NC STATE UNIVERSITY

Tacho Lycos

2023 NASA Student Launch

Preliminary Design Review

High-Powered Rocketry Club at NC State University
911 Oval Drive
Raleigh, NC 27606

October 26, 2022

Common Abbreviations and Nomenclature

AGL	=	Above Ground Level
APCP	=	Ammonium Perchlorate Composite Propellant
APRS	=	Automatic Packet Reporting System
AV	=	Avionics
BP	=	Black Powder
CDR	=	Critical Design Review
CG	=	Center of Gravity
COTS	=	Commercial Off The Shelf
CP	=	Center of Pressure
EIT	=	Electronics and Information Technology
FAA	=	Federal Aviation Administration
FMEA	=	Failure Modes and Effects Analysis
FN	=	Foreign National
FRR	=	Flight Readiness Review
HEO	=	Human Exploration and Operations
HPR	=	High-Power Rocketry
HPRC	=	High-Powered Rocketry Club
IMU	=	Inertial Measurement Unit
L3CC	=	Level 3 Certification Committee (NAR)
LCO	=	Launch Control Officer
LE	=	Leading Edge
LRR	=	Launch Readiness Review
MAE	=	Mechanical & Aerospace Engineering
MSDS	=	Material Safety Data Sheets
MSFC	=	Marshall Space Flight Center
NAR	=	National Association of Rocketry
NCSU	=	North Carolina State University
NFPA	=	National Fire Protection Association
PDR	=	Preliminary Design Review
PLAR	=	Post-Launch Assessment Review
PPE	=	Personal Protective Equipment
RAFCO	=	Radio Frequency Command
RF	=	Radio Frequency
RFP	=	Request for Proposal
RSO	=	Range Safety Officer
SL	=	Student Launch
SLS	=	Space Launch System
SME	=	Subject Matter Expert
SOCS	=	Surrounding Optics and Communication System
SOW	=	Statement of Work
STEM	=	Science, Technology, Engineering, and Mathematics
TAP	=	Technical Advisory Panel (TRA)
TE	=	Trailing Edge
TRA	=	Tripoli Rocketry Association

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1 Summary of Report

1.1 Team Summary

1.1.1 Team Name and Mailing Address

Name: High-Powered Rocketry Club at NC State, Tacho Lycos

Mailing Address: 911 Oval Drive, Raleigh 27606

Primary Contact: Meredith Patterson, mapatter@ncsu.edu, (919) 448-8001

1.1.2 Mentor Information

Name: Alan Whitmore

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1.1.3 Time Spent on PDR Milestone

The team spent an approximate total of 280 hours working towards completion of PDR milestone.

1.1.4 Social Media Accounts

Twitter	Facebook	YouTube	LinkedIn	TikTok	Instagram
@NCSURocketry	/TachoLycos/	ncsurocketry	/tacholycos/	@NCSURocketry	@NCSURocketry

1.2 Launch Vehicle Summary

1.2.1 Official Target Altitude

The official target apogee of the launch vehicle is 4500 ft. AGL.

1.2.2 Motor Choice

The current leading motor choice is the Aerotech L1520T.

1.2.3 Vehicle Size and Mass

The leading launch vehicle design is 104.5 in. long and 6.17 in. diameter. The launch vehicle will be constructed in 6 different sections. All airframe sections will be G12 fiberglass. The nose cone will be a 5:1 ogive. The fin can will be constructed using a removable fin design. Dimensions and design choices are presented in Section 3.3. The leading launch vehicle weight is 43.2 lb. while fully loaded. A breakdown of the weights of components are in Table (7).

1.2.4 Recovery System

The leading recovery system design uses two RRC3 altimeters to control deployment events. At apogee, a Fruity Chutes 18 in. classic elliptical parachute will be used as the drogue parachute. The main parachute will be an Iris Ultracompact 120 in. Fruity Chutes parachute deployed at 550 ft. AGL. This combination of parachutes is expected to have a descent time of 75.27 seconds and a maximum drift distance of 2208 ft. Recovery specifics are discussed in Section 3.4.3.

1.3 Payload Summary

The payload for this year's competition is the Surrounding Optics and Communication System (SOCS). SOCS consists of a RAFCO system and a camera system in the fin can of the launch vehicle. SOCS will receive RAFCO transmitted over APRS. These commands consist of camera controls and editing commands. These commands are to be interpreted and then carried out by SOCS within 30 seconds of receiving, utilizing an on-board camera system that is capable of rotating 360° around an axis normal to the ground.

SOCS's RAFCO system consists of two dipole antennas mounted on the launch vehicle. The camera system consists of four cameras mounted to four servos attached directly to the primary payload computer. The computer will interpret and act upon RAFCO commands, instructing the system and image editing software. After the command sequence has been completed, the resulting image will be saved on the computer. Payload specifics are discussed in Section 4.

2 Changes Since Last Document

2.1 Changes Made to Launch Vehicle

Several changes were made to the launch vehicle since the proposal was submitted. The changes and their justifications are provided in Table (1) below. Further justification for design choices is made in Section 3.3

Table 1: Changes made to the launch vehicle since proposal submission.

Change	Justification for Change
Nose cone changed from 4:1 ogive to 5:1 ogive	Both nose cones have similar aerodynamic properties, but the 5:1 nose cone is larger and heavier. This provides more weight in the forward end of the launch vehicle moving the CG forward and improving stability.
Fin size decreased	As launch vehicle dimensions and weight estimates were improved, smaller fins were necessary in order to stabilize the launch vehicle. The root chord was changed from 18 in. to 10 in.
Motor choice changed to L1520T	With updated weight estimates, Rocksim predicts that a higher impulse motor is necessary to reach the target apogee.
Fin can length increased	Fin can length was increased from 22 in. to 25 in. to to accommodate the larger motor.
Separation points changed	Separation points were changed to lengthen the aft section of the vehicle, aiding in leveling the payload.
Removed motor tube	With the current fin can design a motor tube is not necessary to hold the motor in place. This fin can design is discussed in detail in Section 3.3.10. This also removes weight from the aft section of the launch vehicle improving stability.
Main Parachute changed	Main Parachute choice was changed from a 120” Rocketman Pro X to a Iris UltraCompact 120” Fruity Chutes
AV Sled Composition	Material that the Avionics Sled was to be made of was changed from ABS 3D printing plastic to PEPG 3D printing plastic

2.2 Changes Made to Payload

Table 2: Changes made to the payload since proposal submission

Change	Justification For Change
Focus on multi-camera fin orientation system, rather than clear body tube	Clear body tubes are difficult to manufacture and easily shatter and get scratched. The multi-camera fin orientation system is more fail-safe.
Four fin design selected	Cameras placed in between fins interfere less with airflow around the fins.
Methodology of RAFCO reception decided	Most reliable and easily implemented methodology of receiving and decoding RAFCO.
Vertically aligned payload sled changed to axially aligned	A vertically aligned payload sled increases ease of access and usable area and decreases required space.
Elimination of gravity assisted orientation	Gravity assistance requires many moving parts and could result in unfavorable oscillations.

2.3 Changes Made To Project Plan

Table 3: Changes made to the project plan since proposal submission.

Description of Change	Justification of Change
Team Derived Requirements have been added and are being enforced in the design.	Required per NASA Student Launch handbook

3 Vehicle Criteria

3.1 Launch Vehicle Mission Statement and Success Criteria

The launch vehicle will be constructed using conventional methods and design. Additionally, the launch vehicle will be used to house all payload electronics and support the mission of the payload. The mission and success criteria of the launch vehicle are described in this section.

3.1.1 Mission Statement

The mission of the launch vehicle is to reach the declared apogee of 4500 ft while securely housing all payload electronics and safely delivering them to the ground. The team will work together to design and construct a launch vehicle to accomplish this mission while being in compliance with all NASA and team derived requirements. The launch vehicle will be designed around safety, reliability, reusability, and fun.

3.1.2 Success Criteria

The launch vehicle will be declared successful if it accomplishes the mission stated above while also maintaining compliance to all NASA and team derived requirements. Criteria for success of the launch vehicle are defined in Table (4) below.

Table 4: Launch vehicle success criteria.

Level of Success	Criteria
Complete Success	- Launch has nominal takeoff and descent - Launch vehicle reaches $\pm$250 ft. of target apogee - Launch vehicle is recovered undamaged - Vehicle could be relaunched the same day - Payload is returned to the ground undamaged
Partial Success	- Successful launch vehicle takeoff and descent - Launch vehicle reaches $\pm$500 ft. of target apogee - Launch vehicle can be repaired at the field - Payload sustains damage but does not loose electronic functions
Inconclusive	- Successful launch vehicle takeoff and descent Launch vehicle apogee is $\pm$750 ft. of target apogee - Launch vehicle can be repaired within a day - Payload can be repaired quickly
Partial Failure	- Successful launch vehicle takeoff and unsuccessful descent - Launch vehicle apogee is below 3000 ft. or above 6000 ft. - Launch vehicle can be repaired within a week - Payload requires extended repairs
Complete Failure	- Launch vehicle is not recovered or unrepairable - Payload is unrepairable

3.2 Launch Vehicle Alternative Designs

All potential launch vehicle designs consist of 6 independent sections. From the forward end of the launch vehicle these components are the nose cone, main parachute bay, avionics bay, drogue parachute bay, payload bay, and fin can. Some of these sections will be connected such that there are only 3 independent sections during descent.

All designs will use airframe material that has an outside diameter of 6.17 in.

3.2.1 Material Selection

All materials must be selected to withstand any forces to which the launch vehicle is subjected. These forces come from the motor and from aerodynamic forces. Materials must be selected for the fins and the airframe.

3.2.1.1 Airframe Material selection

The two airframe materials under consideration are Blue Tube and G12 fiberglass. Both are common materials used in high-power rocketry. The airframe will support any load applied to the launch vehicle and must be able to support these loads with appropriate factors of safety.

3.2.1.1.1 Airframe Force Calculations

The airframe will bear most of the loads that are applied to the launch vehicle. In order to ensure that the airframe does not fail, it is necessary to consider the maximum forces the launch vehicle will experience. To begin this analysis, a free body diagram of the launch vehicle under vertical flight is presented in Fig. 1 below.

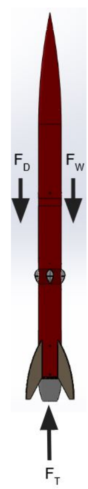

Figure 1: A free body diagram of the launch vehicle during powered flight.

The three largest forces acting on the launch vehicle are aerodynamic drag, thrust, and gravity. The thrust force is assumed to be the maximum thrust of the current motor choice. The maximum thrust of the motor will occur shortly after the vehicle has left the launch rail. Early in the flight, the vehicle should be mostly vertical. Thus, the calculations will assume zero angle of attack.

From the free body diagram the equation of the net force on the launch vehicle can be written as:

$$F_{net} = F_T - F_W - F_D \tag{1}$$

Where the net force, F_{net}, is the sum of the drag, F_D, thrust, F_T, and weight, F_W, forces. Drag force can be calculated using the following equation,

$$F_D = \frac{1}{2}\rho V^2 C_D A \tag{2}$$

Where the drag force, F_D, is a function of the air density, ρ, velocity of the launch vehicle, V, drag coefficient, C_D, and the frontal area, A, of the launch vehicle. The values for each of these variable as well as their sources are shown in Table (5) below.

Table 5: Values used in airframe force calculations and their sources.

	Value	Source
m	43.22 lb	Rocksim and hand measurements
F_T	407.8 lb	Manufacturer motor data
ρ	0.0765 lb/ft^3	Known constant
V	733 ft/s	Rocksim
C_D	0.5	Rocksim
A	0.207 ft^2	Hand calculation
g	32.2 ft/s^2	Known Constant

Evaluating Eqn. 1 using the values in Table (5) shows that the maximum net force on the launch vehicle is expected to be 663.73 lb. The airframe of the launch vehicle must be designed to withstand this force while providing an acceptable factor of safety.

3.2.1.1.2 Blue Tube

Blue tube is an airframe material manufactured by Always Ready Rocketry. Blue tube is made from vulcanized fiber, making it strong and resistant to abrasion and blunt impact. Furthermore, it is more cheap and lightweight than other alternatives such as fiberglass or carbon fiber. The density of blue tube is 0.723 $\frac{oz}{in^3}$. Blue tube is being considered over other lightweight materials such as phenolic tubing because it is also more resistant to shattering than phenolic tubing. Additionally, the cutting and sanding of blue tube does not pose as many occupational hazards as materials such as fiberglass. The biggest drawback of blue tube is that it is not water resistant. The team's home launch field has many irrigation ditches that pose hazards to launch vehicle recovery. As such, water resistance is a large concern when choosing airframe materials. Because of its spiral wound structure, blue tube can also fray and delaminate at cut ends making it harder to work with and limiting reusability. The manufacturer of Blue Tube has posted the maximum compressive strength of blue tube to be approximately 3000lb [11]. This gives blue tube adequate strength to withstand the expected forces on the airframe.

3.2.1.1.3 G12 Fiberglass

G12 fiberglass is a filament wound fiberglass material that was designed for use in large tubing and rocketry. Fiberglass is significantly stronger than blue tube and can easily withstand the expected forces on the airframe. Furthermore, fiberglass is resistant to shattering, scratching, and blunt impacts. The biggest advantage to fiberglass over blue tube is that it is water resistant. Additionally, G12 fiberglass is not as prone to fraying or delamination. These characteristics make it ideal for use in a reusable launch vehicle. The biggest downside of fiberglass is its increased weight with a density of 0.98 $\frac{oz}{in^3}$. This makes fiberglass significantly heavier than blue tube. However, preliminary mass estimates and simulations show that there are several motors that could propel the launch vehicle to within the competition requirements. As such, the increased weight of fiberglass is not of concern. A detailed description of the mass of the leading launch vehicle design can be found in Table (7) below.

3.2.1.2 Fin Material Selection

3.2.1.2.1 Plywood

Aircraft grade birch plywood is the first material choice for constructing the fins of the launch vehicle. Plywood is relatively lightweight with a density of 0.393$\frac{oz}{in^3}$. The plywood that would be used is sold in 1/8-inch-thick sheets. Because of this, two layers of plywood would be laminated together in order to form a single fin. This is the simplest method of fin construction and has been utilized by the team in past years. This method will create fins that are strong enough to endure any forces that are applied to them.

3.2.1.2.2 Composite

Sandwich composites are frequently used in aircraft and rotor blades in order to create lightweight yet strong components. The next method of fin construction under consideration is creating sandwich composite fins using a balsa wood core with 2 layers of 7.5 $\frac{oz}{yd^2}$ fiberglass laminated to either side. This will create a composite structure that is strong yet lightweight. Composite fins would be more difficult and time/cost intensive to fabricate. However, this method of fins construction has been performed by several of the team's mentors. As such the team has access to resources and knowledge that would aid in the fabrication of composite fins. The largest benefit of composite fins is their reduced weight. The density of balsa wood is only 0.0925 $\frac{oz}{in^3}$. This is much less than birch plywood. By using balsa core composite fins, the weight of each fin could be reduced by as much as 47 percent. Reducing the weight in the aft of the launch vehicle will shift the CG forward. Thus, less ballast will be required in order to obtain the minimum required stability.

3.2.2 Nose Cone

Nose cone geometry for subsonic flight creates negligible pressure drag [13]. The main factor impacting the drag of different nose cone geometries is the amount of wetted surface, which creates friction drag. The performance difference of common nose cone geometries, such as ogive, conical, and Von Karman, is a function of surface area that can be approximated as length.

3.2.2.1 4:1 Ogive Nose Cone

A 4:1, referring to the length to diameter ratio, ogive nose cone has a length of 24 in. with the launch vehicle having a diameter of 6 in. Ogive is a common nose cone geometry with mediocre aerodynamic performance in the sub-sonic regime. The nose cone is composed of the same G12 fiberglass as the airframe and will terminate with a screw on metal tip for durability and relaunch capability.

3.2.2.2 5:1 Ogive Nose Cone

A 5:1 ogive nose cone has a length of 30 in. adding 6 in. to the overall length of the launch vehicle relative to the other alternative. This added length increases the skin friction drag of the nose cone with no other increase in aerodynamic performance. The nose cone is composed of the same G12 fiberglass as the airframe and will terminate with a screw on metal tip for durability and relaunch capability.

3.2.3 Nose Cone Bulkhead

The nose cone bulkhead will be used to attach the shock cord to the nose cone. Additionally, any ballast needed to stabilize the launch vehicle will be attached to the forward face of the nose cone bulkhead using a ballast box. The two leading alternatives for construction of the nose cone bulkhead are detailed below.

3.2.3.1 Fixed

A fixed nose cone bulkhead is constructed by permanently epoxying a bulkhead to the inside of the nose cone shoulder. This creates a strong bond between the bulkhead and the airframe which is essential for resisting deployment forces. A permanent bulkhead in the nose cone is the simplest design to manufacture. The biggest drawback of a fixed bulkhead is that it cannot be removed to add ballast to the nose cone. Preliminary simulations indicate that as much as 1 lb. of ballast may need to be added to the nose cone in order to achieve the minimum required stability. For a fixed nose cone bulkhead, the amount of ballast cannot be changed after construction.

3.2.3.2 Removable

A removable nose cone bulkhead allows for the amount of ballast in the nose cone to be changed and is constructed in two parts. First, a centering ring is permanently epoxied into the front of the nose cone shoulder. 1/4-20 T-nuts are affixed to the forward face of this centering ring. Then, a removable bulkhead is bolted to the centering ring using four 1/4-20 steel bolts. This allows for the bulkhead to be removed as necessary. This design maintains the strength of an epoxy bond while allowing for a removable bulkhead. However, this design is more difficult and time intensive to manufacture.

3.2.4 Fin Configuration

Fin configuration refers to the number of fins on the launch vehicle. Changing the number of fins affects the size of the fins as well as the location of the center of pressure and stability. The number of fins also impacts the amount of weight on the fin can.

3.2.4.1 Three-Fin Configuration

A three-fin configuration is a popular choice in high-power rocketry. A three-fin configuration would allow for a stable launch vehicle. With only three fins there are fewer individual components of the launch vehicle to manufacture. One drawback of a three-fin configuration is that each individual fin must have a greater surface area to stabilize the launch vehicle. Additionally, a three-fin configuration is not axially symmetric. In the event of a crosswind, each fin could experience different aerodynamic loading due to the difference in angle of attack.

3.2.4.2 Four-Fin Configuration

A four-fin configuration is viable to stabilize the launch vehicle. Using four fins allows for the surface area of each fin to be smaller than that of a three-fin design. A four-fin design would allow for camera housings to be evenly spaced in between the fins, aiding in the payload design. Additionally, four fins move the center of pressure further aft on the launch vehicle. This is beneficial in order to increase the stability of the launch vehicle.

3.2.5 Fin Design

Fins are instrumental in the stability of the launch vehicle by shifting the point that the pressure force of the vehicle acts, CP, to a more stable location. Fins also increase the cross sectional area subjected to the free stream airflow that create substantial drag on the launch vehicle. Both basic and more advanced fin geometries are considered for the launch vehicle that satisfy the basic function of shifting the CP to the desired location.

3.2.5.1 Trapezoidal Fin

Basic fin designs commonly use trapezoidal profiles with TEs perpendicular to the free stream airflow. This method is simple to modify during the design process to meet the required CP location and influence stability. Trapezoidal fins are also less labor intensive to construct out of common materials such as plywood or fiberglass, utilizing basic power tools. The drawback of this geometry is a lack of aerodynamic optimization to reduce the drag produced by the fins. Fig. 2 displays a three dimensional turbulent energy simulation of a basic fin geometry experiencing airflow at the average speed the launch vehicle will experience.

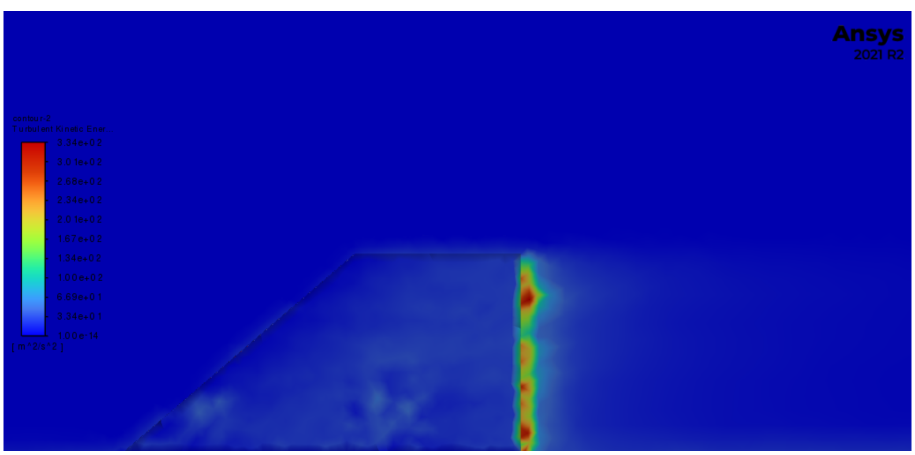

Figure 2: A 3D trapezoidal fin geometry turbulence study in ANSYS.

Looking at the TE of the basic fin in Fig. 2, the dark red is where the air flow devolves into disorganized directional change due to inefficiencies in the fin design. The turbulent wake of the fin creates a change in the flow velocity demonstrated in Fig. 3. Due to the conservation of momentum the reduction in airflow speed is experienced by the launch vehicle as drag.

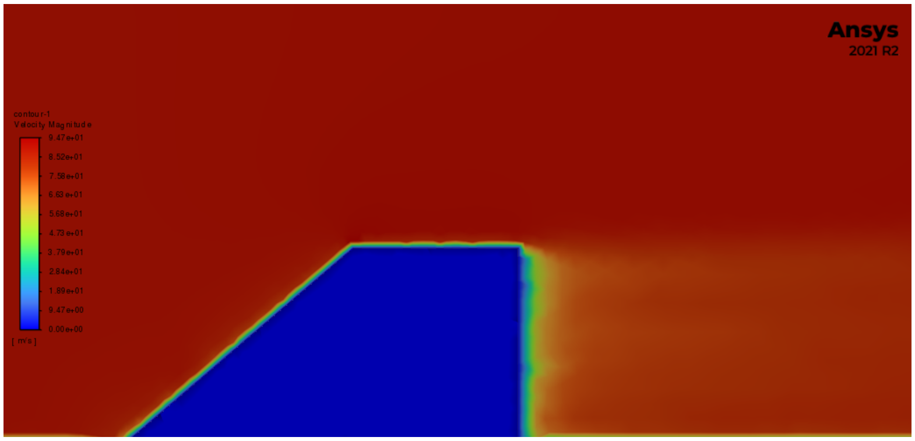

Figure 3: A 3D trapezoidal fin geometry velocity study in ANSYS.

3.2.5.2 Ogive Fin

The following design utilizes a geometry similar to a nose cone to streamline the LE and sweep past the root cord of the fin. The exact dimensions of this design are detailed in Section 3.3.12.

3.2.5.2.1 Geometry

To form the LE of the ogive fin, a circle is created with the diameter being a function of the desired span and chord of the fin. Bisecting the circle with the span and chord creates the desired ogive shape demonstrated below in Fig. 4[19].

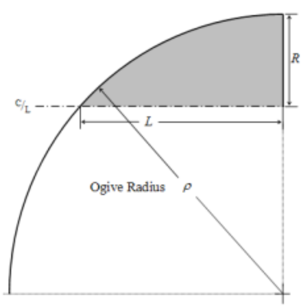

Figure 4: Ogive nose cone radius diagram.

Ogive nose cones commonly refer to the ratio of length to diameter, for instance a 5:1 ogive nose cone, which is utilized on this vehicle. The TE of the fin sweeps back at a significant angle until reaching a flat TE perpendicular to the air stream.

3.2.5.2.2 Aerodynamic Improvements

All edges of the fin that influence airflow will have a significant non-uniform bevel that rounds the edge while slowly transitioning into the cord. By sweeping the TE back past the root cord, the CP of the fin and therefore vehicle are moved farther aft than a trapezoidal geometry of the same surface area. This design has a increases in efficiency due to its lower cross sectional area and swept profile for the same CP influence.

By rounding the profile and profile edges, zones of high turbulence are reduced as shown in Fig. (5), which decreases the non-skin friction drag of the fin.

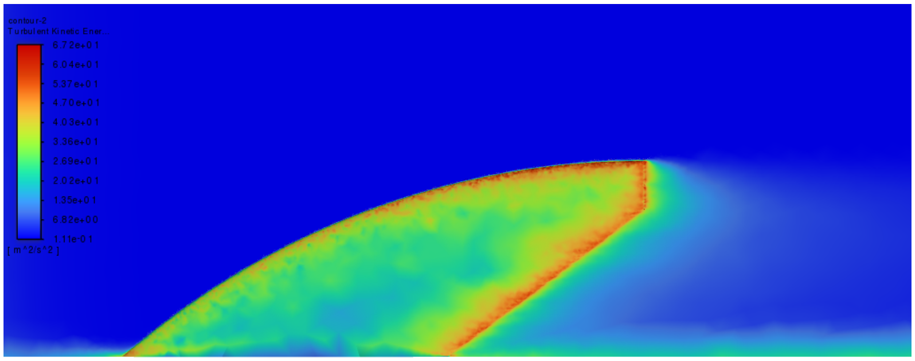

Figure 5: An 3D ogive fin geometry turbulence study in ANSYS.

The LE of this design also has a decrease in turbulent flow resulting in almost no LE decrease in flow velocity shown in Fig. 6. Fin wake velocity decrease is also reduced and with more of a velocity decrease on the tip perpendicular TE than the swept section.

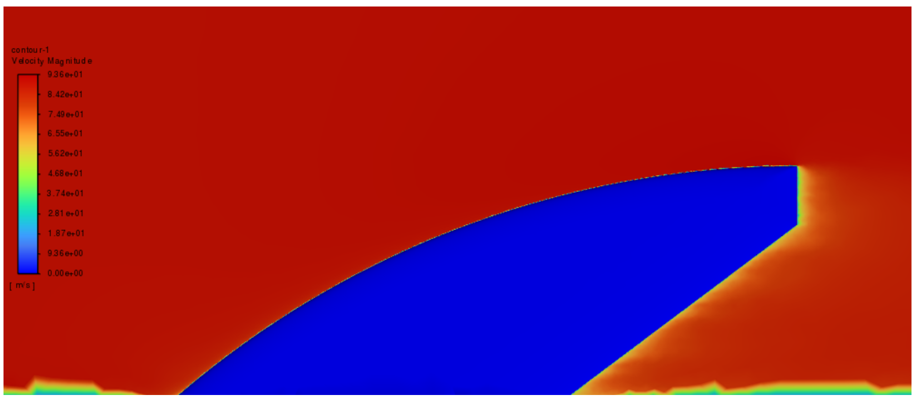

Figure 6: An 3D ogive fin geometry velocity study in ANSYS.

The ANSYS simulation performed demonstrate the ogive fin geometries flow interaction with a obvious visual advantage over a basic fin.

3.2.6 Tail Cone Design

The purpose of the tail cone is to reduce the turbulent wake of the launch vehicle and thus decrease drag. The current design of the launch vehicle uses a tail cone for this purpose. The tail cone will be manufactured out of 3D printed PETG plastic. Alternative shape designs for the tail cone are detailed below.

3.2.6.1 Ogive

The first tail cone design for is an ogive shape, similar to the shape used on many nose cones. This shape is designed to reduce drag but may be difficult to manufacture.

3.2.6.2 Conical

The next design would utilize a conical shaped tail cone. This design would provide many of the same aerodynamic benefits as the ogive tail cone with only a slight loss in performance. The advantage of a conical design is that it would be easier to manufacture.

3.2.7 Fin Can Designs

The fin can will secure the motor into the launch vehicle as well as provide and attachment point for the fins. Because of the motor and fins, the fin can is under high stresses. The two leading alternatives for fin installation are detailed below.

3.2.7.1 Fixed Fin Design

The first method of fin can construction is a fixed fin design. In this design, there are three centering rings inside of the fin can. Each centering ring is permanently epoxied into the airframe. The motor tube runs through the center of all three centering rings. A tab on the fins is then slotted through the airframe and epoxied to both the motor tube and the airframe. The two aft most centering rings sandwich the tab on the fins to hold them securely in place. Finally, epoxy fillets are added to the external surface between the fins and airframe.

This design is commonly used in high-power rocketry. The permanent connection creates a strong bond between all components. However, this design transfers all of the force of the motor directly into the epoxy joint between the centering rings and the airframe. Although the epoxy forms a strong bond with an ultimate stress of 7200 psi, this represents a weak point in the design. These epoxy joints could be subject to failure especially under the higher temperatures of the fin can and under repeated loading.

Furthermore, it is difficult to replace components with the fixed fin design. If a fin were to break during flight or upon landing, replacing it would require the entire fin can to be cut apart. This limits launch vehicle's reusability.

3.2.7.2 Removable Fin Design

The next method of fin can construction allows for removable fins and centering rings. Once again, this design has three internal centering rings. Unlike the fixed fin design, however, only the forward most centering ring is epoxied into place. The two aft most centering rings are held into the launch vehicle using four #8-32 machine screws per centering ring. The aft two centering rings are then connected by plywood runners that slot into the centering rings. These runners are held onto the centering rings via epoxy. Fins are then slotted into the fin can and bolted to the runners. This entire assembly (aft two centering rings, runners, and fins) can then be removed from the launch vehicle in the event that a fin breaks. Threaded rods are also run between the aft two centering rings for structural stability and support. This greatly improves the reusability of the launch vehicle. If any one part of the fin can were to break, it can be easily removed and replaced. Additionally, this design allows for a thrust plate to be secured to the aft end of the fin can. This distributes the load of the motor directly onto the airframe instead of relying on epoxy joints. The main drawback of this design is that there are more parts to fabricate and assemble. Additionally, it requires tighter tolerances during manufacturing to ensure that all components fit properly.

3.2.8 Rail Button Attachment

Rail buttons are used to attach the launch vehicle to the launch rail. The rail buttons ensure that the vehicle remains in vertical flight until it has gained enough speed to be stable on its own. The largest challenge in rail button attachment is ensuring that the rail buttons are far enough away from the launch vehicle to ensure they do not interfere with the external camera housings. Currently, there are two leading designs for the attachment of rail buttons.

3.2.8.1 Extended Rail Buttons

The first method of rail button attachment is to bolt it into the airframe of the rocket. A 3D printed rail button extender would be used in order to hold the rail button above the top of the camera housing. The rail button extender would the be permanently epoxied onto the airframe of the rocket. An example of a rail button extender is shown in Fig. (7).

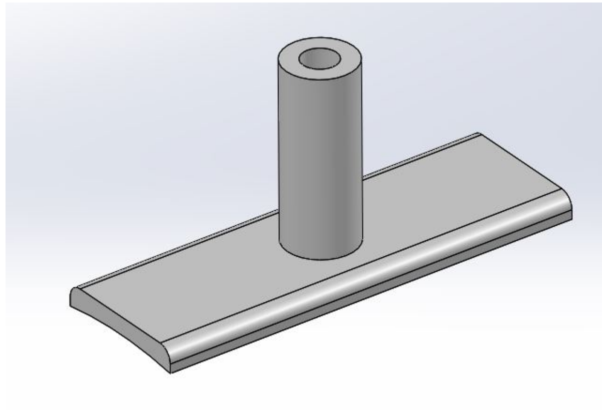

Figure 7: An example of a rail button extender. The rail button would be bolted to this assembly.

3.2.8.2 Fly-Away Rail Guides

Fly away rail guides are an option that allow for rail buttons to be used in guiding the launch vehicle without the need for permanently attaching rail buttons. This system is composed of two spring-loaded halves which clamp around the launch vehicle. While on the launch rail, a pair of rail buttons hold the assembly closed. As soon as the rail buttons clear the launch rail, the guides spring open and fall away from the launch vehicle. An image of this system is shown in Fig. (8) below.

Figure 8: An example of fly-away rail guides.

This system is an effective way to secure smaller rockets to the rail. However, for larger diameter launch vehicles it is likely that the team would have to manufacture a custom system.

3.2.9 Alternative Separation Points

Separation points are sections of the launch vehicle that physically separate during deployment of the main and drogue parachutes. Changing the separation points changes both the configuration of the parachutes and shock cord as well as the mass of each independent section. As such, separation point alternatives are considered for recovery purposes.

3.2.9.1 Nose Cone and AV-Drogue

The first option for separation points will separate the launch vehicle between the nose cone and main parachute bay as well as between the AV bay and the drogue parachute bay. This configuration is shown in Fig. 9 below.

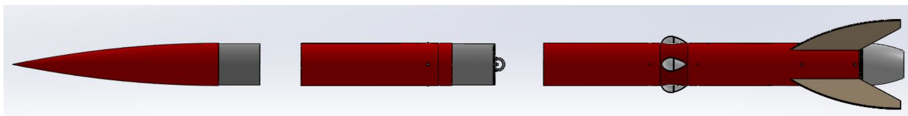

Figure 9: First configuration of separation points that separated between the nose cone and main parachute bay and between the AV and drogue parachute bay.

In this configuration, the longest and heaviest section of the launch vehicle is the aft section. This section is comprised of the drogue parachute bay, payload bay, and the fin can. The increased length of the aft section, however, will cause the launch vehicle to land at a shallower angle. This is helpful to the payload as less corrective action is required to take a level picture.

3.2.9.2 Main and Drogue-Payload

The next option for separation points is to separate the launch vehicle between the main parachute bay and the AV bay as well as between the drogue parachute bay and the payload bay. This more evenly distributes the weight of the launch vehicle if it is necessary to meet kinetic energy and descent rate requirements. This configuration is shown in Fig. 10 below.

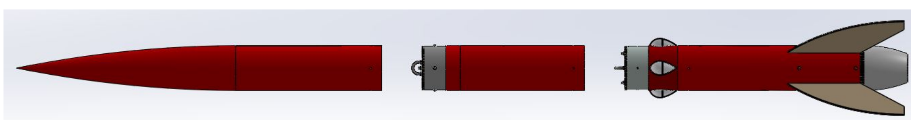

Figure 10: Second configuration of separation points that separated between the main parachute bay and the AV bay and between drogue parachute bay and the payload bay.

3.2.10 Motor Alternatives

Three motors are under consideration for use in the launch vehicle: Aerotech L1390G, Aerotech L1420R, and the Aerotech L1520T. All three motors are capable of propelling the launch vehicle to within the acceptable range of competition altitudes. Only Aerotech motors are being considered for use in the launch vehicle. This is due to their known reliability and legacy use. The specification of each motor under consideration is shown in Table (6) below.

Table 6: Leading motor alternatives.

Motor	Total Impulse (N-s)	Initial Thrust (N)	Maximum Thrust (N)	Average Thrust (N)	Burn Time (s)	Length (mm)
L1390G [1]	3949.0	1416.5	1675.0	1390	2.6	530
L1420R [2]	4603.0	1458.3	1814.0	1490	3.2	665
L1520T [3]	3715.9	1545.4	1765.3	1520	2.4	518

The L1390-G contains 3,949 N. of thrust with a slightly rounded thrust curve shown in Fig. 11. The Mojave Green fuel this motor uses has the potential for a non-uniform startup sequence due to the high activation energy of the chemicals used to produce the green coloring. Additionally, green propellant grains tend to be hygroscopic which could make them more difficult to ignite.

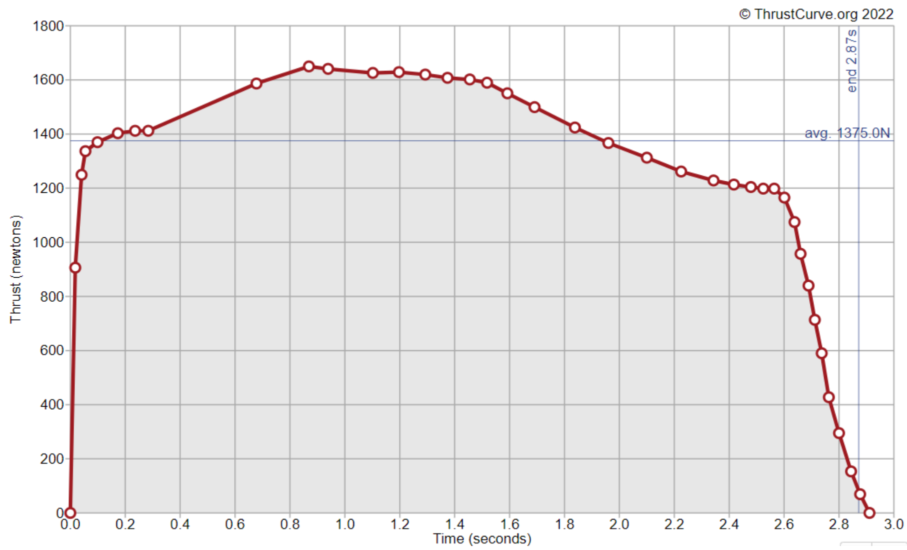

Figure 11: Thrust curve for a L1390-G motor [1]

With a total impulse of 4,603 N. the L1420R has the highest thrust of the motors considered. The Redline fuel mixture is a potential downside due to reports of unreliability. Additionally, the Redline propellant tends to have the same ignition issues as Mojave Green due to high activation energy and being hygroscopic. The thrust curve of the motor shown in Fig. (12) demonstrates a flat operational thrust with almost 3.2 seconds of burn time.

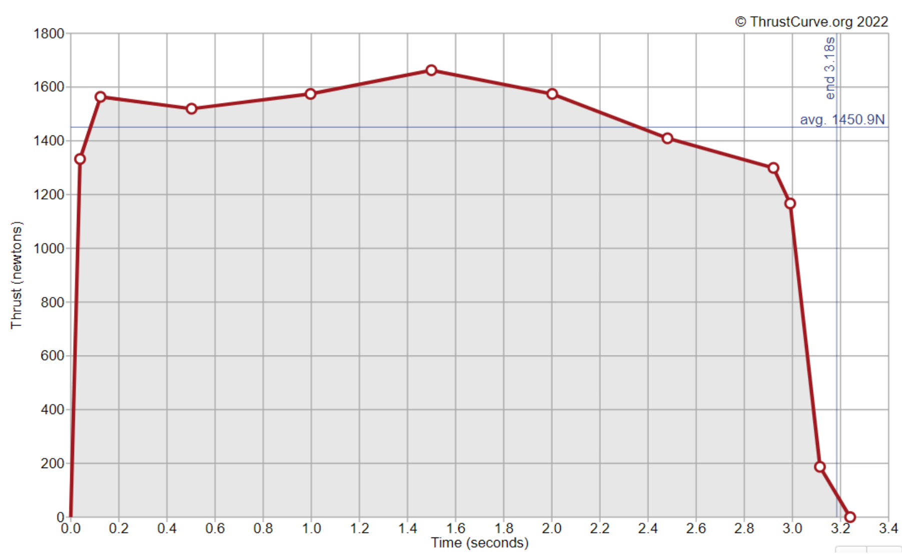

Figure 12: Thrust curve for an L1420-R motor [2]

The L1520T has the highest average thrust of the potential motors at 1,567 N. In Fig. (13) the motor's high flat burn is demonstrated with the motor producing high amounts of thrust shortly after ignition.

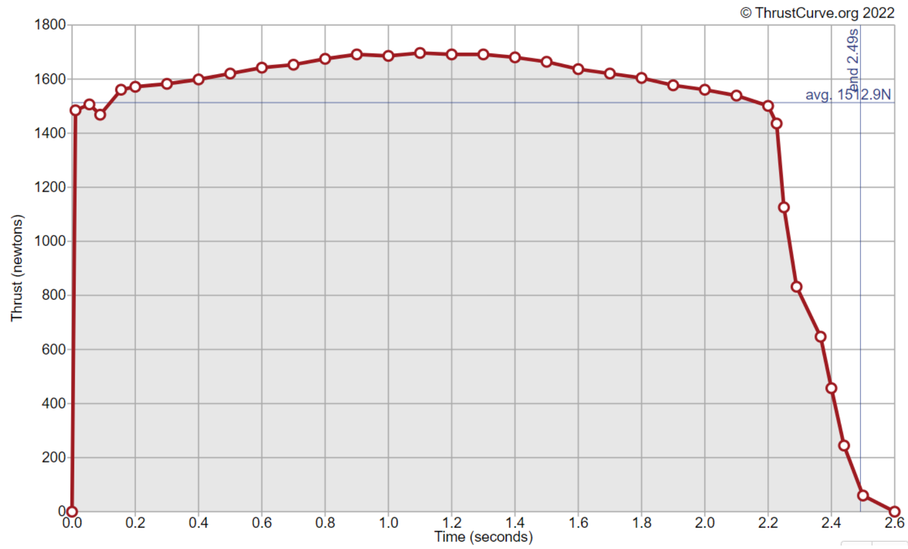

Figure 13: Thrust curve for an L1520T motor [3]

3.3 Leading Launch Vehicle Design

3.3.1 Launch Vehicle Sections and Layout

The current design of the launch vehicle is comprised of 7 sections: Nose cone, main parachute bay, avionics bay, drogue parachute bay, payload bay, and fin can. With an expected weight of 43.3 lb. concentrated at a CG of 61.2 in. from the tip of nose cone. With a four fin configuration the CP of the launch vehicle is at 74.9 in. from the tip of the nose cone. A stability margin of 2.15 is expected. A diagram of the rocket and its components are shown in Fig. (14).

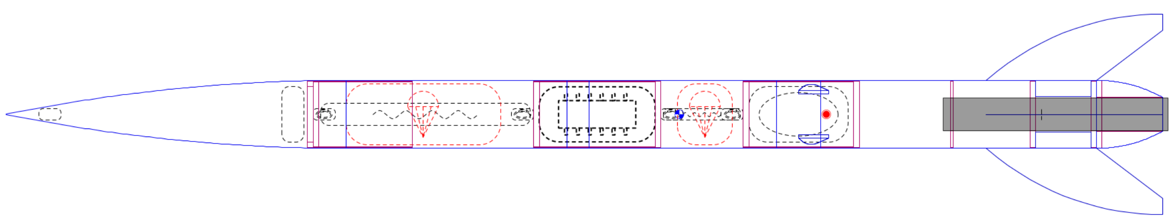

Figure 14: Diagram of the leading launch vehicle design.

The main parachute bay will be bolted to the AV bay using #8-32 machine screws. Likewise, the drogue parachute bay, payload bay, and fin can will all be bolted together. This is done so that the launch vehicle is only separated in 3 independent sections upon descent. The overall length of the launch vehicle is 104.5 in. with a diameter of is 6.17 inches. Overall dimensions of the launch vehicle are shown in Fig. 15 below.

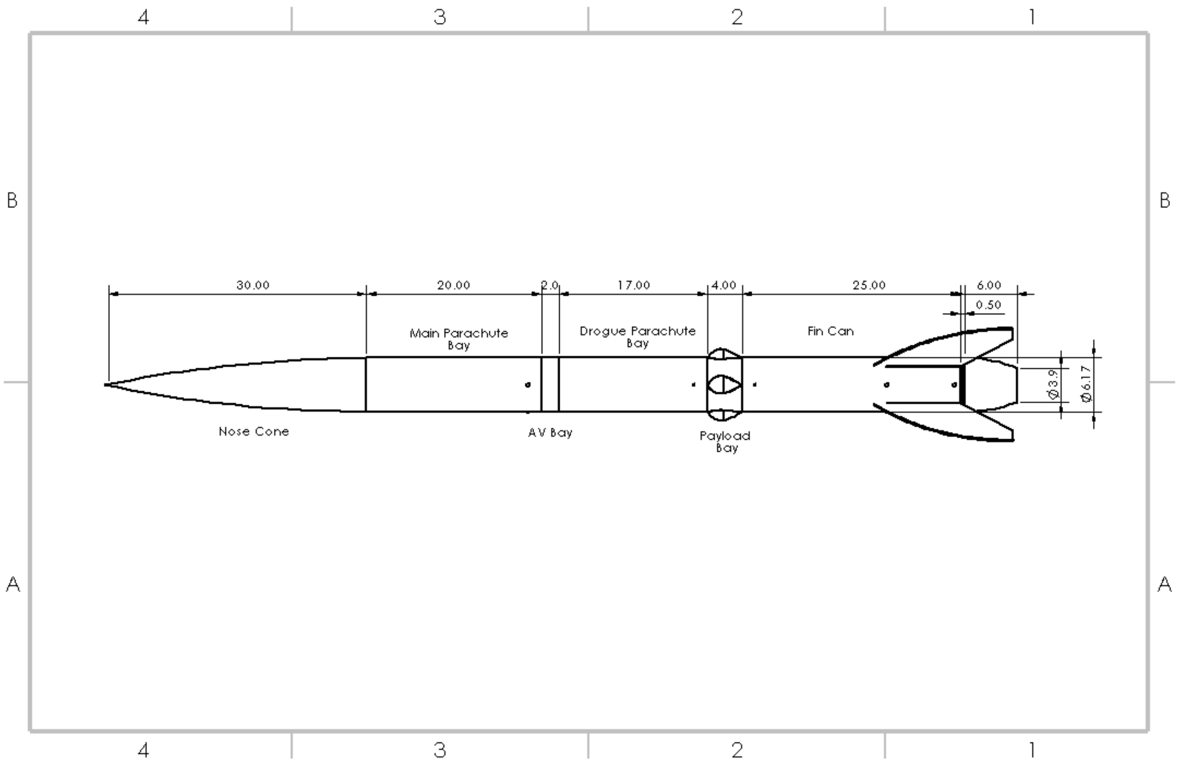

Figure 15: Overall dimensions of the launch vehicle. The overall length is 104.5 in. and the diameter of all sections is 6.17 in.

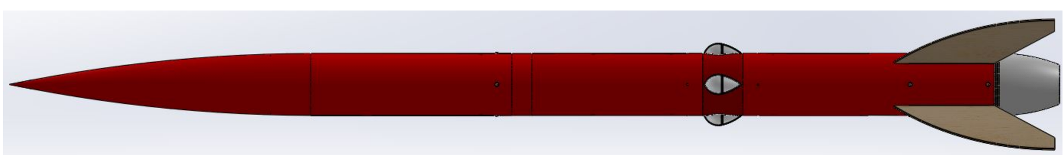

Figure 16: CAD model of the entire assembled launch vehicle.

3.3.2 Separation Points

The current launch vehicle design will separate in two places. The first separation point is between the nose cone and the main parachute bay. The seconds separation point is between the AV bay and the drogue parachute bay. In this design, energetics will be located on either side of the AV bay. This configuration is shown in Fig. 17 below. All separation points will be held together during flight using four 4-40 nylon shear pins. These pins will then break under the force created by the ejection charges allowing the sections to separate.

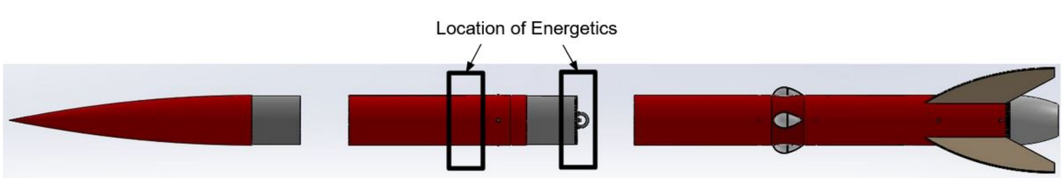

Figure 17: Second configuration of separation points that separated between the main parachute bay and the AV bay and between drogue parachute bay and the payload bay.

3.3.3 Airframe Material Selection

G12 Fiberglass is the current leading design choice for the airframe material. This was chosen because of its high strength and water resistance. The increased strength of fiberglass is necessary in order to withstand the predicted forces. Furthermore, the waterproof nature of fiberglass aids in reusability as previously discussed. Preliminary mass estimates show that a fiberglass launch vehicle will weight approximately 43.2 lbs. A detailed mass breakdown of the launch vehicle is shown in Table 8 below. Additionally, simulations show that even at this weight the launch vehicle will reach a height of 4500 ft. which is well within the range of acceptable altitudes for the competition.

3.3.4 Nose Cone

The nose cone used will be a 5:1 tangent ogive and constructed out of fiberglass with an anodized aluminum tip. The increased length of the nose cone allows for the ballast to be placed farther forward than the 4:1 alternative, decreasing the overall weight of the ballast with the same influence on the center of gravity. A 5:1 ogive for the vehicle geometry is also commonly stocked in most online stores. Unique and optimized nose cone designs are expensive and not readily available. The total length of the nose cone will be 30 in. A nose cone shoulder will be made from a 10 in. long section of coupler. The shoulder will sit 4 in. into the nose cone leaving an exposed coupler length of 6 in. on the outside of the nose cone. A dimensioned drawing of the nose cone is shown in Fig. (18) below.

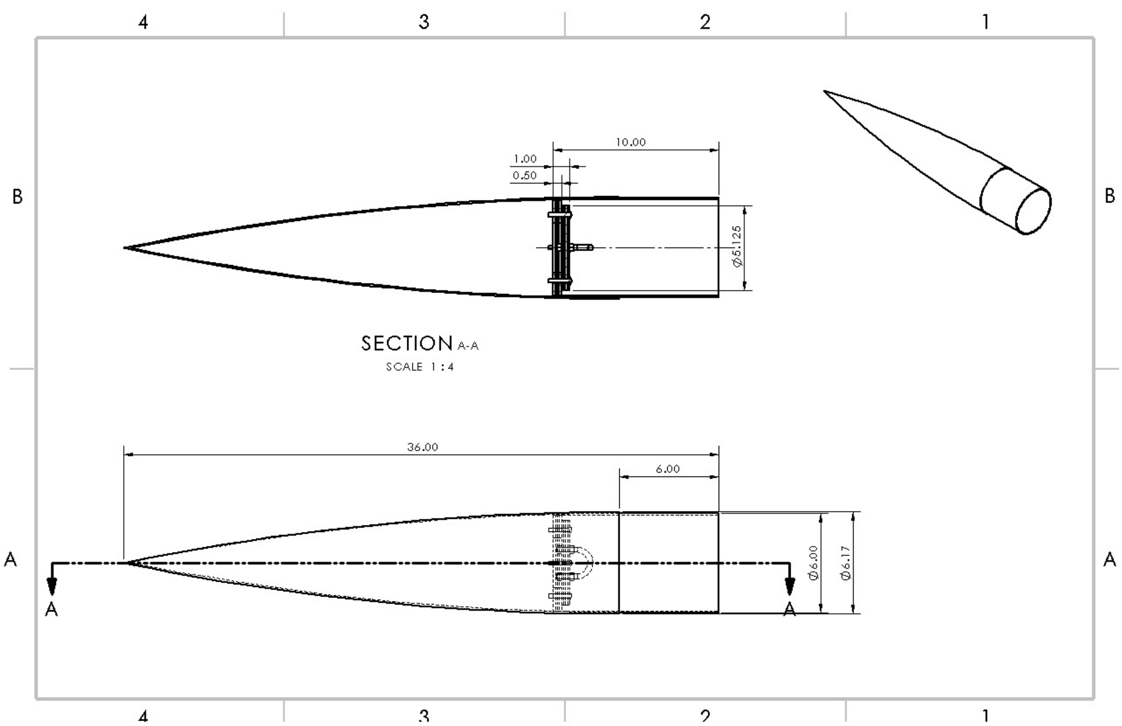

Figure 18: CAD model of the nose cone with the removable bulkhead assembly and dimensions. The nose cone is a 5:1 ogive shape and constructed from G12 fiberglass.

3.3.5 Nose Cone Bulkhead

The nose cone bulkhead assembly will comprise of 2 parts: A permanent centering ring and a removable bulkhead. Diagrams of this assembly are shown in Fig. 19 below. The centering ring will be 1/2 in. thickness and have a 3 in. hole in the center. Four 1/4-20 T-nuts will be secured to the forward face of the centering ring. The centering ring will then be epoxied into the forward end of the nose cone shoulder.

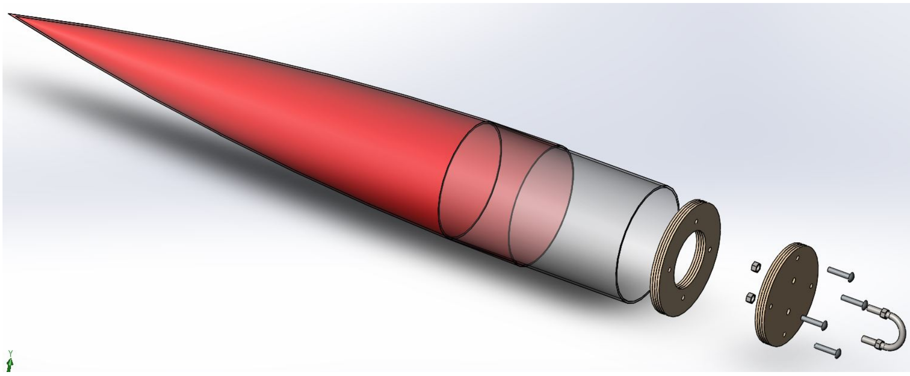

Figure 19: Exploded view model of the nose cone bulkhead/centering ring assembly inserted into the nose cone.

The nose cone bulkhead will have a diameter of 5.125 and will have a U-bolt mounted to the center for shock cord attachment. The nose cone bulkhead will be 1/2 in. thick and will be secured to the centering ring in the nose cone shoulder using four 1/4-20 bolts. A u-bolt will be secured to the center of the bulkhead for attaching shock cord.

This design allows for the nose cone bulkhead to be removed in the event that ballast must be added to the nose cone. Any ballast added to the nose cone will be secured to the forward side of the nose cone bulkhead using a 3D printed ballast box.

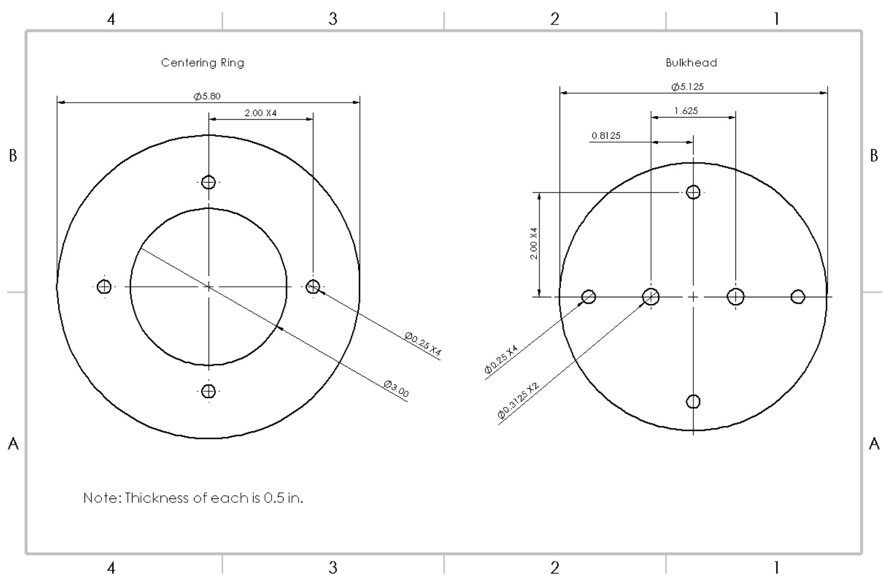

Figure 20: Dimensioned drawing of the nose cone centering ring (left) and the nose cone bulkhead (right). Each component is 3/4 in. thick.

3.3.6 Main Parachute Bay

The main parachute bay will be constructed out of a single section of fiberglass airframe that is 20 in. long. The main parachute bay is located between the nose cone and the avionics bay. It will be secured to the AV bay using four #8-32 machine screws. The main parachute bay will house the main parachute as well as its shock cords.

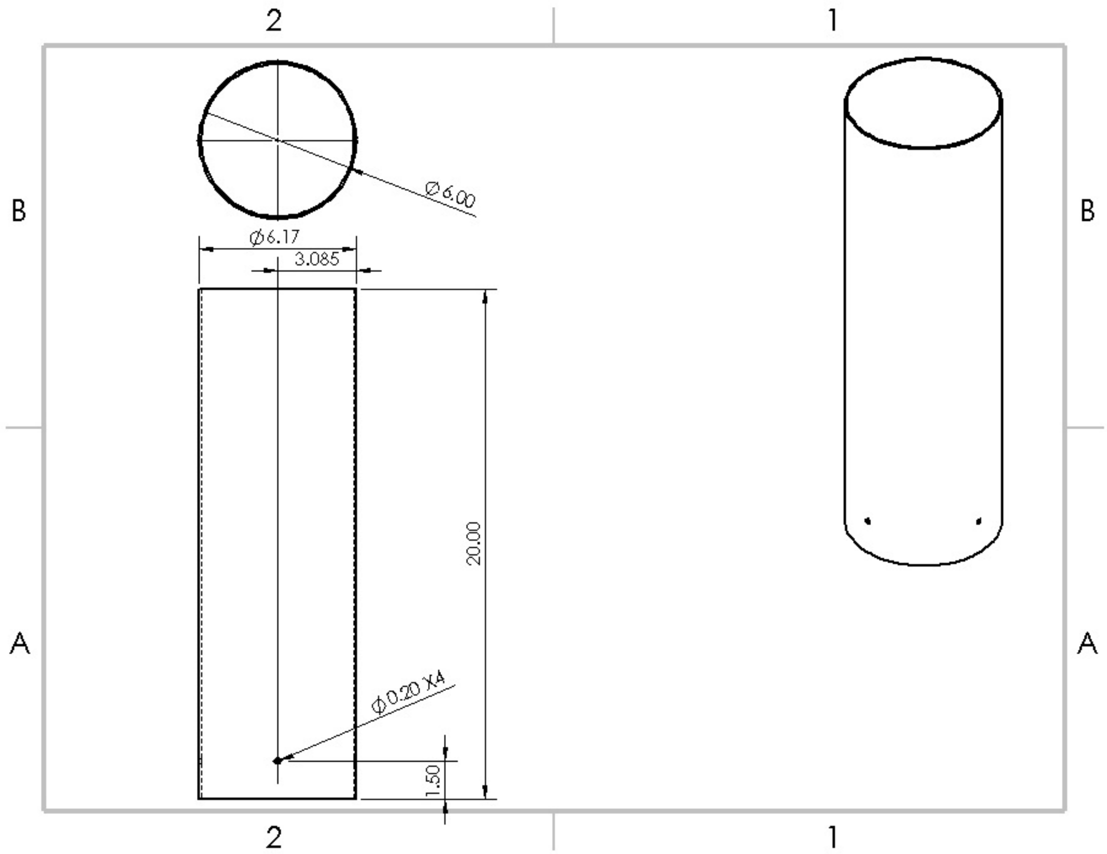

Figure 21: Dimensioned drawing of the main parachute bay.
The section will be constructed out of a 20in. long section of G12 fiberglass.

3.3.7 Avionics Bay

The avionics bay houses all recovery electronics and is positioned between the main and drogue parachute bays. It will be constructed out of an 11 in. long section of coupler and a 2 in. long band of airframe material. The airframe band will be epoxied to the coupler 3 in. from the forward end. Dimensions of the AV Bay are shown in Fig. 22 below. This airframe band will have several holes drilled into it for altimeter pressure ports as well as access to altimeter arming switches.

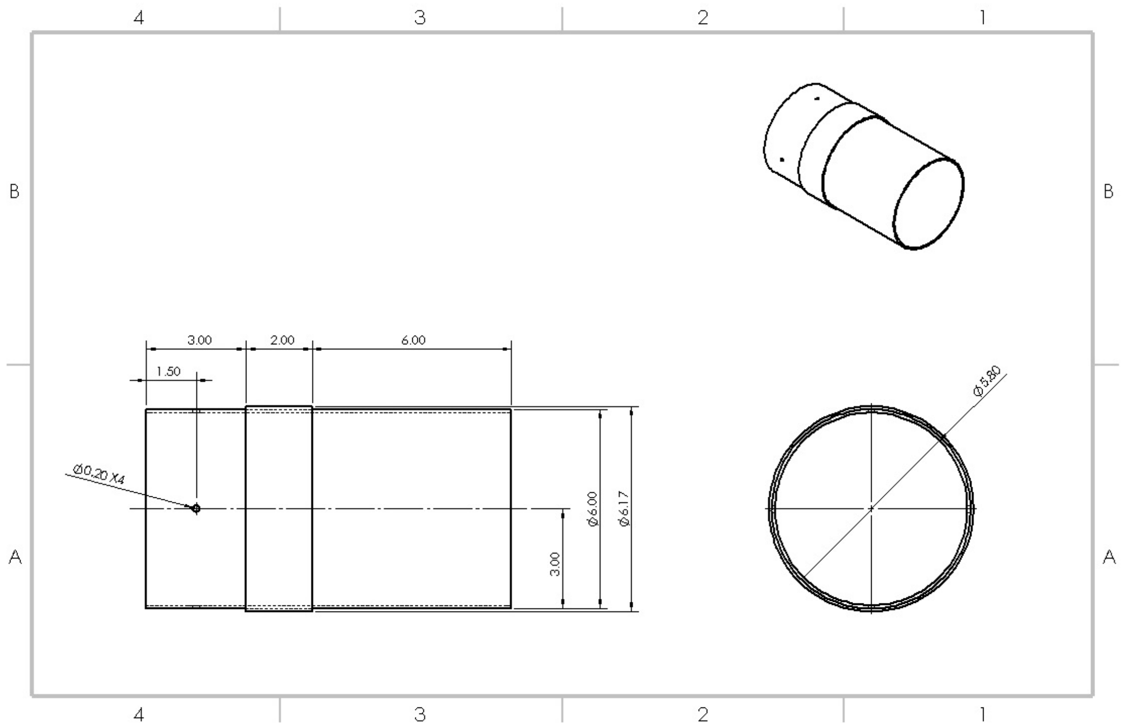

Figure 22: Dimensioned drawing of the avionics bay.

There will be a bulkhead on either side of the avionics bay. These bulkheads will be connected using two 5/16 in. threaded rods that run the entire length of the AV bay. These threaded rods will also be used in order to hold an electronics sled carrying altimeters and recovery electronics. Each Bulkhead will have a U-bolt for attaching shock cords. The total thickness of the AV bay bulkheads are expected to be 1/2 in. thick in order to withstand the expected deployment forces with an acceptable factor of safety. This thickness will be verified using FEA and destructive structural testing. Dimensioned drawings of the AV bay bulkheads are shown in Fig. 23 below.

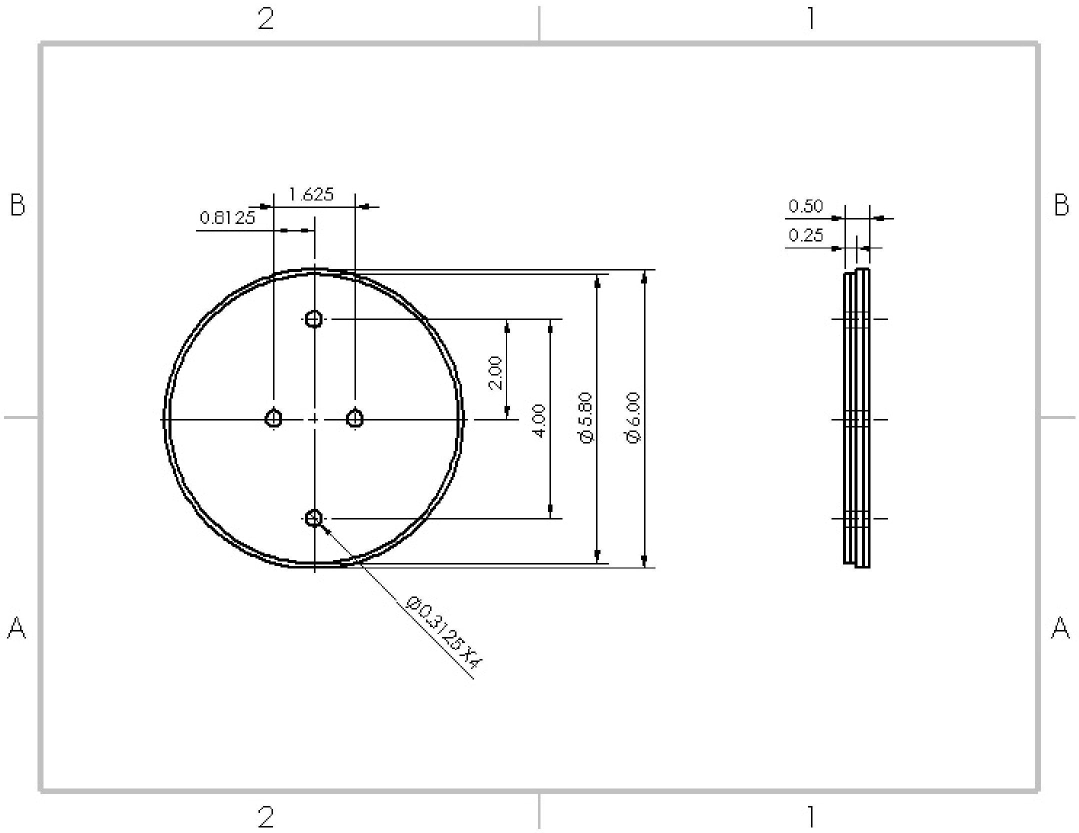

Figure 23: Dimensioned drawing of the avionics bay bulkheads.

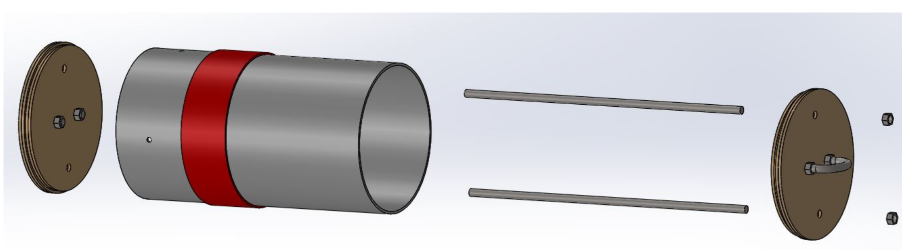

Figure 24: Exploded view of the avionics bay.

3.3.8 Drogue Parachute Bay

The drogue parachute bay is located in between the AV bay and the payload bay. It is constructed out of a single 17 in. section of airframe. The drogue parachute bay will hold the drogue parachute as well as its chock cord and will be connected to the payload bay using four #8-32 machine screws.

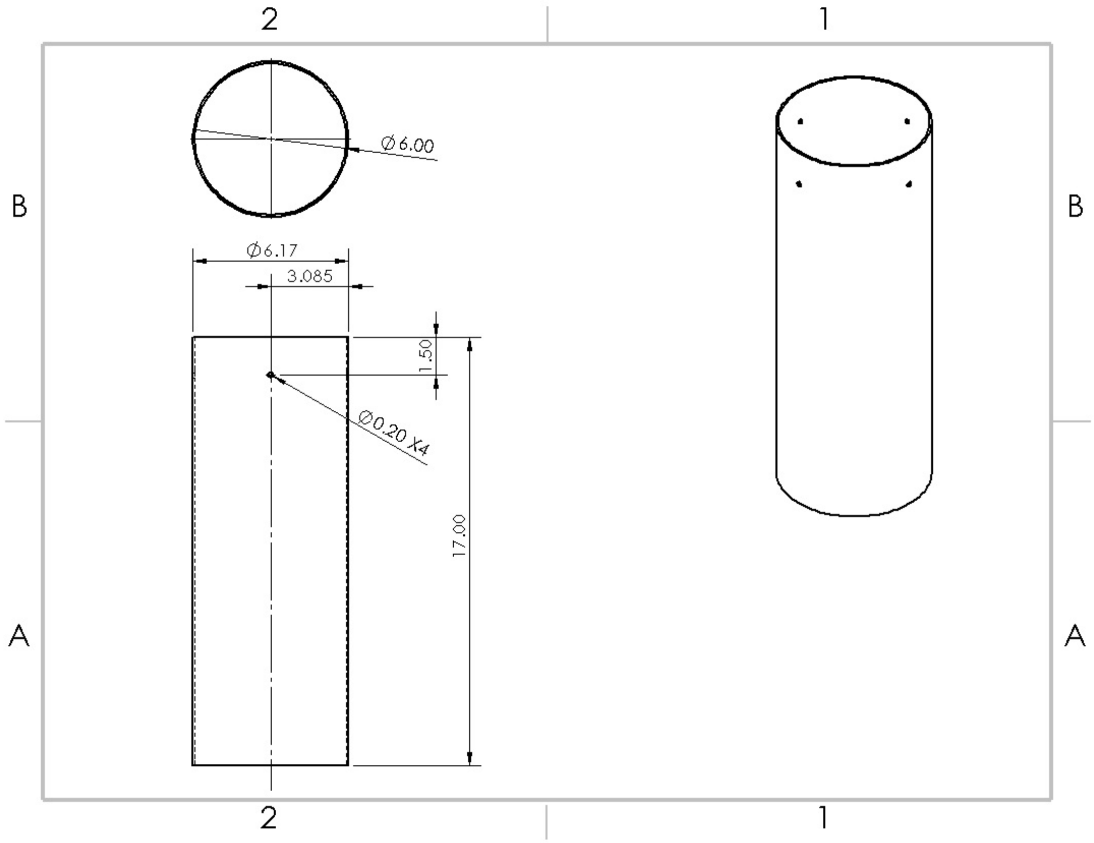

Figure 25: Dimensioned drawing of the drogue parachute bay.
The section will be constructed out of a 17in. long section of G12 fiberglass.

3.3.9 Payload Bay

The payload bay will sit between the drogue parachute bay and the fin can. It will be secured to both the parachute bay and the fin can using four #8-32 machine screws at each connection point. The payload bay will house all of the payload electronics.

The payload bay will be made from a 10 in. long coupler section and a 4 in. long band of airframe. This band of airframe will be epoxied to the center of the coupler section.

Four teardrop shaped holes will be cut into the airframe band of the payload bay and will be evenly spaced radially around the launch vehicle. These holes will allow camera's to extend out of the payload bay. The cameras will be covered with clear camera housings to protect the cameras and minimize the aerodynamic impact. More detailed descriptions of the layout of the payload bay can be found in Section 4.5 below. A dimensioned drawing of the payload bay can be seen in Fig. 26 below.

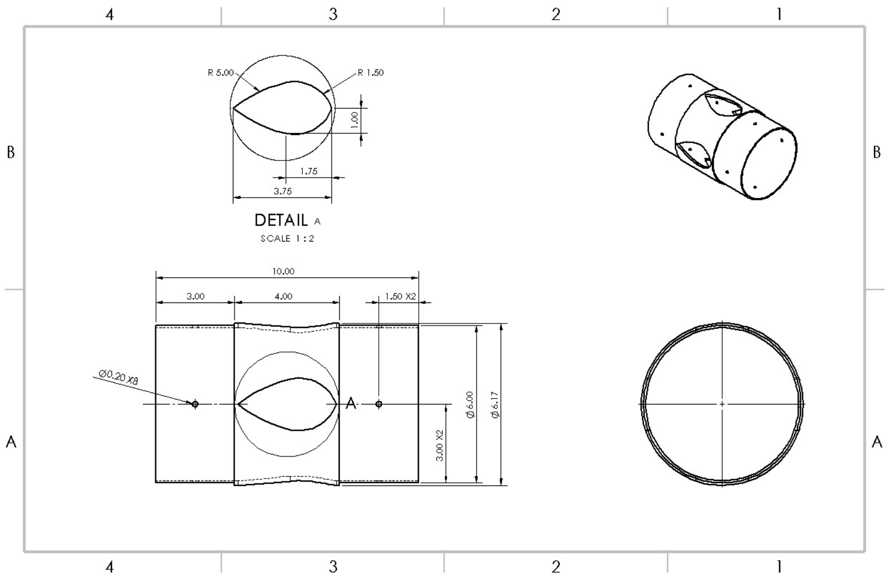

Figure 26: Dimensioned drawing of the payload bay. The bay will be constructed out of a 4 in. long airframe section that is epoxied to the center of a 10 in. long coupler section. Four teardrop shaped holes are cut to allow for camera housings.

The orientation of the payload bay will be such that the camera housings are located in between the fins. This is done to minimize the aerodynamic effect that the camera housings will have on airflow over the fins.

There will be a bulkhead on either side of the payload bay. These bullheads will be held together using a 5/16 in. threaded rod. These threaded rods will also be used to hold a sled carrying payload electronics. The forward bulkhead will also have a U-bolt for shock cords to attach to. The bulkheads are estimated to be 1/2 in. thick in order to withstand all deployment forces.

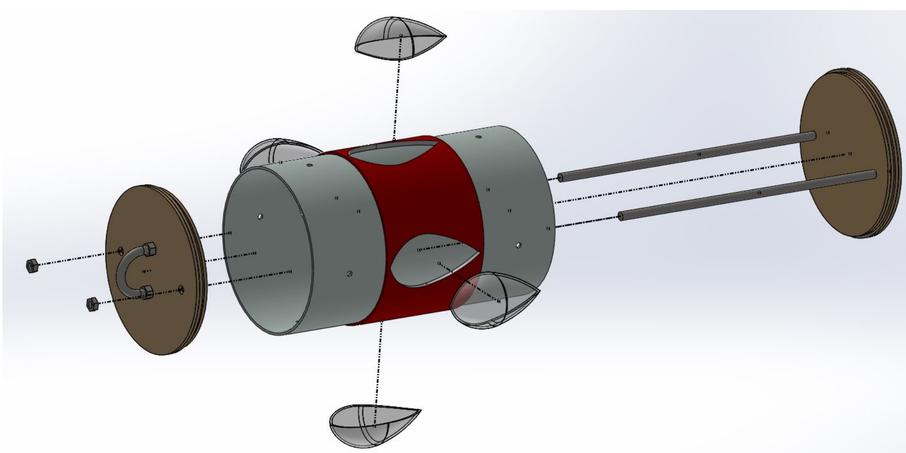

Figure 27: Exploded view of the payload bay showing four housings that will be bolted to the airframe. The bulkheads are joined by a threaded rod that runs the entire length of the bay.

3.3.10 Fin Can

The fin can section of the launch vehicle will be composed of several sub-assemblies: The airframe, removable fin assembly, and the thrust plate. When put together, the fin can section will hold the motor as well as the fins and connect them to the rest of the launch vehicle. An assembled isometric view of the fin can is shown in Fig. 28 below. This view shows how the removable fin assembly will fit inside of the airframe.

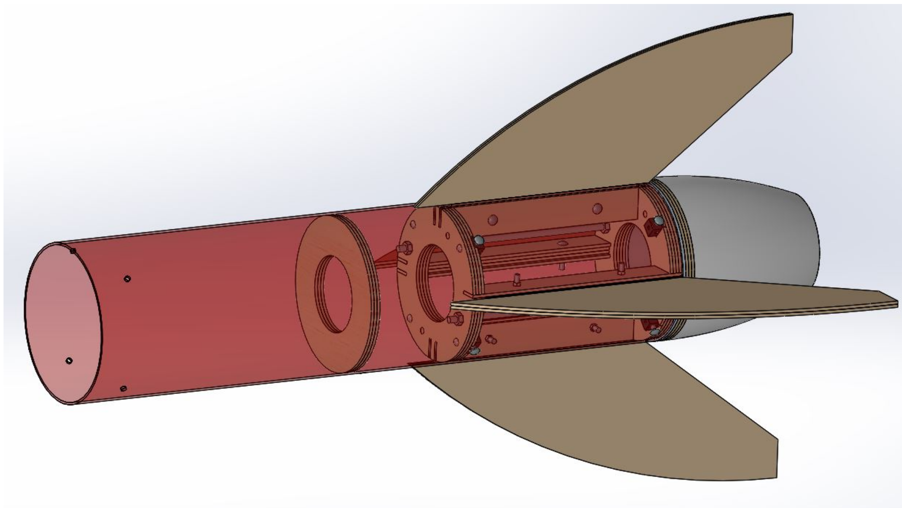

Figure 28: Assembled CAD model of the fin can.

3.3.10.1 Design and Construction

The airframe of the fin can will be a 25 in. long section of G12 fiberglass tube. The aft end of the fin can will have four slots 1/4 x 8.5 in. slots cut into the end. These slots allow for the removable fin assembly to be slid into the airframe. Additionally, there will be 8 holes drilled in between the slots used to secure the removable fin assembly to the airframe using bolts. A centering ring will be permanently epoxied near the forward end of the airframe section. The purpose of this centering ring is to help hold the motor axially aligned in the airframe section. This centering ring is not expected to be load bearing. A dimensioned drawing of the fin can is shown in Fig. 29 below.

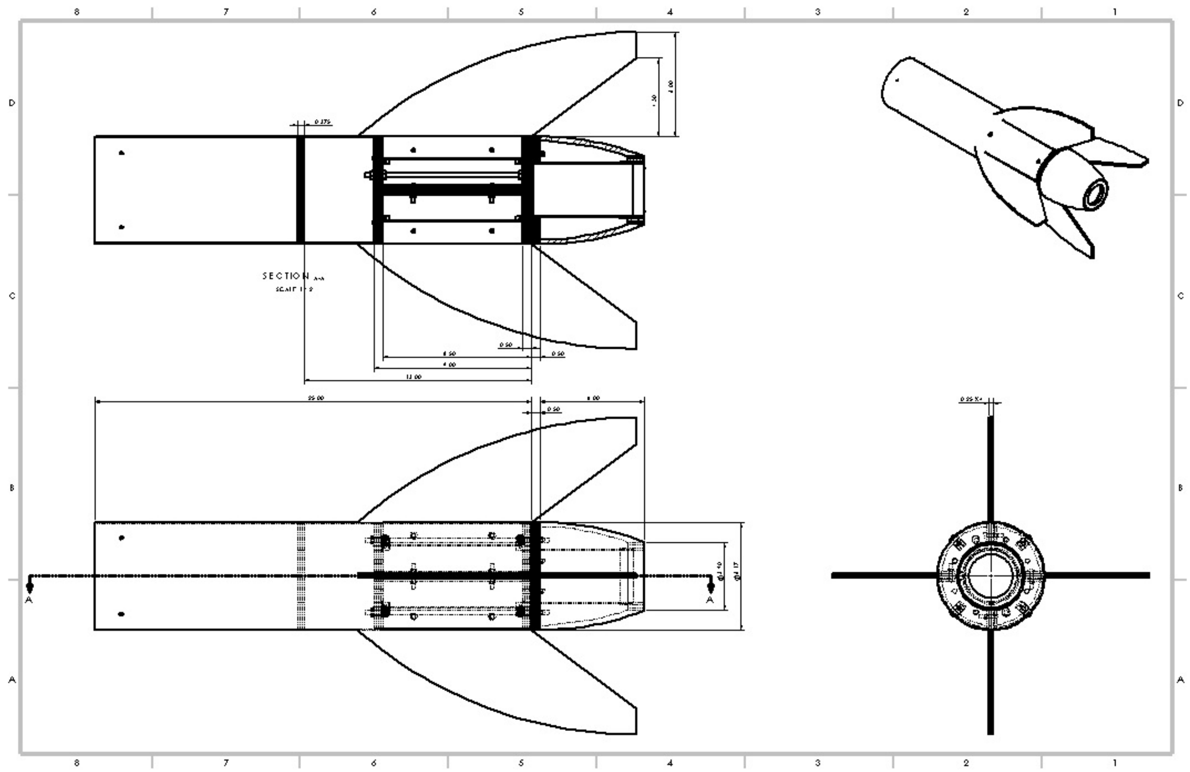

Figure 29: Overall dimensions of the entire fin can assembly.

The removable fin assembly will be constructed out of two centering rings. These centering rings will be connected using plywood runners permanently epoxied into slots on the centering rings. Additionally, two 1/4-20 threaded rods will be used to connect the centering rings. The purpose of these threaded rods is to provide structural rigidity and prevent buckling. These rods will also be used to secure the thrust plate to the rest of the removable fin assembly. Fins will then be bolted onto the plywood runners using two #8-32 machine screws per fin. The entire assembly can then be slotted into the airframe section and is secured using eight #8-32 machine screws. An exploded view of the removable fin assembly is shown in Fig. 30 below.

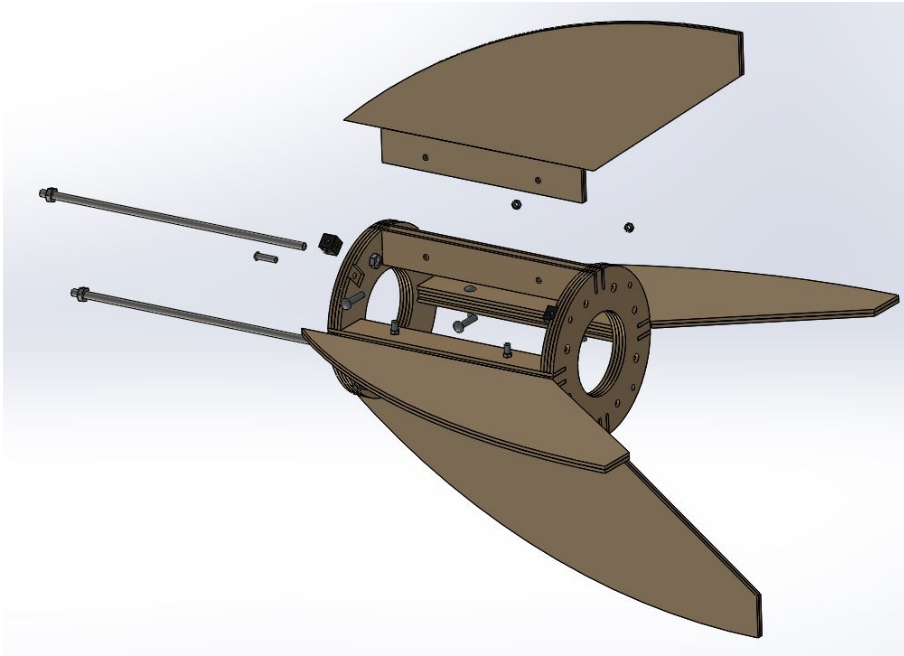

Figure 30: Exploded view of the removable fin assembly. A fin is bolted to the runners between centering rings.Threaded rods re-inforce the entire assembly

This removable fin assembly was chosen because it allows for greater reusability of the launch vehicle. If any component of the assembly is damaged during flight, it can be easily removed and replaced. Removable fins also allow for different fin designs for different flights. This could be beneficial in changing the CP of the launch vehicle to modify stability. Furthermore, this design does not require a motor tube that runs the entire length of the fin can. The motor is adequately held in place using the three centering rings that are housed inside the airframe as well as an Aeropack motor retainer attached to the thrust plate. The use of a thrust plate to transfer the force to the airframe also allows for the removal of the motor tube. This is further discussed later in this section.

The thrust plate is designed to transfer the force of the motor directly into the airframe of the launch vehicle. An exploded view of the proposed design is shown in Fig. 31 below. The thrust plate will be made of a combination of plies of 1 ply of aluminum and 3 plies of plywood. The aluminum ply will be the forward most ply that makes contact with the airframe. The purpose of this ply is to bear most of the load of the motor. Aluminum was chosen because, unlike plywood, it will not compress under the force of the motor or under repeated loading. The aluminum ply will then be bolted to the plywood plies using four #8-32 machine screws. A short 6 in. long section of motor tube will then be epoxied to the plywood thrust plate plies. The purpose of this motor tube section is to hold the motor retainer. The tail cone will then also be epoxied to the motor retainer and is further discussed later in this section. The thrust plate will be secured to the rest of the launch vehicle by being bolted to the 1/4-20 threaded rods that run through the removable fin assembly. The thrust plate will be 6 in. diameter and the overall length, including tail cone and motor tube, will be 6.5 in. A dimensioned drawing of this assembly is shown in Fig. 32 below.

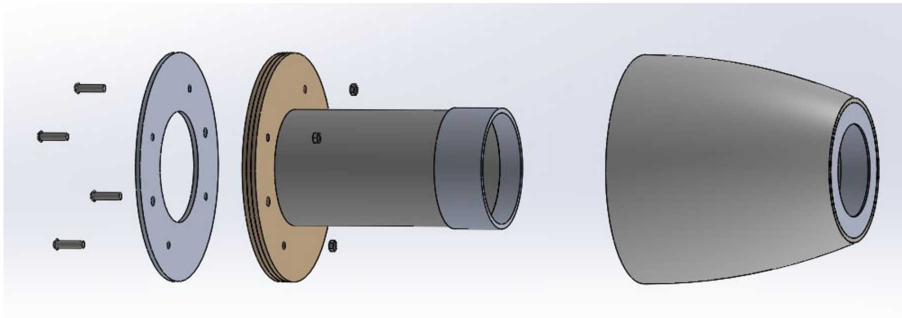

Figure 31: Exploded view of the thrust plate assembly. An aluminum plate is bolted to a plywood centering ring which allows for the motor tube to be epoxied into place and the tail cone to be attached to the motor retainer.

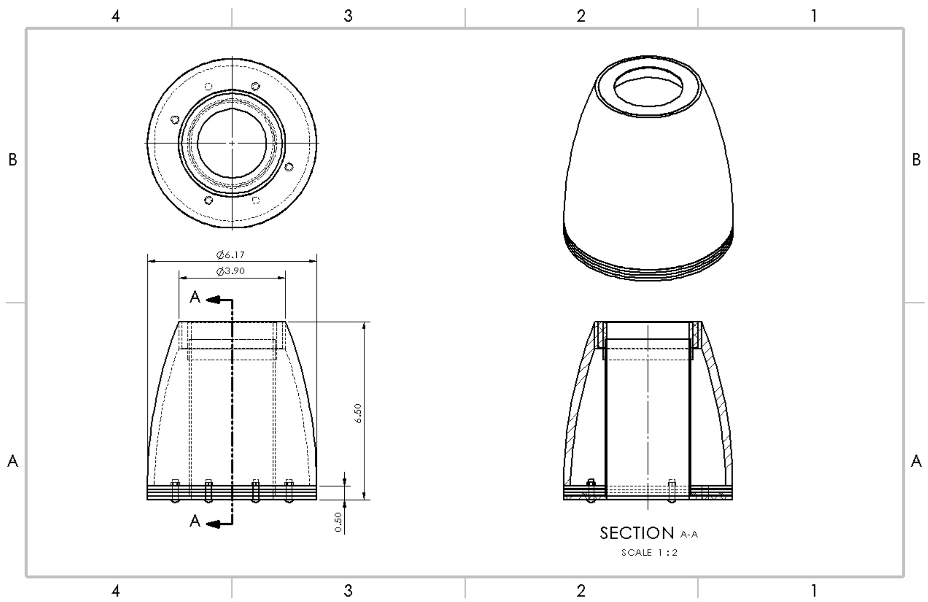

Figure 32: Dimensioned view of the entire thrust plate assembly.

The use of a thrust plate allows all of the force of the motor to be directly applied to the airframe of the launch vehicle. This is beneficial because the fiberglass airframe is the strongest component of the launch vehicle. Additionally, the centering rings in the fin can are no longer load-bearing and can be made thinner and lighter. Furthermore, this design allows for the removal of the motor tube. With a thrust plate, a motor tube is not necessary. In a fixed fin design, the purpose of the motor tube would be to transfer the force to the centering rings and then the airframe. The inclusion of a thrust plate accomplishes this by transferring the force of the motor straight to the airframe. As such, a motor tube that runs the full length of the fin can is not needed and the motor can be held axially aligned by the centering rings alone.

The tail cone will be epoxied to the motor retainer this will allow it to be removed via a screw on cap. Additionally, this method of attaching the tail cone does not require the tail cone to support any loads and only acts as aerodynamic improvement. An exploded view of the entire fin can showing how

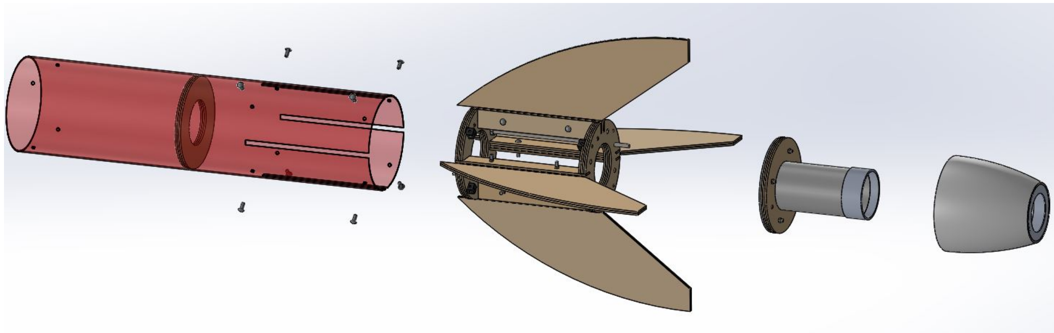

Figure 33: Exploded view of the entire fin can.

3.3.10.2 Preliminary Shear on Fin Can Bolts

Calculations of the maximum shear expected to be endured by the #8-32 bolts on the fin can were performed in order to ensure an adequate factor of safety. To perform the necessary calculations, the bolts were assumed to fail in single shear. The shear force in a bolt is given by Eqn. 3 below.

$$\tau = \frac{F}{A} \tag{3}$$

The shear stress, τ, is a function of the force applied to the bolt, F, and the area that were to be exposed should the bolt fail, A. The diameter of the bolts is known and the yield stress is a material property that can be found in reference texts. Using these values, Eqn. 3 was re-arranged and solved for the maximum force that can be applied to the bolts before yield. This force was found to be 658.74 lb. which gives a factor of safety of 12.9.

These calculations used the maximum force of the most powerful motor under consideration as 407.8 lb. The type of steel used was 304 stainless as this is a common steel used in small fasteners. The assumption was made that this force was evenly distributed over all 8 bolts so that each one only needed to support 1/8 of the thrust of the motor. In reality, the thrust plate will transfer most, if not all, of the force of the motor to the airframe of the launch vehicle. This greatly reduces the load supported by the bolts decreasing the chances of failure. These calculations show that it is feasible to use eight #8-32 machine screws to hold the removable fin assembly into the airframe. Further analysis and FEA will be done to ensure that this assembly is fail safe.

3.3.10.3 Preliminary Buckling of Fin Can Rods

Hand calculations were also performed to ensure that the threaded rods running along the removable fin assembly would not buckle under the force of the motor. The threaded rods were treated as thin columns that are pinned at both ends. This is a valid assumption because the rods have radius much less than their length and are fixed to a centering ring at either end. The critical load under which buckling occurs is then found using Eqn. 4 below [15].

$$P_{cr} = \frac{\pi^2 EI}{l_e^2} \tag{4}$$

Where the critical load, P_{cr}, is given as a function of the column's modulus of elasticity, E, area moment of inertia, I, and the effective length, l_e, of the column. For a column that is fixed at both ends, the effective length is taken to be ½ of the total length of the column. Modulus of elasticity was found online [4] and the moment of inertia was calculated from the radius of the rod. Using these values, the critical load of the column was found to be 3307.6 lb. with a factor of safety of 16.2. This factor of safety was determined by assuming that each rod supported half of the maximum force of the motor. Because the thrust plate will transfer most of the motor's force to the airframe, it is likely that these rods will endure much smaller loads.

3.3.11 Fin Configuration

The launch vehicle will have 4 fins. This allows for camera housings to be epoxied in between the fins. The camera housings need to be in between the fins in order to ensure that they have minimal impact on the aerodynamics of the fins. Furthermore, the four-fin configuration allows for the surface area of each individual fin to be minimized while maintaining adequate stability.

3.3.12 Fin Design

This launch vehicle will utilize a ogive LE fin design that decreases drag while increasing the stabilization efficiency. All edges of the fin that influence airflow will have a significant non-uniform bevel that rounds the edge while stretching several times the thickness of the fin into the cord. By rounding the profile and profile edges, zones of low pressure are reduced, which decreases the non-skin friction drag of the fin. By sweeping the TE back past the root cord, the CP of the fin and by extension the vehicle are moved farther aft than traditional perpendicular TEs for the same surface area. Fig.(34) shows a dimensioned 2D drawing of the selected fins.

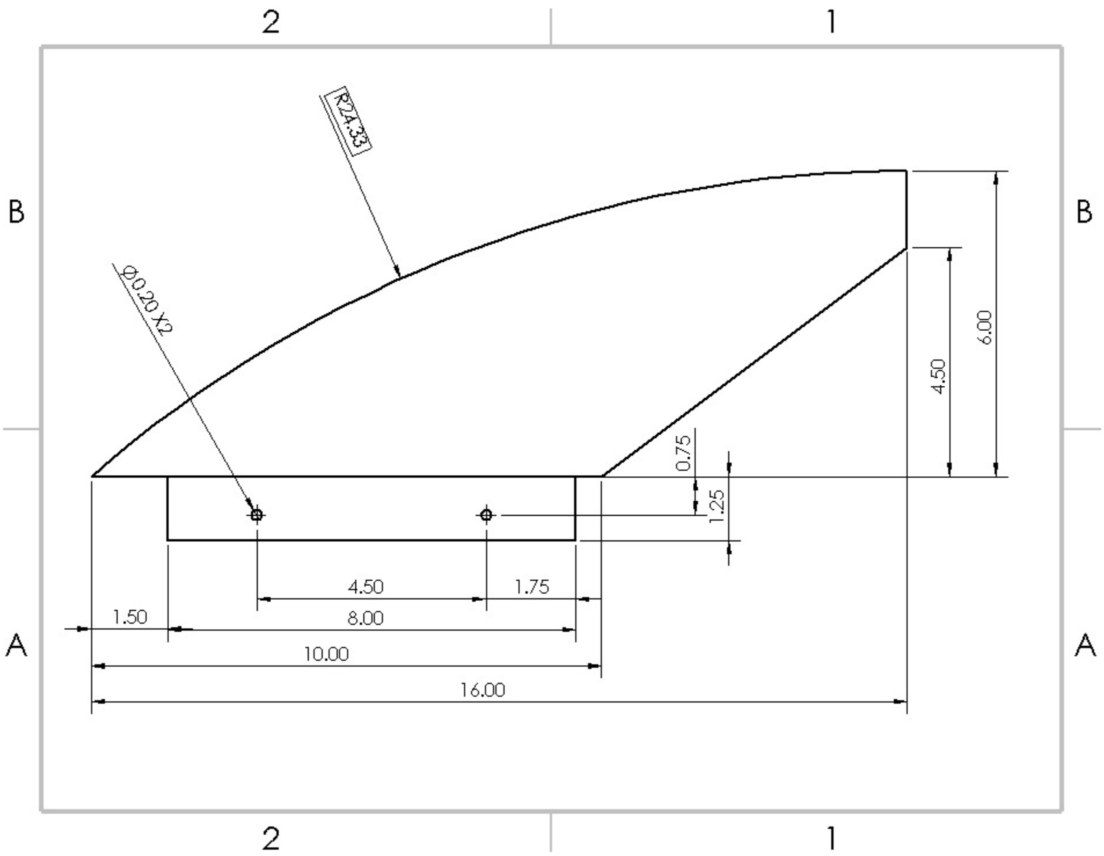

Figure 34: Dimensioned drawing of the geometry of the ogive fin. Root chord is 10 in. and the fins will span 6 in. out of the launch vehicle.

3.3.13 Fin Material and Construction

The fins of the launch vehicle will be constructed as a sandwich composite. The core of the composite will be made from 1/4 in. thick balsa wood. Two plies of 7.5 $\frac{oz}{yd^2}$ plain weave fiberglass cloth will be laminated on each side of the balsa core. West Systems 105 Epoxy Resin and 206 Slow Hardener will be used for the composite layup. Once the fiberglass has been applied, the entire fin will be sealed in a vacuum bag until the epoxy has cured. A diagram of this layup is shown in Fig. 35 below. After the epoxy has cured, the LE of the fin will be sealed with an additional coating of epoxy. This step is necessary in order to ensure that the fiberglass plies do not delaminate from the core.

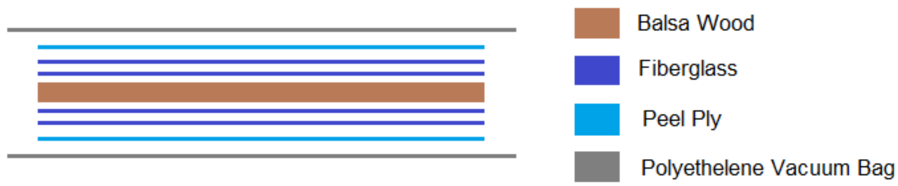

Figure 35: Diagram of the different layers in a composite fin layup. A balsa wood core is laminated with 4 layers of fiberglass and cured in a vacuum bag for 24 hours.

This design will provide fins that are lightweight while maintaining structural stability. It is expected that composite fins will save 47 percent weight over the solid plywood alternative. Expected weight savings was calculated based on the density of each type of wood as well as the density of fiberglass.

3.3.14 Tail Cone

A tail cone will be included on the launch vehicle in order to reduce the bluff body aerodynamic drag on the launch vehicle. The tail cone will be secured to the launch vehicle by being expoxied to the motor retainer.

3.3.14.1 Shape

The tail cone will be manufactured with an Ogive profile. This design of tail cone was chosen because it offers the best aerodynamic benefits. The one drawback is that it may be harder to manufacture. However, the team has access to many 3D printers that should be capable of printing this design. The tail cone will be 6.17 in. diameter at the widest point and 6 in. long.

3.3.14.2 Material and Manufacturing

The tail cone will be 3D printed using PETG filament. 3D printing is a relatively easy manufacturing method when compared to alternatives. PETG was chosen over other alternative filaments, such as PLA or ABS, because of its higher melting temperature of 500 degrees Fahrenheit.

3.3.14.3 Thermal Insulation

The tail cone is in close proximity to the heat of the motor, therefore heat resistance is important. In order to insulate the tail cone from the motor, the motor tube will be wrapped in a layer of cork that is 1/8 in. thick. Cork has a very low thermal conductivity at 0.04 $\frac{W}{m \cdot K}$ [21]. Furthermore, cork is a flexible material that is easy to cut allowing for easy manufacturing.

3.3.15 Bulkhead Sizing

Hand calculations were performed in order to estimate the required bulkhead thickness in order to withstand all loads. For these calculations, the bulkheads are assumed to be circular flat plates. This is an accurate assumption because the thickness of the bulkhead is much less than its radius. Additionally, the force will be approximated as a single force evenly distributed over a small circular area. This is accurate because the force by the u-bolt will be relatively evenly distributed. This is supported by the fact that large washers will be used to distribute the force. The following equation can then be used in order to find the moment on a bulkhead based on the applied force [12].

$$M_r = \frac{W}{4\pi}[(1+\nu)ln(\frac{a}{r}) - 1 + \frac{(1-\nu)r_0^{'2}}{4r^2}] \tag{5}$$

where,

$$r_0^{'} = \sqrt{1.6r_0^2 + t^2} \quad \text{for} \quad r_0 < 0.5t$$

or

$$r_0' = r_0 \quad \text{for} \quad r_0 \geq 0.5t$$

Based on these equations, the moment on the bulkhead, M_r, can be found as a function of the bulkhead loading, W, Poisson ratio, ν, radius, r, the radius of the loading area, r_0.

If we assume a 1/2 in. thick bulkhead, the maximum moment the bulkhead can withstand can be found from the materials ultimate stress and material properties. Material properties were taken from a study published in 2022 [5]. The maximum moment the bulkhead can withstand is then given by [15]:

$$M_x = D(\frac{1}{\rho_x} + \frac{\nu}{\rho_y}) \tag{6}$$

$$M_y = D(\frac{1}{\rho_y} + \frac{\nu}{\rho_x}) \tag{7}$$

where the moment, M, is a function of the flexural rigidity, D, Poisson ratio, ν, as well as the radius of curvature, ρ of the deformed plate. The radius of curvature was found as a function of the max expected displacement of the bulkhead as well as its strain. Thus, by setting the moment found by Eqn. 6 or 7 equal to the moment in Eqn. 5, the maximum load that a 1/2 in. bulkhead can withstand can be found.

These calculations are able to show that a 1/2 in. thick bulkhead has adequate strength to withstand the expected deployment forces. It is important to note that there were many assumption made in these calculations. While this estimate should still be accurate it is important to perform additional calculations in order to verify that the bulkheads will not fail. More advanced ANSYS Structural simulations will be performed to verify that all bulkheads have proper thickness to support the required loads.

3.3.16 Vehicle Weight Estimates

Estimated weights were obtained for each component of the launch vehicle. The current mass of the launch vehicle is 43.22 lb. A detailed breakdown of the weight of all components of the launch vehicle is shown in Table (7) below.

Table 7: Overall weights of every section of the launch vehicle. Overall weight is expected to be 43.22 lb.

Section	Weight (lb)
Nose Cone	6.66
Main Parachute Bay	6.05
Avionics Bay	3.25
Drogue Parachute Bay	3.05
Payload Bay	8.08
Fin Can	16.13
Total Weight	43.22

This weight estimate was obtained by a combination of weighing different components of the launch vehicle as well as calculating weight based on density. A breakdown of the mass of individual components in each section of the launch vehicle is shown in Table (8) below.

Table 8: Breakdown of the weight of each component for each independent section of the launch vehicle.

Avionics Bay		Drogue Parachute Bay		Payload bay	
Component	**Weight (lb)**	**Component**	**Weight (lb)**	**Component**	**Weight (lb)**
Sled	0.18	Airframe	2.24	Airframe	0.52
Bulkheads	0.8	Drogue Parachute	0.07	Coupler	1.33
Threaded Rods and U-bolts	0.55	Shock Cord	0.45	Payload Mass	6
Airframe	0.26	Quick Links	0.14	U-Bolt	0.15
Coupler	1.46	Nomex	0.140625	Quick Link	0.07
Nose Cone		**Main Parachute bay**		**Fin Can**	
Component	**Weight (lb)**	**Component**	**Weight (lb)**	**Component**	**Weight (lb)**
Nose Cone	4.6	Airframe	2.64	Airframe	3.3
Bulkhead	0.34375	Coupler	1.2	Motor Tube	0.3
Centering Ring	0.25	Main Parachute	1.375	Fins	0.75
Ballast Mount	0.25	Shock Cord	0.45	Thrust Plate, Tailcone, and Retainer	0.5
Ballast	1	Quick Links	0.14	Motor Casing and Propellant	10
U-Bolt and Quick Links	0.22	Deployment Bag	0.25	Centering Rings	0.625
				Threaded Rods and Misc. Hardware	0.42

3.3.17 Motor Choice

Based on simulations and hand calculations, the motor best suited to propel the launch vehicle to the desired competition altitude is the L1520T. The motor provides and initial thrust of 1,545 N, making the initial thrust to weight ratio of 8.03. The launch vehicle will clear the launch rail with a velocity of 74.6 ft/sec.

3.4 Recovery Subsystem

3.4.1 Description of Recovery Events

Two dual deploy altimeters will be used on the launch vehicle for redundancy. Prior to launchpad assembly, both altimeters will be programmed and tested to ensure continuity and usability on the launchpad and during flight. This involves using a pressure chamber to simulate launch vehicle ascent and descent to confirm the altimeters are operating correctly and can correctly sense pressure change during flight. Once both altimeters are confirmed to be operating correctly, they will be disconnected from their power source, installed on the avionics sled, and set aside for launch. The power source for the altimeters will also be checked to confirm a full charge for launch.

Prior to drogue parachute bay and main parachute bay assembly, both parachutes will be connected to their respective shock cords via metal quick links. The shock cords are then subsequently attached to their respective bulkheads by quick links. Both the main parachute and the drogue parachute will be wrapped securely in Nomex cloth in order to protect the parachutes from the ejection charges. In addition the shock cords will be insulated using dog barf insulation to protect the Kevlar from the ejection charges. The dog barf insulation and the Nomex cloths contribute to the reusability of the launch vehicle.

After the completely assembled launch vehicle has been mounted onto the launch pad, both altimeters will be armed. Arming altimeters after the launch vehicle has been positioned on the launch rail and before the motor has been armed reduces the risk of involuntary detonation of any vehicle separation charges. Additionally, the altimeters must be armed before a motor igniter is inserted. This is done so that in the event of a premature motor ignition, the launch vehicle will return to the ground under parachute. Once armed, the altimeters audio outputs will once again be checked and recorded to confirm operation, continuity of the altimeters to the ejection charges, and battery voltages. If any errors are displayed during this process the launch process will be halted and the proper altimeter functionality will be restored before resuming the launch process. After altimeter functionality is confirmed, the full launch procedure will commence.

After launch, once the primary altimeter detects an apogee event it will send a signal to the primary drogue ejection charge, with the secondary altimeter following suit with a one second delay. This will cause the avionics bay to separate from the drogue parachute bay by breaking the four 4-40 nylon shear pins that ensure connection during ascent. Once the two sections are separated, the drogue parachute will be released form the bay and deploy completely.

The launch vehicle will proceed to descend under drogue with the fin can, payload bay, and drogue parachute bay hanging approximately two feet lower than the other sections of the launch vehicle. This is done so that the sections of the launch vehicle do not hit each other during descent. The descent velocity under drogue is constant and can be calculated by using the known coefficient of drag of the drogue parachute and the weight of the entire launch vehicle, using equation 8. The drogue descent velocity is higher than the acceptable kinetic energy on landing but low enough such that no damage will occur to the launch vehicle upon main deployment. This shock force is calculated using equation 28. The launch vehicle will continue to descend under drogue until the primary altimeter detects an altitude of 550 feet. At this point the altimeter will send a signal to the main primary ejection charge, which will separate the coupled avionics bay and main parachute bay from the nose cone. At an altitude of 500 feet the secondary altimeter will send a signal to the main secondary ejection charge to ensure main parachute deployment. This separation will allow the main parachute to deploy and slow the launch vehicle down to meet the landing kinetic energy requirements. The nose cone will be secured to the main parachute bay utilizing 4-40 nylon shear pins, and the ejection charge will be calculated to be sufficient in shearing the pins, shown in Section 3.4.2.9.

3.4.2 Recovery Alternative Designs

3.4.2.1 Tracking Device Alternatives

Per NASA requirements, each independent section of the launch vehicle must have a tracking device to locate the launch vehicle upon landing. Since all sections will be held together by shock cord upon descent and landing, only one tracker needs to be selected. Out of all the possible options for remote GPS trackers aboard the launch vehicle, there are four leading candidates: the Big Red Bee 900 and Beeline, the Eggfinder GPS, and the LightAPRS-W. Out of these four options, two have been used and are owned by the club, and all four of the options are popular trackers across the hobby rocketry community.

Table 9: Tracking device alternatives under consideration.

Tracker	License Required	Transmitter Power	Transmitter Frequency	Range	Owned by Team	Cost	Additional Comments
Big Red Bee 900	No	250 mW	900 MHz	6 miles	Yes	$199.00	Simplest Option
Big Red Bee Beeline	Yes	100 mW	420 - 450 MHz	40+ miles	No	$359.00	Most expensive, comes in multiple models
Eggfinder GPS	No	100 mW	900 MHz	2 miles	Yes	$90.00	Most experienced use in club, well within maximum drift distance, cheapest
LightAPRS-W	Yes	10 mW	134-174 MHz	100 miles	No	$115.00	Largest range and lowest power required, also has temperature and pressure sensors

Two of these options operate on the 900 MHz band spectrum, the Big Red Bee 900 and the Eggfinder GPS, while the other two operate in amateur radio frequency ranges. This requires a special license for operation which would need to be acquired prior to use.

The Big Red Bee 900 is the simplest of the options. It consists of a transmitter mounted to a microcontroller board alongside an antennae. This system is powered by a single cell LiPo battery, and as it transmits in the 900 MHz frequency no extra license is required for its operation. This board would be fastened to the avionics sled of the launch vehicle. A coupled handheld receiver displays the battery voltage and location coordinates of the transmitter. This receiver can in turn be interfaced with a laptop to display the transmitter location utilizing Google Maps. Some drawbacks to this unit are limited durability and reliability issues.

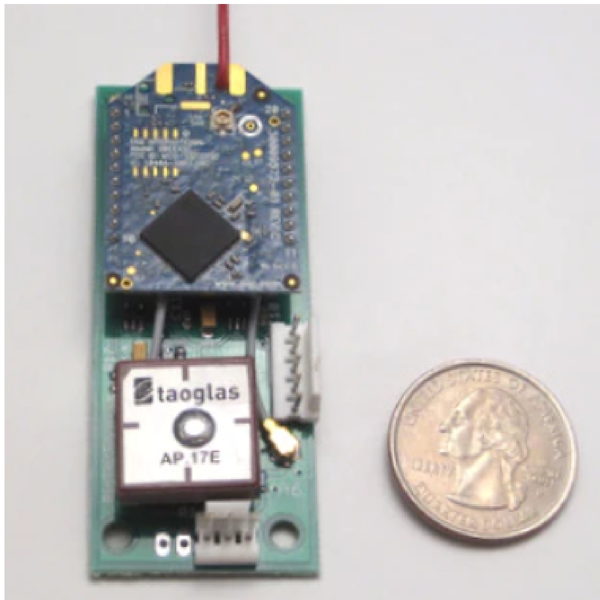

Figure 36: A BigRedBee 900 tracker.

The Big Red Bee Beeline is an advanced version of the Big Red Bee 900. It has the advantages of almost seven times the range at more than 40 miles, as well as operating in the 420-450 MHz range, making it much more reliable and less responsive to interference. This device is again operated with a coupled handheld receiver, however, this version is not capable of interfacing with a laptop. Instead it only uses a directional antennae that allows the users to hone in on the transmitter. While this makes the device much more accurate, it is also more expensive, and requires a license to operate on those specific frequencies.

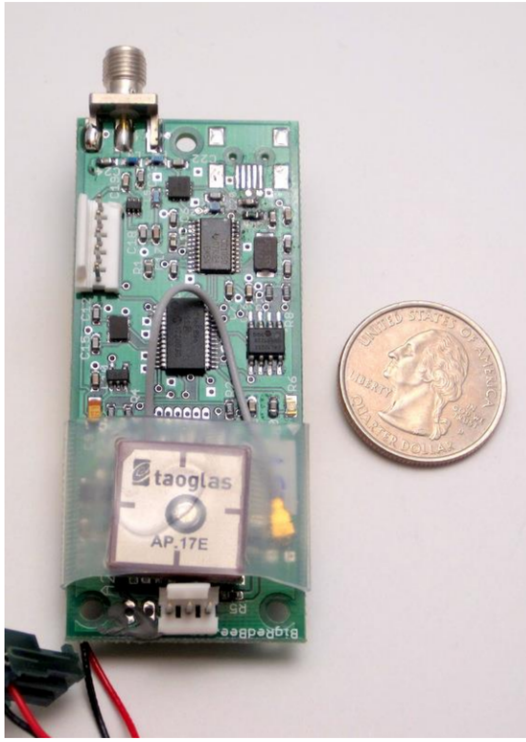

Figure 37: A BigRedBee Beeline tracker.

The Eggfinder GPS has significant legacy use within HPRC. The system uses an Eggfinder TX transmitter with a GPS module, an antenna, and a 2S (7.4V) LiPo battery. A handmade handheld receiver is used to receive the transmitter's signal. However, the lack of documentation for both the transmitter and the receiver complicates troubleshooting and purchasing new transceivers. As the Eggfinder transmits on the 900 MHz wavelength, no license is required for its operation.

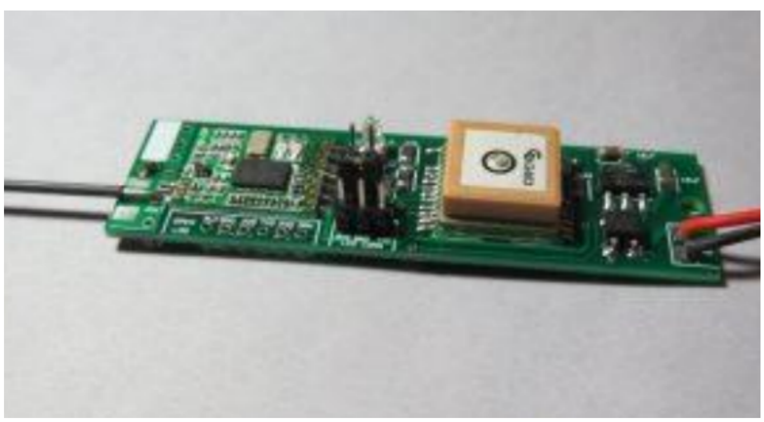

Figure 38: An Eggfinder TX transmitter.

The LightAPRS-W tracker is the final alternative for the launch vehicle tracking device. A much more robust option, it relies on an ATmega microcontroller board paired to a high frequency APRS transmitter and a WSPR transmitter. Since the transmission is on amateur radio frequencies it does not require a specialized receiver and a handheld radio receiver or a receiver connected to a laptop instead. This method is by far the most precise out of all the options and can be used at a much larger range than any of the others. However, it is highly unlikely that such large a range would ever be necessary, and the high cost, radio licenses, and complexity make this choice much less appealing an option. Additonally, it is possible that this tracker could interfere with payload electronics because they operate on very similar frequency bands.

Figure 39: A LightAPRS-W tracker.

3.4.2.2 Altimeter Alternatives

Two redundant dual deploy altimeters will be used to control the parachute deployment for the launch vehicle. Table (10) shows the four leading alternatives for altimeters aboard the launch vehicle.

Table 10: Possible altimeter options for use on the launch vehicle.

Altimeter	Deployment Variability	Altitude Recording Resolution	Dimensions	Data Recorded	Price	Owned by Club
RRC3	300 to 3000 ft 100 ft Increments	1 ft	3.92” x .925”	Altitude, Velocity, Temperature	$96.50	Yes
Entacore AIM	100 to 100,000 ft 1 ft increments	1 ft	2.75” x .984”	Altitude, Velocity, Temperature	$121.15	Yes
Stratologger CF	100 to 9999 ft 1 ft Increments	1 ft	2” x .84”	Altitude, Voltage, Temperature	$69.95	Yes
EasyMini	100 to 100,000 ft 100 ft increment on ascent 10 ft increment on descent	1 ft	1.5” x .8”	Altitude, Velocity, Acceleration, Temperature, Voltage	$96.93	No

The altimeter is a vital component of the recovery system as it controls the ejection charges that the launch vehicle uses to deploy the parachutes, as well as recording the competition altitude. Out of the four choices, the club owns the RRC3, the Entacore AIM, and the Stratologger CF, and has multiple years of experience on both the RRC3 and the Stratologger. Since all four options meet all requirements for dual deployment and altitude recording, the individual altimeters' form factor, reliability, user friendliness when programming, as well as precision are the defining characteristics looked at when making a final selection.

As demonstrated previously in the club, the RRC3 altimeter and the Stratologger altimeters have user friendly programming capabilities, and the Entacore is not far behind. The RRC3 and the Entacore both have large form factors that can occasionally clutter a subscale avionics sled. The stratologger and Entacore have both been known to occasionally fail in parachute deployment, and are both less precise. The club has previously had trouble with the wiring of Entacore altimeters, and online reviews of the EasyMini suggest similar factors.

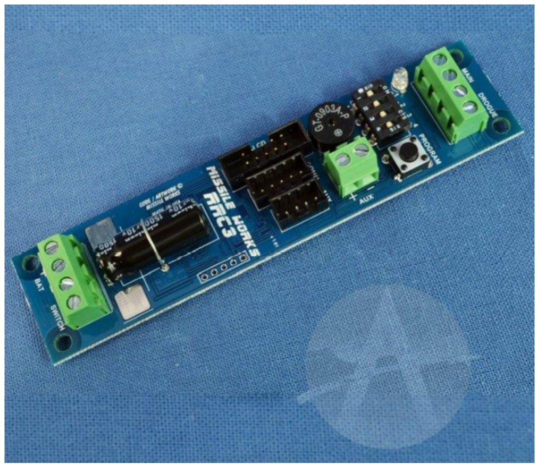

Figure 40: A Missile Works RRC3 ”Sport” altimeter.

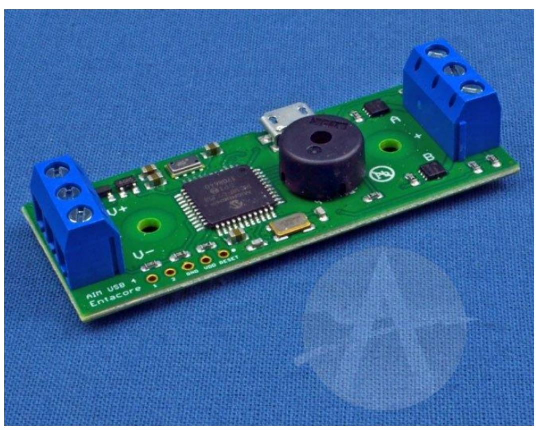

Figure 41: An Entacore AIM USB 3.0 altimeter.

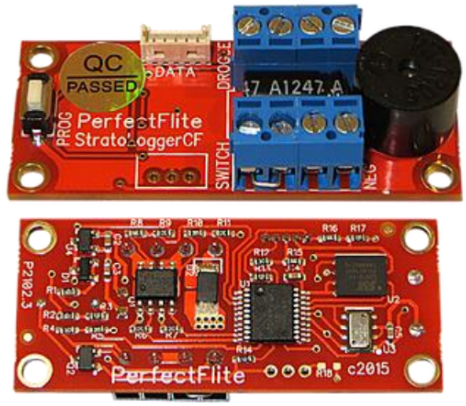

Figure 42: A PerfectFlite StratoLogger CF altimeter.

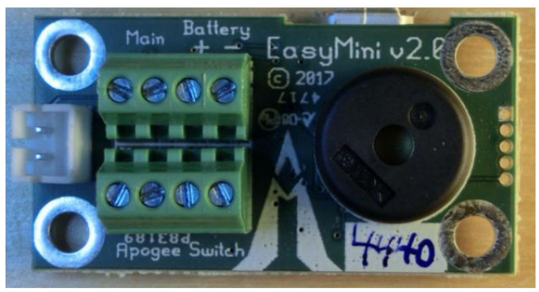

Figure 43: An Altrus Metrum EasyMini altimeter.

3.4.2.3 Altimeter Arming Alternatives

As stated by NASA requirements 3.5 and 3.6, the onboard altimeters need to be both armed by a mechanical switch from the exterior of the launch vehicle, as well be able to be locked in the on position. Two possibilities for these switches are being considered: the pin switch and the screw switch.

The most common method, and the one that the club is well versed in, are screw switches. These switches are very cheap and easy to assemble. They consist of a small PCB that acts as a break in the circuit. Two wires are soldered to the board and a single 3/16” size screw is able to be screwed into and out of the terminal in order to complete the circuit. This switch is mounted to the avionics sled in a way such that once the avionics bay is completely assembled, a screwdriver can be slipped through a machined hole in the body tube and tighten the screw in order to complete the circuit. It is a very simple mechanism that has worked for this club in the past. However, there have frequently been issues with locating the screw on the avionics sled once the avionics bay has been completely assembled in order to activate the altimeters.

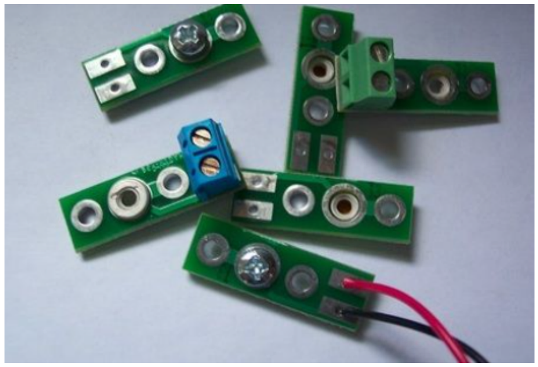

Figure 44: A 6-32 Screw Switch from Chris' Rocket Supplies.

Pull-pin switches is another popular method of providing an external mechanical means to arm an altimeter. They consist of a limit switch inside a housing that allows a pin to hold the circuit in the off position. Once this pin is removed, the circuit is completed. Commonly, a tag or a handle will be on the side of the pin made to be pulled to make the process easier, simpler, and smoother. For this design the switch would be mounted to the avionics sled similar to the method for the screw switches, that allow for the pull-pins to be able to be externally added and removed during the vehicles assembly process. The pull-pin method has the benefit of not being susceptible to inabilities to arm the altimeters while on the launchpad as all that needs to be done is remove the pin externally.

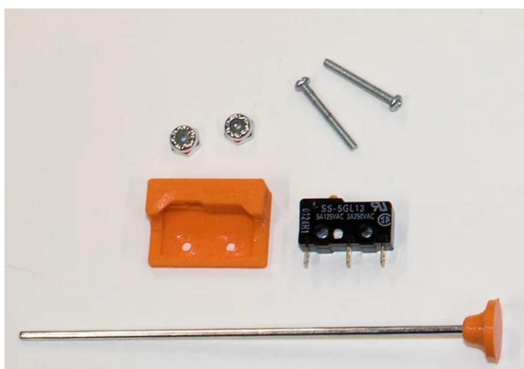

Figure 45: A Pull Pin Switch Kit from Lab Rat Rocketry.

3.4.2.4 Avionics Sled Alternatives

The four leading choices for avionic sled material are birch plywood, aluminum, fiberglass, and PEPG 3D printing filament. As the avionics sled is the mounting device for all the onboard altimeters and GPS tracking systems, it serves as an important piece of the launch vehicle. It is imperative that throughout flight, all of these components are securely mounted to this sled. Thus the sled itself must be capable of enduring up to 13 ft-lbf of force as calculated in section 3.5.12. As all mounted electronics must be easily accessible prior to and proceeding launch, it is vital that the design of the sled prioritize this same ease of access.

Typically the club has favored using birch plywood to make the avionics sled, due to the similar fabrication process of the bulkheads and fins. Using plywood also means that the avionics sled is capable of withstanding the large forces exerted on the launch vehicle during ascent and landing impact, while still being lightweight and easy to manufacture. Using a jigsaw design approach, a laser cutter can be used to cut the plywood with correct dimensions with epoxy resin being used to glue everything together. A plywood avionics sled also means that any additional holes can simply be drilled into the wood with no fear of damaging the surrounding material.

Another leading option is to use PEPG 3D printing filament to manufacture the sled. The benefits to this design include a greatly reduced complexity in the manufacturing process, as only one print would be necessary to complete the sled. This approach also leaves the overall sled much lighter than any of the alternative options.

3.4.2.5 Block Schematic of Recovery Electronics and Proof of Redundancy

The below schematic shows all electronic components involved in this recovery system as well as the connections between them.

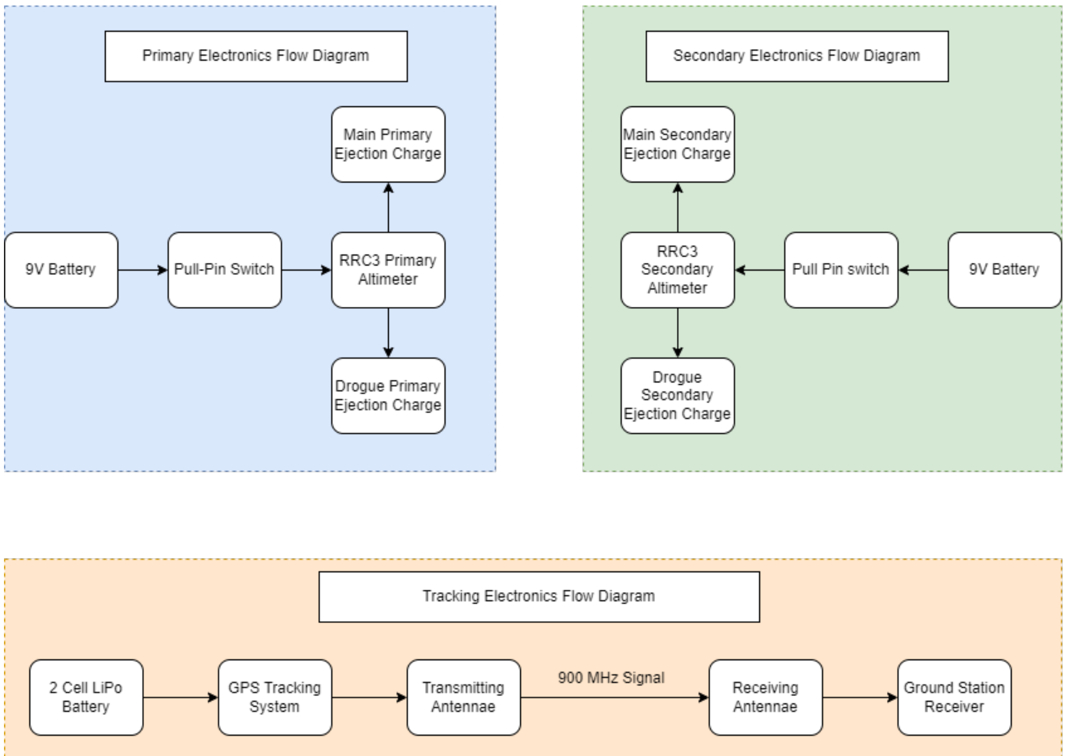

Figure 46: Flow diagram of recovery electronics.

The dashed blue box represents all electronic components of the primary recovery method including the power supply, the mechanical arming switch that activates the system, the altimeter used, one ejection charge to separate the avionics bay from the lower payload bay and release the drogue parachute, and one ejection charge to separate the main parachute bay from the nose cone and release the main parachute. The dashed green box represents the components that have the same function as those in the dashed blue box but are there for redundancy. This includes another power source, another mechanical arming switch, another altimeter, and two more ejection charges to release the drogue parachute and the main parachute.

The primary altimeter is what will be used to record the competition altitude. Each altimeter is connected to a pair of terminal blocks via a quick connect soldered connection. All four ejection charges are connected to the other side of the terminal blocks and wired through the bulkheads with plumbers putty sealing the connection from the ejection charges to the rest of the recovery electronics. Each secondary ejection charge is larger than the primary ejection charge by half a gram.

The orange box below represents the tracking system and its components. This system includes the onboard components as well as the receiving components on the ground. The onboard components include a power source, the onboard GPS tracker, and the antennae that transmits the location. The ground components include the receiving antennae and the receiving electronics.

The separation between these three systems is readily apparent in this diagram, as well as the redundancy incorporated into the overall system. By using two separate altimeters wired to two separate power sources we ensure that there are two independent systems that if one fails, the other will seamlessly takeover.

3.4.2.6 Drogue Parachute Alternatives

Drogue parachute selection is mainly a factor of descent rate. If this descent rate is too low, then the total descent time of the launch vehicle will exceed the maximum allotted descent time stipulated by NASA requirement 3.11. A slow drogue descent rate will also increase the total drift distance which needs to be below 2500 feet. On the other hand, if the descent rate under drogue is too high then the shock the launch vehicle experiences when the main parachute is deployed will be too excessive. This force is calculated in section 3.5.12 where the maximum force is primarily based on the maximum force that the bulkheads that connect to the shock cord containing the main parachute can withstand. This limit is calculated in section 3.3.15. The velocity of the launch vehicle under drogue can be calculated using the following equation where V is the descent velocity, g is the acceleration caused by Earth's gravity, m is the burnout mass of the launch vehicle A is the area of the parachute, C_D is the coefficient of drag of the parachute, and ρ is the density of the parachute.

$$V = \sqrt{\frac{2gm}{AC_D\rho}} \tag{8}$$

The other two calculated values of estimated descent time and wind drift distance are calculated using equation 25 and 26 in sections 3.5.9 and 3.5.10.

Table 11: Drogue parachute choices.

Parachute	Drag Coefficient	Descent Velocity	Descent time from Apogee to Main Deployment	Wind Drift (20 mph) From Apogee To Main Deployment	Does the Club Own It?
Fruity Chutes 12" Classic Elliptical	1.34	172.91 ft/s	22.84 seconds	670 ft	No
Fruity Chutes 15" Classic Elliptical	1.37	136.69 ft/s	28.89 seconds	847 ft	No
Fruity Chutes 18" Classic Elliptical	1.43	111.6 ft/s	35.39 seconds	1038 ft	Yes
Fruity Chutes 24" Classic Elliptical	1.47	82.43 ft/s	47.92 seconds	1406 ft	Yes
Fruity Chutes 24" Compact Elliptical	1.41	84.37 ft/s	46.81 seconds	1373 ft	Yes

Table (11) shows the drag coefficients, descent velocity, descent time, and wind drift of the launch vehicle under drogue. Since NASA does not set requirements for specifically descent under drogue or main, only for the overall launch vehicle descent, the drogue data needs to be combined with data from the main parachutes selected shown in Table (12). However, it is a team derived requirement that the drogue descent rate does not exceed 120 ft/s due to the shock caused when the main parachute is deployed. Based on this factor, both the 12" and the 15" fruity chutes parachutes will not be selected.

3.4.2.7 Main Parachute Alternatives

The main parachute selection provides the determining factor of impact kinetic energy. The impact kinetic energy is calculated in Section 3.5.8 using Eqn. 24, and the velocity used is calculated using the same descent velocity Eqn. 8 from Section 3.4.2.6. The values of descent time and drift distance are calculated the same way as the drogue values.

Table 12: Main parachute alternatives.

Parachute	Drag Coefficient	Landing Velocity	Impact Kinetic Energy	Descent time from Main Deployment	Wind Drift (20 mph) From Main Deployment	Does the Club Own It?
Fruity Chutes 72” Iris UltraCompact	2.033	23.38 ft/s	173.04 ft-lbf	23.51 seconds	690 ft	No
Fruity Chutes 84” Iris UltraCompact	2.134	19.56 ft/s	121.1 ft-lbf	28.11 seconds	825 ft	No
Fruity Chutes 84” Iris Ultra Standard	2.131	19.58 ft/s	121.27 ft-lbf	28.09 seconds	824 ft	Yes
Fruity Chutes 96” Iris UltraCompact	2.087	17.31 ft/s	94.8 ft-lbf	31.77 seconds	932 ft	No
Fruity Chutes 120” Iris UltraCompact	2.105	13.79 ft/s	60.16 ft-lbf	39.88 seconds	1170 ft	Yes
Fruity Chutes 144” Iris UltraCompact	2.118	11.46 ft/s	41.52 ft-lbf	48.01 seconds	1408 ft	No
Rocketman 120” Pro-X	.855	21.64 ft/s	148.19 ft-lbf	25.41 seconds	745 ft	Yes

3.4.2.8 Shock Cord Alternatives and Sizing

During descent, each separated section of the launch vehicle is connected to the others and the parachutes utilizing shock cord. This shock cord must be strong enough not only to endure the forces the parachute feels during drogue and main descent and be able to hold the weight of each section of the launch vehicle, it also needs to handle the immense loads felt by the launch vehicle during main deployment. As stated in section 3.5.12, the maximum load the shock cord needs to endure is about 310 ft-lbf at main deployment. In addition, the shock cord needs to be able to withstand the force, heat, and pressure of the black powder ignition charges when detonated. Common options for hobby rocketry shock cords are 1 inch nylon webbing cord, or 5/8 inch tubular Kevlar cord. While Kevlar is much stronger and resilient than the nylon webbing, the nylon webbing cord is much cheaper. Several lengths of kevlar shock cord are already owned by the club.

Shock cord lengths are typically 3 to 5 times the length of the launch vehicle used. As the predicted launch vehicle length is 104.5 in. and shock cords are sized in lengths of 5 ft., the shock cord must range in length of 320 to 520 inches. Since a lighter launch vehicle is always advantageous, the leading design is to have a 30 foot shock cord connect both the nosecone, main parachute, main parachute bay, and avionics bay, and another 30 foot cord connecting the avionics bay to the drogue parachute, drogue bay, and the lower half of the launch vehicle.

Once the launch vehicle has reached apogee height and the drogue parachute is ejected, the two section connected cannot be level with each other. This is because the descent of the launch vehicle involves a lot of movement for all sections of the vehicle, and if the sections are at the same level there will be collisions. For this reason it is appropriate to have at least a five foot separation between sections on descent. The upper section closer to the drogue parachute must be the section containing the main parachute, as once released it needs to open unimpeded by any hardware or other obstacles from the launch vehicle. If reversed the fin can, payload bay, and drogue bay could potentially fall on and puncture the main parachute resulting in a recovery failure and a potential loss of vehicle.

Once the main parachute has been ejected the launch vehicle will be in three separate sections. It is beneficial to have the fin can be the first section to contact the ground because it is the heaviest section. The nose cone will be the highest section of the launch vehicle during main descent and will hang above the main parachute bay.

The diagram below shows the shock cord lengths as well as the launch vehicle lengths for descent under both main and drogue parachute.

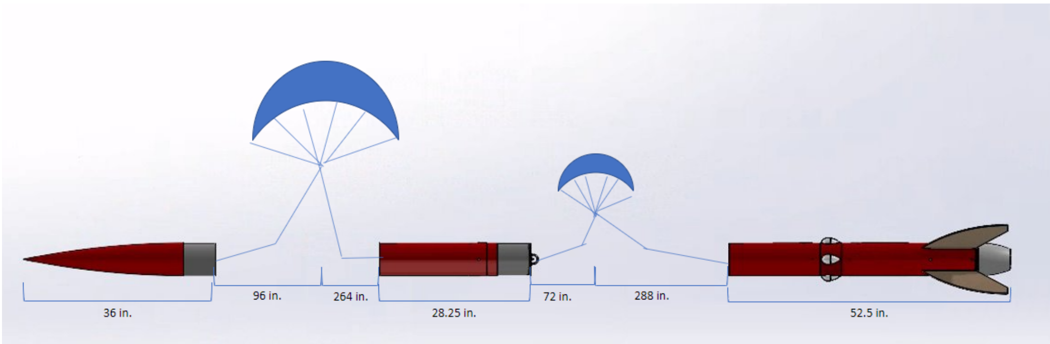

Figure 47: Diagram of the leading parachute placement.

3.4.2.9 Ejection Charge Sizing

The ejection charge that will be utitlized aboard the launch vehicle is 777 grade FFF black powder. This grade of black powder was choen for the fineness of the grains, as the finer the grain the quicker the combustion. A quicker combustion in turn leads to a cleaner break, and elss residue and unburnt black powder being left over in the compartments after the detonation of the charges.

the mass of the charges chosen is determined by the volume of the compartments where the separation points lie. The empty volume is calculated by using the total volume of the compartment and subtracting the volumes of the packed parachutes as well as the shock cord and other hardware in the compartment. By first calculating the volume of the compartment and the pressure the compartment needs to be at to break the shear pins, the ideal gas equation, shown below, is used to obtain the mass of the charge needed to separate the launch vehicle sections.

$$PV = mRT \tag{9}$$

In this equation, P represents the pressure of the compartment, V represents the empty volume of the compartment, m represents the mass of the black powder charge, R represents the known gas constant of the black powder combustion products, and T is the temperature of the black powder during combustion. The pressure needed to break the sections apart has been calculated by multiplying the number of shear pins used by the force needed to shear each individual pin. The pressure calculated is 10 psi, the known gas constant of black powder is approximately 22.16 ft-lbf, and the combustion temperature is 3307 degrees Rankine.

As stated in Section 3.4.2.5, a secondary black powder charge is used for each parachute for redundancy in order to ensure vehicle separation. These secondary charges will be .5 grams larger than the primary charge in order to ensure vehicle separation while at the same time not being large enough to cause damage to the bulkheads or structure of the launch vehicle. The masses of black powder ejection charges are given below.

Table 13: Ejection charge sizing for each separation point.

Point of Separation	Volume of Compartment	Mass of Primary Charge	Mass of Secondary Charge
Nose Cone and Main Parachute Bay	387.26 cubic inches	2.0 grams	2.5 grams
Avionics Bay and Drogue Parachute Bay	171.19 cubic inches	0.9 grams	1.4 grams

Following NASA requirement 3.2, the day before each launch there will be a ground ejection test of both primary charges to ensure separation between the sections. This allows for an experimental verification that all section can separate and that the ejection charges are sized correctly, as well as the launch procedure is accurately laid out and precisely documented. In the condition that the vehicle fails to separate during one of the tests, the ejection charge will be increased by .2 grams and the test will be repeated until separation occurs.

3.4.3 Recovery Leading Design

3.4.3.1 Avionics Bay

In the current leading recovery design, the avionics bay, located forward of the drogue parachute bay and aft of the main parachute bay, controls and contains all of the recovery electronics. The avionics bay itself is made up of an avionics sled made of PETG 3D printing filament, two threaded rods that run parallel through the bay, as well as two bulkheads that protect the sled and all the onboard electronics. The avionics sled will be made out of PETG 3D printing filament due to the ease of access that the club has to a 3D printer and the exact specifications that the sled can be cut to, as well as save on time and money printing and fabricating the sled. A sheet of aluminum foil will be placed on the flat surface of the avionics sled to prevent interference from the onboard altimeters to the GPS tracking system. Wired through the bulkhead there will be a total of four ejection charges rigged to detonate at apogee for the aft section and at 550 feet for the forward section to release the drogue and main parachutes respectively.

3.4.3.2 Parachute Selection

The drogue parachute that has been selected for the 2023 competition flight is the Fruity Chutes 18” classic elliptical parachute. This particular chute has been selected due to the fact that it is the smallest parachute that fruity chutes produces that results in the launch vehicle descending at a velocity below 120 ft/s. This consideration means that it lets the launch vehicle get from drogue deployment at apogee to main deployment at 550 feet the quickest, thus reducing the overall descent time of the vehicle as well as the overall drift distance. The drogue drift time associated with this parachute is calculated to be approximately 34.59 seconds and the maximum wind drift under 20 mph wind is approximately 1015 feet under a descent velocity of 111.6 ft/s.

The main parachute that has been selected for the 2023 competition launch is the Iris Ultracompact 120” fruity chutes parachute. This parachute has been chosen because it is one of two parachutes that lets the vehicle make ground impact with a kinetic energy less than 65 ft-lbf, with an estimated impact of 60.16 ft-lbf. The only other parachute that provides an acceptable impact kinetic energy is the Fruity Chutes 144” Iris Ultracompact parachute. However usage of this parachute increases total descent time of the vehicle under main parachute by approximately 8 seconds, as well as increasing the wind drift by approximately 250 feet from 1170 feet to 1408 feet. In addition the 144” parachute is not currently owned by the club.

With both the leading drogue and main parachute choices, the total descent time is estimated to be 75.27 seconds, and the maximum total drift distance is estimated to be 2208 feet. This selection satisfies both conditions with a considerable margin, and in fact qualifies our team for the bonus points awarded by recovering the launch vehicle in under 80 seconds. Using the same drogue parachute by changing the main parachute to the 144” fruity chutes would also have satisfied both criteria for the competition, but our team would not have been awarded the bonus points met by having the launch vehicle descend in under 80 seconds as the estimated descent time for that combination is 83.4 seconds.

3.4.3.3 Recovery Electronics

The leading recovery electronics include two RRC3 sport altimeters acting as primary and secondary altimeters, as well as the Big Red Bee 900 being the onboard GPS system. The RRC3 altimeters were primarily chosen because of their advanced functionality and capability to deploy multiple drogues and/or main parachutes if the design changes and becomes necessary, or even for future competitions proceeding this one where its advanced functionality would be extremely helpful. The use of the RRC3 on this year’s launch vehicle does not guarantee the use on subsequent years’ vehicles, but it makes it easier to pass down information and techniques. The Big Red Bee was chosen as it was the system with the best interface as well as being one that the team already owns and has experience in. It was chosen over the Eggfinder primarily because of the simplicity in receiving the transmitter’s signal on a handheld tracking device. The receiver will be manufactured by the team which will be interfaced with on a laptop.

3.5 Mission Performance Predictions

3.5.1 Launch Day Target Apogee

The target apogee of the launch vehicle is 4,500 ft., which was calculated via simulations in RockSim validated with calculations. Further description of the apogee prediction methods is shown below.

3.5.2 Flight Profile Simulations

RockSim simulations are the primary method of determining the flight parameters of the launch vehicle. Using RockSim, the expected apogee, velocity, and acceleration of the leading launch vehicle design are obtained. Utilizing a L1520T Aerotech motor for propellant, the expected flight profile of the launch vehicle is shown in Fig. (48) below. The parameters of the simulation are shown in Table (14).

Table 14: Launch simulation parameters used in Rocksim.

Parameter	Assumption	Justification
Launch Rail Angle	5°	Handbook 1.12
Launch Rail Length	144 in.	Handbook 1.12
Wind Speed	10 mph.	Median Flight Condition
Launch Direction	Into Wind	Standard Procedure

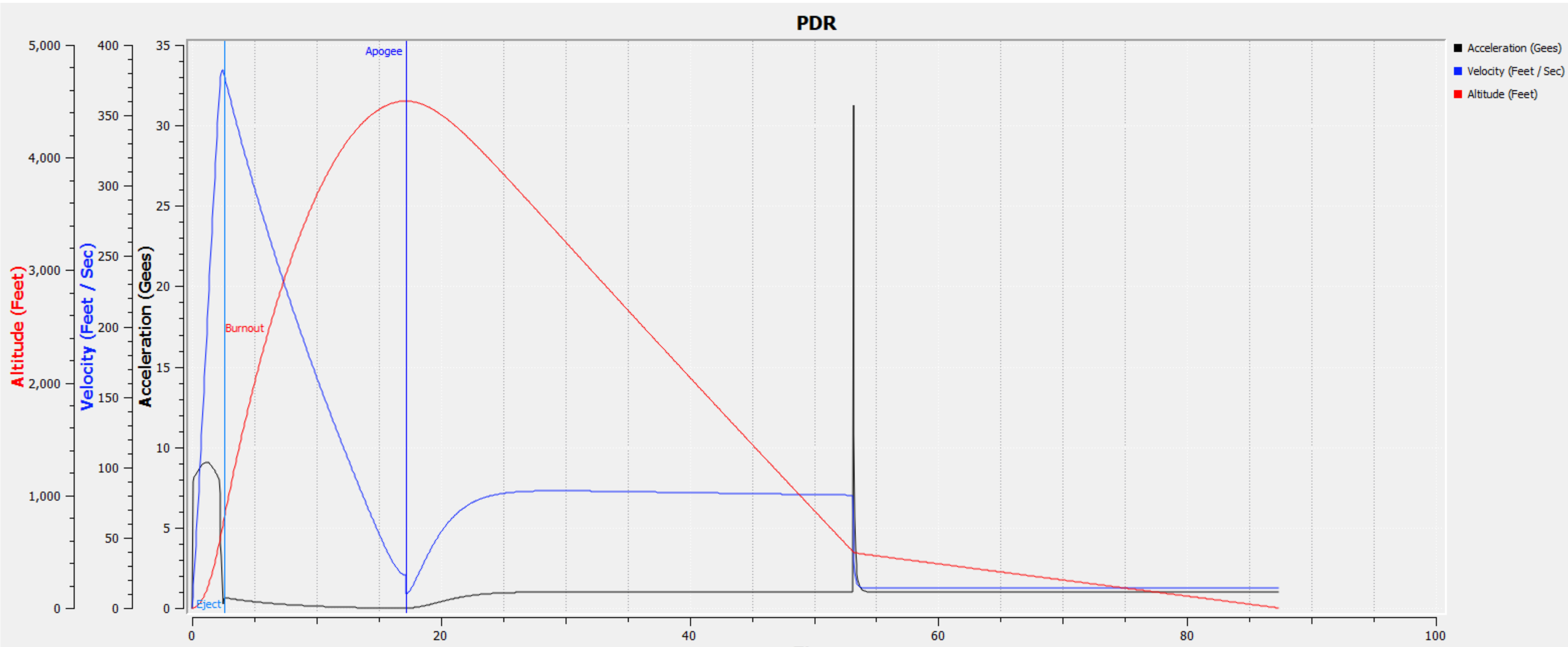

Figure 48: RockSim flight profile of the launch vehicle using an L1520T motor.

According to the flight profile in Fig. (48) above, we can see that the launch vehicle achieves a maximum velocity of 385 ft/s. The maximum projected acceleration undergone by the launch vehicle during take off is 9 Gs.

3.5.3 RockSim Apogee Study

One of the largest factor to impact the expected apogee is the the potential variation in the payload weight. Changing the weight of the payload by even as little as 1 lb. impacts the apogee of the launch vehicle by several hundred feet. This phenomenon is analyzed in Fig. (49) where the designed mass of the payload is illustrated in green and deviation from 6 lb. target weight transitions into the red.

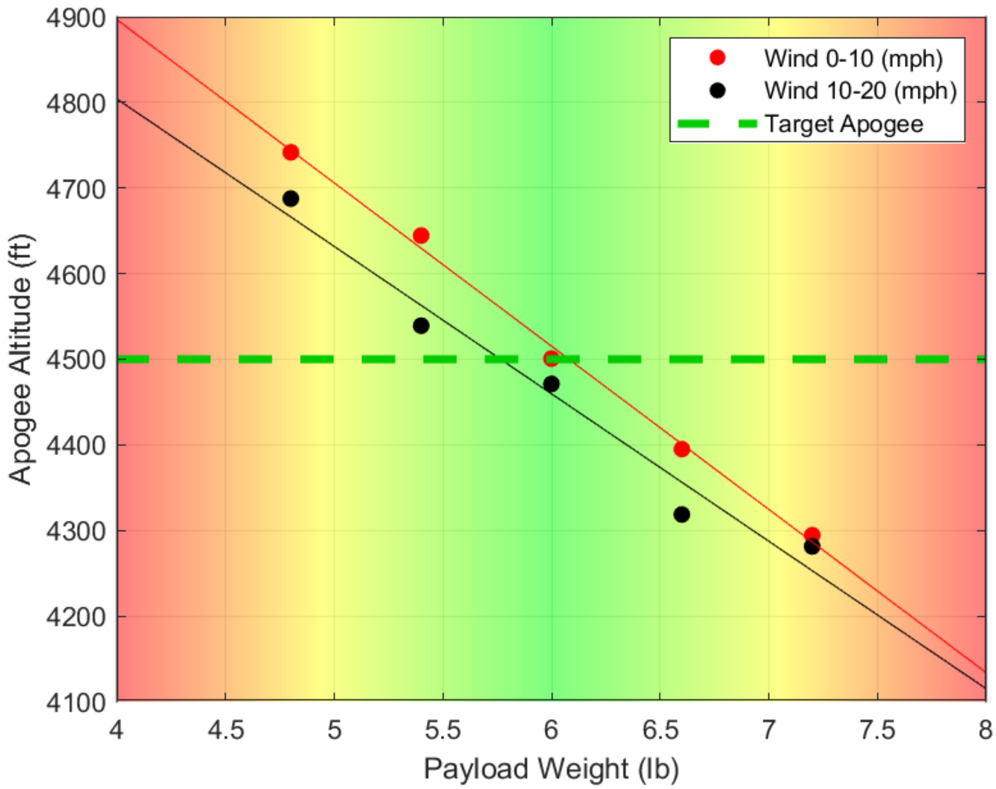

Figure 49: Simulation of altitude variation with payload weight.

The launch vehicle reaches the expected apogee of 4500 ft. at a payload weight of 6 lb. Above and below this value the apogee begins to vary linearly with weight. Additionally, the weight of the payload will affect the overall CG of the launch vehicle. In turn, this could potentially impact the stability. Thus, it is important to accurately estimate the payload's weight. Currently, the expected payload weight is 6 lb. to maintain stability and reach the target apogee. Further details on payload weight are given in Section 4.5 and the impact on stability in Section 3.5.5 below.

Smaller, but compounding, weight discrepancies in multiple sections of the vehicle can influence the apogee of the launch vehicle by hundreds of feet for less than a 10% variability of the total weight. Fig. 50 graphically shows the variation from the expected total weight of 43.3 lb. relationship with apogee.

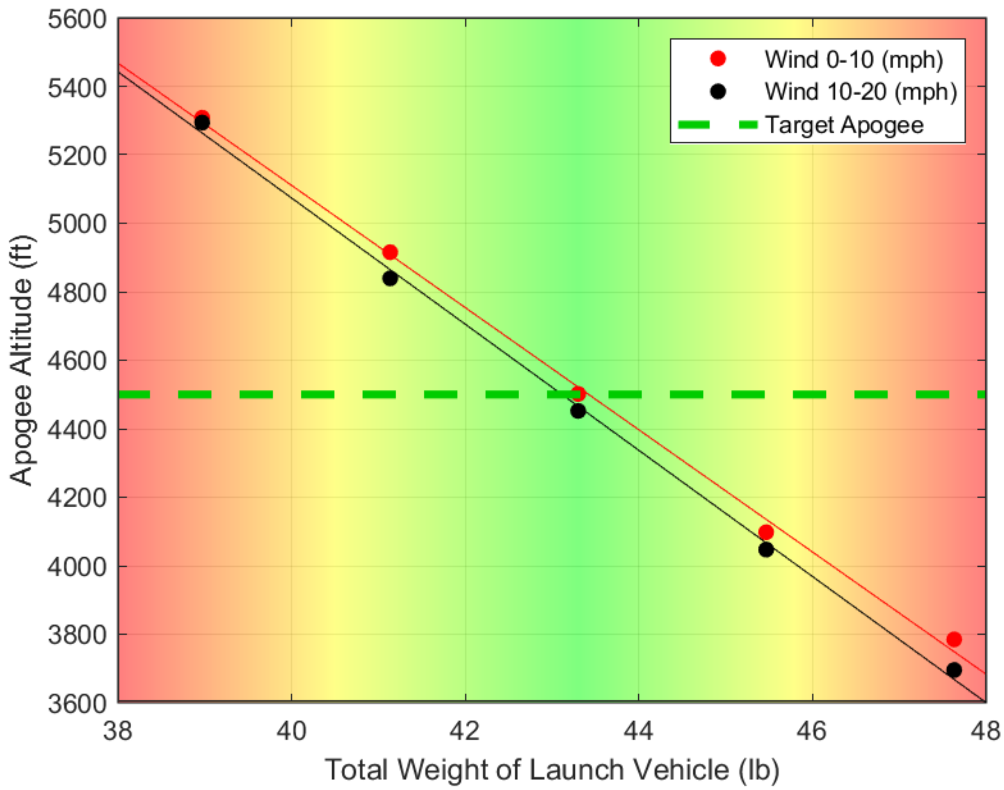

Figure 50: RockSim simulation of the relationship between payload total weight and apogee altitude.

The leading design includes 2 lb. of ballast detailed in Section 3.5.7 that will allow for the total weight of the launch vehicle to be reduced in the event of an over weight situation. Similarly, additional ballast can be added to predetermined points on the vehicle for the tuning of the total weight in an under weight situation.

3.5.4 Apogee Verification Calculations

A calculation of expected apogee was performed below using Eqns. 10 through 16 to validate RockSim simulation results. Table (15) contains necessary values for the hand calculations and were taken from the geometry of the launch vehicle, L1520T motor reference [3] data, and standard environmental constants.

Table 15: Values used to algebraically solve for apogee

Quantity	Variable	Value	Units
Mass	M	19.6	kg
Frontal Area	A	0.01936	m^2
Gravitational Acceleration	g	9.81	m/s^2
Total Impulse	I	3715	N·m
Average Thrust	T	1567	N
Burn Time	t	2.4	s
Air Density	ρ	1.225	kg/m^3
Drag Coefficient	C_D	0.2469	N/A

To start the hand calculations the aerodynamic drag is the calculated in equation (10) used in calculating q and x.

$$k = \frac{1}{2}\rho C_D A = 0.002928\frac{kg}{m} \tag{10}$$

$$q = \sqrt{\frac{T - Mg}{k}} = 685.3\frac{m^2}{s^2} \tag{11}$$

$$x = \frac{2kq}{M} = 0.2047\frac{m}{s^2} \tag{12}$$

The maximum velocity can now be calculated using the results of Eqns. (11) and (12).

$$v_{max} = q\frac{1 - e^{-xt}}{1 + e^{-xt}} = 165.1\frac{m}{s} \tag{13}$$

Maximum velocity will be at the point where the motor stops producing thrust. This height is calculated using equation (14).

$$h_{boost} = -\frac{M}{2k}\ \text{ln}\ \frac{T - Mg - kv^2}{T - Mg} = 200m \tag{14}$$

Gravity and drag then slow the launch vehicle down during the coast phase of the flight and the height gained is calculated using equation (15).

$$h_{coast} = \frac{M}{2k}\ \text{ln}\ \frac{Mg + kv^2}{Mg} = 1161m \tag{15}$$

Finally, by summing the calculated altitudes, the apogee was calculated:

$$h_{total} = h_{boost} + h_{coast} = 1361m \rightarrow 4455ft \tag{16}$$

This apogee is now compared to the altitude given by RockSim:

Table 16: Apogee comparison between RockSim and hand calculations.

Method	Result	Comparison
RockSim	4500	$\%_{diff} = 1.005\%$
Algebraic	4455	

These two calculation methods are strikingly similar despite the variety of factors influencing the RockSim's analysis. The 1% difference in apogee from the two methods results from RockSim's wind shear and turbulence simulations. However, real world factors will produce far more variability in the flight than the difference between the simulations calculated.

3.5.5 RockSim Stability Margin Study

The current launch vehicle design has a stability margin of 2.15 calipers before launch and a rail exit stability margin of 2.22 calipers. Stability margin is calculated by dividing the difference in location of the center of pressure and center of gravity by the outer diameter of the launch vehicle. Stability during the duration of the ascent is showing in Fig. (51).

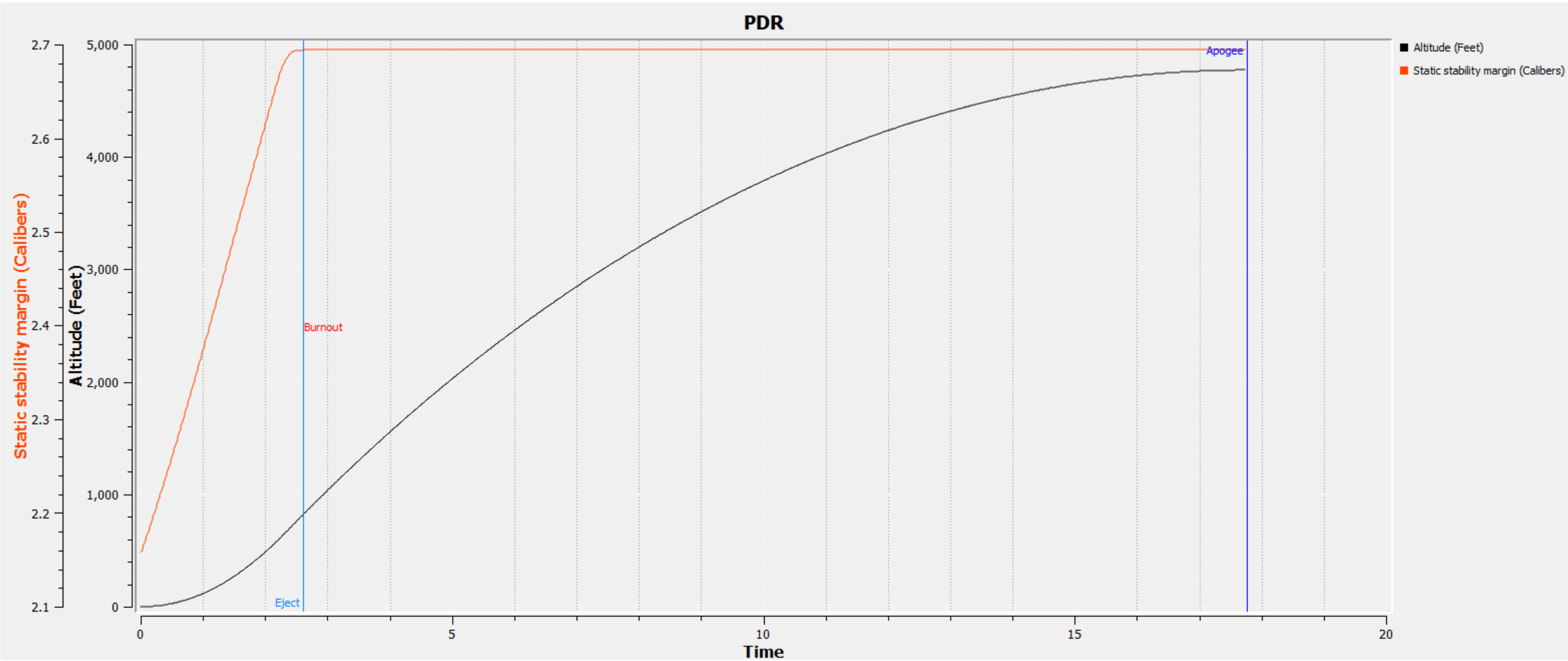

Figure 51: RockSim simulation of stability margin variation during flight.

As the motor burns, the center of gravity moves forward. After motor burnout, the CG and stability margin remain constant. It is also important to consider how the stability of the launch vehicle changes with a variable payload mass. Varying both overall mass of the payload as well as its location within the payload bay will change the CG and by extension the stability. An analysis of these changes is shown in Fig. (52) below.

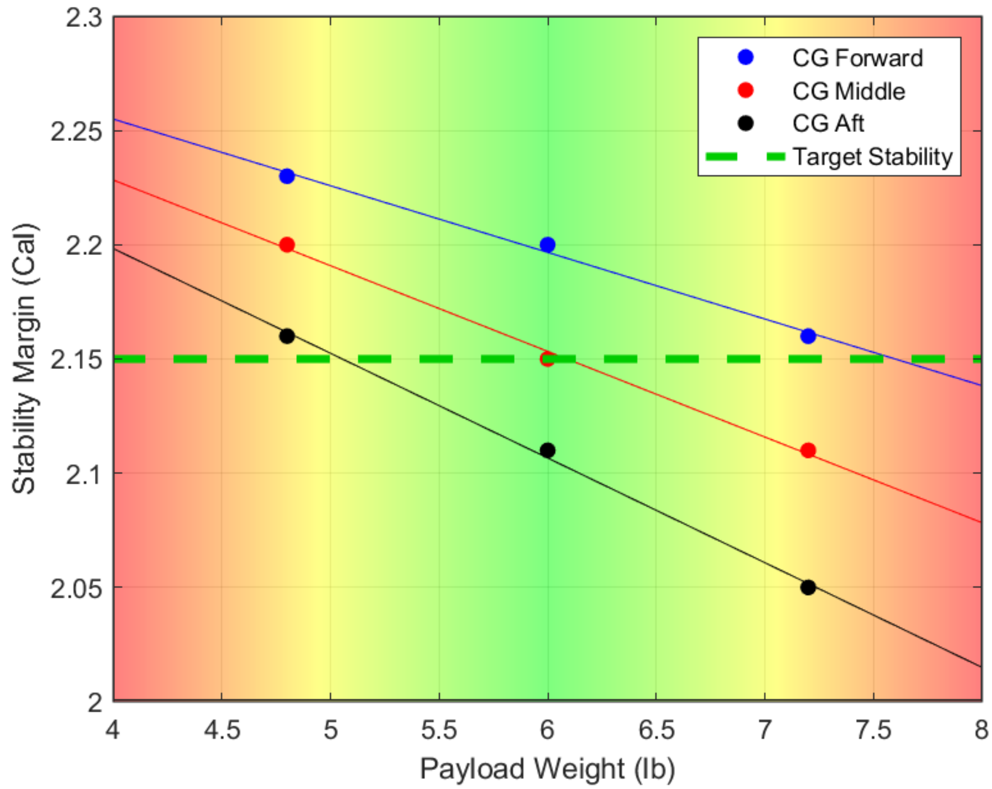

Figure 52: RockSim simulation of the impact of payload location and weight on launch vehicle stability.

This Fig. 52 shows that increasing payload mass will decrease the stability. Therefore, it is essential that payload mass be minimized. If the payload mass must exceed the 6 lb. estimation the launch vehicle stability can be improved by moving the payload center of gravity as far forward as possible.

3.5.6 Stability Margin Calculation

RockSim is known to generate accurate CG information but CP calculation is a more complex process. To verify the stability margin is reasonable, the CP of the launch vehicle is calculated using an additional method than RockSim. By splitting the launch vehicle into two aerodynamic shapes, the launch vehicle CP can be calculated using Barrowman's method [10]. Ogive nose cones have an understood pressure coefficient of 2 with an arm that is a function of the nose cone length as shown in Eqn. 17. Barrowman's equations utilize standard trapezoidal fin geometries which are not compatible with the the ogive geometry chosen for the launch vehicle. To circumvent the incompatibility, a modified version of the ogive fin is constructed as shown below in Fig. 53.

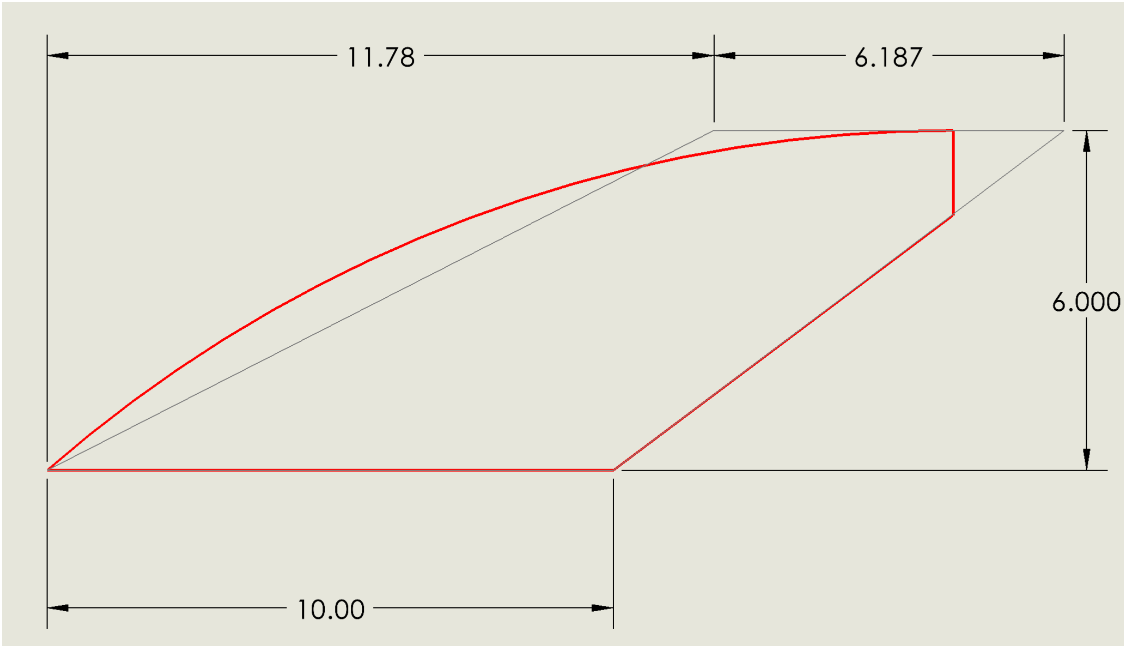

Figure 53: Annotated modification to the ogive fin for Barrowman's equations.

The following Table (17) contains the variable names and values for the stability equations detailed below.

Table 17: Measured values for stability calculations.

Variable	Description	Value	Units
C_N	Nose Cone Coefficient	2	N/A
L_N	Length of Nose Cone	30	Inches
R	Radius	3.085	Inches
S	Fin Semi-Span Length	6	Inches
C_R	Fin Root Chord	10	Inches
C_T	Fin Tip Chord	6.19	Inches
N	Number of Fins	4	N/A
X_B	Nose Cone Tip to Root Chord LE	88	Inches
X_R	Fin Sweep from Root Chord LE to Tip LE	11.78	Inches
CG	Center of Gravity from RockSim	61.2	Inches

$$X_n = 0.466 * L_N \tag{17}$$

Finding the fin coefficient and arm length is more complex utilizing the basic fin geometry Eqns. 18 and 19.

$$\theta = 90^\circ - tan^{-1}(\frac{S}{X_R}) = 63.01^\circ \tag{18}$$

$$L_F = \sqrt{S^2 + (\frac{1}{2}C_T - \frac{1}{2}C_R + \frac{S}{tan(\theta)})^2} = 6.109in. \tag{19}$$

Eqn. 20 and 21 are calculating the center of pressure coefficient and its moment arm due to the fin geometry.

$$C_F = \left[1 + \frac{R}{S+R}\right]\left[\frac{4N(\frac{S}{2R})^2}{1+\sqrt{1+(\frac{2L_F}{C_R+C_T})^2}}\right] = 8.997 \quad (20)$$

$$X_F = X_B + \frac{X_R}{3}\frac{C_R+2C_T}{C_R+C_T} + \frac{1}{6}\left[(C_R+C_T-\frac{C_RC_T}{C_R+C_T})\right] = 91.89in. \quad (21)$$

A weighted average of the aerodynamic moment arm from the nose cone and fin geometries is calculated in Eqn. 22.

$$X_{CP} = \frac{C_NX_N + C_FX_F}{C_N+C_F} = 77.71in. \quad (22)$$

This simplified CP calculation is then used to calculate the stability of the launch vehicle using Eqn. 23.

$$\frac{X_{CP}-X_{CG}}{2R} = 2.67cal \quad (23)$$

A calculated CP of 77.71 in. is different than the 74.9 in. CP RockSim is generating from Section 3.3.1. This creates a significantly different stability margin shown in Table (18).

Table 18: Calculated stability values.

Variable	Description	Value	Unit
θ	Sweep Angle	63.01	Degrees
L_F	Middle Chord Length Line	6.109	Inches
C_F	Fin Coefficient	8.997	N/A
X_F	Fin Arm Length	91.89	Inches
X_{CP}	CP Location	77.71	Inches

Table 19: Stability margin comparison between RockSim and hand calculations.

Method	Result	Comparison
RockSim	2.15 calibers	$\%_{diff} = 21.58\%$
Barrowman's Method	2.67 calibers	

Hand calculations only utilizing the nose cone and approximate fin geometry were understandably significantly different than RockSim. The modification of the fin geometry to fit the Barrowman's equations is a significant potential source of error. Rocksim, unlike the Barrowman's method, also factors the camera housing geometry into its center of pressure calculations. A stability margin of 2.67 calibers is an over stable condition that is not inherently detrimental to the flight vehicle but may result in unfavorable weather cocking. The actual stability of the launch vehicle is confirmed to be reasonable and is likely much closer to the RockSim calculations than the Barrowman's method.

3.5.7 Ballast Placement for Stability and Altitude Tuning

The launch vehicle design considers ballast placement in three locations for efficient movement of the CG and total weight of the vehicle. To move the CG forward with the least amount of weight, the ballast is placed as far forward as possible. Using epoxy, 1 lb. of ballast is permanently installed in the inside tip of the nose cone. The final mass of the launch vehicle and its exact CG will likely vary sightly from the current design. For the purpose of maintaining the desired stability margin, an additional 1 lb. of ballast is designed into the forward side of the nose cone bulkhead using a removable system. If the CG needs to move forward, more weight can be added to this location. Similarly, if the CG needs to move aft to meet the desired stability margin, a portion of the 1 lb. can be easily removed to accomplish this.

To tune the apogee of the launch vehicle, the overall weight of the launch vehicle can be adjusted. To increase the weight of the vehicle without impacting the stability of the vehicle, ballast is added to either side of the CG. Forward of the CG, ballast is added in the previously described fin can bulkhead. Aft of the CG, ballast is secured inside of the fin to the aft centering ring. Utilizing the removable fin design, this ballast location is easily accessed and modifiable.

3.5.8 Kinetic Energy Landing

The landing kinetic energy of the launch vehicle is calculated using the following kinetic energy equation derived in Newtonian mechanics.

$$E = \frac{1}{2}mV^2 \tag{24}$$

The maximum impact energy allowed by NASA requirement 3.3 is 75 ft-lbf with additional points being awarded for being below 65 ft-lbf. Using this equation we can calculate the maximum impact velocity that each section of the launch vehicle must impact the ground at under main parachute descent, shown in Table (20) below. The descent velocity used is the descent velocity calculated for the current leading main parachute option stated in Section 3.4.3.

Table 20: Impact velocity necessary to meet NASA requirements by launch vehicle section

Section	Mass of Section	Descent Velocity Necessary to be Awarded Points	Descent Velocity Necessary to be Awarded Bonus Points
Nose Cone	.207 slugs	26.92 ft/s	25.06 ft/s
Main Parachute Bay and Avionics Bay	.220 slugs	26.11 ft/s	24.31 ft/s
Drogue Parachute Bay, Payload Bay, and Fin Can	.632 slugs	15.4 ft/s	14.34 ft/s

As shown above, the fin can, payload bay, and drogue bay section is the heaviest section and is the determining factor of the maximum impact velocity. This will also be the lowest hanging section of the launch vehicle under main descent. Based on the above calculations, the Fruity Chutes 120” Iris UltraCompact parachute has been selected as the main parachute that will be used in competition as described in Section 3.4.3. Using Eqn. 8 to calculate the descent rate, and subsequently using Eqn. 24 to calculate 3.5.8 each section’s kinetic energy, the impact energies of each section under main parachute are shown in the Table (21) below.

Table 21: Impact kinetic energy under main parachute by launch vehicle section

Section	Mass of Section	Velocity Under Main Parachute	Impact Energy
Nose Cone	.207 slugs	6.11 ft/s	4.247 ft-lbf
Main Parachute Bay And Avionics Bay	.220 slugs	9.16 ft/s	9.230 ft-lbf
Drogue Parachute Bay, Payload Bay, and Fin Can	.6332 slugs	13.79 ft/s	60.206 ft-lbf

The velocity of each section under main parachute was calculated by considering the weight pulling on the parachute at each point before the next section makes impact with the ground; The velocity of the second section to impact the ground used the weights of the first and second sections as well as the recovery hardware (ie shock cord, Nomex cloth, and quick links) pulling down on the main parachute using Eqn. 8. The subsequent kinetic energies were again calculated using Eqn. 24. As shown in Table (21), all 3 section impacts are under the 65 ft-lbf requirement set by NASA.

3.5.8.1 Kinetic Energy Alternative Calculation Methods

Using Apogee Rockets' RockSim simulation, the impact velocity under main parachute is estimated to be 14.25 ft/s. Once again using equation 24 to estimate the impact energy, we find that the impact energy on the lower section of the rocket is 64.16 ft-lbf. This just barely allows the bonus points to be awarded for getting the impact energy below 65 ft-lbf.

3.5.9 Expected Descent Time

Expected descent times are calculated by first assuming that the drogue parachute immediately deploys at apogee and the entire launch vehicle starts descending at the drogue descent rate stated in Section 3.4.2.6. Likewise it is assumed that the main parachute also deploys at the deployment height and the launch vehicle immediately slows to the descent velocity under main as stated in Section 3.4.2.7. Assuming these two heights, the following equations is used based on Newtonian kinematics.

$$t = \frac{h_a - h_m}{v_d} + \frac{h_m}{v_m} \tag{25}$$

Where h_a is the apogee altitude, h_m is the main deployment altitude, v_d is the velocity under drogue, and v_m is the velocity under main. Total descent times of the launch vehicle under drogue and under main are stated in Tables (11) and (12). For the selected drogue parachute and main parachute the total descent time is estimated to be 75.7 seconds, well within the 80 second limit described in NASA requirement 3.11.

3.5.10 Expected Drift

Expected Drift calculations are made by assuming that the launch vehicle drift speed is equal to the wind speed at a specific altitude. This is a large overestimation because the only forces experienced by the launch vehicle under drogue and main descent are the force imparted on the vehicle by wind, the drag caused by the parachute, and the force of gravity. Therefore calculating the expected wind drift using this velocity will lead to a much lower total wind drift experienced during competition flight. The wind drift experienced by the launch vehicle is calculated using the following equation were v_w is the maximum wind speed allowed for competition flight, t is the estimated descent time, and D is the expected drift.

$$D = v_w t \tag{26}$$

The total wind drift is calculated by adding the wind drift distance under drogue to the wind drift distance under main parachute. The results are tabulated below.

Table 22: Wind drift distances for different wind velocities

Wind Velocity	Drift Distance
0 mph	0
5 mph	552
10 mph	1104
15 mph	1656
20 mph	2208

Using this equation the expected wind drift experienced by the launch vehicle under maximum wind conditions allowable by NASA of 20 mph is 2208 feet, well under the 2500 feet limit defined in NASA requirement 3.10.

3.5.11 Alternative Descent time and Wind Drift Calculations

Using a Rocksim simulation of the launch vehicle's flight path, an alternate descent time is estimated to be 87.43 seconds. This calculation accounts for the time each parachute takes to unfurl as well and therefore the amount of drag generated as it does so. This results in a much smoother curve for descent time as shown below.

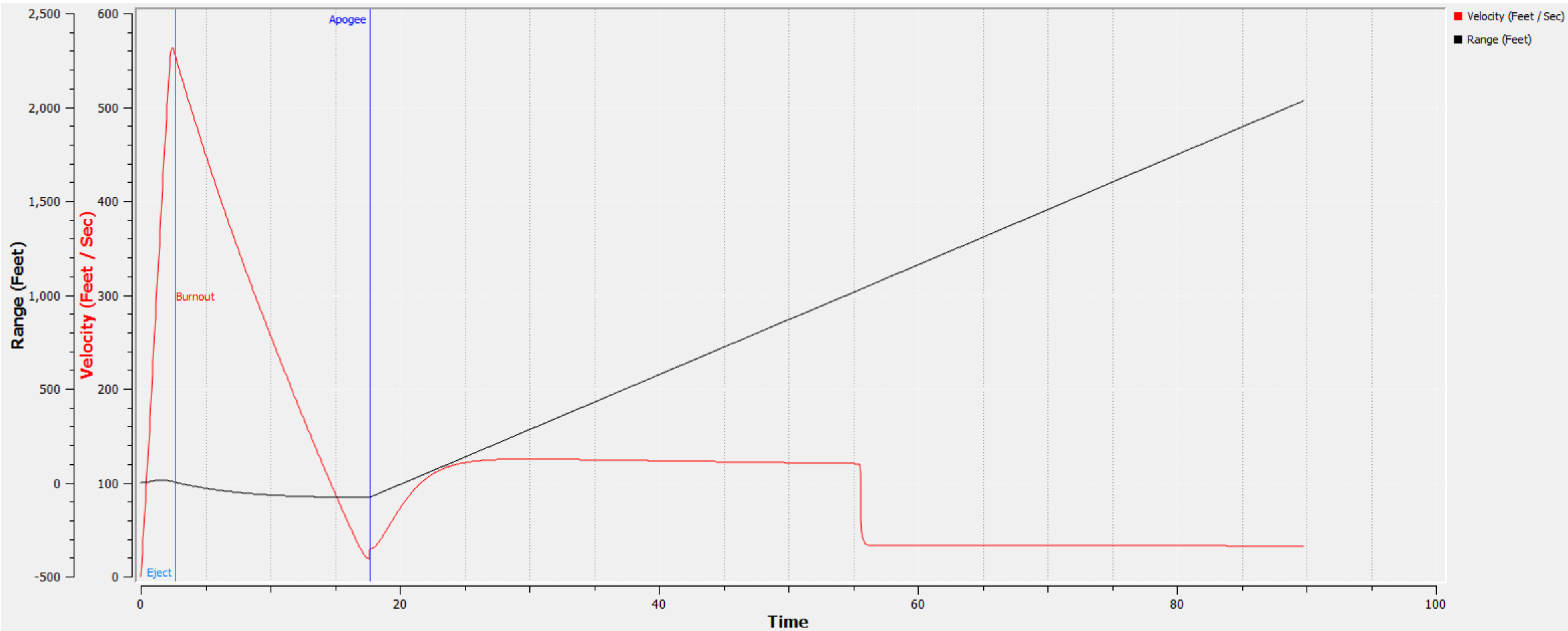

Figure 54: Altitude, range, and descent time simulated in RockSim.

Using RockSim again to estimate total wind drift calculations, an estimated range of 2029 feet was calculated. The main difference between the hand calculations for drift distances and the RockSim analysis is that the RockSim analysis accounts for the additional time of descent, as well as not analyzing wind as a constant force and uses simulated wind thermal simulations. This results in an extended drift time and therefore affects the drift distance, although it does not make the same assumption that the drift velocity is equal to the wind velocity. However, both calculations result in the total drift distance being below the 2500 foot requirement set by NASA.

3.5.12 Parachute Opening Shock Calculations

The forces experienced on the launch vehicle during parachute deployment are amongst the highest experienced by the launch vehicle. This force is calculated by first calculating the time the parachute takes to open, and then using that time as the change of time it takes for the vehicle to change velocities, as shown in the following two equations where r is the radius of the parachute that is opening, v is the drogue descent velocity, m is the mass of the launch vehicle after burnout without the main parachute, Δv is the difference between drogue descent velocity and main descent velocity, and F is the total shock felt by the launch vehicle.

$$t = \frac{8r}{v} \tag{27}$$

$$F = \frac{m\Delta v}{t} \tag{28}$$

The coefficient of 8 in Eqn. (27) is used based on a study by W. Ludtke that elaborated on how to calculate the opening shock forces for parachutes [14]. Using these two formulas, the descent rates of 111.6 ft/s, and the main parachute radius of 5 ft calculated for the launch vehicle using the leading drogue and main parachutes in Table (11) and (12), the time the main parachute takes to unfurl is .356 seconds and the maximum shock experienced by the vehicle is estimated to be 316.73 ft-lb.

4 Payload Criteria

4.1 Payload Objective

The objective of this payload mission is to design and construct a camera system housed inside of the Launch Vehicle. This camera system will be able to receive commands over APRS, decode and interpret those commands, and perform the series of actions the commands describe. The camera should be level with the ground and be able to rotate 360° about an axis normal to the ground. After photos are taken, the on board computer will add a time stamp, and perform any image editing commanded, including image rotation, filter application and removal, and color removal and reapplication. The camera system will then store these images on-board.

The payload system is the Surrounding Optics and Communication System (SOCS). SOCS receives commands using an RTL-SDR dongle adapter, interprets those commands inside the on-board Raspberry Pi, and transmits those commands to one of four servos attached to a camera unit mount. The servo of choice will then rotate the camera and capture images as the RAFCO communication delineates. These images are then sent back to the Pi, edited, and timestamped.

4.2 Payload Success Criteria

The Surrounding Optics and Communication System (SOCS) is the primary payload design for the 2023 competition year. Success criteria for this system are described in Table (23) below.

Table 23: Payload success criteria

Success Level	Payload Aspect	Safety Aspect
Complete Success	SOCS autonomously receives RF signal and performs the specified series of tasks using the chosen onboard camera.	No one is harmed or injured during the execution of payload operations.
Partial Success	SOCS receives RF commands, but does not do the correct tasks in the necessary order. OR SOCS communicates the tasks to the wrong camera.	There are one or more close calls during the execution of payload operation that puts individuals at risk, although no one is harmed or injured.
Partial Failure	SOCS receives RF command, and omits two or more of the tasks.	Individuals incur minor injuries during the execution of payload operation.
Complete Failure	SOCS does not receive RF commands and does not do its subsequent tasks.	There are major injuries during the execution of payload operation.

4.3 Alternative Payload Design

Design alternatives for the payload system are discussed in the following sections. While each component has individual criterion to be evaluated under, in general, the selected alternative should have high reliability, ease of integration into the rest of the payload system, and availability or ease of manufacturing.

4.3.1 Camera Mounting

Four cameras, one for each fin, must be mounted or secured to the rest of the payload system. Several methods are considered below to secure cameras, allow them to move as necessary, and transmit data successfully between cameras, servos, and the on-board Pi.

4.3.1.1 Hard-Mount

In this mounting method, four transparent camera housings, one for each camera, will be placed over the cameras to shield them from debris and aerodynamic forces. A 3D model of this configuration can be seen in Figure (56) below. One camera sits in between each of the four fins and is supported by a servo that controls its rotation. This camera-servo assembly is attached to a camera unit mount, shown in Figure (55), which is bolted or riveted to the inside of the payload bay immediately prior to flight.

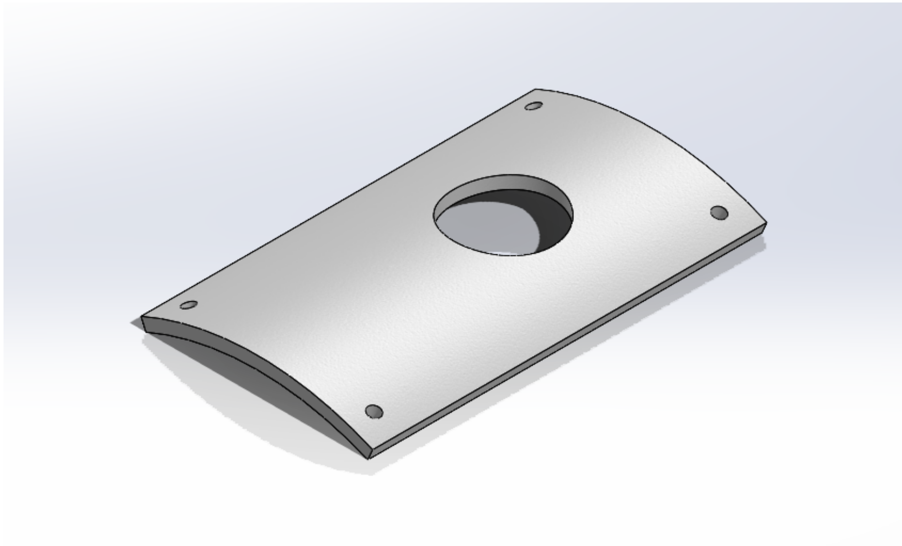

Figure 55: CAD model of the camera unit mounting plate.

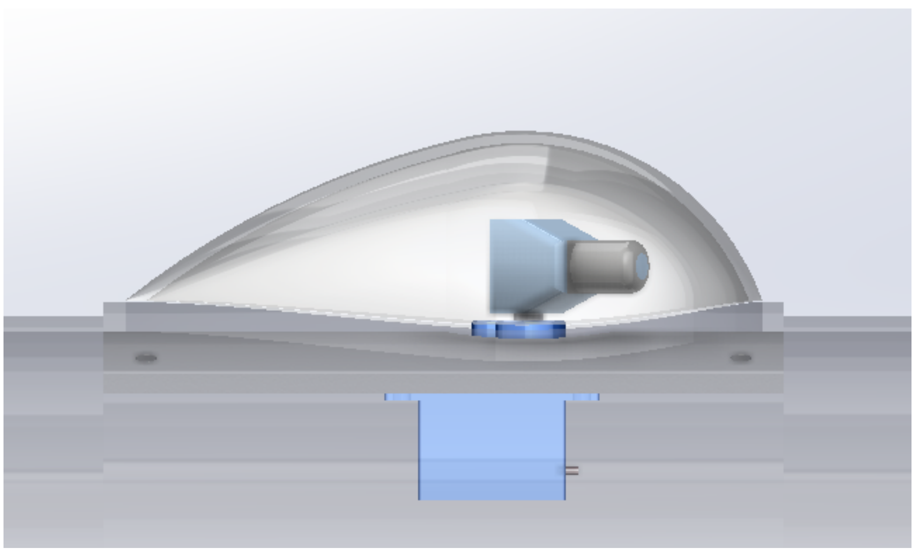

Figure 56: CAD model of the hardware mounting schematic.

An advantage of this method is its simplicity. This method requires no moving parts during or after flight, is easily moved on and off of the vehicle, and is replaceable in the event that pieces of the camera unit mounts shear or wear down from in-flight forces. It is easy to manufacture via 3D printing, and can be secured tightly to the airframe. Next, because the cameras do not translate, cables providing power to each servo and camera are shorter and cable management is simpler.

This method has several drawbacks, the first of which being structural instability. Given that the camera unit mounts will likely be 3D printed using PLA plastic, strong forces could crumple layers of 3D printed filament. Additionally, If more than two bolted connections become unstable during flight, the camera and servo attached to the compromised mount will be subject to rotational and vibrational motion that could compromise image quality and system reusability.

4.3.1.2 Camera Extension

An extending arm is proposed to achieve payload success in this alternative. In this mounting method, a hole or gate is cut into the vehicle, and an arm deploys one of four cameras from the inside of the airframe. This arm is powered by one or more motors that convert rotational motion into linear actuation. Once the camera and the servo controlling its rotation have been fully deployed by the arm, the camera's servo can be rotated and an image can be captured.

This design allows the camera of choice to be unobstructed by fins, airframe sections, and recovery hardware that may otherwise be in an image captured by a camera closer to the airframe. Giving the camera a large distance between itself and the ground gives a better view of the surrounding field and, thus, a clearer set of final pictures.

However, this method is costly and complicated. Firstly, motors with the accuracy and power required to maneuver such a system are expensive and troublesome. Stepper motors take up too much space, small servo motors are not powerful enough to push the amount of weight comprising an arm, and DC motors are too heavy to fit inside of the vehicle without disrupting the vehicle's CG. Secondly, this system is complex and has many unique points of failure. A long lever arm leaves the camera arm vulnerable to snaps and breaks from as little as a gust of wind on the field. Lastly, housing camera arms inside of the vehicle prior to landing leaves little to no room for essential payload electronics.

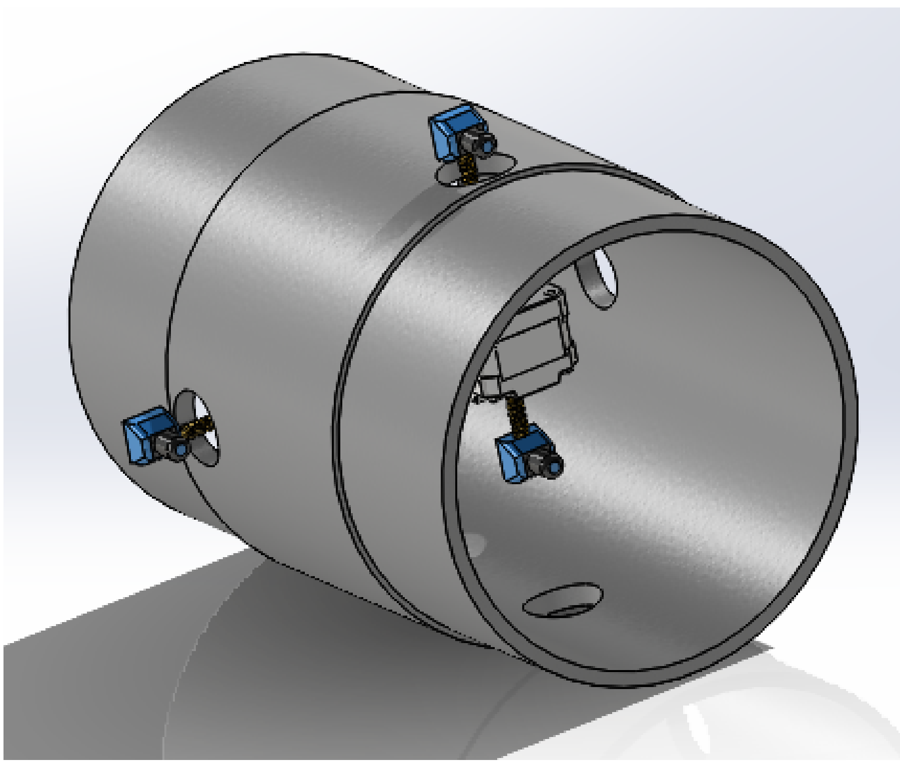

Figure 57: CAD model of the camera extension design.

4.3.2 Camera Housing

Cameras must be housed during flight to protect them from debris, dislocation, and power loss prior to receipt of RAFCO commands. Three options are considered to protect cameras from harm during flight and secure other important payload components.

4.3.2.1 Protruding Camera Housing

A protruding camera housing is defined as a round, transparent dome that sits on top of the launch vehicle and protects payload cameras and electronics from debris and dislocation during flight. Cameras sit fixed inside of the housings and rotate via a servo when called to do so by RAFCO commands. Housings allow cameras to have a clear view of the launch field upon landing, without the need for camera translation out of the vehicle. Thus, a design utilizing protruding camera housings is simpler and has fewer points of failure than a non-fixed camera system. Additionally, protruding housings do an excellent job of shielding cameras from dirt and debris and can be made to be removable in the case that they are scratched or scraped.

This is also a drawback of protruding housings, as any scratch or scrape on their surface will influence image quality. If the fin can rolls or slides, each of the four transparent camera housings have the potential to be cracked or scratched, decreasing the quality of captured images and potentially damaging the cameras. Housings can be difficult to manufacture, especially if they have a complex shape. These complex shapes may be necessary to reduce drag, which is a factor to consider in any protuberance from the airframe.

These housings can come in a variety of shapes and sizes which each provide unique benefits and drawbacks. These pros and cons are discussed in the sections below and in Table (27) via a Pugh Matrix feasibility study.

4.3.2.1.1 Dome-Shaped Housing

A CAD model of a simple dome-shaped housing design can be seen in Figure (58) below. This design can be easily manufactured via vacuum forming or purchased as a COTS component. The dome-shaped housing design is easily procurable and interfaces well with the airframe. Additionally, it does not require any abnormally shaped holes to be drilled into the airframe.

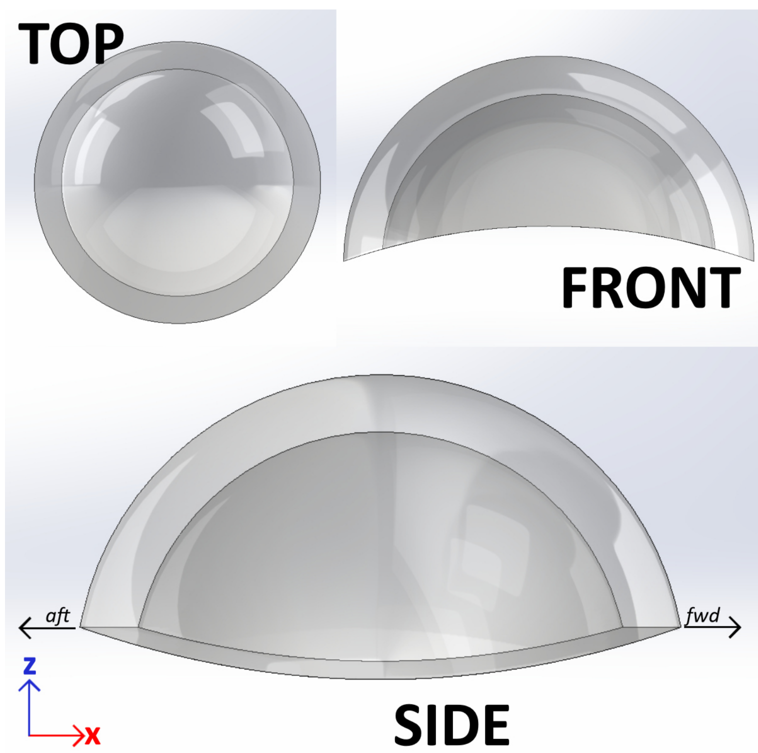

Figure 58: CAD model of a dome-shaped camera housing.

One major drawback of this housing design is its contribution to overall vehicle drag. ANSYS simulations, shown in Figure (59) below, showed that a 2D cross-section of this design would produce 7 N of drag per housing on the full-scale vehicle. This leads to an apogee decrease of up to 200 feet.

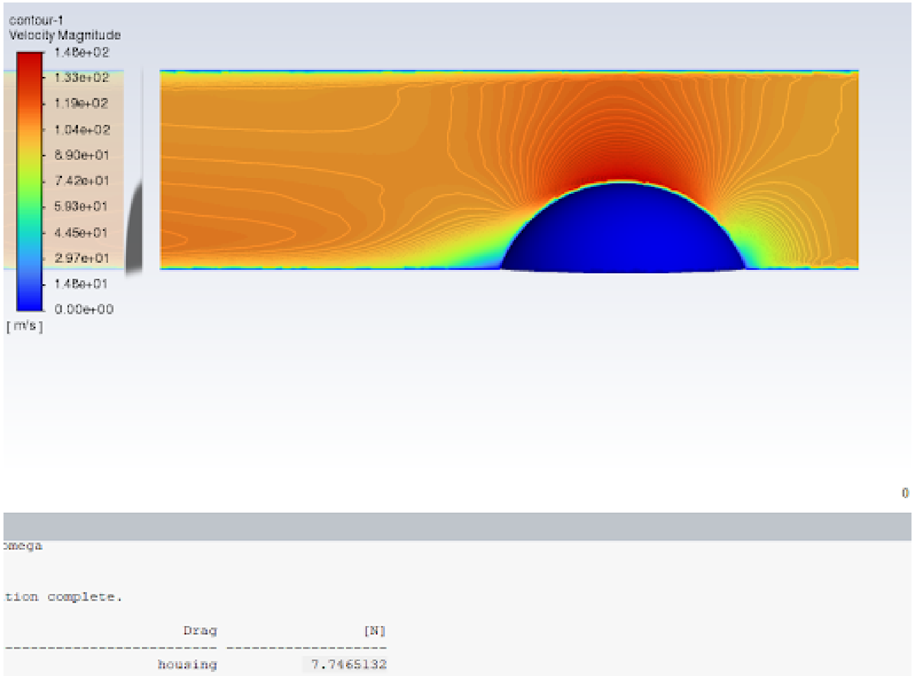

Figure 59: ANSYS simulation of the dome-shaped camera housing.

4.3.2.1.2 Teardrop-Shaped Housing

A CAD model of a simple teardrop-shaped camera housing design can be seen in Figure (60) below. This design is more aerodynamically favorable than the dome design, as its tapered TE decreases wake and turbulent vortices. Its cross section, seen in the side view of the below figure, closely resembles that of an airfoil, known to reduce drag.

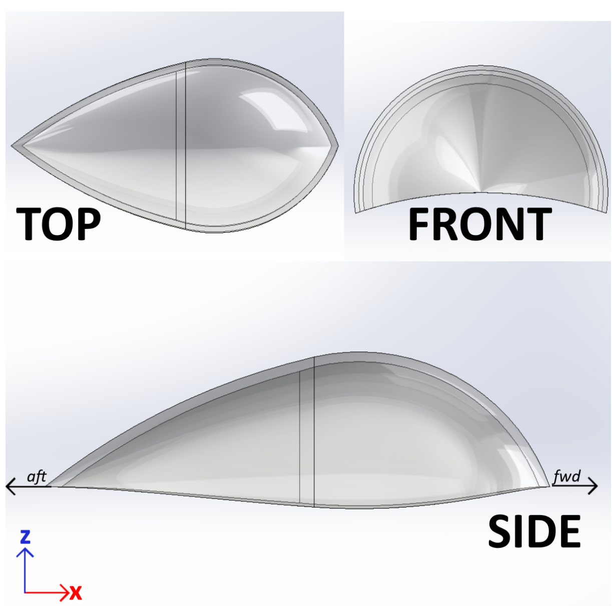

Figure 60: CAD model of a teardrop-shaped camera housing.

From the ANSYS simulation below in Figure (61), it can be seen that the drag per housing is drastically lower than that of the dome-shaped housing. These teardrop-shaped housings produce only 0.6 N of drag per housing, and only a marginal impact on expected apogee.

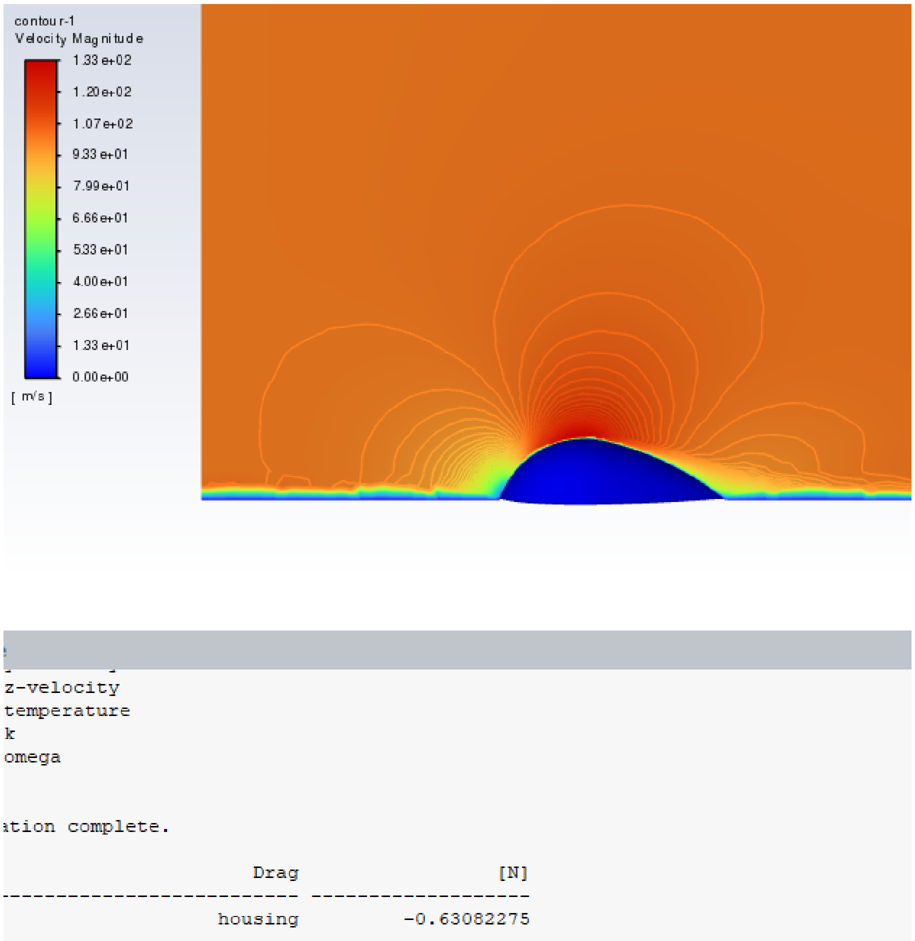

Figure 61: ANSYS drag simulation of the teardrop-shaped camera housing.

The thicker these complex camera housings are, the more distorted the view from each payload camera will be. Post processing in MATLAB using MATLAB's undistortImage() function can be performed on distorted images, and thinner-walled housings can be used to reduce overall distortion.

This design's complex shape renders it difficult to manufacture, requiring specialty 3D printing equipment or vacuum forming to ensure transparency and structural integrity. 3D printing ensures that the camera housing structure can withstand compressive forces, but parts printed with transparent resin must undergo significant post-processing to be rendered fully transparent. Alternatively, sheets of PETG plastic can be vacuum-formed to 3D-printed molds in a much simpler process yielding more transparent camera housings. However, vacuum forming requires thin sheets of plastic which do not resist compressive forces as well as 3D printed resin do.

4.3.2.2 Hole

Two alternatives to a protruding camera housing design are discussed. The first is a simple rectangular-shaped hole in the airframe through which one of four cameras will peek through after the vehicle lands and the payload system begins receiving RAFCO commands. This design is strongly considered for its aerodynamic favorability, as four holes produce less drag than four proturbing camera housings. However, this design is more technically complicated as it requires the use of rotating **and** translating parts. The camera must first be extended from the launch vehicle via the design discussed in section 4.3.1.2, righted normal to the horizon using an IMU and servo, and then rotated according to RAFCO commands received by the payload and interpreted by the APRS decoding software. Figure 62 below shows a CAD model of this design.

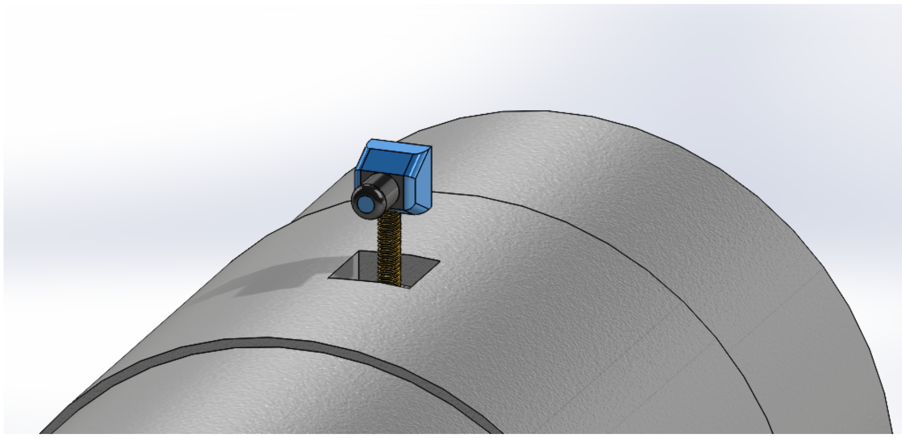

Figure 62: CAD model of the hole design.

In addition to requiring complicated and costly equipment to function properly, this design would offer no protection against the elements. Dirt, dust, and other airborne particulates would have free access to cameras and other payload electronics. Irrigation ditches present on the Bayboro, NC launch field contain water that, in this design, would easily get inside of the vehicle and cause significant damage to the payload electronics.

4.3.2.3 Gate

A gate-based design follows the same principles as the hole design described above. A hinging door remains closed over a hole drilled in the airframe and protects internal components during flight. After landing, this gate is either pushed open by payload components or actuated using a latch. This design offers advantages not found in the hole design, including protecting components from debris during flight and offering even more aerodynamic advantage. However, this design also has disadvantages, such as the need to actuate or push a latched gate poses a failure risk, because if the latch fails closed, no images can be taken.

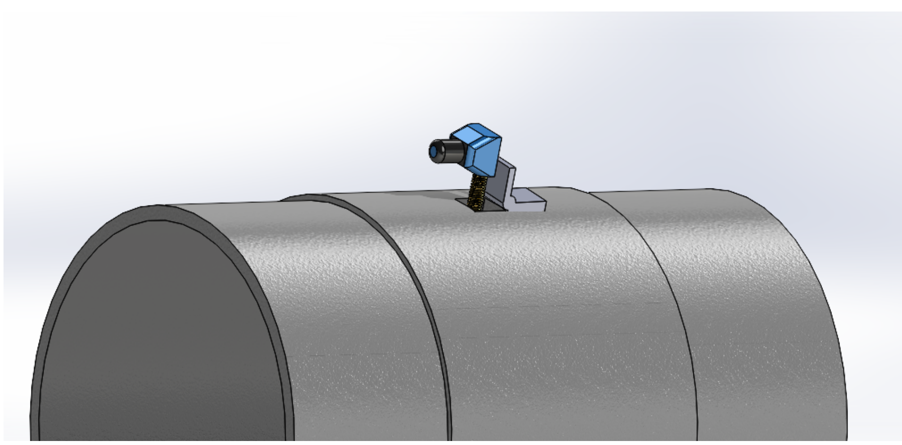

Figure 63: CAD model of the gate design.

4.3.3 Antenna Type

An antenna designed to receive the frequencies used during competition is required in order to properly receive transmitted commands. There are numerous options for antennas, but many are too bulky or have unwanted or undesired properties. The following alternatives are considered due to their efficacy, availability or ease of construction, variety of mounting options, and ease of use.

4.3.3.1 Dipole

Dipole antennas are the simplest form of antennas. They consist of two conductive wires connected to a receiver in the middle. Incoming RF waves polarize the wires, creating positive and negative charges. These charges change over time in accordance to the incoming RF wave. This change in charge is propagated down into the receiver, which can then interpret the RF wave. In order to receive RF signals of a specific wavelength, the dipole antenna must be (approximately) one-half the length of that frequency's wavelength. Thus, dipole antennas are also often referred to as half-wave dipoles. Figure 64 shows a basic view of how a dipole antenna is constructed and its operating principle.

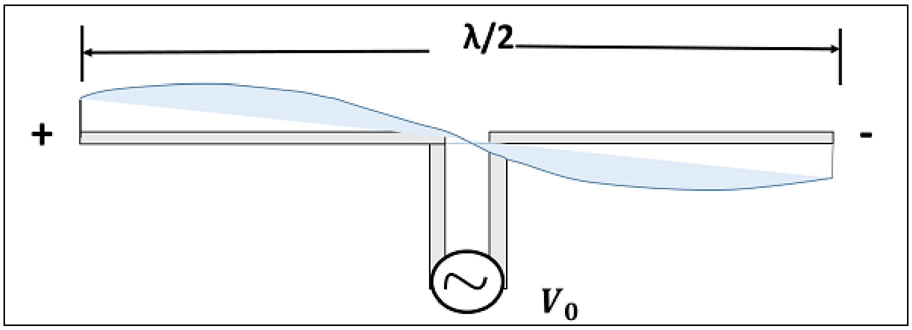

Figure 64: A basic diagram of a half-wave dipole. [7]

Dipole antennas have several advantages. They are relatively easy to construct, requiring only two wires and an insulating element. They also do not need to be perfectly straight, and thus may be bent or curved slightly as needed. Dipole antennas are also essentially omnidirectional, although they are weakly directional when horizontal, and cannot pick up signals axially. This means that antenna orientation will not play a significant factor in reception efficacy, unless the antenna lies entirely axially to the transmitter. Additionally, as a half-wave antenna, an additional ground is not required, and thus the antenna can easily be mounted to the launch vehicle.

One of the major drawbacks to a half-wave dipole antenna is the necessary length. A half-wave dipole antenna for the approximate competition frequency range would be around 40 inches long. While dipole antennas can handle some curvature, they do still have to be relatively straight. Thus, there would need to be 40 inches of linear mounting space available on the launch vehicle.

4.3.3.2 Whip

A whip antenna is a type of flexible monopole antenna. They are commonly used for non-stationary mounts, such as cars or hand-held radios. Whip antennas consist of a wire and a ground plane, with a receiver in between. RF signals cause charge differences along the length of the wire, which results in a voltage that the receiver is able to interpret. While there are numerous options for whip antennas, the most common type is a quarter-wave antenna, in which the antenna is approximately the length of a quarter-wavelength of the specified frequency. Figure 65 is an example of a common form-factor for quarter-wave whip antennas.

Figure 65: Example of a quarter-wave whip antenna. [18]

One primary advantage of quarter-wave whip antennas is their length. Since they are only as long as a quarter of the target frequency's wavelength, a whip antenna would be approximately 20 inches long. Whip antennas can also be made shorter than a quarter-wavelength through the addition of an inductor in series with the antenna, although these shortened antennas have much lower gain. Quarter-wave and shorter whip antennas are also easily available commercially in the desired frequency, and thus there are a number of options available to best fit the payload's needs. Whip antennas have a similar radiation pattern to dipole antennas, meaning that they can act as omnidirectional antennas except along their axis.

There are two significant disadvantages to a quarter-wave whip antenna, the first being that they require a large grounding plane. Typically, quarter-wave whip antennas are grounded to a vehicle body or the physical ground itself. The size and construction of the launch vehicle makes such grounding impossible or impractical. Additionally, a whip antenna could not be mounted along the body of the launch vehicle. It would have to be either mounted normal to the body of the launch vehicle, creating a large protuberance that would result in instabilities, or internally, in which an internal space long enough to accommodate the antenna would need to be created, and there would be a significant reduction in signal quality due to the RF signals having to pass through the body of the launch vehicle, potentially blocking them entirely.

4.3.4 Antenna Mounting

Independent of antenna type, the mounting scheme used has several available alternatives. While the type of antenna used must be factored in to the antenna mounting scheme used, since length and orientation are determined by this selection, additional considerations include attenuation, ease of maintenance, and integrity.

4.3.4.1 Internal

In this antenna mounting scheme, the antenna would be mounted inside the payload bay, along with the rest of the payload electronics. This approach means that only a single antenna is necessary for proper reception. There are several significant disadvantages to this design. In order to fit the antenna fully inside of the launch vehicle, the payload bay would have to be between 20-40 inches depending on the antenna being used. In addition, there is the high likelihood of attenuation of RF signals due to material interference. There is also the possibility of electromagnetic interference from the other electronic devices inside of the payload bay.

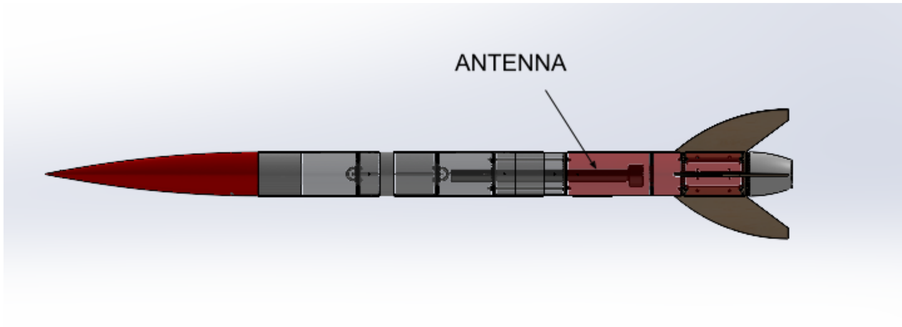

Figure 66: CAD model of the antenna inside of the vehicle.

4.3.4.2 Along Launch Vehicle

Another mounting scheme has the antenna placed along the LE of the fin. In this approach, two antennas are placed along opposing fins to reduce aerodynamic drag. This allows the vehicle to have one of the antennas pointing upright away from the ground no matter how the fin can lands. The antenna feeds into the payload bay through a small hole on the side of the vehicle and allows the payload system to receive RF Signals clearly. The main disadvantage of the design is that, there is a chance that the antennas might get damaged during launch or landing.

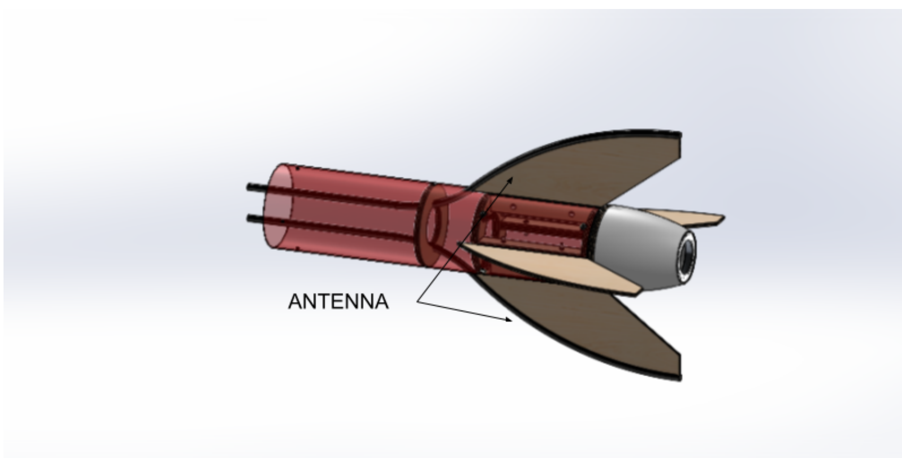

Figure 67: CAD model of the antenna along the fins.

4.3.4.3 Internal to Fin

In this mounting scheme, the antenna runs through the inside of the fin. During the fin manufacturing process a groove is drilled along the fins to host each antenna. Because the fins are removable, multiple fins with antennas may be installed such that no matter in which orientation the fin can lands, one or more of the antennas will be upright to receive RF signals. There are several disadvantages to this design. The manufacturing process would be very complex and it might put the structural stability of the fins at risk if not done correctly. There will also be attenuation due to the antenna being in between two pieces of wood.

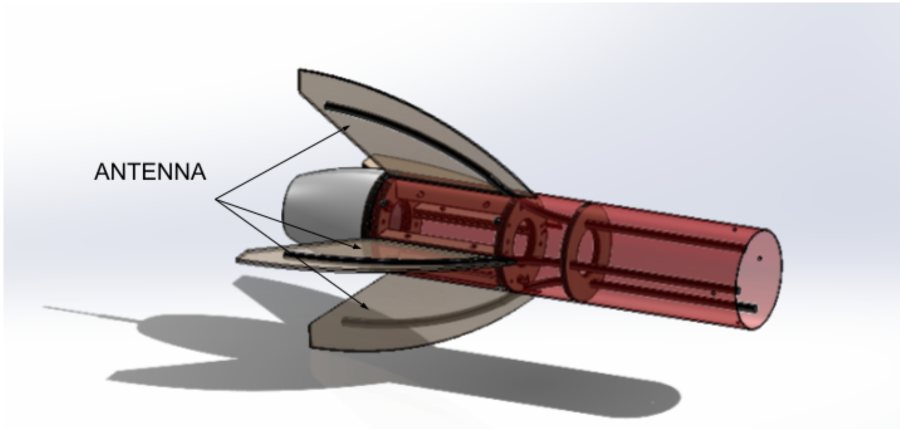

Figure 68: CAD model of the antenna inside of the fin.

4.3.5 Power Switch Type

The methodology used to power-on the payload system is an important component for which to analyze the alternatives. This power switch should be connected in between the positive voltage source and the powered component, as close (component-wise, i.e. no components between the power source and switch) as possible. Power switch alternatives should keep in mind ease of mounting, ease of use, stability, and the launch and assembly conditions in which it will need to be used.

4.3.5.1 Screw Switch

Screw switches have been used traditionally to power on payloads. They consist of a PCB with three screw holes, two of which are for mounting. The third screw is used to close an electrical connection between the two wires soldered onto the board. These screw switches are typically mounted to the payload sled at an angle, which allows them to be accessed from the outside of the launch vehicle via a long screw driver and a hole in the launch vehicle body. Figure 69 shows a typical screw switch.

Figure 69: A commercially available screw switch. [16]

Screw switches are easily mounted to the payload for outside access, with a simple angled bracket allowing for proper positioning. They also do not require much preparation for use, simply being attached to the payload with the connection screw being loosely threaded in. The payload with the screw switches attached can easily be slid into the body of the launch vehicle.
Screw switches have been used extensively by the team in the past, and several significant issues can be noted. While electrically sound, problems may often arise during payload activation on the launch pad. The screw hole cannot easily be seen through, and there are two structural screws in addition to the screw providing electrical continuity. These structural screws can often be mistaken for the continuity screw, which can lead to undesirable results. If one of these structural screws is unsecured due to the assumption that it is the continuity screw, the screw switch itself may shift out of reach from the screw hole, in addition to there now being loose hardware inside of the launch vehicle. Similarly, if the continuity screw is fully unsecured, it will

fall inside the body of the launch vehicle. In order to re-prime the switch, the launch vehicle will have to be removed from the launch pad and disassembled.

4.3.5.2 Pull-pin Switch

Pull-pin switches consist of a limit switch (Single Pull, Double Throw) inside of a housing, which allows for a pin to hold the switch in position such that one of the terminals is unconnected to the others. When the pin is removed, the connection is made. There is a handle connected to the end of the pin, allowing it to be secured to the side of the launch vehicle by adhesive tape, as well as easy removal by hand for payload activation. The body of the switch is mounted onto the payload sled using the included hardware. The following figure (70) shows a commercially available pull-pin switch.

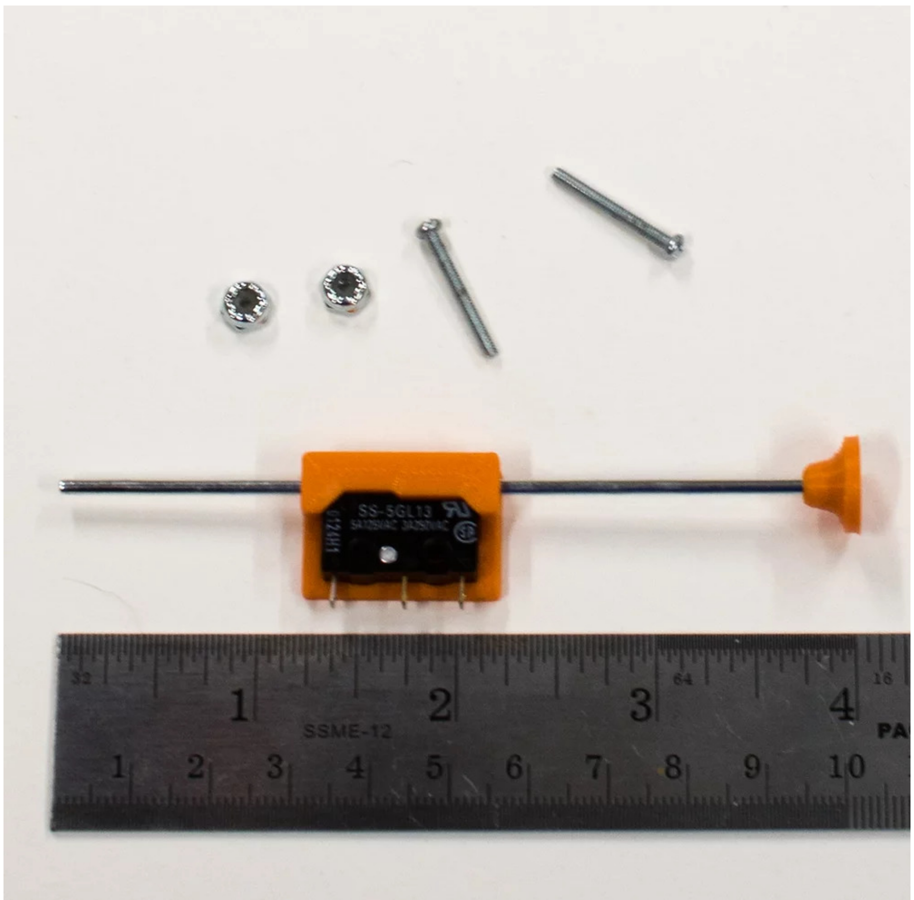

Figure 70: A commercially available pull pin switch. [17]

Unlike screw switches, pull-pin switches require no tools to activate, and there is no chance of components falling inside the launch vehicle during activation. The pin is also easily reinserted, aiding in easy recovery and troubleshooting. Because of the pin coming from the inside of the launch vehicle to the outside, the orientation of the payload is fixed, unlike with screw switches where alignment marks must be used.
Because these switches have an element that goes through the body of the launch vehicle, the payload must first be turned on, inserted into the launch vehicle, and then turned off with the insertion of the pin. While this is a relatively complicated procedure, the efficacy has been proven through implementation in non-team projects by team members, including its use as the power switch for an altimeter in a successful TRA L1 attempt.

4.3.6 Camera Type

Several types of cameras are considered below to satisfy system success criteria. Cameras must be lightweight and small enough to fin inside of the fin can and payload bay sections, powerful enough to capture a wide field of view, and must interface easily with the on-board computer, a Raspberry Pi. Three types of cameras are discussed below: FPV drone cameras, Raspberry Pi camera modules, and webcam cameras.

4.3.6.1 FPV Camera

First-Person View (FPV) cameras are commonly used on COTS drones to capture flight footage. These cameras come in a range of sizes, fields of view, and resolutions, which influence which brand and type of FPV camera is optimal for use. FPV cameras are small, require around 5V of power, and can easily physically interface with the launch vehicle. The RunCam Nano 2, an example of an FPV camera, can be seen in Figure (71) below.

Figure 71: A commercially available FPV camera, the RunCam Nano 2. [20]

However, these cameras are not standard to the Raspberry Pi and, thus, have no standardized method of connection to a Pi. Little information is available on the hardware and software required to connect FPV cameras to Raspberry Pi, so this interface must be designed by the team.

4.3.6.2 Raspberry Pi Camera Module

The second camera option is a Raspberry Pi-compatible Camera Module. This type of camera integrates well with the on-board Pi, as it is made specifically for this purpose. Many manufacturers produce cameras compatible with Raspberry Pi hardware and software, and this method of camera connection comes with benefits and drawbacks. Firstly, cameras that interface with Pis by default communicate via CSI cables. These cables are complex, fragile, and are difficult to connect in parallel. This means that the four cameras used in the payload system will be difficult to communicate with the Pi via a relay system. Additionally, most standard Pi cameras have fields of view around 60 degrees, which is too low to capture useful images. The Arducam IMX219 camera module, an example of a Pi-compatible camera, is shown below in Figure (72)

Figure 72: A commercially-available Raspberry Pi camera, the Arducam IMX219. [8]

4.3.6.3 Webcam

A final option for payload cameras is a webcam-style camera that utilizes a traditional USB interface. These cameras are flexible and often quite large, but can easily be extended from the vehicle to capture images upon landing. Below, in Figure (73), is an example of a commercially-available webcam.

Figure 73: A commercially-available webcam camera. [6]

Webcams can often be bulky and difficult to physically integrate into the vehicle. However, they are designed to integrate well with computers and can use the Raspberry Pi USB input as an interface. Of the four USB ports on the Raspberry Pi, one is taken by the RTL-SDR dongle. Thus, if this option is chosen, a USB hub must be added to the Pi to allow for the required number of ports.

4.4 Feasibility Study

In choosing a final payload design, a feasibility study is conducted to assess the benefits and drawbacks of each design component. Pugh matrices were chosen to conduct feasibility studies. In each table, the critical elements of each design component are delineated, the impacts of those elements are enumerated on a scale of 1 to 5 (where 1 is least impactful and 5 is most impactful), and ratings for each design are displayed.

Table 24: Design rating impact on critical elements.

Strong Positive Impact	Little to No Impact	Strong Negative Impact
+1	0	-1

Table 25: Pugh matrix of camera mounting scheme.

Critical Element	Project Impact	Arm Mount	Hard-Mount
Simplicity	5	-1	1
Ease of Assembly	4	-1	1
Cost of Manufacturing	3	1	-1
Ease of Manufacturing	3	0	-1
Weight/CG Change	3	-1	1
Structural Integrity	3	-1	0
Coding Required	2	-1	1
Picture Quality	1	1	-1
RAW TOTALS		-3	1
WEIGHTED TOTALS		-13	7

Table 26: Pugh matrix of camera housing methods.

Critical Element	Project Impact	Protrusion	Hole	Gate
Aerodynamics	2	-1	0	1
Structural Integrity	3	1	-1	0
Simplicity	5	1	-1	-1
Picture Quality	1	-1	-1	-1
Manufacturing Cost	3	1	0	0
Coding Required	2	1	0	0
Weight + CG Impact	4	1	0	0
RAW TOTALS		3	-3	-2
WEIGHTED TOTALS		14	-9	-4

Table 27: Pugh matrix of camera housing shape alternatives.

Critical Element	Project Impact	Dome	Teardrop
Aerodynamic Impact	4	-1	1
Cost of Manufacturing	3	1	0
Ease of Manufacturing	3	1	0
Structural Integrity	3	0	1
Distortion	2	0	-1
RAW TOTALS		1	1
WEIGHTED TOTALS		2	6

Table 28: Pugh matrix of Antenna Alternatives.

Critical Element	Project Impact	Whip	Dipole
Length	5	1	-1
Size	2	-1	1
Mounting Configuration	3	-1	1
Signal Quality	3	1	0
Simplicity	4	0	1
Antenna Orientation	5	0	1
RAW TOTALS		0	3
WEIGHTED TOTALS		3	9

Table 29: Pugh matrix of antenna mounting scheme.

Critical Element	Project Impact	Inside Payload	Along Fin	Inside Fin
Signal Quality	5	-1	1	-1
Aerodynamics	3	1	-1	0
Ease of Construction	3	1	1	-1
Cost	2	1	0	-1
Simplicity	2	0	0	-1
RAW TOTALS		4	-1	-4
WEIGHTED TOTALS		3	5	-12

Table 30: Pugh matrix of power switch alternatives.

Critical Element	Project Impact	Pull-Pin	Screw
Weight	1	0	1
Size	2	0	1
Simplicity	3	-1	1
Reliability	5	1	-1
RAW TOTALS		0	2
WEIGHTED TOTALS		2	1

4.5 Leading Payload Design

The leading payload design utilizes teardrop-shaped camera housings and four protruding cameras in a system known as the Surrounding Optics and Communication System (SOCS). Protruding aerodynamic camera housings are chosen as the method of payload camera retention and the cameras do not linearly actuate away from the vehicle. Static cameras afford the system more benefits than drawbacks when compared to cameras that extend out of the vehicle, and interface well with a protruding housing design. Teardrop-shaped housings scored the highest on the weighted Pugh matrix feasibility study, and are the best candidate to house and protect payload cameras. A 3D model of SOCS is shown below in Figure (74). In this figure, the airframe is transparent for visual clarity only. The final full-scale airframe is made of opaque G12 fiberglass. SOCS takes in RAFCO commands using one of two dipole antennas (chosen via IMU data and a relay board), parses those commands using an in-house APRS decoding software running on a Raspberry Pi, and executes those commands by sending data to and from the camera-servo assembly. Details of this system can be seen in the flowchart in Figure (76). Electronics that power and control SOCS are housed on SOCS sled and retained in the vehicle using two four-layer, 1/2-in thick bulkheads on either side of the payload bay. The sled is secured to these bulkheads using two 1/4-20 threaded rods. Further system details are discussed in Section 4.5.2. Table (31) shows the estimated weight for the various payload components, along with the overall weight.

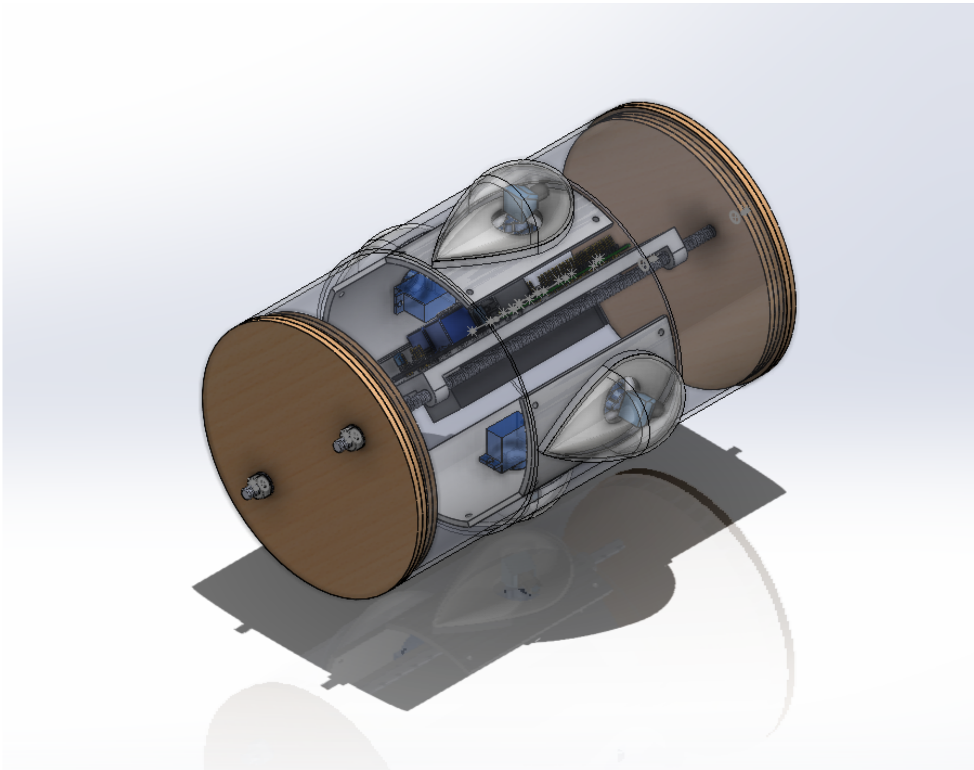

Figure 74: CAD model of the payload bay housing SOCS.

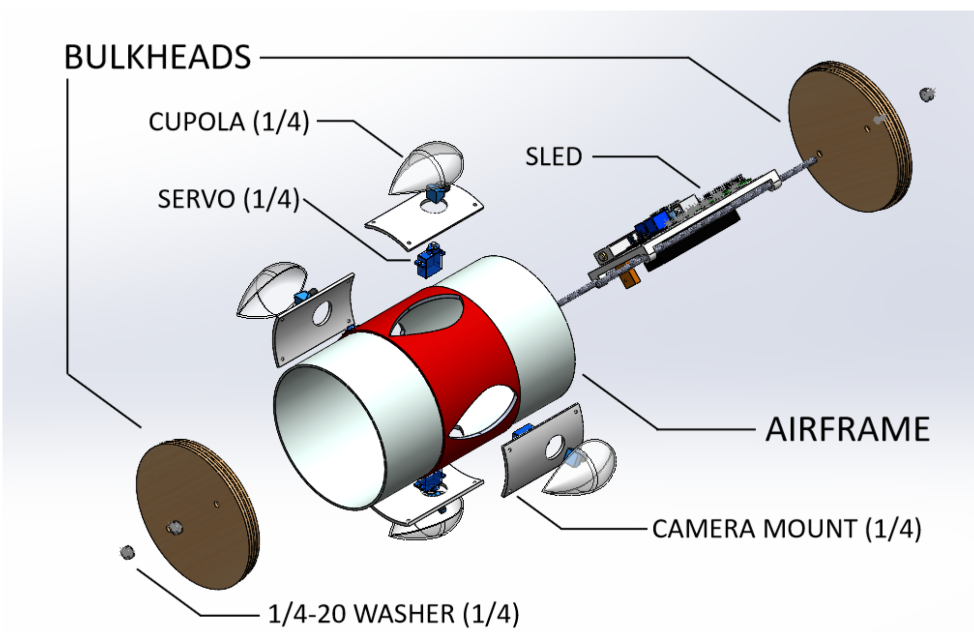

Figure 75: Exploded view of the payload bay with internal electronics.

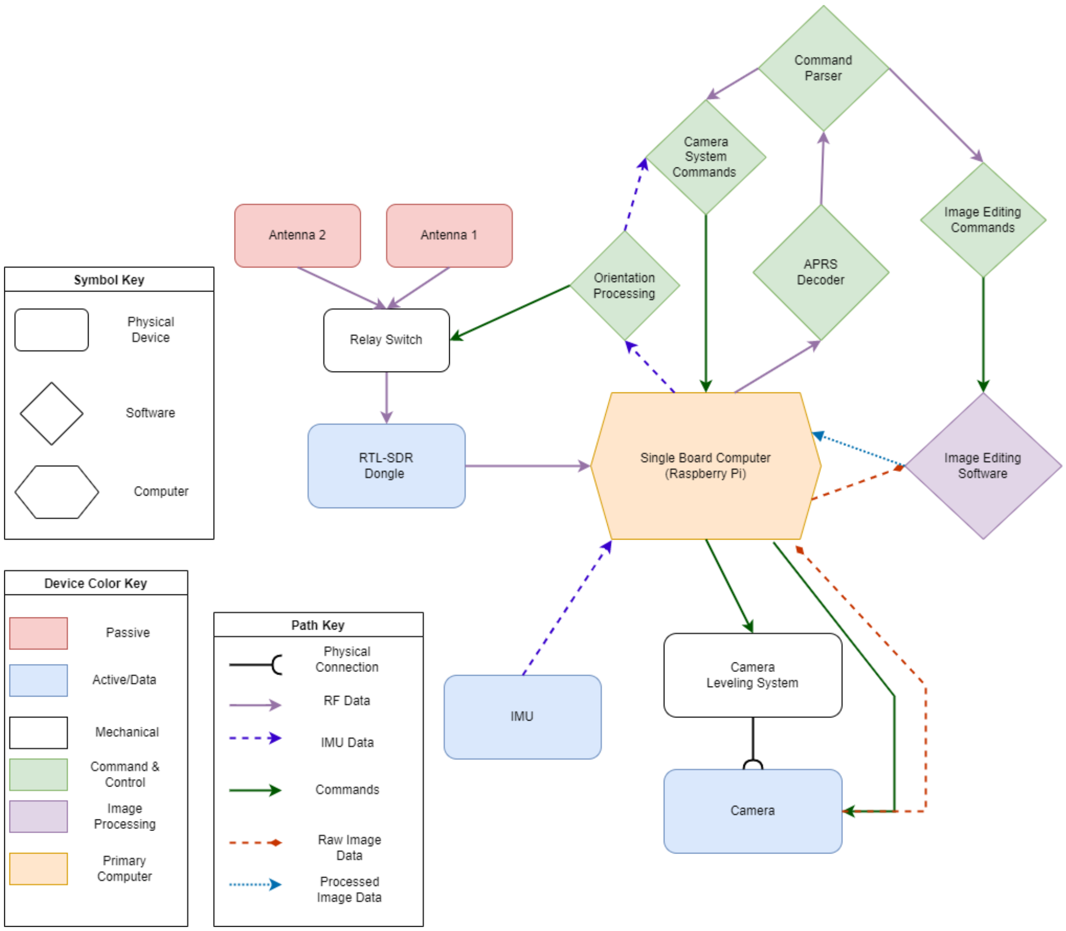

Figure 76: Current payload system operational flowchart.

Table 31: Payload weight estimates.

Component	Individual Weight (lb)	Count	Net Weight (lb)
3 Cell Lipo Battery	0.85	1	0.85
Raspberry Pi	0.1	1	0.1
2-Channel Relay	0.1	1	0.1
Dipole Antenna	0.2	2	0.4
RTL-SDR Dongle	0.07	1	0.07
Servo	0.02	4	0.08
Camera	0.03	4	0.12
Camera Housing	0.08	4	0.32
Threaded Rods	0.25	2	0.5
Payload Sled	0.15	1	0.15
Camera Mount	.08	4	0.32
Pull-pin Switch	0.05	1	0.05
		Overall Weight	3.06

4.5.1 Payload Electronics

The payload electronics will be activated using a pull-pin switch. This switch was chosen over a screw-switch due to its reliability and ease of use. Only a single pull-pin between the 3-cell LiPo battery and the rest of the

electronics system is required.
The leading camera choice is a Pi-compatible Arducam camera module. These modules interface well with both a Pi and the corresponding Arducam Multi Camera Adapter Module, shown below in Figure (77). This module allows for up to four cameras to be attached to and addressed by the Raspberry Pi.

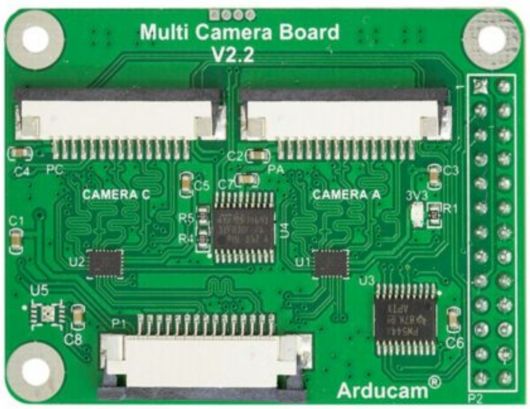

Figure 77: The Arducam Multi Camera Module for the Raspberry Pi. [9]

Unlike many other Pi cameras, Arducam cameras have high fields of view that are within the bounds of competition and team derived requirements. Additionally, third party CSI cables that are reinforced to survive repeated twisting are available commercially.
The leading antenna design consists of two Dipole antennas mounted on opposite sides of the launch vehicle, on the fin can. This ensures that at least one antenna will always be a considerable distance off of the ground. Dipole antennas were chosen for their ease of construction, longitudinal mounting capability, and lack of need for an additional ground plane. External antenna mounting was chosen due to the length of the antennas, as they would be too long to reasonably fit in either a section of the launch vehicle nor inside the fins, and lack of signal degradation due to material interference. These two antennas are attached to a two-channel relay. Using the data provided by the IMU, the same orientation determination that is used to decide which camera is facing upright is used. The Raspberry Pi will then command the relay to select the proper antenna to connect to. This antenna will then attach to a Nooelec NESDR SMArt RTL-SDR dongle, which contains a REALTEK RTL2832U, allowing for the processing of analog RF signals by the Pi. The RAFCO commands are then decoded and parsed by software on the Pi, and the desired commands are sent to the camera system and the image processing software respectfully. The previously described servo and camera commits the described commands, and passes the captured image to the image processing software, which performs the requested edits. The image is then saved on-board for later retrieval. The following figure (78) is the electrical block diagram for the current payload system.

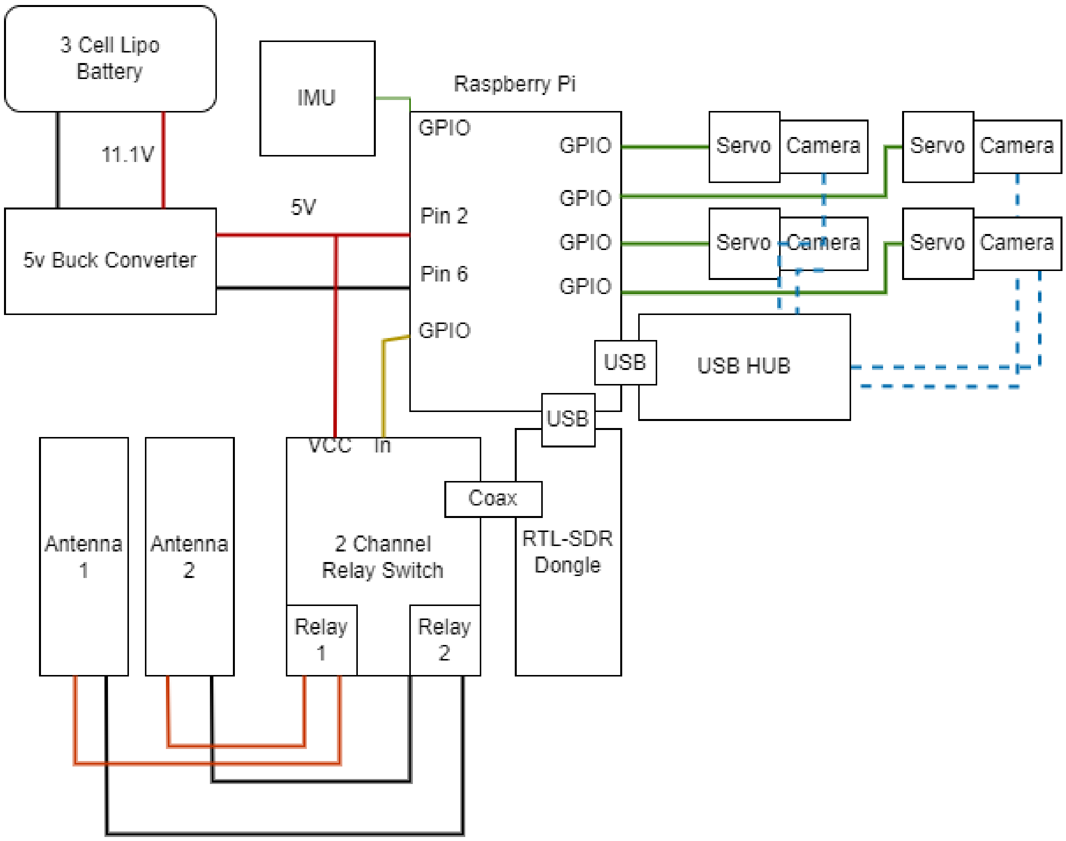

Figure 78: Current payload electrical block diagram.

4.5.2 Payload Retention and Interface

The payload bay, a 10-inch vehicle section that is comprised of 4 inches of body tube in between two 3-in long coupler sections, houses SOCS. SOCS consists of a payload sled housing electronics, four cameras attached to their respective servos and mounts, bulkheads and threaded rods that retain the system and integrate it to the vehicle, and antennas that receive commands.

All four cameras that comprise SOCS protrude through teardrop-shaped holes in the launch vehicle airframe that correspond to the teardrop-shaped bottom of each housing. These cameras are attached to one servo each, which is riveted or bolted to the inside of the payload bay airframe. Each housing has a bottom lip that allows the housings to sit between the plastic camera unit mount plate and the airframe. A 3D CAD model of this assembly is shown below in Figure (79).

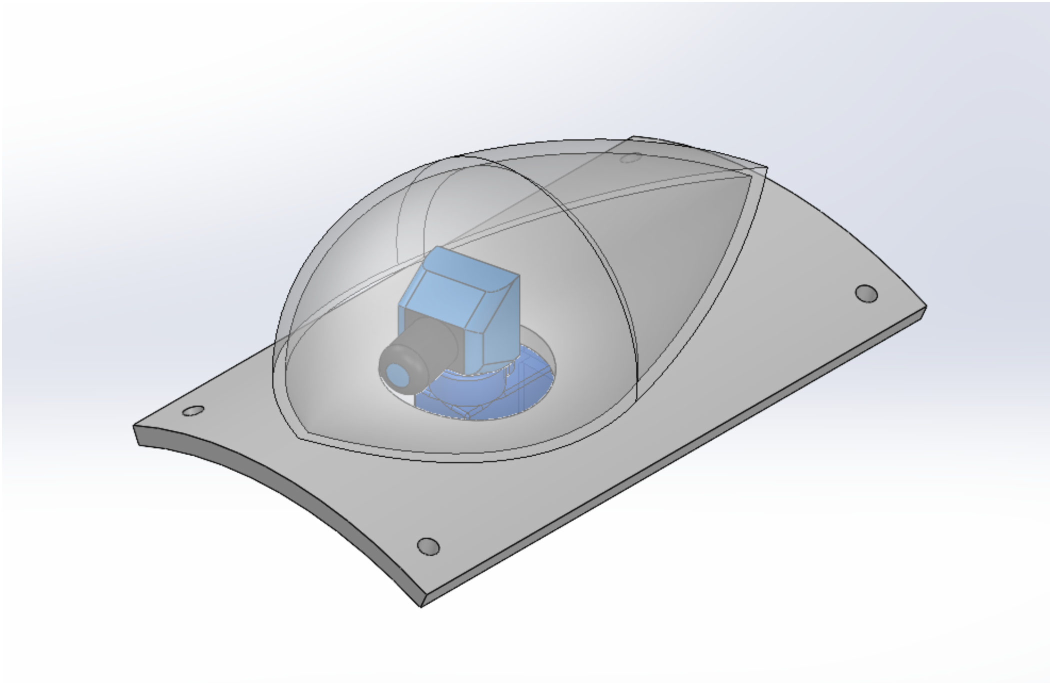

Figure 79: Camera, mount, servo, and camera housing assembly.

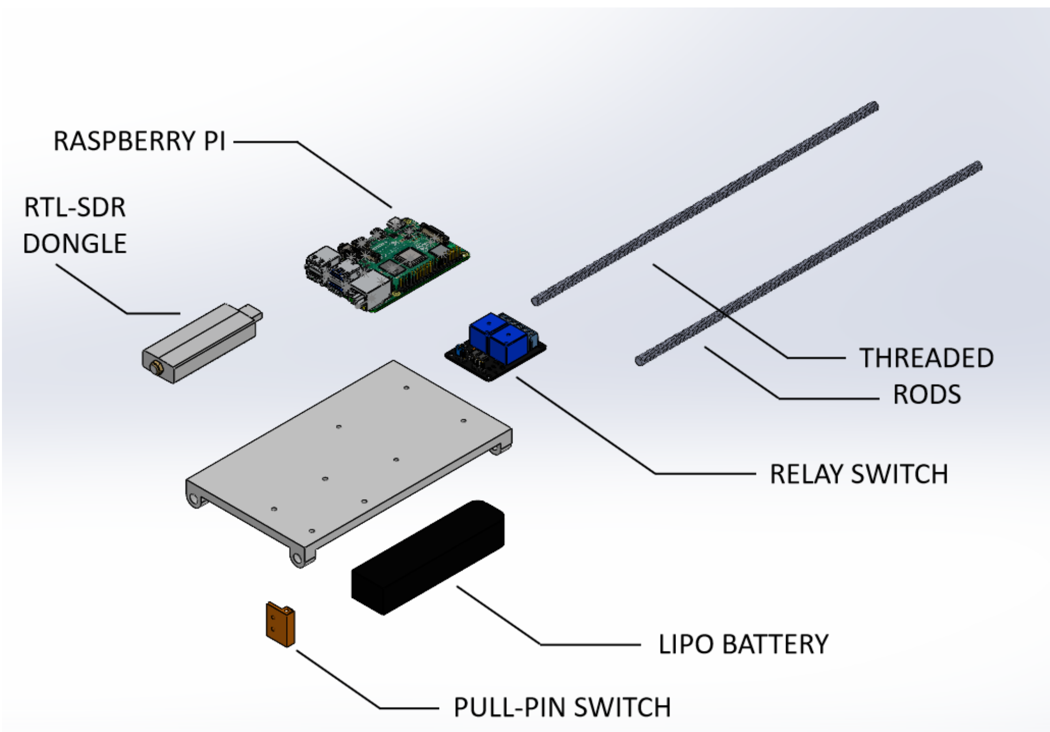

Figure 80: Exploded view of the payload sled.

The holes in which the camera housings will sit and through which the cameras will look will be created by first using a drill press to make initial pilot holes. Then, once an outline of each camera housing is marked on the airframe using pencil, a Dremel tool will precisely grind away fiberglass until the desired teardrop shape is achieved. Because of the dangerous nature of fiberglass shards, Fiberglass sanding and grinding will take place in a ventilated area while personnel wear respirators, well-fitting safety glasses, and other necessary protective equipment.

4.5.2.1 Camera Housing Manufacturing Techniques

Vacuum forming using PETG is the camera housing manufacturing method of choice at the time of PDR. Vacuum-formed PETG is more transparent than SLA-printed material and requires less specialty equipment to manufacture. Additionally, vacuum forming is more cost-efficient than SLA printing, which must be ordered from a 3D printing facility.

The Mayku Vacuum Former (shown in Figure (81) below) will be used to vacuum-form payload components for the fullscale vehicle and is accessed through NCSU's E-Garage facilities.

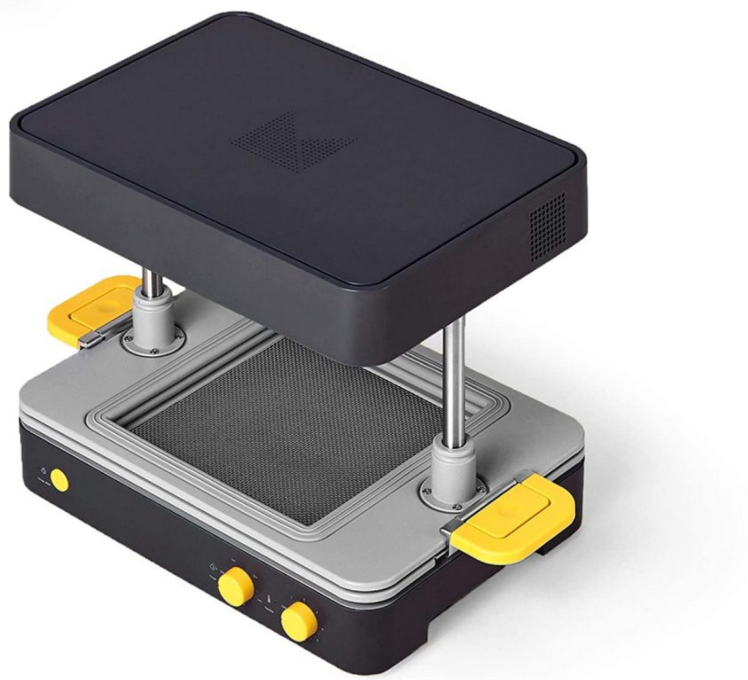

Figure 81: The Makyu vacuum former used to manufacture fullscale camera housings.

5 Safety

5.1 Safety Documentation Methods

This year, safety documentation will be performed through FMEA analysis and Likelihood-Severity (LS) matrices. Verification of safety procedures is checked through various sources, including but not limited to, inspection, Launch Day checklists, NAR Safety Code, TRA Safety Code, and HPRC standards.

Below is the Likelihood-Severity matrix upon which all of the FMEA tables are based.

Table 32: FMEA Matrix

		Level of Severity			
		1 Low Risk	2 Medium Risk	3 High Risk	4 Severe Risk
Likelihood of Occurrence	A Very Unlikely	1A	2A	3A	4A
	B Unlikely	1B	2B	3B	4B
	C Likely	1C	2C	3C	4C
	D Very Likely	1D	2D	3D	4D

5.2 FMEAs

Table 33: Launch Vehicle Structure FMEA

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After
Hazards to and from Bulkheads						
S.B.1	Failure of U-bolts	Excessive deployment force	Ballistic reentry	4A	Distribution of load during construction	4A
S.B.2	Failure of nose cone bulkhead bolts			4A	Load management during construction	4A
S.B.3	Bulkhead cracking	Excessive stress around bolt connections	Separation of bulkhead from airframe	3D		3B
S.B.4	Bulkhead delaination	Excessive axial stress from shock cord connections		3D		3B
S.B.5	Separation of bulkhead from airframe	Softened epoxy	Stabilization of LV is changed	3D		3B
		Excessive force from latch connections				
S.B.6	Exposure of bulkhead to hot ejection gases	Black powder explosions	Singes/burns on bulkhead	3B	Ensure LV is kept in optimal temperatures	3A
Hazards to and from Removable Fin System						
S.F.1	Failure of #8 bolts securing assembly to airframe	Excessive force from motor, excessive force on landing	CATO or loss of stability, damage to LV components (potentially repairable)	3B	Bolts and rods are designed to have a high safety factor, as supported by preliminary calculations.	3A
S.F.2	Buckling of fin can threaded rods or fin runners	Excessive force from motor, excessive force on landing	CATO, loss of stability	3B		3A
S.F.3	Thrust plate failure		CATO, damage to airframe	3A	Material decision during design phase	2A
S.F.4	Fin breaks	Excessive force upon landing, fin flutter	Loss of stability during flight	3B	Fiberglass reinforcement during construction	1C
S.F.5	Centering ring cracks or delaminates	Excessive force from motor	Motor is not securely held, CATO, loss of stability	3A	Proper construction techniques	1A
S.F.6	Motor retainer comes un-epoxied	Epoxy is weakened by heat	Motor descends separately from the launch vehicle	2B	Ensure epoxy is rated for expected temperatures	1A
Hazards to and from Airframe						
S.A.1	Fin can body tube cracking	Hoop stress from internal pressure	Jettison of motor and motor tube, CATO, Inability to relaunch LV	4A	Propellant grains are securely fastened in a motor tube and motor construction is assisted by Tripoli personnel	2A
S.A.2	AV bay body tube cracking		Inadequate axial force generated by black powder to separate LV sections	3B	Calculations are performed in order to get an accurate measurement for the correct amount of black powder, ejection tests are performed prior to each and every flight	3A
S.A.3	Body tube zippering	Excessive forces from shock cord		2B		

Continued on next page

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After
		Parachute ejection at excessively low altitude			Fiberglass body tube and rigid, appropriately sized couplers are used in the full scale LV to	2A
S.A.4	High-energy impact with ground	Late or no parachute deployment	Airframe rupture	3B	An appropriate recovery system is used to slow the LV for a safe landing	3A
S.A.5	LV section collision	Shock cord of insufficient length		3B		3A
S.A.6	Airframe exposure to water	LV touchdown in wet area of launch field	Airframe disintegration/rupture	2C	The full scale LV airframe is made of waterproof fiberglass	2B
		Inclement weather	CATO		The subscale LV is made of blue tube, so it will not be launched in inclement weather conditions	
S.A.7	Airframe exposure to burning black powder	Uncontrolled ejection charges	Airframe disintegration/rupture	1D	The airframes of both the subscale and the full scale LVs are constructed from heat-resistant materials	1C
S.A.8	Body tube abrasion	High energy impact with the ground	Changes in LV center of pressure/stability, irreversible damage to LV	1C	An appropriate recovery system is used to slow the LV for a safe landing	1B
		Parachute re-inflation upon landing, causing the body tube to be dragged			Launches do not occur in high winds	1A

Table 34: Recovery System FMEA

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After
Hazards to/from Parachutes and Shock Cord						
R.C.1	No parachute deployment	Insufficiently powered altimeters	LV goes ballistic, LV high-energetic touchdown	2D	Altimeter battery checked immediately prior to flight	2A
R.C.2		Insufficient black power charges to separate LV components		3D	Dual-redundant black powder charges which are tested for efficacy before flight	3B
R.C.3		Shear pins of an excessive diameter		2D	4-40 shear pins are used	2C
R.C.4		Excessively moist black powder		1D	Black powder is properly stored, no launches occur in inclement weather	1C
R.C.5	Parachute rips and tears	Parachute contact with hot ejection gases	Poor parachute performance, high-energetic touchdown	4C	Parachutes are wrapped in fireproof Nomex like a burrito in order to insulate them from heat and tension	4B
R.C.6		Parachute contact with motor flame		1C		1B
R.C.7		Parachute entanglement during separation or deployment		2C		2B
R.C.8	Shock cord disconnection	Loose quick links	LV goes ballistic, LV high-energetic touchdown	3D	Quick links are tested immediately prior to flight	3B
R.C.9	Shock cord rip	Excessive flight forces on shock cord		1D	Shock cords are rated for up to 6000 lb of flight forces	1C
R.C.10	Excessive force on shock cord	Late parachute deployment		3D	Estimated forces on the recovery system are calculated and verified	3C
R.C.11	Parachute falls off	Manufacturing defect		4A	No mitigation possible, manufacturing defect is likely to go unnoticed until the parachute goes through the stresses of flight	4A
Hazards to/from Avionics and Black Powder						
R.A.1	Main parachute deployed at apogee	Wires for main and drogue parachutes mixed up, wires misrouted	Wind drift LV tree landing Personnel made to walk considerable distances during recovery	3C	Avionics utilize labeled quick-connects to prevent mix-ups	2A
R.A.2	Shear pins shear prematurely	Shear pins are of an insufficient diameter	Premature section separation	3B	4-40 shear pins are utilized and have been chosen for their strength	3A
R.A.3	Premature section separation	Premature black powder detonation	Failure to reach intended apogee	3C	Altimeters are set properly and verified by team leads	3B
R.A.4	Late section separation	Delayed black powder detonation	Excessive force on shock cords, ballistic landing	3D		3C
R.A.5	Excessive wind drift	Premature parachute deployment	LV tree landing	3B	Avionics utilize labeled quick-connects to prevent mix-ups by personnel	2A
R.A.6	Premature black powder detonation	Sympathetic main/drogue black powder detonation	Parachute damage	1C	AV blast caps face in opposite directions	1B

Continued on next page

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After
R.A.7	Abrupt pressure changes in AV bay	LV separates while altimeters are still armed	Black powder detonates while altimeters are being armed	3C	Pull-pin switches are used to arm altimeters, preventing jostling that can occur with screw switches	2B
R.A.8	LV flight without armed altimeters	No section separation	Ballistic descent	1D	Recovery checklists are used to ensure altimeters are installed in rockets	1B

Table 35: Payload System FMEA

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After
Hazards to/from Payload Structures						
Pa.S.1	Scratches on cupula surface	High energy contact between cupola and ground	Blurred or obscured camera image	3C	Triple redundant, top cupola will not be in contact with the ground	3B
Pa.S.2	Structural deformation of cupula		Obscures camera image and damages the structure of the mounted camera	2C	Triple redundant, top cupola will not be in contact with the ground, installing cupolas without epoxy so they are removable	2B
Pa.S.3	Cracking/breaking to the antenna	High energy impact between the antenna and the ground	Antenna will no longer be able to receive commands	2D	Two antenna on different sides of LV to ensure that one always lands sky-upwards, recovery system in place to drastically slow down LV to minimize force between LV and ground, the whole antenna will be secured with tape and a protective cover will be added on the leading edge of the fin	2C
Pa.S.4	Cracking/breaking of camera system	High energy impact between camera and launch vehicle	Camera will no longer be able to execute commands received by the antenna	3D	Camera will be fixed in place to the outside of the LV, camera mount designed to prevent movement from the camera upon high energy impact, recovery system in place to minimize force between LV and ground	2D
Pa.S.5	Cracking/breaking of payload sled	Impact between the components of the payload sled and the inside of the launch vehicle during launch	Loss of power to payload electronics, loss of communication between the Raspberry Pi and the antenna	1D	Payload sled attached to two 1/4-20 threaded rods which will be secured by at least two nuts per side on a bulkhead that will keep the sled from moving around inside the payload bay, payload sled will be 3D printed so that it can withstand the impact force when the LV comes in contact with the ground	1C
Pa.S.6	Payload electronics cables shear/fraying	Friction due to contact between cables and sled		3D	Payload sled will be designed to secure each cable in place while allowing enough room to not constrict cables	3C
Pa.S.7	Camera obstruction	Payload parachute, fins, fin can, and/or shock cord in line of sight of the camera	Failure of camera to take clear pictures of launch vehicle's surroundings	3A	Each camera will be placed between the fins and near the top of the fin can to minimize fin obstruction, launches not conducted during inclement weather	2A
Hazards to/from Payload Electronics						
Pa.E.1	Antenna does not have clear view	Landing orientation	Weak/corrupted signals or no reception of commands	2D	Multiple antennas on different sides of the launch vehicle to ensure that one will always land sky-upwards	2C
Pa.E.2	Damage to LiPo battery connection/low power	LiPo battery is not charged fully Friction due to contact between cable and sled	Loss of power to Raspberry Pi and camera	3D	Voltage of battery measured before each launch, all connections and wires are secured	2D
Pa.E.3	Overvoltaging of electronic components	Voltage from LiPo battery is higher than components are able to withstand	Electronics get fried and are no longer useable	2D	Use of buck converters to regulate voltage going into components	1D

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Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After
Pa.E.4	Wires shorting together on circuit board	Wires are too loosely connected and come in to contact with each other	Incorrect voltages are passed through the circuit, excessive current flow, possible fire hazard	2D	All exposed wire is covered in shrink wrap and secured with electrical tape	1D

Table 36: Aerodynamic FMEA

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After
Hazards to/from Motor						
Ae.M.1	Uneven pressure buildup in motor tube	Defects in propellant grain	CATO	2D	AeroTech Motors are used for their low likelihood of catastrophic failure	2C
Ae.M.2	Motor ejection	Loose motor retainer	CATO	2D	Motor assembly performed under the guidance of certified Tripoli professionals	2C
		Centering ring epoxy fail			Epoxy is allowed to fully cure	
		Crack in fin can body tube			Body tubes are made of fiberglass in order to combat axial compression	
Ae.M.3	Premature ignition	Ignition system static discharge	Severe personnel burns, LV flight without armed altimeters	3D	Ignition systems provided by certified NAR/Tripoli professionals	3B
Ae.M.4	Cracks in propellant grain	Torsional load applied to grains during assembly	Uneven thrust curve, CATO	3D	Motor assembly performed under the guidance of certified Tripoli professionals	3C
Hazards to Aerodynamics						
Ae.A.1	bird strike	bird	CATO	4C	The LV has an optimal stability margin calculated in RockSim and measured immediately prior to launch	4B
Ae.A.2	LV weathercocking	Vehicle over-stability	Failure to reach intended apogee	3A		2A
Ae.A.3	LV divereges from expected trajectory	Vehicle instability	Failure to reach intended apogee; Ballistic descent	3D		3C
Ae.A.4	Fin flutter	Transonic LV speeds	Fin structural damage	1C	The LV will not fly at speeds in the transonic range	1B
Ae.A.5	Centering ring structural failure	Unexpectedly high motor thrust	Motor goes through LV	2D	Fins are made with aircraft-grade epoxy using proven construction methods	2C
		Weak epoxy connections				
Ae.A.6	Fin damage	Fin flutter	Fin loss	1C	The LV will not fly at speeds in the transonic range	1B
			Divergence from intended trajectory			
			Vehicle instability			
Ae.A.7	Abnormal thrust curves	Propellant grain gaps, cracks, holes, and bubbles	Failure to achieve intended apogee	2B	AeroTech Motors are used for their low likelihood of catastrophic failure	2A
			Excessive force applied to structural bulkheads			

Table 37: Personnel Safety FMEA

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After
Hazards to Skin and Soft Tissue						
Pe.S.1	Slips, trips, and falls	Material spills around the lab	Skin abrasion/bruising	3B	After handling of liquid/powder assembly materials, lab floors will be inspected for material spill.	1B
		Wet/uneven launch field conditions			Only required recovery personnel are allowed on launch field to recover rocket; closed toe, heavy duty shoes are required.	
Pe.S.2	Personnel fingers caught in bandsaw blade	Bandsaw blade contact with clothes and/or jewelry	Skin and muscle tear/abrasion	2D	Personnel working with equipment are trained in proper use of machinery and PPE.	2C
		Personnel misunderstanding of bandsaw operation				
Pe.S.3	Contact with hot soldering iron	Personnel misunderstanding of soldering system	Mild to severe burns	3C		3B
Pe.S.4	Personnel collision with LV	Launch rail tipping with assembled LV	Skin and muscle abrasion/tear	2C	Launch rails, provided by TRA personnel, have a locking mechanism that is engaged when the LV is righted.	2B
Pe.S.5		Sideways propulsion from severe instability		2B	The stability margin of the LV is no less than 2.0.	1B
Pe.S.6		LV touchdown within close proximity to personnel		1B	The LV is angled 20° away from personnel; personnel are instructed to keep eyes on all falling LVs and keep others around them aware.	1A
Pe.S.7	High load placed on personnel muscle	Lifting heavy LV components	Muscle strain/tear	4C	At least two persons carry the LV while it is fully assembled and proper lifting techniques are utilized	4A
Pe.S.8	Bug sting/bite	Prolonged exposure to wildlife during launch day activities	Itchiness, rash, and/or anaphylaxis	4A	Bug spray is provided to team members during launch day and there is appropriate knowledge team member allergies as well as on the proper usage of EpiPens	3A
Pe.S.9	Personnel contact with ejection charges	Contact with unknown black powder after touchdown	Mild to severe burns and abrasions	3C	Personnel approaching the LV are provided with Nomex gloves; LV sections are inspected for unblown charges prior to handling.	3B
Pe.S.10	Contact with large, airborne shrapnel	CATO	Severe skin abrasion/laceration	2D	Personnel are separated from the launch pad according to guidelines set by the RSO. AeroTech motors are chosen for their low likelihood of catastrophic failure.	2B
Pe.S.11	Contact with small, airborne shrapnel	Sanding, cutting, or drilling brittle or granular materials	Cuts and bruises	3C	Protective eye and face equipment are provided to personnel working with power tools.	2C
Pe.S.12	Exposure to uncured epoxy fluid	Working with epoxy	Skin rash, skin irritation	3A	Nitrile gloves and other appropriate forms of PPE are provided to personnel working with hazardous liquid/vapor materials.	2A
Pe.S.13	Exposure to vaporous chemicals	HazMat off-gassing		2A		2A

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Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After
Pe.S.14	Excessive amount of walking	Far away LV touchdown	Muscle sprain, shin splints	3A	The LV is equipped with a GPS tracker, and if the LV is sufficiently far away from the launch site, recovery personnel are driven to the recovery site.	2A
Hazards to Bones and Joints						
Pe.B.1	Slips, trips, and falls	Material spills around the lab, wet/uneven launch field conditions	Bone fracture, bone bruise, dislocation	1D	After handling of liquid/powder assembly materials, lab floors will be inspected for material spill. Only required recovery personnel are allowed on launch field to recover rocket; closed toe, heavy duty shoes are required.	1C
Pe.B.2	Excessive amount of walking	Far away LV touchdown	Stress fracture	2D	The LV is equipped with a GPS tracker; the LV is sufficiently far away from the launch site, recovery personnel are driven to the recovery site.	2C
Pe.B.3	Personnel fingers caught in bandsaw blade	Bandsaw blade contact with clothes and/or jewelry Personnel misunderstanding of bandsaw operation	Broken bone	2D	Personnel working with equipment are trained in proper use of machinery and PPE.	2C
Pe.B.4	Contact with large, airborne shrapnel	CATO	Bone fracture requiring immediate medical attention. Limb loss.	2D	Personnel are instructed by the RSO to stand a minimum distance away from the launch pad. AeroTech motors are chosen for their low likelihood of catastrophic failure.	2C
Hazards to Respiratory System						
Pe.R.1	Exposure to epoxy fumes	Working with epoxy	Difficulty breathing, respiratory irritation.	2C	Personnel working with epoxy are provided particle masks. An oxygen sensor in the lab goes off when there is insufficient oxygen.	2C
Pe.R.2	Exposure to COVID-19	Working in close proximity with infected personnel	Respiratory infection, hospitalization, death, outbreak amongst teammates.	4D	Personnel are highly encouraged to follow updated University guidelines for COVID-19. Personnel who have been exposed to or are infected with COVID-19 are encouraged to attend meetings virtually.	4C
Pe.R.3	Exposure to carcinogenic particulates	Working with fillet epoxy	Respiratory irritation and/or infection, cancer	4D	Personnel working with fillet epoxy are provided particle masks.	4C
Pe.R.4	Inhalation of aerosolized particulates	Sanding, cutting, and/or drilling	Respiratory irritation, difficulty breathing.	4B	Personnel working with materials prone to particulate production are provided with particle masks.	4A
Pe.R.5	Inhalation of spray paint fumes	Working with spray paint for rocket aesthetics		4B	Personnel in the vicinity of burning chemicals are provided with particle masks. Personnel stand a minimum distance away from burning motors.	4A
Pe.R.6	Inhalation of combustion reactants	Close proximity to LV motors and ejection charges		3B		3A
Hazards to Head						
Pe.H.1	Personnel contact with high-energy LV components	High-energy LV sections are in proximity to personnel at touchdown		2D	The LV has a dual-redundant recovery system. Personnel are instructed by the RSO to stand a minimum distance away from the launch pad.	2C
Pe.H.2	Launch vehicle tipping during assembly	Launch rail assembly		3D	Launch rails, provided by TRA personnel, have a locking mechanism that is engaged when the LV is righted.	3C

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Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After
Pe.H.3	Slips, trips, and falls	Attempting to jump through or over launch field irrigation ditches	Concussion, brain damage, memory loss, skull fracture	3D	Personnel members are made aware that jumping over ditches is strictly forbidden.	3D
Pe.H.4	Contact with large, airborne, shrapnel	CATO		2D	Personnel are instructed by the RSO to stand a minimum distance away from the launch pad. AeroTech motors are chosen for their low likelihood of catastrophic failure.	2B
Pe.H.5	Impact with ballistic lander	Premature lander ejection from LV		2D	The latch release system will be tested for its ability to withstand flight forces prior to launch.	2B
Hazards to Eyes						
Pe.E.1	Exposure to epoxy fumes	Working with epoxy	Eye irritation, temporary blindness (from tear production), permanent or semi-permanent blindness	3D	Personnel working with epoxy will be provided with safety glasses	3C
Pe.E.2	Exposure to aerosolized particulates	Working with spray paint Sanding, cutting, or drilling		2D	Personnel cutting, sanding, or drilling will be provided with safety glasses.	2B
Pe.E.3	Eye contact with the sun/bright sky	Maintaining eye contact with falling rockets	Temporary or permanent blindness	1B	Personnel maintaining eyes with falling rockets are encouraged to wear sunglasses.	1A

Table 38: Environmental Safety FMEA

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After
Hazards to Wildlife						
E.W.1	Fire on launch field	Motor ignition	Crop damage, wildlife injury, personnel burns	3D	Ground areas around the launch pad are free of flammable debris, launch rails are fitted with blast plates to deflect exhaust gases away from the ground	3B
E.W.2		Black powder ignition		2C	Recovery personnel are equipped with a fire extinguisher	1C
E.W.3		Payload battery explosion		2D	Payload batteries are isolated from moisture, abrasion, and heat	2B
E.W.4	Payload battery explosion	Puncture of battery leading to contact with moisture	HazMat leakage onto the launch field, water contamination, fire on launch field	3D		3B
		Excessive heat surrounding battery				
E.W.5	Contact between LV components and birds	Birds flying in proximity to LV flight path	Wildlife injury, wildlife death, obstruction of bird migration patterns	1C	Airways in the LV flight path are confirmed to be clear by the RSO	1A
E.W.6	Permanent jettisoning of Nomex sheet	Rips and tears in Nomex	Contamination of wildlife habitats, food, supply, and water supply	2A	Nomex is rated to withstand flight forces and is flame resistant	1A
E.W.7		Nomex connection breakage		1A	Nomex sheets are connected to shock cord by steel quick links, which are confirmed to be tight by the safety officer prior to flight	1A
E.W.8	HazMat deposit in irrigation ditch	Battery explosion	Toxic chemicals remain in soybeans by wildlife and humans	2B	All protective insulation is biodegradable, payload batteries are protected from puncture and heat	2A
E.W.9		Explosion byproducts		3D		3C
	Wildlife consumes toxins	HazMat littering	Wildlife develop digestive issues or incur injury or death			
E.W.10	CATO	Motor defects	Wildlife incur injury or death, water supply contaminated	2D	AeroTech motors are selected for their low likelihood of catastrophic failure	2C
E.W.11	LV landing in tree	Premature parachute deployment	Destruction of habitats	4C	Recovery systems are tested away from trees	4B
		High wind drift				
E.W.12	Emission of microplastics	High usage of single-use plastics	Wildlife infertility, bodily inflammation, choking/digestive hazard	4B	Use of reusable containers encourages team-wide	4B
Hazards to Land						
E.L.1	Forceful impact of LV with ground	Late or no deployment of parachute	Permanent ruts or dips in the launch field, inability for field to be used for future farming endeavors	3A	Use of altimeters in parachute deployment to ensure accuracy	2A
E.L.2	Non-recoverable landing in tree	Premature parachute deployment	Permanent damage to tree	4C	The recovery system is labeled and documented so that assembly mistakes are unlikely	4B
		High wind drift				
E.L.3	Fire at launch field	CATO	Tree destruction, inability for field to be used for future farming endeavors	2D	Ground areas around the launch pad are free of flammable debris, launch rails are fitted with blast plates to deflect exhaust gases away from the ground	2B
		Motor ignition				
		Black powder detonation				
		Payload battery explosion				

<table>
<tr><th>Label</th><th>Hazard</th><th>Cause</th><th>Effect</th><th>LS Before</th><th>Mitigation</th><th>LS After</th></tr>
<tr><td colspan="7">Hazards to Air/Water</td></tr>
<tr><td rowspan="3">E.A.1</td><td rowspan="3">Greenhouse gas emissions</td><td>Transportation to/from launch field</td><td rowspan="2">Air pollution, contribution to climate change</td><td rowspan="3">4A</td><td rowspan="3">Team members are highly encouraged to carpool to launches and to either take public transportation, bike, or walk to regularly scheduled meetings</td><td rowspan="3">4A</td></tr>
<tr><td>Motor and black powder combustion by-products</td></tr>
<tr><td>Use of power-drawing electronics</td><td></td></tr>
<tr><td>E.A.2</td><td>Emission of microplastics</td><td>Use of single-use plastics</td><td>Air and water contamination</td><td>4A</td><td>Use of single-use plastics will be limited in LV design</td><td>4A</td></tr>
<tr><td>E.A.3</td><td>Chemical off-gassing</td><td>Working with HazMats</td><td rowspan="6">Air pollution</td><td>1B</td><td>HazMats that off-gas are only used in well-ventilated areas</td><td>1A</td></tr>
<tr><td>E.A.4</td><td rowspan="4">Smoke emission</td><td>CATO</td><td>2B</td><td>AeroTech Motors are used for their low likelihood of catastrophic failure</td><td>2A</td></tr>
<tr><td>E.A.5</td><td>Motor ignition</td><td>2B</td><td rowspan="2">Under nominal circimstances, LV operation produces few combustion byproducts</td><td>2A</td></tr>
<tr><td>E.A.6</td><td>Black powder detonation</td><td>1B</td><td>1A</td></tr>
<tr><td>E.A.7</td><td>Man-made wildfire</td><td>2D</td><td>Heat sources are not allowed within 25 feet of LV motors</td><td>2B</td></tr>
<tr><td>E.A.8</td><td>Creation of vaporized hydrochloric acid</td><td>APCP combustion byproduct contact with water</td><td>1B</td><td>AeroTech motors do not produce enough by-product to create hydrochloric acid</td><td>1B</td></tr>
</table>

Table 39: Environmental Factors FMEA

Label	Hazard	Cause	Effect	LS Before	Mitigation	LS After
Hazards to LV Structure						
E.S.1	LV contact with water	LV touchdown in irrigation ditch	Airframe structural damage	4C	The full scale LV is made of fiberglass, a water-resistant material	4B
E.S.2	Contact between LV and birds	Birds flying in proximity to LV flight path	Airframe abrasion/rupture	2B	Airways in the flight path of the LV are confirmed to be clear before flight by the RSO	2A
E.S.3	LV landing in tree	Large gusts of wind contributing to wind drift	Inability to recover rocket, mission failure	3D	Launches do not occur if wind speed at the launch field exceeds 20 mph	3C
Hazards to Personnel						
E.Pe.1	Personnel contact with sunlight and heat	Lack of personal protective equipment and devices Hot launch conditions	Heat stroke, sunburn	4B	Personnel are provided with sunscreen and are highly encouraged to bring sunglasses, a tent is set up for personnel to take shelter	2B
E.Pe.2	Personnel slips, trips, and falls	Uneven ground Sharp rocks on the ground Working near/next to irrigation ditches	Bruising, broken bones, concussion	4C	Personnel are required to wear closed toe shoes to launch day activities, only recovery and launch pad personnel are permitted on the launch field itself.	3C
E.Pe.3	Rain or hail	Inclement weather conditions	Rips, dents, holes in airframe, personnel injury	3C	Launches are not conducted during inclement weather. If inclement weather develops, the launch may be postponed and personnel will take shelter	3A
E.Pe.4	Lightning strike			1D		1A
E.Pe.5	Wet and icy terrain		Personnel slips, trips, and falls	2C		1C
Hazards to Payload System						
E.Pa.1	Payload contact with water	LV touchdown in irrigation ditch or other wet area	Payload electronics damage	1C	Mitigation pending	1C
E.Pa.2	Lightning strike	Inclement weather conditions		3C	Launches are not conducted in inclement weather	2A
Hazards to Mission Success						
E.M.1	Damp propellant grains	High humidity conditions	No motor ignition, inability of LV to fly	1D	Launches are not conducted in inclement weather	1B
E.M.2	Damp black powder grains			2D		1B
E.M.3	LV flight in proximity to bird flight	Birds flying in proximity to LV in flight	Diverted flight path, failure of LV to reach intended apogee	2B	Airways in the flight path of the LV are confirmed to be clear before flight by the RSO	2A

6 Project Plan

6.1 Requirements Verification

Table 40: NASA Requirements

NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
1.1	Students on the team SHALL do 100% of the project, including design, construction, written reports, presentations, and flight preparation with the exception of assembling the motors and handling black powder or any variant of ejection charges, or preparing and installing electric matches (to be done by the team's mentor). Teams SHALL submit new work. Excessive use of past work SHALL merit penalties.	The students of the High Powered Rocketry Club at NC State design and construct a solution to the requirements as listed in the Student Launch Handbook using past ideas and methods while also integrating new ideas.	Inspection	Project Management	Not Verified	Students complete the project using all original work performed only by the students.
1.2	The team SHALL provide and maintain a project plan to include, but not limited to the following items: project milestones, budget and community support, checklists, personnel assignments, STEM engagement events, and risks and mitigations.	The project management team, consisting of the team lead, vice president, treasurer, secretary, safety officer, webmaster, and social media lead manage the project planning tasks pertaining to this requirement.	Inspection	Project Management	Not Verified	See Section 6 for current project plan.
1.3	The team SHALL identify all team members who plan to attend Launch Week activities by the Critical Design Review (CDR).	The team lead identifies and reports the team members that attend the launch week by January 9, 2023, with the submission of CDR milestone documentation.	Inspection	Project Management	Not Verified	TBD
1.3.1	Team members attending competition SHALL include students actively engaged in the project throughout the entire year.	The project management team identifies the students that have been actively engaged throughout the year to be invited to launch week activities.	Inspection	Project Management	Not Verified	TBD
1.3.2	Team members SHALL include one mentor (see requirement 1.1.2).	The team lead invites the mentor listed in section 1.2 to attend launch week activities.	Inspection	Project Management	Not Verified	TBD
1.3.3	Team members SHALL include no more than two adult educators.	The team lead invites the adult educator listed in section 1.2 to attend launch week activities.	Inspection	Project Management	Not Verified	TBD
1.4	Teams SHALL engage a minimum of 250 participants in Educational Direct Engagement STEM activities in order to be eligible for STEM Engagement scoring and awards. These activities can be conducted in person or virtually. To satisfy this requirement, all events SHALL occur between project acceptance and the FRR due date. A template of the STEM Engagement Activity Report can be found on pages 39–42.	The outreach lead implements STEM engagement plans with K12 student groups throughout the project lifecycle and submits all STEM Engagement Activity Reports within two weeks of the event's conclusion.	Inspection	Project Management	Not Verified	TBD

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
1.5	The team SHALL establish and maintain a social media presence to inform the public about team activities.	The webmaster and social media officer cooperate to maintain our website and social media platforms to inform the public about all activities and events that the team performs throughout the year. Our social media platforms include, but are not limited to: our club website, Facebook, Instagram, and Twitter.	Inspection	Project Management	Verified	All forms of social media related to team activities have been sent to the NASA project management team.
1.6	Teams SHALL 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 SHALL be sufficient. Late submissions of PDR, CDR, FRR milestone documents SHALL be accepted up to 72 hours after the submission deadline. Late submissions SHALL incur an overall penalty. No PDR, CDR, FRR milestone documents SHALL be accepted beyond the 72-hour window. Teams that fail to submit the PDR, CDR, FRR milestone documents SHALL be eliminated from the project.	The team lead sends all deliverables to the NASA project management team prior to each specified deadline. In the event that the deliverable is too large, the webmaster posts the document on the team's website, and the team lead sends the NASA project management team a link to the file.	Inspection	Project Management	Not Verified	The team emails all deliverables to the NASA project management team by each specified deadline.
1.7	Teams who do not satisfactorily complete each milestone review (PDR, CDR, FRR) SHALL be provided action items needed to be completed following their review and SHALL be required to address action items in a delta review session. After the delta session the NASA management panel SHALL meet to determine the teams' status in the program and the team SHALL be notified shortly thereafter.	Team members complete and submit each milestone review document before the provided deadline. In the event that a document is not satisfactorily completed, the team completes the action items provided and attends the delta review session to maintain their status in the program.	Inspection	Project Management	Not Verified	The team completes a satisfactory milestone review document and submits before the deadline.
1.8	All deliverables SHALL be in PDF format.	The team lead converts all deliverables to PDF format prior to submission to the NASA project management team.	Inspection	Project Management	Verified	This report is submitted in PDF format.
1.9	In every report, teams SHALL provide a table of contents including major sections and their respective sub-sections.	The team lead creates and manages a Table of Contents in each milestone report.	Inspection	Project Management	Verified	See the Table of Contents above.
1.10	In every report, the team SHALL include the page number at the bottom of the page.	For each milestone report, the team uses a document template which includes page numbers at the bottom of each page.	Inspection	Project Management	Verified	The page number has been listed at the bottom of every page in this report.

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
1.11	The team SHALL 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.	Each team member participating in the video teleconference obtains the necessary equipment for them to perform a video teleconference with the review panel.	Inspection	Project Management	Not Verified	The team plans to provide their own equipment to engage in a video teleconference with the review panel.
1.12	All teams attending Launch Week SHALL be required to use the launch pads provided by Student Launch's launch services provider. No custom pads SHALL be permitted at the NASA Launch Complex. At launch, 8-foot 1010 rails and 12-foot 1515 rails SHALL be provided. The launch rails SHALL be canted 5 to 10 degrees away from the crowd on Launch Day. The exact cant SHALL depend on Launch Day wind conditions.	The aerodynamics lead designs a launch vehicle to be launched from either an 8-foot 1010 rail or a 12-foot 1515 rail. The structures lead fabricates the launch vehicle according to the aforementioned design.	Inspection	Aerodynamics; Structures	Not Verified	The team plans to use all provided equipment for Launch Day.
1.13	Each team SHALL identify a "mentor." A mentor is defined as an adult who is included as a team member, who SHALL be supporting the team (or multiple teams)throughout the project year, and may or may not be affiliated with the school, institution, or organization. The mentor SHALL maintain a current certification, and be in good standing, through the National Association of Rocketry (NAR) or Tripoli Rocketry Association (TRA) for the motor impulse of the launch vehicle and must have flown and successfully recovered (using electronic, staged recovery) a minimum of 2 flights in this or a higher impulse class, prior to PDR. The mentor is designated as the individual owner of the rocket for liability purposes and must travel with the team to Launch Week. One travel stipend SHALL be provided per mentor regardless of the number of teams he or she supports. The stipend SHALL only be provided if the team passes FRR and the team and mentor attend Launch Week in April.	The team lead identifies qualified community members to mentor team members throughout the design process.	Inspection	Project Management	Verified	See section 1.1.2
1.14	Teams SHALL track and report the number of hours spent working on each milestone.	The team reports the number of hours spent on each milestone in the associated milestone report.	Inspection	Project Management	Verified	See Section 1.1.3

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
2.1	The vehicle SHALL deliver the payload to an apogee altitude between 4,000 and 6,000 ft. above ground level (AGL). Teams flying below 3,500 ft. or above 6,500 ft. on their competition launch SHALL receive zero altitude points towards their overall project score and SHALL not be eligible for the Altitude Award.	The aerodynamics lead designs a launch vehicle to reach an apogee between 4,000 and 6,000ft. AGL. The team then constructs the vehicle as designed.	Analysis; Demonstration	Aerodynamics	Not Verified	See Section 3.5
2.2	Teams SHALL declare their target altitude goal at the PDR milestone. The declared target altitude SHALL be used to determine the team's altitude score.	The aerodynamics lead reports the team's target altitude goal in the PDR milestone report, submitted by October 26, 2022.	Inspection	Aerodynamics	Verified	See Section 3.5.1
2.3	The launch vehicle SHALL be designed to be recoverable and reusable. Reusable is defined as being able to launch again on the same day without repairs or modifications.	The recovery and structures leads design a recovery harness system that allows the launch vehicle to be recovered upon ground impact with minimal damage.	Demonstration	Recovery; Structures	Not Verified	See Section 3.4
2.4	The launch vehicle SHALL 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.	The aerodynamics and recovery leads design the vehicle to have a maximum of 4 independent sections.	Inspection	Aerodynamics; Recovery	Verified	See Section 3.4 regarding the launch vehicle design.
2.4.1	Coupler/airframe shoulders which are located at in-flight separation points SHALL be at least 2 airframe diameters in length. (One body diameter of surface contact with each airframe section).	The aerodynamics lead designs a airframe with couplers at in-flight separation points at least two airframe diameter in length. The structures lead constructs the couplers to the determined lengths	Inspection	Aerodynamics	Not Verified	See Section 3.3 regarding the leading launch vehicle design.
2.4.2	Nosecone shoulders which are located at in-flight separation points SHALL be at least ½ body diameter in length.	The aerodynamics lead designs the airframe such that nosecone shoulders at in-flight separation points are at least 1/2 body diameter in length.	Inspection	Aerodynamics	Verified	See section 3.3 regarding the leading launch vehicle design.
2.5	The launch vehicle SHALL be capable of being prepared for flight at the launch site within 2 hours of the time the Federal Aviation Administration flight waiver opens.	The project management and safety teams develop launch day checklists that can be executed in under two (2) hours.	Demonstration	Project Management: Safety	Not Verified	TBD
2.6	The launch vehicle and payload SHALL 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.	The project management and safety teams monitor the power consumption of each electrical launch vehicle and payload component and verify functionality of each component after two (2) hours.	Demostration	Project Management: Safety	Not Verified	TBD

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
2.7	The launch vehicle SHALL be capable of being launched by a standard 12-volt direct current firing system. The firing system SHALL be provided by the NASA-designated launch services provider.	The project management and safety teams choose a motor ignitor that can be ignited from a 12-volt direct current firing system.	Demonstration	Project Management; Safety	Not Verified	TBD
2.8	The launch vehicle SHALL require no external circuitry or special ground support equipment to initiate launch (other than what is provided by the launch services provider).	The project management and safety teams ensure the launch vehicle is designed such that no external circuitry or ground support equipment is required for launch.	Demonstration	Project Management; Safety	Not Verified	Currently, there is no plan to use external circuitry which can be seen in section 3.3.
2.9	Each team SHALL use commercially available ematches or igniters. Hand-dipped igniters SHALL not be permitted.	The project management and safety teams ensure proper purchase and use of commercially available ematches and igniters.	Inspection	Project Management; Safety	Not Verified	TBD
2.10	The launch vehicle SHALL 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).	The aerodynamics lead selects a commercially available solid motor propulsion system using APCP that is approved by NAR, TRA, and/or CAR for use in the launch vehicle.	Inspection	Aerodynamics	Not Verified	See section 1.2.2 regarding the motor selected.
2.10.1	Final motor choices SHALL be declared by the Critical Design Review (CDR) milestone.	The aerodynamics lead declares the team's final motor choice in the CDR milestone report by January 9, 2023.	Inspection	Aerodynamics	Not Verified	TBD
2.10.2	Any motor change after CDR SHALL be approved by the NASA Range Safety Officer (RSO). Changes for the sole purpose of altitude adjustment SHALL not be approved. A penalty against the team's overall score SHALL be incurred when a motor change is made after the CDR milestone, regardless of the reason.	The project management team requests approval from the NASA RSO for motor changes following submission of the CDR milestone report.	Inspection	Project Management	Not Verified	TBD
2.11	The launch vehicle SHALL be limited to a single motor propulsion system.	The aerodynamics lead designs the launch vehicle such that it only utilizes a single stage.	Inspection	Aerodynamics	Not Verified	See section 3.3 regarding the leading vehicle design.
2.12	The total impulse provided by a College or University launch vehicle SHALL not exceed 5,120 Newton-seconds (L-class).	The aerodynamics lead chooses a motor that does not exceed 5,120 Newton-seconds of total impulse.	Inspection	Aerodynamics	Not Verified	See section 1.2.2 for motor selection.
2.13	Pressure vessels on the vehicle SHALL be approved by the RSO.	The structures lead provides the necessary information on any onboard pressure vessels to the NASA RSO and home field RSO.	Inspection	Structures	Not Verified	TBD

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
2.13.1	The minimum factor of safety (Burst or Ultimate pressure versus Max Expected Operating Pressure) is 4:1 with supporting design documentation included in all milestone reviews.	The minimum factor of safety (Burst or Ultimate pressure versus Max Expected Operating Pressure) is 4:1 with supporting design documentation included in all milestone reviews.	Analysis; Inspection	Structures	Not Verified	TBD
2.13.2	Each pressure vessel SHALL 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.	The structures lead selects certain onboard pressure vessels such that they include a pressure relief valve that sees the full pressure of the tank and is more then capable of withstanding the maximum pressure and flow rate of the tank.	Analysis; Inspection	Structures	Not Verified	TBD
2.13.3	The full pedigree of the tank SHALL be described, including the application for which the tank was designed and the history of the tank. This SHALL 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.	The structures lead records the full history of each pressure vessel, including the number of pressure cycles, the dates of pressurization/depressurization, and the name of each person or entity administering the pressure events.	Inspection	Structures	Not Verified	TBD
2.14	The launch vehicle SHALL 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.	The aerodynamics lead designs the launch vehicle such that it has a minimum static stability margin of 2.0 at the point of rail exit.	Analysis	Aerodynamics	Not Verified	See section 3.5.5 regarding the projected stability margin.
2.15	The launch vehicle SHALL have a minimum thrust to weight ratio of 5.0 : 1.0.	The aerodynamics lead designs the launch vehicle such that it has a thrust to weight ratio of at least 5.0:1.0.	Analysis; Inspection	Aerodynamics	Not Verified	See section 1.2.2 regarding motor choice.
2.16	Any structural protuberance on the rocket SHALL be located aft of the burnout center of gravity. Camera housings SHALL be exempted, provided the team can show that the housing(s) causes minimal aerodynamic effect on the rocket's stability.	The aerodynamics lead designs the launch vehicle such that all structural protuberances are located aft of the burnout center of gravity. If any camera housings are included, the aerodynamics lead shows that the housings cause minimal aerodynamic effects on launch vehicle stability.	Analysis; Inspection	Aerodynamics	Not Verified	See section 3.3 regarding location of cupulas on the launch vehicle.
2.17	The launch vehicle SHALL accelerate to a minimum velocity of 52 fps at rail exit.	The aerodynamics lead designs the launch vehicle such that a velocity of 52 fps or greater is achieved by the launch vehicle at the rail exit.	Analysis	Aerodynamics	Not Verified	See section 1.2.2 regarding the projected velocity of the launch vehicle.

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
2.18	All teams SHALL successfully launch and recover a subscale model of their rocket prior to CDR. Success of the subscale is at the sole discretion of the NASA review panel. The subscale flight may be conducted at any time between proposal award and the CDR submission deadline. Subscale flight data SHALL be reported in the CDR report and presentation at the CDR milestone. Subscales are required to use a minimum motor impulse class of E (Mid-Power motor).	The management team launches a subscale model of the launch vehicle using an impulse motor of E or greater. The management and safety teams successfully recover the subscale model of the launch vehicle. The team reports subscale flight data in the CDR milestone report by January 9, 2023.	Demonstration	Project Management	Not Verified	See section 6.4 regarding the project timeline.
2.18.1	The subscale model should resemble and perform as similarly as possible to the full-scale model; however, the full-scale SHALL not be used as the subscale model.	The aerodynamics lead designs a unique subscale launch vehicle which performs similarly to the full- scale launch vehicle.	Inspection	Aerodynamics	Not Verified	TBD
2.18.2	The subscale model SHALL carry an altimeter capable of recording the model's apogee altitude.	The recovery lead installs an altimeter in the subscale launch vehicle capable of recording the vehicle's apogee altitude.	Inspection	Recovery	Not Verified	See Section 3.4.3 regarding selected altimeter.
2.18.3	The subscale rocket SHALL be a newly constructed rocket, designed and built specifically for this year's project.	The team constructs a new subscale launch vehicle, designed to meet the specifications for this year's project.	Inspection	Project Management	Not Verified	TBD
2.18.4	Proof of a successful flight SHALL be supplied in the CDR report.	The team includes proof of a successful subscale flight in the CDR milestone report by January 9,2023.	Inspection	Project Management	Not Verified	TBD
2.18.4.1	Altimeter flight profile graph(s) OR a quality video showing successful launch, recovery events, and landing as deemed by the NASA management panel are acceptable.	The recovery lead creates an altimeter flight profile graph that includes all altitudes recorded from liftoff through landing.	Analysis	Recovery	Not Verified	TBD
2.18.4.2	Quality pictures of the as landed configuration of all sections of the launch vehicle SHALL be included in the CDR report. This includes but not limited to nosecone, recovery system, airframe, and booster.	The recovery team takes pictures of the configuration of all sections of the launch vehicle after landing and include them in the CDR report to be submitted before January 9, 2023.	Analysis; Demonstration	Project Management; Recovery	Not Verified	TBD
2.18.5	The subscale rocket SHALL not exceed 75% of the dimensions (length and diameter) of your designed full-scale rocket. For example, if your full-scale rocket is a 4" diameter 100" length rocket your subscale SHALL not exceed 3" diameter and 75" in length.	The aerodynamics and structures leads design a subscale launch vehicle such that it does not exceed 75% of the dimensions of the designed full-scale launch vehicle.	Inspection	Aerodynamics; Structures	Not Verified	TBD

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
2.19.1	Vehicle Demonstration Flight—All teams SHALL successfully launch and recover their full-scale rocket prior to FRR in its final flight configuration. The rocket flown SHALL be the same rocket to be flown for their competition launch. The purpose of the Vehicle Demonstration Flight is to validate the launch vehicle's stability, structural integrity, recovery systems, and the team's ability to prepare the launch vehicle for flight. A successful flight is defined as a launch in which all hardware is functioning properly (i.e. drogue chute at apogee, main chute at the intended lower altitude, functioning tracking devices, etc.). The following criteria SHALL be met during the full-scale demonstration flight:	The team launches and recovers the full-scale launch vehicle in its final flight configuration prior to the FRR milestone.	Demonstration	Project Management	Not Verified	See section 6.4 for project timeline regarding planned VDF date.
2.19.1.1	The vehicle and recovery system SHALL have functioned as designed.	No anomalies are detected in the performance of the launch vehicle and its recovery system.	Demonstration	Project Management	Not Verified	TBD
2.19.1.2	The full-scale rocket SHALL be a newly constructed rocket, designed and built specifically for this year's project.	The team constructs a new full-scale launch vehicle that is designed and built according to the specifications for this year's project.	Inspection	Project Management; Aerodynamics	Not Verified	TBD
2.19.1.3.1	If the payload is not flown, mass simulators SHALL be used to simulate the payload mass.	If the payload is not flown during the VDF, the structures lead installs mass simulators to simulate intended payload mass.	Inspection	Structures	Not Verified	TBD
2.19.1.3.2	The mass simulators SHALL be located in the same approximate location on the rocket as the missing payload mass.	If the payload is not flown during the VDF, the structures lead installs mass simulators in the same approximate location on the launch vehicle as the missing payload mass.	Inspection	Structures	Not Verified	TBD
2.19.1.4	If the payload changes the external surfaces of the rocket (such as camera housings or external probes) or manages the total energy of the vehicle, those systems SHALL be active during the full-scale Vehicle Demonstration Flight.	If the payload changes the external surfaces or manages the total energy of the launch vehicle, the project management team activates those systems during the VDF.	Inspection	Project Management	Not Verified	TBD
2.19.1.5	Teams SHALL fly the competition launch motor for the Vehicle Demonstration Flight. The team may request a waiver for the use of an alternative motor in advance if the home launch field cannot support the full impulse of the competition launch motor or in other extenuating circumstances.	The aerodynamics lead installs the Launch Day motor for the VDF.	Inspection	Aerodynamics	Not Verified	TBD

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
2.19.1.6	The vehicle SHALL be flown in its fully ballasted configuration during the full-scale test flight. Fully ballasted refers to the maximum amount of ballast that SHALL be flown during the competition launch flight. Additional ballast may not be added without a re-flight of the full-scale launch vehicle.	The aerodynamics lead decides on the final ballast configuration. The structures lead installs all required ballast for the VDF.	Inspection	Structures; Aerodynamics	Not Verified	TBD
2.19.1.7	After successfully completing the full-scale demonstration flight, the launch vehicle or any of its components SHALL not be modified without the concurrence of the NASA Range Safety Officer (RSO).	After successful completion of the VDF, the project management team does not allow further modification of the launch vehicle or any of its components without approval from the NASA RSO.	Inspection	Project Management	Not Verified	TBD
2.19.1.8	Proof of a successful flight SHALL be supplied in the FRR report.	The project management team provides all proof of successful VDF in the FRR milestone report.	Inspection	Project Management	Not Verified	TBD
2.19.1.8.1	Altimeter flight profile data output with accompanying altitude and velocity versus time plots is required to meet this requirement. Altimeter flight profile graph(s) that are not complete (liftoff through landing) SHALL not be accepted.	The recovery lead includes all altimeter data from the VDF in the FRR milestone report.	Inspection	Recovery	Not Verified	TBD
2.19.1.8.2	Quality pictures of the as landed configuration of all sections of the launch vehicle SHALL be included in the FRR report. This includes but not limited to nosecone, recovery system, airframe, and booster.	The recovery lead includes all pictures of the landing configuration of the launch vehicle in the FRR milestone report.	Inspection	Recovery	Not Verified	TBD
2.19.1.9	Vehicle Demonstration flights SHALL be completed by the FRR submission deadline. No exceptions SHALL be made. If the Student Launch office determines that a Vehicle Demonstration Re-flight is necessary, then an extension may be granted. THIS EXTENSION IS ONLY VALID FOR RE-FLIGHTS, NOT FIRST TIME FLIGHTS. Teams completing a required re-flight SHALL submit an FRR Addendum by the FRR Addendum deadline.	The team completes the VDF by the FRR milestone report submission deadline. If a re-flight is required, the team submits an FRR addendum by the FRR addendum deadline.	Inspection	Project Management	Not Verified	TBD

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
2.19.2	Payload Demonstration Flight—All teams SHALL successfully launch and recover their full-scale rocket containing the completed payload prior to the Payload Demonstration Flight deadline. The rocket flown SHALL be the same rocket to be flown as their competition launch. The purpose of the Payload Demonstration Flight is to prove the launch vehicle's ability to safely retain the constructed payload during flight and to show that all aspects of the payload perform as designed. A successful flight is defined as a launch in which the rocket experiences stable ascent and the payload is fully retained until it is deployed (if applicable) as designed. The following criteria SHALL be met during the Payload Demonstration Flight:	The team successfully launches and recovers the full-scale launch vehicle containing the completed payload prior to the PDF deadline.	Inspection	Project Management	Not Verified	See section 6.4.1 for the project timeline regarding PDF deadline.
2.19.2.1	The payload SHALL be fully retained until the intended point of deployment (if applicable), all retention mechanisms SHALL function as designed, and the retention mechanism SHALL not sustain damage requiring repair.	The payload is fully retained until the point of intended deployment, with each retention mechanism functioning as designed and not sustaining damage requiring repair during the PDF.	Inspection	Integration	Not Verified	TBD
2.19.2.2	The payload flown SHALL be the final, active version.	The payload flown during the PDF is the final, active version of the payload.	Inspection	Project Management	Not Verified	TBD
2.19.2.3	If the above criteria are met during the original Vehicle Demonstration Flight, occurring prior to the FRR deadline and the information is included in the FRR package, the additional flight and FRR Addendum are not required.	The project management team ensures all criteria for the VDF are met and submitted before the FRR deadline. In the event that all criteria are not properly met, the team submits the additional flight and FRR addendum required.	Inspection	Project Management	Not Verified	TBD
2.19.2.4	Payload Demonstration Flights SHALL be completed by the FRR Addendum deadline. NO EXTENSIONS SHALL BE GRANTED.	The team completes the PDF by the FRR Addendum deadline.	Inspection	Project Management	Not Verfied	See section 6.4 for project timeline regarding PDF.
2.20	An FRR Addendum SHALL be required for any team completing a Payload Demonstration Flight or NASA required Vehicle Demonstration Re-flight after the submission of the FRR Report.	If the team is completing the PDF or a NASA- required VDF re-flight after the submission of the FRR Report, the team lead submits an FRR Addendum by the FRR Addendum deadline.	Inspection	Project Management	Not Verified	TBD

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
2.20.1	Teams required to complete a Vehicle Demonstration Re-Flight and failing to submit the FRR Addendum by the deadline SHALL not be permitted to fly a final competition launch.	The project management team manages the schedule to ensure that a PDF is successfully completed by the FRR Addendum deadline.	Inspection	Project Management	Not Verified	TBD
2.20.2	Teams who successfully complete a Vehicle Demonstration Flight but fail to qualify the payload by satisfactorily completing the Payload Demonstration Flight requirement SHALL not be permitted to fly a final competition launch.	The project management team successfully completes VDF and PDF before the FRR Addendum deadline.	Demonstration	Project Management	Not Verfied	TBD
2.20.3	Teams who complete a Payload Demonstration Flight which is not fully successful may petition the NASA RSO for permission to fly the payload at launch week. Permission SHALL not be granted if the RSO or the Review Panel have any safety concerns.	The project management team petitions the NASA RSO for permissions to fly the payload at launch week in the event that PDF is not fully successful.	Inspection	Project Management	Not Verified	TBD
2.21	The team's name and Launch Day contact information SHALL be in or on the rocket airframe as well as in or on any section of the vehicle that separates during flight and is not tethered to the main airframe. This information SHALL be included in a manner that allows the information to be retrieved without the need to open or separate the vehicle.	The project management team includes the team name and contact information on the launch vehicle such that it can be retrieved without the need to open or separate the vehicle.	Inspection	Project Management	Not Verified	TBD
2.22	All Lithium Polymer batteries SHALL be sufficiently protected from impact with the ground and SHALL be brightly colored, clearly marked as a fire hazard, and easily distinguishable from other payload hardware.	The project management and safety teams clearly mark all lithium polymer batteries as a fire hazard and ensure they are sufficiently protected from impact with the ground.	Analysis; Inspection	Project Management; Safety	Not Verified	TBD
2.23.1	The launch vehicle SHALL not utilize forward firing motors.	The aerodynamics lead designs the launch vehicle to not utilize forward firing motors.	Inspection	Aerodynamics	Not Verified	Currently, there are no plans to utilize forward firing motors which can be seen in section 3.3 for launch vehicle design.
2.23.2	The launch vehicle SHALL not utilize motors that expel titanium sponges (Sparky, Skidmark, MetalStorm, etc.).	The aerodynamics lead designs the launch vehicle to not utilize motors that are capable of expelling titanium sponges.	Inspection	Aerodynamics	Not Verified	See section 1.2.2 regarding motor selection.
2.23.3	The launch vehicle SHALL not utilize hybrid motors.	The aerodynamics lead designs a launch vehicle that does not utilize hybrid motors.	Inspection	Aerodynamics	Not Verified	See section 1.2.2 regarding motor selection.

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
2.23.4	The launch vehicle SHALL not utilize a cluster of motors.	The aerodynamics lead designs a launch vehicle such that it utilizes only one motor.	Inspection	Aerodynamics	Not Verified	See section 1.2.2 regarding motor selection.
2.23.5	The launch vehicle SHALL not utilize friction fitting for motors.	The structures lead designs a motor retention system such that it does not utilize friction fitting to hold the motor in place.	Inspection	Structures	Not Verified	See section 1.2.2 regarding vehicle design..
2.23.6	The launch vehicle SHALL not exceed Mach 1 at any point during flight.	The aerodynamics lead designs the launch vehicle such that it does not exceed Mach 1 at any point during the flight.	Analysis	Aerodynamics	Not Verified	See section 1.2.2 regarding vehicle velocity during flight.
2.23.7	Vehicle ballast SHALL 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).	The aerodynamics lead designs the launch vehicle such that the vehicle ballast does not exceed 10% of the total unballasted weight of the launch vehicle.	Analysis; Inspection	Aerodynamics	Not Verified	See section 1.2.2 regarding vehicle design.
2.23.8	Transmissions from onboard transmitters, which are active at any point prior to landing, SHALL not exceed 250 mW of power (per transmitter).	The recovery and payload leads select onboard transmitters that do not exceed 250mW of power for each transmitter.	Analysis	Recovery; Payload	Not Verified	TBD
2.23.9	Transmitters SHALL not create excessive interference. Teams SHALL utilize unique frequencies, handshake/passcode systems, or other means to mitigate interference caused to or received from other teams.	The recovery and payload leads choose a transmitter that creates minimal interference. The safety lead then enforces the usage of unique frequencies to mitigate interference with other teams.	Analysis; Demonstration	Safety; Recovery; Payload	Not Verified	TBD
2.23.10	Excessive and/or dense metal SHALL not be utilized in the construction of the vehicle. Use of lightweight metal SHALL be permitted but limited to the amount necessary to ensure structural integrity of the airframe under the expected operating stresses.	The structures lead designs the launch vehicle such that the amount of metal utilized in the construction of the vehicle is minimized.	Inspection	Structures	Not Verified	TBD
3.1	The full scale launch vehicle SHALL 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.	The recovery lead designs a dual-deployment recovery system.	Demonstration	Recovery	Not Verified	See section 3.4.3 for leading recovery system design.
3.1.1	The main parachute SHALL be deployed no lower than 500ft.	The recovery lead designs a recovery system that deploys the main parachute no lower than 500 ft.	Demonstration	Recovery	Not Verified	See section 3.4.3 for recovery system design.

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
3.1.2	The apogee event may contain a delay of no more than 2 seconds.	The recovery lead designs a recovery system that has an apogee event delay of no more than 2 seconds.	Demonstration	Recovery	Not Verified	See section 3.4.3 for recovery system design.
3.1.3	Motor ejection is not a permissible form of primary or secondary deployment.	The recovery lead designs a recovery system where the motor does not separate from the launch vehicle.	Inspection	Recovery	Not Verified	See section 3.4.3 for recovery system design.
3.2	Each team SHALL perform a successful ground ejection test for all electronically initiated recovery events prior to the initial flights of the subscale and full scale vehicles.	The recovery lead conducts ejection tests prior to each launch confirming electronics function properly.	Demonstration	Recovery	Not Verified	See section 3.4.3 for recovery system design.
3.3	Each independent section of the launch vehicle SHALL have a maximum kinetic energy of 75 ft-lbf at landing. Teams whose heaviest section of their launch vehicle, as verified by vehicle demonstration flight data, stays under 65 ft-lbf SHALL be awarded bonus points.	The recovery lead designs a recovery system such that the maximum kinetic energy experienced by the heaviest section of the launch vehicle does not exceed 65 ft-lbf.	Analysis	Recovery	Not Verified	See section 3.5.6 for kinetic energy calculations.
3.4	The recovery system SHALL contain redundant, commercially available barometric altimeters that are specifically designed for initiation of rocketry recovery events. The term "altimeters" includes both simple altimeters and more sophisticated flight computers.	The recovery lead designs a recovery system that utilizes a primary and secondary altimeter, each individually independent from the other.	Inspection	Recovery	Not Verified	See section 3.4.3 detailing altimeters.
3.5	Each altimeter SHALL have a dedicated power supply, and all recovery electronics SHALL be powered by commercially available batteries.	The recovery lead designs a recovery system that utilizes separate power sources for each altimeter used.	Inspection	Recovery	Not Verified	See section 3.4.3 for recovery system design.
3.6	Each altimeter SHALL 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.	The recovery lead designs a recovery system that utilizes pin switches to activate each altimeter accessible from the exterior of the launch vehicle.	Inspection	Recovery	Not Verified	See section 3.4.3 for recovery system design.
3.7	Each arming switch SHALL be capable of being locked in the ON position for launch (i.e. cannot be disarmed due to flight forces).	The recovery lead utilizes arming switches that can be locked in the ON position.	Inspection	Recovery	Not Verified	See section 3.4.3 for recovery system design.
3.8	The recovery system, GPS and altimeters, electrical circuits SHALL be completely independent of any payload electrical circuits.	The recovery lead designs a recovery system that ensures all recovery electronics are all independent of the payload electronics.	Inspection	Recovery	Not Verified	See section 3.4.3 for recovery system design.

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
3.9	Removable shear pins SHALL be used for both the main parachute compartment and the drogue parachute compartment.	The recovery lead designs a recovery system that utilizes removable shear pins to secure separable sections of the launch vehicle together on the pad.	Inspection	Recovery	Not Verified	See section 3.4.3 for recovery system design.
3.10	The recovery area SHALL be limited to a 2,500 ft. radius from the launch pads.	The recovery lead designs a recovery system that prevents the launch vehicle from drifting more than 2,500 ft. radius from the launch pad in launch pad condition.	Analysis; Demonstration	Recovery	Not Verified	See section 3.4.3 for recovery system design.
3.11	Descent time of the launch vehicle SHALL be limited to 90 seconds (apogee to touch down). Teams whose launch vehicle descent, as verified by vehicle demonstration flight data, stays under 80 seconds SHALL be awarded bonus points.	The recovery lead designs a recovery system that safely descends the launch vehicle in under 80 seconds.	Analysis; Demonstration	Recovery	Not Verified	See section 3.4.3 for recovery system design.
3.12	An electronic GPS tracking device SHALL be installed in the launch vehicle and SHALL transmit the position of the tethered vehicle or any independent section to a ground receiver.	The recovery lead designs a recovery system that utilizes a GPS tracking system that transmits the location of the launch vehicle at all points during flight.	Inspection; Demonstration	Recovery	Not Verified	See section 3.4.3 for recovery system design.
3.12.1	Any rocket section or payload component, which lands untethered to the launch vehicle, SHALL contain an active electronic GPS tracking device.	The recovery lead designs a GPS system and implements it on any payload component that lands separate from the launch vehicle.	Inspection	Recovery	Not Verified	See section 3.4.3 for recovery system design.
3.12.2	The electronic GPS tracking device(s) SHALL be fully functional during the official competition launch.	The recovery lead tests and ensures all GPS devices remain fully functional the day of official competition launch.	Inspection; Demonstration	Recovery	Not Verified	See section 3.4.3 for recovery system design.
3.13	The recovery system electronics SHALL not be adversely affected by any other on-board electronic devices during flight (from launch until landing).	The recovery lead designs a recovery system that recovery avionics are not affected by any other electronics onboard the launch vehicle.	Inspection; Demonstration	Recovery	Not Verified	See section 3.4.3 for recovery system design.
3.13.1	The recovery system altimeters SHALL be physically located in a separate compartment within the vehicle from any other radio frequency transmitting device and/or magnetic wave producing device.	The recovery lead designs an avionics bay that is physically located in a separate compartment than any other radio frequency transmitting or magnetic wave producing devices.	Inspection	Recovery	Not Verified	See section 3.4.3 for recovery system design.
3.13.2	The recovery system electronics SHALL be shielded from all onboard transmitting devices to avoid inadvertent excitation of the recovery system electronics.	The recovery lead designs an avionics bay that is shielded from other onboard transmitting devices.	Inspection	Recovery	Not Verified	See section 3.4.3 for recovery system design.

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
3.13.3	The recovery system electronics SHALL 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.	The recovery lead designs an avionics bay that is shielded from onboard magnetic wave generating devices.	Inspection	Recovery	Not Verified	See section 3.3 for launch vehicle design.
3.13.4	The recovery system electronics SHALL be shielded from any other onboard devices which may adversely affect the proper operation of the recovery system electronics.	The recovery lead designs an avionics bay that is shielded from any other onboard devices that may affect recovery system operations.	Inspection	Recovery	Not Verified	See section 3.4.3 for recovery system design.
4.1	Teams SHALL design a payload capable upon landing of autonomously receiving RF commands and performing a series of tasks with an on-board camera system. The method(s)/design(s) utilized to complete the payload mission SHALL be at the team's discretion and SHALL be permitted so long as the designs are deemed safe, obey FAA and legal requirements, and adhere to the intent of the challenge.	Payload team designs a payload system that is capable of receiving and interpreting RF commands and performing tasks with an onboard camera system, while obeying safety, FAA, and legal requirements, and adhering to the intent of the challenge.	Demonstration	Payload Systems; Payload Electronics; Payload Structures; Integration; Safety	Not Verified	See section 4.5 for leading payload design.
4.2.1	Launch Vehicle SHALL contain an automated camera system capable of swiveling 360º to take images of the entire surrounding area of the launch vehicle.	Payload system contains a camera system that is able to rotate 360º, take pictures, and is able to take pictures of the entire area surrounding the launch vehicle.	Demonstration	Payload Systems; Payload Electronics; Payload Structures; Integration	Not Verified	See section 4.5 for leading payload design.
4.2.1.1	The camera SHALL have the capability of rotating about the z axis. The z axis is perpendicular to the ground plane with the sky oriented up and the planetary surface oriented down.	Camera system rotational axis is about the described z axis.	Demonstration	Payload Systems; Payload Electronics; Payload Structures	Not Verified	TBD
4.2.1.2	The camera SHALL have a FOV of at least 100º and a maximum FOV of 180º.	Camera used in payload system has a FOV of at least 100º and at most 180º.	Inspection	Payload Electronics	Not Verified	TBD
4.2.1.3	The camera SHALL time stamp each photo taken. The time stamp SHALL be visible on all photos submitted to NASA in the PLAR.	Payload system adds a time stamp to each photo taken before saving.	Demonstration	Payload Electronics; Payload Systems	Not Verified	TBD
4.2.1.4	The camera system SHALL execute the string of transmitted commands quickly, with a maximum of 30 seconds between photos taken.	Camera system takes less than 30 seconds to execute commands between photos.	Demonstration	Payload Electronics; Payload Systems; Integration	Not Verified	TBD

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
4.2.2	NASA Student Launch Management Team SHALL transmit a RF sequence that SHALL contain a radio call sign followed by a sequence of tasks to be completed.	The payload system is able to determine if the correct call sign is used, and then accept and perform RF commands.	Demonstration	Payload Electronics; Payload Systems	Not Verified	TBD
4.2.3	The NASA Student Launch Management Panel SHALL transmit the RAFCO using APRS.	The payload system is able to accept RAFCO using APRS.	Demonstration	Payload Systems; Payload Electronics	Not Verified	TBD
4.2.3.1	The NASA Management Team SHALL transmit the RAFCO every 2 minutes.	The payload system is able to accept RAFCO commands continuously.	Demonstration	Payload Systems; Payload Electronics	Not Verified	TBD
4.2.3.3	The payload system SHALL not initiate and begin accepting RAFCO until AFTER the launch vehicle has landed on the planetary surface.	The payload system is designed such that it does not accept RAFCO until after launch vehicle landing.	Demonstration	Payload Systems; Payload Electronics	Not Verified	TBD
4.2.4	The payload SHALL not be jettisoned.	The payload system is designed such that no components are jettisoned.	Inspection	Payload Systems; Payload Electronics; Payload Structures; Integration	Not Verified	See section 4.5 for leading payload design.
4.2.5	The sequence of time-stamped photos taken need not be transmitted back to ground station and SHALL be presented in the correct order in your PLAR.	The sequence of time-stamped photos are presented in correct order in the teams PLAR.	Demonstration	Payload Systems; Payload Electronics	Not Verified	TBD
4.3.1	Black Powder and/or similar energetics are only permitted for deployment of in-flight recovery systems. Energetics SHALL not be permitted for any surface operations.	The payload recovery system is designed such that any energetics are only utilized in flight.	Inspection	Integration	Not Verified	See section 4.5 for leading payload design.
4.3.2	Teams SHALL abide by all FAA and NAR rules and regulations	The safety team verifies payload compliance with all FAA and NAR rules and regulations.	Demonstration	Safety	Not Verified	See section 5 regarding all safety rules and regulations.
4.3.6	Any UAS weighing more than .55 lbs. SHALL be registered with the FAA and the registration number marked on the vehicle.	Any UAS weighing more than .55 lbs is registered with the FAA and the registration number of the vehicle is marked.	Inspection	Payload Systems	Not Verified	TBD
5.1	Each team SHALL use a launch and safety checklist. The final checklists SHALL be included in the FRR report and used during the LRR and any Launch Day operations.	Checklists are included in the FRR and are used during LRR and Launch Day activities.	Validation of Records	All Subteams	Not Verified	TBD
5.2	Each team SHALL identify a student safety officer who SHALL be responsible for all items in section 5.3.	The student safety officer, Megan Rink, upholds all responsibilities detailed in safety requirement 5.3.	Validation of Records	Safety	Verified	TBD

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
5.3.1.1	The safety officer SHALL monitor team activities with an emphasis on safety during design of vehicle and payload.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.1.2	The safety officer SHALL monitor team activities with an emphasis on safety during construction of vehicle and payload components.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.1.3	The safety officer SHALL monitor team activities with an emphasis on safety during assembly of vehicle and payload.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.1.4	The safety officer SHALL monitor team activities with an emphasis on safety during ground testing of vehicle and payload.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.1.5	The safety officer SHALL monitor team activities with an emphasis on safety during subscale launch test(s).	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.1.6	The safety officer SHALL monitor team activities with an emphasis on safety during full-scale launch test(s).	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.1.7	The safety officer SHALL monitor team activities with an emphasis on safety during competition launch.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.1.8	The safety officer SHALL monitor team activities with an emphasis on safety during recovery activities.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.1.9	The safety officer SHALL monitor team activities with an emphasis on safety during STEM engagement activities.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety	Not Verified	TBD
5.3.2	The safety officer SHALL implement procedures developed by the team for construction, assembly, launch, and recovery activities.	The safety team writes and implements procedures and checklists for assembling, launching, and recovering the launch vehicle.	Demonstration	Safety	Not Verified	TBD
5.3.3	The safety officer SHALL manage and maintain current revisions of the team's hazard analyses, failure modes analyses, procedures, and MSDS/-chemical inventory data.	The student safety officer manages all safety documentation for the team.	Inspection	Safety	Verified	TBD

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NASA Req No	SHALL Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
5.4	During test flights, teams SHALL abide by the rules and guidance of the local rocketry club's RSO. The allowance of certain vehicle configurations and/or payloads at the NASA Student Launch does not give explicit or implicit authority for teams to fly those vehicle configurations and/or payloads at other club launches. Teams should communicate their intentions to the local club's President or Prefect and RSO before attending any NAR or TRA launch.	The safety team ensures all rules and regulations from the local rocketry club are followed by all team members.	Demonstration	Safety	Not Verified	TBD
5.5	Teams SHALL abide by all rules set forth by the FAA.	The safety team ensures all rules from FAA are followed.	Demonstration	All Subteams	Verified	TBD

Table 41: Team Derived Requirements: Safety

ID	Description	Justification	Success Criteria	Verification Method	Status	Status Description
Functional Requirements						
SDR 1	All epoxy SHALL be left to cure for at least 24 hours before a load is applied.	Uncured epoxy weakens the structural stability of the launch vehicle, which increases the likelihood of structural failure.	Epoxied parts are labeled and remain untouched until the date and time listed on their label.	Inspection	Not Verified	Current manufacturing procedures specify at least a 24-hour cure period for all epoxied parts.
SDR 2	Safety glasses SHALL be provided to each personnel working with or around power tools.	Using PPE reduces the risk of skin and eye injury from debris in the air due to power tool operation.	Safety glasses for every working team member of HPRC are contained in the lab's PPE cabinet.	Inspection	Not Verified	There are 25 pairs of safety glasses in the PPE cabinet which exceeds lab capacity.
SDR 3	Nitrile gloves, safety glasses, and particulate masks SHALL be provided to all personnel working with volatile liquid and/or powder chemicals.	Using PPE reduces the risk of skin and eye injury from debris caused by volatile liquids and/or powders.	Gloves, glasses, and masks for every working team member are contained in the lab's PPE cabinet.	Inspection	Not Verified	There are 25 glasses, 8 boxes of nitrile gloves, and 3 cases of masks in the lab's PPE cabinet which exceeds lab capacity.
SDR 4	All launch day attendees SHALL maintain a walking pace at all times on the launch field, including during assembly, launch, and recovery of the launch vehicle.	Maintaining a steady walking pace decreases the risk of slipping, tripping, and falling.	Team members will maintain a walking pace at all times during launch day.	Inspection	Not Verified	Team members will be briefed before launch on launch field etiquette.
SDR 5	Hazards identified as orange or red in the risk assessment matrix SHALL be decreased to yellow or green in the CDR through mitigations.	Mitigating potentially dangerous and/or frequent hazards provides a more robust launch vehicle and payload system.	All hazards identified in the CDR document will fall in the yellow or green zones after mitigation is applied.	Inspection	Not Verified	% of current hazards fall in the green or yellow zones.
SDR 6	All hazardous/flammable liquids and/or powder chemicals will be stored in a designated flame cabinet whenever it is not being used.	Storing all hazardous liquids in a fireproof cabinet decreases the risk of injury to students and damage to lab equipment.	All hazardous liquids remain in the flame cabinet until they are used by team members. After use, the liquids are immediately returned.	Inspection	Not Verified	All hardeners, resins, lubricants, cleaners, aerosol paints, black powder, oxidizers, and igniters used by the team are stored in a JUSTRITE Flammable Liquid Storage Cabinet.

Table 42: Team Derived Requirements: Launch Vehicle

ID	Description	Justification	Success Criteria	Verification Method	Status	Status Description
Functional Requirements						
LVF 1	The launch vehicle SHALL not exceed a velocity of Mach 0.7.	Exceedingly high speeds and acceleration undergone by the launch vehicle will increase the risk of damage to the payload other structural components inside the launch vehicle.	Simulations are completed in RockSim to calculate the launch vehicle's maximum velocity.	Analysis	Verified	The calculated maximum velocity of the launch vehicle is Mach 0.34. See section 3.5 for mission performance predictions.
LVF 2	The launch vehicle SHALL no exceed an acceleration of 14Gs during its ascent.	Accelerations undergone by the launch vehicle that are higher than 14Gs increase the risk for damage to both the payload and the structural components inside the launch vehicle.	Simulations are completed in RockSim to calculate the launch vehicle's maximum acceleration during its ascent.	Analysis	Verified	The maximum acceleration during flight of the launch vehicle is calculated to be 9Gs. See section 3.5 for mission performance predictions.
Design Requirements						
LVD 1	All structural components of the launch vehicle SHALL be designed with a minimum factor of safety of 1.5.	This ensures that the launch vehicle will remain structurally stable during flight despite experiencing higher than expected loads. Additionally, it prevents unexpected failure of the launch vehicle during flight.	The factor of safety of each critical component is reported in the documentation and calculated by structural analysis and testing.	Analysis; Testing	Not Verified	The factor of safety is calculated through simulations and testing. See section 3.3.15 for the simulation details. Future documentation will include additional simulations and testing.
LVD 2	The inner diameter of the launch vehicle SHALL be no larger than 6 in.	Limiting the size of the launch vehicle makes it easier to construct, cuts down on weight and decreases aerodynamic drag.	The inner diameter of the airframe is no larger than 6 in.	Inspection	Not Verified	See section 3.3.1 for launch vehicle size.
LVD 3	The launch vehicle SHALL have symmetrical fins.	Ensures that the launch vehicle is aerodynamic. Also ensures that the CG is on the center with equal aerodynamic forces on each side and an equal weight distribution.	Launch vehicle has four fins, equally spaced from each other around the airframe.	Inspection	Not Verified	See Section 3.3 for fin design.
LVD 4	The launch vehicle SHALL use at least 2 centering rings to support the motor tube.	Ensures that the motor tube has the adequate support to handle the high force caused by the motor during launch.	The launch vehicle has three centering rings to support the motor tube during flight.	Inspection	Not Verified	See section 3.3.10 for leading fin can design.

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ID	Description	Justification	Success Criteria	Verification Method	Status	Status Description
LVD 5	The launch vehicle SHALL have a stability margin between 2 and 2.7 upon rail exit.	To meet the NASA requirement 2.14, a stability margin of 2.0 or greater is required. The maximum value of 2.7 was selected because excessively high stability margins cause undesirable weather cocking of the launch vehicle in high winds.	The calculated launch vehicle's stability margin is calculated to be between 2.0 and 2.7.	Analysis	Not Verified	See section 3.5.6 for the stability margin analysis of the leading launch vehicle design.
LVD 6	Each separated section of the launch vehicle SHALL have a connection point for a shock cord capable of sustaining the maximum loads experienced in flight to our defined minimum safety factor of 2.	The nose cone, avionics bay, payload bay, and fin can are all tethered to the launch vehicle and need to withstand loads during flight without completely separating from the launch vehicle.	ANSYS simulations and structural tests are done on the bulkheads and hardware of each section to confirm the factor of safety.	Analysis; Inspection	Not Verified	See section 3.3.15 for the bulkhead calculations. Additional testing and analysis will be presented in future documentation.
LVD 7	The launch vehicle blast caps SHALL be exposed accessible.	Accessible energetics allow for safer and easier installation of black powder charges.	The avionics bay is designed to have blast caps that are easily accessible.	Inspection	Not Verified	See section 3.4.3.1 for the leading avionics bay design.
Environmental Requirements						
LVE 1	The airframe of the launch vehicle SHALL be capable of launching in temperatures between 20 and 100 degrees Fahrenheit.	The launch vehicle will be used in a variety of launch fields and seasons, including winter in North Carolina and spring in Alabama.	The airframe material is rated to remain undamaged and undeformed under the stated temperatures.	Inspection; Analysis	Not Verified	See section 3.3.3 for the selected material for the leading launch vehicle design.
LVE 2	The launch vehicle SHALL be water-resistant.	The team's home launch field contains several irrigation ditches that are often filled with water. Having a water resistant airframe will help mitigate any potential damage as a result of the launch vehicle landing in one of the ditches. Additionally, this will help protect the structural integrity of the launch vehicle in situations involving high humidity or rainfall.	The airframe is not damaged nor deformed upon exposure to water.	Inspection; Demonstration	Not Verified	See section 3.3.3 for the selected material for the leading launch vehicle design.

Table 43: Team Derived Requirements: Payload

ID	Description	Justification	Success Criteria	Verification Method	Status	Status Description
Functional Requirements						
PF 1	Each camera SHALL have a field of view greater than 120 degrees and less than 180 degrees.	The camera system must take an image of the launch vehicle's surroundings. Increasing the required minimum field of view mitigates the impact on image quality as a result of unforeseen obstructions.	The cameras used will have a field of view greater than 120 degrees and less than 180 degrees.	Analysis; Demonstration	Not Verified	TBD
PF 2	All electronic components in the launch vehicle SHALL be removable.	Removable electronics allow for easier adjustments to the payload design.	None of the electronic components in the launch vehicle are permanently fixed in place.	Inspection; Testing; Demonstration	Not Verified	Current electronic design has not been completed. Further design and testing will be presented in future documentation.
PF 3	The RTL-SDR dongle SHALL only accept RF commands from one antenna.	Antennas that are connected in parallel interfere with each others' signals and decrease signal quality.	When RAFCO commands are being given, a relay switch allows for input from the upward- facing antenna to be sent to the RTL-SDR dongle which is then sent to the Raspberry Pi.	Testing; Demonstration	Not Verified	See section 4.5 regarding the leading payload design. Further design and testing will be presented in future documentation.
Design Requirements						
PD 1	SOCS SHALL have a combined weight of no more than 8 lb.	A weight limit for all of the payload components allows for the launch vehicle to reach the desired altitude.	SOCS has a maximum combined weight of 8 lb.	Inspection	Not Verified	See section 3.5.3 for payload weight estimations.
PD 2	SOCS SHALL have 4 symmetrical housings placed equidistant around the launch vehicle and centered at the midpoint between each fin.	Each camera being placed at the midpoint between two fins ensures that one camera will always be facing upward regardless of landing orientation. This design allows mitigates the obstruction of the fins in the camera's field of view.	SOCS will consists of four cameras and housings. Each housing is placed at the midpoint between two fins.	Inspection	Not Verified	See section 4.5 for leading payload design.

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ID	Description	Justification	Success Criteria	Verification Method	Status	Status Description
PD 3	The antenna SHALL face upwards upon landing regardless of landing orientation.	An upward-facing antenna prevents structural damage occurring to the antenna as a result of high-energy impact with the ground. Additionally, it mitigates obstructions from the terrain surrounding the launch vehicle that might interfere with receiving RF commands.	Two antennas will be used on the launch vehicle. They will be placed along the leading edge of an opposing fin such that regardless of landing orientation, there will always be one antenna upward-facing.	Inspection	Not Verified	TBD
PD 4	Each housing SHALL withstand all loads encountered during the flight and landing of the launch vehicle.	Housings are the only structural component protecting the cameras in the SOCS. Each housing must be able to withstand the loads encountered during the flight and landing of the launch vehicle to mitigate any damages to the cameras.	Each housing is designed to withstand the loads encountered during flight and landing of the launch vehicle.	Analysis; Testing	Not Verified	TBD
PD 5	Each housing SHALL extrude no more than 1 in. from the launch vehicle.	Housings that extrude further than 1in. from the launch vehicle add unnecessary weight and aerodynamic drag to the launch vehicle.	Each housing is designed such that it does not extrude more than 1in. from the outside of the launch vehicle.	Analysis; Inspection	Not Verified	See section 4.5 regarding leading payload design.
Environmental Requirements						
PE 1	Each housing in SOCS SHALL be water-tight.	The team's home launch field contains several irrigation ditches that are often filled with water. Having a water resistant housing will help mitigate any potential damage to the cameras as a result of the launch vehicle landing in one of the ditches.	Each camera in SOCS is not damaged as a result of the launch vehicle being exposed to water.	Inspection Demonstration	Not Verified	See section 4.5 regarding leading payload design.

Table 44: Team Derived Requirements: Recovery

ID	Description	Justification	Success Criteria	Verification Method	Status	Status Description
Functional Requirements						
RF 1	The descent velocity under drogue SHALL be less than 120ft/s.	High descent velocities under drogue parachute lead to a larger load on main parachute when it deploys.	The chosen parachute will have a terminal velocity of less than 120ft/s with the given mass of the rocket.	Analysis	Not Verified	The selected drogue parachute has a descent velocity of 111. 6ft/s.
RF 2	The secondary black powder charges SHALL be larger than the primary black powder charges.	The secondary black powder charges are used to separate the sections of the launch vehicle should the main charge not generate enough pressure to initially separate the sections. The secondary charges need to be larger than the initial charges to ensure the sections are completely separated during flight.	The amount of black powder added to the secondary blast cap will be greater than the amount added to the primary blast cap.	Inspection	Not Verified	See section 3.4.2.9 regarding ejection charge sizing.
RF 3	Fully charged 9V batteries SHALL be used for the altimeters before every flight.	Black powder might not be properly ignited if there is insufficient voltage to the blast cap.	Batteries will be verified to be fully charged at 9 volts before being placed on the AV sled.	Inspection; Analysis	Not Verified	See section 3.4.1 regarding recovery procedures.
Design Requirements						
RD 1	Only U-Bolts SHALL be used for all shock cord connections.	U-Bolts are designed to provide two points where shock can go through the bulkhead. Dispersing the shock through multiple points increases the bulkhead stability.	U-Bolts are installed in every bulkhead that is used as an anchor point for the recovery harness.	Inspection	Not Verified	See section 3.3 for leading launch vehicle design.
RD 2	Threaded quick-links SHALL be used to attach all recovery harnesses to the launch vehicle attachment points.	Threaded quick-links are very unlikely to detach during flight. Due to their design, they are very easy to attach around the U-Bolt.	Quick-links will be used to attach all recovery harnesses to their respective anchor points.	Inspection	Not Verified	See section 3.4.1 regarding recovery description.
RD 3	Nomex cloth SHALL be used to protect the main parachute from ejection gases.	Ejection gases will burn/melt the fabric of the main parachute upon exposure, causing the parachute to fail.	The main parachute will be folded and stored inside a Nomex cloth before being attached to the main shock cord inside the main parachute bay.	Inspection	Not Verified	See section 3.4.1 regarding use of Nomex cloth.
Environmental Requirements						
RE 1	All protective insulation SHALL be biodegradeable.	Insulation protecting the shock cords and parachutes might fall out during/after flight. Since it is hard to prevent all of the insulation from falling out, we require insulation that does not negatively impact the environment.	Insulating used in all parachute bays will be checked to ensure it is biodegradable.	Inspection	Not Verified	See section 3.4.1 regarding insulation.

6.2 Budget

Table 45 below details the year-long budget for the 2022-2023 Student Launch Competition.

Table 45: 2022-2023 NASA Student Launch Competition Budget

	Item	Quantity	Price Per Unit	Item Total
Subscale Structure	Plastic 4 in. 4:1 Ogive Nosecone	1	$ 29.80	$ 29.80
	4 in. Blue Tube	2	$ 43.95	$ 87.90
	4 in. Blue Tube Pre-Slotted	1	$ 53.50	$ 53.50
	AeroTech J420R-14 Motor	2	$ 93.08	$ 186.16
	Aero Pack 38mm Retainer	2	$ 29.17	$ 29.17
	AeroTech RMS-38/600 Motor Casing	1	$ 112.34	$ 112.34
	Large Rail Button -1515	1	$ 7.87	$ 7.87
	Standard Rail Button - 1010	2	$ 4.25	$ 8.50
	Blast Caps	4	$ 1.80	$ 7.20
	Terminal Blocks	4	$ 3.00	$ 12.00
	Double Pull Pin Switch	2	$ 11.95	$ 23.90
	Subtotal:			**$ 611.58**
Full Scale Structure	6 in. Nosecone Fiberglass Ogive 4:1	1	$ 149.99	$ 149.99
	6 in. G12 Fiberglass Tube (60 in.)	1	$ 259.00	$ 259.00
	6 in. G12 Fiberglass Tube (48 in.)	1	$ 207.20	$ 207.20
	6 in. G12 Fiberglass Coupler	4	$ 77.50	$ 310.00
	AeroTech High-Power L1390G-P Motor	2	$ 223.54	$ 447.08
	Aero Pack 75mm Retainer	1	$ 59.50	$ 59.50
	AeroTech RMS-75/3840 Motor Casing	1	$ 526.45	$ 526.45
	Large Rail Button -1515	2	$ 4.25	$ 11.40
	U-Bolts	8	$ 1.00	$ 8.00
	Blast Caps	4	$ 1.80	$ 7.20
	Terminal Blocks	4	$ 3.00	$ 12.00
	Double Pull Pin Switch	2	$ 11.95	$ 23.90
	Subtotal:			**$ 2021.72**
Payload	FPV Cameras	4	$ 19.99	$ 79.99
	Acrylic Sheets	2	$ 11.22	$ 44.88
	Stepper Motor	2	$ 173.97	$ 347.97
	Raspberry Pi	1	$ 120.00	$ 120.00
	IMU	1	$ 15.30	$ 15.30
	RTL-SDR Dongle	1	$ 39.95	$ 39.95
	Whip Antenna	4	$ 25.00	$ 100.00
	Servo	8	$ 10.00	$ 80.00
	Subtotal:			**$ 805.59**
	Iris Ultra 120 in. Standard Parachute	1	$ 475.71	$ 475.71
	Iris Ultra 60 in. Standard Parachute	1	$ 212.85	$ 212.85
	18 in. Compact Elliptical Parachute	1	$ 70.95	$ 70.95
	RRC3 Sport Altimeter	4	$ 96.50	$ 386.00
	Eggfinder TX Transmitter	1	$ 70.00	$ 70.00
	6 in. Deployment Bag	1	$ 54.40	$ 54.40
	4 in. Deployment Bag	1	$ 47.30	$ 47.30
	18 in. Nomex Cloth	1	$ 26.40	$ 26.40
	13 in. Nomex Cloth	1	$ 17.60	$ 17.60
	5/8 in. Kevlar Shock Cord (per yard)	25	$ 6.99	$ 174.75
	3/16 in. Stainless Steel Quick Links	16	$ 2.00	$ 32.00

	AeroTech Ejection Charge - 1.4g	24	$ 1.25	$ 30.00
	Subtotal:			**$ 1,702.72**
Miscellaneous	Paint	12	$ 18.00	$ 216.00
	Domestic Birch Plywood 1/8 in.x2x2	12	$ 14.82	$ 177.84
	West Systems 105 Epoxy Resin	2	$ 109.99	$ 219.98
	West Systems 206 Slow Hardener	2	$ 62.99	$ 125.98
	ABS 3D Printer Filament Spool	1	$ 23.00	$ 23.00
	ClearWeld Quick Dry 2-Part Epoxy	1	$ 20.28	$ 20.28
	Wood Glue	1	$ 7.98	$ 7.98
	Misc. Bolts	1	$ 20.00	$ 20.00
	Misc. Nuts	1	$ 10.00	$ 10.00
	Misc. Washers	1	$ 8.00	$ 8.00
	Tinned Copper Wire Kit	1	$ 25.00	$ 12.00
	Zip Ties Pack	1	$ 6.59	$ 6.59
	Hook and Loop Strips Box	1	$ 10.00	$ 10.00
	9V Battery Pack	1	$ 12.00	$ 12.00
	Misc. Tape	1	$ 20.00	$ 20.00
	Estimated Shipping			$ 1,000.00
	Incidentals (replacement tools, hardware, safety equipment, etc.)			$ 1,500.00
	Subtotal:			**$ 3,410.63**
Travel	Student Hotel Rooms – 4 nights (# Rooms)	8	$ 898.45	$ 7,187.60
	Mentor Hotel Rooms – 4 nights (# Rooms)	2	$ 1022.03	$ 2,044.06
	NCSU Van Rental (# Vans)	1	$ 798.00	$ 2,394.00
	Subtotal:			**$ 11,625.66**
Promotion	T-Shirts	40	$ 15.00	$ 600.00
	Polos	15	$ 25.00	$ 375.00
	Stickers	500	$ 0.43	$ 215.00
	Subtotal:			**$ 1,190.00**
Total Expenses:				**$ 21,356.86**

As highlighted in Figure 82, our expenses can be divided into different sub-sections with travel funds taking up the majority of our spending for this year.

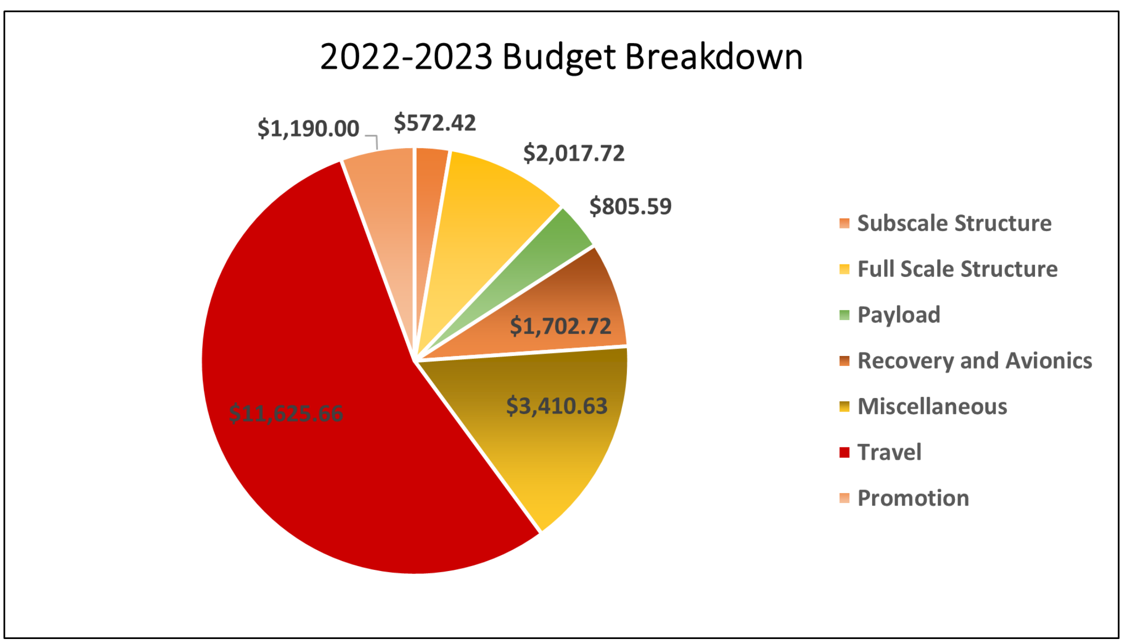

Figure 82: Budget breakdown of the 2022 - 2023 competition year.

6.3 Funding Plan

HPRC receives the majority of funding from a variety of NC State's funding sources, as well as North Carolina Space Grant (NCSG). Below is an in depth breakdown of the team's current funding sources.

NC State's Student Government Association's (SGA) Appropriations Committee is responsible for distributing university funding to nearly 600 different organizations on campus. Each semester the application process persists of a proposal where we outline what we are requesting from SGA, how much money we estimate to receive from other sources, and our anticipated club expenses for the academic year. We then meet with representative from SGA and give a presentation outlining our club activities and how we benefit the university. SGA then collectively allocates money to each organization on campus. In the 2021-2022 academic year, HPRC received $1,766.81 from SGA: $451.81 in the fall semester and $1,315 in the spring semester. For this academic year, a request of $2,000 was submitted for the fall semester and another $2,000 request will be submitted in the spring semester, assuming SGA regulations and budget will remain the same.

The Educational and Technology Fee (ETF) is an NC State University fund that allocates funding for academic enhancement through student organizations. In the 2021-2022 academic year, we received $3,000 from ETF and the team anticipate to receive $2,500 for this academic year. This funding will be used primarily to pay for the student and team's faculty advisors' travel costs.

Student and mentor travel costs will primarily be covered by NC State's College of Engineering Enhancement Funds. These funds come from a pool of money dedicated to supporting engineering extracurricular activities at NC State. Based on the 2021-2022 academic year, it is estimated we will receive $7,500.

In addition to funding through NC State organizations, North Carolina Space Grant is a large source of HPRC's funds. NCSG accepts funding proposals during the fall semester and teams can request up to $5,000 for participation in NASA competitions. NCSG will review the proposal and inform the club of the amount awarded. In previous academic years, this has been the maximum amount of $5,000, which will be available for use starting November 2022.

In the past, HPRC has held sponsorships with Collins Aerospace, Jolly Logic, Fruity Chutes, and more. The team is currently seeking out new sponsorships and reaching out to past sponsors. The team has found that companies are more likely to donate gifts in kind rather than provide monetary sponsorship. The team estimates to receive $2,500 in gifts of kind this academic year.

These totals are listed in Table 46 below, which outlines the projected costs and incoming revenue for the 2021-2022 academic year.

Table 46: Projected Funding Sources

Organization	Fall Semester	Spring Semester	Academic Year
Educational and Technology Fee	$0	$2,500	$2,500
Engineering Enhancement Fund	$0	$7,500	$7,500
NC State Student Government	$2,000	$2,000	$4,000
North Carolina Space Grant	$5,000	$0	$5,000
Sponsorship	$1.000	$1,500	$2,500
Total Funding:			**$21,500.00**
Total Expenses:			**$21,317.70**
Difference:			**$182.30**

6.4 Project Timelines

6.4.1 Competition Deliverables Timeline

Table 47: Competition Deadlines

Event/Task	Start Date	End Date/Submission
Request for Proposal Released	Aug. 17, 2022	N/A
Proposal Submission	Aug. 17, 2022	Sep. 19, 2022, 8:00 a.m. CST
PDR Submission	Oct. 26th, 2022	Oct. 26, 2022, 8:00 am CST
PDR Team Teleconference	(Tentative) Nov. 01 - 21, 2022	
Subscale Launch Opportunity	Nov. 19, 2022	Jan. 09, 2023
CDR Submission	Dec. 05, 2022	Jan. 09, 2023, 8:00 am CST
CDR Team Teleconference	(Tentative) Jan. 17 – Feb. 07, 2023	
Full-Scale Launch Opportunity	Feb. 18, 2023	Mar. 06, 2023
Final Launch Vehicle Design RockSim file submission	Dec. 05, 2022	Mar. 06, 2023, 8:00 am CST
FRR Submission	Jan. 26, 2023	Mar. 06, 2023, 8:00 am CST
FRR Team Teleconference	(Tentative) Mar. 13 - 31, 2023	
Payload/Vehicle Demonstration Re-Flight (if needed)	Mar. 13 2023	Apr. 03, 2023
FRR Addendum (if needed)	Mar. 13 2023	Apr. 03, 2023, 8:00 am CST
Team Travel to Huntsville, AL	Apr. 12, 2023	N/A
Launch Readiness Review	Apr. 12, 2023	N/A
NASA Safety Briefing	Apr. 13, 2023	N/A
Rocket Fair and MSFC Tours	Apr. 14, 2023	N/A
Launch Days	Apr. 15, 2023	Apr. 16, 2023
Post-Launch Assessment Review	Apr. 15, 2023	May 01, 2023, 8:00 am CDT

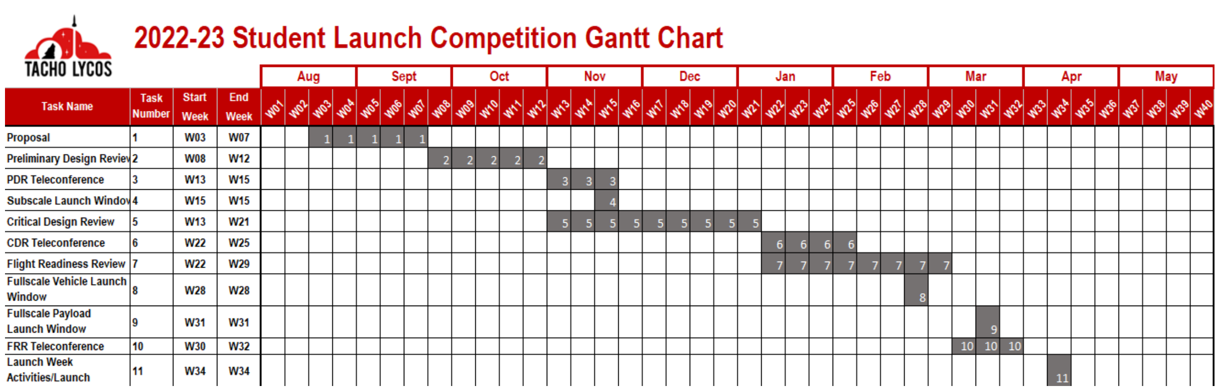

Figure 83: A Gantt chart containing the official NASA competition deadlines.

6.4.2 Funding Timeline

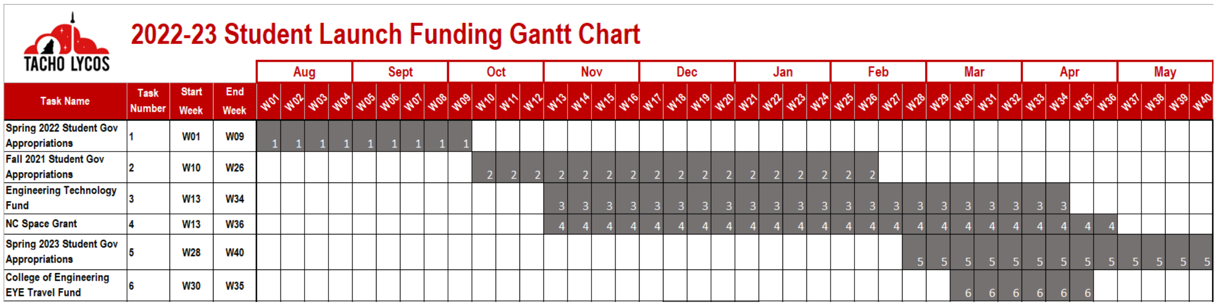

Figure 84: A Gantt chart detailing when our funding sources are available for use.

6.4.3 Development Timelines

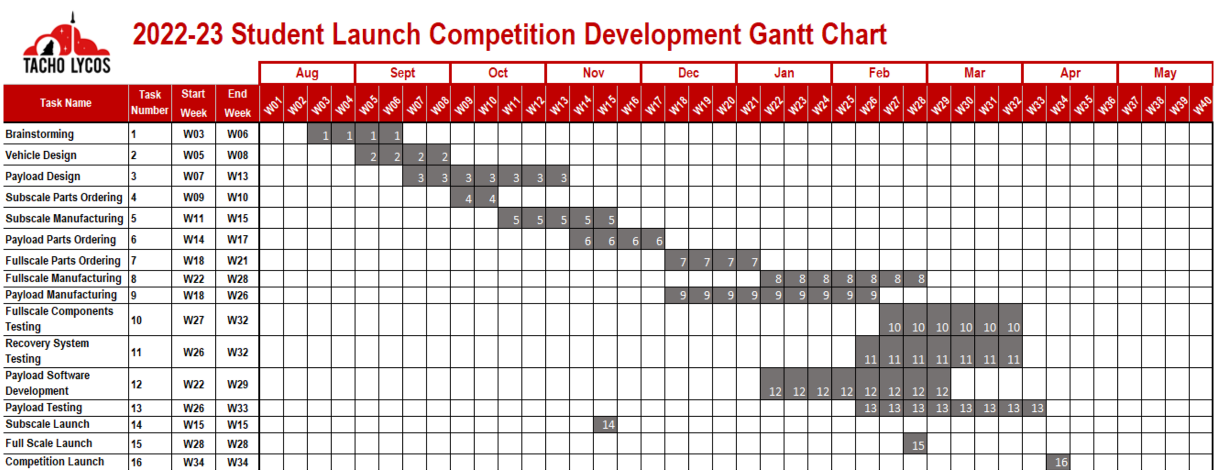

Figure 85: A Gantt chart detailing launch vehicle and payload development schedules.

Table 48: Weekly Club Schedule

Monday	3-5 pm: Vehicle Subteam Meeting- joint meeting for aerodynamics, structures, and recovery subteams
Tuesday	6:30-7:30 pm: WolfWorks Experimental- Club experimental projects unrelated to student launch, currently working on airbrakes system
Wednesday	3-5pm: Payload Subteam Meeting- meeting for development, building, and testing of payload system
Thursday	5:30-6:30pm: Structures Fabrication Meetings 6:30-7:30pm: Club Officer Meetings 7:30-8:30pm: General Body Club Meetings
Friday	10:40am-12:30pm: Senior Design Meeting 2:30-3:30pm: WolfWorks Experimental Meeting 5-8pm: Launch Prep (select weeks)
Saturday	7am-7pm: Launch Day Activities (select weeks)

October						
Sunday	**Monday**	**Tuesday**	**Wednesday**	**Thursday**	**Friday**	**Saturday**
10/2	10/3	10/4	10/5	10/6	10/7	10/8
	- Cut Body Tubes - Cut Nose Cone Shoulder - AV and payload Bay airframe epoxy			Before Meeting: - Bulkhead Layups (NC, Fin Can) - 3D print alignment ring	Fall Break	Fall Break
10/9	10/10	10/11	10/12	10/13	10/14	10/15
Fall Break	Fall Break	Fall Break		Before Meeting: - Sand Bulkheads - Epoxy T-nuts to NC centering Ring - NC centering Ring Epoxy to NC - Epoxy motor tube to thrust plate - Fillet Coupolas (Ashwin)	- Design and Print Tail Cone	
10/16	10/17	10/18	10/19	10/20	10/21	10/22
	- Epoxy retainer to motor tube - Cut AV threaded rods - Blast caps -Drill shear pin and rivett holes			Before Meeting: - Epoxy tail cone to retainer - Fillet Motor Retainer - File Slots in Fin centering rings - Mark and cut Fin can Slots	PDR soft Deadline	
10/23	10/24	10/25	10/26	10/27	10/28	10/29
	- Payload Bulkhead Layups - Cut Fin can threaded rods - Prepare for fin layups (cut balsa and fiberglass) - Epoxy fin centering rings/runners together		PDR Due	- Composite Fin Layup - Epoxy forward fin can centering ring		
10/30	10/31					
	- Composite Fin Layups - Drill pressure port/switch holes - Rail Buttons					
November						
Sunday	**Monday**	**Tuesday**	**Wednesday**	**Thursday**	**Friday**	**Saturday**
		11/1	11/2	11/3	11/4	11/5
				Before Meeting: - Composite Fin Layups (if nessecary) - Ensure all Hardware is mounted to bulkheads		
11/6	11/7	11/8	11/9	11/10	11/11	11/12
	Paint	Paint	Paint	Paint	Paint	
11/13	11/14	11/15	11/16	11/17	11/18	11/19
				Ejection Testing	Dry Run and Packing	Launch

Figure 86: The subscale build calendar is pictured above. Items highlighted in green have been completed as of the writing of this document.

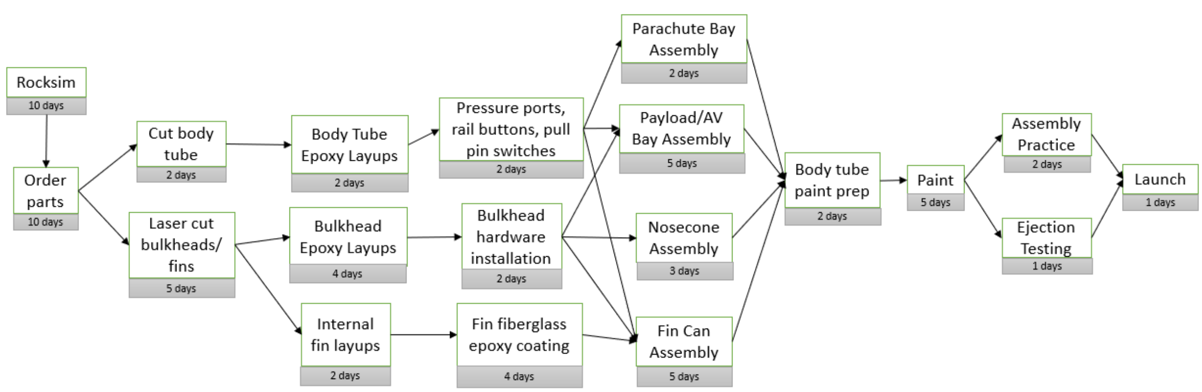

Figure 87: A PERT chart for the construction of the launch vehicle.

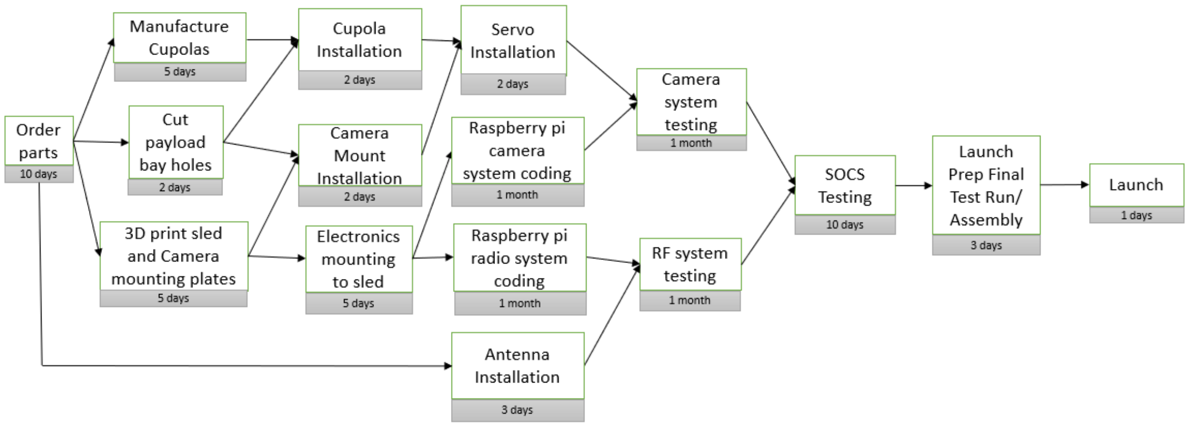

Figure 88: A PERT chart for the construction of the payload.

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