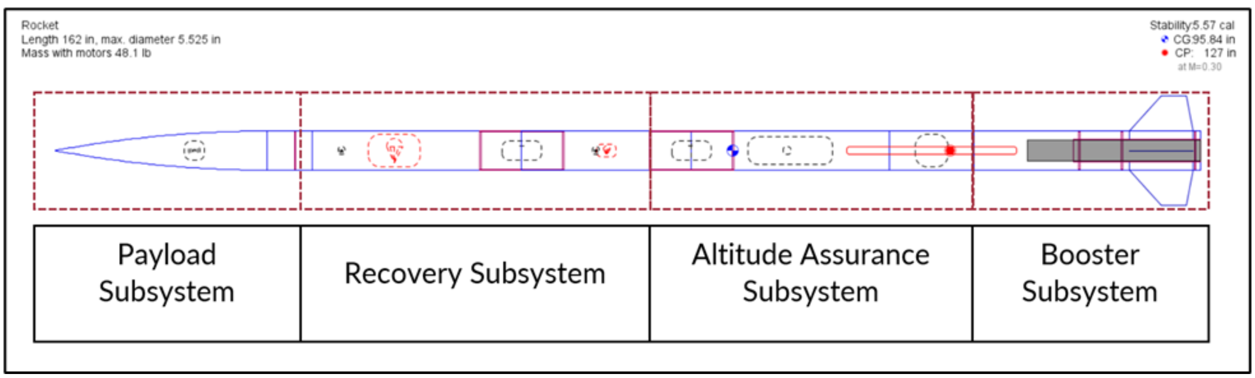


Figure 3.2: Final OpenRocket Schematic

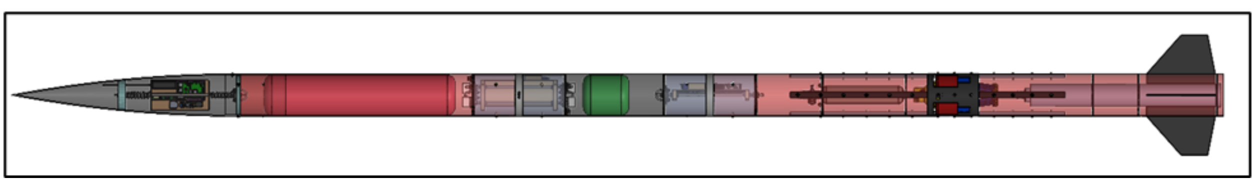


Figure 3.2 Full-Scale CAD Model: Transparent

3.3.1 Airframe Structural Design: Hybrid Choice

For the airframe structural design, cylindrical monocoque, cylindrical semi-monocoque, cylindrical sandwich structure, and conical monocoque airframes were considered. The final design alternative chosen for the Airframe Subsystem geometry is to use a cylindrical monocoque airframe of a large-scale length and diameter. The cylindrical monocoque structure was chosen primarily due to availability and simplicity. Because airframes of this type are commonly available from vendors, and are purpose-built for launch vehicle airframes, variability is eliminated in construction methods. Therefore, we believe their usage in Project Silverstein would provide the mission with the highest possible outcome of success [1-3]. The team has decided to use a large-scale sizing for the airframe, since many complications of subsystem development are minimized by way of designing around more relaxed spatial constraints. This was deemed to be an acceptable tradeoff to drag concerns, added project costs, and motor impulse limits.

3.3.2 Aerodynamics: Nose Cone and Aft Fins

For the aerodynamics design, a nose cone, blunted body, aft fin set, and no fin set were considered. The final design alternatives chosen for the Aerodynamics Subsystem are use of a full nose cone and aft fins. The nose cone allows for a significant improvement to drag performance, and the decreased spatial constraints of using a blunted body are not considered substantial to the design of the vehicle. Placing fins to the aft of the vehicle allows for increased control over the stability of the vehicle in flight, and the constraints placed on ballast mass by the competition rules limit the stability that can be achieved without these aerodynamic features [4].

3.3.3 Deployment: Fixed Payload and Dual-Point Separation

For the deployment design, a deploying payload, fixed payload, dual-point recovery, and single-point recovery were considered. The team has decided to use dual-point separation for the deployment system. The decision to use this type of deployment was simple: the team has no experience using the chute release mechanism required for single-point separation.

3.3.4 Altitude Assurance: Drag-Producing Devices

For the altitude assurance design, passive mass adjustment, jettisonable ballast mass, thrust modulation, and drag-producing devices were considered. The final design alternative chosen for the altitude assurance subsystem is utilizing drag flaps (dubbed “Rose Petals”) to actively control deceleration. Specifically, the avionics system monitors its flight path through a combination of accelerometer and barometer data, and compares its predicted apogee to the target apogee for the mission. If its projected apogee at any point in the flight exceeds that of the target, it will increase its aerodynamic drag to correct the discrepancy by deploying the Rose Petals. This method has been chosen as the leading alternative due to a large range of control authority in a variety of conditions, relative to the other alternatives considered.

3.3.5 Propulsion: Fast-Burn Reloadable Motor System

For the propulsion design, a disposable motor system, reloadable motor system, slow-burn motor, and fast-burn motor were considered. The final design alternative chosen for the Propulsion Subsystem is to use a reloadable motor system with a fast burn rate. The leading motor brand and designation is the Cesaroni Technology Inc. L2375WT-P. This motor's high average thrust allows the rocket to achieve sufficient rail velocity. Additionally, the Cesaroni L2375WT distinguishes itself from the other leading alternative, the Aerotech L2200G, by its much more consistent thrust curve, meaning that onboard electronics can more easily detect and compensate for the acceleration applied by the motor.

Table 3.2: Final Alternatives Summary

Vehicle Subsystem	Final Alternative(s)
Airframe	Monocoque Airframe, Semi-Monocoque Subsection
Aerodynamics	Nose Cone
	Aft Fins
Deployment	Fixed Payload
	Staged Charges, Dual Separation
Altitude Assurance	Drag-Producing Devices
Propulsion	Reloadable Motor System
	Fast Burn Motor

3.4 Energetic Device Identification and Points of Separation

Energetic devices within the vehicle are located fore and aft of the Avionics Bay in the form of black powder recovery charges and within the booster section of the vehicle in the form of the final selected motor. The vehicle will separate into three sections between the Payload and Recovery Subsystems and between the recovery and Altitude Assurance Subsystems.

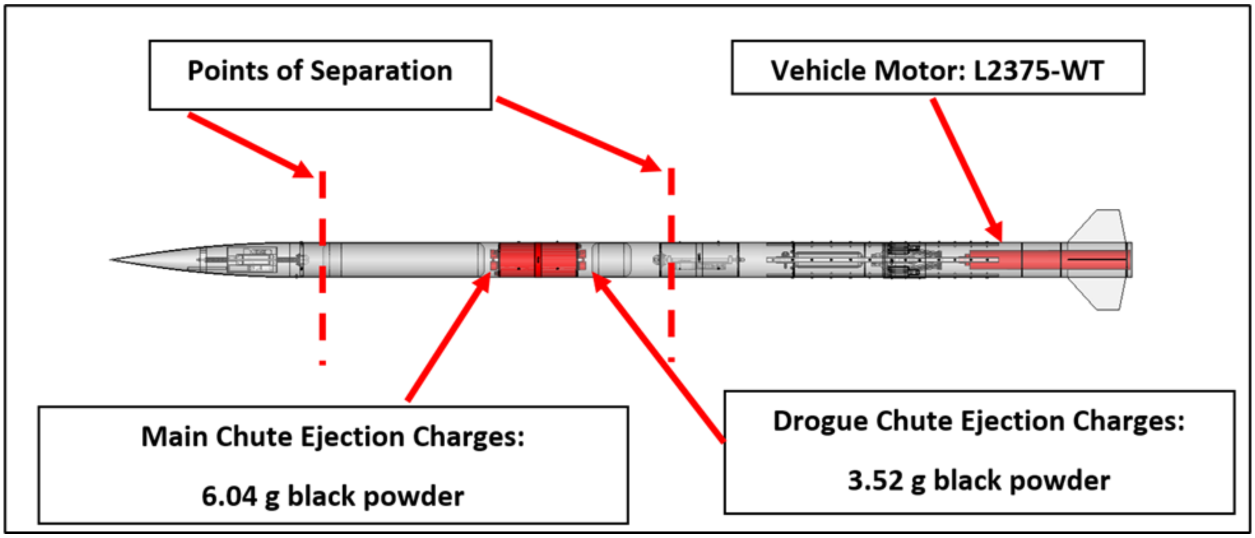


Figure 3.4 Energetic Device Identifications and Points of Separation

3.5 Vehicle Subsystem and Component Review

The following section will exhibit each subsystem of the vehicle and review the design at a component level. The subsystems will be reviewed from the forward end of the vehicle to the aft end of the vehicle. This is done to illustrate where each subsystem lies within the vehicle and illustrate the interactions with the neighboring subsystems.

3.5.1 Payload Subsystem Review

The payload will be placed in the nosecone of the launch vehicle and have a mass of 6.1 lbm. It will be shock mounted to reduce the acceleration of the electronics, thereby minimizing the error by time-averaging the response mechanically. This both avoids saturating the IMU and decreases the risk of misrepresenting the duration or magnitude of shock loads due to a limited sampling frequency. The payload electronics, antenna, and battery will be mounted to a sled which is free to move. Two threaded rods guide the payload sled between a cylindrical lower bulkhead and wedge-shaped upper bulkhead, termed the Payload Bulkhead and Wedge respectively. Screws through the skin of the nose cone in the Payload Bulkhead keep the system in place from underneath, while the forward wedge bulkhead is compression-fit.

The Payload Sled consists of an Upper Plate, Lower Plate, and Payload Backplane, which will all be made of MDF to reduce mass. Aluminum tubes will be placed between the upper and lower plates to directly resist the compressive load of the suspension springs. The wedge will be made out of Extruded Polystyrene (XPS) foam while the Payload Bulkhead will be made from ½ in. nylon 66. The decision to use XPS foam for the wedge was to increase the shock-absorbing properties of the payload bay. The Payload Bulkhead, on the other hand, is structural and is directly loaded at recovery events. All electronics on the Payload Sled will be screwed to the sled with the exception of the FUNcube Pro+, which will be mounted using cable ties. The locations of components were selected in order to simplify wiring - GPIO components are near the GPIO pins on the Raspberry Pi for easy wiring. The battery is placed on the opposite side of the Payload Sled to balance out the weight. The battery is mounted securely using a 3D printed holder, which will be screwed to the sled. The USB ports for the FUNcubePro+ and Raspberry Pi 4 are not at the same height, which will be adjusted out using a 3D printed case for the Pi. Electronics will be armed using a stationary, side-mounted screw switch. Wires connecting from the payload sled to the arming switch will be coiled and sleeved to minimize risk of damage in-flight. As seen in Figure 3.5, two springs on either side of the payload sled (where the electronics are mounted) will provide shock-mounting. The springs used are 302 stainless steel compression springs. They have a rating of 10 lbs/in and a maximum compression of 1.57 inches at a max load of 19.34 lbs. The payload springs are sized according to the expected shock load they must absorb. By equating Newton's 2nd Law of Motion and Hooke's Law for springs, and knowing their compression distance, stiffness, and the approximate weight of the payload (.45 kg), the acceleration that it can absorb from rest is 381 m/s^2. This is 1.5 times our expected acceleration during deployment of 250 m/s^2.

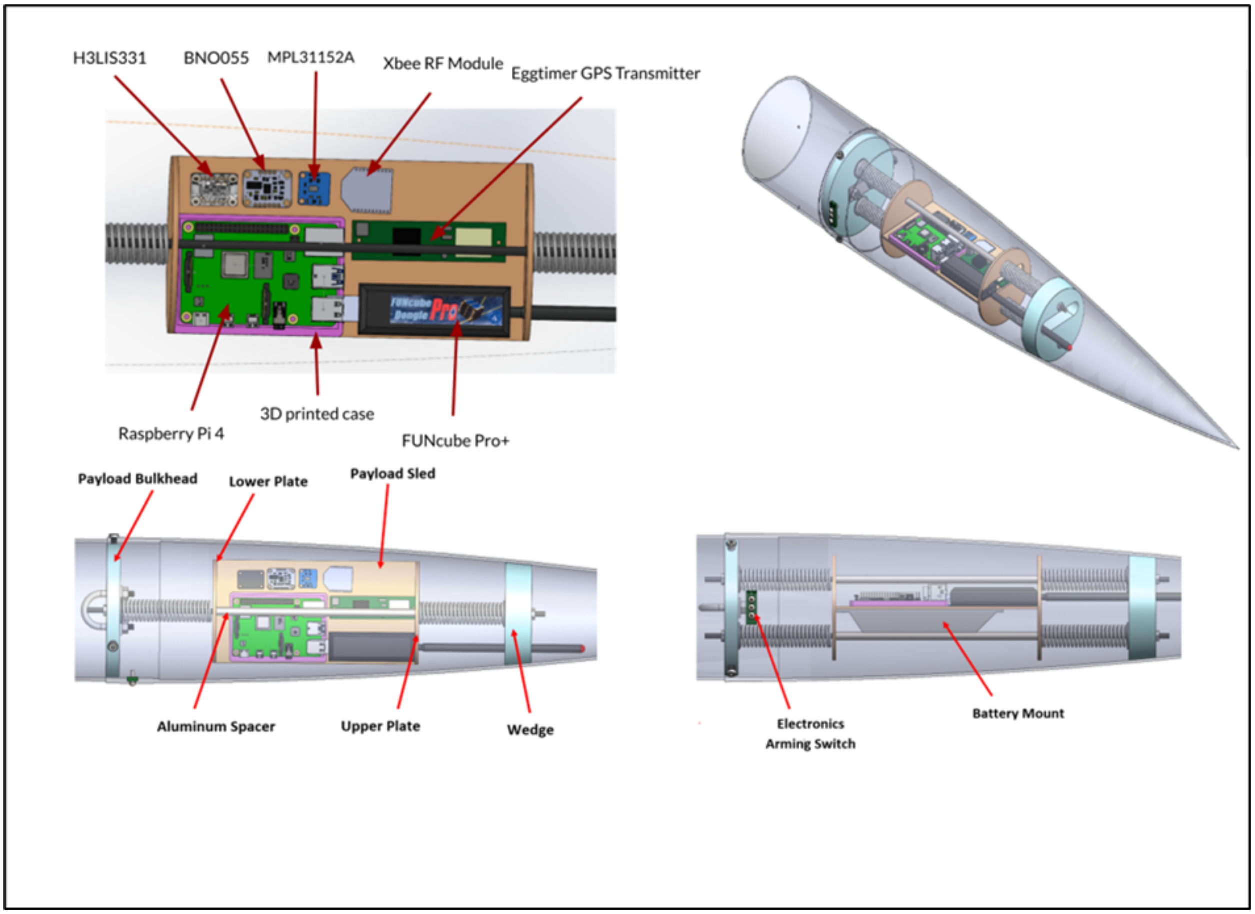


Figure 3.5: Payload Bay Component Diagram

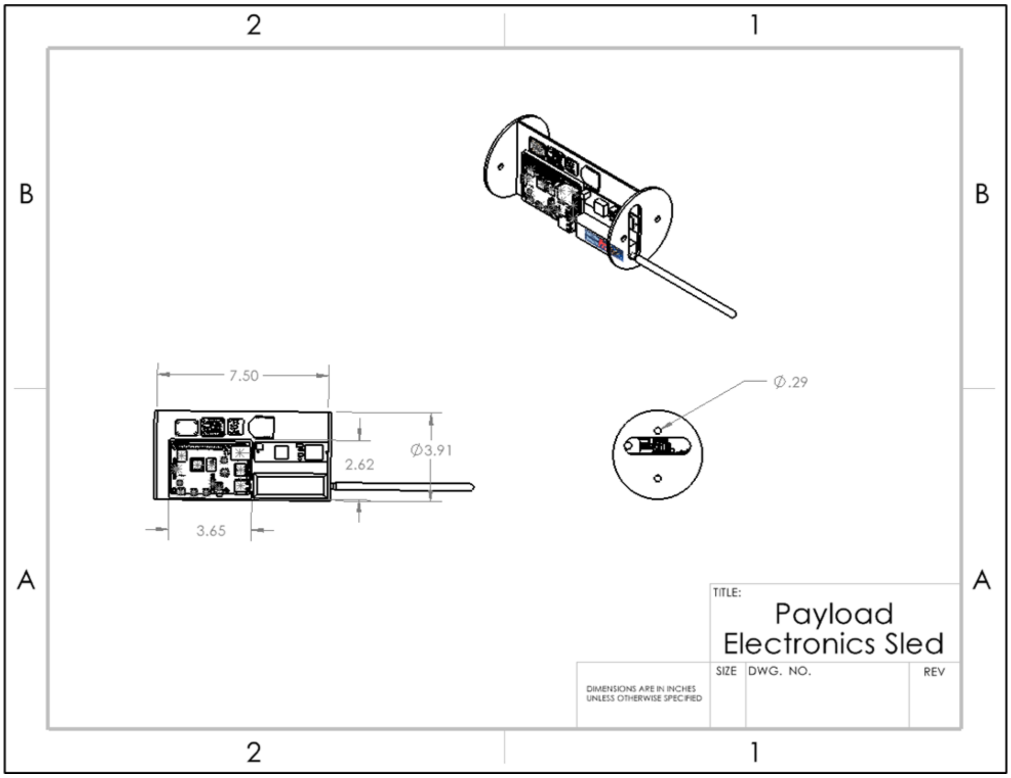


Figure 3.6: Payload Electronics Sled Dimensional Drawing

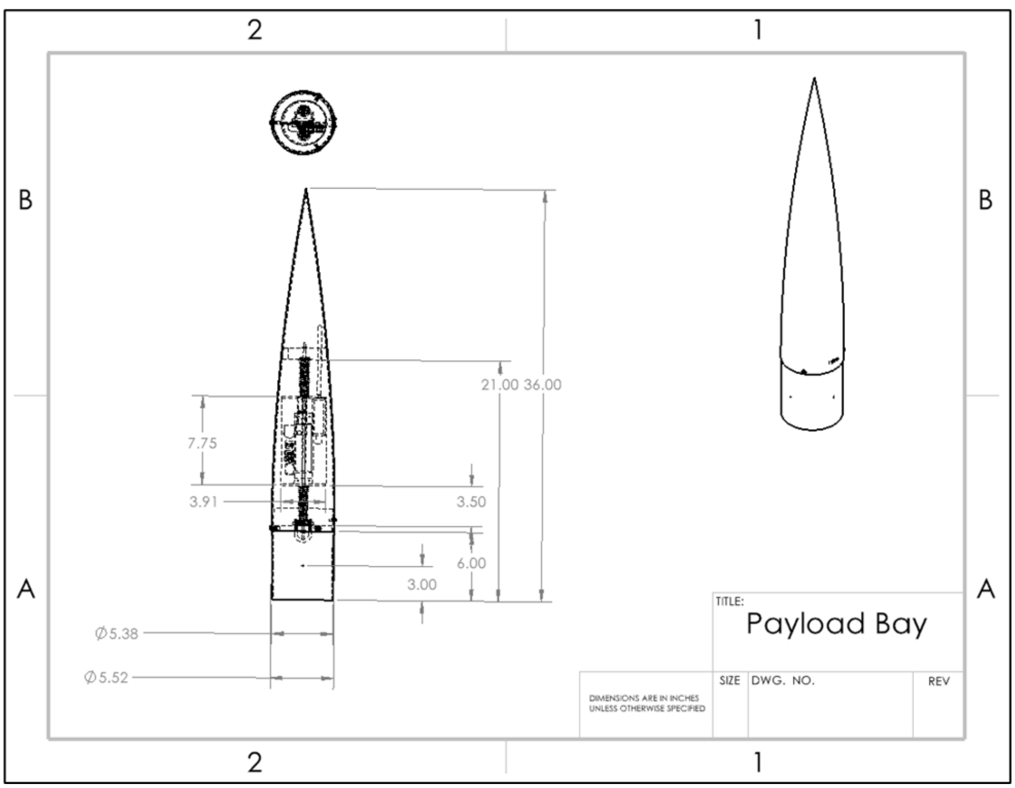


Figure 3.7: Payload Bay Dimensional Drawing

3.5.1.1 Structural Analysis of Payload Bulkhead

Finite Element Analysis was performed on the Payload Bulkhead, as it is a critical component in integrity of both the payload and recovery subsystems. In order to efficiently use available space, this bulkhead will see direct shock loading from recovery events at the U-bolt plate. Figure 3.7 shows the equivalent Von Mises stress results obtained by static structural analysis. A force of 3000 N is applied at the U-bolt plate in the aft direction while compression-only supports are located within the three mounting holes on the side of the bulkhead. The force applied is based on the proof load of the recovery harness, scaled proportionally to the mass of the payload section versus the remainder of the rocket.

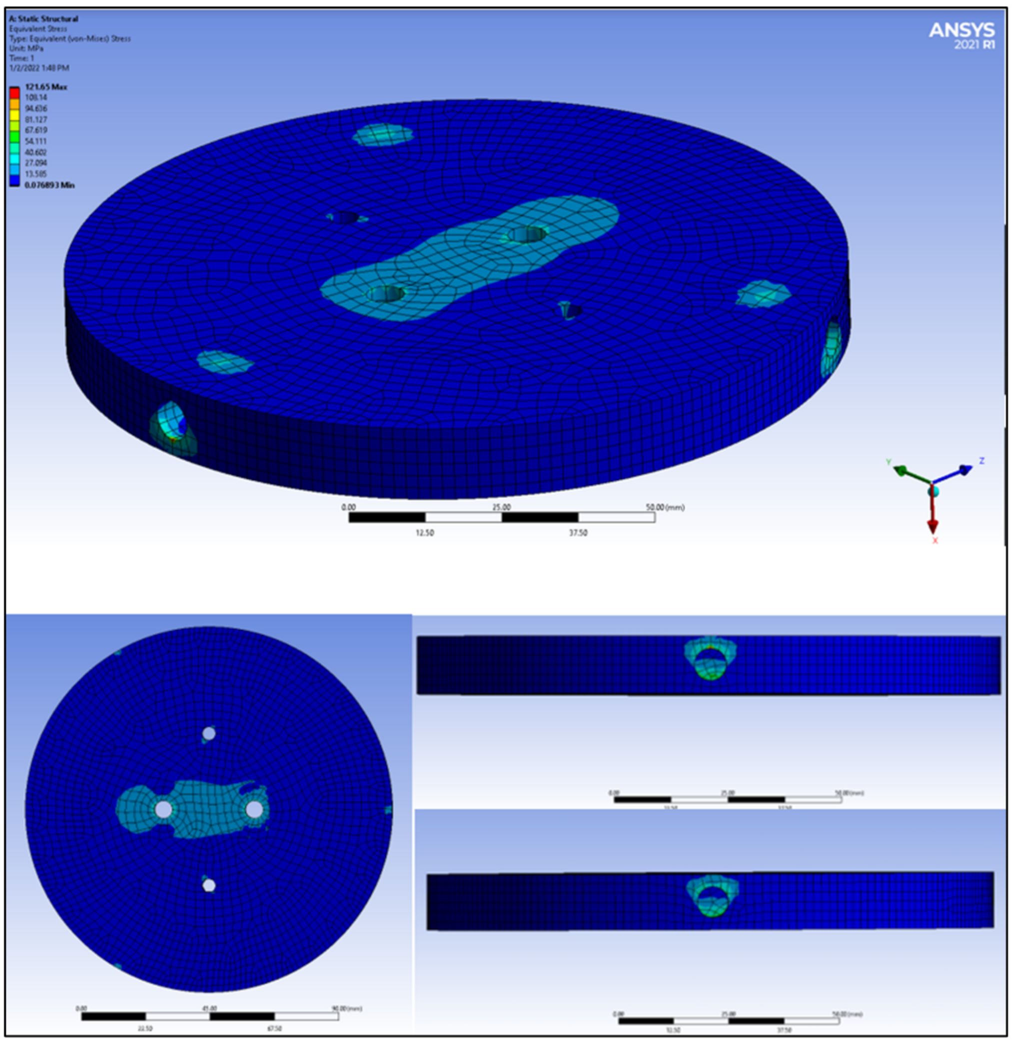


Figure 3.8: Structural Simulation of Payload Bulkhead

Analysis yielded a maximum load of 121.65 MPa at one of the mounting holes. However, this load is a simulation artifact caused by a point constraint. This constraint was used to mathematically stabilize the problem and is not physical to the payload structure.

Discarding this point stress and evaluating the stress of nearby nodes, most stress occurs at the mounting holes on the side, however it is far below the yield strength of 3D printed nylon at 66 MPa. From these results, we conclude that our material choice and dimensioning are sufficient for providing structural integrity during recovery events.

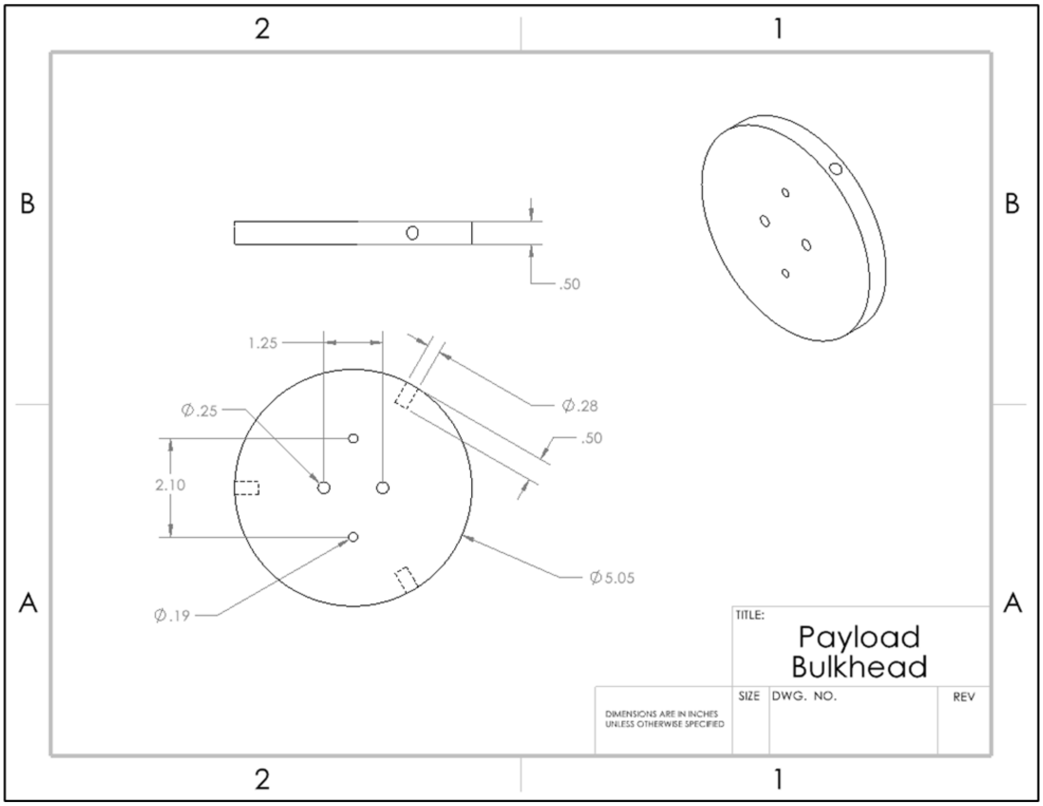


Figure 3.9: Payload Bulkhead Diagram

3.5.2 Recovery Subsystem Review

Table 3.3: Final Recovery Subsystem Alternatives

Component	Final Alternative(s)
Altimeters	MissileWorks RRC3 and Altus Metrum EasyMini
Recovery System	Drogue Deploy at Apogee then Main Deploy
Recovery Harness	Kevlar Harness
Power Delivery	7.4v Lithium Polymer Battery
Main Chute	SkyAngle Cert3 XXL
Drogue Chute	SkyAngle Cert3 Drogue

The Recovery Subsystem consists of the Avionics Bay, Main Chute, Drogue Chute, and the sections of the airframe that house these components. It will have a mass of 14.6 lbm. The parachutes selected for the mission are shown in Table 3.3. These chutes are tethered via ⅜ in. quick links to the airframe by 7/16 in. tubular kevlar recovery harnesses: 35 feet tethering the drogue section to the Altitude Assurance Controls Bay and 25 feet tethering the main section to the Payload Bulkhead. Sizing of the airframe sections were based on the parachute manufacturer's recommended sizing [5].

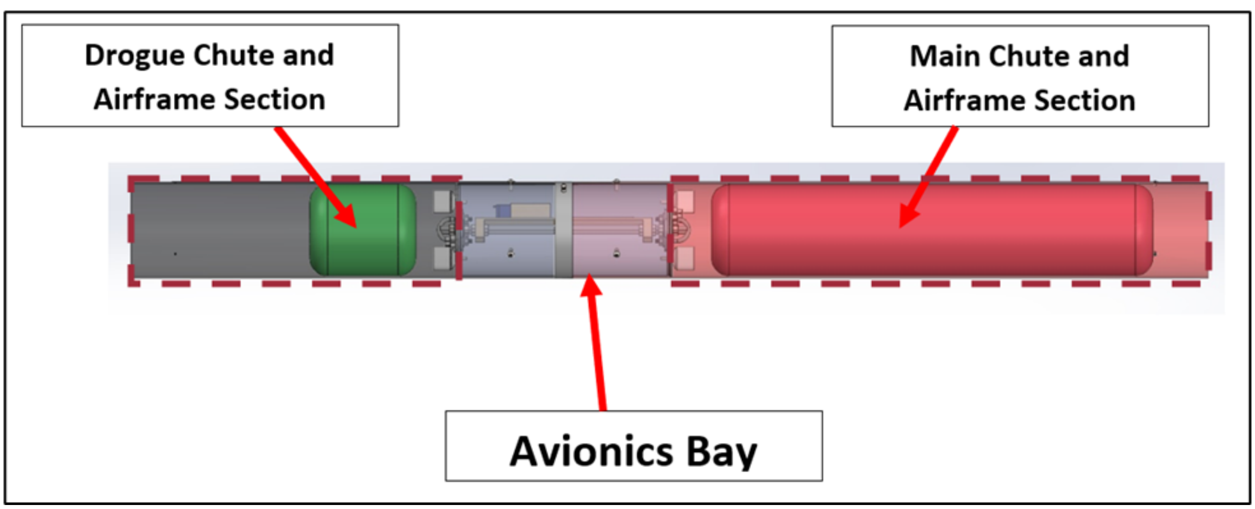


Figure 3.10: Recovery Sections

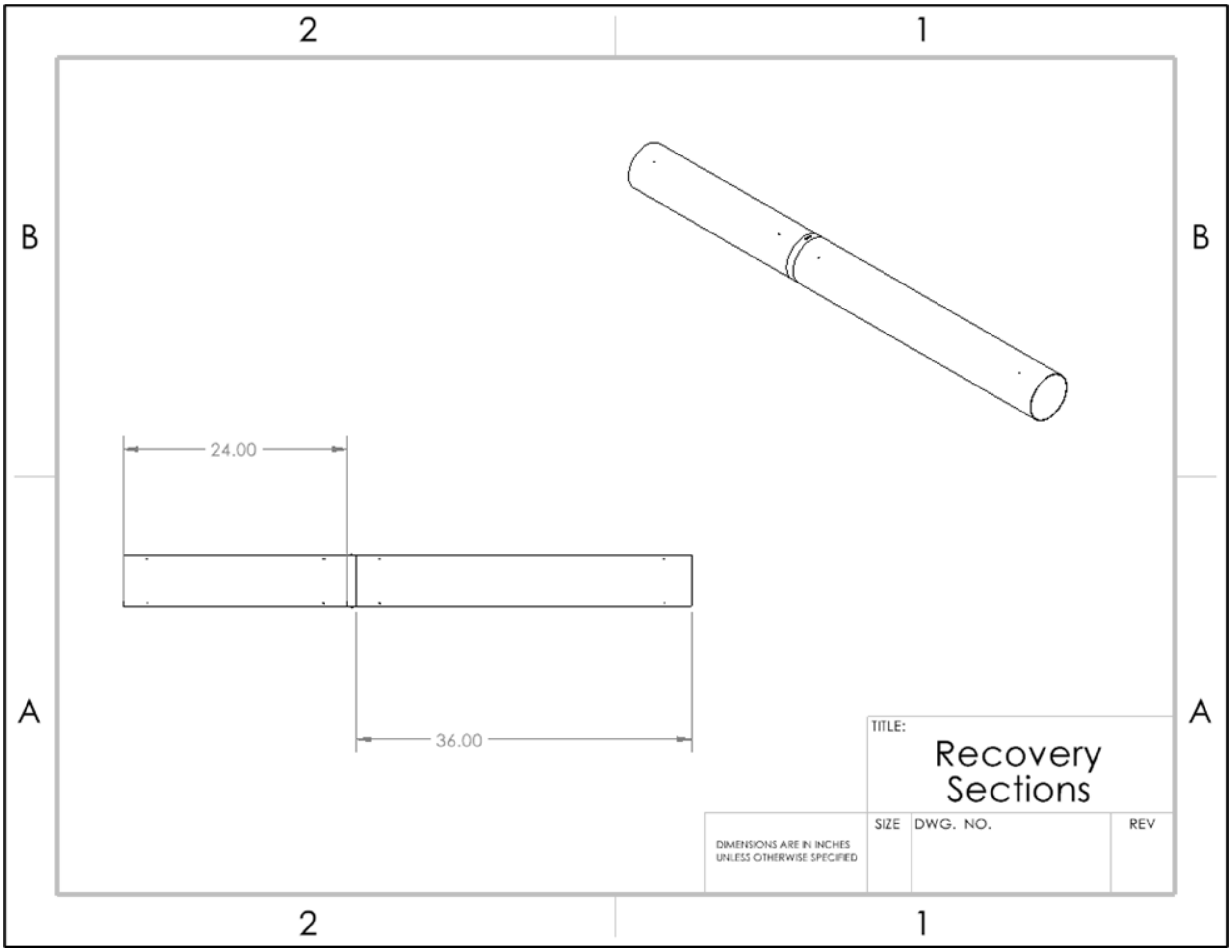


Figure 3.11: Recovery Sections Dimensional Drawing

3.5.2.1 Avionics Bay

The Avionics Bay consists of recovery electronics, an electronics sled, two externally accessible arming switches, four charge wells, two U-bolts mounted on two load distributor plates. The electronics are mounted on a plywood sled that is secured to two ¼ in. threaded rods in tension. These threaded rods also provide support from one end to the other during respective recovery events. The recovery forces directly apply to two ¼ in. U-bolts which clamp between the coupler bulkhead and a distributor plate. On each bulkhead, two charge wells direct the blast toward the chutes and away from the bay. The Avionics Bay will be electronically armed using two externally accessible arming switches. Remove Before Flight tags will be pinned to the side of the vehicle (not depicted here).

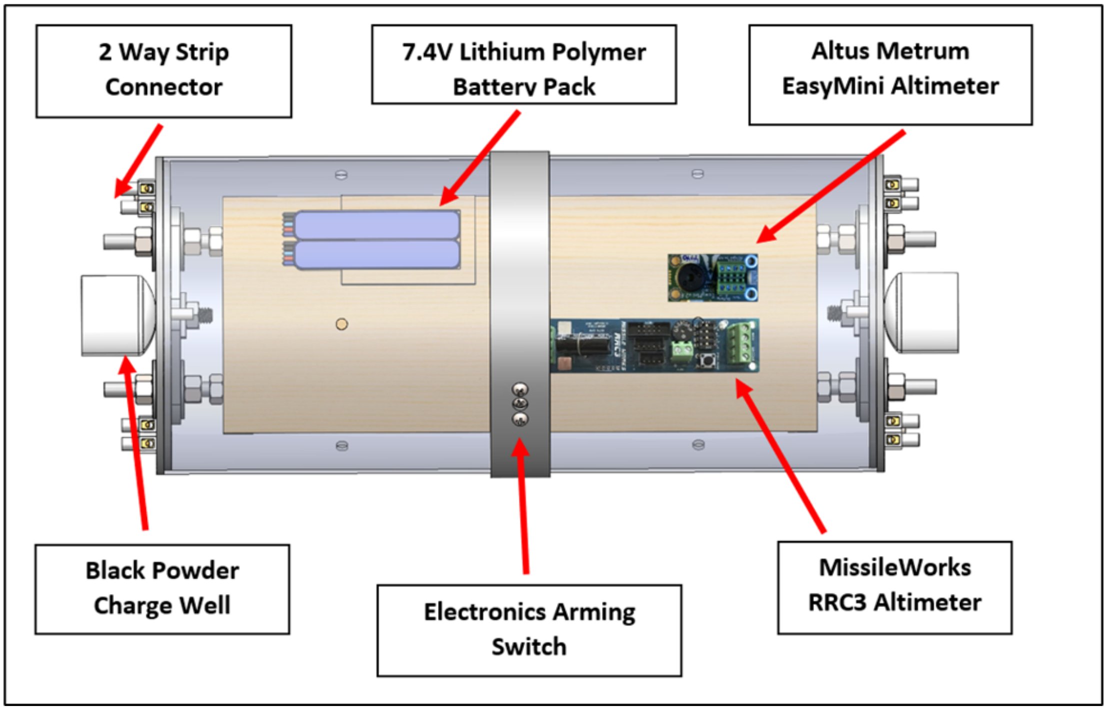


Figure 3.12: Avionics Bay Components

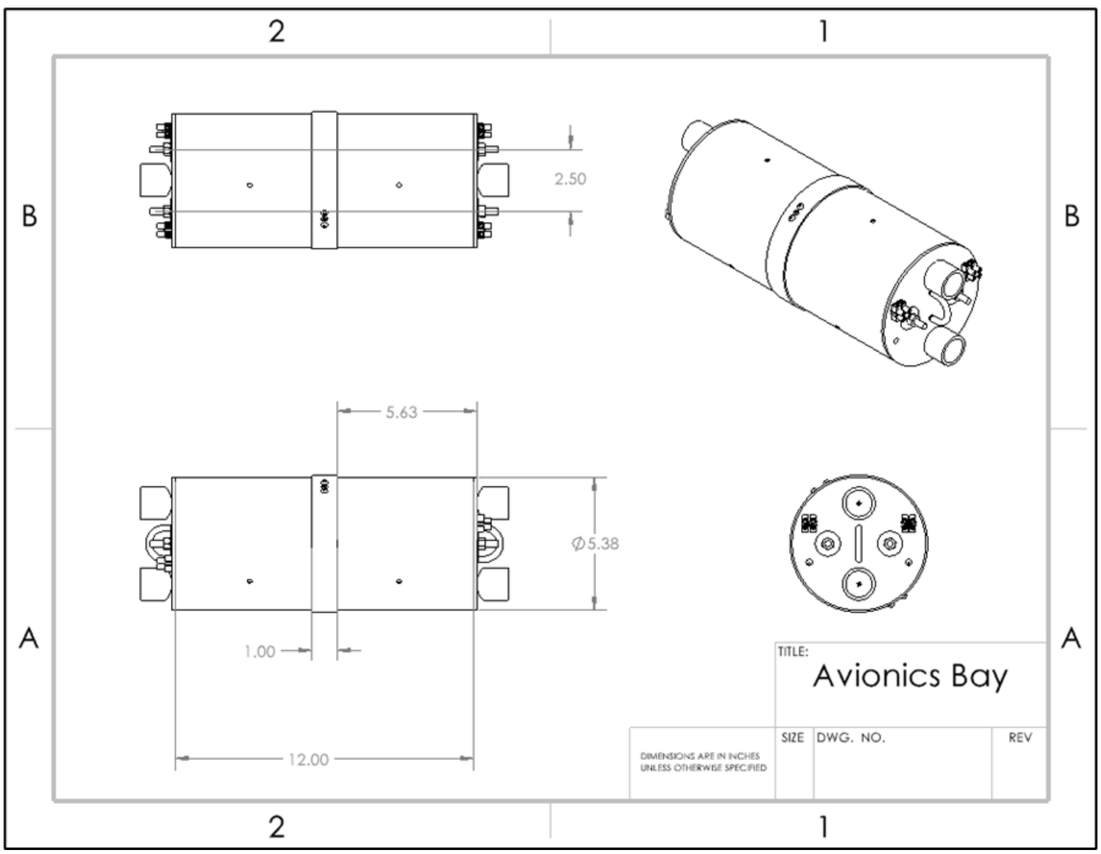


Figure 3.13: Avionics Bay Dimensional Drawing

3.5.2.2 Recovery Electronics

The Avionics Bay electronics used for recovery are summarized in Table 3.4. The final altimeters chosen for the mission are the MissileWorks RRC3 Altimeter and the Altus Metrum EasyMini Altimeter. These altimeters provide the benefit of team experience working with these altimeters, which supports reliability of the mission. They are also low-cost which puts less strain on the project budget. Redundancy in the avionics design exists in the dissimilarly redundant altimeters which control one drogue chute and one main chute deployment each. The RRC3 Altimeter will fire the primary charges while the EasyMini Altimeter will fire the backup charges. These altimeters are electrically isolated, being armed with their own switches and supplied their own power from two dedicated lithium polymer batteries. The redundant electronics and wiring of the avionics bay can be seen in Figure 3.14.

Table 3.4: Final Recovery Electronics Selection

Recovery Electronics
MissileWorks RRC3 Altimeter
Altus Metrum EasyMini Altimeter
Electronic Screw Switch (x2)
7.4v Lithium Polymer Battery (x2)

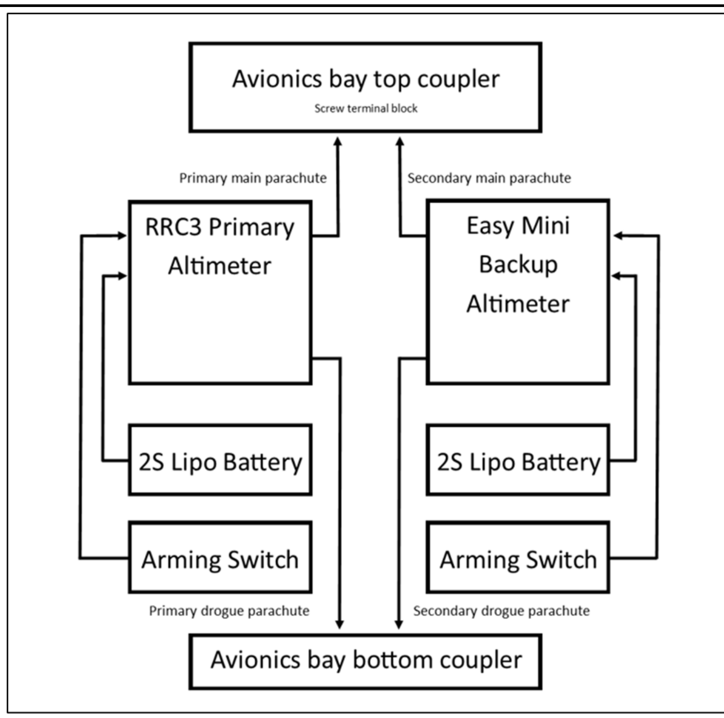


Figure 3.14: Avionics Wiring Diagram

3.5.2.3 Structural Analysis of Coupler Bulkheads

A static structural analysis was performed on the forward bulkhead, as it will see the highest shock load during recovery events. The recovery load of 23.5 kN in our analysis was based on the proof load of our recovery harness. The rationale behind this is that our recovery harnesses will never see loading that large during the mission and so designing the rest of the recovery system to that strength will ensure structural integrity of all recovery components. The initial analysis considered a bulkhead modeled as an orthotropic material comparable to G10 fiberglass and supported by 1 in. washers under a load from the U-bolt plate.

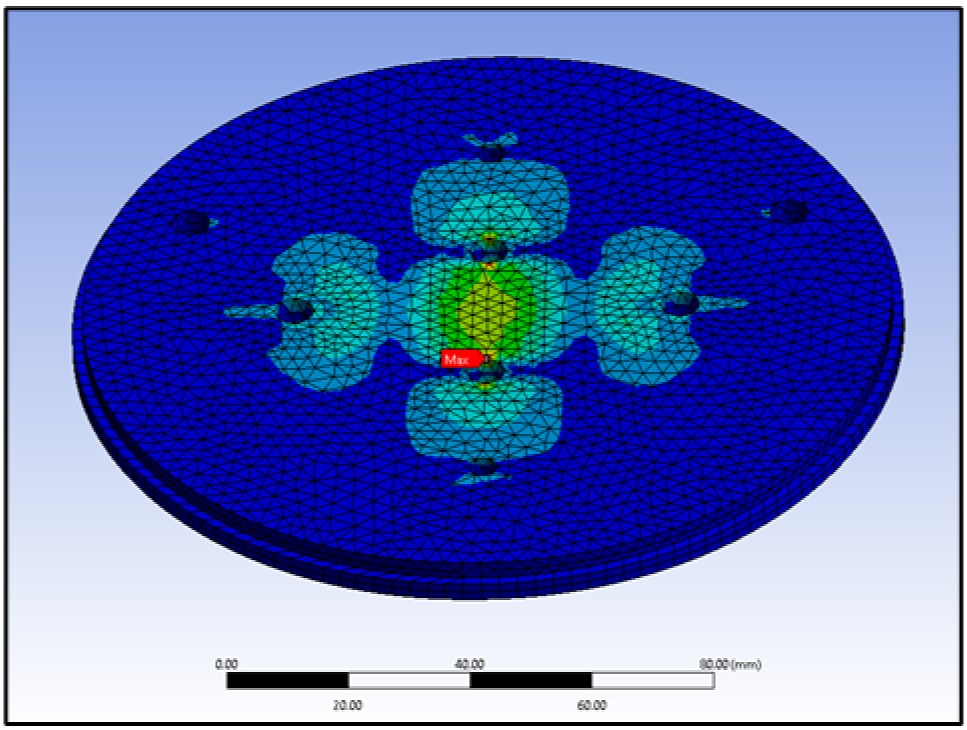


Figure 3.15: Structural Simulation of Initial Coupler Bulkhead Design

The analysis showed a maximum load of 814 MPa concentrated toward the edges of the U-bolt holes. Since this load would surpass the yield strength of the fiberglass, this design was not acceptable. A redesign of the bulkhead employed the use of a 4mm thick aluminum distributor plate to spread the load across the bulkhead. The analysis was repeated modeling the distributor plate as an isotropic material, highlighted in Figure 3.16.

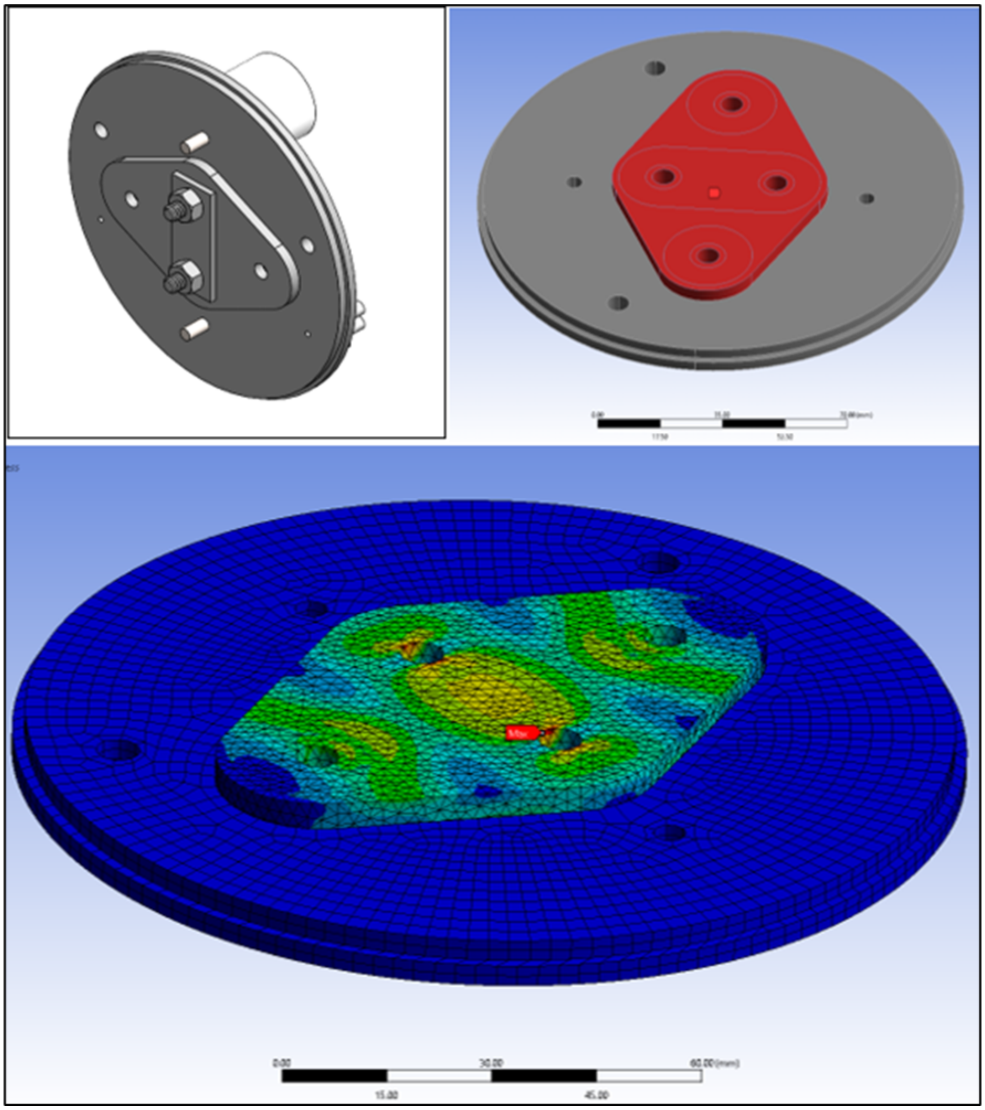


Figure 3.16: Structural Simulation of Coupler Bulkhead with Distributor Plate

The analysis determined a maximum load of 188 MPa concentrated at the edges of the U-bolt holes on the distributor plate. Since the maximum loading for both the aluminum distributor plate and the fiberglass bulkhead is less than either of their yield strengths, this design was determined to be acceptable.

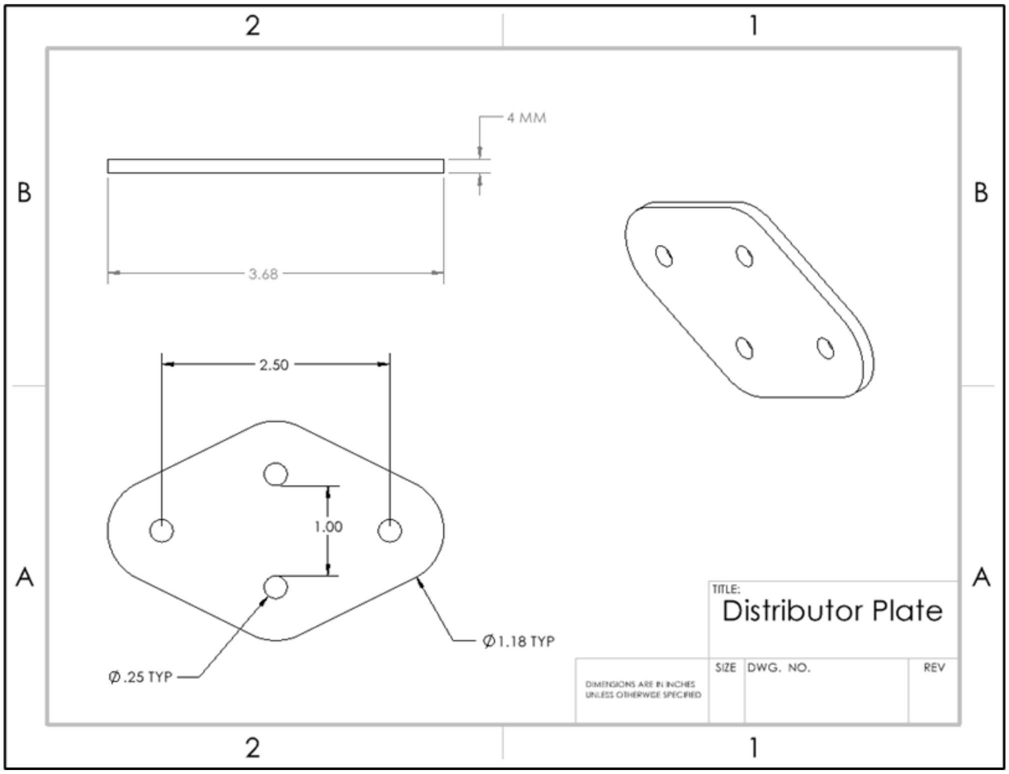


Figure 3.17: Distributor Plate Dimensional Drawing

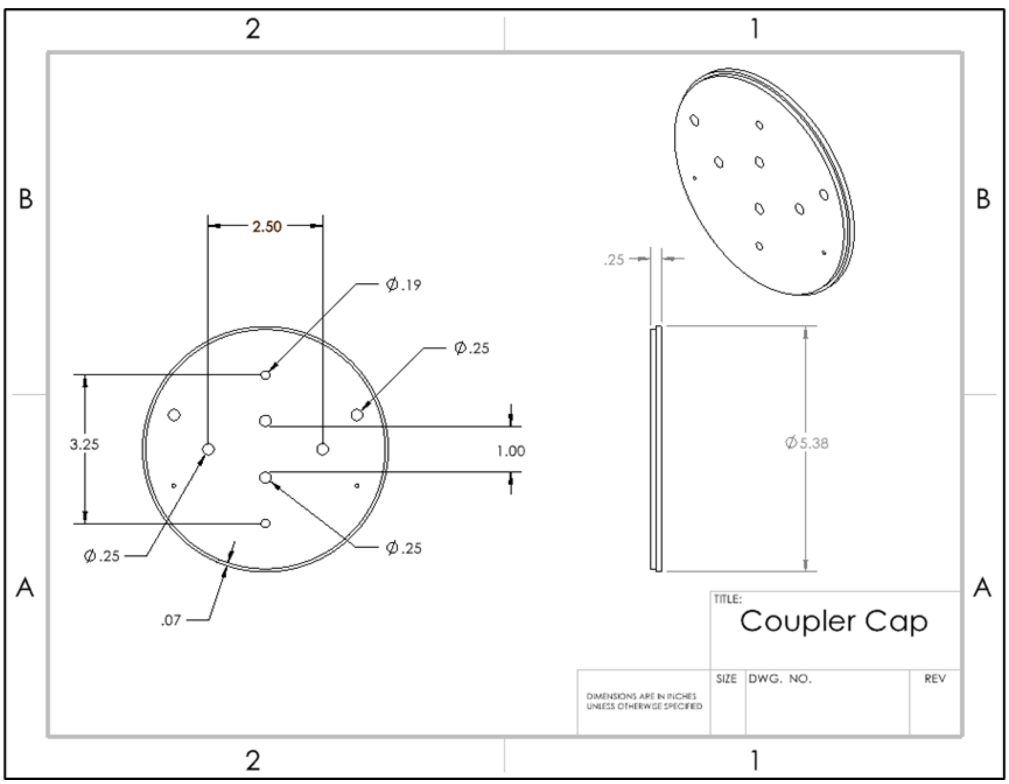


Figure 3.18: Coupler Cap Dimensional Drawing

3.5.2.4 Shear Pin and Ejection Charge Sizing

The deployment subsystem of the rocket was designed around the use of 4-40 nylon shear pins due to easy availability of tooling and well-documented use in high-power rocketry. The shear strength of the pins was determined by the following:

$$A_{pin} = \pi R_{minor}^{2}$$

$$A_{pin} = \pi (0.04025\, in)^{2} = 0.00509\, in^{2}$$

$$F_{shear} = A_{pin} \tau_{nylon}$$

$$F_{shear} = (0.00509\, in^{2})(10\,000\, psi) = 50.9\, lbf$$

The most destructive failure mode that is preventable by shear pins is early deployment during burn or just after burnout. While in a conventional rocket the main cause of this failure mode would be air pressure differential or drag force, the altitude assurance subsystem design presents the additional risk of pneumatics leaks causing the drogue compartment to become pressurized. Therefore, the shear pins will be sized to contain a full failure of the pneumatics system without deploying the parachute.

The worst case of a pneumatics failure results in the reservoir's air being vented solely to the control area and drogue bay above it for a total length of 24 inches, plus the 0.42L of the tank. The differential pressure may be calculated by Boyle's Law:

$$P_{tube} V_{tube} = P_{tank} V_{tank}$$

$$A_{cross-section} = \pi (2.69\, in)^{2} = 22.7\, in^{2}$$

$$P_{tube} = (150\, psi)(0.42L)/(22.7 in^{2} * 24 in + 0.42L)$$

$$P_{tube} = 6.74 psi$$

The force may be calculated, and therefore the minimum number of shear pins:

$$F = P_{tube} A_{cross-section}$$

$$F = 6.74 psi * 22.7 in^{2}$$

$$F = 153\, lbf$$

$$n = F/F_{shear} = 153\, lbf/50.9\, lbf = 3.01$$

Therefore, 4 shear pins must be used to guarantee that the parachute does not eject accidentally; this value will be carried over to the forward section to ease assembly procedures. From this, a minimum bulkhead pressure can be determined:

$$P_{bulkhead} = 4F_{shear} / A_{cross-section}$$

$$P_{bulkhead} = 4\,(50.9\ lbf) / (22.7\ in^2) = 8.97\ psi$$

From the NASA High Powered Video Series [6], we can estimate the amount of black powder required to eject each parachute from the volume of section:

$$V_{M/D} = A_{cross-section}\, L_{M/D}$$

$$V_M = (22.7\ in^2)\,(24\ in) = 545\ in^3$$

$$V_D = (22.7 in^2)(12\ in) = 272\ in^3$$

$$n_{BP} = (P_{bulkhead} V_{M/D}) / (RT_{combustion,BP})$$

$$n_{BP,M} = [(8.97\ psi)(545\ in^3)] / [(266\ in\ lbf/lbmR)(3307R)]\ (454\ grams/1lbm) = 2.52g$$

$$n_{BP,D} = [(8.97\ psi)(272\ in^3)] / [(266\ in\ lbf/lbmR)(3307R)]\ (454\ grams/1lbm) = 1.26g$$

Additionally, this calculation assumes that all of the black powder will be burned completely. However, team experience has shown that this is rarely the case. As a result, both estimates will be doubled under the assumption that around 50% of the powder will burn. Therefore, the final charge amounts will be 5.04g for the main and 2.52g for the drogue.

3.5.3 Altitude Assurance Subsystem Review

3.5.3.1 Overview

The Altitude Assurance system consists of pneumatically actuated, 3D printed flaps that deploy from a 3D printed housing bolted to aluminum rails. It will have a mass of 13.4 lbm. The aluminum rail structure bolts directly to fiberglass sections above and below the system, and is strengthened in torsion by a 3D printed nylon fairing. The system uses pneumatic actuation, which was chosen for power, rapid motion, and design flexibility. The drawback to this choice was primarily increased weight, which remained well within what was allowed for the system. Compressed air will be stored in a COTS paintball tank, which was chosen due to its low weight, low cost, and extremely high pressure rating. An integrated regulator and pop-off valve will ensure working pressure is well-controlled and meets competition requirements. A solenoid attached to the Altitude Assurance computer operates a single dual-action piston located in the center of the petal assembly. This piston actuates the flaps through linkages which move the flaps outside of the housing into the airstream around the rocket. The design is radially symmetrical and 3D printed out of nylon for strength and manufacturability. M5 steel bolts provide the pin structure and the aluminum rails provide a rigid skeleton so that the rocket airframe remains rigid and structurally sound.

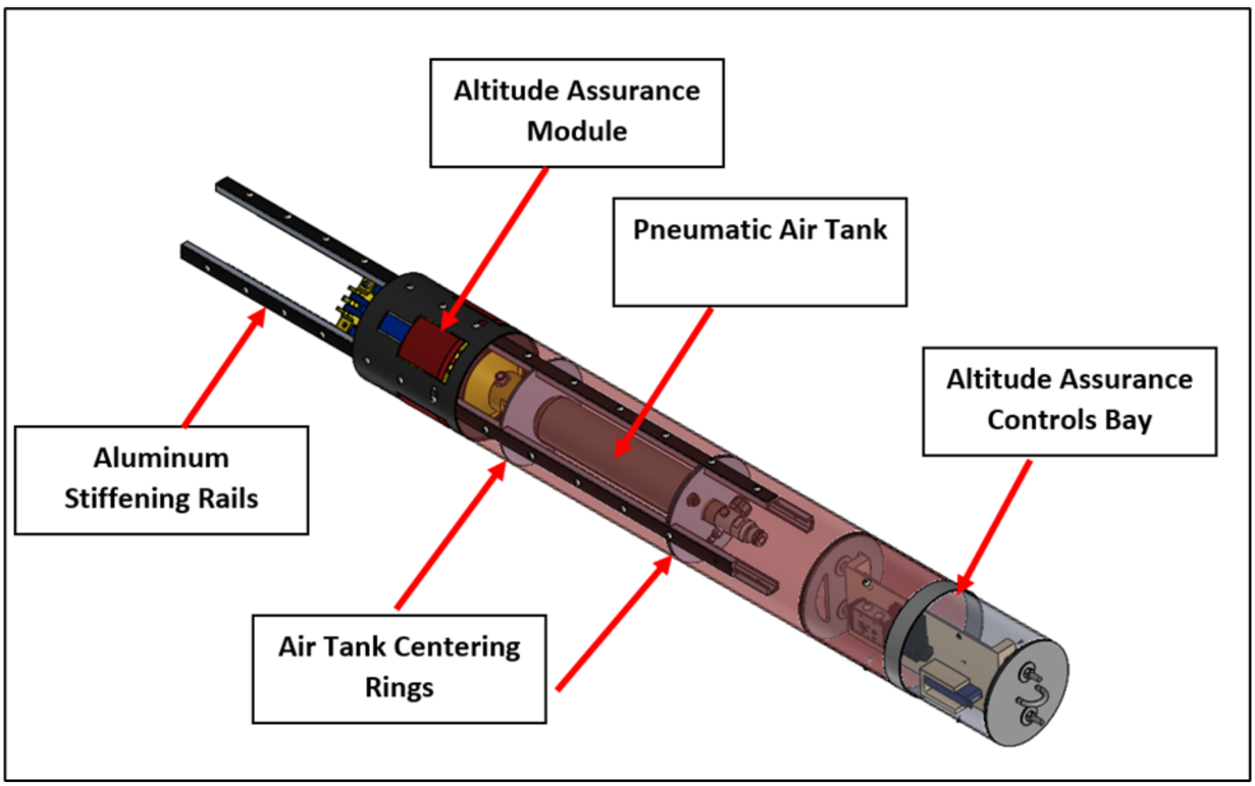


Figure 3.19: Altitude Assurance Subsystem Components

3.5.3.2 Simulated Actuation Forces

In order to design the geometry of the Altitude Assurance actuation mechanism, a simulation was performed using MATLAB which used rigid-body statics to determine the force required from the piston to actuate the flaps under a maximum aerodynamic loading. Once the geometry was finalized, this simulation was then expanded to inspect the loading across all components.

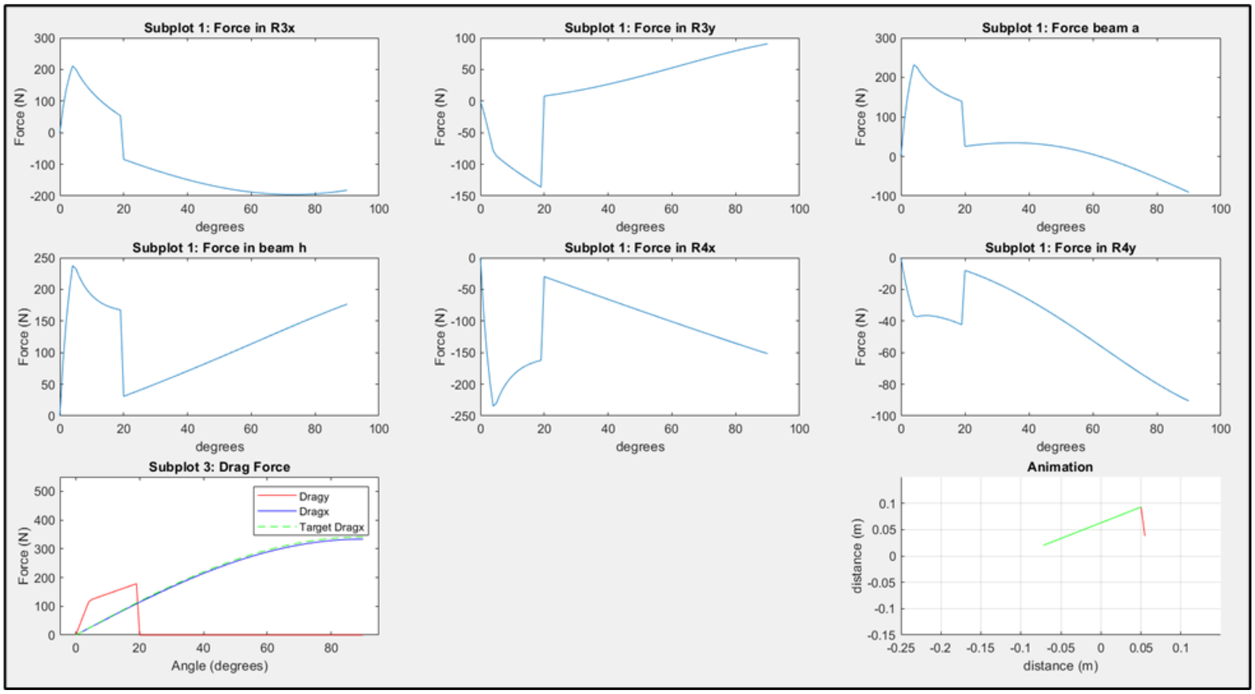


Figure 3.20: Simulated loading on Altitude Assurance Components

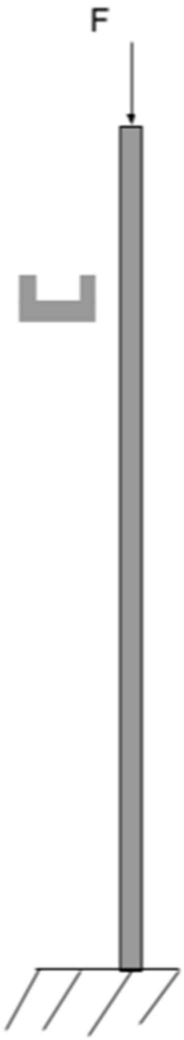
Table 3.5: Maximum Loading on Structural Altitude Assurance Components

Structural Altitude Assurance Component	Maximum Load (N)	Maximum Load with Safety Factor (N)
Linkage	250	1000
Linkage Mount	325	1300
Rose Petal	75	300

A maximum aerodynamic force on the all four flaps of 340N was determined based on the aerodynamic drag required for the vehicle to lower its apogee by 1000ft. From this, the force on each critical linkage was determined and a safety factor of 4.0 was applied to validate the design.

3.5.3.3 Aluminum U-channel Design

Standard 6063 Aluminum U-channels of 8ft length will be purchased and cut to length. See Figure 3.21: Load Calculations for AL Channels for strength analysis. These channels failed by buckling in the subscale test, so that is the strength calculation we used to evaluate the rails.



Failure by Buckling:

$F = n \pi^2 E I / L^2$

F = allowable load (N)

n = factor for the end conditions = 4

E = Modulus of Elasticity (Pa (N/m^2)) = 68.9 GPa = 68.9*10^9 Pa

L = length of column (m) = 0.210 m

I = Moment of Inertia (m^4) = 909.6 mm^4 = 909.6 e-12 m^4

Allowable Load = $n \pi^2 E I / L^2$ = 56900 N per channel

Expected Load = 2800N / 4 = 700 N per channel

FoS = 80.1

Failure in Compression (Shear):

Stress (sigma) = F/A

F = Force (N) = 700N per channel

A = Area (m^2) = 83.3 mm^2 = 83.3 e-6 m^2

Expected Stress = F/A = 7.00 MPa

Allowable Stress = Shear Strength/cos(45) = 214 MPa

FoS = 30.7

Figure 3.21: Load Calculations for AL Channels

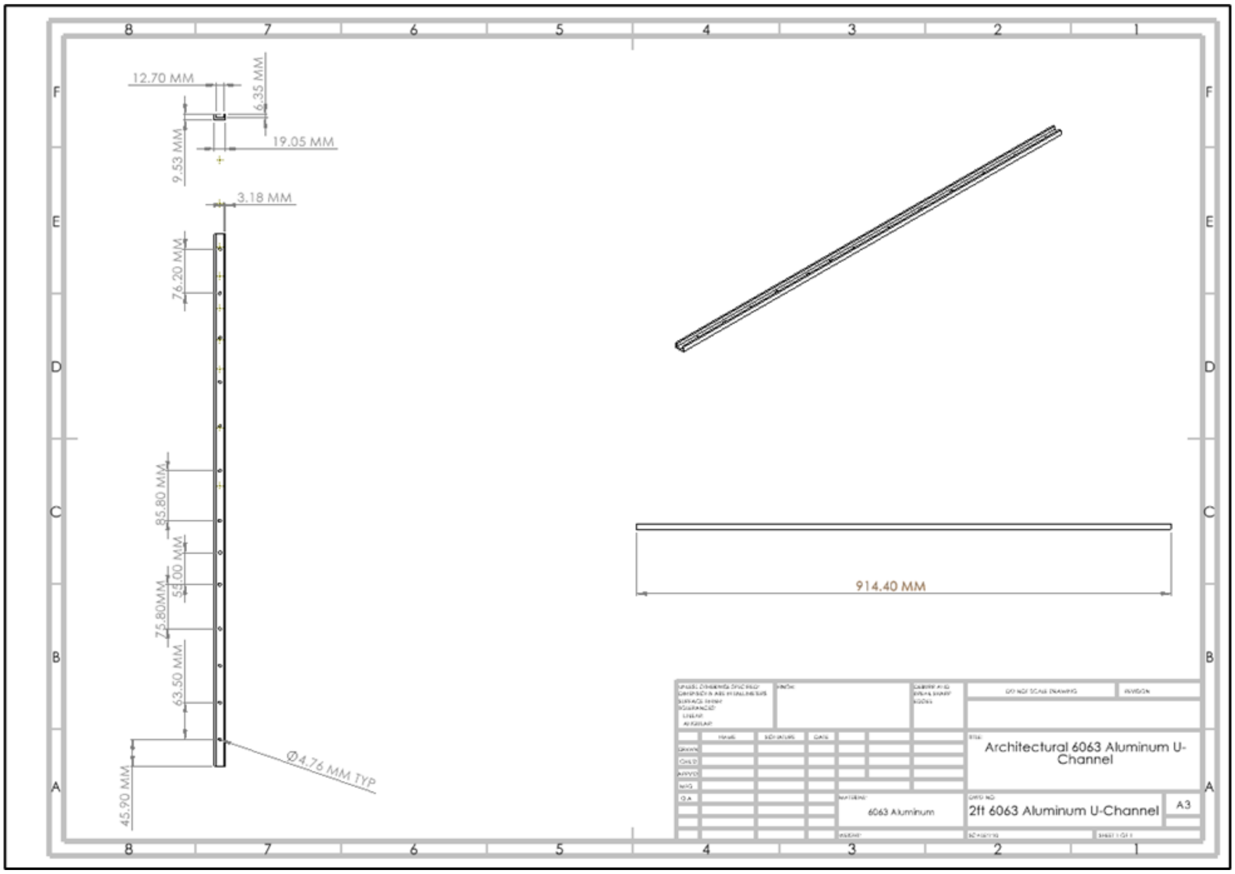


Figure 3.22: 6063 Aluminum Channel Dimensional Drawing

3.5.3.4 Channel Insert Design

In order to interface the flat, aluminum c-channel to the curved interior of the rocket body, spacers will be needed. There are two types of 3D printed channel inserts, A and B, each with the same purpose but with different corresponding lengths. They are designed to fit the aluminum channel to the fiberglass housing. Both types of inserts will be printed out of nylon.

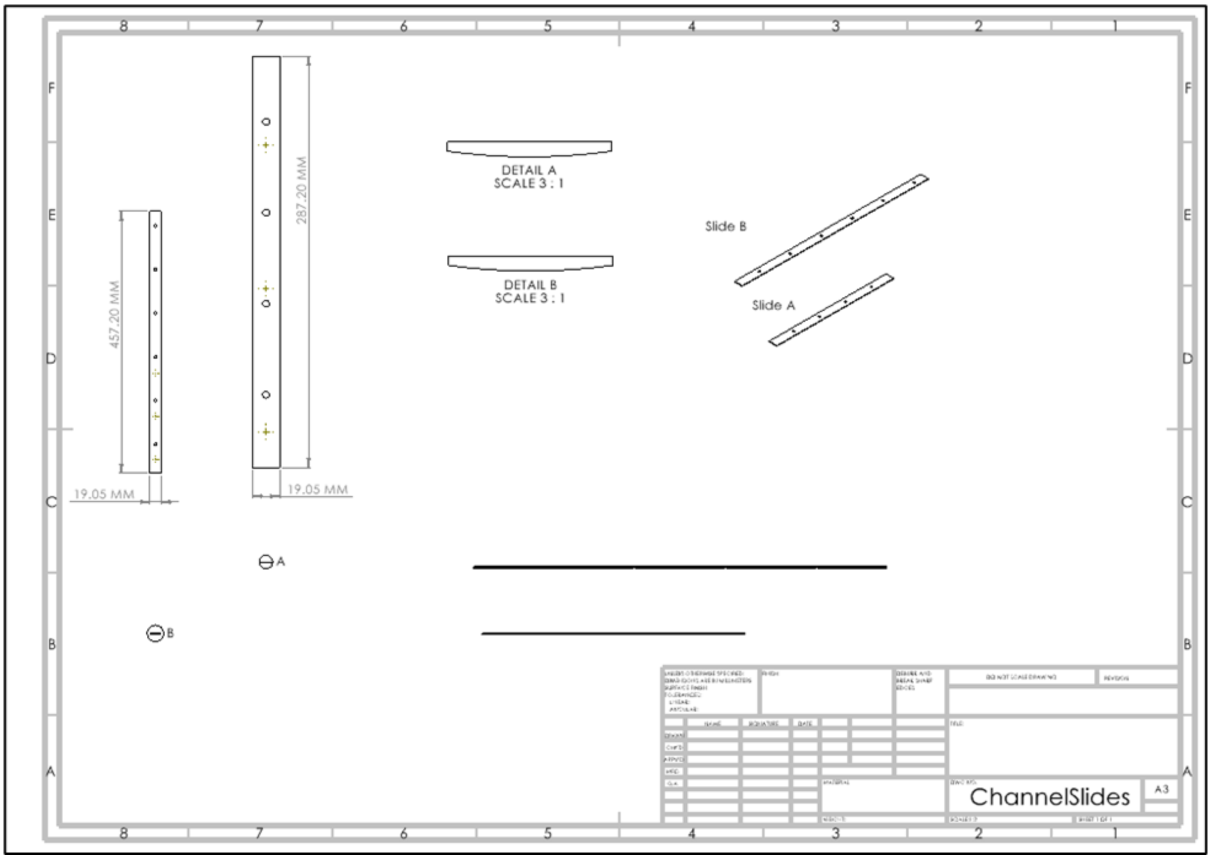


Figure 3.23: Channel Insert Dimensional Drawing

3.5.3.5 Housing

The Altitude Assurance housing will be 3D printed out of nylon. The housing is designed as an aerodynamic shell with cutouts for the petals and linkages.

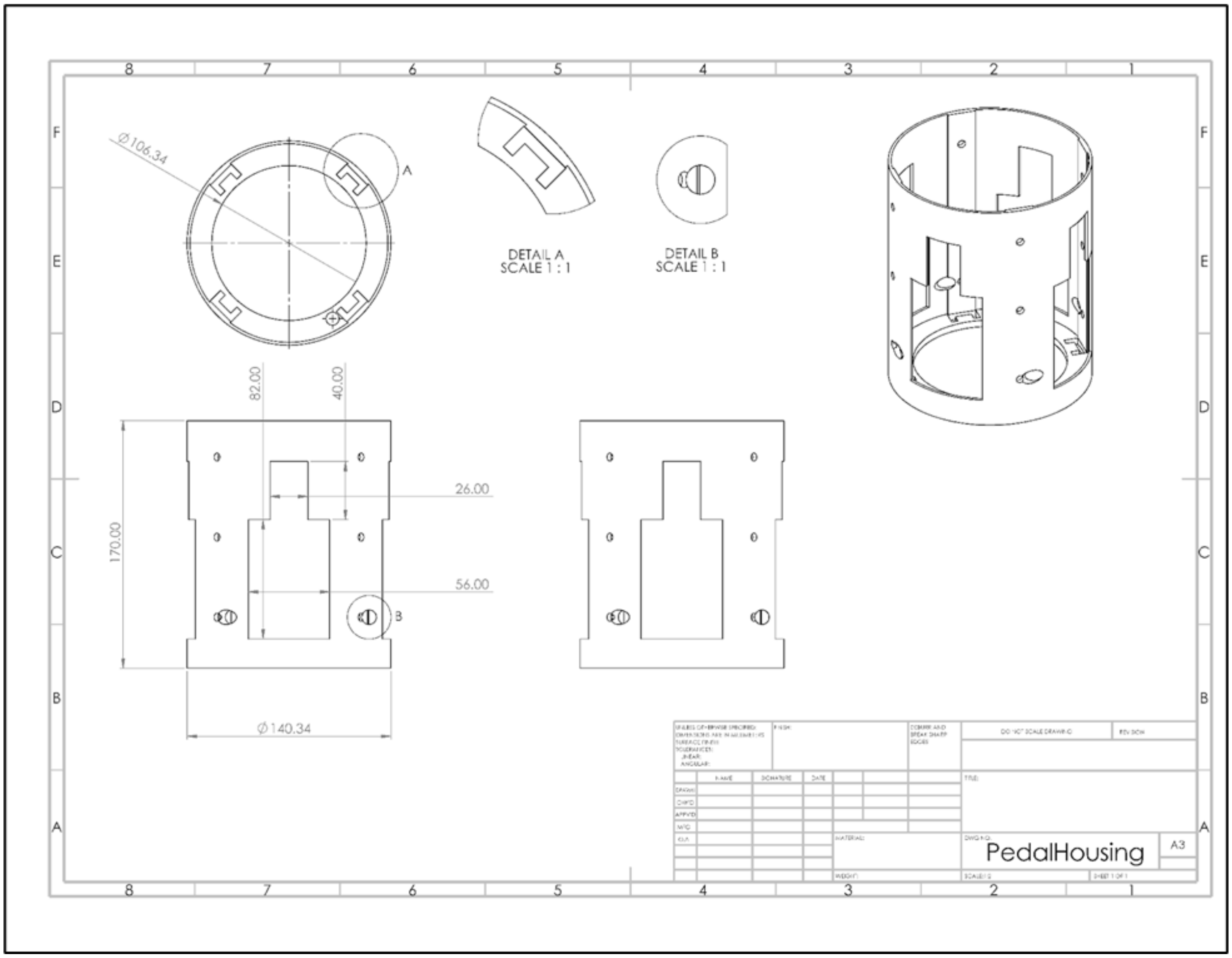


Figure 3.24: Altitude Assurance Housing Dimensional Drawing

3.5.3.6 Brake Mount

The Altitude Assurance brake mount will be 3D printed out of nylon. The brake mount is the primary structure for the actuation, providing a pivot point for the petals and a mount for the pneumatic piston. The brake mount is assembled to the housing by sliding it through the top of the housing, gluing it to the lip on the housing and bolting it directly to the aluminum u-channels.

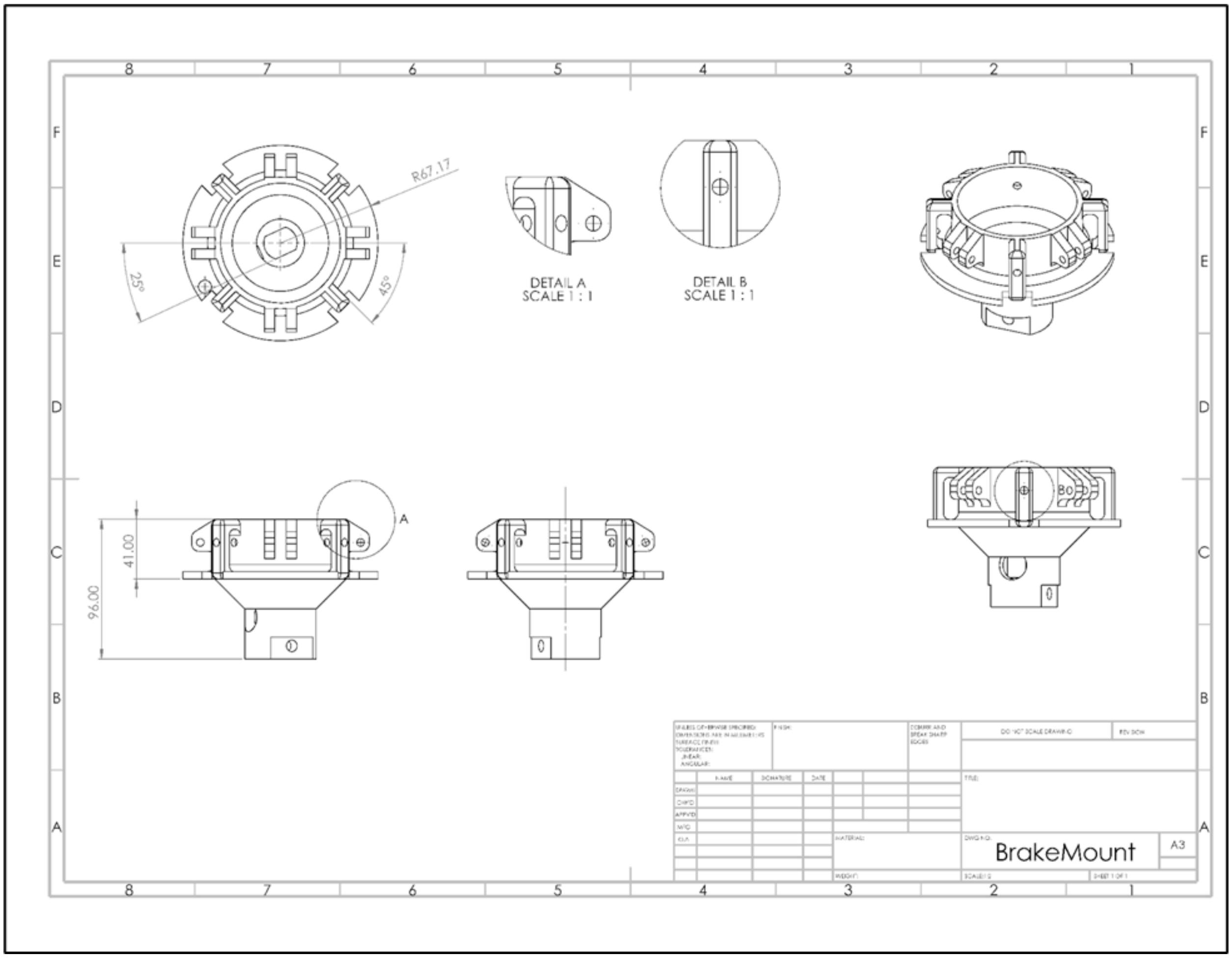


Figure 3.25: Brake Mount Dimensional Drawing

3.5.3.7 Linkages

The Altitude Assurance linkages will be 3D printed out of nylon. These linkages are designed to actuate the petal and fill in the gap in the housing created for their motion.

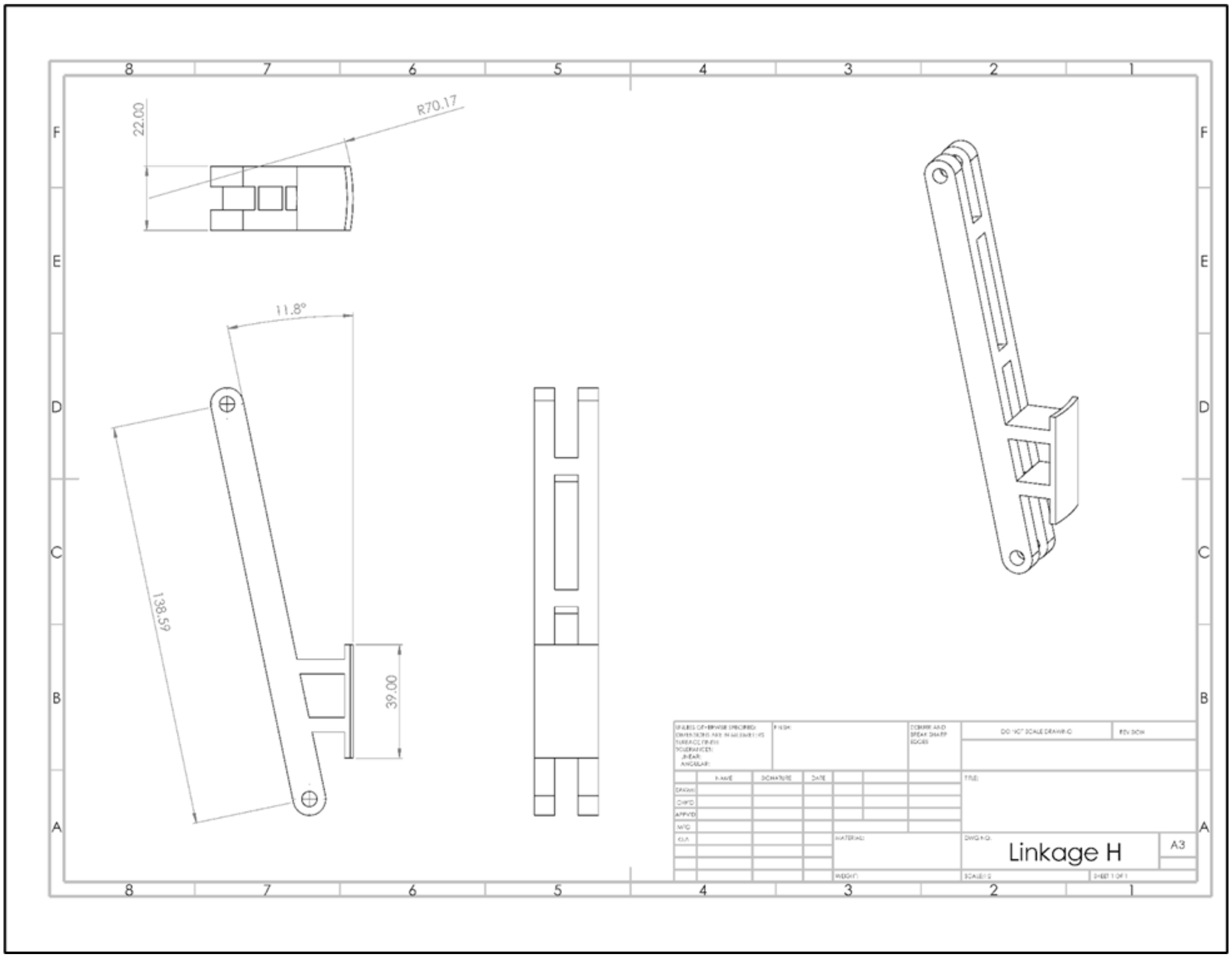


Figure 3.26: Altitude Assurance Linkage Dimensional Drawing

Structural analysis was performed on the linkage to ensure the structural suitability of nylon. Finite element analysis was performed to validate the strength of the linkage under the maximum loading with the applied safety factor of 4.0. A static structural analysis was performed with a bearing load of 1000N applied at the bolt-hole interface of the linkage and the linkage mount. A compression-only support was applied at the bolt-hole interface of the linkage and the Rose Petal.

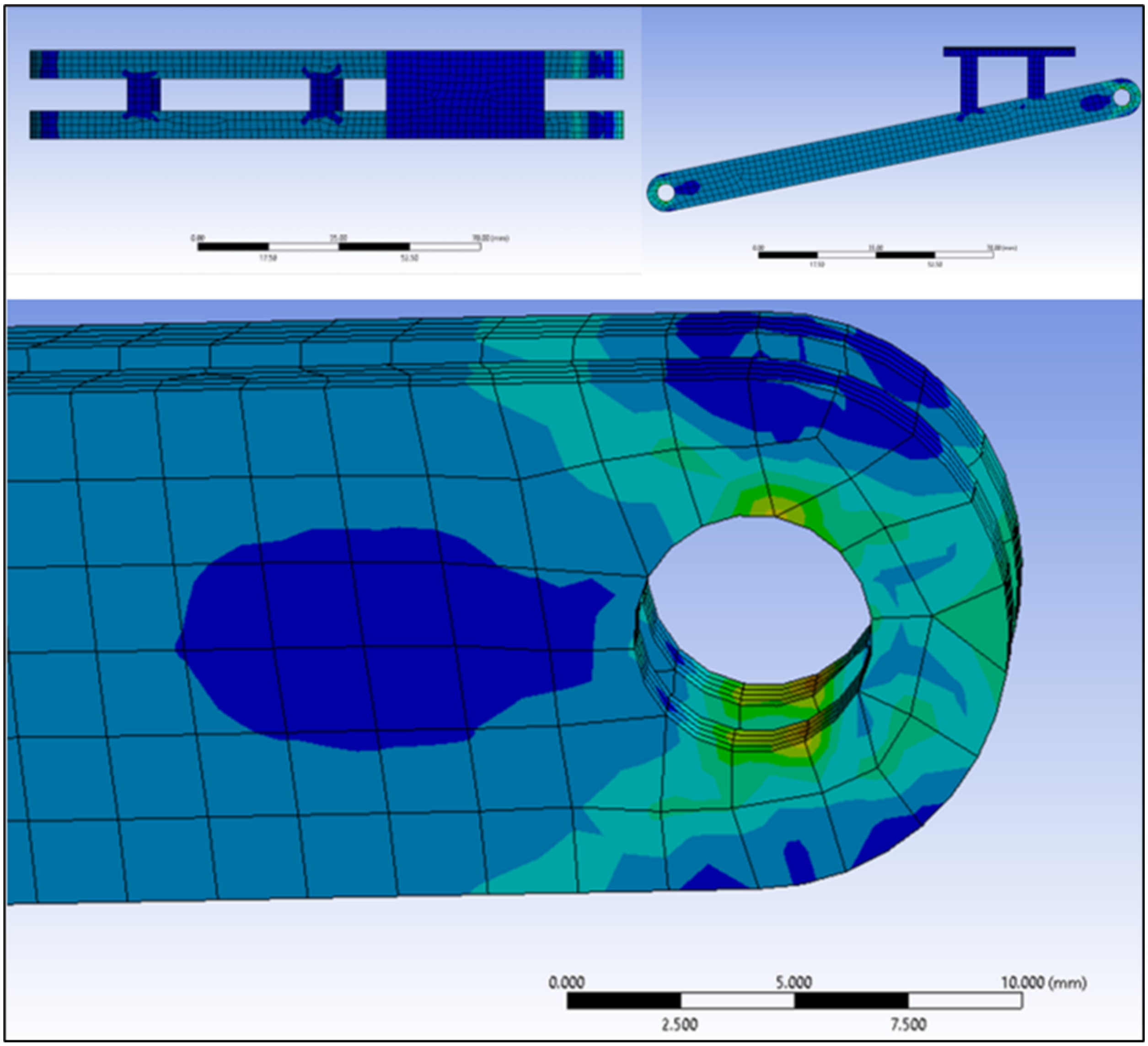


Figure 3.27: Altitude Assurance Linkage Structural Analysis

From the analysis, it was determined that the maximum loading of 53.1 MPA occurred at the loaded bolt-holes. Because this is less than 66 MPa, the tensile strength of 3D printed nylon at yield, the dimensioning and material choice for the linkage are valid.

3.5.3.8 Linkage Mount

The Altitude Assurance linkage mount will be 3D printed out of nylon. This part connects to the end of the piston and is secured by two nuts on the threaded piston rod.

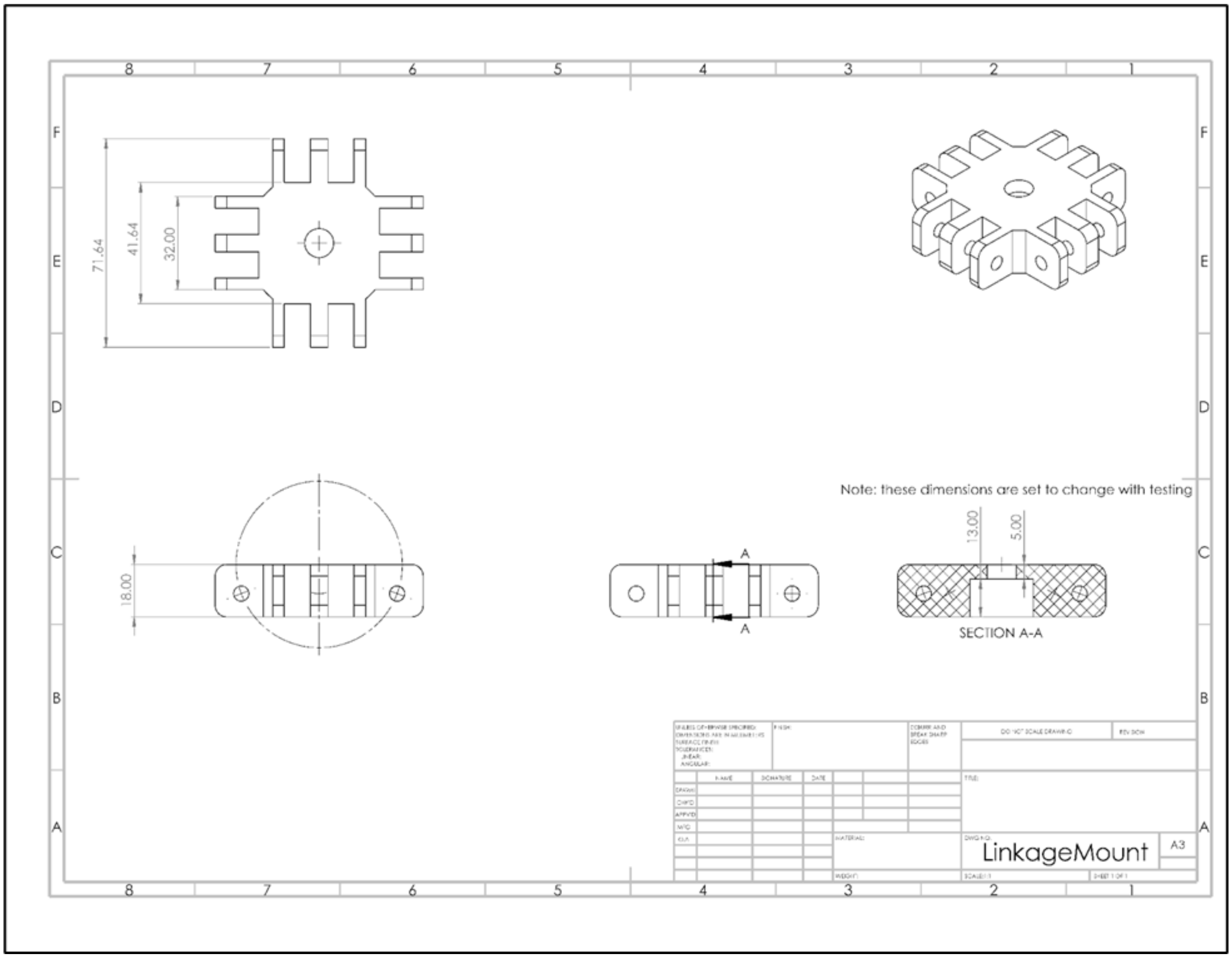


Figure 3.28: Altitude Assurance Linkage Mount Dimensional Drawing

Structural analysis was performed on the linkage mount to ensure the structural suitability of nylon. Finite element analysis was performed to validate the strength of the linkage under the maximum loading with the applied safety factor of 4.0. A static structural analysis was performed with a bearing load of 1300N applied at the bolt-hole interface of the linkage mount and the linkage. A compression-only support was applied at the center bolt-hole interface of the linkage mount and the pneumatic piston.

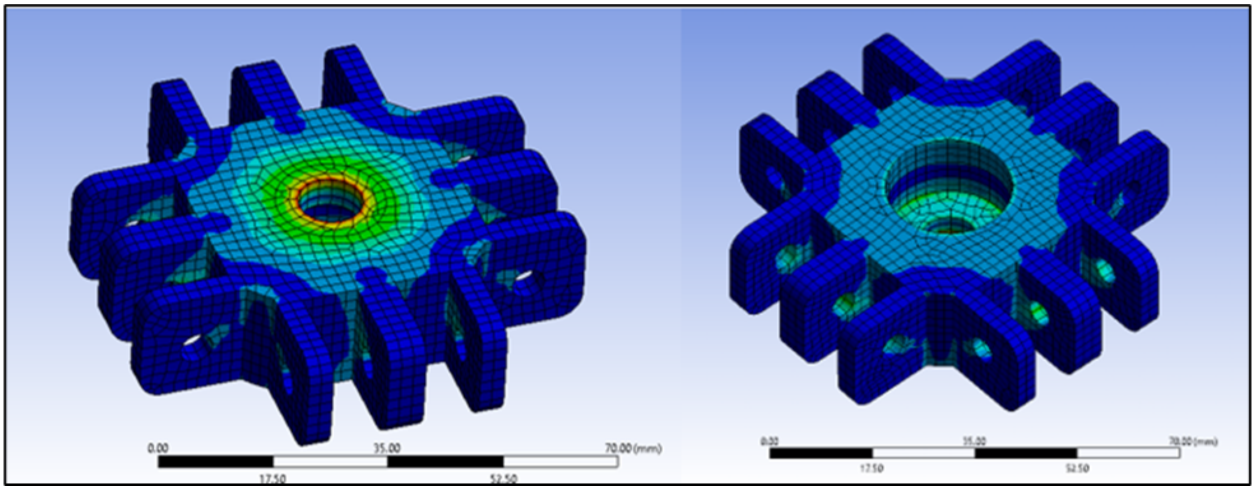


Figure 3.29: Altitude Assurance Linkage Mount Structural Analysis

From the analysis, it was determined that the maximum loading of 27.9 MPA occurred at the center bolt-hole. Because this is less than 66 MPa, the tensile strength of 3D printed nylon at yield, the dimensioning and material choice for the linkage mount are valid.

3.5.3.9 Antifold Disk

The Altitude Assurance anti-fold disk will be 3D printed out of Nylon. This part prevents the air pressure and handling from pushing the petals and linkages too far inside the rocket.

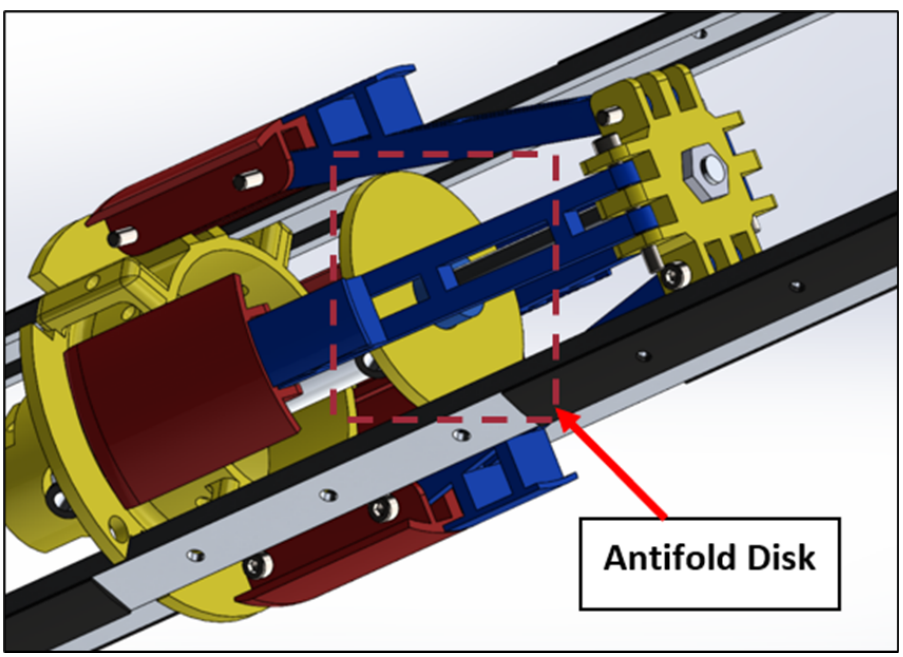


Figure 3.30: Altitude Assurance Antifold Disk Location

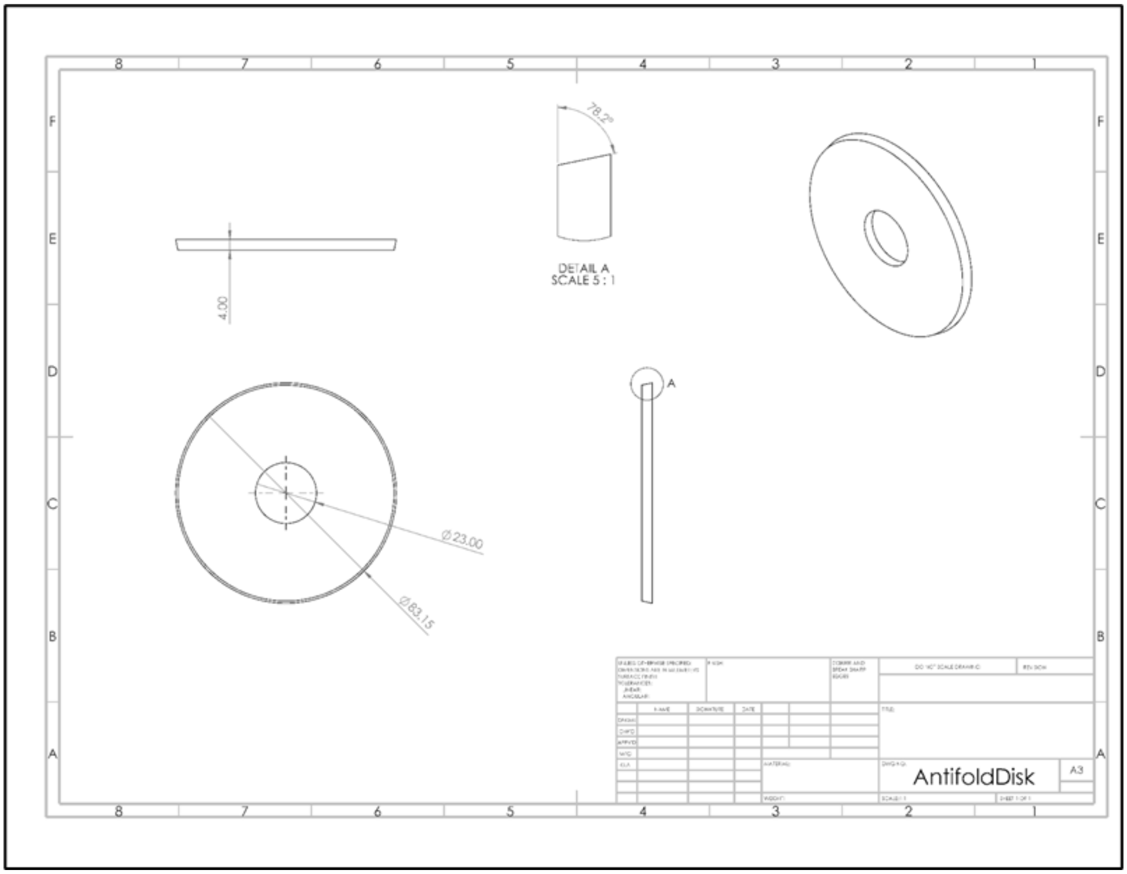


Figure 3.31: Altitude Assurance Antifold Disk Dimensional Drawing

3.5.3.10 Rose Petals

The Altitude Assurance Rose Petals will be 3D printed out of Nylon. These parts were designed according to the drag requirements for the vehicle. The OpenRocket software was used to calculate the required drag force and then an aerodynamic model for a flat plate was used to calculate the required effective area for the pedals. This flat plate drag model is given by

$$F_{drag} = 0.5 * C_d * A_{effective} * \rho * v_{airstream}^2$$

Where F_{drag} is drag force, C_d is the drag coefficient, $A_{effective}$ is the effective area, ρ is the air density, and $v_{airstream}$ is the velocity of the airstream the pedals fold into. This folding fulfills our requirements by rotating 90 degrees into the airstream, causing symmetrical drag below the center of gravity.

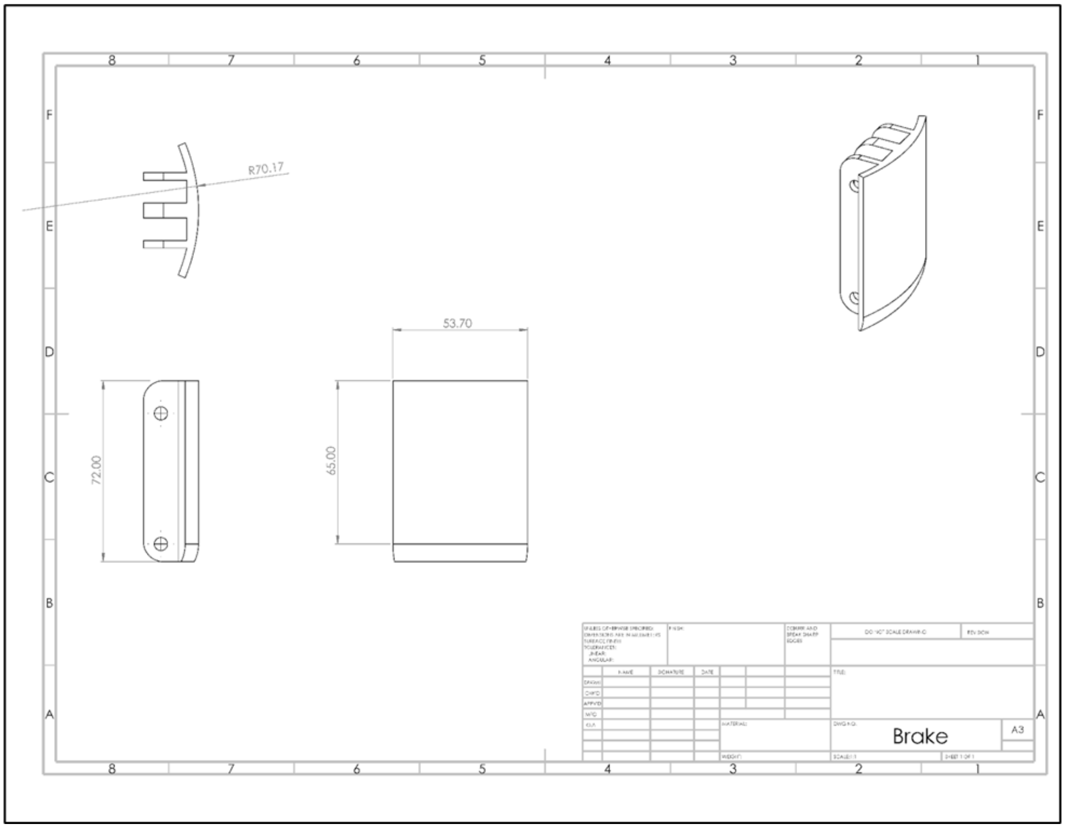


Figure 3.32: Rose Petal Dimensional Drawing

Structural analysis was performed on the Rose Petals to ensure the structural suitability of nylon. Finite element analysis was performed to validate the strength of the linkage under the maximum loading with the applied safety factor of 4.0. A static structural analysis was performed with a bearing load of 300N applied at the bolt-hole interface of the Rose Petals and the brake mount. A compression-only support was applied at the bolt-hole interface of the Rose Petals and the linkage.

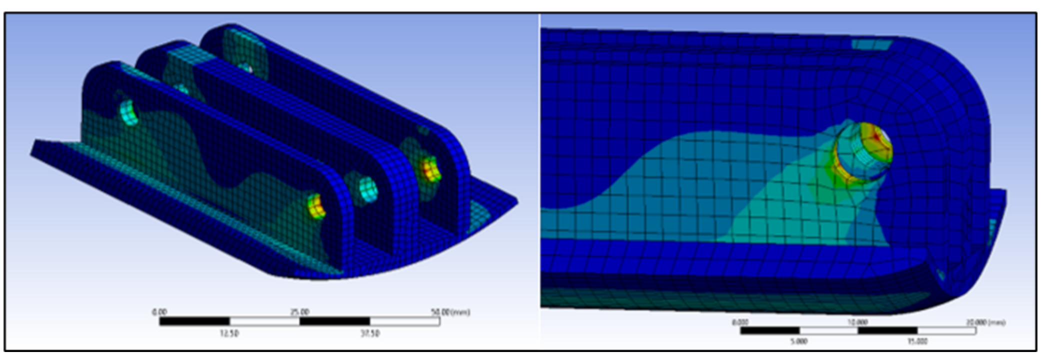


Figure 3.33: Rose Petal Structural Analysis

From the analysis, it was determined that the maximum loading of 9.9 MPA occurred at the loaded bolt-holes. Because this is less than 66 MPa, the tensile strength of 3D printed nylon at yield, the dimensioning and material choice for the Rose Petals are valid.

3.5.3.11 Controls Bay and Air Tank

The Altitude Assurance Controls Bay and Air Tank provide the means of actuation for the rest of the Altitude Assurance subsystem. The air tank is stored aft of the Controls Bay and is secured to the vehicle through the use of three centering rings. Two of these centering rings position the air tank while the remaining ring clamps onto the throat of the air tank to secure it in place. The positioning rings are permanently bonded to the U-channel using TotalBoat High Performance Epoxy Resin in order to maximize stiffness of the system. The clamping ring bolts to the forward positioning ring and serves to allow easy removal and installation of the air tank while preventing forward motion of the air tank over the duration of the flight. The two forward rings are drilled and tapped to accept ¼ in. to ¼ in. push-to-connect fittings. These fittings serve as a seal between the Altitude Assurance Controls Bay and the aft-end of the vehicle.

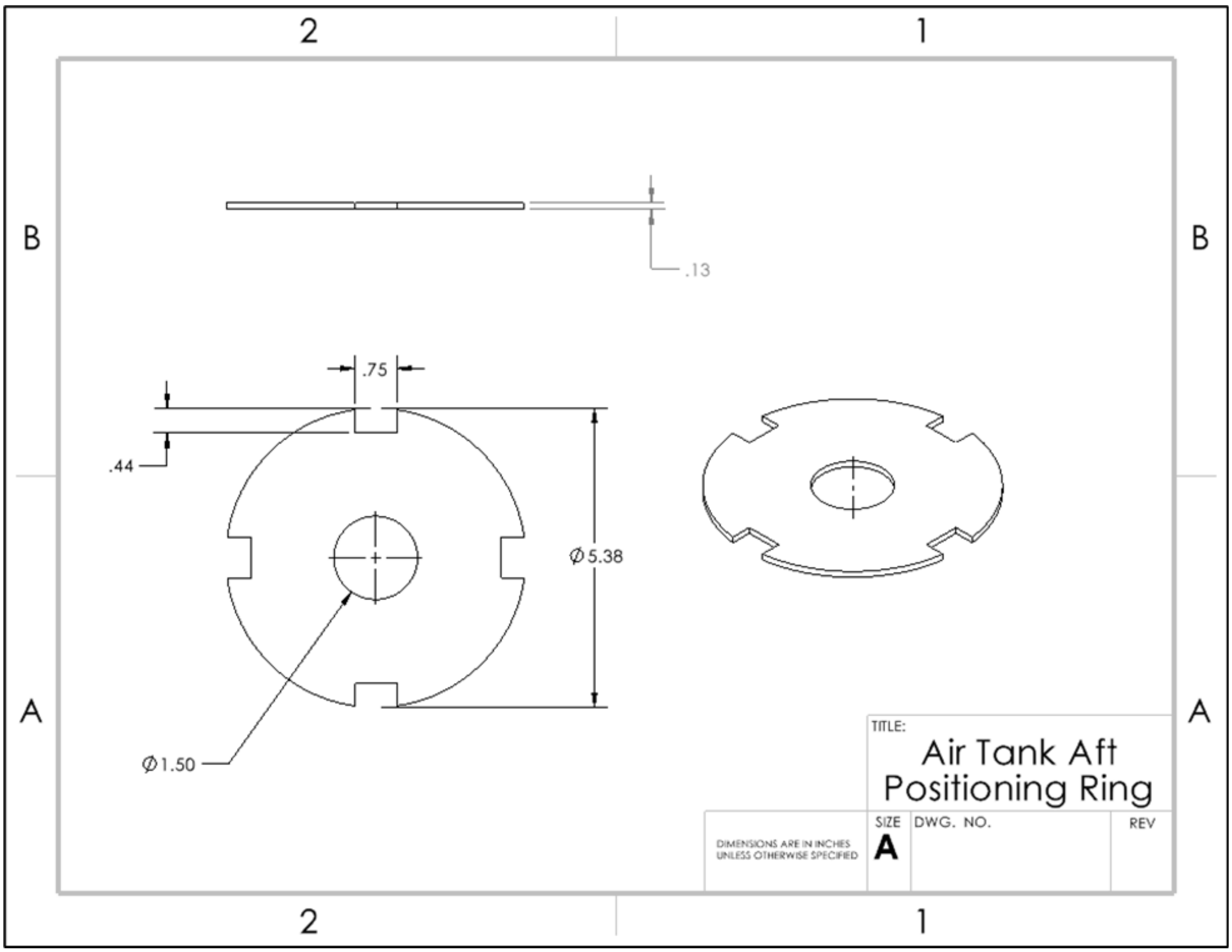


Figure 3.34: Air Tank Aft Positioning Ring Dimensional Drawing

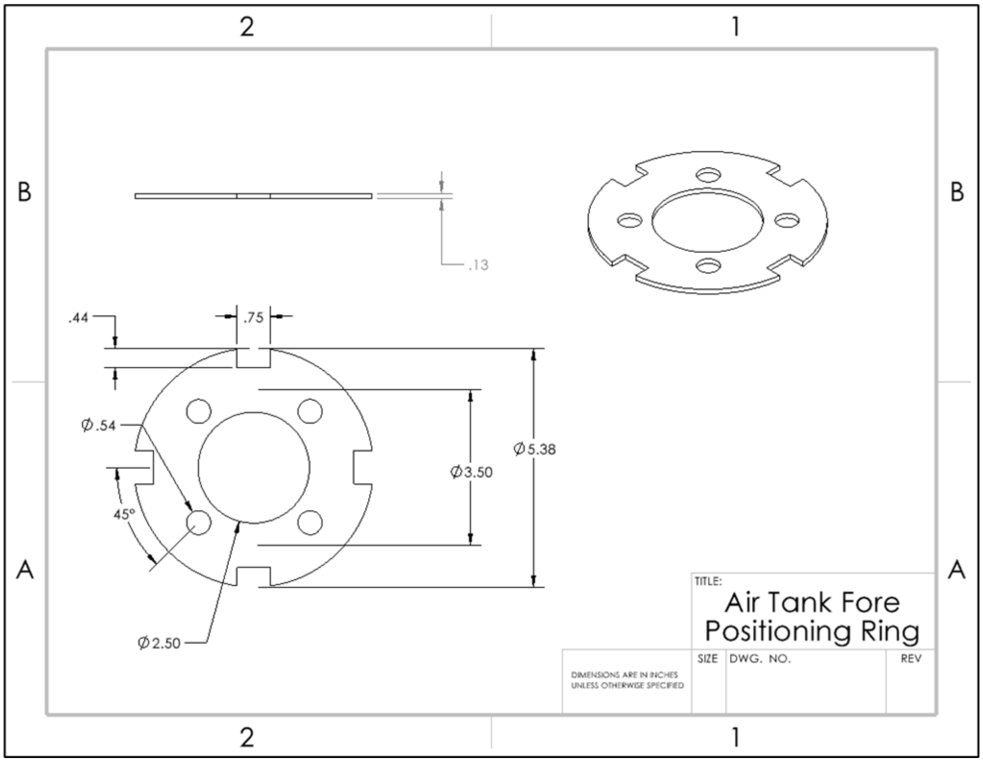


Figure 3.35: Air Tank Fore Positioning Ring Dimensional Drawing

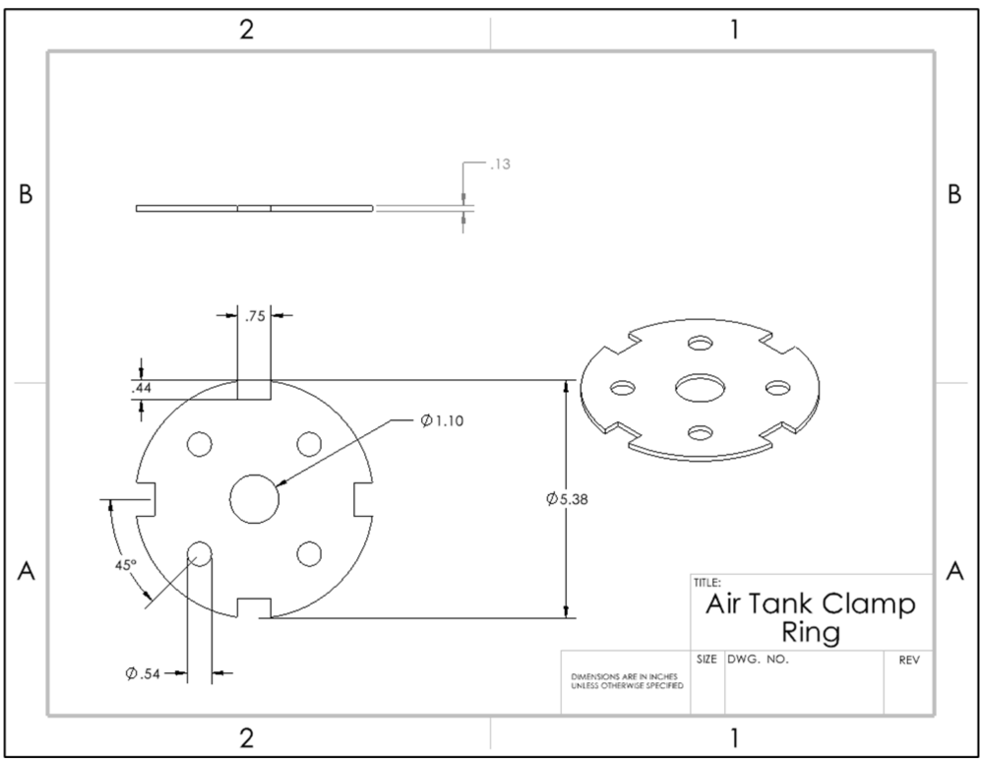


Figure 3.36: Air Tank Clamp Ring Dimensional Drawing

The Altitude Assurance Controls Bay is seated between the air tank and the drogue chute bay. In this bay, the control electronics and solenoids command high pressure air to the Altitude Assurance pneumatic cylinder for actuation. The electronics within the bay consist of a single-acting solenoid, a Teensy 4.1 Development Board, a BMP280 barometric pressure sensor, a BNO055 IMU, a 2S lithium-polymer battery, a buck-boost converter, and an arming switch. These electronics are mounted to a sled that in turn mounts to threaded rods held in tension by two coupler caps. The aft cap has an opening for pneumatic tubing to pass through. The forward cap holds a U-bolt mounted to a distributor plate (identical to the distributor plates used in the Recovery Subsystem). This U-bolt then secures a recovery harness for the drogue section of the Recovery Subsystem.

As evident from the subscale test of the Altitude Assurance system, the barometric pressure sensor needs to be sealed from the aft section of the Altitude Assurance system and should only be exposed to the atmosphere surrounding the vehicle. Because of this, the Controls bay needs to pass air without exposure to the turbulence of the Altitude Assurance. The design accomplishes this by using four push-to-connect fittings as seals which are mounted on the forward air tank centering rings. Two of the fittings are used for commanding air to the cylinder while the other two are used for air input to the solenoid and purged air.

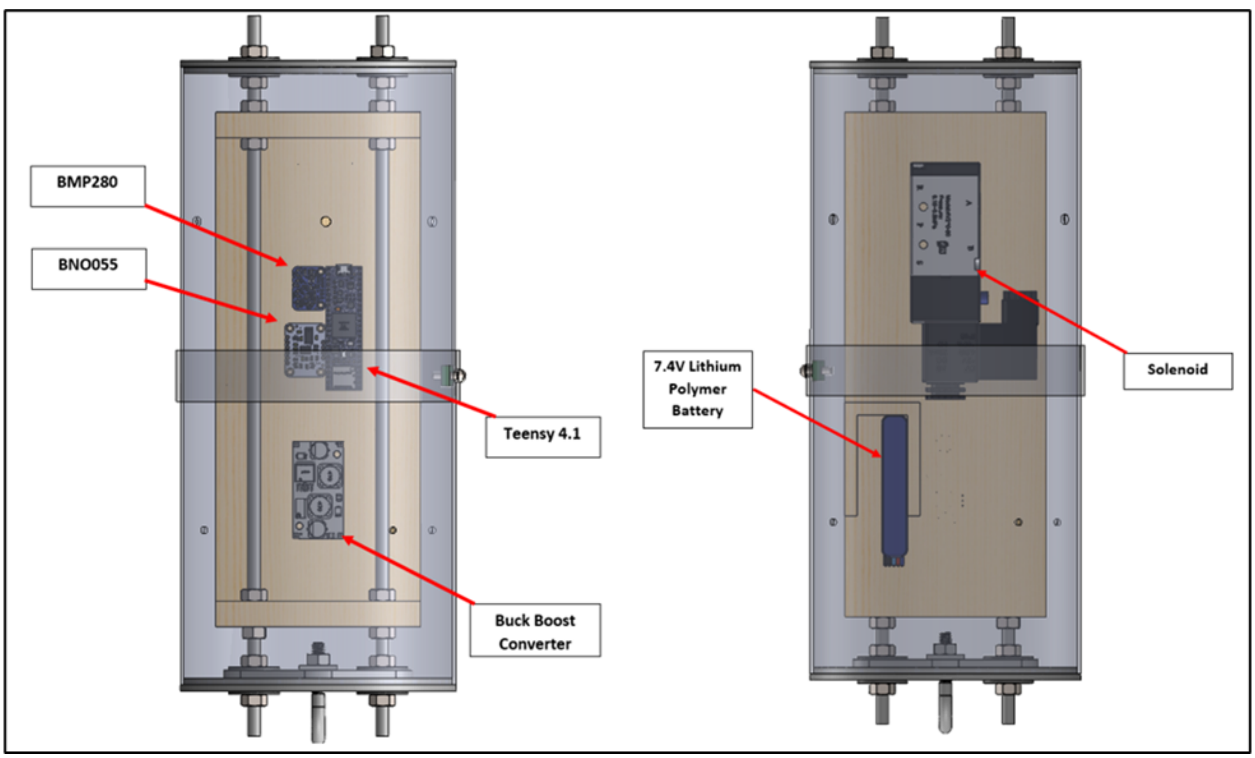


Figure 3.37: Altitude Assurance Controls Bay Electronics

3.5.4 Booster Subsystem

The Booster Subsystem consists of the L2375-WT motor, four G10 fiberglass trapezoidal fins, and the aft section of the airframe. It will have a mass of 14 lbm. Fins are permanently bonded to both the motor tube and the airframe section by epoxy resin adhesive. Centering rings are similarly bonded to the airframe using epoxy resin adhesive. Careful consideration to the epoxy construction will be given for creating filets between adhering surfaces. The decision to join the fins and centering rings by epoxy resin adhesive was made in consideration to team experience, ease of construction, and common use in the construction of high-power rockets.

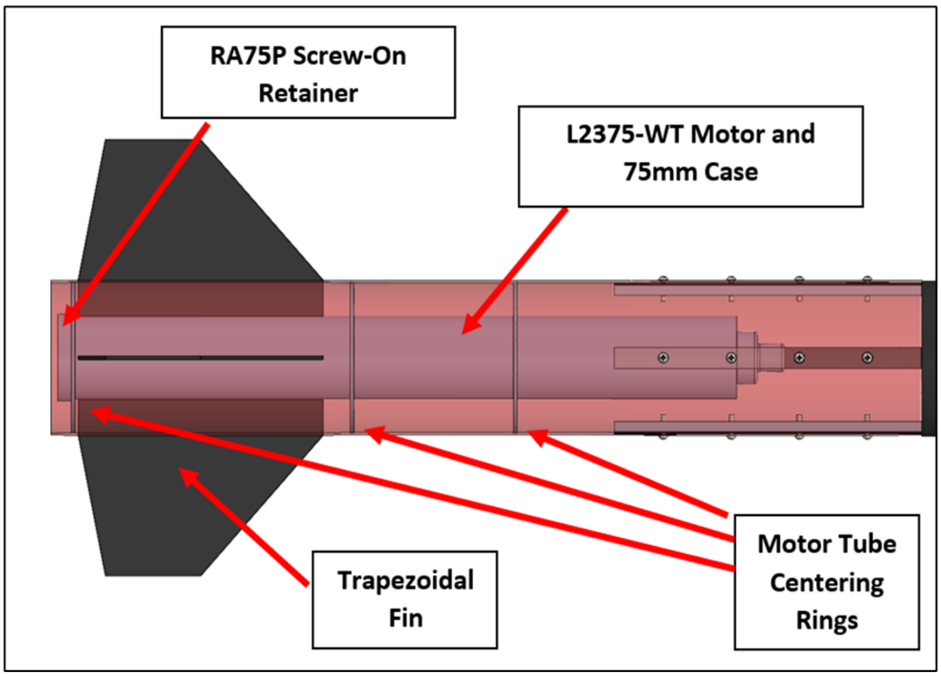


Figure 3.38: Booster Subsystem Components

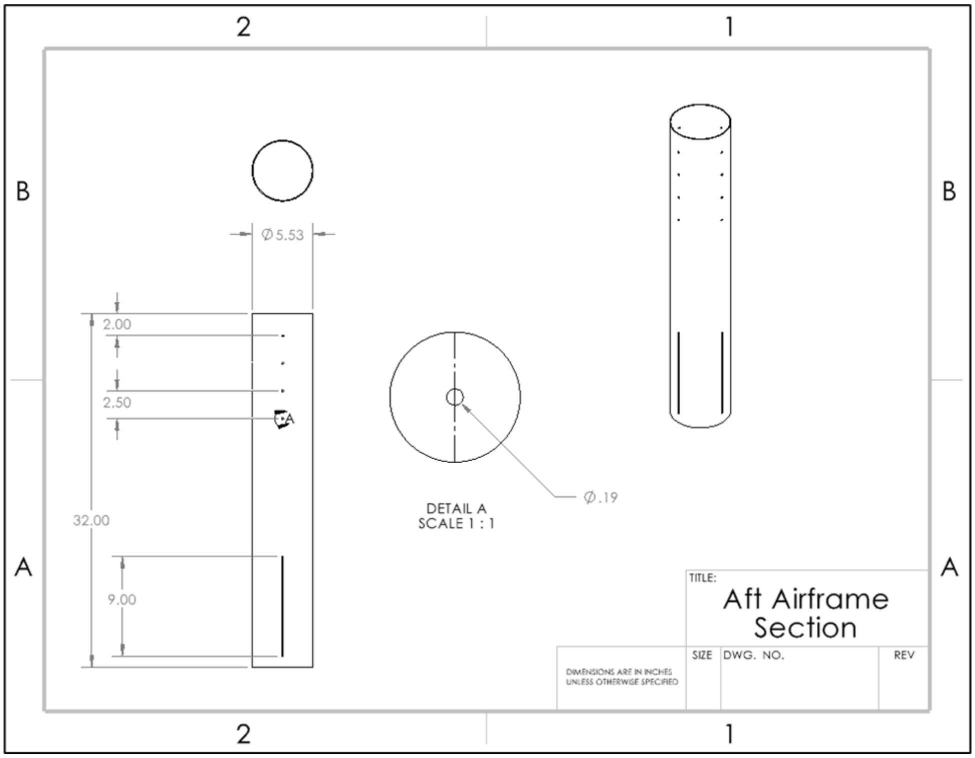


Figure 3.39: Aft Airframe Section Dimensional Drawing

3.5.4.1 Fin Design

The selected fin design is a trapezoidal shape with root chord 9 in, height 5 in, tip chord 3.5 in, and sweep length 4.5 in from the leading edge. This design offers fair aerodynamic efficiency while being easy to manufacture and providing a more-than-sufficient stability margin for the vehicle as designed. Additionally, it reduces chances that the rocket lands fins-first in comparison to a swept or clipped-delta design, making airframe damage less likely in a recovery-failure scenario.

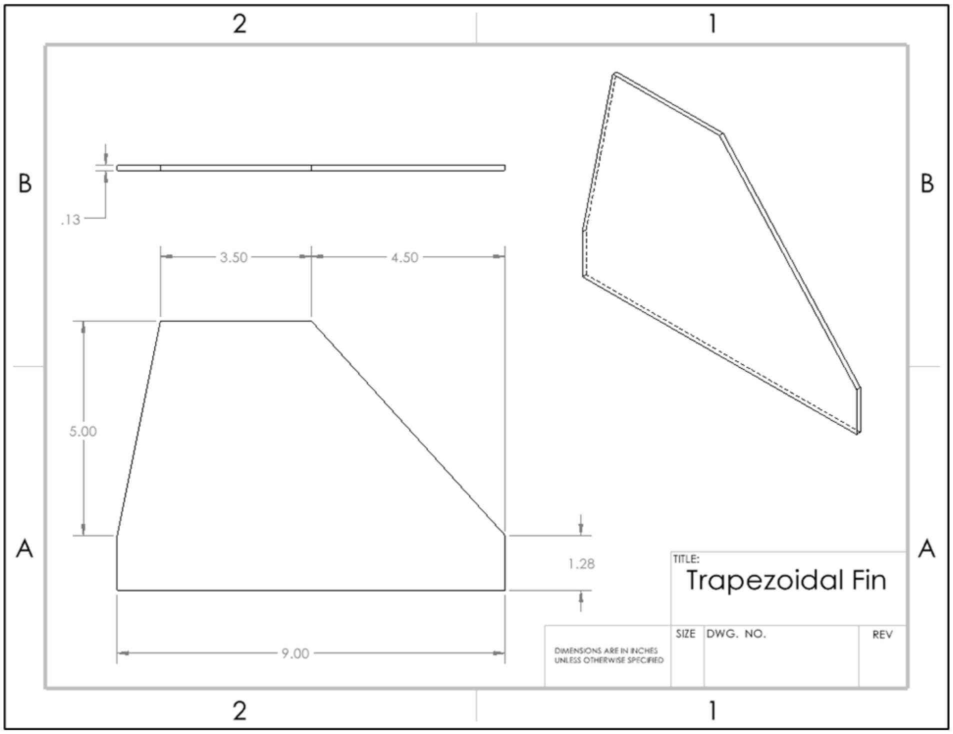


Figure 3.40: Trapezoidal Fin Dimensional Drawing

3.5.4.2 Motor Retention Method

The motor will be retained by an Aeropack RA75P screw-on retainer. This retainer design is commonly used throughout both this team and high-power rocketry in general for its ease of use and reliability. Additionally, it offers the advantage of motor-length independence, meaning that a shorter motor may be installed for partial-flight-profile tests should that become necessary.

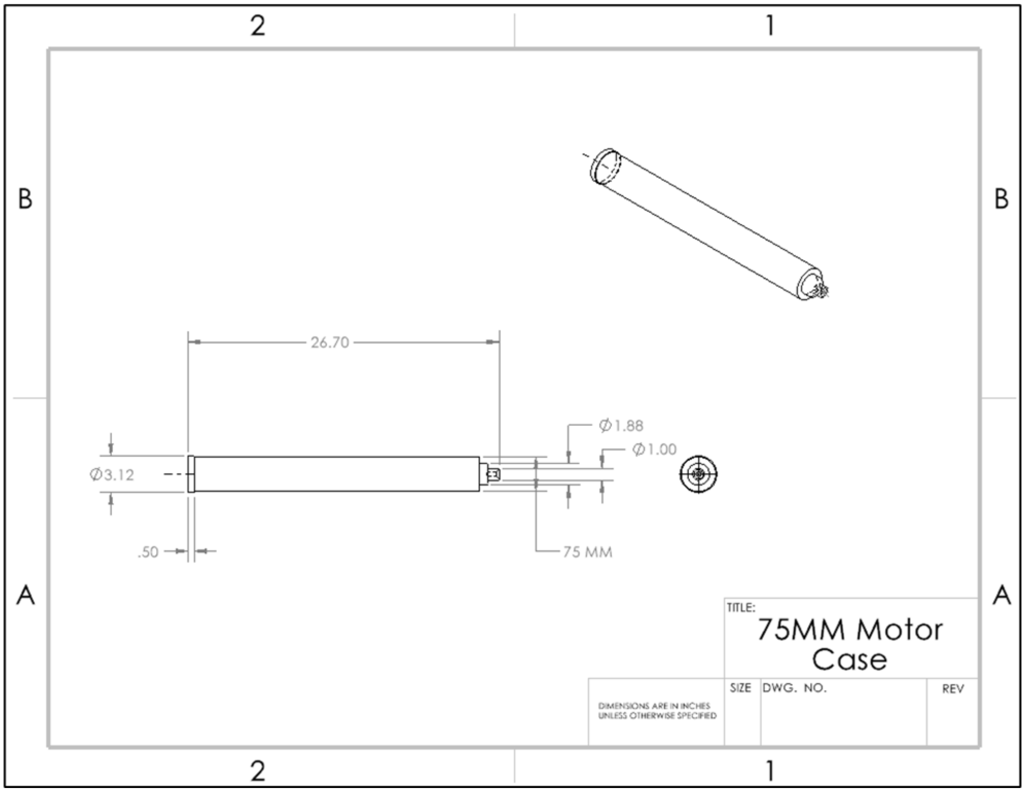


Figure 3.41: 75mm Motor Case Dimensional Drawing

3.5.5 Subsystems Mass

Table 3.6 shows a summary of the individual subsystems mass. The masses reported are representative of the fully configured vehicle ready to launch on the pad.

Table 3.6: Summary of Subsystems Mass

Vehicle Subsystem	Mass (lbm)
Payload	6.1
Recovery	14.6
Altitude Assurance	13.4
Booster	14.0
Total	48.1

3.6 Vehicle Subscale Flight Results

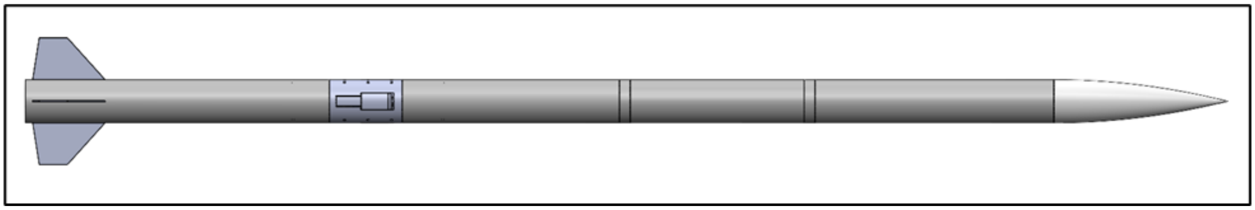


Figure 3.42: Subscale CAD Model

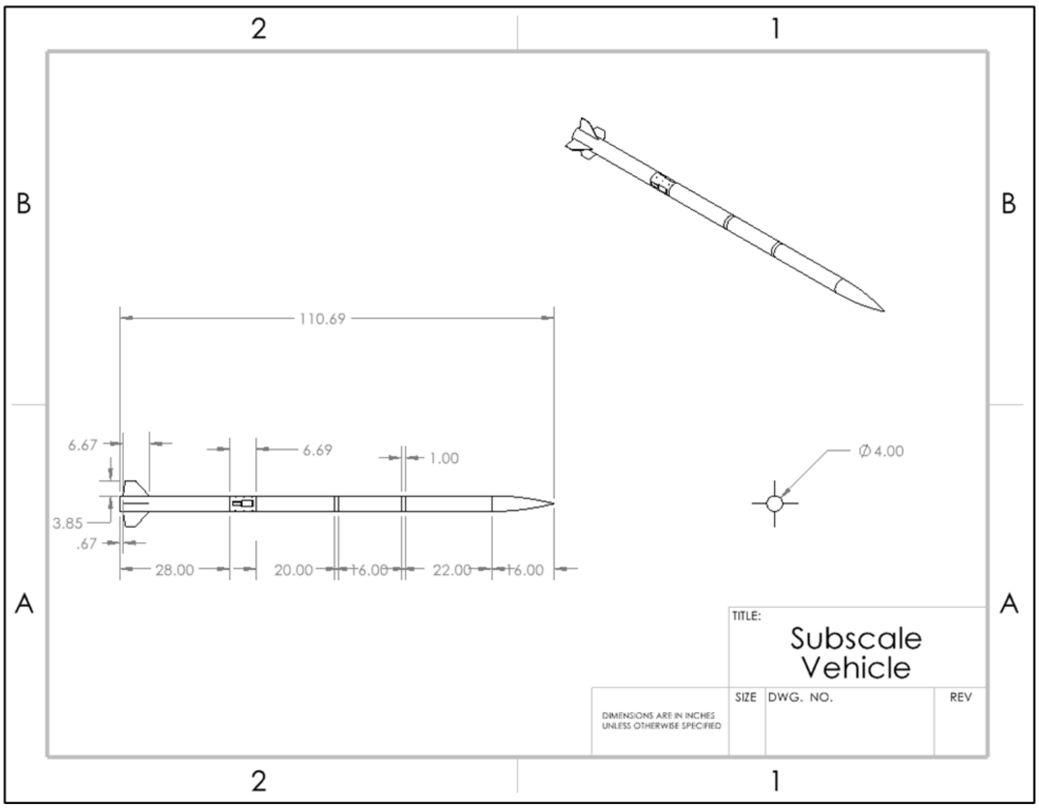


Figure 3.43: Subscale Dimensional Drawing

3.6.1 Scaling Factors

3.6.1.1 Airframe

The subscale air frame was scaled to 0.72:1 in both length and outside diameter. The team was constrained by the size of the Altitude Assurance system, which required a 4” diameter air frame. The choice of a larger subscale design also allowed the use of fabrication techniques and materials directly applicable to the full scale competition vehicle.

3.6.1.2 Altitude Assurance

Due to limited prior experience, the team deemed the inclusion of the Altitude Assurance system in the subscale flight necessary to demonstrate flight readiness and safety. The actuation mechanism of the Altitude Assurance system used the size designed in the full-scale design to validate electromechanical function. This includes pneumatics, 3D printed drag flaps, linkages, U-channels, and electronics. The only design alterations were the reduction of the number of air brakes from four to three and decreasing the diameter of the air brake housing to fit a 4” air frame.

3.6.1.3 Recovery System

The recovery hardware for the subscale launch was selected based on component availability. The main parachute, drogue parachute, and black powder ejection charges were selected specifically to meet the flight profile needs of the subscale build and provide a safe descent. Parachute sizing was simulated in OpenRocket to ensure a safe descent time and kinetic energy. Ground testing of the subscale rocket was conducted to verify a safe amount of black powder used for deployment separation. This method of parachute scaling ensures a safe vehicle recovery and does not impact mission performance since no payload experiment is affected by parachute sizing.

The recovery harness was scaled using the same 0.72:1 ratio as the air frame. Then, we selected the closest commercially available recovery harness which was equal to or greater than our scaled size. This resulted in a 30’ tubular Kevlar shock cord for the drogue parachute and a 25’ tubular Kevlar shock cord for the main parachute, both supplied from Wildman Rocketry.

3.6.1.4 Vehicle Occupation Length

The length allotted to each component of the vehicle was scaled proportionally with the overall subscale length. For example, the length of the airframe allotted to the main parachute in the subscale when compared to the full scale is 0.72:1. This is noteworthy because it ensures that any system capable of fitting in the subscale will fit inside the full scale vehicle. Specifically, because our Altitude Assurance system was scaled 1:1 with the full scale, we can guarantee there will be ample allotted space for the system in the full scale after launching it on the subscale.

3.6.1.5 Fins

The fins of the subscale vehicle were scaled proportionally to the full scale vehicle by a ratio of 0.72:1.

3.6.1.6 Flight Profile and Propulsion

The motor for the subscale vehicle was selected to match the flight profile of the full scale vehicle at a 1:1 scale. This was important to testing subsystems as we wanted to ensure that the subsystems could withstand the loads that they would see on the final vehicle. This was particularly important in testing the Altitude Assurance module as the maximum drag force withstanded by the Altitude Assurance Module scales with velocity squared.

3.6.2 Subscale Flight Analysis

3.6.2.1 Flight Conditions

The subscale flight was performed on December 19, 2021 in Pence, Indiana. The vehicle flew on a Cesaroni Technology K780BS and a functioning Altitude Assurance module. The local elevation was 700ft above sea level. The air temperature was 38°F and the air pressure was 30.33 inHg. The average wind speed was 3 mph. The predicted flight model was evaluated using a custom Simulink and MATLAB simulator at these conditions.

3.6.2.2 Predicted Flight Model

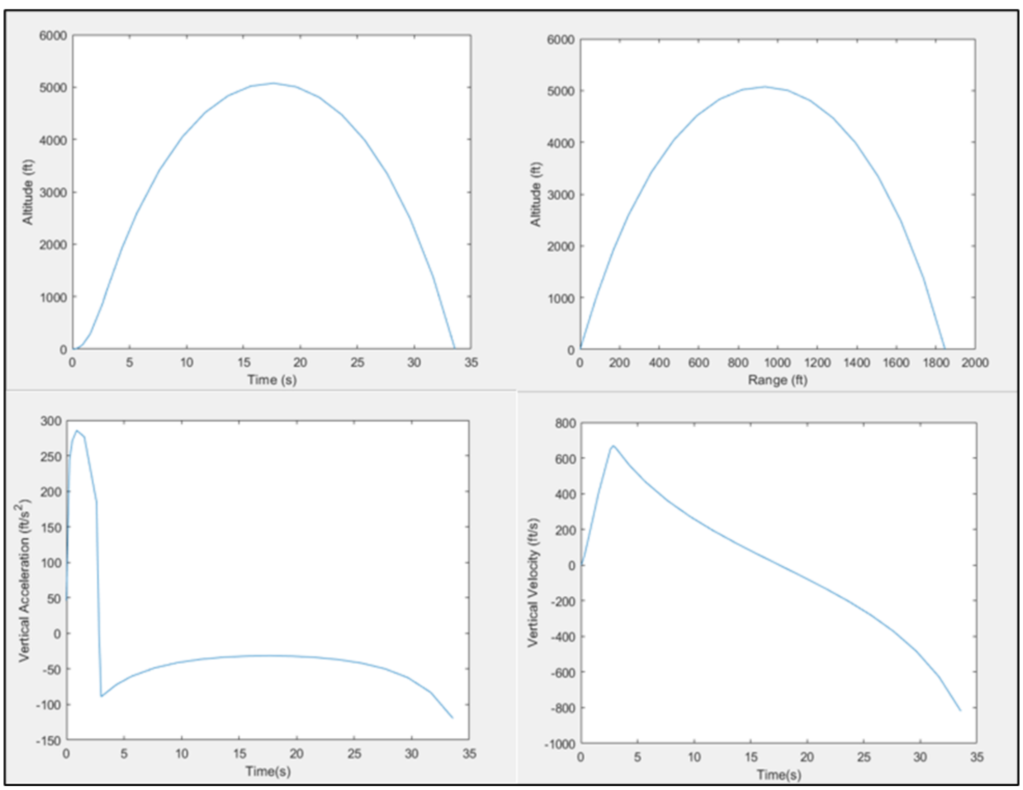


Figure 3.44: Subscale Predicted Flight Model

Table 3.7: Subscale Predicted Flight Model Summary

Apogee	5074 ft
Drift	1846 ft
Maximum Vertical Velocity	667.3 ft/s
Maximum Vertical Acceleration	285.7 ft/s^2

The flight model used the conditions from the subscale launch. A parabolic trajectory was appropriate for this as the actual subscale flight failed to deploy recovery devices. This flight model considers the base performance of the vehicle and does not model the performance with the Altitude Assurance Module.

3.6.2.3 Flight Data

Although the subscale flight failed to deploy recovery devices, flight data was recoverable from the vehicle. The subscale was equipped with an experimental Altitude Assurance Module to slow the vehicle down in flight.

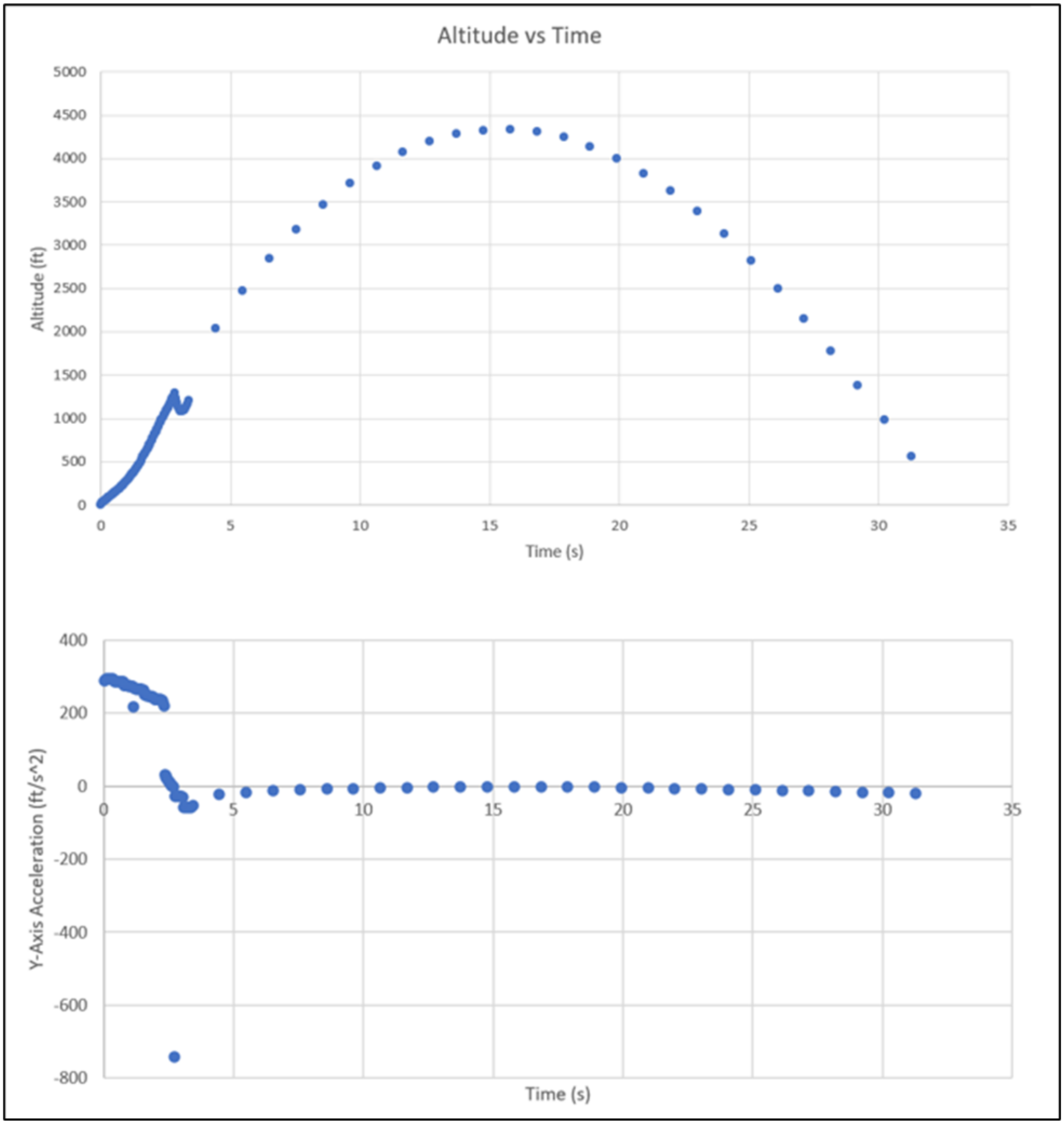


Figure 3.45: Subscale Flight Altitude and Acceleration Data

Table 3.8: Subscale Flight Data Summary

Apogee	4430 ft
Drift	2500 ft
Maximum Y-Axis Acceleration	295.6 ft/s^2

After the launch of the subscale vehicle, the landing location was estimated to be roughly 2500 ft from the pad. Because significant effort was required post-launch to recover flight hardware and extract data, no precise measurement of landing location was taken.

3.6.2.4 Analysis of Flight Data

Based on analysis of the flight data, we have determined that the Altitude Assurance Module was functioning as expected and is capable of deploying and slowing the vehicle down in flight. The accelerometer Y-axis detects a sudden deceleration near 3s into the flight, corresponding to motor burnout. The Altitude Assurance Module quickly detects a false apogee roughly 0.4s later and so the petals are retracted. The indication of a false apogee detection is evident by the decreased data sample frequency, which was imposed by design in order to avoid inflating the SD card with recovery data where a high sampling rate is not necessary.

3.6.2.5 Discussion

Recovery of flight data allowed for identification of a significant design flaw in the Altitude Assurance controls bay. When the Altitude Assurance Module is not deployed, the pneumatic control piston is fully extended and pressurized. When the solenoid controlling the pneumatic piston purges the air in the cylinder, the pressure within the airframe quickly rises. This registers as an altitude decrease by the barometric altimeter, triggering extreme unintended behavior. This explains the sharp drop in altitude and immediate recovery observed around 2.8s. After identifying this issue, it was hypothesized that aerodynamic pressurization or depressurization of the chamber could also occur. In general, the control hardware must be in a separate volume of air from both the mechanism and purge outlet to accurately assess altitude.

This design flaw has been corrected in the full scale design by sealing the Altitude Assurance Controls Bay from the Altitude Assurance Module below, using full bulkheads around the air tank. An additional pneumatic line running from the Altitude Assurance Controls Bay to the Altitude Assurance Module will allow purged air to escape aft of the sealed region. Pneumatic lines will pass through this bulkhead using slip-fit tubing connectors as seals.

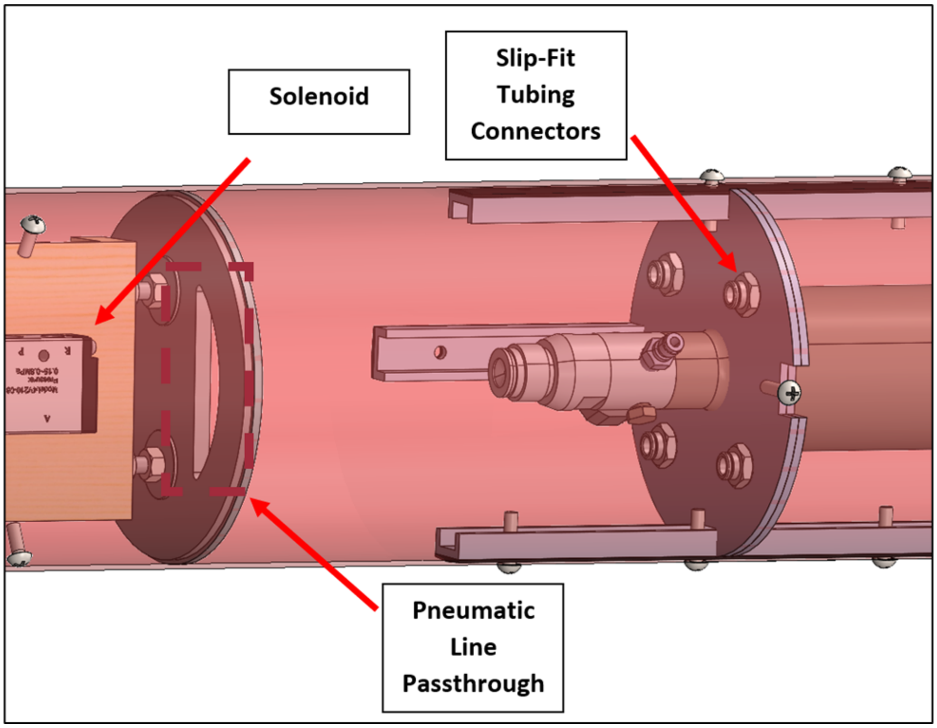


Figure 3.46: Air Tank Bulkhead Seal

The parabolic flight profile as seen in the altitude data shows that neither parachutes were able to deploy successfully. This was due to a launch operation failure to arm the altimeters. This heavily influenced the priorities of the team to place heavy emphasis on operations checklists to ensure that a similar failure will not occur on subsequent flights.

In comparison to the predicted flight model, the flight data shows that the vehicle achieved a lower apogee and similar maximum acceleration. The lower apogee is expected since it is evident that the Altitude Assurance Module had been deployed which is designed to lower the apogee of the vehicle in comparison to the base performance. Since the custom simulator does not account for wind, a lower drift is to be expected as well.

3.6.3 Pre-Subscale Rocket Flight Analysis

3.6.3.1 Flight Conditions

The pre-subscale flight was performed on December 11 2021 in Pence, Indiana. The local elevation was 700ft above sea level. The air temperature was 38°F and the air pressure was 39.93 inHg. The average wind speed was 15 mph. The predicted flight model was evaluated using OpenRocket at these conditions.

3.6.3.2 Predicted Flight Model

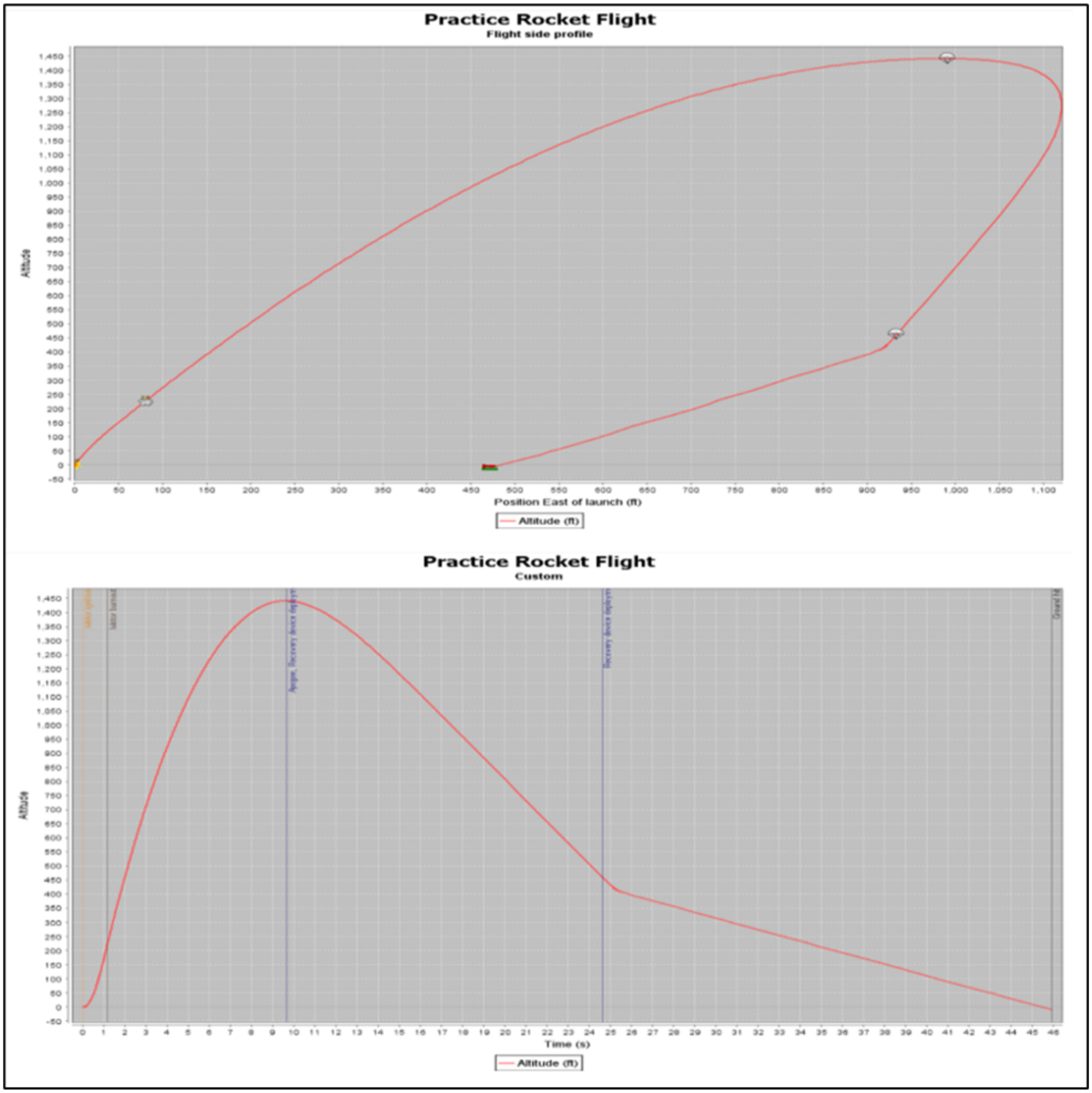


Figure 3.47: Practice Rocket Predicted Flight Model

Table 3.9: Practice Rocket Predicted Flight Model Summary

Apogee	1441 ft
Drift	472 ft

The flight model used the conditions from the subscale launch. The drogue chute was configured to deploy at apogee and the main chute was configured to deploy at an altitude of 500 ft. Because no acceleration or velocity data was gathered, the model did not report those profiles.

3.6.3.3 Flight Data

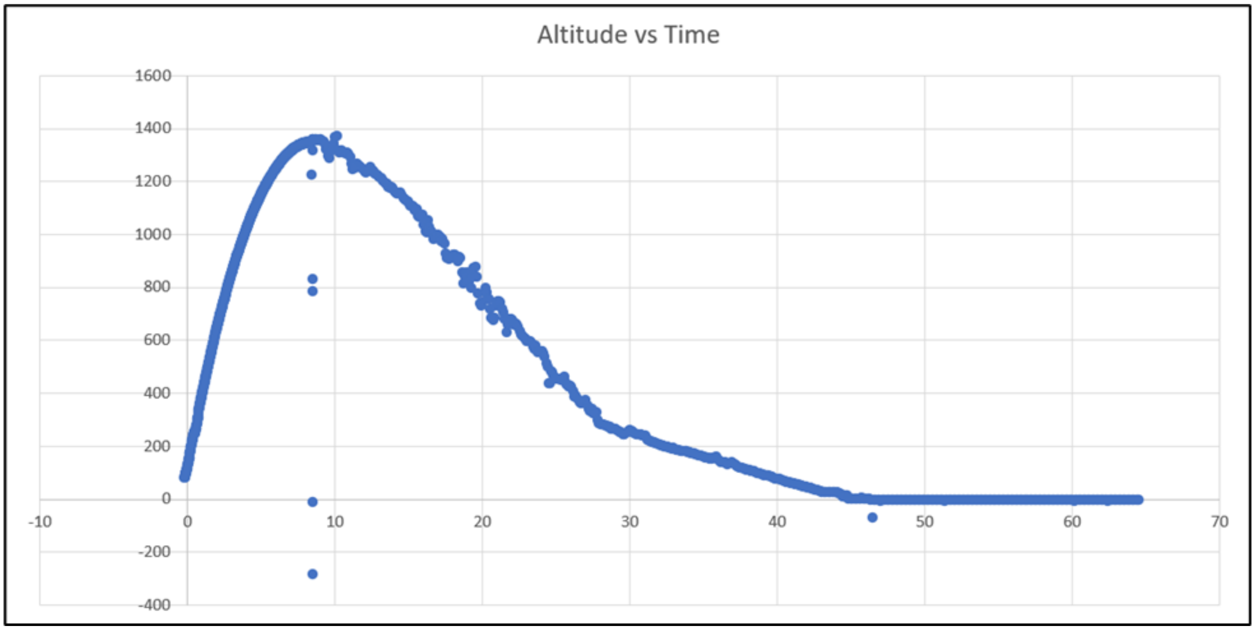


Figure 3.48: Practice Rocket Flight Altitude Data

Table 3.10: Practice Rocket Flight Data Summary

Apogee	1377 ft
Drift	520 ft

The drift was calculated using consumer-grade GPS measurements of the location of the launch pad and rocket recovery.

3.6.3.4 Analysis of Flight Data

From the data, it is evident that both parachutes deployed successfully as the flight profile is not parabolic. Two descent rates can be seen for the two parachutes deployed.

3.6.4 Drag Coefficient Calculations

The drag coefficient of the full-scale vehicle was estimated based on the rate of deceleration of the subscale just after motor burnout. Additionally, the drag coefficient of the rocket with Rose Petals deployed was calculated by the same method to validate the drag calculations.

Upon motor burnout, the onboard accelerometer of the petal control computer recorded a maximum deceleration of 0.856 G or 8.42 m/s^2, as pictured in the graph below. At this point, the motor's propellant has been exhausted and the rocket's mass is reduced to 8.24 kg, and integrating measured acceleration from the moment of launch up to this data frame yields a vertical velocity of 176.9 m/s.

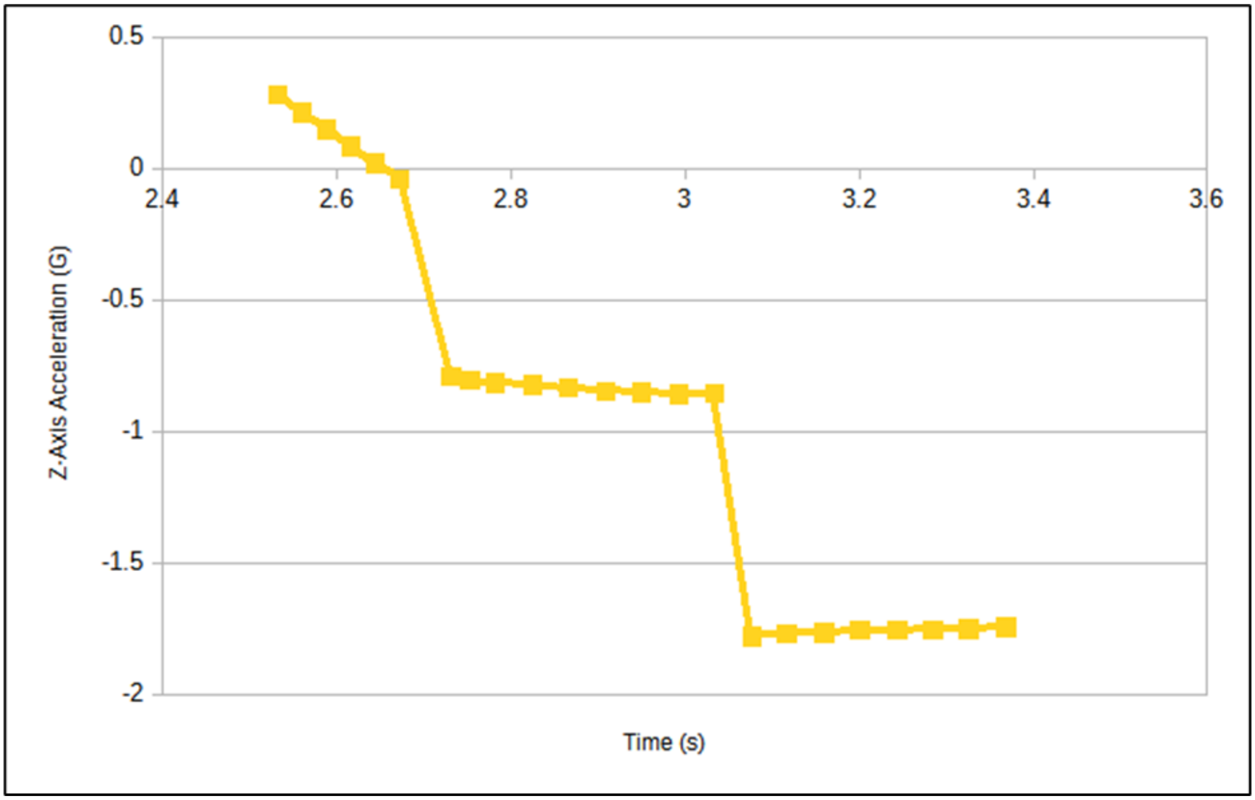


Figure 3.49: Z-Axis Acceleration vs. Flight Elapsed Time During Burnout

This information, along with the physical parameters of the rocket, is sufficient to calculate the drag coefficient of the rocket via the definition of quadratic drag:

$$F_{drag} = 1/2 * \rho * v^2 * C_d * A$$

$$m * a = 1/2 * \rho * v^2 * C_d * A$$

$$m = 8.24\ kg$$

$$a = 8.42\ m/s^2$$

$\rho = 1.20\ kg/m^3$ (based on environmental conditions of launch site)

$v = 176.9\ m/s$ (found by numerical integration of accelerometer data)

$$A = 8.2045 * 10^{-3} m^2$$

Plugging in variables yields:

$$69.4\,N = 154\,N * C_d$$

$$C_d = 0.451$$

Repeating the same calculation for the maximum drag measured after petal deployment gives the drag coefficient with petals deployed:

$$m * a = 1/2 * \rho * v^2 * C_d * A$$

$$a = 17.45\,m/s^2$$

$$v = 175.8\,m/s$$

$$144\,N = 154\,N * C_d$$

$$C_d = 0.934$$

3.7 Mission Performance Predictions

3.7.1 Summary of Mission Performance Calculations

Table 3.11: Mission Performance Calculations Summary

Official Target Competition Launch Altitude	5000 ft.
Landing Kinetic Energies	20.1 ft-lbf, 28.9 ft-lbf, 70.5 ft-lbf
Expected Descent Time	84.9s (1)
Expected Maximum Drift	2490 ft (1)

(1) Descent predictions are for the worst-case scenario, in which the altitude assurance system does not function at all and apogee is overshot. Nominal flights will yield lower drift.

Mission performance simulations were performed using the open-source software OpenRocket. The software is capable of modeling rockets and simulating flights. This software is free to use and is publicly available to download. The simulation was done using the extended Barrowman equations with six degrees of freedom (three translational axes and three rotational axes). A fourth order Runge-Kutta integration method was selected, along with a spherical approximation of the earth, which is sufficiently accurate for the purposes of this design.

Verifications for mission performance were performed using a custom MATLAB and Simulink simulator. This simulator numerically solves vehicle kinematics by accounting the vehicle's momentum in flight.

3.7.2 Flight Profile Simulation using OpenRocket

The flight profile simulation was performed using an 8° rail cant in 10 mph wind speed and a turbulence intensity of 10%. These results assume no functionality of Altitude Assurance. The results of the flight profile simulation performed using OpenRocket are shown below.

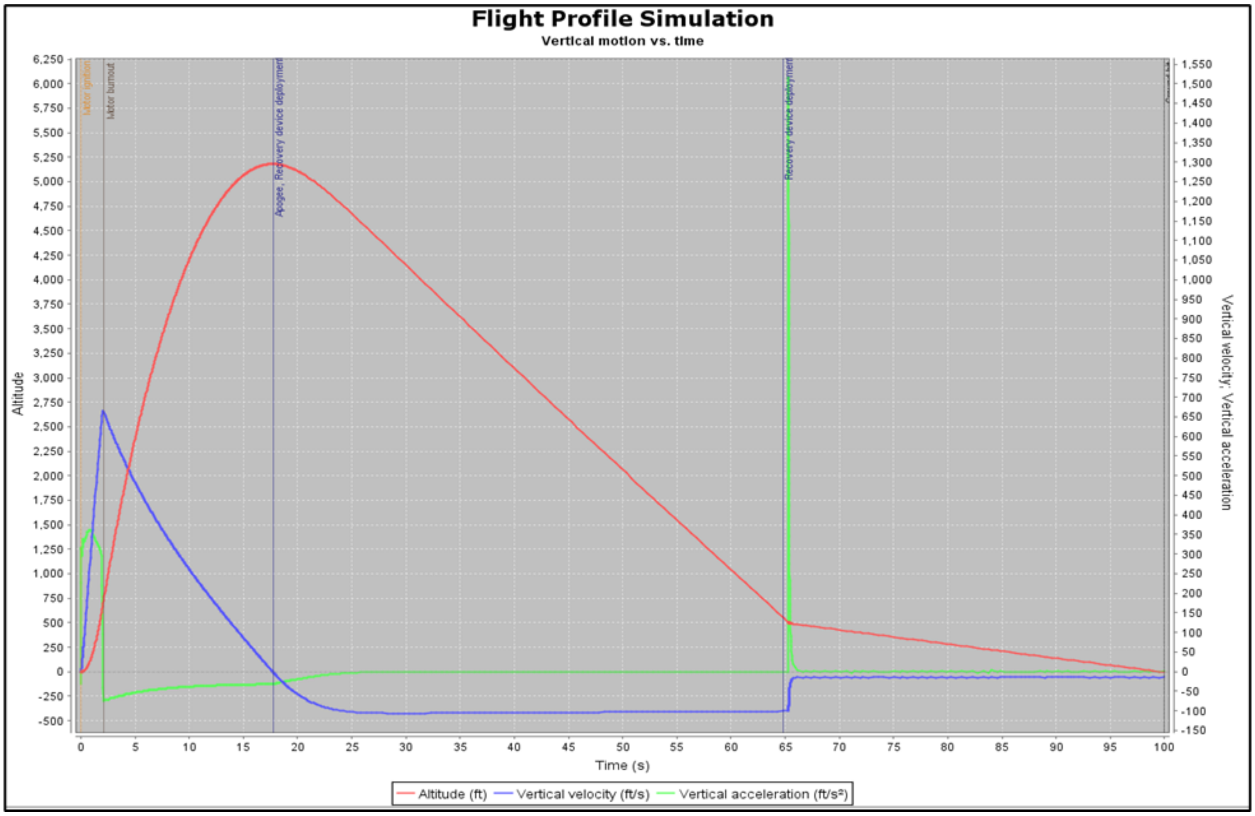


Figure 3.50: OpenRocket Flight Profile Simulation

Our target apogee for this mission is 5000ft. The flight profile simulation predicts an apogee of 5180 ft. which is valid for the project as our Altitude Assurance module will be capable of lowering the apogee to hit the project target.

3.7.2.1 Flight Profile Verification using Custom Simulator

The flight profile simulation verification was performed in similar conditions. The results of the flight profile simulation performed using the custom simulator are shown below.

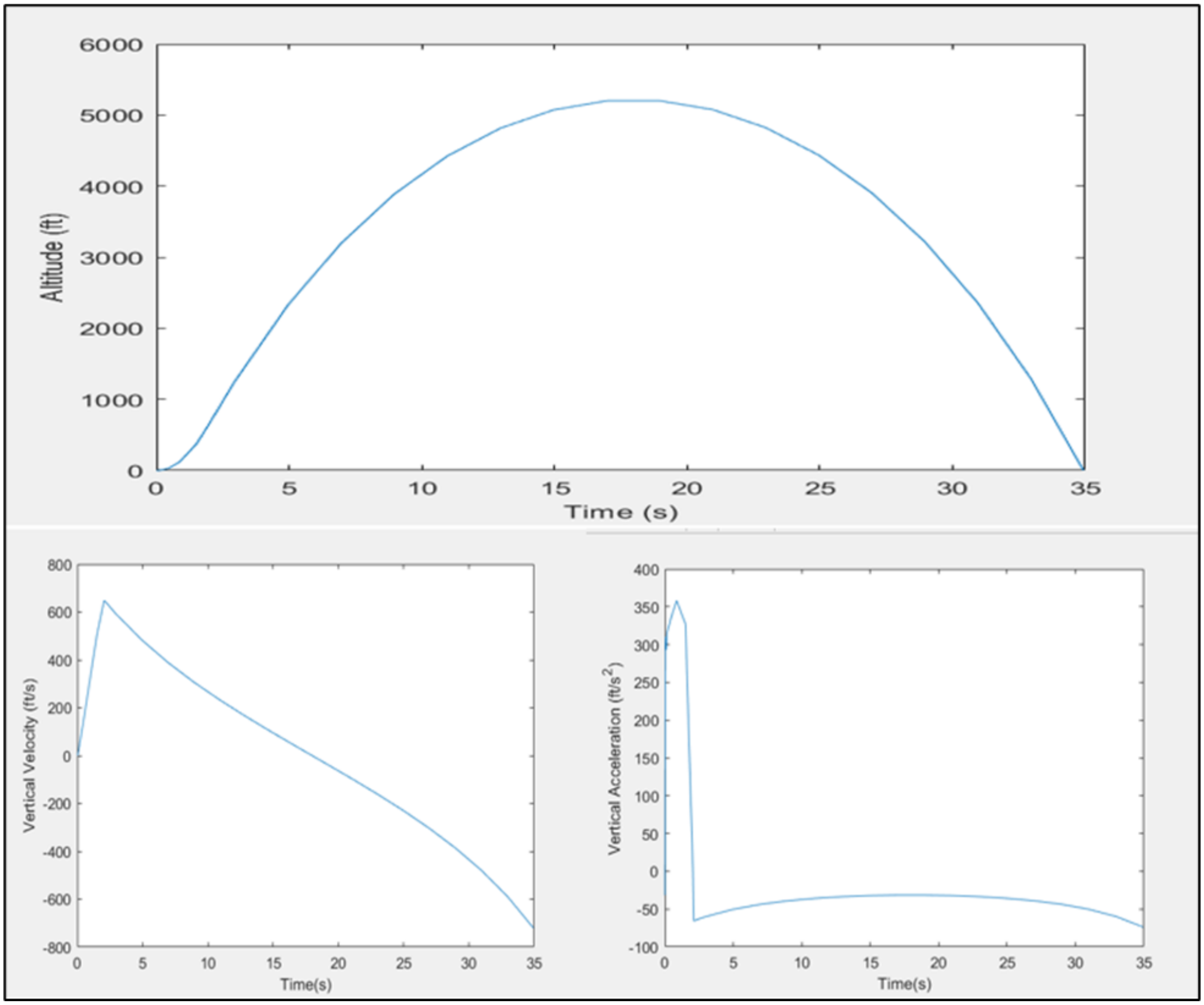


Figure 3.51: Custom Simulator Flight Profile Simulation

Table 3.12: Comparison Between OpenRocket and Custom Simulator Summary

	OpenRocket	Custom Simulator
Apogee (ft)	5180	5057
Max Vertical Velocity (ft/s)	665.8	645.1
Max Vertical Acceleration (ft/s^2)	362.3	357.6

The OpenRocket and custom simulator yielded very similar results. The major differences between these simulations is that the OpenRocket predicts that the vehicle will achieve higher apogee, maximum vertical velocity, and maximum vertical acceleration. This is to be expected because the custom simulator, for the sake of simplicity, neglects to model various attributes of the flight profile: namely, the deployment of recovery devices and an altitude varying air density. The latter contributes to a lower performing vehicle as high ground-level air density is used throughout the flight. This increases the drag seen by the vehicle from what would be physical in actual flight.

3.7.3 Static Stability Margin and CG/CP Relationships

The static stability margin of the vehicle was simulated using OpenRocket. The schematic and summary table below shows the static stability margin and relationships between the center of gravity and center of pressure.

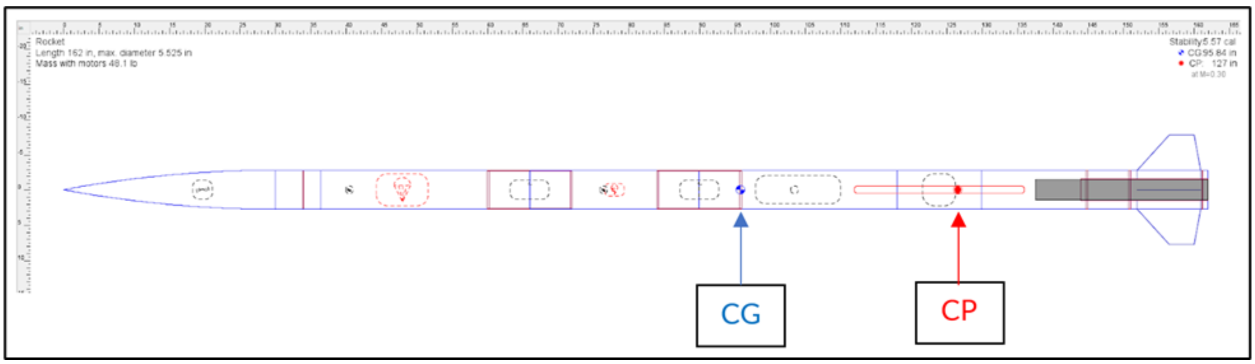


Figure 3.52: Static Stability Margin and CP/CG Relationships

Table 3.13: Summary of Static Stability Margin and CP/CG Relationships

Static Stability Margin	5.57 calibers
CG *	95.84 in.
CP *	127 in.

*Reported from the forward end of the rocket

3.7.4 Kinetic Energy Calculations

Kinetic energy calculations for each tethered section were performed for ground-hit events. These calculations used the ground-hit velocity reported by OpenRocket and the mass of each tethered section. A 20 mph wind speed was used to determine the worst-case results. The table below summarizes the kinetic energy calculations.

Table 3.14: Summary of Kinetic Energy Calculations

	Forward Section	Mid Section	Aft Section
Mass (lbm)	6.08	8.72	21.3
Ground-Hit Velocity (ft/s_	14.6	14.6	14.6
Kinetic Energy (ft-lbf)	20.1	28.9	70.5

Since the worst-case kinetic energy of each independent section does not exceed the 75 ft-lbf maximum mandated by the competition, the vehicle recovery design was deemed to be acceptable.

3.7.5 Expected Descent Time Calculation

The expected descent time calculations were based on the OpenRocket recovery simulation. The table below summarizes the expected descent time in a worst-case event of 5° launch rail cant with no Altitude Assurance functionality to ensure a 5000ft apogee.

Table 3.15: Summary of Expected Descent Time Calculation

Flight Time (s)	104
Time to Apogee (s)	18.4
Expected Descent Time (s)	85.6

3.7.6 Drift Calculations

The drift calculations were based on the OpenRocket recovery simulation. The main parachute is set to deploy at 600ft while the drogue parachute is set to deploy directly at apogee.These calculations assumed apogee was achieved directly over the launch pad and assumed realistic launch degree cant angles for each wind speed. A summary of these calculations are shown in the table below.

Table 3.16: Summary of Expected Drift Calculations

Wind Speed (mph)	Launch Rail Cant (degrees)	Expected Drift (ft)
0	5	206.4
5	5	356.2
10	10	919.1
15	10	1347
20	10	1961

4. Payload Criteria

4.1 Experiment and Criteria for Success

The vehicle payload will determine the location of its landing site. The launch field will be divided into a 250ft edge-length grid. The payload will determine which box the vehicle landed in by averaging two independent calculation methods. Our mission will be considered successful when, upon landing, our payload autonomously returns the correct gridded position of its landing location to the team, verified by GPS. This operation should be performed independently of a successful landing; no aspect of the payload should affect the deployment hardware and vice versa.

4.2 Overview of Payload Systems

Similar to the vehicle design process, the payload has been divided into individual subsystems, each with a specific purpose. Table 4.1 lists the payload subsystems and their objectives with varied purposes and names from the PDR.

Table 4.1: Payload Systems Summary

Payload Systems	Objective
Flight Computer	Process the data collected from the other subsystems, determine the location of the payload, control and power all electrical components on the rocket. The Flight Computer is the Raspberry Pi situated on the rocket.
Telemetry	Consists of an altimeter, independent GPS Transmitter, and the Xbee RF Modules that send the payload position and data back to the ground station. Located on the rocket.
Inertial Measurement Unit (IMU)	Includes the two accelerometers: the main accelerometer with a lower maximum acceleration, and a backup with a large range (BNO055 and H3LIS331, respectively) on the rocket that will measure distance traveled through use of time-integration analysis.
RF System	Consists of two SDRs, one rotating antenna at the ground station, one stationary antenna on the rocket, a bandpass filter, and an amplifier that are collectively

	used to determine the position of the payload via RF.
Power Delivery	Stores and delivers power to the payload on the rocket. Consists of a battery and buck converter on the rocket.
Ground Station Computer	Run the RF System at the ground station and manage the motion of the rotating Yagi-Uda antenna as well as receive and display telemetry data via a GUI. Consists of Raspberry Pi at the ground station.

The figure below (Figure 4.1) reiterates the information above with clear connections and locations for each subsystem.

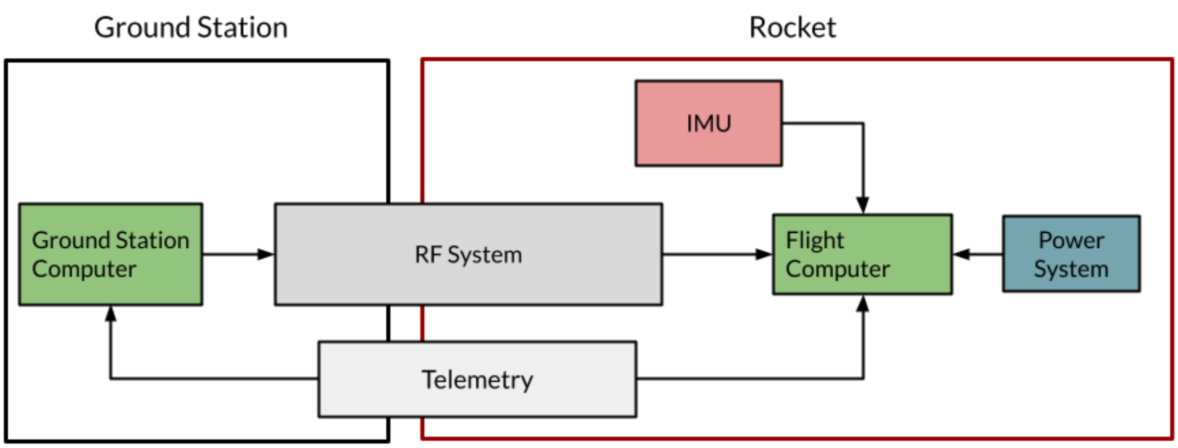


Figure 4.1: Payload Systems Block Diagram

The block diagram below (Figure 4.2) describes the components that make up the payload and how they will interface with each other to accomplish the payload goals.

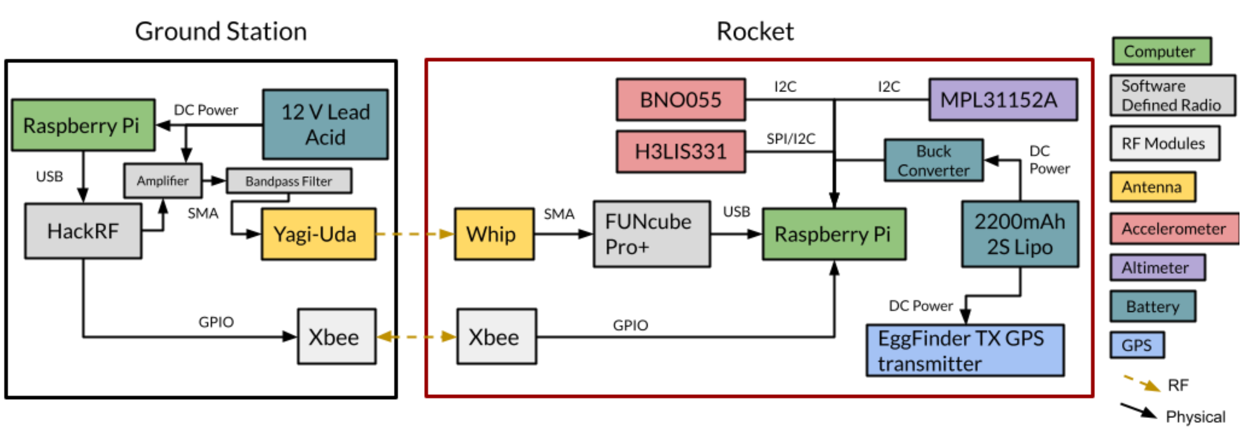


Figure 4.2: Payload and Ground Station Integration Block Diagram

The two locations that make up the payload system are the Rocket and the Ground Station. The Ground Station consists of the Ground Station Computer (Raspberry Pi 4), part of the RF System (HackRF One, Low Noise Amplifier, 426 MHz bandpass filter and the Yagi-Uda directional rotating antenna). It will have its own Xbee RF module for receiving the payload location and telemetry. The Rocket will have the other half of the RF System (Whip antenna, FUNcube Pro+), the Flight Computer (Raspberry Pi), the Power Delivery System (2200mAh 2S LiPo and buck converter), the IMU (BNO055 and H3LIS331), and the Telemetry System (Eggfinder TX GPS Transmitter, Xbee, MPL31152A). The Eggfinder TX GPS Transmitter will operate at a frequency of 915 MHz.

4.3 Expected Descent Time Calculations

On launch day at Huntsville, the team will record the location in GPS coordinates of the Ground Station and program it into the Flight Computer. This is necessary because the RF System locates with the Ground Station as a reference frame, so it needs to be placed on our grid. A screw switch turns on the power to the payload on the rocket. Once the rocket is set up on the rail, the screw switch will be screwed in and the payload will enter sleep mode. This means that the only subsystems active are the IMU, Fight Computer, and Power System.

Once an acceleration is detected by the IMU, the payload will enter active mode. The accelerometers and altimeter will begin collecting data, the Raspberry Pi will begin logging, and the Xbee will send data back to the ground station for monitoring purposes. The GPS Transmitter will log and transmit throughout the flight independent of the rest of the system. The payload will continue working as described until the main parachute deploys. At this point, the Yagi-Uda Antenna and the HackRF on the ground station will begin determining the angle of the payload. To do this, the rotating Yagi-Uda will swivel in 1 degree increments and send its angle to the rocket payload continuously. The angle where the strongest signal is received will be the angle from the ground station to the rocket.

Once the angle is determined, the Ground Station will send out another signal. The FUNcube Pro+ will record the power of the recorded signal. The Flight Computer will use this value and known parameters to determine the distance to the ground station according to the Friis Transmission Formula (Equation 1) and solving for distance.

$$P_r = \frac{P_t G_t G_r \lambda^2}{(4\pi R)^2}$$

(Equation 1)

Where P_r is the power at the receiving antenna, λ is the wavelength, G_r is the gain of the receiving antenna, G_t is the gain of the transmitting antenna, P_t is the power at the receiving antenna, and R is the distance between the antennas. After solving for distance, the Flight Computer will translate the polar coordinates (distance and angle) and translate them to rectangular coordinates. The Flight Computer will also, during the flight, double integrate the IMU data to determine location and will constantly be sending this information back to the Ground Station. The Flight Computer will then average the positions from both locating systems. The Flight Computer will have a preprogrammed grid that will include the coordinates (in feet) of each grid box. With the determined average location, the Flight Computer will find which box the coordinates correspond to and send it back via the Xbee module.

A design alternative from the Preliminary Design Review was chosen for each subsystem of the payload. The justification is outlined in each subsystem section as well as its integration with the payload system.

4.4 Flight Computer

Table 4.2: Control Systems Alternatives

Flight Computer	Pros	Cons	
Embedded Microcontroller	- Proven within other competition teams - Flexibility and Robust GPIO Support - Wide industry adoption	- Limited computation - Highly specialized programming - Low level software	
Discrete Electronics	- Speed - No sampling of analog signals - No toolchain	- Complex design - Relatively inflexible - Hard to debug - Larger footprint	
Embedded Linux	- Linux tools - Performance - High level languages - Native development - Code portability - Prior experience	- Size - High power draw - Complex software	✔
No Control System	- Simple - Reliable	- Sensors must directly determine position	

4.4.1 Design Decision

The team has chosen an embedded Linux computer as our leading payload control system. As justified in the PDR, no control system was quickly ruled out due to it being overly restrictive on the payload design. A control system using discrete electronics was ruled out due to its complexity and extensive prerequisite knowledge required. An embedded microcontroller was considered due to its real world uses, variety of GPIO pins, and readily available support from other Rose-Hulman competition teams and the internet; however, the team utilizes complex signal processing algorithms that will be better carried out in a high level programming language unavailable on embedded microcontrollers. Additionally, the low barrier to entry provided by being able to develop natively on the embedded linux board is something we value, especially as a rookie team.

The embedded Linux computer we decided on is a Raspberry Pi 4. Our team has extensive experience with Raspberry Pis, and a Pi can fulfill our needs for complex algorithms with its computing power and can integrate easily with our desired components. We picked a Raspberry Pi 4 because it has the most computing power of the Raspberry Pi line.

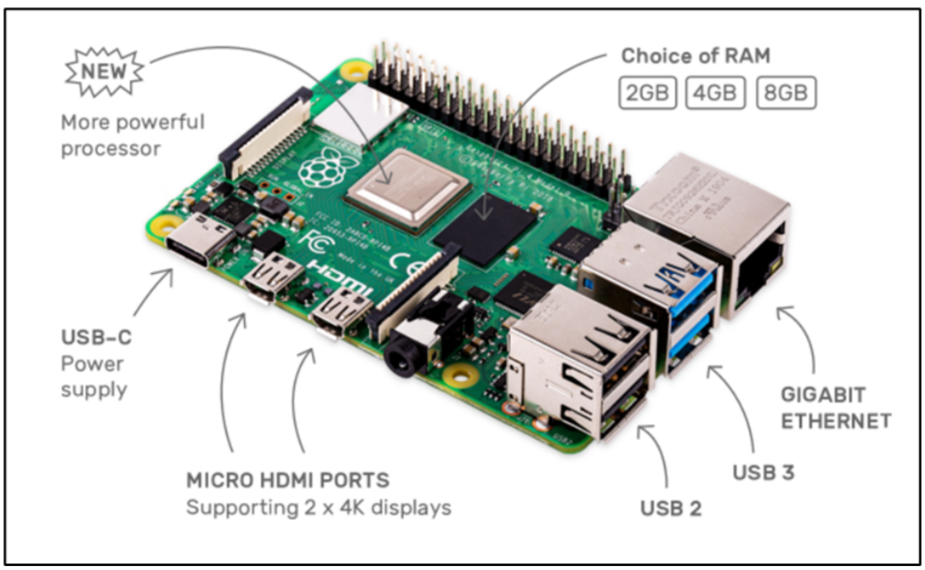


Figure 4.3: Raspberry Pi 4 Diagram

Source: raspberrypi.com/products/raspberry-pi-4-model-b/

4.4.2 Integration

The Flight Computer will integrate with the RF System through USB 2, the IMU through I2C and SPI, and the telemetry through serial. The Pi has specific pins for the I2C, SPI, and serial protocols. The Pi will handle all power distribution as no electrical component needs more than 5V. The battery will power the Pi through its voltage input pins. The voltage from the battery will be stepped down through a Buck Converter to achieve the Pi's 5V requirement. All the systems will be written in Python.

The code will take the information from the two locating systems (4.5 and 4.6) and translate it to cartesian coordinates relative to the known location of the ground station and landing grid. The RF and IMU systems will give data that will be interpreted as a distance from the launchpad and an angle, from here it is a simple calculation to convert to cartesian coordinates and then detect which segment of the grid contains the payload.

4.5 Telemetry Design

Table 4.3: Telemetry Alternatives

Telemetry Design	Pros	Cons	
Discrete component radio frequency (RF) circuit	- Mechanical robustness - No software failure	- Relatively inflexible - Difficult to test - Requires extensive background knowledge	
Commercial off the shelf (COTS) module	- Easy to develop and use - Proven technology - Experience within the team - Smaller footprint	- Software failure	✔

4.5.1 Integration

To transmit the data compiled and processed by the Flight Computer, the leading alternative telemetry system is a commercial off the shelf radio transmitter. The COTS module was chosen for its simplicity, reliability, ease of use, and prior team experience. The amount of knowledge required to create a discrete component radio frequency circuit makes this solution infeasible.

We determined an XBee RF module would be most effective as the module was already in the team's possession and team members have successfully implemented it in a prior project. Each XBee module is configured to only work with another specified, so it will not interfere with transmission from other teams. The model we chose is Digi XBee-PRO 900HP RF Module. This module transmits at a frequency that will not interfere with our RF System (426 MHz) as its range is 902-928 MHz. We will choose whatever frequency has the smallest interference at the launch site. In addition, it has more than sufficient range (9mi), so we can guarantee a link with the Ground Station.

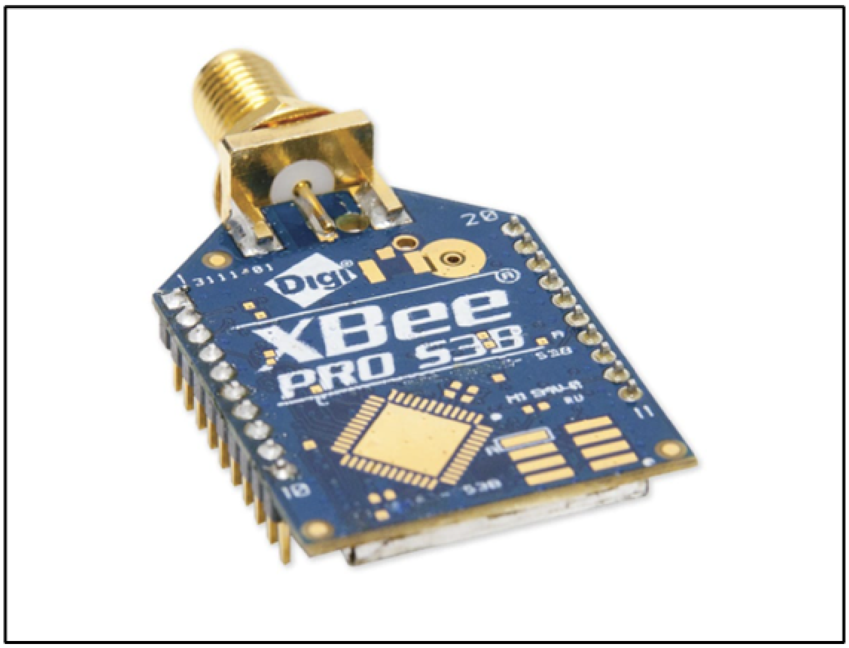


Figure 4.4: XBee Pro S3B Image

Source:
digi.com/products/embedded-systems/digi-xbee/rf-modules/sub-1-ghz-rf-modules/xbee-pro-900hp

The Xbee will interface with the Raspberry Pi through serial communication. GPIO pins 14 and 15 on the Pi are serial pins. The Xbee works with the RPi. GPIO Python library that has extensive online support. It will be powered through the Pi's power distribution at 5V.

In order to minimize RF interference between the transmitters and the electrical equipment on the payload, each non-transmitting component will be wrapped in conducting foil for electromagnetic shielding. To prevent short circuiting the foil will be placed between two insulating layers before being used on the payload system.

4.6 Inertial Measurement Unit System

Table 4.4: Payload Locating Alternatives

Sensor and Data Acquisition Design	Pros	Cons	
Time Integration	- No equipment external to the vehicle is required - Easy to compute - Easy to test	- Complex signal processing required to filter data - Possibly large amounts of error	✓
Radar	- No additional hardware on rocket	- Not in spirit of competition	
Passive RF Field	- Accurate - Significant prior art - No error accumulation	- Large hardware investment - Computationally complex - Needs hardware at ground station to work	✓
Visual Simultaneous Localization And Mapping (vSLAM)	- Can correct accumulated error - No external hardware is needed	- Complex signal processing - Computationally expensive	

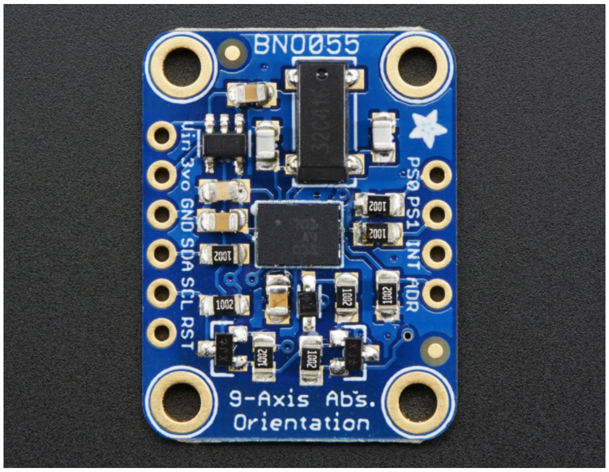
4.6.1 Design Decision

The determination of the location of the payload upon landing is planned to be distributed between two systems: Passive RF Field and Time Integration. These systems have been renamed to the RF System and the IMU System. Radar was eliminated as it was felt it is not in the spirit of the competition, and vSLAM was eliminated for being too out of scope to implement with the team's expertise. We chose two separate systems to ensure redundancy. Testing will be done to determine an algorithm to minimize the final location error with the independently derived location data of the IMU system and the RF system.

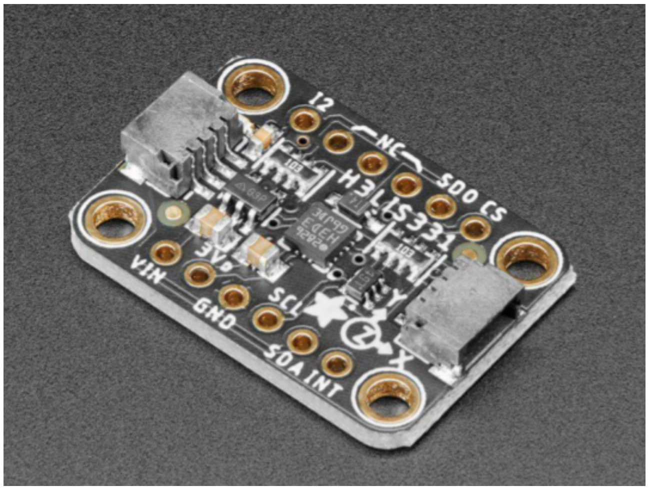
During the rocket flight, data from the low-range IMU will be used to calculate the position by integrating the acceleration data twice. During separation and chute deployment, the acceleration exceeds the maximum of the low range detector, saturating the data leading to critical error. Thus, at deployment phase, acceleration data from the high range IMU will be recorded and integrated with to minimize error. The high range

IMU cannot be used for low-acceleration phases of the flight as its resolution is low, which could lead to accumulated error with integration.

The two accelerometers chosen are the BNO055 (low-range) and the H3LIS311 (high-range) from Adafruit. The BNO055 can measure accelerations up to 16g, and the H3LIS311 can measure up to 30g. Our recovery is estimated to result in an acceleration of around 26g, which is within the range of the high-range accelerometer. At lower accelerations, the high-range has a higher noise level which justifies the use of the low-range to decrease error. We chose Adafruit hardware because of the extensive documentation as well as prior team experience.



Source:
learn.adafruit.com/adafruit-bno055-absolute-orientation-sensor



Figures 4.5, 4.6: Adafruit BNO055, Adafruit H3LIS311

Source:
cdn-learn.adafruit.com/downloads/pdf/adafruit-h3lis331-and-lis331hh-high-g-3-axis-accelerometers.pdf

4.6.2 Integration

Both accelerometers use I2C communication to interface with the Flight Computer. Raspberry Pis have two I2C buses on GPIO pins 2 and 3 and pins 0 and 1, but we will chain the two modules together. Adafruit provides Python libraries for both modules. The Flight Computer will record and log the data from both accelerometers throughout the flight and in real-time calculate distance from the data. The Flight Computer will translate the distance traveled to the gridded launch field, and, just before landing, will transmit the calculated location via the Xbee to the Ground Station. The accelerometers will be mounted with the rest of the payload on a shock-resistant plate.

4.7 RF System

The RF system consists of the following subsystems:

1) Ground Station System
2) Payload System

4.7.1 Ground System Station

This subsystem consists of a HackRF One SDR that transmits signals through a Yagi-Uda directional antenna. Between the HackRF One and the antenna are connected a low noise amplifier and a 426 MHz bandpass filter. This is done because transmissions will also be made at 426 MHz, within the amateur radio 440MHz frequency allocation. The antenna is mounted on a pole that is connected to a motor to facilitate rotation.

The rotating directional antenna has been chosen because it is simpler and more cost effective than a phased array antenna for transmission. The Yagi-Uda antenna has high directional gain that will help ensure as much power is transmitted to the rocket as possible. The HackRF is an COTS module with significant documentation and can be interfaced through GNUradio, a signal-processing and radio interfacing toolkit. GNU Radio Companion is the chosen software to be used on the ground station as it is compatible with Python that will be used on the Raspberry Pi. The HackRF One is connected to the Ground Station Computer (detailed in Section 4.8).

The Yagi-Uda antenna will be slowly rotated on a pole using a GT2 toothed belt drive attached to a NEMA 17 stepper motor. Using a transmission frequency determined by testing discussed in Section 6.1, a signal at 426 MHz is sent with the angle data. This is the angle between the line x=0 in the defined grid and the direction the Yagi-Uda antenna is facing. The exact angle and number of times the signal will be sent out will be determined through testing detailed in Section 6.1 (Locating Test for RF 2).

4.7.2 Payload Station

The FUNcube Pro+ has the highest resolution (16-bit) of any commercial SDR available and is also directly compatible with GNU Radio. The FUNcube Pro+ has a whip antenna and is connected to the payload computer.

The payload on the rocket receives the signals sent by the HackRF and records angle as well as signal strength.

In order to calculate the angle the payload is located with respect to the vertical axis in the coordinate system defined in Section 4.9, each recorded angle is weighted with respect to the signal strength and averaged. This method allows for much higher precision than is afforded with using individual signals and comparing maximum strength as the Yagi-Uda antenna has a signal range of 52°.

The distance is estimated using the loss equation described in Section 4.2.1. The final angle and distance is converted to cartesian coordinates on the payload computer using the relations $x = rcos\theta,\ y = rsin\theta$.

4.7.3 Ground Station SDR Design Decision Table

Table 4.5: Ground SDR Decision

Ground station SDR	Pros	Cons	
LimeSDR Mini	- Low cost	- High noise figure (2 dB)	
HackRF One	- GNU Radio compatibility - Low cost - Large user base (extensive documentation)	- Low sampling rate	✔
BladeRF 2.0 Micro	- High sampling rate	- Increasing noise with increasing frequency	
Ettus USRP B210	- GNU Radio compatibility	- High cost	

4.7.4 Payload SDR Design Decision Table

The most important aspect of the receiving SDR is the ADC resolution. It describes the accuracy at which the SDR translates the analog RF signal to a digital signal. The ADC resolution is important for the payload receiving antenna as accurate readings are required to calculate the distance of the payload from the ground station using the signal loss.

Table 4.6: Payload SDR Decision

Payload SDR	ADC Resolution	Price (estimate)	
RTL-SDR	8 bits	$20	
Funcube Pro+	16 bits	$200	✔
Airspy	10 bits	$200	
SDRPlay	8 bits	$150	

(source: https://www.rtl-sdr.com/about-rtl-sdr/)

4.8 Power Delivery

Table 4.7: Power Delivery

Power Delivery Design	Pros	Cons	
Disposable Cell	Durable Highest energy density Use in high-power rocketry well documented	Requires regular battery replacement Additional mounting and securing considerations	
Rechargeable Battery	High energy density Use in high-power rocketry well documented	- Lower cycle count - Susceptible to critical burst failure - Dangerous if not handled correctly	✔
Hybrid Supercapacitor	Low fire risk Rapid charging Extremely high power density	Unproven technology Poor charge stability compared to batteries	

4.8.1 Design Decision

The power delivery decision is a rechargeable LiPo battery, a standard rechargeable battery used for high-powered rocketry. We want to ensure every launch starts with a full battery, and replacing a disposable battery even if it is not empty is not economical. Hybrid supercapacitors would likely be the preferred option due to their weight efficiency, space efficiency, preferable charging characteristics, and low fire risk, but the novelty of the technology means they are expensive and have limited commercial availability.

4.8.2 Integration

The LiPo we select must supply all the rocket payload for a minimum of 2 hours according to requirement 2.7 in the handbook. We decided it should last at least 3 hours to ensure functionality during launch. The transmitter and altimeter selected for telemetry have sleep functionality, which will be engaged until the IMU detects launch; therefore, calculations assume that the transmitter and altimeter will be asleep for 2.5 hrs and awake for 0.5 hrs. This is an extremely conservative estimate relative to the flight time allotted by competition requirements. However, the power requirement for 2 hrs of continuous wakeful operation was also calculated to ensure competition power requirements can be met even in the case of payload malfunction. The table below summarizes all hardware components and their power draw for battery capacity determination. All power draw estimates are upper-bounds.

Table 4.8 Rocket Payload Component Power Draw

Component	System	Operating Voltage	Operating Current Draw	Power Required	Energy Required	Fail-safe Energy
Raspberry Pi 4	Flight Computer	5V	600 mA	3 W	9000 mWh	6000 mWh
FUNcube Pro+	RF System	5V	200 mA	1 W	3000 mWh	2000 mWh
BNO055	IMU System	3.6V	12 mA	44 mW	130 mWh	88 mWh
H3LIS331	IMU System	3.6V	.30 mA	1.1 mW	3.3 mWh	2.2 mWh
Xbee	Telemetry	3.6V	2.5 μA (sleep) 215 mA (transmit)	9.0 μW (sleep) 0.77 W (transmit)	250 mWh (limited)	1500 mWh
MPL3115A 2	Telemetry	3.6V	2.0 μA (sleep) 2.0 mA (record)	6.0 μW (sleep) 6.0 mW (record)	3.0 mWh	12 mWh
GPS module	Telemetry	4.6V	100 mA	100mA*3 hrs= 300 mAh	46 mWh	46mWh
				Totals:	12410 mWh	9650 mWh

The total energy required for every component in the rocket payload is 12410 mWh. However, conversion between voltages using a buck converter is not fully efficient, and power consumption while using internal voltage regulators is not well-documented.

Assuming a conversion efficiency of 80%, the energy required for all components is 15500 mWh. Voltage conversion is usually most efficient when stepping down and when the differential is minimized, and so a 2S battery (7.4v) was selected, meaning the required charge capacity is 2100 mAh. Although this exact capacity can be purchased, 2200mAh batteries are more common, and cells with more advanced chemistry can be purchased, decreasing weight. By selecting a 2200mAh, 2S battery, we can be confident our payload will last the minimum 2 hours on the rail, while minimizing required payload mass and volume.

The LiPo wires will be terminated with a XT60, and a corresponding connector will go into the arming switch and then to the buck converter. The buck converter will ensure 5V going to the Pi and will terminate on pin 2 and a ground pin on the Pi.

4.9 Ground Station Computer

The use of the Ground Station Computer is to interface with the Telemetry and RF Systems. The primary roles of the ground station computer are

1) Control rotational directional antenna motion
2) Interface with HackRF One for transmitting angle signal
3) Receive final coordinate from Payload with an XBee module

The Ground Station Computer will be a Raspberry Pi 4 as well because the HackRF needs to interface with a USB device, and a Linux computer works best with GNUradio. There will also be no new programming language or libraries needed to work with for the Xbee. A Python-based code will be executed on the ground station computer that rotates the transmitting antenna to a certain angle. Data that contains this angle is relayed through the computer to the HackRF One module that transmits the signal through the amplifier, bandpass filter, and finally the Yagi antenna. The signal is transmitted multiple times to reduce the error of recorded loss at the payload receiver.

After this process completes, the antenna changes its angle, and the process is repeated over 360^{o}.

Once the payload has identified the grid cell it is located in, it transmits its coordinates through XBee modules. The receiving XBee module is connected to the ground station computer and the payload location is recorded.
This system will undergo testing as elaborated in Section 6.1.

Table 4.9: Ground Station Computer Decision

Computer System	Pros	Cons	
Personal Computer	- No additional cost - Maximum computing power - Most convenient to use GNU Radio Companion	- Interfacing rotating directional antenna and SDR complicated - Additional error mode as PC is used for multiple purposes.	
Arduino	- Interfacing rotating directional antenna and SDR simplified	- Lowest computing power	
Raspberry Pi 4	- Interfacing rotating directional antenna and SDR simplified	- Higher upfront cost	✔

4.9.1 Integration

The Raspberry Pi will be connected to a monitor through HDMI, and a GUI displayed on it will include the GPS coordinates, altimeter reading, status checks during the flight, and finally, the location of the payload according to the IMU and the RF Systems. A mock-up of the GUI layout with example output in light gray is shown in Figure 4.7. The Updates section at the top of the GUI will display messages with status updates throughout the flight. We will have messages displayed every 10 min during sleep mode. It will send additional descriptive messages when the payload system enters active mode (launch), when the parachutes deploy, when the RF determines the angle and distance, and finally when it detects landing. This will aid in our final decision to use the location of the RF System or the IMU. A power strip will connect to a 12V Lead Acid car battery, and the Raspberry Pi will be connected with a 5V power regulator through the USB-C port on the Pi. A 5-sided metal (conducting) box that is open at the bottom will be placed over the ground station computer and other non transmitting electronics to minimize interference of transmitted RF signals from the antenna with the electronics of the system.

Figure 4.7: Mock-up GUI to be displayed at the Ground Station with example output

4.10 Gridded Launch Field

The required gridded launch field is shown in the figure below. Each box is 250ft on each edge and it is centered on the launch pad coordinates given. The grid is 20 by 20 for total dimensions of 5000ft by 5000ft. Each box is given a designation with a number and a letter.

Map data ©2022 Google

Figure 4.8: Launch Field Grid

4.11 Payload Integration into the Launch Vehicle

A structural analysis and overview of the payload bay is described in Section 3.4.1. The electronic components will be integrated via screws to the Payload Sled, and foam will fill the empty space to prevent jousling (not pictured for clarity). All components requiring the GPIO pins are near the said pins on the Raspberry Pi for easy wiring. The Battery and buck converter are on the opposite side of the Payload Sled to balance out the weight. The battery slides into the battery retainer and is zip-tied in place. The retainer is screwed into the payload sled. The arming switch is put on the flat section of the nose cone and is wired in series with the battery. The wires to the arming switch will be secured to the nose cone to prevent jouseling. The rest of the wiring will be soldered to each component.

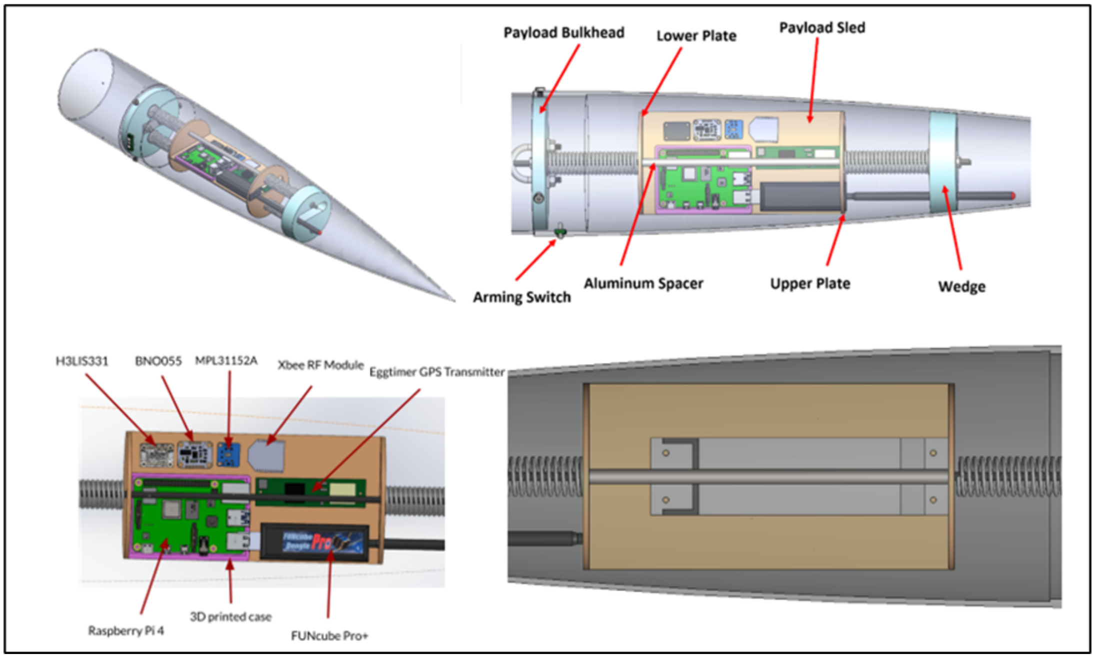


Figure 4.9: Payload Integration Overview

4.11.1 Integration

After the payload is screwed into the sled on launch day, it will be ensured that the arming switch is unscrewed. The arming switch will complete the circuit from the battery to the rest of the payload. When screwed in, the switch will put the payload into sleep mode until the accelerometers detect the launch acceleration. When the payload is successfully armed, the payload will beep once, and the Raspberry Pi's built-in programmable LED will turn on. We will use that to ensure the payload is successfully in sleep mode before putting it on the rail. The LED will turn on when the computer detects responses from all components. If a component is not responding or anything else is not working, the LED will blink and the Pi will continuously beep.

5. Safety

5.1 Launch Concerns and Operation Procedures

In order to ensure that various steps of the project in its construction and assembly are completed safely, the team has created a set of descriptive checklists and procedures for team members to follow for launch preparation.

These procedures and checklists are included in Appendix sections 7.2 through 7.5.

5.2 Summary of Hazard Analysis Methodology

Table 5.1: Probability of Event

Category	Value	Description
Improbable	1	Less than 10% chance
Unlikely	2	10-35% chance
Possible	3	35-65% chance
Likely	4	65-90% chance
Probable	5	Greater than 90% chance

Table 5.2: Severity of Event

Category	Value	Human Impact	Equipment Impact	Mission Impact
Negligible	1	Minor or none	Minor or none	No disruption
Marginal	2	Minor injury	Minor damage	Proceed with caution
Moderate	3	Moderate injury	Repairable equipment failure	Flight delayed until event resolved
Critical	4	Serious injury	Partially irreparable equipment failure	Flight does not proceed until system removed
Catastrophic	5	Life threatening or debilitating injuries	Failure resulting in total loss of system or equipment	Flight canceled or destroyed

Table 5.3: Mapped Risk Assessment Matrix

Category	Negligible	Marginal	Moderate	Critical	Catastrophic
Improbable	1	2	3	4	5
Unlikely	2	4	6	8	10
Possible	3	6	9	12	15
Likely	4	8	12	16	20
Probable	5	10	15	20	25

5.3 Personnel Hazard Analysis

Table 5.4: Personnel Hazard Identification

Identified Hazard	Causes	Effects	Mitigations
Fire	- Open flames - Mishandling of equipment - Improper wiring	- Severe burns - Loss of part or project - Death	- Store flammable substances in flammables cabinet, fire extinguisher placed nearby, no open flames, test circuitry before use
Airborne particle exposure	- Sanding dust - Metal shavings - Paint - Aerosols	- Skin laceration or irritation - Eye damage - Respiratory distress	- Proper use of PPE and safety training, use paint booth and ventilated workspace where necessary
Electric Shock	- Improper wiring - Device failure - Test equipment misuse	- Extreme personal injury - Hardware damage/loss - Mission delays	- Members will not work alone and will be trained on use of high-voltage electrical equipment
Entanglement with machines	- Improper use of machinery - Machinery failure	- Severe lacerations - Crushed limbs - Fatal injuries	- Use PPE, follow dress codes in machine shops, adhere to required safety training
Epoxy Contact	- Surface contamination - Broken PPE - Resin spill	- Skin irritation - Eye irritation - Epoxy sensitivity	- Discard broken PPE, limit exposure, wear proper PPE, limit use to specified working surfaces

Eye Irritants	- Solder and epoxy fumes - Flying debris - Airborne particles	- Possible temporary vision loss - Eye irritation - Blindness	- Wear proper PPE, document irritants and limit exposure, use workspace ventilation booth, locate and train on use of eyewash station for every team member
Falling tools or materials	- Mounting failure - Improper use of storage racks	- Tool damage - Storage rack damage - Personal injury	- Store frequently used tools in easy to access locations, adhere to 5S standards of lean production
Fiberglass Contact	- Airborne particles created during fabrication - Fiberglass skin irritation	- Skin irritation - Respiratory Issues - Splinters	- Wear N95 respirators during fabrication, only sand fiberglass in sanding booth
Flying debris	- Improper use of machinery - Machinery failure	- Blunt force trauma - lacerations	- Maintain a safe distance from machines under operations, ensure those working on machinery are properly certified by the BIC
Exposure to Hazardous Fumes	- Working with inadequate ventilation - Improper soldering and welding practices - Epoxy handling - Activities from other teams in shared workspace	- Eye irritation/damage - Lung irritation/damage - Lightheadedness - Shortness of breath and nausea - Possible nerve damage	- Maintain proper PPE when working with fuming materials or maintain a safe distance from fuming materials in a well-ventilated environment
Hazardous Waste Contact	- Chemical spills - Incidental contamination	- Skin contact may cause rashes to burns - May require hospitalization	- Follow hazardous waste disposal techniques set by BIC/KIC

Exposure to Unsafe Noise Levels	- Use of BIC/KIC machine shop - Loud power tools - Other BIC/KIC teams	- Increased rate of higher frequency hearing damage	- Use proper PPE, maintain a safe distance from active machinery
Improper use of tools	- Use of BIC/KIC machine shop - Soldering irons	- Damage to equipment is unlikely - Injury may range from deep lacerations - Burns to lost fingers	- Ask BIC/KIC personnel or team Safety Officer before using high-risk tools, attend BIC safety training
Soldering or Welding Injuries	- Worker inattentiveness - Distractions during fabrication - Lack of fixturing equipment	- Second or third-degree burns - Hardware damage due to reflex response	- Only solder and weld during work hours and in predefined locations, make sure all personnel are aware when work is being performed, use sufficient fixturing equipment
Tripping	- Carrying unsafe loads - Unclean workspace - Worker inattentiveness	- Equipment damage - Sprains and bruises - Fractured bones, concussion, death (unlikely)	- Maintain well lit work areas. Adhere to 5S workspace standards of organization. Maintain walking areas.
Contact with Launch Vehicle Debris	- Faulty parachute ejection - Severe winds	- Blunt damage to the rocket or payload - Concussion - Fractured skull - Death	- Keep a close eye on the vehicle or have someone spot the vehicle for those who are unable
Launchpad Fire	- Flammable debris blown across launch pad - Flammable fuel spilled	- Heat damage to parachute - Motor - Electronics	- Remove brush, dry debris, and other flammables around the launch pad area and have a fire extinguisher on hand

Personnel Injury from Terrain	- Uneven footing, potholes, nails, etc.	- Sprained or broken ankles - Small puncture wounds	- Watch footing around terrain, travel in groups, maintain cell phone contact
Airborne Debris	- High wind speeds - Systems on the rocket breaking mid-flight	- Blunt force trauma - Lacerations	- Maintain a reasonable and safe distance from energetic devices
Contact Burns	- Contact with motor after flight - Standing too close to the launchpad	- Mild to severe burns	- Proper handling of the rocket will be used - NAR-mandated setback distances will be observed
Heat Stroke	- Prolonged exposure in a high-temperature environment	- Possible hospitalization	- Ensure team members limit exposure to dangerously high temperatures - Provide water
Hypothermia	- Failure to wear appropriate clothing	- Possible hospitalization	- Ensure team members limit exposure to dangerously low temperatures
Dehydration	- High environment temperature - Low fluid consumption	- Fatigue - Dizziness - Confusion - Immediate medical treatment	- Ensure access to cool drinking water at team events - Provide shaded areas available for rest

Table 5.5: Personnel Hazard Mitigation

Identified Hazard	Pre - Mitigation Risk (Probability/Severity/Total)			Post - Mitigation Risk (Probability/Severity/Total)		
Fire	2	5	10	2	4	8
Airborne particle exposure	3	3	9	2	2	4
Electric Shock	2	4	8	2	3	6
Entanglement with machines	3	5	15	2	5	10
Epoxy Contact	4	2	8	2	2	4
Eye irritation	3	4	12	2	4	8
Falling tools or materials	2	4	8	2	2	4
Fiberglass Contact	3	3	9	1	2	2
Flying debris	2	4	8	2	1	2
Exposure to Hazardous Fumes	4	3	12	1	3	3
Hazardous Waste Contact	2	3	6	2	2	4
Exposure to Unsafe Noise Levels	3	3	9	3	1	3
Improper use of tools	3	3	9	1	2	2
Soldering or Welding Injuries	4	2	8	3	1	3
Tripping	2	3	6	2	2	4
Contact with Launch Vehicle Debris	1	5	5	1	3	3

Launchpad Fire	2	3	6	1	3	3
Personnel Injury from Terrain	2	2	4	1	2	2
Airborne Debris	3	3	9	3	2	6
Contact Burns	1	4	4	1	3	3
Heat Stroke	3	3	9	2	2	4
Hypothermia	1	3	3	1	2	2
Dehydration	3	3	9	2	2	4

5.4 Failure Modes and Effects Analysis (FMEA)

5.4.1 Vehicle System FMEA

Table 5.6: Vehicle Systems FMEA Hazard Identification

Identified Hazard	Causes	Effects	Mitigations
Structural Failure Under Intended Loading	- Inadequately-designed structure - Not all failure modes considered during analysis - Material defects during construction	- Unpredictable competition performance - Vehicle cannot be reflown - Falling debris exceeds competition limits for kinetic energy upon landing	- Design airframe to withstand compression load at a safety factor of 2
Airframe Overloaded During Launch	- Motor improperly packed - Loose components cause local shock loading - High winds - Improper parachute deployment	- Falling debris exceeds competition limits for kinetic energy upon landing	- Multiple checks to internal packing - System testing with a variety of parameters

Hidden Structural Damage Prior To Launch	- Accidental damage during transportation or construction	- Falling debris exceeds competition limits for kinetic energy upon landing	- Check for cracks and material inconsistencies during construction
Structural Damage During Landing	- Miscalculation of landing energy or improper parachute deployment	- Significant repairs needed	- Test recovery system extensively
Bond Line Failure	- Lack of checks to bond line Rushed construction	- Falling debris exceeds competition limits for kinetic energy upon landing	- Multiple checks to bond lines
Component Mounting Failure During Launch	- Failure to utilize correct mounting techniques	- Launch failure - Destruction of component	- Multiple checks to mounting - Tests of mounting techniques
Structural Failure Of Deployment Systems	- Improper design of deployment subsystem - Construction errors	- Falling debris exceeds competition limits for kinetic energy upon landing	- Multiple checks of deployment systems during launch - Tests of deployment systems
Structural Failure During Deployment	- Insufficient damping in parachute attachment - Construction errors - Jammed structures	- Mission failure	Same as above
Aerodynamic Instability	- Location of masses change within the vehicle - Dynamic instability due to drag flaps	- Vehicle exceeds competition limits for kinetic energy on landing	- Static stability margin is measured as part of preflight checklist - Final vehicle configuration is tested at Vehicle Demonstration Flight

			- Drag flaps will command closed if high vibrations are detected
Electronics Failure Of Deployment Systems	- Parts dead on arrival - Insufficient charge of battery - Damage from aerodynamic forces	- Unpredictable competition performance - Vehicle does not separate - Vehicle exceeds competition limits for kinetic energy upon landing - Personal injury	- Remove-before-flight tag arms vehicle - Dissimilar redundancy in altimeter selection - Test altimeters upon arrival and before flight
Electronics Fire	- Overcharge of battery - Short circuit wiring	- Vehicle and/or falling debris exceeds competition limits for kinetic energy upon landing	- Teach all members the proper handling of the batteries and wiring - Multiple checks for proper wiring
Battery Depletion During Launch	- Unintended draw on electronics - Battery is not charged prior to launch	- Deployment electronics not functional - Flight altimeter not functional for scoring - Vehicle exceeds competition limits for kinetic energy upon landing	- Tests of battery under launch conditions - Potential redundant battery systems
Failure Of Airframe To Separate	- Over-tight fitting tolerances between airframe components	- Vehicle exceeds competition limits for kinetic energy upon landing	- Tests of airframe separation

	- Unintended mechanical locking between airframe components		
Internal Hardware Damaged During Separation	- Damage to internal electronics	- Failure to successfully calculate and to test the recovery system	- Test the recovery system multiple times
Recovery Hardware Does Not Eject	- Damage to airframe, electronics, and possible damage to property	- Vehicle exceeds competition limits for kinetic energy upon landing	Same as above
Damage To Parachute	Same as above	Same as above	Same as above
Parachute Does Not Open	Same as above	Same as above	Same as above
Excessive Vehicle Drift During Recovery	- Failure to test and successfully simulate recovery system	- Vehicle exceeds competition limits for recovery drift	Same as above
Altitude Assurance Initialization Failure	- Failure to test, successfully simulate, and properly construct altitude assurance	- Flaps do not actuate, apogee overshoot - Flaps actuate before burnout, destabilization	- Extensively test, validate simulations, and carefully construct altitude assurance
Altitude Assurance Control Scheme Failure	- Excessive loads jam control mechanism - Faulty control logic - Incorrect apogee prediction model	Same as above	- Final vehicle configuration is tested at Vehicle Demonstration Flight
Altitude Assurance Does Not Halt At Apogee	Same as above	Same as above	- Final vehicle configuration is tested at Vehicle Demonstration Flight

Mechanical Failure Of Altitude Control Hardware	Same as above	Same as above	Same as above
Structural Failure Of Altitude Control Hardware	Same as above	- Falling debris exceeds competition limits for kinetic energy upon landing	- Altitude Control Structure will be designed with a factor of safety appropriate for critical systems.
Uneven Deployment Of Drag Flaps	- Failure to test and successfully simulate drag flaps	- Aerodynamic instability of launch vehicle - Failure to deploy recovery systems - Vehicle exceeds competition limits for kinetic energy upon landing	- Testing and successfully simulating drag flaps
Motor Cannot Ignite	- Faulty product or packing of motor - Faulty igniter installation	- Vehicle fails to launch - Failure to compete with all other systems	- Test motor packing and ensure product is in good condition - Multiple sign-offs on motor assembly and installation Igniter retention using support rod
Motor Does Not Provide Design Thrust	- Faulty product or packing of motor	- Vehicle fails to reach 4000 ft	- Altitude Assurance actively adjusts flight trajectory if too much thru
Motor Explodes	- Imperfections in motor grain packing cause localized high pressure regions	- Mission fails	- Test motor and check datasheets for verification

Motor Retention Mechanism Breaks	- Imperfections in motor grain packing cause localized high pressure regions	- Falling debris exceeds competition limits for kinetic energy upon landing	Same as above
Motor Misalignment	- Poor construction quality of motor mount	- Unpredictable vehicle trajectory	Same as above
Motor Damages Internal Components	- Heat conduction through structure - Failure of bulkhead	Same as above	Same as above

Table 5.7: Vehicle Systems FMEA Hazard Mitigation

Identified Hazard	Pre - Mitigation Risk (Probability/Severity/Total)			Post - Mitigation Risk (Probability/Severity/Total)		
Structural Failure Under Intended Loading	2	3	6	2	2	4
Airframe Overloaded During Launch	2	4	8	2	2	4
Hidden Structural Damage Prior To Launch	1	4	4	1	2	2
Structural Damage During Landing	3	3	9	2	3	6
Bond Line Failure	3	4	12	2	3	6
Component Mounting Failure During Launch	2	4	8	1	3	3
Structural Failure Of Deployment Systems	3	4	12	2	2	4
Structural Failure During Deployment	3	3	9	2	2	4
Aerodynamic Instability	4	3	12	3	3	9
Electronics Failure Of Deployment Systems	2	4	8	2	2	4
Electronics Fire	1	5	5	1	3	3
Battery Depletion During Launch	2	4	8	2	2	4
Failure Of Airframe To Separate	4	5	20	3	4	12
Internal Hardware Damaged During Separation	2	3	6	1	3	3
Recovery Hardware Does Not Eject	3	5	15	2	4	8
Damage To Parachute	2	4	8	1	4	4

Parachute Does Not Open	3	5	15	2	5	10
Excessive Vehicle Drift During Recovery	2	2	4	2	1	2
Altitude Assurance Initialization Failure	2	2	4	2	1	2
Altitude Assurance Control Scheme Failure	2	2	4	2	1	2
Altitude Assurance Does Not Halt At Apogee	3	2	6	2	2	4
Mechanical Failure Of Altitude Control Hardware	3	4	12	2	3	6
Structural Failure Of Altitude Control Hardware	3	2	6	2	2	4
Uneven Deployment Of Drag Flaps	2	4	8	2	3	6
Motor Ignition Incapability	1	4	4	1	3	3
Motor Does Not Provide Design Thrust	2	4	8	1	3	3
Motor Explodes	1	5	5	1	4	4
Motor Retention Mechanism Breaks	1	4	4	1	3	3
Motor Misalignment	2	4	8	1	3	3
Motor Damages Internal Components	2	4	8	1	3	3

5.4.2 Payload and Payload Integration FMEA

Table 5.8: Payload and Payload Integration FMEA Hazard Identification

Identified Hazard	Causes	Effects	Mitigations
Mounting Failure During Flight	- Rushed implementation or lack of training	- Damaged payload bay	- Multiple checks
Mounting Failure During Landing	Same as above	Same as above	Same as above
Hardware Misassembly	Same as above	Same as above	- Bench test payload prior to launch
Faulty Control Logic	- Oversight or lack of checks	Same as above	- Multiple checks from multiple people to ensure correct logic
Failure to Arm Electronics	- Oversight or lack of checks	- Mission Failure	- Embed a master switch to enable the electronics for the vehicle - Train students to enable switch when not enabled
Failure to Detect Landing	- Failure to test sensors - Incorrect wiring	- Premature determ ination of vehicle location	- Testing of sensors under multiple conditions
Wiring Failure Between Controller and Hardware	- Oversight or lack of checks - Improper placement of electronics bay - Loose or misassembled components	- Electronics fire - Effects range from small burnout on pins to explosion mid flight	- All electronics will be checked by all students before the launch

Telemetry Transmission/Reception Failure	- Interference - Parachute Interrupts Telemetry	- Miscommunication with other sensors and main controller	Same as above
Sensor Hardware Failure	- Parachute covers sensors - Aerodynamic effects influence barometric readings - Mismounting or misalignment of	- Bad readings to determine location	Same as above
Battery Depletion Prior to Data Transmission	- Lack of testing	- Loss of the sensor data Failure of payload competition	- Test the battery under launch conditions
Debris	- Debris not removed from launch site	- Interference with the launch vehicle causing a postponed launch to mission failure	- Clear area before the launch
Premature Deployment	- Deployment charge self-ignites - Deployment electronics trigger charge early	- Vehicle exceeds competition drift limit	- Testing of the launch vehicle and verification of simulations
Late Deployment	- Failure to successfully calculate and to test the recovery system	- Vehicle exceeds competition limits for kinetic energy upon landing	- Testing of the launch vehicle and verification of simulations
Failure To Arm Electronics	- Oversight of electronics arming	- Vehicle exceeds competition limits for kinetic energy upon landing	- Remove-before-flight tag arms vehicle - Electronics arming is made explicit in pre-flight checklist

Table 5.9: Payload and Payload Integration FMEA Hazard Mitigation

Identified Hazard	Pre - Mitigation Risk (Probability/Severity/Total)			Post - Mitigation Risk (Probability/Severity/Total)		
Mounting Failure During Flight	2	5	10	2	4	8
Mounting Failure During Landing	3	3	9	2	2	4
Hardware Misassembly	2	4	8	2	3	6
Faulty Control Logic	3	5	15	2	5	10
Failure to Arm Electronics	4	2	8	2	2	4
Failure to Detect Landing	2	4	8	2	2	4
Wiring Failure Between Controller and Hardware	3	3	9	1	2	2
Controller Hardware Failure	2	4	8	2	1	2
Telemetry Transmission/Reception Failure	4	3	12	1	3	3
Sensor Hardware Failure	2	3	6	2	2	4
Battery Depletion Prior to Data Transmission	3	3	9	1	2	2
Debris	1	3	3	1	2	2
Premature Deployment	2	2	4	2	1	2
Late Deployment	2	2	4	1	2	2
Failure To Arm Electronics	2	4	8	2	2	4

5.5 Environmental Concerns

Table 5.10: Environmental Hazards Identification

Identified Hazard	Causes	Effects	Mitigations
Launchpad fire	- Dry environment - Flammables near launchpad during motor ignition	- Grass fire - Charred launch field	- Launch pad cleared as part of pre-flight checklist
Fire at landing site	- Dry environment - Unintentional motor ejection	- Launch field fire	- Motor will not protrude past aft end of vehicle
Collision with spectator drones	- Launch environment carelessness	- Possible complete mission failure	- Visually verify safe launch conditions prior to ignition, and coordinate with range safety officers to verify conditions at time of launch
Vehicle Fouled by Foreign Objects	- Unclean team preparation area	- Cascaded mission hazards	- Vehicle and payload inspection as part of pre-flight checklist
Inclement Weather	- Poor launch planning	- Component material embrittlement	- Independently measure launch conditions, and/or coordinate with other teams and range safety officers to verify conditions at time of launch

Wet Launch and Landing Sites	- Prior inclement weather effects present launch conditions	- Component material weathering	- Design vehicle to withstand wet environments
Components overheat on launchpad	- Overexposure to sun - High temperature launch day conditions	- Component material melting or failure	- Ensure proper protection of mission components on launch day as part of launch day guidelines
Launch debris left on site	- Rocket ejects debris during flight - Failure to collect waste generated during mission operations - Catastrophic mission failure	- Littering during launch operations	- Track waste generated during launch operations and provide trash bags for immediate disposal - Design vehicle to fail in minimal independent sections - Construct external vehicle components from materials that can be visually identified at the launch site - Visual environmental inspection as part of post flight checklist
Vehicle lost on recovery	- Recovery subsystem failure - Vehicle destruction	- Failed mission - Littering during launch operations	- Ensure redundancy in recovery design
Team equipment left on site	- Negligence of launch day operations	- Equipment must be repurchased	- Post flight checklist
Launch vehicle stuck in tree	- Unintended collision trajectory	- Potential vehicle and payload loss	- Do not perform test launches at sites with trees.

Launch vehicle collision with structures	- Unintended collision trajectory - Wind turbines and buildings present at launch fields	- Launch vehicle and payload destruction - Potential damage to structures	- Evaluate launch day conditions with special consideration to intended vehicle trajectory as part of pre-flight checklist

Table 5.11: Environmental Hazards Mitigation

Identified Hazard	Pre - Mitigation Risk (Probability/Severity/Total)			Post - Mitigation Risk (Probability/Severity/Total)		
Launchpad fire	3	4	12	2	3	6
Fire at landing site	2	4	8	1	2	2
Collision with spectator drones	2	4	8	1	4	4
Vehicle Fouled by Foreign Objects	1	3	3	1	2	2
Inclement Weather	1	5	5	1	1	1
Wet Launch and Landing Sites	2	2	4	1	2	2
Components overheat on launchpad	3	3	9	2	3	6
Launch debris left on site	2	3	6	1	3	3
Vehicle lost on recovery	3	5	15	2	5	10
Team equipment left on site	2	3	6	1	3	3
Launch vehicle stuck in tree	2	5	10	1	5	5
Launch vehicle collision with structures	2	5	10	1	5	5

5.6 Project Risks

Table 5.12: Project Risk Hazards Identification

Identified Hazard	Causes	Effects	Mitigations
Time	- Poor time management - Improper delegation of tasks - Students shifting focus away from competition	- Document Writing/Vehicle Fabrication is rushed - Failure to meet deadlines	- Establish a reasonable timeline and adhere to it - Evenly distribute tasks among students
Miscommunication	- Students not requesting help - Poor attitude towards people and leadership	- Project requirements are completed incorrectly - Project requirements are not completed because they are assigned to no one	- Have a good relationship with the team - Foster a friendly and inviting atmosphere
Scope	- Failure to maintain focus on core design - Adding too many features that may deviate from requirements	- Project becoming infeasible due to complexity	- Stay on track of project plan - Regularly reevaluate our design requirements
Resource	- World-wide shortages - Equipment breaking down - Students unable to participate	- Insufficient resources to complete project	- Order parts as early as possible
Budget	- SGA not providing us enough funding - No sponsorships	- Insufficient funds to finish vehicle advancements	- Request for funding early on in the process to avoid late delivery

Performance	- Wrong motor type or poor selection of vehicle aerodynamics	- Not enough thrust to reach desired apogee - Overshooting the vehicle beyond 6000 feet	- Testing in environments similar to launch site

Table 5.13: Project Risk Hazards Mitigation

Identified Hazard	Pre - Mitigation Risk (Probability/Severity/Total)			Post - Mitigation Risk (Probability/Severity/Total)		
Time	5	5	25	4	2	8
Communication	3	3	9	2	2	4
Scope	2	3	6	2	2	4
Resource	3	4	12	2	4	8
Budget	4	4	16	4	3	12
Performance	3	4	12	3	2	6

6. Project Plan

6.1 Testing

6.1.1 Recovery

6.1.1.1. Black powder recovery test

Prior to every vehicle launch, the full vehicle and recovery system must be tested to ensure a successful recovery on launch day. The test objective will be the vehicle recovery system, which includes parachute packing, black powder charges, and electronics. The team will vary the amount of black powder used, starting with the amount calculated in section 3.5.2.4

The test will begin with a full assembly of the launch vehicle, including both parachutes properly packed. This will not include any batteries, black powder, pressurized air, or other energetic devices. This will demonstrate the vehicles readiness to be assembled on launch day. The vehicle will then be disassembled, taken to an approved testing site outdoors on campus, and reassembled with blackpowder. The procedure outlined in Appendix 7.6 will be followed. The recovery system will be deployed using a 12 volt battery system. The recovery will be considered a success when, for each parachute, the parachute and deployment bag or chute proctor are fully ejected from the launch vehicle. And, the recovery harness is completely pulled out of the vehicle. The team safety officer and an adult educator must be present for the test, inspect the recorded test footage, and confirm a successful test. If the test is not successful, the team will conduct the test again with more or less black powder, depending on the failure of the test. The adjustment will be at the recommendation of the safety officer and adult educator.

6.1.1.2. Battery Test

Per requirement 2.7, the avionics and recovery system must be capable of remaining in a launch ready configuration for at least two hours and testing is needed to verify this functionality. The testing objective will be ensuring functionality of the avionics and recovery electronics. The testing variable will be a chosen battery for the avionics and recovery system. This test will be considered a success when the avionics and recovery system has been powered on and flight ready for at least three hours.

To conduct this test, the avionics and recovery system will be placed on the team's workbench, powered on. The system will be left alone, but not unattended, for three hours. If the system is still on, it will remain plugged in and checked every 30 minutes until it is no longer powered on. The test is considered a success and the team has an accurate idea of how long a chosen battery can power the system. If the system was powered off at the three hour check, a new battery will be chosen and the test repeated.

6.1.1.3. Avionics Bay Shock Testing

The flight hardware as-assembled must be capable of withstanding the g-forces caused by parachute deployment. To validate the robustness of construction, a simple drop test will be conducted as seen in Figure 6.1. The Avionics bay will start at the top of an 8020 rail, and will be dropped with a linear guide to control the trajectory. The G-force experienced on impact with the base will be recorded using the avionics computers, and the height of the drop will be increased until the controllers are fully saturated. This ensures the hardware as-assembled can robustly withstand the shock loading of parachute deployment.

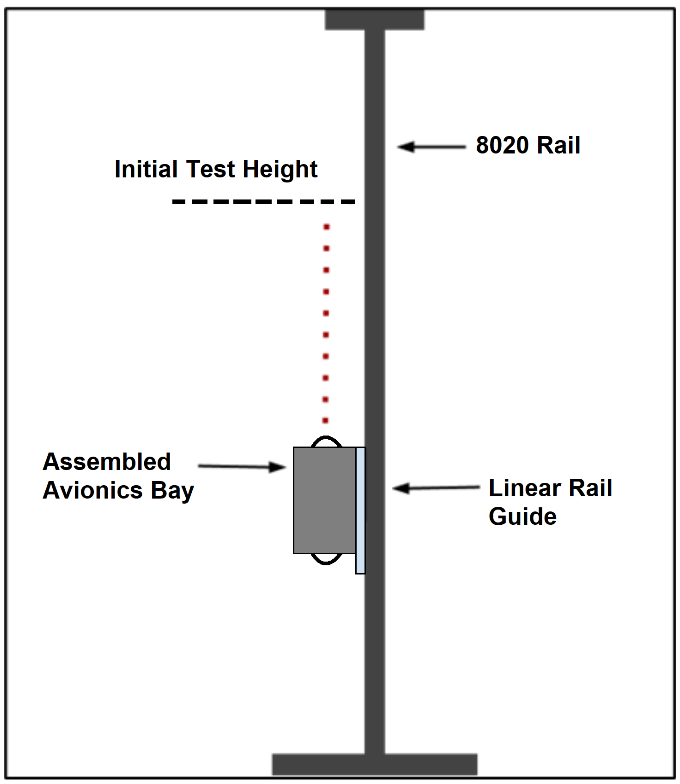


Figure 6.1 Avionics Bay Shock Load Test

6.1.2 Payload

6.1.2.1. Test of Payload RF Interference

After construction of the Payload Sled, detailed in Section 4.10, a test is needed to ensure the XBee RF module, EggTimer GPS transmitter, and FUNCube Pro+ SDR do not interfere with each other at anypoint during payload operation. The test objective is that no component decreases functionality as a result of close proximity (within 3 in of each other). A test will be considered successful only after it is determined no RF transmitter impacts the functionality of the others. This will be measured in the change in the signal-to-noise range and the change in the noise floor. If either changes by less than 10%, the test will be considered a success.

6.1.2.2. Battery Test

With the calculations given in 4.7, we expect our payload to easily last at least 3 hours. The calculations were conservative, but in practice, the duration could vary greatly. Thus, we need to create a test to ensure the battery will power the payload electronics for at minimum 3 hours. Success will be achieved when it lasts for 2.5 hours in sleep mode and .5 hours in transmitting mode. We will write simple test code to achieve these time intervals. The battery we will use is the 2S 2200mAh (Section 4.7). We will charge the battery fully before the test. During the test, we will also record the voltage and amperage that each component is using to validate the nominal values provided by the manufacturer.

This test is necessary to achieve requirement 2.7 with room for error. We will not know how long our vehicle is on the launch pad waiting to be launched, so we must ensure it will last an expected amount of time. If the battery does not last for the intended amount of time, it would be necessary to use a larger battery for the final flight. If this occurs, we may increase the capacity to 2500 mAh or 3000 mAh. It would also be necessary to identify which component was taking up more power than anticipated and diagnose the problem.

6.1.2.3. Locating test for RF 1

We will conduct a test to verify if the system for locating the payload with RF is viable while on the ground. We will place the payload in a field with a visible flag marking its location, and perform the angle and distance measurements by measuring the received signal strength at the payload from the rotating Yagi at another location in the field. We will try a variety of distances to test the limits of the system. The measurement will be performed three times to compare variance in results. The actual distance and angle will be determined with GPS.

6.1.2.4. Locating test for RF 2

The second test that will be done is to determine the maximum interval between transmitted angle values of the Yagi-Uda rotating antenna for precise angle calculations on the payload 2500 feet away, as well as the number of iterations over which the final value is averaged for precise readings.

The payload will be placed 2500 feet away from the ground station at an angle of 45° from the central axis (i.e, x=0 in the cartesian plane in the defined grid). The Yagi-Uda antenna will be rotated from -90° to 90°, transmitting the value of the angle every 0.5°. This process will be repeated 10 times. Necessary calculations as described in Section 4.6 will be carried out on the payload computer averaging separately from 1 to 10 times the signal is sent. This process will be repeated with angle intervals of 1°, 1.5°, and 2°. Actual coordinates will be found using GPS placed at the payload.

The trial with angle and number of iterations which most closely matches the GPS data will be implemented in the final RF system design.

Once the above steps are completed, the accuracy of the system will be tested when the payload is located 0°, 75°, and 90° from the central axis. The payload will be placed at one of these angles 2500 ft away from the ground station, and the transmission cycle will be completed once at each location and compared with GPS data. If there is a discrepancy in the data, the first process of optimum iteration and angle interval testing will be done at that angle and the data will be analyzed to choose the most suitable final interval and iteration count.

6.1.2.5. Locating test for RF 3

The next test for RF will be in determining the angle and distance while the payload is moving as is planned for the final flight. The goal for this test is to reliably communicate with a moving payload. Success criteria is to have an 80% communication rate at a distance of 2500ft.

To conduct this test, we will attach the necessary payload electronics to a drone. The altimeter, Flight Computer, GPS, and rocket side of the payload system will be present. We will move the drone around 500ft away and 700 ft up (height of final launch vehicle at main deployment) and point the Yagi-Uda antenna towards it. We will send a signal with the HackRF, and the payload will be programmed to send back to the Yagi-Uda the power at which it received the signal. We then continue this process as the drone moves down at around 15 ft/s (the estimated speed of descent for the final launch vehicle). The goal is to get at least 10 transmissions and receipts before the drone lands.

The percentage of receipts from the payload should be around 80%. If it is not, we will assess where the problems lie. They could be in the orientation of the Yagi, the transmission power, or something as of yet unaccounted for. We could increase the power transmission from the Ground Station to get better signal receipt on the payload, or we might adjust the positioning of the Yagi.

6.1.2.6. Locating test for RF 4

Once the drone-deployed system has been tested, the RF payload system will be fitted onto the payload of the full-scale rocket as on final launch day in order to test whether the system works cohesively over the rocket flight. The payload will be deployed from the rocket after firing as on the final day. Success will be if the final coordinates match with those of the GPS system on the full-scale test payload.

6.1.2.7. Locating test for IMU, drone deployed

Before a full scale test of the IMU substem can be completed, the system needs to go through an initial round of validation and software debugging. As the team has limited full-scale launches due to finances and availability of the launch field, a drone test will be adequate for initial testing. The initial validation test will consist of a drone carrying an aft section of a level 1 high powered rocket to an altitude of 400 ft and releasing it under parachute. The 400 ft altitude is a restriction due to the school's close proximity to an airport. The payload will record prior to drone lift off, during ascent, and during descent under parachute. This will roughly simulate the flight profile of the full scale rocket. The IMU will calculate its position relative to the drone take off location. This will be compared to the actual position measured using GPS. This test will be successful when the error between the measured and calculated position is less than 125 feet, half of an image grid square.

The results of this test will drive software and hardware changes as necessary: examples of expected design changes include mounting techniques, shock absorption, and algorithm adjustments. It is expected that this first round of system testing will lead to bug fixes and performance enhancements.

6.1.2.8. Locating test for IMU, rocket deployed

Similar to the locating test for the RF system, a full-scale test of the accuracy and functionality of the IMU system is necessary to ensure its viability for the final payload that will launch on Launch Day. The objective for this test is for the IMU system to locate the rocket with an error of less than 125ft (half the length of a grid square). This will also be a test for the grid locating system to ensure it reports a grid accurate to the IMU data. The meeting of this objective will be considered success criteria.

The team will launch a rocket with the payload and IMU system on board. It is necessary that the rocket chosen experiences similar accelerations to the final launch vehicle. The purpose of the test is to understand that the acceleration the rocket undergoes can be accurately measured by the accelerometers chosen and programmed; thus the acceleration should be similar to the final launch vehicle. We will use GPS to corroborate the IMU's position and measure accuracy.

If the accuracy is sufficient, no changes need to be made. If not, the team will determine where the error accumulated. We will analyze the acceleration data after launch and ensure there were no obvious failures of the hardware such as saturation or noise. If there is saturation, we may consider another high range accelerometer in lieu of the H3LIS311. If there is too much noise on either hardware device, that would also warrant a change in hardware. If the accelerometer data is reasonable but the grid box reported an error of more than 125ft, the problem is likely in the coding. More tests may be necessary after the main test to ensure code integrates and reports properly.

6.1.2.9. Test of Shock-mount

The payload must withstand all forces during launch, so we are planning on shock-mounting the entire rocket payload system. The design for the shock-mounted system can be found in 4.10. The objective of this test is to significantly decrease the amount of acceleration the payload will experience. The payload will experience about 112 m/s^2 of acceleration during the launch and 250 m/s^s after main parachute deployment. The test will be considered a success if the acceleration is decreased by 50% .

In order to test the amount of decreased G-force, we will create a simple drop test as seen in Figure 6.2. The "Dummy" Payload will start at the top of the 8020 rails, and it will be dropped and hit the rail at the bottom. The "Dummy" Payload will contain just an accelerometer for the first test as a control, and the second test will contain the shock mount as well as an accelerometer.

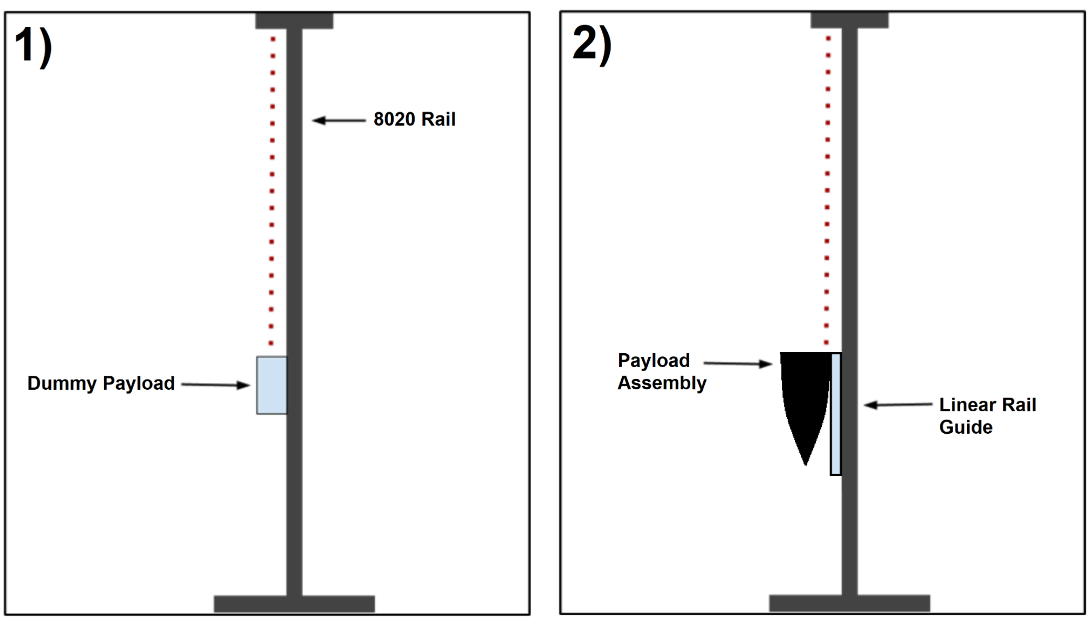


Figure 6.2 Shock Mount Test Setup and Procedure

After both trials are completed (and repeated to ensure consistency), the team will compare the reduction in acceleration to the success criteria. If it does not meet the success criteria, we will have to re-evaluate our design. The problem may be in the chosen spring's stiffness, the length the spring is allowed to compress, or how the weight is distributed. The changes that will be made would remedy any of the problems mentioned.

6.1.3 Altitude Assurance

6.1.3.1. Altitude Assurance Battery Test

Per requirement 2.7, the altitude assurance system must be capable of remaining in a launch ready configuration for at least two hours and testing is needed to verify this functionality. The testing objective will be the altitude assurance electronics, excluding the pneumatic systems. The testing variable will be a chosen battery for the altitude assurance system. This test will be considered a success when the altitude control system has been powered on and flight ready for at least three hours.

To conduct this test, the altitude assurance system will be placed on the team's workbench, powered on, and without pneumatics. The system will be left alone, but not unattended, for three hours. If the system is still on, it will remain plugged in and checked every 30 minutes until it is no longer powered on. The test is considered a success and the team has an accurate idea of how long a chosen battery can power the system. If the system was powered off at the three hour check, a new battery will be chosen and the test repeated.

6.1.3.2. Altitude Assurance Pneumatic Test

Per requirement 2.7, the altitude assurance system must be capable of remaining in a launch ready configuration for at least two hours and testing is needed to verify this functionality. The testing objective will be the pneumatic pressure vessel, piston, solenoid, and supporting hardware for the altitude assurance system. The testing variable will be the assembly of the pneumatics system itself. The test will be considered a success when the pneumatics system has been pressurized and, after three hours, still maintains 95% of its starting pressure.

To conduct this test, the altitude assurance system will be assembled and placed into the launch vehicle in their launch ready configuration. The pneumatics system will be pressurized following the procedure in appendix 7.2. The system will be left in this configuration, but under the supervision of the safety officer, for three hours. After three hours have elapsed, the tank pressure will be inspected. If the tank pressure is less than 95% of the test starting pressure, the altitude assurance pneumatics will be inspected for leaks and reassembled. The test will be repeated until the system has met the success criteria.

6.1.3.3. Altitude Assurance Strength Test

The altitude assurance system is capable of altering the launch vehicle's trajectory. If an air brake were to fail, the produced drag from the altitude assurance system would be asymmetrical and the vehicle's flight would no longer be predictable. Testing of the strength of the air brakes is needed to verify performance calculations. The testing objective will be a single air brake flap. The air brake flap will be placed with the side which experiences aerodynamic forces facing upward on a workbench. Weights will be placed on top of the air brake, starting at 2 pounds. The weight will be increased until the total weight reaches 17 pounds. At this point the test will be considered a success. If the air brake shows signs of structural failure before 17 pounds is rached, the air brake will be inspected for the cause of failure and redesigned.

6.2 Requirements Compliance Plan

6.2.1 Competition Requirements Verification

Section	Requirement
1.1	Students on the team will do 100% of the project, including design, written reports and presentations. Teams will submit new work. Excessive use of past work will merit penalties.

Verification Plan:

Because the team has not previously participated in NASA SL, no past work exists and no verification is needed around re-use. In order to ensure students complete 100% of work, team members may only consult with outside help, and must individually complete all design work, written reports, and presentations. Advisors will be given access to team documents for supervising, but will not be not given editing privileges. This requirement will be verified by the team president and vice president by reviewing all documentation prior to submission and inspecting all physical construction after each work day.

Section	Requirement
1.2	The team will provide and maintain a project plan to include, but not limited to the following items: project milestones, budget and community support, checklists, personnel.

Verification Plan:

The team president is in charge of the project plan. This plan will be inspected for completeness by the vice president and team advisor. This project plan will be recorded and maintained in Click Up project management software, which will be available for all team members at all times. The president will maintain deadlines, determine milestones, and log actionable items in the software.

The team treasurer will maintain the budget. This budget will be inspected for completeness by the president after each purchase request. The team treasurer will maintain an updated budget spreadsheet located in the team's Google Drive account, which is viewable to all team members at all times.

Section	Requirement
1.3	Foreign National (FN) team members must be identified by the Preliminary Design Review (PDR) and may or may not have access to certain activities during Launch Week due to security restrictions. In addition, FN's may be separated from their team during certain activities on site at Marshall Space Flight Center.

Verification plan:
The team president will identify Foreign Nationals on the team and compile a list for inclusion with competition documents. This list will be inspected for completeness by the vice president immediately prior to submission.

Section	Requirement
1.4	The team must identify all team members who plan to attend Launch Week activities by the Critical Design Review (CDR). Team members will include: 1.4.1. Students actively engaged in the project throughout the entire year. 1.4.2. No more than two adult educators.

Verification plan:
The team president will maintain a list of members interested in attending Launch Week. This requirement will be verified by demonstration of completion to all active team members. A list of active team members who will attend launch week will be assembled by polling all team members at least one week prior to submission. An up-to-date list will be submitted to SL Management along with the CDR submission package, and all active team members will be included on this message via blind carbon copy.

Section	Requirement
1.5	The team will engage a minimum of 250 participants in direct educational, hands-on science, technology, engineering, and mathematics (STEM) activities. These activities can be conducted in-person or virtually. To satisfy this requirement, all events must occur between project acceptance and the FRR due date.

Verification plan:
The team public affairs officer will coordinate STEM engagement events. These will be verified by demonstration to the team president. The public affairs officer will record the number of participants for each event in a spreadsheet available to the team. The team president will assess the progress of STEM engagement via checkpoints set at the end of every month. If engagement is not meeting the checkpoints, the team vice-president will assist in the planning and coordination of engagement events.

Section	Requirement
1.6	The team will establish and maintain a social media presence to inform the public about team activities.

Verification plan:
The team will have a social media presence established and run by the public affairs officer. The social media presence will be demonstrated to the team and community by regular posting and activity. Additionally, the public affairs officer may give any team member access to any social media accounts in order to facilitate a more engaging social media presence.

Section	Requirement
1.7	Teams will email all deliverables to the NASA project management team by the deadline specified in the handbook for each milestone. In the event that a deliverable is too large to attach to an email, inclusion of a link to download the file will be sufficient. Late submissions of milestone documents will be accepted up to 72 hours after the submission deadline. Late submissions will

	incur an overall penalty. No milestone documents will be accepted beyond the 72-hour window. Teams that fail to submit milestone documents will be eliminated from the project.

Verification plan:
The team president will monitor and track all deliverable deadlines in Click Up per requirement 1.2, maintaining a project plan. The vice-president will be responsible for periodically inspecting Click Up and ensuring the team's progress towards completion of competition deliverables. Additionally, both the president and vice-president receive all email notifications from the NASA management team.

Section	Requirement
1.8	All deliverables must be in PDF format.

Verification plan:
The team president will draft the email that contains the deliverables, and the Vice President will inspect the email before it is sent and check for the PDF documents.

Section	Requirement
1.9	In every report, teams will provide a table of contents including major sections and their respective sub-sections.

Verification plan:
The Vice President creates the initial document with preliminary sections, including the table of contents. They will update the table as writing continues, and before the document is submitted, the President will inspect the table of contents prior to document submission.

Section	Requirement

1.10	In every report, the team will include the page number at the bottom of the page.

Verification plan:
The Vice President is in charge of creating the initial document with preliminary sections, including page numbers at the bottom of the page. The page numbers will automatically update as the writing continues, and the President will verify this requirement before submitting the document.

Section	Requirement
1.11	The team will provide any computer equipment necessary to perform a video teleconference with the review panel. This includes, but is not limited to, a computer system, video camera, speaker telephone, and a sufficient Internet connection. Cellular phones should be used for speakerphone capability only as a last resort.

Verification plan:
The President is in charge of acquiring the equipment necessary to perform a video teleconference. This will include an external camera and stand to ensure high quality video from our university's Information Technology Department. The presentation team will perform a test presentation prior to the selected presentation date to ensure all equipment is fully functional and provide time to resolve any technical difficulties.

Section	Requirement
1.12	Teams will track and report the number of hours spent working on each milestone.

Verification plan:
Every individual member will keep track of the amount of time they work on each document. The president will compile each person's individual time after document completion and prior to document submission.

Section	Requirement

2.1	The vehicle will deliver the payload to an apogee altitude between 4,000 and 6,000 feet above ground level (AGL). Teams flying below 4,000 feet or above 6,000 feet on their competition launch will receive zero altitude points towards their overall project score and will not be eligible for the Altitude Award.

Verification plan:
The Vice President is responsible for overseeing the development and testing of the Altitude Assurance system which will assure that the vehicle reaches a target altitude of 5000 ft as prescribed in the Project Silverstein PDR report. The Vice President will lead testing of the Altitude Assurance system during the subscale reflight, vehicle demonstration flight, and payload demonstration flight. Simulation of the launch vehicle performance will be inspected by the Vice President at least one week before each test flight.

Section	Requirement
2.2	Teams shall identify their target altitude goal at the PDR milestone.

Verification plan:
In our PDR milestone, we identified a target altitude of 5000ft. The PDR was inspected by the Vice President to ensure this was included in the document.

Section	Requirement
2.3	The vehicle will carry, at a minimum, two commercially available barometric altimeters that are specifically designed for initiation of rocketry recovery events (see Requirement 3.4).

Verification plan:

The Vice President is responsible for overseeing the final design of the launch vehicle. The Vice President will inspect the final design of the launch vehicle at least one week before the submission of the CDR deadline. If two commercially available barometric altimeters are not present in the Avionics Bay, a redesign of the Avionics Bay will be issued.

Section	Requirement
2.4	The launch vehicle will be designed to be recoverable and reusable. Reusable is defined as being able to launch again on the same day without repairs or modifications.

Verification plan:
This requirement will be accomplished via demonstration and analysis. In the Recovery section, we determined what parachutes, chord, and black powder charges will be necessary to achieve a low landing kinetic energy. We will demonstrate these calculations are complete by launching the full-scale rocket and inspecting any damage. Any damage that is sustained will be analyzed, and the factor of safety will be increased before the next flight. The Vice President will lead this effort.

Section	Requirement
2.5	The launch vehicle will have a maximum of four (4) independent sections. An independent section is defined as a section that is either tethered to the main vehicle or is recovered separately from the main vehicle using its own parachute.

Verification plan:
The team Vice President will be responsible for ensuring compliance of the launch vehicle architecture. Compliance will be verified by demonstrating the design is complete. A vehicle design consisting of 3 independent sections has been demonstrated to all members of the team in a joint meeting prior to the completion of the CDR.

Section	Requirement
2.5.1	Coupler/airframe shoulders which are located at in-flight separation points will be at least 1 body diameter in length.

Verification plan:
The team Vice President will be responsible for ensuring compliance of the launch vehicle architecture. Compliance will be verified by demonstrating the design is complete. A vehicle design in which each coupler located at a point of separation contained a shoulder of at least six inches has been demonstrated to the team prior to the completion of the CDR.

Section	Requirement
2.5.2	Nosecone shoulders which are located at in-flight separation points will be at least ½ body diameter in length.

Verification plan:
The team vice president will be responsible for ensuring compliance of the launch vehicle architecture. Compliance will be verified by demonstrating the design is complete. A vehicle design in which the nose cone contained a shoulder of at least three inches has been demonstrated to the team prior to the completion of the CDR.

Section	Requirement
2.6	The launch vehicle will be capable of being prepared for flight at the launch site within 2 hours of the time the Federal Aviation Administration flight waiver opens.

Verification plan:

With the checklists created by the Safety Officer, Vice President, and other members, we have an order in which the rocket should be compiled before and on launch day. These are arranged such that as much work as can be done before is done with verification by several members of the team. The 2 hour minimum will be achieved through testing of our preparation time before launch day.

Section	Requirement
2.7	The launch vehicle and payload will be capable of remaining in launch-ready configuration on the pad for a minimum of 2 hours without losing the functionality of any critical on-board components, although the capability to withstand longer delays is highly encouraged.

Verification plan:
The team vice president will be responsible for ensuring compliance of critical on-board components. Compliance will be verified by hand calculations of critical components' power usage and estimated power on time for a selected battery, shown in section 4.7. And, by a test which demonstrates the critical components' ability to remain in launch-ready configuration for at least three hours. This test will take place prior to the first full scale vehicle demonstration.

Section	Requirement
2.8	The launch vehicle will be capable of being launched by a standard 12-volt direct current firing system. The firing system will be provided by the NASA-designated launch services provider.

Verification plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final vehicle design. Because the final motor selected for the mission ships with an igniter capable of firing off a standard 12-volt DC firing system and no obstructions exist for the igniter in the design, a redesign was not issued.

Section	Requirement
2.9	The launch vehicle will require no external circuitry or special ground support equipment to initiate launch (other than what is provided by the launch services provider).

Verification plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final vehicle design. Because the final vehicle design does not employ the use of external circuitry or special ground support equipment, a redesign was not issued.

Section	Requirement
2.10	The launch vehicle will use a commercially available solid motor propulsion system using ammonium perchlorate composite propellant (APCP) which is approved and certified by the National Association of Rocketry (NAR), Tripoli Rocketry Association (TRA), and/or the Canadian Association of Rocketry (CAR).

Verification plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final motor choice. Because the leading motor choice described in the report does use a commercially available APCP propulsion system approved by NAR and TRA, a reselection was not issued.

Section	Requirement
2.10.1	Final motor choices will be declared by the Critical Design Review (CDR) milestone.

Verification plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final vehicle design. Because the leading motor choice described in the report does not exceed 5120 N-s in impulse, a reselection was not issued.

Section	Requirement
2.10.2	Any motor change after CDR must be approved by the NASA Range Safety Officer (RSO). Changes for the sole purpose of altitude adjustment will not be approved. A penalty against the team's overall score will be incurred when a motor change is made after the CDR milestone, regardless of the reason.

Verification plan:
The Treasurer will purchase the decided motor and extras as soon as possible to mitigate worries of delayed shipping. Should the motor still not arrive on time

Section	Requirement
2.11	The launch vehicle will be limited to a single stage.

Verification plan:
The design of the rocket as decided by the Vice President and the rest of the team does not include a second stage. By simple inspection, this requirement is fulfilled.

Section	Requirement
2.12	The total impulse provided by a College or University launch vehicle will not exceed 5,120 Newton-seconds (L-class)

Verification plan:
The Vice President is responsible for the final design of the launch vehicle. At least one week before the submission of the PDR report, the Vice President has inspected the vehicle design. Because the leading motor choices described in the report did not exceed

5120 N-s in impulse, a reselection was not issued. Because of this, the selection for the final motor choice did not exceed 5120N-s in impulse.

Section	Requirement
2.13	Pressure vessels on the vehicle will be approved by the RSO

Verification Plan:
The team Safety Officer will be responsible for acquiring the approval for any on board pressure vessels by the RSO. The safety officer will communicate RSO approval to the president and vice president. The launch vehicle is prohibited from launching until approval is received.

Section	Requirement
2.13.1	The minimum factor of safety [for a pressure vessel on the vehicle] (Burst or Ultimate pressure versus Max Expected Operating Pressure) will be 4:1 with supporting design documentation included in all milestone reviews.

Verification Plan:
The team Safety Officer will be responsible for ensuring that a selected pressure vessel and system design maintain at least a 4:1 factor of safety for burst and max operating pressure.

Section	Requirement
2.13.2	Each pressure vessel will include a pressure relief valve that sees the full pressure of the tank and is capable of withstanding the maximum pressure and flow rate of the tank

Verification Plan:
The team Vice President will be responsible for the final design of any vehicle system utilizing a pressure vessel. The design will be inspected for the inclusion of a pressure relief valve that sees full tank pressure. Additionally, the pressure relief valve will be

inspected to ensure its operational range is suitable for use in the chosen pressure vessel design. The vice president will issue a redesign of the pressure vessel system if the relief valve is omitted or does not meet the pressure requirements of the system.

Section	Requirement
2.13.3	The full pedigree of the tank will be described, including the application for which the tank was designed and the history of the tank. This will include the number of pressure cycles put on the tank, the dates of pressurization/depressurization, and the name of the person or entity administering each pressure event

Verification Plan:
The team Safety Officer will be responsible for maintaining a complete and accurate log of all pressure tank events and uses. This log will include a description of the tank, relevant safety information, and dated entries for each pressurization, depressurization, and the person or persons administering each event. This log will be periodically inspected by the president and included in all milestone reports.

Section	Requirement
2.14	The launch vehicle will have a minimum static stability margin of 2.0 at the point of rail exit. Rail exit is defined at the point where the forward rail button loses contact with the rail.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final vehicle simulations and calculations. Because simulations have shown that the static stability margin of the vehicle is above the minimum static stability margin, a redesign was not issued.

Section	Requirement
2.15	The launch vehicle will have a minimum thrust to weight ratio of 5.0 : 1.0

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final vehicle simulations and calculations. Because neither simulations nor calculations have shown that the thrust to weight ratio of the vehicle is below 5.0:1.0, a redesign was not issued.

Section	Requirement
2.16	Any structural protuberance on the rocket will be located aft of the burnout center of gravity. Camera housings will be exempted, provided the team can show that the housing(s) causes minimal aerodynamic effect on the rocket's stability

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final vehicle design. Because all structural protuberances on the final vehicle design are located aft of the burnout center of gravity, a redesign was not issued.

Section	Requirement
2.17	The launch vehicle will accelerate to a minimum velocity of 52 fps at rail exit.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final vehicle simulations and calculations. Because neither simulations nor calculations have shown that the vehicle accelerates below 52 fps at rail exit, a redesign was not issued.

Section	Requirement
2.18	All Lithium Polymer batteries will be sufficiently protected from impact with the ground and will be brightly colored, clearly marked as a fire hazard, and easily distinguishable from other payload hardware.

Verification Plan:
The Safety Officer is responsible for ensuring that all batteries are marked and colored, but the members of the team working on the design of the bays will ensure they are protected to avoid combustion if recovery is to fail. The Vice President will inspect the batteries before any launch ensuring that they are clear of deformation or puncters and they are clearly marked and labeled. The vice president will also inspect the final design of any launch vehicle system utilizing lithium polymer batteries for addicate protection from impact prior to CDR submission. The vice president will issue a redesign if the current design does not adequately protect the batteries.

Section	Requirement
2.19.1	The launch vehicle will not utilize forward firing motors.

Verification Plan:
As the Vice President is in charge of the final design, they will inspect the design to ensure that the launch vehicle will not utilize forward firing motors. This inspection will occur at least 2 weeks before the submission of the CDR.

Section	Requirement
2.19.2	The launch vehicle will not utilize motors that expel titanium sponges (Sparky, Skidmark, MetalStorm, etc.)

Verification Plan:
The Vice President is responsible for the review of the final rocket design, and will verify that no motors that expel titanium sponges are utilized or referenced within. The design review will occur at least 2 weeks before the submission of the CDR.

Section	Requirement
2.19.3	The launch vehicle will not utilize hybrid motors.

Verification Plan:
The Vice President is responsible for the review of the final rocket design, and will verify that no hybrid motors are utilized or referenced within. The design review will occur at least 2 weeks before the submission of the CDR.

Section	Requirement
2.19.4	The launch vehicle will not utilize a cluster of motors.

Verification Plan:
The Vice President is responsible for the review of the final rocket design, and will verify that a cluster of motors is not used or referenced within. The design review will occur at least 2 weeks before the submission of the CDR.

Section	Requirement
2.19.5	The launch vehicle will not utilize friction fitting for motors

Verification Plan:
The Vice President is responsible for the final design and review of the launch vehicle, and will verify that the motors for the launch vehicle are secured without the use of friction fitting. The design review will occur at least 2 weeks before the submission of the CDR. This review will include the verification of how the motors are secured and that none of the design utilizes friction fittings for the motors.

Section	Requirement
2.19.6	The launch vehicle will not exceed Mach 1 at any point during flight.

Verification Plan:
The Vice President is responsible for reviewing the launch vehicle, and will verify that the launch vehicle cannot exceed Mach 1 in rocket simulation before the launch vehicle is utilized.

Section	Requirement
2.19.7	Vehicle ballast will not exceed 10% of the total unballasted weight of the rocket as it would sit on the pad (i.e. a rocket with an unballasted weight of 40 lbs. on the pad may contain a maximum of 4 lbs. of ballast).

Verification Plan:
The Vice President is responsible for the final design and review of the launch vehicle, and will verify that the vehicle ballast will not exceed 10% of the total unballasted weight of the rocket as it would sit on the pad. This verification will occur at least 2 weeks before the submission of the CDR. This review will include recalculation of the total unballasted rocket weight and vehicle ballast weight.

Section	Requirement
2.19.8	Transmissions from onboard transmitters, which are active at any point prior to landing, will not exceed 250 mW of power (per transmitter).

Verification Plan:
The vice president is responsible for reviewing the final design of the launch vehicle and payload. The vice president will inspect the design presented by team members prior to the submission of the CDR. A redesign will be issued for any design which includes a transmitter exceeding 250mW of power prior to landing.

Section	Requirement
2.19.9	Transmitters will not create excessive interference. Teams will utilize unique frequencies, handshake/passcode systems, or other means to mitigate interference caused to or received from other teams.

Verification Plan:
The payload team will inspect the frequencies used by other teams and inquire about interference. As a preemptive measure, the team has a range of frequencies it can transmit at, and all telemetry will be encoded. If a team relies heavily on one frequency, all of our transmitters have a range of at least 15mHz that they can transmit, so we can change our transmission frequency to comply with this requirement.

Section	Requirement
2.19.10	Excessive and/or dense metal will not be utilized in the construction of the vehicle. Use of light[1]weight metal will be permitted but limited to the amount necessary to ensure structural integrity of the airframe under the expected operating stresses.

Verification Plan:
The vice president will be responsible for the final design of the launch vehicle. They will inspect the design for the use of any dense or lightweight metals. Designs utilizing dense metals will not be allowed. Team members must demonstrate to the vice president through analysis, such as FEA, the necessity of any lightweight metals included on the launch vehicle. The vice president will issue a redesign of the launch vehicle if the analysis does not justify the use of a chosen lightweight metal.

Section	Requirement
3.1	The full scale launch vehicle will stage the deployment of its recovery devices, where a drogue parachute is deployed at apogee, and a main parachute is deployed at a lower altitude. Tumble or streamer recovery from apogee to main parachute deployment is also permissible, provided that kinetic energy during drogue stage descent is reasonable, as deemed by the RSO.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final vehicle design. Because the final vehicle design incorporates dual deployment of a drogue chute at apogee and a main chute at 600ft., a redesign was not issued.

Section	Requirement
3.1.1	The main parachute shall be deployed no lower than 500 feet

Verification Plan:
The team safety officer will be responsible for the configuration of the recovery altimeters. Altimeter configuration will be inspected by the team president prior to launch day and again at the team's work table on launch day. The launch vehicle will not be allowed to fly until both altimeters are configured with main parachute deployment greater than 500 feet.

Section	Requirement
3.1.2	The apogee event may contain a delay of no more than 2 seconds

Verification Plan:
The team safety officer will be responsible for the configuration of the recovery altimeters. Altimeter configuration will be inspected by the team president prior to launch day and again at the team's work table on launch day. The launch vehicle will not be allowed to fly until an event delay of 2 seconds or less is configured.

Section	Requirement
3.1.3	Motor ejection is not a permissible form of primary or secondary deployment.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final vehicle design. Because the design incorporates electronic deployment of both the drogue and main chutes and a motor ejection design is not used, a redesign was not issued.

Section	Requirement
3.2	Each team will perform a successful ground ejection test for all electronically initiated recovery events prior to the initial flights of the subscale and full scale vehicles.

Verification Plan:
The team safety officer is responsible for the coordination and planning of all ground ejection test. The team will not be allowed to travel to the launch site until a successful ground test is demonstrated to the team vice president and advisor.

Section	Requirement
3.3	Each independent section of the launch vehicle will have a maximum kinetic energy of 75 ft-lbf at landing.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final vehicle simulations and calculations. Because neither simulations nor calculations have shown that the maximum kinetic energy of any independent section does not exceed 75 ft-lbf at landing, a redesign was not issued.

Section	Requirement
3.4	The recovery system will contain redundant, commercially available altimeters. The term “altimeters” includes both simple altimeters and more sophisticated flight computers.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least one week before the submission of the CDR report, the Vice President will have inspected the vehicle design and made sure that the altimeters used on the launch vehicle are redundant and commercially available.

Section	Requirement
3.5	Each altimeter will have a dedicated power supply, and all recovery electronics will be powered by commercially available batteries.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least one week before the submission of the CDR report, the Vice President will have inspected the vehicle design and ensured that each of the redundant altimiters have a dedicated power supply and that the recovery electronics are powered by commercially available batteries.

Section	Requirement
3.6	Each altimeter will be armed by a dedicated mechanical arming switch that is accessible from the exterior of the rocket airframe when the rocket is in the launch configuration on the launch pad.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least one week before the submission of the CDR report, the Vice President will have inspected the design and verified that each altimeter is armed by a dedicated mechanical arming switch accessible from the exterior of the rocket airframe when the rocket is in the launch configuration on the launch pad.

Section	Requirement
3.7	Each arming switch will be capable of being locked in the ON position for launch (i.e. cannot be disarmed due to flight forces).

Verification Plan:
The Vice President is responsible for the review of the final design of the launch vehicle. Thus, during the review of the final design, which occurs at least one week before the submission of the CDR report, the Vice President will verify that the design of the arming switch on the launch vehicle will allow for the arming switch to be locked in the ON position, and unable of being disarmed due to flight forces.

Section	Requirement
3.8	The recovery system electrical circuits will be completely independent of any payload electrical circuits.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least one week before the submission of the CDR report, the Vice President will have inspected the design and verified that the recovery system electronics are independent of all payload electronic systems.

Section	Requirement
3.9	Removable shear pins will be used for both the main parachute compartment and the drogue parachute compartment.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final vehicle design. Because the design incorporates removable shear pins for deployment of both the drogue and main chutes, a redesign was not issued.

Section	Requirement
3.10	The recovery area will be limited to a 2,500 ft. radius from the launch pads.

Verification Plan:
The Vice President will finalize the design of the rocket which includes the maximum drift based on several wind conditions (Section 3.7.1). On top of analyzing our predicted drift, during our practice flights, the Vice President will determine our experimental drift to ensure it meets this requirement.

Section	Requirement
3.11	Descent time of the launch vehicle will be limited to 90 seconds (apogee to touch down).

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least one week before the submission of the CDR report, the Vice President will have inspected the design and verified through CFD that the drag produced by the drogue and main parachutes is low enough to limit descent time to 90 seconds while also meeting the Section 3.2 requirement.

Section	Requirement
3.12	An electronic GPS tracking device will be installed in the launch vehicle and will transmit the position of the tethered vehicle or any independent section to a ground receiver.

Verification Plan:
The payload team has a Eggtimer GPS Transmitter that will be constantly transmitting the GPS location of the rocket throughout the flight. The Vice President will inspect its functionality before the FRR to ensure this requirement is met.

Section	Requirement
3.12.1	Any rocket section or payload component, which lands untethered to the launch vehicle, will contain an active electronic GPS tracking device.

Verification Plan:
The vice president is responsible for the final design of the launch vehicle and payload. The final design will be inspected by the vice president and verified by a secondary inspection of the safety officer for the inclusion of a GPS tracking device on any rocket section or untethered payload.

Section	Requirement
3.13	The recovery system electronics will not be adversely affected by any other on-board electronic devices during flight (from launch until landing).

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least one week before the submission of the CDR report, the Vice President will have inspected the design and ensured that the recovery system is adequately shielded. Ground testing will also be conducted in order to verify that no adverse effects occur to the recovery system as a result of other electronic systems.

Section	Requirement
3.13.1	The recovery system altimeters will be physically located in a separate compartment within the vehicle from any other radio frequency transmitting device and/or magnetic wave producing device.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least one week before the submission of the CDR report, the Vice President will have inspected the design and ensured that the recovery system altimeters are in another compartment separate from other RF transmitters or magnetic wave producing devices.

Section	Requirement
3.13.2	The recovery system electronics will be shielded from all onboard transmitting devices to avoid inadvertent excitation of the recovery system electronics.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least one week before the submission of the CDR report, the Vice President will have inspected the design and ensured that the recovery system is adequately shielded. Testing will be conducted in order to verify that the shielding is adequate to avoid excitation of the recovery system by other transmitting devices.

Section	Requirement
3.13.3	The recovery system electronics will be shielded from all onboard devices which may generate magnetic waves (such as generators, solenoid valves, and Tesla coils) to avoid inadvertent excitation of the recovery system.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least one week before the submission of the CDR report, the Vice President will have inspected the design and ensured that the recovery system is adequately shielded. Testing will be conducted in order to verify that the shielding is adequate to avoid excitation of the recovery system by other transmitting devices.

Section	Requirement
3.13.4	The recovery system electronics will be shielded from any other onboard devices which may adversely affect the proper operation of the recovery system electronics.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least one week before the submission of the CDR report, the Vice President will have inspected the design and ensured that the recovery system is adequately shielded. Testing will be conducted in order to verify that the shielding is adequate to avoid interference from other onboard devices.

Section	Requirement
4.1	Teams shall design a payload capable of autonomously locating the launch vehicle upon landing by identifying the launch vehicle's grid position on an aerial image of the launch site without the use of a global positioning system (GPS). The method(s)/design(s) utilized to complete the payload mission will be at the teams' discretion and will be permitted so long as the designs are deemed safe, obey FAA and legal requirements, and adhere to the intent of the challenge.

Verification Plan:
The payload team has created a system that, in design, fulfills this requirement (Section 4.5 and 4.6), but testing and analysis is required after the system is built. The payload team will conduct testing on the RF and IMU Systems using the Flight Computer, and ensure that this requirement is met (see Section 6.1.2)

Section	Requirement
4.2.1	The dimensions of the gridded launch field shall not extend beyond 2,500 feet in any direction; i.e., the dimensions of your gridded launch field shall not exceed 5,000 feet by 5,000 feet

Verification Plan:
The Vice President is responsible for ensuring this requirement is met. The gridded launch field shown in Section 4.9 has dimensions of 2,500 ft on both sides. The Vice President inspected the gridded launch field.

Section	Requirement
4.2.1.1	Your launch vehicle and any jettisoned components must land within the external borders of the launch field.

Verification Plan:
The Vice President is responsible for ensuring drift calculations are performed for the launch vehicle and any jettisoned components. Our current launch calculations meet this requirement

Section	Requirement
4.2.2	A legible gridded image with a scale shall be provided to the NASA management panel for approval at the CDR milestone.

Verification Plan:
The President is responsible for sending the gridded image to the NASA management panel for approval, but the Vice President is responsible for ensuring that the image is

legible. This requirement will be completed through inspection before the image is submitted.

Section	Requirement
4.2.2.1	The dimensions of each grid box shall not exceed 250 feet by 250 feet.

Verification Plan:
The Vice President is responsible for ensuring that the gridded image has box dimensions that do not exceed 250 ft. In its CDR state, the image has box dimensions of 250ft on each side (Section 4.9).

Section	Requirement
4.2.2.2	The entire launch field, not to exceed 5,000 feet by 5,000 feet, shall be gridded

Verification Plan:
The Vice President will ensure that the launch field image is accurately gridded before its submission to the NASA management panel.

Section	Requirement
4.2.2.3	Each grid box shall be square in shape.

Verification Plan:
The Vice President is responsible for ensuring that the gridded image has boxes that are square.

Section	Requirement
4.2.2.4	Each grid box shall be equal in size, it is permissible for grid boxes occurring on the perimeter of your launch field to fall outside the dimensions of the launch field. Do not alter the shape of a grid box to fit the dimension or shape of your launch field.

Verification Plan:
The Vice President is responsible for ensuring that each grid box is equal in size before its submission to the NASA management panel.

Section	Requirement
4.2.2.5	Each grid box shall be numbered

Verification Plan:
The Vice President is responsible for ensuring that each grid box is numbered before its submission to the NASA management panel.

Section	Requirement
4.2.2.6	The identified launch vehicle's grid box, upon landing, will be transmitted to your team's ground station.

Verification Plan:
The payload design accounts for this requirement using the GUI controlled by the Ground Station Computer. The Flight Computer does the work to determine the grid box, and it will send via the Telemetry System the determined box to the Ground Station Computer. The Vice President will inspect the work of the payload design team before the FRR to ensure the requirement is met.

Section	Requirement
4.2.3	GPS shall not be used to aid in any part of the payload mission.

Verification Plan:
The Vice President is responsible for the oversight of the final launch vehicle design and will verify that GPS is not used or referenced in any part of the payload mission's documentation or hardware.

Section	Requirement
4.2.3.1	GPS coordinates of the launch vehicle's landing location shall be known and used solely for the purpose of verification of payload functionality and mission success.

Verification Plan:
The Vice President is responsible for ensuring the completion of this requirement through analysis and inspection. The payload team has created a payload design that transmits the GPS coordinates continuously throughout the flight. The GPS has its own transmitter and is thus completely separate from the rest of the payload. The Flight Computer determines the location of the rocket and does not have access to the GPS data. The Vice President will ensure the payload functions as designed before the FRR by overseeing the full payload test.

Section	Requirement
4.2.3.2	GPS verification data shall be included in your team's PLAR.

Verification Plan:
The President is responsible for the filing of all team documents and will verify, before the submission of the PLAR, that GPS verification data is included within.

Section	Requirement
4.2.4	The gridded image shall be of high quality, as deemed by the NASA management team, that comes from an aerial photograph or satellite image of your launch day launch field.

Verification Plan:
The Vice President is responsible for the quality of the gridded image and will ensure that it is an aerial photograph of satellite image.

Section	Requirement
4.2.4.1	The location of your launch pad shall be depicted on your image and confirmed by either the NASA management panel for those flying in Huntsville or your local club's RSO. (GPS coordinates are allowed for determining your launch pad location).

Verification Plan:
The Vice President is responsible for ensuring the completion of this requirement. Up to this point, the launch pad coordinates that are used for the gridded image depicted in Section 4.9 are from a frequently asked questions post on the NASA.gov website (https://www.nasa.gov/stem/studentlaunch/faqs.html). The Vice President will confirm with the NASA management panel two weeks before the FRR is due via email to ensure our coordinates are correct and have not been updated.

Section	Requirement
4.2.5	No external hardware or software is permitted outside the team's prep area or the launch vehicle itself prior to launch

Verification Plan:
The Safety Officer will be responsible for ensuring that no external hardware or software exists, intentionally or accidentally, outside of the team prep area. The Safety Officer will take physical steps to bring hardware or software back to the prep area should it be identified outside of the prep area.

Section	Requirement
4.3.1	Black Powder and/or similar energetics are only permitted for deployment of in-flight recovery systems. Energetics will not be permitted for any surface operations.

Verification Plan:
The vice president is responsible for the final payload and vehicle mission design. Prior to CDR submission the, the vice president will inspect the payload and vehicle mission design to ensure no energetic devices are used.

Section	Requirement
4.3.2	Teams shall abide by all FAA and NAR rules and regulations.

Verification Plan:
The Safety Officer is responsible for ensuring all team members and all team-related projects abide by all FAA and NAR rules and regulations.

Section	Requirement
4.3.3	Any experiment element that is jettisoned during the recovery phase will receive real-time RSO permission prior to initiating the jettison event, unless exempted from the requirement at the CDR milestone by NASA.

Verification Plan:
The team has determined an experiment element which jettisons from the launch vehicle is not necessary to successfully complete the payload mission. Because of this, there will be no verification needed for RSO permission prior to a jettison event.

Section	Requirement
4.3.4	Unmanned aircraft system (UAS) payloads, if designed to be deployed during descent, will be tethered to the vehicle with a remotely controlled release mechanism until the RSO has given permission to release the UAS.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final vehicle design. Because the design does not incorporate the use of any unmanned aircraft system to be deployed during descent, a redesign was not issued.

Section	Requirement

4.3.5	Teams flying UASs will abide by all applicable FAA regulations, including the FAA's Special Rule for Model Aircraft (Public Law 112-95 Section 336; see https://www.faa.gov/uas/faqs).

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final vehicle design. Because the design does not incorporate the use of any unmanned aircraft system to be deployed during descent, a redesign was not issued.

Section	Requirement
4.3.6	Any UAS weighing more than .55 lbs. will be registered with the FAA and the registration number marked on the vehicle.

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final vehicle design. Because the design does not incorporate the use of any unmanned aircraft system to be deployed during descent, a redesign was not issued.

Section	Requirement
5.1	Each team will use a launch and safety checklist. The final checklists will be included in the FRR report.

Verification Plan:
The President is the primary administrator of the team and is responsible for the filing of all USLI documents. It is the Safety Officer's responsibility to complete the checklists, but the President will verify that a launch and safety checklist is complete, completed, and included in the FRR report.

Section	Requirement
5.2	Each team shall identify a student safety officer who will be responsible for all items in section 5.3.

Verification Plan:
The President is responsible for the overseeing of the team Officers and will verify, at every change of the team's roster, that a student Safety Officer has been identified and elected by the team. If there is no Safety Officer, the President will ensure that there is an election and that a new Safety Officer is selected at the next meeting in which a majority student population is present.

Section	Requirement
5.3.1	The safety officer will monitor team activities with an emphasis on safety during: 5.3.1.1. Design of vehicle and payload 5.3.1.2. Construction of vehicle and payload components 5.3.1.3. Assembly of vehicle and payload 5.3.1.4. Ground testing of vehicle and payload 5.3.1.5. Subscale launch test(s) 5.3.1.6. Full-scale launch test(s) 5.3.1.7. Competition Launch 5.3.1.8. Recovery activities 5.3.1.9. STEM Engagement Activities

Verification Plan:
The team Safety Officer has and will be an important part of every design decision. They have and will green-light every design decision with an emphasis on safety by being present at design meetings and reading over all aspects of technical documents. They have been and will be present at every launch and ground test in order to ensure the completion of checklists and the following of RSO rules. All construction of the launch

vehicles will take place during broadcasted meeting times. If the Safety Officer cannot attend, they will appoint someone present to oversee construction with an emphasis on safety.

Section	Requirement
5.3.2	The safety officer will implement procedures developed by the team for construction, assembly, launch, and recovery activities.

Verification Plan:
The Safety Officer is responsible for fulfilling this requirement. They have made checklists located in the Appendix that list the procedures for payload, recovery electronics, pneumatic, recovery, and rocket motor preparation. The Safety Officer is in charge of implementing these checklists, and the Vice President or the President will verify their completion during any activity they are needed for.

Section	Requirement
5.3.3	The safety officer will manage and maintain current revisions of the team's hazard analyses, failure modes analysis, procedures, and MSDS/chemical inventory data.

Verification Plan:
The Safety Officer will check over and update the teams Safety Section (containing hazard analyses, failure modes analysis) at least 2 weeks before the submission of any document. In addition, the Safety Officer created procedures for preparing our rocket for launch (see Appendix) and will change them as needed. The President will verify the completion of updating of all the required documents.

Section	Requirement
5.3.4	The safety officer will assist in the writing and development of the team's hazard analyses, failure modes analysis, and procedures.

Verification Plan:
The team president is responsible for assigning responsibilities for the rest of the team leadership. The team president will inspect the progress made and work done by each

member of the team leadership. As the hazard analyses, failure modes analyses, and procedures were developed, the president verified that the team safety officer was involved.

Section	Requirement
5.4	Teams will abide by all rules set forth by the FAA

Verification Plan:
The Safety Officer is responsible for ensuring the team's adherence to FAA guidelines.

6.2.2 Updated Derived Requirements

6.2.2.1. Vehicle Derived Requirements

Requirement	Justification
Metallic components may only be used when non-metallic alternatives are proven insufficient.	The competition rules prohibit the use of excessive and/or dense material in the construction of the vehicle per Req. 2.23.10. This is to ensure that the vehicle is constructed with minimal use of metallic materials.

Verification Plan:

The Team Vice President is responsible for the final design of the launch vehicle. Adherence to this derived requirement requires testing, analysis, and final inspection of the vehicle.

Metallic component use must be justified via one of several alternative methods:

- Material failure calculations, simulations, and/or testing of non-metallic alternatives shows that metals are required.
- COTS components are utilized and no non-metallic alternatives exist
- Where fiber composites are structurally sound, a detailed feasibility study shows their use to be infeasible
- Hardware is determined to be in the critical load path of the recovery harness, since metals are well-characterized materials with ductile failure

An audit of all metallic components has been performed in advance of submission of the Critical Design Review, and all components have been appropriately justified. No later than one week prior to the submission of the Flight Readiness Review, the Vice President will perform an audit of all mechanical testing to ensure any components with testing-based rationale have been appropriately validated.

Requirement	Justification
The airframe design and construction must be able to accommodate multiple internal arming switches which have clear external access.	Per Req. 3.6, altimeters must be activated by a dedicated arming switch which is externally-accessible. Per Req. 3.7, these arming switches must not be able to be disarmed during flight. Internal arming switches for altimeters and other electronics must be internal to the airframe to protect these switches from aerodynamic manipulation.

Verification Plan:

The Team Vice President is responsible for the final design of the launch vehicle. This derived requirement is enforced via inspection of the design to ensure internal arming switches are present in the design, can be wired into their respective systems from the locations selected, and are freely accessible from the outside of the vehicle. An audit of the vehicle design has been performed before the submission of the Project Silverstein CDR.

Requirement	Justification
Each vehicle subsystem must have a center of mass along the centerline of the vehicle.	Rocket trajectory is simulated using masses lumped to the centerline of the vehicle. Asymmetry in the mass may cause unexpected deviation from the flight profile.

Verification Plan:
The Team Vice President is responsible for the final design of the launch vehicle. This derived requirement will be verified by analysis of the design and inspection of the as-built system. Led by the Vice President, each vehicle subsystem team will perform an audit of their respective subsystem to ensure mass components are strategically placed. During the audit, members will verify that the current system design is symmetrically balanced through the use of an appropriate CAD model or hand calculation, or demonstrate the ability for components to be easily rearranged. An example of this would be a 3d-printed mounting bracket for the subsystem, which can be easily modified and re-printed. This audit will take place during the construction of the full scale launch vehicle and must be completed at least 48 hours prior to the full-scale test flight. Any subsystem found to not be meeting this requirement will have its mass adjusted accordingly.

Requirement	Justification
The airframe will be restricted from designs utilizing asymmetric structural response.	Rocket trajectory is simulated by ignoring structural response. Asymmetry in the structural response may cause unexpected deviation from the flight profile.

Verification Plan:
The Team Vice President is responsible for the final design of the launch vehicle. This derived requirement will be verified by either inspection of the design or by simulation of the structural response. Where symmetric geometries are utilized on the vehicle, inspection of the design to confirm symmetry will be completed no less than 1 week before the completion of the Critical Design Review. Where asymmetric geometries are used, structural analysis must be performed to demonstrate that off-axis deformation of the structure is no greater than 1% of the total deformation under load. This audit has

been performed in advance of submission of the Critical Design Review, and all systems have been found to be compliant.

Requirement	Justification
Altitude Assurance System will be restricted to extending drag-producing devices aft of the burnout CG.	Extended drag-producing devices that are a part of the altitude assurance system are classified as structural protuberances by the RR-SL team. Per Req. 2.16, these devices may only act aft of the burnout CG.

Verification Plan:
The Team Vice President is responsible for the final design of the launch vehicle. Under guidance of the Vice President, the Altitude Assurance team will perform an audit of the Altitude Assurance subsystem to ensure that the device is positioned below the burnout CG, using both analysis and inspection of the as-built rocket. The burnout CG is known from both OpenRocket calculations and physically balancing the assembled rocket with no propellant. This audit has been performed in advance of submission of the Critical Design Review, and all systems have been found to be compliant.

Requirement	Justification
The Altitude Assurance System must be capable of decreasing launch vehicle apogee by 1700 ft.	Performance calculations, petal performance, margin, req 2.1

Verification Plan:
The Team Vice President is responsible for the final design of the launch vehicle. Under guidance of the Vice President, the Altitude Assurance team will verify this requirement via analysis of the design. The team will perform an audit of the Altitude Assurance subsystem every time that a change is made to the flight model to ensure that the drag produced by the petals is sufficient to decrease the launch vehicle apogee by 1700 ft. These calculations will be additionally refined with every test of the Altitude Assurance subsystem to ensure that the drag model accurately represents the flap behavior. This audit has been performed in advance of submission of the Critical Design Review, and all systems have been found to be compliant.

6.2.2.2. Recovery Derived Requirements

Requirement	Justification
All energetic devices must be handled using COTS electronics.	The team is not experienced in experimenting with energetic devices. Handling energetic devices with COTS electronics will remove variability

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle, and will verify this requirement via inspection of the design. All energetic devices will be identified individually, and any electronics used to interface with these devices will be subsequently identified. Any non-COTS components identified during this audit will be listed, and the responsible team members will select alternatives. This audit has been performed in advance of submission of the Critical Design Review, and all hardware has been found to be compliant.

6.2.2.3. Payload Derived Requirements

Requirement	Justification
The method used for locating the rocket will be strictly applicable to communication with a probe on another planet	Derived from Req. 4.1, the team determined that the phrasing "adhere to the intent of the challenge" as indication that our solution should be viable on another planet with no existing technology

Verification Plan:
As this derived requirement is deeply integrated into the design of the payload, the payload team is responsible for ensuring compliance, which will be done via analysis and inspection of the design. The team has and will analyze the current methods in use for communicating with other planets, and has eliminated any methods deemed "not in the spirit of the competition." The team will continue to seek input from the NASA panel of judges and advisors in order to conceptually verify that our approach to locating the rocket represents a viable solution to interplanetary probe communication. The Vice President will further verify this requirement by inspecting the work of the payload team. Both the audit internal to the payload team and the separate audit from the vice president have been performed in advance of submitting the Critical Design Review.

Requirement	Justification
The payload experiment must fully fit inside the nose cone	The vehicle team has concluded that the payload must fit entirely inside the nose cone

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle, and will verify this requirement via inspection of the system during both design and construction. Under direction of the vice president, the payload team will perform an audit of the payload experiment every time that the nose cone is changed to ensure that the system fits entirely inside the nose cone. Their design will be further verified by the CAD model to fit the nose cone. This audit has been performed in advance of submission of the Critical Design Review, and all systems have been found to be compliant. Once construction of the physical system has begun, test-fits of the payload into the nose cone will be performed to ensure continued compliance for the as-built system.

Requirement	Justification
The payload must not deploy from the launch vehicle	The vehicle team has determined that the added safety and mission risk caused by payload deployment are not necessary to successfully complete this year's mission

Verification Plan:
The Vice President is responsible for the final design of the launch vehicle. At least two weeks before the submission of the CDR report, the Vice President has inspected the final vehicle design. Since the design does not incorporate the deployment of the payload system, a redesign was not issued.

6.3 Budget

6.3.1 Line Item Budget

Component Level Budget				**TOTAL:**	**$16,328.00**
Item	**Price**	**Qty**	**Shipping**	**Total**	**Vendor**
Equipment					
Voron 2.4	$940.00	1	$0.00	$940.00	3d Printers Bay
Voron 0.1	$493.00	1	$0.00	$493.00	3d Printers Bay
LiPo Battery Charger	$80.00	1	$0.00	$80.00	Hobby King
LiPo Battery Bag	$5.00	2	$0.00	$10.00	Hobby King
Soldering And Rework Station	$200.00	1	$0.00	$200.00	Amazon

Wire Brush	$15.00	1	$0.00	$15.00	Amazon
Electrical Vise	$30.00	1	$0.00	$30.00	Amazon
Solder Hands	$25.00	1	$0.00	$25.00	Amazon
Hand Clamp	$8.00	2	$0.00	$16.00	Amazon
Bar Clamp 4 Pack	$16.00	1	$0.00	$16.00	Amazon
Cobalt Drill Index	$200.00	1	$0.00	$200.00	Amazon
Pliers, Wrenches	$94.00	1	$0.00	$94.00	Amazon
			Section Total:	**$2,119.00**	
General Consumables					
Solder	$25.00	1	$0.00	$25.00	Amazon
B/W/R 22 Gauge	$12.00	3	$0.00	$36.00	Amazon
B/R 18 Gauge	$10.00	2	$0.00	$20.00	Amazon
Gf30 Nylon 3d Printer Filament	$185.00	1	$15.00	$200.00	3dxtech
Pla Plus Filament	$25.00	3	$0.00	$75.00	Amazon
Epoxy	$172.00	1	$0.00	$172.00	Total Boat
Fine Adjustment Cable Ties	$17.00	1	$8.00	$25.00	Mcmaster Carr
Electrical Tape	$4.00	6	$0.00	$24.00	Amazon
Solo Cups	$5.00	1	$0.00	$5.00	Amazon
Rail Buttons	$8.00	4	$5.00	$37.00	Rail Buttons
M2/M3/M4/M5 Bolts	$25.00	2	$0.00	$50.00	Amazon
Popsicle Sticks	$4.00	1	$0.00	$4.00	Amazon
Duct Tape	$13.00	1	$0.00	$13.00	Amazon
Aluminum Wide Rivets	$13.00	1	$3.00	$16.00	Mcmaster Carr
Aluminum Narrow Rivets	$10.00	1	$3.00	$13.00	Mcmaster Carr
Aluminum Billet	$146.00	1	$0.00	$146.00	Mcmaster Carr
Protoboard	$12.00	1	$0.00	$12.00	Amazon
			Section Total:	**$873.00**	
Rocket Body					
G12 Body Tube	$46.00	10	$27.00	$487.00	Wildman Rocketry
Nosecone	$150.00	1	$15.00	$165.00	Wildman Rocketry
Mica Insulation Sheets	$85.00	1	$14.00	$99.00	Mcmaster Carr
Spray Paint	$6.00	3	$0.00	$18.00	Amazon
14" Coupler	$78.00	2	$14.00	$170.00	Madcow Rocketry
G10 Sheet	$18.00	4	$20.00	$92.00	Wildman Rocketry

		Section Total:		**$1,904.00**	
Altitude Assurance					
2 Ft X 1/4" Diameter Uhmwpe Rod	$3.00	1	$11.00	$14.00	Mcmaster Carr
Ptfe Film Tape	$15.00	1	$0.00	$15.00	Amazon
16mmx75mm Air Cylinder	$12.00	2	$0.00	$24.00	Amazon
2-Way Solenoid Valve	$17.00	2	$0.00	$34.00	Amazon
Altimeter	$10.00	2	$10.00	$30.00	Adafruit
Control Computer	$15.00	2	$8.00	$38.00	Digikey
Absolute Position Encoder	$8.00	6	$12.00	$60.00	Sparkfun
		Section Total:		**$215.00**	
Motor					
Motor Case	$560.00	1	$20.00	$580.00	Wildman
Motor	$350.00	3	$40.00	$1,090.00	Wildman
75mm Motor Tube	$40.00	1	$7.00	$47.00	Madcow Rocketry
75mm Motor Retainer	$65.00	1	$7.00	$72.00	Wildman Rocketry
		Section Total:		**$1,789.00**	
Subscale					
54mm Motor Retainer	$31.00	1	$0.00	$31.00	Wildman Rocketry
Motor Reload Kit 38mm 720 Case	$104.00	1	$0.00	$104.00	Wildman Rocketry
Centering Ring	$7.00	3	$0.00	$21.00	Madcow Rocketry
Motor	$120.00	1	$40.00	$160.00	Wildman Rocketry
54mm Motor Tube	$30.00	1	$7.00	$37.00	Madcow Rocketry
4" Airframe Tube	$272.00	1	$23.00	$295.00	Madcow Rocketry
4" Coupler	$29.00	1	$16.00	$45.00	Madcow Rocketry
4" 4:1 Ogive Nose Cone	$38.00	1	$18.00	$56.00	Madcow Rocketry
		Section Total:		**$749.00**	
Payload					
Raspberry Pi 4 Kit	$120.00	2	$0.00	$240.00	Amazon
Cots Telemetry Modules	$80.00	2	$10.00	$170.00	Sparkfun
750 Mah 4s Battery	$38.00	2	$0.00	$76.00	Getfpv
Sd Cards	$9.00	4	$0.00	$36.00	Amazon
Mountable Xt60 Plugs	$12.00	1	$0.00	$12.00	Amazon
22awg Silicone Wire	$15.00	1	$0.00	$15.00	Amazon

18awg Silicone Wire	$15.00	1	$0.00	$15.00	Amazon
Accelerometer	$20.00	3	$10.00	$70.00	Adafruit
Altimeter	$10.00	4	$10.00	$50.00	Adafruit
750 Mah 4s Battery	$38.00	1	$0.00	$38.00	Getfpv
			Section Total:	**$722.00**	
		Recovery			
Rrc3 Altimeter	$74.00	2	$7.00	$155.00	Wildman
Rocket Locator				$0.00	
Recovery Harness	$72.00	2	$7.00	$151.00	Wildman
Avionics Bay	$50.00	2	$10.00	$110.00	Madcow Rocketry
750 Mah 4s Battery	$38.00	1	$0.00	$38.00	Getfpv
Hybrid Supercapacitor	$11.00	2	$4.00	$26.00	Digikey
Nylon Shear Pins	$4.00	2	$5.00	$13.00	Apogee Rockets
Skyangle Cert-3 Large	$139.00	1	$13.00	$152.00	Madcow Rocketry
Drogue Chute	$86.00	1	$29.00	$115.00	The Rocket Man
Mica Insulation Sheet	$85.00	1	$14.00	$99.00	Mcmaster Carr
			Section Total:	**$859.00**	
		Travel			
Mileage Reimbursement (4 Per Car)	$415.00	5	$0.00	$2,075.00	N/A
Student Hotel (4 Per Room)	$135.00	20	$0.00	$2,700.00	N/A
Mentor Hotel	$135.00	4	$0.00	$540.00	N/A
Meals (Per Person)	$15.00	40	$0.00	$600.00	N/A
			Section Total:	**$5,915.00**	
		Branding			
Stickers (Bulk Order)	$100.00	1	$4.00	$104.00	Sticker Mule
Team Presentation Polos	$18.00	20	$0.00	$360.00	Bagnoche Sports
Team Event T-Shirts	$10.00	20	$0.00	$200.00	Bagnoche Sports
			Section Total:	**$664.00**	
		Outreach			
Vehicle Mileage (3 Events)	$11.50	6		$69.00	N/A
Meals (10 People, 3 Events)	$15.00	30		$450.00	N/A
			Section Total:	**$519.00**	

6.3.2 Funding Acquisition Plan

Since the PDR, the Rose Rocketry Student Launch Team has secured additional sources of funding. The team submitted and was approved a One Time Funding Request (OTFR) from our Student Government Association (SGA). This approximately $10,000 request secured funding for a majority of this season's tools and material cost. The team also received an anonymous $1,000 donation. This donation will be used as a last resort in the event of unexpected circumstances as a source of emergency funding. The also continues to received funding frim the Branimun

Each year every competition team inside the BIC submits a budget, which is later awarded in full or is adjusted. This academic year, the BIC received a 40% budget cut by school administration, in effort to make up for ongoing COVID-19 expenses. As a result, every BIC team also received a budget cut. Rose Rocketry's BIC budget is $3000 for the 2021-2022 academic year.

In a similar process to the BIC, every club on campus submits a budget to SGA. These budgets are then reviewed, edited, and awarded. However, this process only applies to clubs fully approved and recognized by SGA. Due to miscommunication, unclear instructions, and contradictory SGA policies, Rose Rocketry is not a fully recognized SGA club. Instead, Rose Rocketry currently holds a probationary club status and is ineligible for a full budget. This means we do not have any funds set aside by SGA for the team and no dollar amount we expect to receive. To receive SGA funding, the club must submit special One Time Funding Requests (OTFR). This is a lengthy process which can take anywhere from one week minimum to 4 weeks maximum to obtain funding for the requested items. This places a unique risk on the team of not having funding for parts ordered any less than a month or more in advance. However, due to a majority of club activities and competitions being canceled last academic year, SGA has a surplus of funds and is able to support the setup and operational cost of Rose Rocketry, so long as OTFRs are submitted in a timely manner.

In addition to BIC and SGA funding, Rose Rocketry has received a $1000 donation from an anonymous donor to support team efforts.

6.3.3 Material Acquisition Plan

Due to the timeline issues laid out above with SGA and the ongoing global supply chain issues, the team forseas the ordering and receiving of parts to be one of the biggest challenges faced this competition season. In order to be better prepared for competition, the team has added additional milestones throughout the season, such as the launching of a level 2 fiberglass kit in November to gain experience before building and launching the subscale rocket. Although these additional milestones will benefit the team, they add an additional timeline constraint to an already tight timeline. We have already run into problems with parts being out of stock, such as the RRC3 altimeter, and SGA taking weeks to release funding, such as not having funding for the first 5 weeks of the school year. In order to ensure the team has everything required to complete competition and derived milestones, the team is ordering components as soon as possible and prioritizing discussions of component funding at team meetings. Because we do not have a specified budget from SGA, at any point a component is considered to be a leading contender in a leading design alternative, funding for that component will be submitted through an OTFR and ordered. This is done due to the high likelihood that by the time a system component is finalized, there will not be enough time left to submit an OTFR, wait for approval, wait for shipping, and add the component to its respective system.

6.4 Timeline

6.4.1 Major Project Deadlines

NASA + Indiana Rocketry Schedule + Rose Rocketry Deadlines

- January 3 - Subscale Flight Deadline
- January 3 - Completed gridded map due
- January 3 - CDR, presentation slides, flysheet due
- January 21 - Altitude Assurance PLA prototype construction, Payload Integration deadline
- January 28 - Full-Scale Rocket Construction Deadline
- January 31 - Ground Test of Full-Scale Rocket
- February 5 - Altitude Assurance final version deadline
- February 12-13 - High Power Launch, Vehicle Demonstration Flight
- February 28 - Payload Software Deadline
- March 7 - Vehicle demonstration flight deadline
- March 7 - Flight Readiness Review (FRR) report, presentation slides, and flysheet due to NASA project management team by 8:00 a.m. CST.
- March 11 - Remaining Payload Tuning/Testing Finished
- Saturday & Sunday, March 12-13, High Power Launch, Payload Demonstration Flight
- April 4 - Payload Demonstration Flight and Vehicle Demonstration Re-flight deadlines
- April 4 - FRR Addendum Due
- April 19-20 - Travel to Huntsville, AL
- April 20-24 - Competition Week
- May 9 - Post-Launch Assessment Review (PLAR) Due

7. Appendices

7.1 Citations

[1] “Rocket Motors, kits, and supplies from Wildman Rocketry,” *wildmanrocketry.com*. [Online]. Available: https://wildmanrocketry.com/.

[2] “Rocket Motors, kits, and supplies from Wildman Rocketry,” *wildmanrocketry.com*. [Online]. Available: https://wildmanrocketry.com/.

[3] “High Power Rocketry Supplies, advanced model rocketry, rocket kits,” *LOC Precision / Public Missiles Ltd*. [Online]. Available: https://locprecision.com/.

[4] “Ask us - drag of cylinders & cones,” *Aerospaceweb.org | Ask Us - Drag of Cylinders & Cones*. [Online]. Available: http://www.aerospaceweb.org/question/aerodynamics/q0231.shtml.

[5] *CERT-3 Chutes*. [Online]. Available: http://www.b2rocketry.com/Cert-3.html

[6] Cavender, D., 2022. High Powered Video Series Counterpart Documents. Available at: <https://www.nasa.gov/sites/default/files/atoms/files/sl_video_instruction_book.pdf>

7.2 Flight Preparation Procedure

All steps should be checked by at least two team members.

Payload Preparation

Night before:

- ☐ Charge 2200mAh battery to full
- ☐ Screw in every electrical component on the Payload Sled

At work table on launch day:

- ☐ Attach the battery to the bottom of the Payload Sled
- ☐ Plug in the battery
- ☐ Put the payload retention system in the nose cone and screw in
- ☐ Ensure payload arming switch is turned off to avoid battery drain – there should be no beeping coming from the payload bay

Recovery Electronics Preparation

Before the day of the launch:

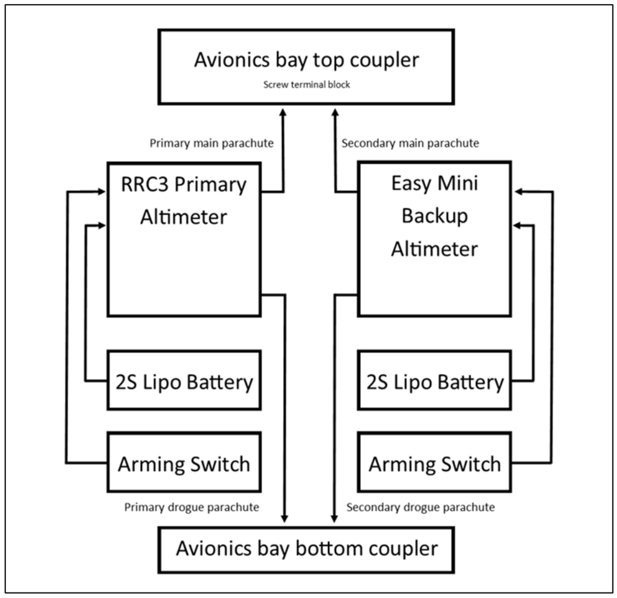
***The following step involves the handling of lipo batteries, a known fire hazard. Lipo batteries should be treated with care, never left unattended, and stored in the team designated fire proof bag.**

- ☐ Inspect lipo batteries for any signs of damage. This includes dents, swelling, broken connectors, exposed wire,etc. Notify the team safety officer of any damaged batteries before proceeding.
- ☐ Charge two 2S lipo batteries to full. One for each altimeter
- ☐ Prepare all relevant software and documentation for altimeters:
    - ☐ EasyMini manual
    - ☐ RRC3 manual
    - ☐ Altus Metrum AltOS configuration software

    ***Launch locations may not have cellular signal, so all documentation must be downloaded ahead of time.**
- ☐ Ensure the range kit has the required items:
    - ☐ Altimeters (may be installed)
    - ☐ Batteries and connectors
    - ☐ Spare wire and wire strippers
    - ☐ Ematches
    - ☐ Black powder (incl. scales and containers)
    - ☐ Eyeglass screwdrivers for screw terminals

    ***Failure to include any of these components will likely make repair or modification of the avbay configuration difficult or impossible.**

- [ ] Ensure that no charges or ematches are connected to the avbay from previous flights. ***All pyrotechnics must be disconnected until final assembly. Even without black powder, ematches are potentially dangerous and should be treated as energetic devices.**
- [ ] Assemble the avbay wiring according to the schematic below. Be sure to match standard wire colors whenever possible.



- [ ] Before plugging in batteries, verify that the polarity of the connectors matches the + and - terminals marked on the altimeter.
    - [ ] Additionally verify that the polarity of the battery and connector match. Hand-made and manufactured connectors alike may have incorrect wire coloring; any that do should be resoldered or discarded.
    ***Connecting polarity incorrectly may permanently damage the altimeters.**
- [ ] Once all schematics have been checked, ensure that switches are opened. **Wear safety glasses and have a Class B fire extinguisher ready while initially connecting batteries, as an accidental short may result in violent sparks or, in extreme cases, fire.**
- [ ] Inspect batteries for any damage. If any damage is found, dispose of batteries in a flammable waste disposal area.
  ***Damage to batteries may result in electrical fires. Therefore, damaged batteries must be disposed of safely and immediately.**
- [ ] Connect batteries. No altimeters should power up; if any do, inspect switch contacts for debris or shorts. Do not continue until the short is cleared.
  ***To minimize risk in the event charges are deployed accidentally, once pyrotechnics are armed, the altimeters absolutely must not be powered on until the rocket is on the launch pad or in another designated safe area as approved by the RSO. A shorted or unreliable switch may cause avionics to become armed in an unsafe location.**

- [ ] Close the switches associated with each altimeter, one at a time. Note the beep code for each altimeter and ensure that it is as expected based on the table of beep codes included in each altimeter's instructions. If a GPS tracker is also included, ensure that it acquires lock; it may need to be brought outdoors to acquire signal.
  *__Diagnosing altimeter issues before launch day allows more opportunity to debug potential issues or mis-configurations while access to club equipment and internet is readily available.__
- [ ] Before packing equipment away, ensure that all batteries are fully charged.
  *__A low battery may power on the computer and read continuity correctly but fail to provide enough current for deployment, resulting in a recovery failure.__

At the worktable on launch day:

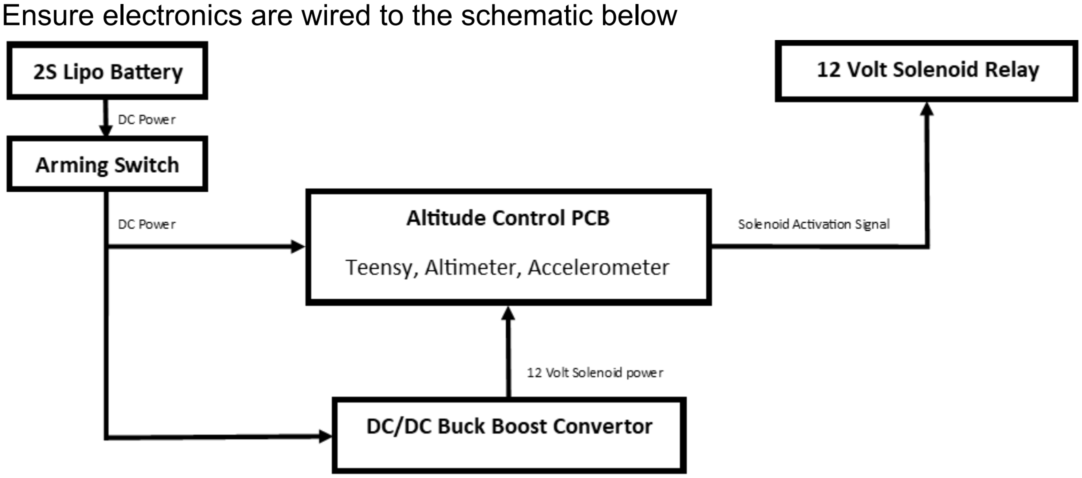
- [ ] Re-check the wiring against the schematic and ensure that no pyrotechnics are installed.
- [ ] Ensure that switches are opened.
- [ ] Inspect, secure and plug in batteries.
  *__The preceding steps mirror the day-before procedure and are intended to ensure that no components have been damaged in transport.__
- [ ] Ensure that all nuts on the sled side of the avbay are tightened.
  A loose sled may damage itself under the acceleration of the rocket or cause wires to become disconnected in flight.
- [ ] Insert the sled assembly into the avbay and secure the nuts on the other bulkhead. Ensure that no wires are caught in the edges of either bulkhead.
  Avbay coupler edges have the potential to tug loose or sever altimeter wires caught in them.
- [ ] As before, switch on each switch one at a time and verify beep codes or GPS lock, then switch all switches entirely off. If beep codes differ from expected, do not proceed until the issue is resolved.
- [ ] Immediately after avionics bay assembly and testing, insert the two Remove Before Flight (RBF) tags into their respective locations next to the arming switches.
  Failure to arm the altimeters will be catastrophic. This is an important step in the procedure checklist to ensure a successful flight.

Airbrake and Pneumatics Preparation

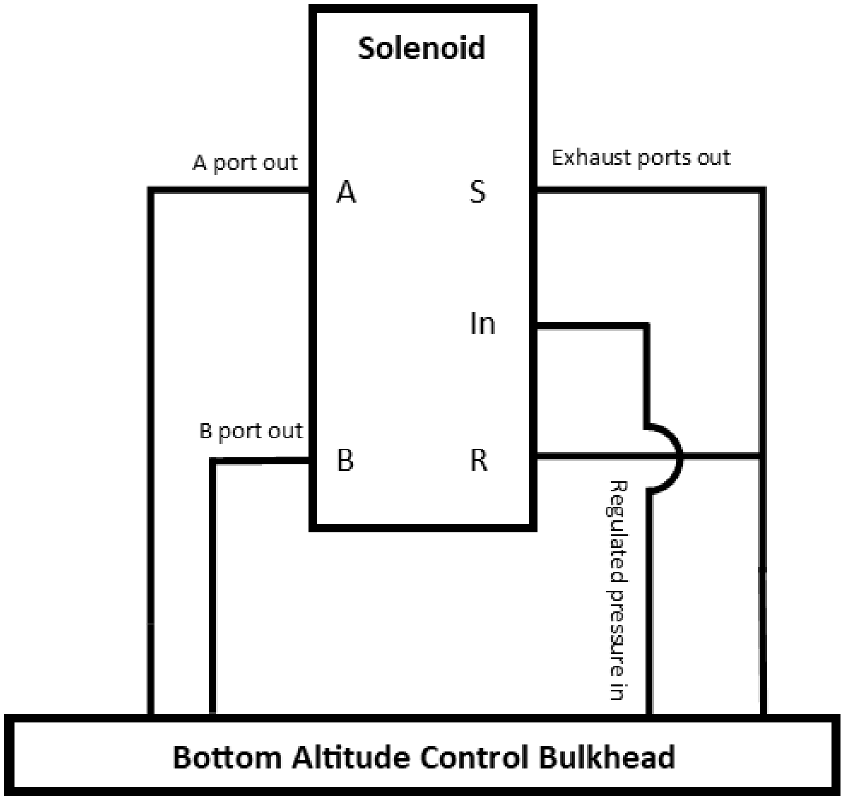
Before Day of Launch:

***The following step involves the handling of lipo batteries, a known fire hazard. Lipo batteries should be treated with care, never left unattended, and stored in the team designated fire proof bag until use.**

- [ ] Inspect lipo batteries for any signs of damage. This includes dents, swelling, broken connectors, exposed wire,etc. Notify the team safety officer of any damaged batteries before proceeding.
- [ ] Charge one 2S lipo battery
- [ ] Prepare all software and dependencies for altitude control computer
  - [ ] Teensy documentation
  - [ ] Altitude control computer code front team github
  - [ ] Arduino IDE
  - [ ] Any external libraries required for code compilation
- [ ] Plug the teensy into your computer and attempt to upload the latest altitude control code. **Do not continue until you are successfully able to compile and upload the latest code. It is important to ensure the altitude control computer is in a known state prior to the launch.**
- [ ] Inspect the altitude control sled and ensure all components are fastened securely.
  - [ ] Solenoid valve
  - [ ] Buck Boost Converter
  - [ ] Teensy
  - [ ] Altimeter
  - [ ] Accelerometer
  - [ ] Electronic wiring
  - [ ] Arming switch
  - [ ] Pneumatic Fitting
- [ ] Inspect Pneumatic tubing and fittings for any cracks, dents, or other defects. Replace tube or fitting if any defects are found.
- [ ] Ensure electronics are wired to the schematic below



☐ Ensure the pneumatics are plumbed according to the diagram below



The following steps involve pressurized air. Safety glasses must be worn to prevent eye injury from flying debris.

☐ Connect the air tank to the external compressor and regulator assembly. Close the tank's pressure relief valve and fill the tank to 150 PSI.

☐ Check that the onboard regulator is set to 90 PSI

At this time the pneumatic system is pressurized. Ensure no person or object is closer than 12 inches to the aero brakes. The brakes should be considered live and capable of actuating at any time.

☐ Inspect the pneumatic tubing and fittings for any signs of leaks. Pay close attention to fitting joints and tube connections.Do not proceed until any leaks are addressed.

☐ Leave the altitude control system pressurized for at least ten minutes. Verify that the tank pressure is still 150 PSI. Note: do not leave the system unattended.

☐ Test deploy the aerobrakesusing the manual override on the solenoid.

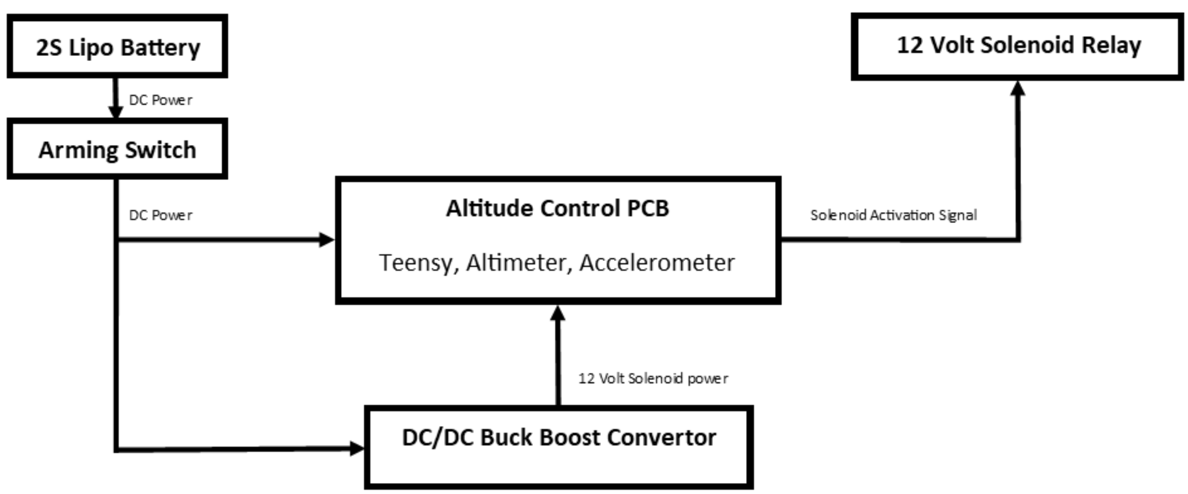
☐ Arm the altitude control computer. Ensure it follows the expected boot sequence for the uploaded software. This includes the deployment and retraction of the aerobrakes under computer control.

Only after successful deployment and retraction of the aerobrakes under computer control is the altitude control system considered ready for launch day.

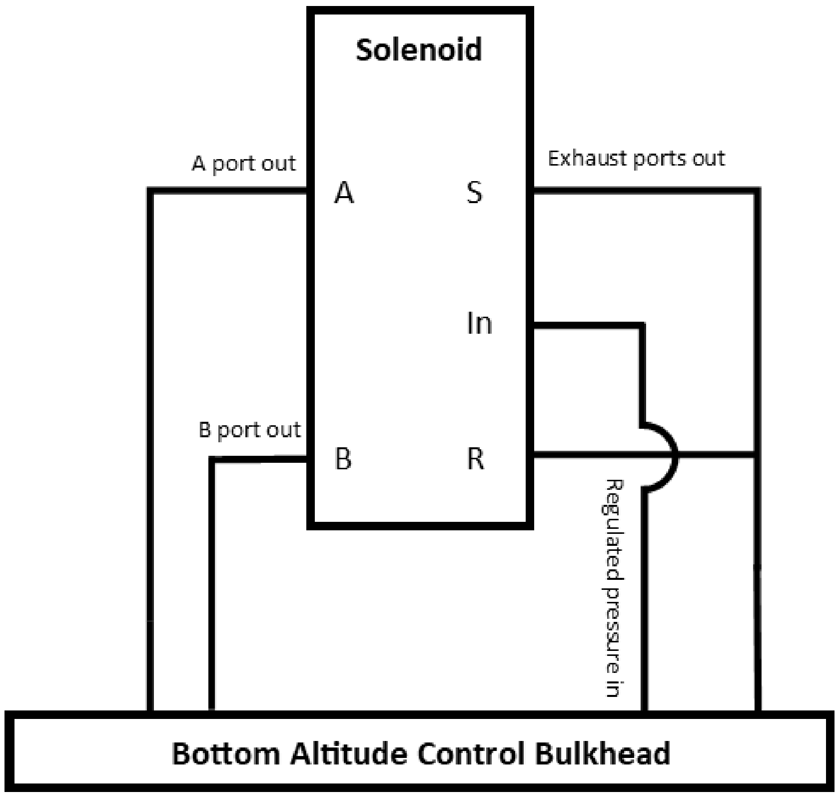
At the worktable on launch day:

***The following step involves the handling of lipo batteries, a known fire hazard. Lipo batteries should be treated with care, never left unattended, and stored in the team designated fire proof bag until use.**

- [ ] Inspect lipo batteries for any signs of damage from transport. This includes dents, swelling, broken connectors, exposed wire,etc. Notify the team safety officer of any damaged batteries before proceeding.
- [ ] Inspect the altitude control sled and ensure all components are fastened securely.
    - [ ] Solenoid valve
    - [ ] Buck Boost Converter
    - [ ] Teensy
    - [ ] Altimeter
    - [ ] Accelerometer
    - [ ] Electronic wiring
    - [ ] Arming switch
    - [ ] Pneumatic Fitting
- [ ] Inspect Pneumatic tubing and fittings for any cracks, dents, or other defects from transport. Replace tube or fitting if any defects are found.
- [ ] Ensure electronics are wired to the schematic below



☐ Ensure the pneumatics are plumbed according to the diagram below



The following steps involve pressurized air. Safety glasses must be worn to prevent eye injury from flying debris.

☐ Connect the air tank to the external compressor and regulator assembly. Close the tank's pressure relief valve and fill the tank to 150 PSI.

☐ Check that the onboard regulator is set to 90 PSI

At this time the pneumatic system is pressurized. Ensure no person or object is closer than 12 inches to the aero brakes. The brakes should be considered live and capable of actuating at any time.

☐ Inspect the pneumatic tubing and fittings for any signs of leaks. Pay close attention to fitting joints and tube connections.Do not proceed until any leaks are addressed.

☐ Leave the altitude control system pressurized for at least ten minutes. Verify that the tank pressure is still 150 PSI. Note: do not leave the system unattended.

☐ Test deploy the aerobrakesusing the manual override on the solenoid.

☐ Arm the altitude control computer. Ensure it follows the expected boot sequence for the uploaded software. This includes the deployment and retraction of the aerobrakes under computer control.

Only after successful deployment and retraction of the aerobrakes under computer control is the altitude control system considered ready for launch day.

Rocket Airframe and Recovery Preparation

***Gloves should be worn while handling fiberglass to avoid splinters.**

- ☐ Inspect all epoxy joints (fins, motor mount, nose cone bulkhead) for cracking or signs of wear.
- ☐ Quick-link the longest portion of the three-loop recovery harness to the top of the booster section.
- ☐ Thread the three-loop harness through the drogue airframe section. Ensure alignment and “this way up” markers are obeyed.
- ☐ Bolt the drogue tube to the booster coupler. Do not force bolts if they do not fit; double-check alignment if problems are encountered.
- ☐ Drogue harness assembly:
    - ☐ Accordion-fold the portion of the cord before the middle loop in a bundle about 12” long and wrap a single loop of masking tape around the center.
      ***Accordion-folding harnesses ensures that they do not become wrapped around the parachute, and the tape breaking provides damping in overly energetic deployments.**
    - ☐ Quick-link the drogue parachute to the middle loop of the harness.
    - ☐ Quick-link the far end of the harness to the bottom of the main avionics bay.
    - ☐ Accordion-fold the top half of the harness as before. Note that the bundle should be smaller than the previous.
    - ☐ Fold the drogue parachute in accordance with Appendix B.
    - ☐ Put both cord bundles into the drogue tube.
    - ☐ Put the wrapped drogue chute into the tube.
      ***The cords must be placed below the parachute so that, in the event of a weak deployment, the tension on the cord will pull the parachute loose.**
- ☐ Main harness assembly:
    - ☐ Connect the main chute and one end of the two-loop harness to the nose cone u-bolt with a quick link.
      ***Ensure that all parts are connected to one quicklink, rather than separate quicklinks on the u-bolt. Placing load across the u-bolt may cause unpredictable strain on the bulkhead.**
    - ☐ Accordion-fold the harness as before, leaving enough unfolded to comfortably reach the other end of the main tube.
    - ☐ Fold the main chute in accordance with Appendix B.
    - ☐ Slide the folded harness into the main tube, followed by the folded parachute. Check direction and alignment markers.
    - ☐ Attach the tube to the nose cone using nylon shear pins.
- ☐ Proceed to motor preparation.

Rocket Motor Preparation

- ☐ Prepare a work surface for motor assembly. It should be clean, dry, sheltered from wind as much as possible, and away from any sources of heat or flame.
  ***Motor reload kits contain many small parts and paper instruction sheets that may blow away in strong winds. Additionally, sources of heat present a risk of accidental ignition, and dirt or debris on the work surface may prevent motor components from forming a reliable seal.**
- ☐ Before beginning motor assembly, have ready:
  - ☐ All required motor hardware (may include cases, retaining rings, spacers, and seal disks as well as tools such as specialized wrenches)
  - ☐ Manufacturer instructions for the motor (2 copies); print ahead of time if possible
  - ☐ Synthetic grease
    From this point onward, anyone handling the motor or reload kit components must wear safety glasses. Additionally, rubber gloves are recommended while handling grease.
- ☐ Read through the instructions in their entirety before beginning.
- ☐ Unpack the reload kit. Identify all parts as specified by the instructions and ensure that nothing is missing.
- ☐ With a partner following along, assemble the motor according to manufacturer instructions. Describe each step out loud as you perform it. Perform any “optional but recommended” steps (for example greasing the liner) unless a clear reason exists not to do so.
  ***Describing steps out loud both allows your partner to verify the step and helps to prevent “autopiloting” that may lead to assembly mistakes.**
- ☐ Have your partner inspect the completed motor. Verify any dimensional information given in the instructions (typical thread depths or fit tolerances).
- ☐ Ensure that no parts from the reload are unused except as specified by instructions.
- ☐ Reinstall nozzle cover to prevent dust ingress.
- ☐ Install the motor in the rocket and hand-tighten the retainer.

Deployment Charges and Final Assembly

- ☐ Prepare charges as in Appendix C.
  ***After the following steps, the airframe will have the potential to separate violently if a charge is accidentally triggered. All personnel should stay clear of the area in front of and behind the rocket.**
- ☐ Install the main tube assembly onto the front of the avbay. Bolt into place, ensuring alignment as with other sections.
- ☐ Install the forward assembly into the front of the booster assembly and secure with shear pins.

Setup and Launch Procedure

Safety glasses should be worn at all times while handling the rocket once charges or the motor have been installed.

- ☐ Obtain approval to launch from the site RSO.
- ☐ Tilt the pad such that the designated “rocket side” of the rail faces upward.
- ☐ While one person steadies the rail, slide the rocket onto the rail until it reaches the lower stop.
- ☐ While steadying the rocket, rotate the pad back to vertical or the angle designated by the RSO.
- ☐ Instruct all non-essential personnel to return to the flight line.
  ***Those not involved in the readying of the rocket must be at a safe distance before charges are armed.**
- ☐ Power on all altimeters. Check continuity beeps as before. Do not proceed unless beeps are as expected.
  ***There is a small chance connections may come loose on the way to the launch pad.**
- ☐ If the configuration calls for GPS to be powered on at the pad, do so and wait for lock.
- ☐ Strip wires as necessary, then twist together the bare leads of the igniter.
  ***Ensuring that the igniter leads are shorted together reduces the risk of static discharge or other accidental energization firing the igniter.**
- ☐ Insert the igniter into the motor until it stops. Pull the igniter out slightly and reinsert to ensure it is not caught on a grain gap.
  ***Motors will only ignite reliably if the igniter is installed all the way to the top of the motor.**
- ☐ Secure the igniter with tape, a plastic cap, or as otherwise specified by the manufacturer.
- ☐ Tap the alligator clips together to check for voltage.
  ***If the controller is accidentally energized, this step will cause sparks to alert you to the issue.**
- ☐ Connect the igniter leads to the alligator clips. Wrap any remaining leads around the outside of the clips.
  ***Additional wrapping of leads helps to eliminate poor connections.**
- ☐ If the launch control system offers a continuity test, use it to ensure that the igniter is functional and connected properly.
- ☐ Return to the flight line and continue with the next procedure.

Flight Procedure

- ☐ Before flight, assign the following roles:
    - ☐ Visual tracker (2 or more)
    - ☐ GPS operator
    - ☐ Videographer (2 if possible)
    - ☐ Flight Event Recorder (2 if possible)
- ☐ Visual trackers: Spread out on the flight line. Ensure that you have a means of communication with the team.
  ***Multiple visual lines on the rocket will allow triangulation in the event of a GPS failure.**
- ☐ Videographer: Ensure you have an unobstructed view of the rocket.
  ***In the event of a catastrophic failure, video may be the only concrete evidence of the flight. Prioritize capturing the entire flight over "detail shots."**
- ☐ GPS operator: Ensure that the tracking setup is ready and transmitting coordinates.
- ☐ Flight Event Recorders: Ready a checklist from Appendix A as well as a writing implement.
  Note: Some items on this checklist refer to "without airframe failure". In the event of a mechanical failure of the airframe in flight, these checkboxes help pinpoint the exact moment of failure.
- ☐ Signal to the RSO that the team is ready.
- ☐ During the flight:
    - ☐ Visual trackers identify landmarks on the horizon as the rocket descends to aid in triangulation.
    - ☐ GPS operators call out altitude figures as they are available. This helps to identify flight events. (Note that GPS units do not always yield reliable altitude numbers.)
    - ☐ Video recorders film the rocket. Sighting over your camera or phone may yield better results than looking at the viewfinder or screen.
    - ☐ Event recorders record the flight in accordance with their checklists.
- ☐ Wait until given a range-clear signal from the RSO to begin searching.
- ☐ During recovery:
    - ☐ Visual trackers stay where they are and direct searchers via radio.
    - ☐ Depending on personnel availability, videographers may either act as visual trackers using a frame of video as reference or join the search.
    - ☐ Event recorders and GPS operators

7.3 Flight Event Checklist

- ☐ Liftoff
- ☐ Burnout (without airframe failure)
    - ☐ Petal deployment (if visible)
    - ☐ Petal retraction before apogee (if visible)
- ☐ Apogee (without airframe failure)
    - ☐ Primary charge
        - ☐ Drogue deploys with primary
    - ☐ Secondary charge
        - ☐ Drogue deploys with secondary
    - ☐ Petal retraction if deployed (if visible)
- ☐ Stable descent under drogue
    - ☐ Primary main charge: ________ feet
        - ☐ Main deploys with primary
    - ☐ Secondary main charge: ________ feet
        - ☐ Main deploys with secondary
- ☐ Touchdown under main

7.4 Parachute Folding

- Draw the parachute lines together with the peak of the parachute opposite them.

- Double the lines in the center of the parachute.

- Fold the parachute in thirds vertically, covering the lines.

- Fold the parachute in thirds horizontally. The number of folds in this step may be varied for tube fitment.

- Roll the parachute vertically (along the axis of the shroud lines).

- Place the parachute in the center of the chute protector. Attach the chute protector's eyelet to the parachute's quicklink.
- Fold the top and bottom of the chute protector over the chute.
- Roll the sides of the chute protector around the parachute. The net result should be a "burrito wrap" shape.
- Ensure that the material of the parachute is not visible from the outside.
  ***If nylon is exposed to ejection gases, it will likely be damaged, resulting in a recovery failure.**

7.5 Charge Preparation

***Black powder is a low explosive and is very easily ignited. Safety glasses must be worn whenever handling black powder, and heat sources or flames must not be allowed within 25 feet of it.**

- Gather materials: measured black powder, funnel, igniter, masking tape, cable ties, marker, scissors, vinyl gloves
- Prepare charge pouches:
    - Cut the vinyl glove at the base of the finger to make a charge pouch. Repeat for necessary charges.
- Prepare the igniter:
    - Pull back on the igniter element cover and remove. Pull back on the exposed wire cover and remove.
    - Stripping the wire for more exposure may be necessary.
- Insert funnel into one charge pouch and slowly pour the measured black powder. Sometimes it is necessary to gently shake the funnel if the flow of black powder is interrupted. **Make sure all the black powder has escaped the funnel before removing the funnel. Failure to do this may result in a chemical spill.**
- Insert igniter into the now filled charge pouch until the element is completely covered with black powder.
- Twist charge pouch around igniter wire **tightly** and secure with a cable tie.
- Wrap the charge pouch **tightly** with masking tape.
- Label the black powder amount on the wire of the igniter
- Verify charge preparation with the team Safety Officer