Tacho Lycos
2021 NASA Student Launch
Preliminary Design Review

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Common Abbreviations & Nomenclature

AGL	=	above ground level
APCP	=	ammonium perchlorate composite propellant
ARRD	=	advanced retention and release device
AV	=	avionics
BP	=	black powder
CDR	=	Critical Design Review
CG	=	center of gravity
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
L3CC	=	Level 3 Certification Committee (NAR)
LCO	=	Launch Control Officer
LOPSIDED	=	Lander for Observation of Planetary Surface Inclination, Details, and Environment Data
LRR	=	Launch Readiness Review
MAE	=	Mechanical & Aerospace Engineering Department
MSDS	=	Material Safety Data Sheet
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
POS	=	Planetary Observation System
PPE	=	personal protective equipment
RFP	=	Request for Proposal
RSO	=	Range Safety Officer
SL	=	Student Launch
SLS	=	Space Launch System
SME	=	subject matter expert
SOW	=	statement of work
STEM	=	Science, Technology, Engineering, and Mathematics
TAP	=	Technical Advisory Panel (TRA)
TRA	=	Tripoli Rocketry Association
UAS	=	Unmanned Aircraft System
UAV	=	Unmanned Aerial Vehicle

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1.1 Team Summary

1.1.1 Team Name and Mailing Address

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1.1.3 Time Spent on PDR Milestone

The team spent a total of 80 hours working towards completion of the PDR milestone.

1.2 Launch Vehicle Summary

1.2.1 Official Target Altitude

The team's official target altitude is 4473 feet. See section 3.3.1 for justification.

1.2.2 Preliminary Motor Choice

The current leading motor selection is the Aerotech L1520T. See section 3.1.5 for justification.

1.2.3 Launch Vehicle Size and Mass

The current leading design calls for a 106.25-inch-long launch vehicle with a diameter of 6 inches. The fully loaded mass of the leading launch vehicle with the leading motor selection is 45.2 lbs. See section 3.1.4 for additional launch vehicle size discussion.

1.2.4 Recovery System

The leading recovery system design is a dual-deployment system controlled by two independent PerfectFlite Stratologger CF altimeters. An 18-inch drogue parachute will be deployed at apogee, and a 120-inch main parachute will be deployed at 675 feet AGL. See Section 3.2.3 for discussion of the recovery system.

1.3 Payload Summary

The lander has been designated the Lander for Observation of Planetary Surface Inclination, Details, and Environment Data, abbreviated to LOPSIDED. The imaging system contained within LOPSIDED is designated the Planetary Observation System, abbreviated as POS. During the descent of the launch vehicle, the LOPSIDED-POS system will be removed from the payload bay by the main parachute recovery harness. The LOPSIDED-POS will then detach itself from the recovery harness by means of an Advanced Retention Release Device (ARRD). The LOPSIDED-POS will descend under its own parachute. After landing, LOPSIDED will record its angle relative to vertical, then disengage two solenoid latches, allowing gravity to rotate two concentric rings such that LOPSIDED is oriented vertically with respect to gravity. LOPSIDED will then record its final orientation. The POS will then capture an image from each of the four onboard cameras. The POS will then stitch these images into a panorama before sending this final image to the team's laptop computer.

2. Changes Made Since Proposal

2.1 Changes Made to Launch Vehicle

Table 2-1 below lists all changes made to the launch vehicle since proposal, along with justification of these changes.

Table 2-1 Changes to Launch Vehicle Since Proposal

Description of Change	Justification of Change
Some bulkheads are no longer 3/4" thick.	Each bulkhead experiences a different amount of force and, therefore, can be designed differently. For example, the middle centering ring in the fin can is mainly used for alignment so it will be 1/2" thick. The nose cone bulkhead will be increased to 1" thick to move CG forward and produce a more desirable stability margin.
Fins have been scaled up 7.7%	After making changes to the design, the launch vehicle's stability margin fell, so the fin size was increased to compensate for this.
Leading motor selection changed from L1150R to L1520T.	After the RockSim model was updated to reflect changes to the payload weight and CG within the payload bay, as well as changes having been made to the bulkheads and fins, a larger motor became more desirable. This will allow for the team to reach the middle of the altitude window while also having a higher velocity at launch rail departure.
Payload parachute changed from Fruity Chutes 60" Classic Elliptical to Fruity Chutes 48" Classic Elliptical.	The payload mass is less than half than the manufacturer certified mass for the larger parachute, making wind drift a concern.

2.2 Changes Made to Payload

Table 2-2 below lists all changes made to the payload since proposal, along with justification of these changes.

Table 2-2 Changes to Payload Since Proposal

Description of Change	Justification of Change
The payload leveling system has changed from an individually actuated leg design to a free-hanging gravity gimbal design.	The original payload levelling design would not have provided an adequate range of leveling angles. The new gimbal design reduces system complexity and allows for a greater range of leveling angles.

The payload chassis has a reduced cross-sectional area near the location of the gimbal leveling system.	Reduced chassis dimensions allow for a greater range of motion from the gimbal leveling system.
The payload integration system no longer has guidance tracks or rollers within the payload bay.	With the new gimbal design, LOPSIDED will be in nearly constant contact with the walls of the payload bay without any additional mechanisms. This design removes the need for additional supports.
The POS will utilize a 433 MHz transmitter rather than a 900 MHz transmitter.	The current leading method of image transmission is designed to work in a different range of frequencies than those stated in the proposal.

2.3 Changes Made to Project Plan

Table 2-3 below lists all changes made to the project plan since proposal, along with justification of these changes.

Table 2-3 Changes to Project Plan Since Proposal

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

3. Vehicle Criteria

3.1 Selection, Design, and Rationale of Launch Vehicle

3.1.1 Launch Vehicle Mission Statement

The mission of this launch vehicle is to reach the target apogee and return safely to the ground in a reusable condition. Additionally, the launch vehicle will support the payload mission by safely ejecting the payload at main deployment.

3.1.2 Launch Vehicle Success Criteria

Mission success is defined by compliance with NASA SL requirements in Table 6-1 as well as the team derived requirements in Table 6-2. Levels of success are defined in more detail in Table 3-1 below.

Table 3-1 Launch Vehicle Mission Success Criteria

Level of Success	Definition
Complete Success	– Launch vehicle recoverable – Nominal launch vehicle takeoff and descent – Lander is undamaged during deployment and after landing – Launch operations can be repeated the same day – Achieved apogee is between 3000 and 6000 feet
Partial Success	– Launch vehicle repairable – Successful launch vehicle takeoff and descent – Lander repairable following landing – Achieved apogee is between 3000 and 6000 feet
Partial Failure	– Launch vehicle repairable – Successful launch vehicle takeoff and unsuccessful descent – Lander repairable following landing – Achieved apogee is below 3000 feet or above 6000 feet
Complete Failure	– Launch vehicle unrecoverable – Lander unrecoverable

3.1.3 Launch Vehicle Alternative Designs

All variations on the launch vehicle design share the same section layout. Starting at the forward end of the rocket, the configuration will consist of the nose cone, the payload bay, the main parachute bay, the AV bay, and the fin can. Additionally, the current launch vehicle design has a diameter of 6 inches.

3.1.3.1 Material Selection

Body tube material is an important consideration when designing the launch vehicle. All other systems contained in the launch vehicle are attached directly to the airframe or indirectly through bulkheads and hardware, so the success of the airframe material is crucial. The chosen material must be able to withstand aerodynamic forces during flight, the motor thrust as well as any loads from other components. Based on the materials referenced in the proposal, the choice of materials was narrowed down to Blue Tube 2.0 and G12 fiberglass as they provided the most promising performance.

To determine the compressive force on the launch vehicle airframe, the following equation was used:

$$F_C = F_D + F_I \tag{1}$$

Where F_D is the aerodynamic drag force and F_I is the inertial force on the launch vehicle. Drag force is calculated using the equation below:

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

Where ρ is the density of air, V is the velocity of the launch vehicle, C_D is the drag coefficient, and A is the frontal reference area of the launch vehicle. All required values excluding the density of air were calculated using RockSim. This equation assumes the launch vehicle is flying at a 0° angle of attack. The inertial force on the launch vehicle can be found using the following equation:

$$F_I = ma \tag{3}$$

Where m is the mass of the launch vehicle and a is its peak acceleration during flight. Using these equations, the total force on the launch vehicle can be calculated. These values are shown below for both Blue Tube 2.0 and Fiberglass.

Table 3-2 Force Calculations for Blue Tube 2.0 Airframe

	Value	Source
m	38.77 lbm	RockSim
C_D	0.284	RockSim
A	0.208 ft^2	RockSim
a	541.95 ft/s^2	RockSim
V	617.61 ft/s	RockSim
rho	0.00237 slugs/ft^3	Constant
F_D	**27.16 lbf**	Formula
F_I	**689.53 lbf**	Formula
F_C	**716.69 lbf**	Formula

Table 3-3 Force Calculations for G12 Fiberglass Airframe

	Value	Source
m	41.98 lbm	RockSim
C_D	0.286	RockSim
A	0.208 ft^2	RockSim
a	542.36 ft/s^2	RockSim
V	565.37 ft/s	RockSim
rho	0.00237 $slugs/ft^3$	Constant
F_D	**21.66 lbf**	Formula
F_I	**745.90 lbf**	Formula
F_C	**767.56 lbf**	Formula

3.1.3.1(a) Blue Tube 2.0

Blue Tube is a vulcanized fiber-based rocket airframe material created by Always Ready Rocketry. It is highly abrasion resistant and durable, made with eco-friendly materials, and much cheaper and more lightweight than alternatives such as carbon fiber and fiberglass, making this an ideal choice for high powered rocketry. While more expensive than phenolic tubing, it is more shatter-resistant making it the better choice. With an average compressive load yield of around 3000 lbf, it is more than suitable for the launch vehicle, which has estimations of its maximum compressive load at around 717 lbf. However, since it is a paper-based product, the material is not completely water-resistant. Should the airframe encounter water, it would be damaged, and the launch vehicle may be rendered unusable. Additionally, due to its spiral wound structure, the spiral will need to be filled and smoothed to decrease drag forces on the rocket, as well as give the rocket a nice finish.

3.1.3.1(b) G12 Fiberglass

G12 fiberglass is another material that could be used for the launch vehicle airframe. G12 tubing is manufactured using fiberglass roving and epoxy with layers wound in a range between 30° and 45° for additional strength. It is a commonly used material as it is extremely durable and damage resistant. It is far stronger than Blue Tube and can easily withstand the 768 lbf loading. However, this increased strength comes at a cost. Compared to fiber-based products like Blue Tube, G12 fiberglass is heavier, with a density of 0.98 oz/in^3 compared to Blue Tube at a density of 0.723 oz/in^3, and it is more expensive at about $29 more per foot. However, using fiberglass for the airframe is preferable to Blue Tube, because unlike Blue Tube it is moisture resistant and will not deform due to high humidity or other water damage. The tube also comes with a smooth finish, eliminating the additional work required by a material like Blue Tube. Additionally, it will naturally last for a longer period of time, due to not having the tendency to fray or unravel like Blue Tube, making this a good option for a reusable launch vehicle airframe.

3.1.3.2 Nose Cone Bulkhead

3.1.3.2(a) Fixed

Having the nose cone bulkhead fixed in place with epoxy offers a strong bond which is important, as the U-bolt attached to this bulkhead serves as an anchor point for the main parachute. This option has a very simple installation process, but the downside is that once the epoxy has cured, the bulkhead cannot be removed without dissolving the epoxy or damaging the nose cone. Should the launch vehicle need ballast, it would need to be installed on the nosecone bulkhead before it is epoxied in place. This design would not allow later adjustments.

3.1.3.2(b) Removable

Having a removable nose cone bulkhead gives the option to adjust the amount of ballast attached to the bulkhead to alter the stability margin of the launch vehicle. Additionally, since the rotary latch for the payload is attached to this bulkhead, having easy access to house electronics from the backside would be incredibly useful. However, to make the bulkhead removable, it would need to be mounted inside the nosecone using L-brackets. Using brackets instead of an epoxy bond may lead to a weaker connection and places stress concentrations on the bolts. Additionally, the bolts would need to be mounted through the nosecone, which placed stress concentrations around the holes in the nose cone as well. The protrusion of the bolt heads could slightly affect the nosecone aerodynamics. However, this effect would likely be inconsequential and while an epoxy connection may be stronger, using sufficiently strong bolts could make up for this.

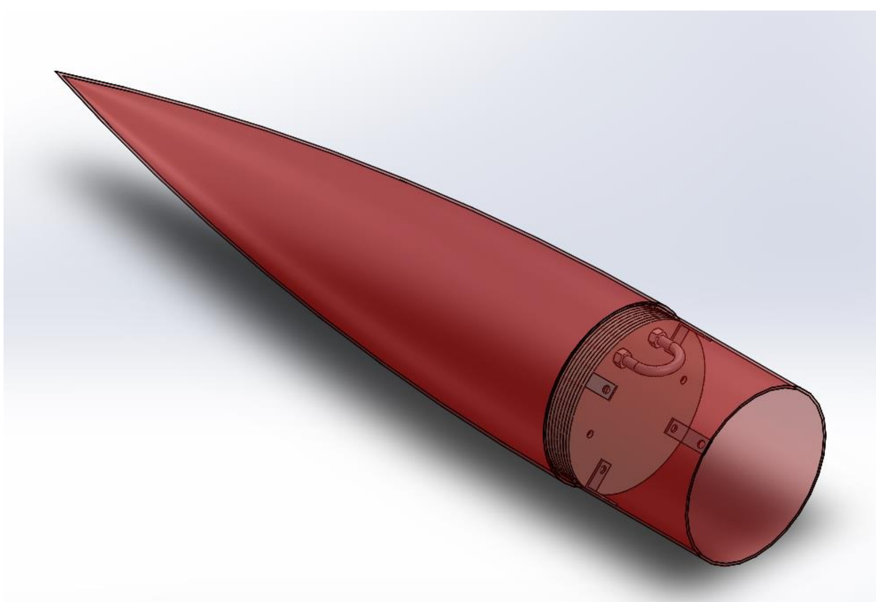

Figure 3-1 Nose Cone with Removable Bulkhead

3.1.3.3 Nose Cone Selection

There are many different shapes of nose cones for different rockets based on what the mission is. However commercial availability of large diameter nosecones narrows the choices for this launch vehicle down to just a few:

3.1.3.3(a) Conical Nose Cone

Conical nose cones are the simplest design for a nose cone, but this simplicity does not yield positive aerodynamic results for subsonic speeds, which the launch vehicle is limited to per NASA 2.22.26. The pointed end of a conic nose cone is not ideal for pushing through the atmosphere when there are no shockwaves.

3.1.3.3(b) Von Karman

Of the commercially available nose cones, the Von Karman was chosen to be the optimum nose cone design for full scale. The Von Karman nosecone is a specific type of Haack nose cone, which is not derived from geometry, but instead were mathematically derived to minimize friction drag. As pressure drag below Mach 0.8 is largely negligible, minimizing friction drag is highly desirable. However, at subscale, there are no commercially available options for a Von Karman nose cone. The production methods available for the team to independently manufacture a Von Karman were either too complicated, expensive, or would yield too fragile of a nose cone, limiting the number of launches the team could perform.

3.1.3.3(c) Ogive

An Ogive nose cone is derived from a segment of a circle in such a way that it has a pointed tip and is tangent to the body of the outer diameter of the launch vehicle. Many Ogive geometries exist, typically expressed as a ratio of the length of the nose cone to the diameter of the nose cone, and they are commercially available at both full scale and subscale dimensions.

3.1.3.4 Anchor Selection

3.1.3.4(a) Eyebolt

An eyebolt has a single point of contact on the surface it is bolted to. It has a metal loop, or eye, through which a quick link can be attached. The single point of contact makes installation and removal easy, and it has a low profile and mass making it an enticing option. However, the single point of contact significantly increases the chance of failure of the bolt due to the increase in stress concentration compared to a U-bolt.

3.1.3.4(b) U-bolt

A U-bolt provides a more secure connection between the bulkhead and shock cord due to the two points of contact it has on the bulkhead. The force of the shock cord deployment on the bolt is split evenly between the two points, reducing the stress on each individually. However, this comes with reduced versatility, especially when trying to secure it in tightly. It also has a higher mass and profile than an eyebolt and takes slightly more time to install. However, the open loop, or "U", makes it easier to attach a quick link compared to an eyebolt.

3.1.3.5 Fin Configuration

3.1.3.5(a) Four-Fin Configuration

The primary advantage of the four-fin configuration is that it is axisymmetric, leading to greater roll stability when encountering a cross wind. Also, due to the symmetry of the four-fin configuration, it is easier to install the fins along the correct alignments. Additionally, a four-fin configuration has the potential to offer much more stability than a three-fin configuration.

A disadvantage of a four-fin configuration is that it requires more labor than a three-fin configuration; an additional fin slot must be cut, and an additional fin must be constructed and filleted. Along with additional labor, a four-fin configuration requires smaller fins to achieve the same stability of a three-fin configuration, which can contribute to the risk of the launch vehicle weathercocking once it leaves the launch rail.

3.1.3.5(b) Three-Fin Configuration

A three-fin configuration requires less cuts to be made than a four-fin configuration, reducing the probability of error. Three fins also have the capability of providing the same stability of four fins with less weight and less drag, which would improve the performance of the launch vehicle. The reason for this decreased weight and drag is that by having three fins with a larger relative root chord, they reach further out of the turbulent boundary layer of the rocket, allowing each fin to be more effective while maintaining the same total area as a four-fin configuration. A three fin-configuration also serves to reduce the likelihood of weathercocking, as the launch vehicle will roll if a crosswind is encountered, reducing the effective area the wind can act on to pitch the rocket.

Aligning three fins to 120 degrees is more difficult and more prone to alignment errors than a four-fin configuration is. A three-fin configuration is also non-axisymmetric when exposed to a crosswind, which would lead to rolling of the launch vehicle about its longitudinal axis.

3.1.4 Leading Launch Vehicle Design

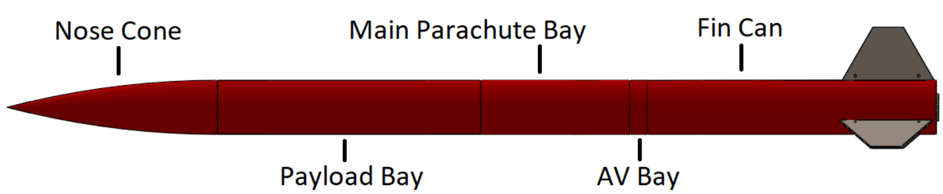

Figure 3-2 Launch Vehicle Layout

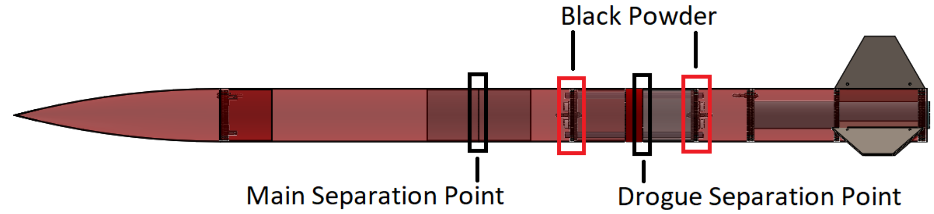

Figure 3-3 Launch Vehicle Points of Separation and Energetic Materials

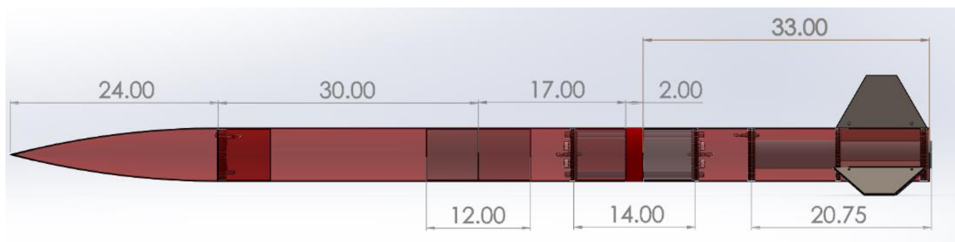

Figure 3-4 Launch Vehicle Dimensions

3.1.4.1 Material Selection

Currently, the leading choice for the body of the launch vehicle is G12 fiberglass. This was selected because of its overall durability and reusability, despite the extra weight and cost that would be included in this model. The biggest factor in this decision was that North Carolina's unpredictable weather might force a launch with high humidity or recent precipitation. Additionally, the irrigation ditches at the team's home launch site are typically filled with water leading to the possibility of landing in water and damaging launch vehicle components. This makes Blue Tube a poor choice since it needs to be specially treated for it to be water resistant. None of the team members have experience with this process and it would add extra time to the manufacturing process. Using the fiberglass option, RockSim has the mass of the launch vehicle estimated at around 45.2 lbs which is a very practical number in terms of reaching the desired apogee of 4473 ft.

3.1.4.2 Anchor Selection

The leading choice of anchor for the bulkheads is the U-bolt. While heavier than an eyebolt, the additional weight is not enough to make a significant difference overall. The installation time is only slightly longer so that will not be an issue. The main benefit of a U-bolt over an eyebolt is the increased reliability from the extra point of contact to split the applied loading and add a secondary point of failure.

3.1.4.3 Nose Cone Selection

The team decided that it was imperative that the subscale accurately model the aerodynamics of the full scale as closely as possible, so an Ogive nose cone design was chosen. Additionally, the fragility that would be introduced to the nose cone by attempting to manufacture a Von Karman was ultimately decided to not be worth the additional aerodynamic performance it would provide. A 4:1 Ogive design was chosen, as it provided the most desired stability margin for the vehicle.

The nosecone and nosecone bulkhead, excluding payload hardware and any ballast, are expected to weigh 3.22 lbs.

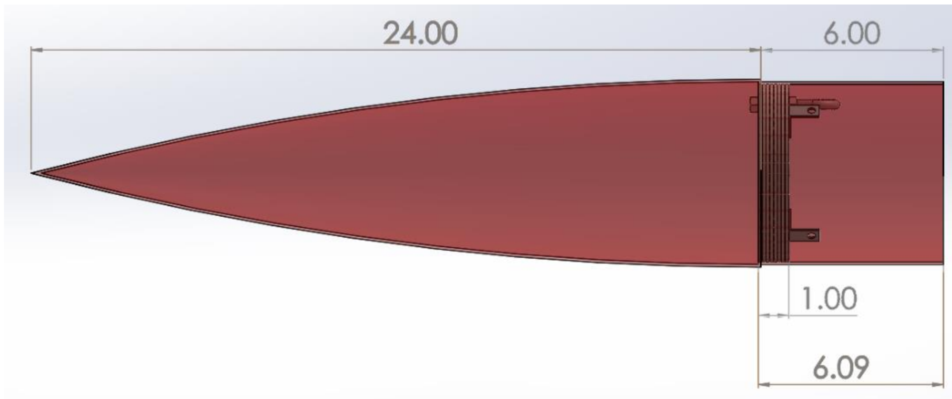

Figure 3-5 Nose Cone Dimensions

3.1.4.4 Nose Cone Bulkhead

A removable bulkhead will be used in the nose cone so that ballast adjustments can be made if the stability of the completed launch vehicle does not perfectly match simulations. The bulkhead will be secured with the forward face flush with the inner edge of the nosecone shoulder 6 inches in. Additionally, a rotary latch will be mounted on the aft side of the bulkhead to hold the payload, and the latch electronics will be mounted to the forward side of the bulkhead along with any necessary ballast. L-brackets will be attached to the bulkhead which will be secured inside the nose cone using bolts through the nose cone and L-brackets. To account for the lack of an epoxy connection, the L-brackets, bolts, and bulkhead will be designed or chosen to withstand deployment forces with a sufficient factor of safety. This bulkhead will also have a U-bolt attached for use in main parachute deployment.

3.1.4.5 Payload Bay

The 30-inch payload bay will be located between the nosecone and main parachute bay. The nosecone will be secured to the payload bay with a bolted connection and the payload bay and main parachute bay coupler will be connected with 4-40 nylon shear pins. Additional shear pin holes will be added to the payload bay in order to hold the payload in place during flight while still allowing it to deploy with the main parachute.

Excluding the LOPSIDED-POS, the payload bay is expected to weigh 2.98 lbs.

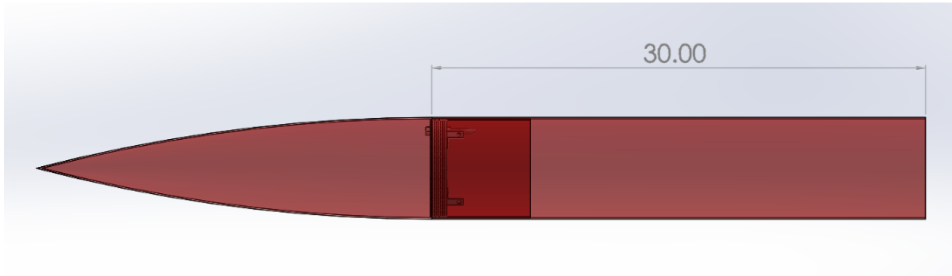

Figure 3-6 Payload Bay Dimensions

3.1.4.6 Main Parachute Bay

The main parachute bay will be located between the payload bay and the AV bay. A 12-inch coupler will be epoxied into the forward end of the body section with 6 inches exposed to connect to the payload bay. This connection will be secured with shear pins. The aft end of the main parachute bay will be secured to the forward end of the AV coupler with a bolted connection. The body portion of the main parachute bay will be 17 inches long. The leading main parachute alternative has a packed volume of 5 inches D x 7 inches L. As the main parachute bay holds most of the shock cord making up the main recovery harness, the recovery harness and parachutes can be estimated as taking up 10 inches of the main parachute bay. The remainder of the length is left open to allow a sufficient volume for the black powder ejection charge to expand into. Making this open volume will allow for a smoother pressure curve over the course of the detonation event, reducing wear and stress on the airframe.

Excluding the main parachute, the main parachute bay and attached coupler is expected to weigh 3.22 lbs.

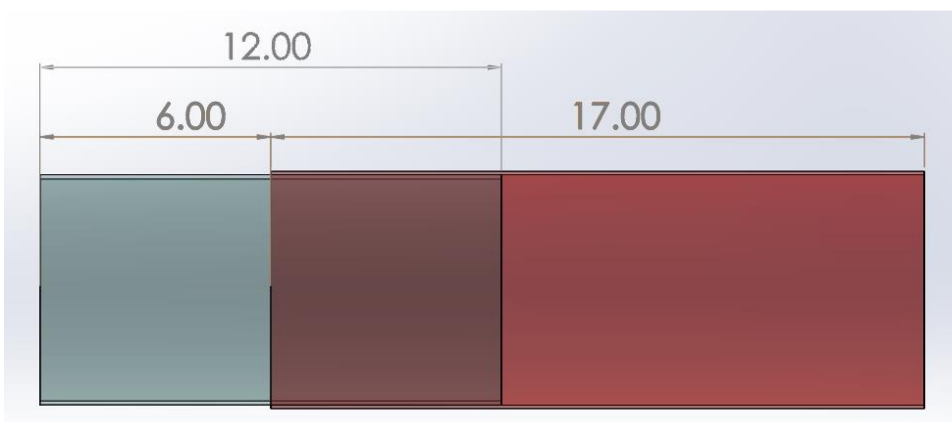

Figure 3-7 Main Parachute Bay Dimensions

3.1.4.7 AV Bay

The AV bay will be located between the main parachute bay and fin can. It will be a modular design, able to fully separate from the rest of the launch vehicle, allowing for a simple construction process and an easily accessible AV sled. This also allows for easy loading of black powder and allows the AV bay to be worked on and assembled separately from the rest of the launch vehicle.

The AV bay consists of a 14-inch-long coupler section with a 2-inch-thick band of body tube centered on the outside. The forward end is connected to the main parachute bay with a bolted connection and the aft end is secured to the fin can with shear pins. Each AV bay bulkhead will have a U-bolt for use in main and drogue parachute deployment. The bulkheads also each have two blast caps mounted on the outside to house the primary and secondary black powder charges for main and drogue parachute deployment. The AV sled is mounted inside the coupler on threaded rods that run between the two bulkheads.

Excluding the AV sled and black powder charges, the AV bay is expected to weigh 4.38 lbs.

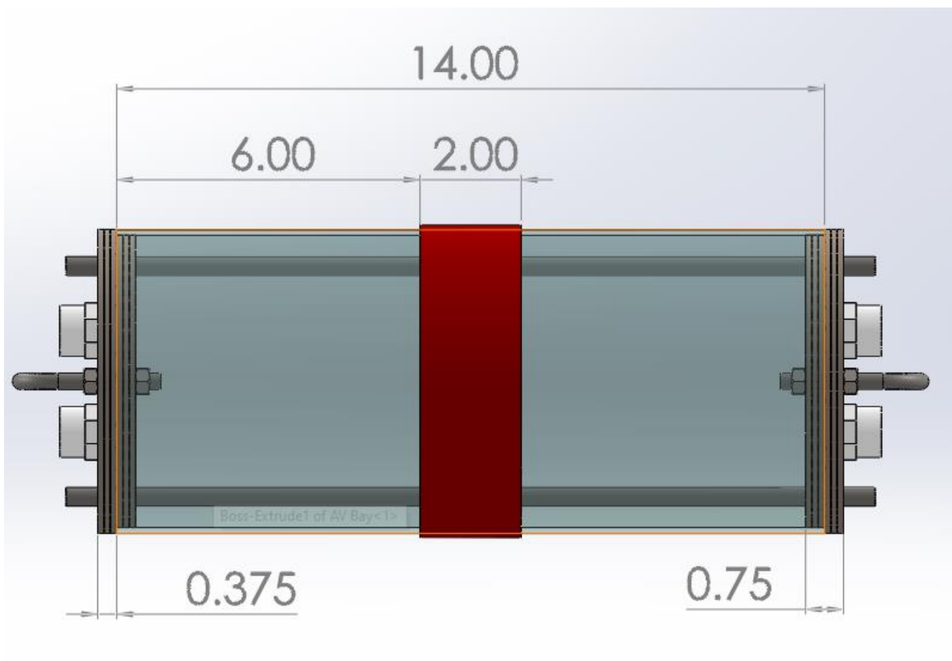

Figure 3-8 AV Bay Dimensions

3.1.4.8 Fin Can

The 33-inch-long fin can assembly houses the motor tube and the fins. The assembly consists of a forward bulkhead, engine block, two centering rings, the motor tube, and fins. The forward bulkhead and the aft centering ring will support most of the load from the motor while the middle centering ring is used primarily for motor tube and fin alignment. The fins will be designed with a fin slot that fits between the two centering rings for additional support. Half of the layers used in the forward bulkhead are centering ring layers so the forward bulkhead can also assist with motor tube alignment. Additionally, a U-bolt will be attached to the forward bulkhead to be used for drogue parachute deployment. The drogue parachute is housed in the additional space in front of the forward bulkhead. The fin can is connected to the AV bay using 4-40 nylon shear pins.

Excluding the motor, the fin can assembly is expected to weigh 6.92 lbs.

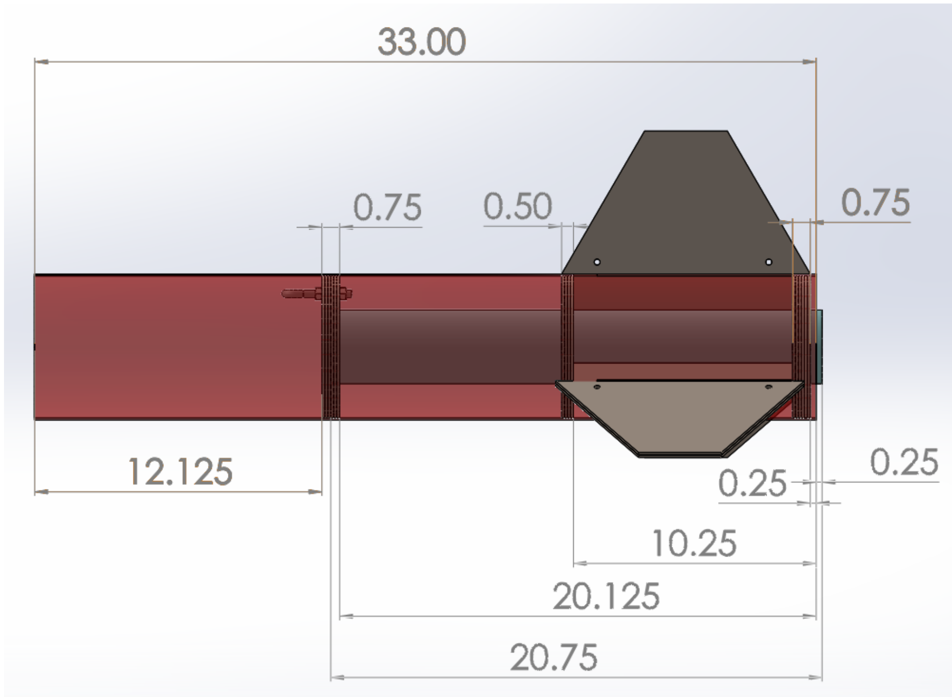

Figure 3-9 Fin Can Dimensions

3.1.4.9 Fin Configuration Selection

The team chose the three-fin configuration for the launch vehicle. The reduction in drag and overall weight, as well as the lower chance of weathercocking, were the primary drivers for this decision. The fins are designed to mirror the properties of an isosceles trapezoid and are dimensioned by sweep length and semispan rather than sweep angle for ease of manufacturing. The leading-edge sweep reduces drag and pushes the center of pressure of the fin aft wards, increasing stability. The trailing-edge sweep also reduces drag while helping protect the fins from potential damage if the launch vehicle lands fin first. The fins will be 0.25” thick and will be made of two 0.125” thick plywood cutouts that will be held together by an epoxy resin. This greater thickness is to reduce the likelihood of fin flutter during the ascent of the launch vehicle.

3.1.4.10 Bulkhead Sizing

The bulkhead dimensions shown in the figures above are all approximations. These approximations are based on successful past designs and initial finite element analysis done in the proposal using estimates for deployment forces, but more analysis and testing will need to be done to finalize the sizing. When more accurate values for deployment forces are found and the launch vehicle design has been solidified, the ideal bulkhead thickness can be estimated using hand calculations and FEA.

To perform the necessary hand calculations, the book Roarks Formulas for Stress and Strain[1] was referenced. Chapter 11 of the text covers the behavior of flat plates under loading, which the bulkheads can be approximated to be. Although a U-bolt has two anchor points where force is applied, they will be approximated as a small circular area. This is a safe assumption as flanges will be used on the U-bolt to distribute the loading more evenly. In this case, a basic bulkhead with a centered U-bolt can be approximated as a uniform load over a very small central circular area with a fixed edge. The following equations are used to find the resulting moment on the bulkhead:

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

$$r_0' = \sqrt{1.6r_0^2 + t^2} \text{ if } r_0 < 0.5t \tag{5}$$

$$\text{or } r_0' = r_0 \text{ if } r_0 \geq 0.5t \tag{6}$$

Where W is the total applied load, v is the Poisson's ratio of the bulkhead material, a is the distance from the fixed edge to the loading area, r is the radius of the bulkhead, r_0 is the radius of the circular area where the loading is applied, and t is the bulkhead thickness.

On constant-thickness bulkheads with a non-centered U-bolt, a different formula for maximum moment is used:

$$M_{max} = -\frac{W}{8\pi}\left[2 - \left(\frac{r_0'}{a-p}\right)^2\right] \tag{7}$$

Where a is the radius of the bulkhead and p is the distance from the center of the bulkhead to the area where the loading is applied.

Hand calculations only work well for these specific instances, and even then, approximations were made. Formulas get far more complex for centering rings where the inner edge is fixed to the motor tube. Additionally, not all bulkheads are of uniform construction. Half of the layers of the engine block are solid disks and the other half have the center cut out to hold the motor tube. This mixed construction along with an off-center U-bolt would lead to a very difficult and imprecise hand

[1] Young, Warren C., and Richard G. Budynas. "Flat Plates." *Roark's Formulas for Stress and Strain*, McGraw-Hill, New York, NY, 2002, pp. 427–524.

calculation. Because of this, structural analysis will be done in ANSYS to get more precise results.

3.1.5 Motor Alternatives

When starting the design process, the team was already in possession of an RMS-75/3840 motor casing and decided to avoid purchasing additional motor casings. Using this motor casing, the options available were the 850W, 1150R, 1256WS, 1390G, and 1520T from Aerotech. The 1150R was initially included in potential design considerations, but after examining previous team experience with the motor, it was removed from consideration due to inconsistent and unreliable launch performances.

Table 3-4 Motor Data

Motor	Total Impulse (lbf-s)	Initial Thrust (N)	Max Thrust (lbf)	Burn time (s)	Weight (oz)
L850W	819.74	1000.9	419.56	4.4	132
L1256WS	850.94	1352.4	339.14	3.0	132.49
L1390G	887.81	1416.5	370.80	2.6	136.83
L1520T	835.41	1545.4	396.88	2.4	128.79

Per the club's past issues with weathercocking, motors with a low burn time and high initial thrust were preferred during consideration. Motors with these qualities are more likely to provide a higher velocity upon departure of the launch rail, which will move the center of pressure of the launch vehicle aft and provide greater stability. For this reason, the L850W was removed from consideration, despite achieving an apogee between 3500 and 5500 feet in RockSim simulations.

Simulations conducted in RockSim were the primary source of information used to select between the L1256WS, L1390G, and L1520T. All simulations were run at zero degrees of tilt and no wind to develop a preliminary understanding of how each motor would perform. These results can be seen in the table below.

Table 3-5 Motor Simulation Results

Motor	Apogee (feet)	Velocity at Launch Rail Departure (ft/s)
L1256WS	4393	69.07
L1390G	4742	69.09
L1520T	4490	73.82

From the results of these simulations, the team chose the L1520T for the launch vehicle. Not only did the 1520T have the highest launch rail velocity for the three motors chosen, it also had an estimated apogee closest to the middle of launch window, 4500 feet. The club also has experience with the 1520T, which only helped solidify the team's choice. The 1520T will give the launch vehicle a thrust to weight ratio of 7.94. The thrust curve of the 1520T is shown below.

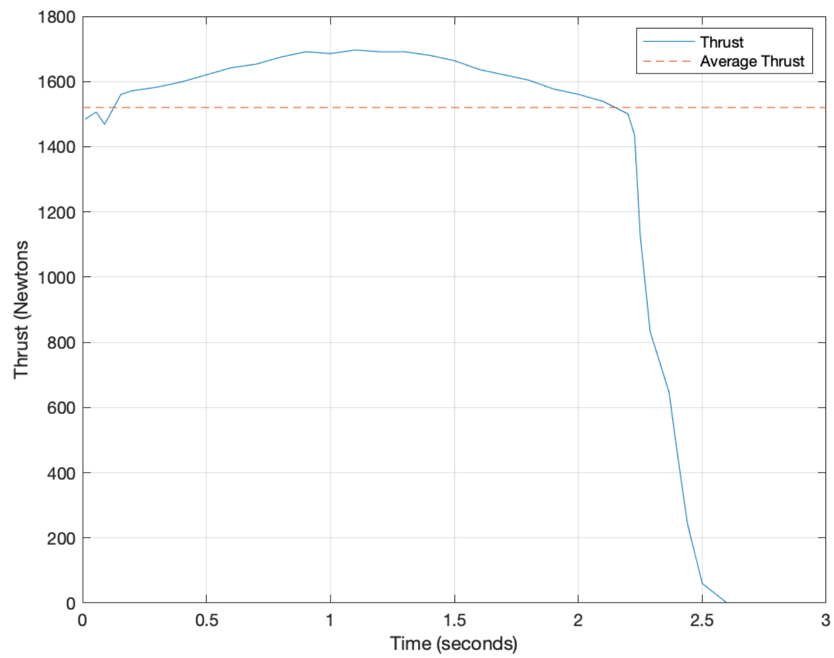

Figure 3-10 Aerotech L1520T Thrust Curve

3.2 Recovery Subsystem

3.2.1 Description of Recovery Events

Once the launch vehicle is mounted to the launch rail and raised into launch position, the altimeters will be armed from outside the launch vehicle by closing the connected switches. Before proceeding in the launchpad sequence of events, the altimeter audio readouts will be listened to and recorded to confirm proper functionality. Battery voltage, proper altimeter mission programming, and continuity to the black powder ejection charges are all confirmed during this startup sequence. Once proper altimeter functionality is confirmed, the launch procedure will proceed. In the event of improper altimeter functionality, the launch procedure will be halted, and proper altimeter functionality will be restored before proceeding.

At or shortly following apogee, both onboard altimeters will detect apogee and initiate drogue parachute deployment. When apogee is detected, the primary altimeter will send a current to an attached E-match, which will in turn detonate the primary black powder charge in between the fin can and AV bay. The black powder charge is contained in a PVC pipe end cap, with the E-match resting on the bottom of the cap and any remaining space packed with wadding made from shredded paper towels. The cap is then sealed with electrical tape to prevent leakage of black powder and to retain the E-match in place. Details on black powder ejection charge sizing and calculations are provided in Section 3.2.2.11. Detonation of this black powder charge will produce sufficient force to break the shear pins securing the fin can to the rest of the launch vehicle and separate the sections. The fin can and remainder of the launch vehicle are secured by a length of shock cord linked to the drogue parachute. More details on the arrangement and selection of the recovery harness components are provided in Section 3.2.2.10. One second after apogee is detected, the secondary altimeter will send a current to the E-match in a secondary black powder ejection charge, ensuring redundancy in the event the primary black powder ejection charge fails to detonate or separate the body sections. The one second delay on the secondary charge guards against over pressurization of the launch vehicle section that would occur should both charges detonate simultaneously.

During the first stage of descent under drogue parachute, the launch vehicle will descend at a constant velocity considerably greater than the required landing velocity. The drogue descent velocity will be such that descent time requirements are met while sufficiently slowing and stabilizing the launch vehicle for successful main parachute deployment. Drogue descent velocity is controlled through selection of the drogue parachute, as described in Section 3.2.2.6. Upon reaching an altitude of 675 ft, the main altimeter will detect the main parachute deployment altitude and fire the primary main black powder ejection charge. This charge is constructed and wired identically to the drogue charges, save for being connected to different altimeter terminals. The main parachute charges are sized in the same manner as the drogue parachute charges as described in Section 3.2.2.11. As with the drogue parachute, a secondary ejection charge will be detonated following a ~1 s delay after the primary ejection charge. To meet this the secondary altimeter will send current to the main secondary ejection charge once it detects an altitude of 650 ft to ensure system redundancy. Once the body sections separate, again tethered by a shock cord, the main parachute will inflate and slow the launch vehicle to a safe landing velocity.

Once the launch vehicle reaches an altitude of 500 ft, an altimeter onboard the LOPSIDED will detect this and fire the ARRD to separate the LOPSIDED-POS from the main recovery harness. The LOPSIDED-POS will initially be slowed and stabilized by the furled payload parachute, secured by a parachute retention device. Once an altitude of 200 ft is detected, the parachute retention device will be released, and the payload parachute allowed to inflate. The payload parachute will slow the LOPSIDED-POS to a safe landing velocity, while the main parachute performs the same role in safely landing the launch vehicle. The main parachute shall be sized such that all sections make a safe landing even if payload deployment is unsuccessful.

Tracking of the launch vehicle and the LOPSIDED POS will be primarily performed via visual tracking. Should visual tracking of the launch vehicle and LOPSIDED-POS be unsuccessful, both the launch vehicle and the LOPSIDED-POS will contain an electronic tracking device. Should visual contact be lost, the ground recovery team will use these radio tracker devices to locate the launch vehicle and LOPSIDED-POS.

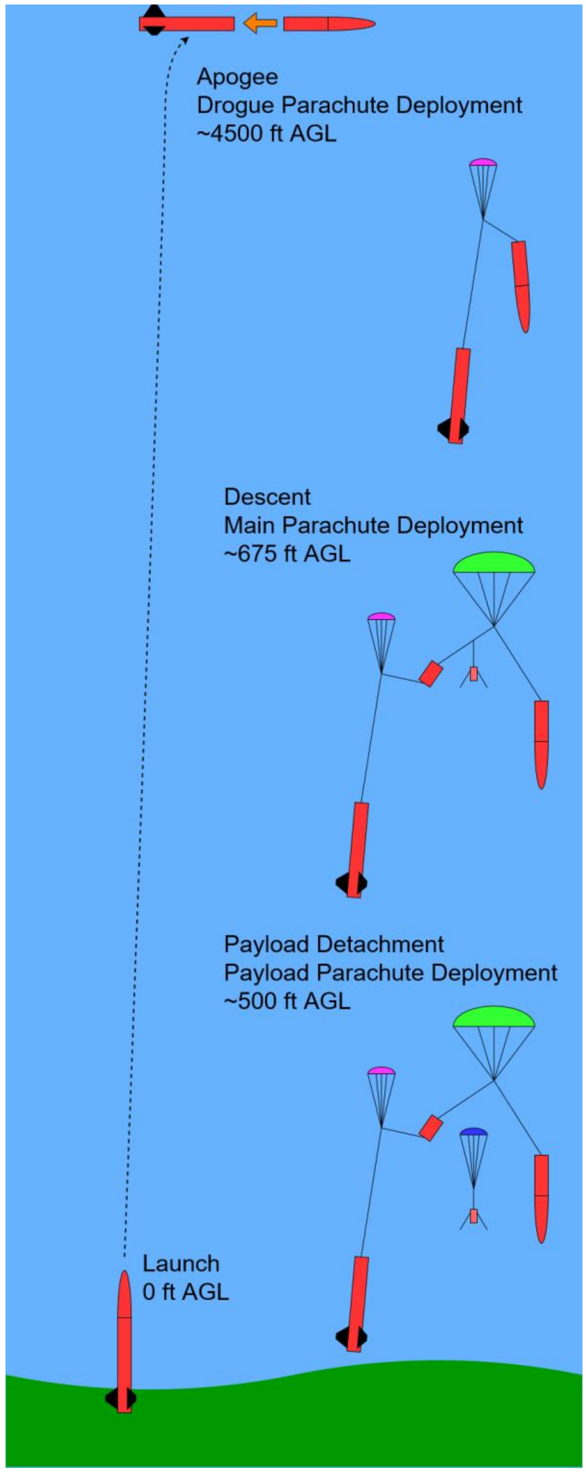

Figure 3-11 Recovery Sequence of Events

All body sections intended to separate during recovery will be secured together using 4 removable nylon shear pins. These shear pins will be capable of supporting the launch vehicle under its own weight. Black powder ejection charges will be sized to overcome the force needed to shear these pins and overcome any frictional forces holding body sections together.

3.2.2 Recovery Alternative Designs

3.2.2.1 Tracking Device Alternatives

Four tracking devices are under consideration for use in tracking the launch vehicle and the LOPSIDED-POS during descent and after landing. Two of these, the BigRedBee BeeLine and the QRP Labs LightAPRS+W, transmit on amateur radio frequencies. The other two tracking devices, the Eggfinder GPS and the BigRedBee 900 make use of 900 MHz ISM band. All four of these systems are designed for the tracking of rockets or other unmanned vehicles. Table 3-6 shows a comparison of the different tracking devices and several important parameters

Table 3-6 Tracking Device Comparison

Tracker	Cost	License	Transmitter Power	Transmitter Frequency	Owned by the Team
BigRedBee 900	$378.00	No	250 mW	900 MHz	Yes
BigRedBee BeeLine	$85.00	Yes	15 mW	420-450 MHz	No
Eggfinder GPS	$95.00	No	100 mW	900 MHz	Yes
LightAPRS+W	$115.00	Yes	10 mW (WSPR)	2.5 kHz-200 MHz	No

The BigRedBee 900 tracker represents the simplest, plug-and-play option of the four tracking devices. This tracker consists of a 900 MHz transmitter mounted to a microcontroller board alongside a GPS antenna and a single cell LiPo battery. As it transmits on the 900 MHz ISM band, no license is required to operate the transmitter. A handheld unit receives data and displays battery voltage and location coordinates. The handheld receiver is also capable of being plugged into a laptop to display real-time location using Google Earth. Downsides of this system include limited range, narrow frequency availability, and known durability and reliability issues as encountered by the team in previous operation of the BigRedBee 900. Advantages include ease of use, the tracker being already owned by the team, and real time location data.

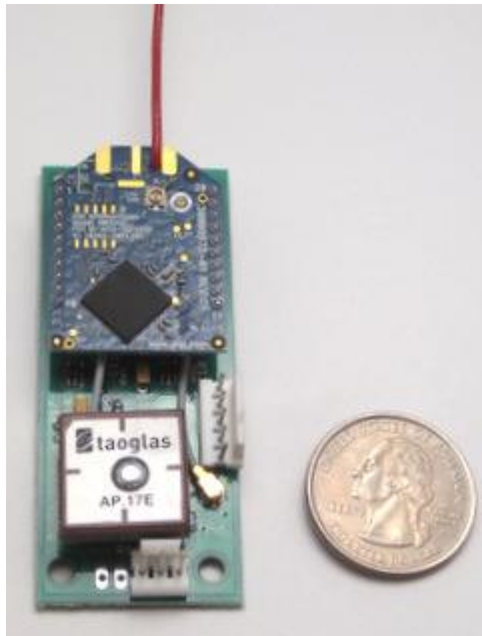

Figure 3-12 BigRedBee 900

The Eggfinder GPS is similar in many regards to the BigRedBee 900. The Eggfinder TX GPS consists of an Eggfinder TX transmitter with a GPS module, an antenna, and a 2S (7.4V) LiPo battery. The Eggfinder TX transmitter will be paired to an Eggfinder RX receiver Bluetooth dongle. The Eggfinder GPS system transmits on the 900 MHz ISM band as well, with paired frequencies within this band for the transmitter and receiver. The Eggfinder RX is capable of being paired to a phone via Bluetooth such that the location data can be plotted on the phone by a member of the field recovery team. This alternative is already owned by the team.

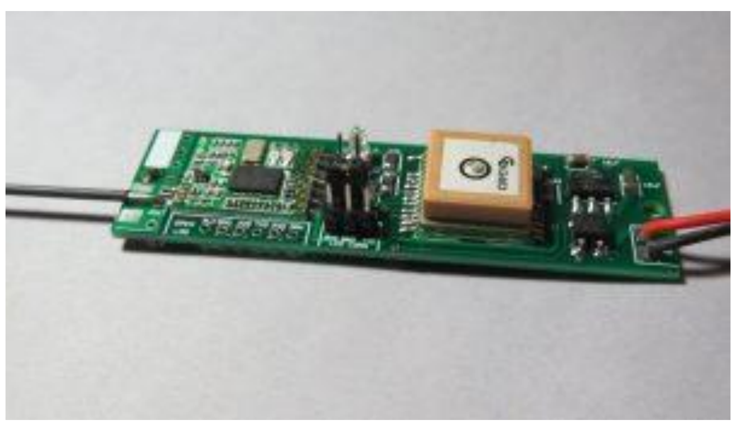

Figure 3-13 Eggfinder TX Transmitter

Representing a more robust option, the QRP Labs LightAPRS+W consists of an ATmega microcontroller board paired to a very high frequency (VHF) APRS transmitter and a high frequency (HF) WSPR transmitter. Since it operates on amateur radio frequencies, no dedicated receiver is required, and instead a standard amateur radio handheld receiver is used. The WSPR and APRS protocols are both capable of transmitting standardized location data, which can in turn be plotted by a computer connected to the receiver. For competition purposes, only the WSPR mode is within the transmitter power limit, restricting the possibilities of this tracker.

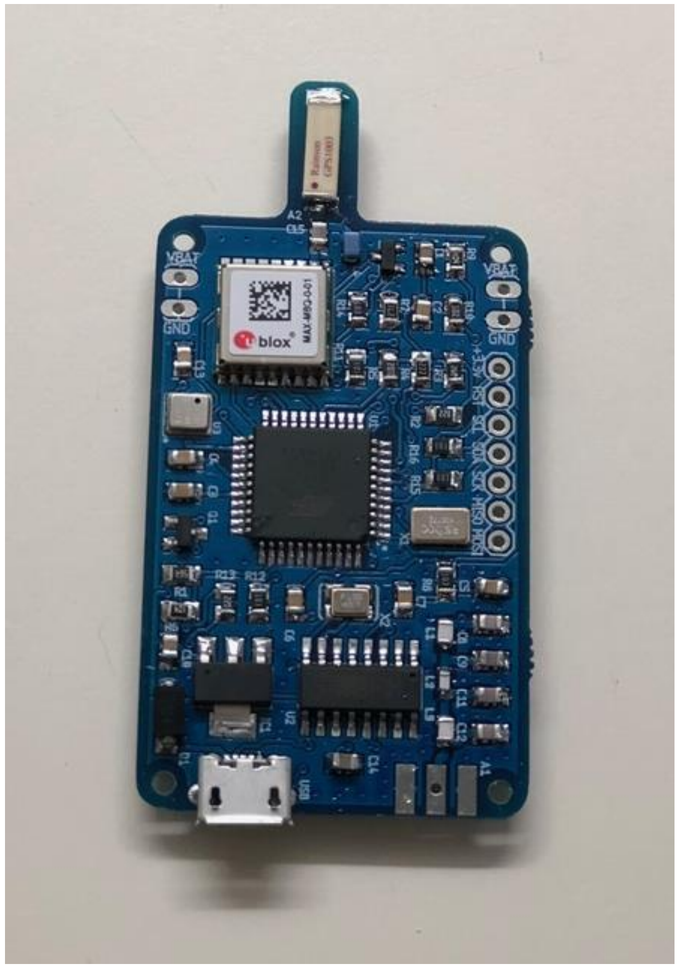

Figure 3-14 LightAPRS+W Tracker

A second amateur radio tracker, the BigRedBee BeeLine, represents a simpler approach to amateur radio band options. The BigRedBee BeeLine consists of a 70 cm VHF transmitter that broadcasts a homing signal and a LiPo battery. The BigRedBee BeeLine does not contain a GPS or other form of location tracking device, rather the field recovery team must home in on it manually using a radio receiver and a directional antenna. This approach to tracking offers the advantages of simplicity, low cost, and a durable transmitter. These advantages come at the cost of a more involved and subtle tracking method that must be employed to successfully locate the launch vehicle or LOPSIDED-POS.

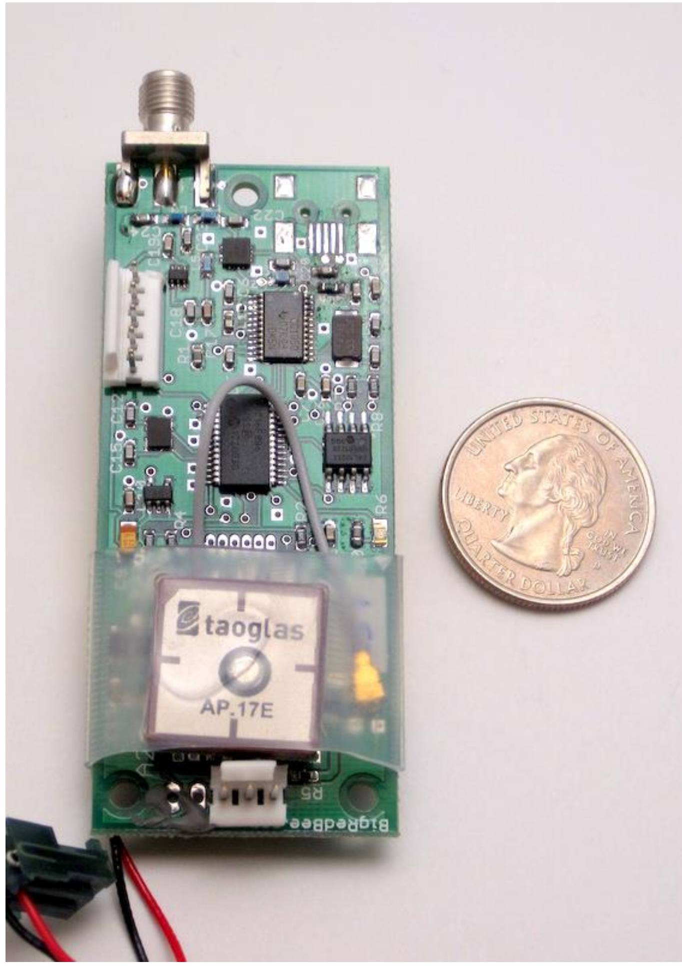

Figure 3-15 BigRedBee BeeLine Tracker

3.2.2.2 Recovery Electronics Alternatives

Table 3-7 Altimeter Alternative Comparison

Altimeter:	RRC3	Entacore AIM	StratoLogger CF	EasyMini
Main Deployment Variability	300 to 3000 ft; Increments of 100 ft	100 to 999 ft; Increments of 1 ft	100 to 9999 ft; Increments of 1 ft	Any
Delay After Apogee	0 to 30 sec; Increments of 1 sec	Available	0 to 5 sec; Increments of 1 sec	Available
Minimum Apogee	100 to 300 ft	N/A	100 ft	N/A
Altitude Logging Resolution	1 ft	1 ft	1 ft	1 ft
Dimensions	3.92” L x 0.925” W	2.75” L x 0.984” W	2” L x 0.84” W	1.5” L x 0.8” W

Data Logged	Altitude, velocity, temperature, voltage	Altitude, velocity, temperature, voltage, continuity	Altitude, voltage, temperature, voltage	Altitude, velocity, acceleration, temperature, voltage

Four altimeter alternatives were considered for the recovery system based on availability and possession of a dual-deployment system. The PerfectFlite StratoLogger CF, Missile Works RRC3, Entacore AIM USB 3.0, and Altus Metrum EasyMini were evaluated for suitability. All four altimeters can meet the minimum requirement of being dual deploy and thus can control the recovery process. In considering altimeters for use in the launch vehicle reliability, form factor, measurement precision, and variability of main parachute deployment altitude are considered of greatest importance.

The Missile Works RRC3 and Entacore AIM USB 3.0 both have a long, narrow form factor which takes up excessive space on the avionics sled. The Entacore AIM USB 3.0 uses a common ground terminal for the ejection charges and lacks a dedicated switch terminal, making wiring difficult. The common ground terminal, wherein two wires are secured in a single terminal port, creates an insecure mechanical connection. Should this terminal fail such that one or both wires come loose, this design increases the likelihood of failure of both main and drogue parachute ejection charges for this altimeter. This design is deemed unacceptable to the team. The Missile Works RRC3, while a robust flight computer with useful functionalities including a multi-flight recording unit, is only capable of setting main parachute deployment altitudes in increments of 100 ft. This lack of flexibility is undesirable should design changes require main parachute deployment at other altitudes.

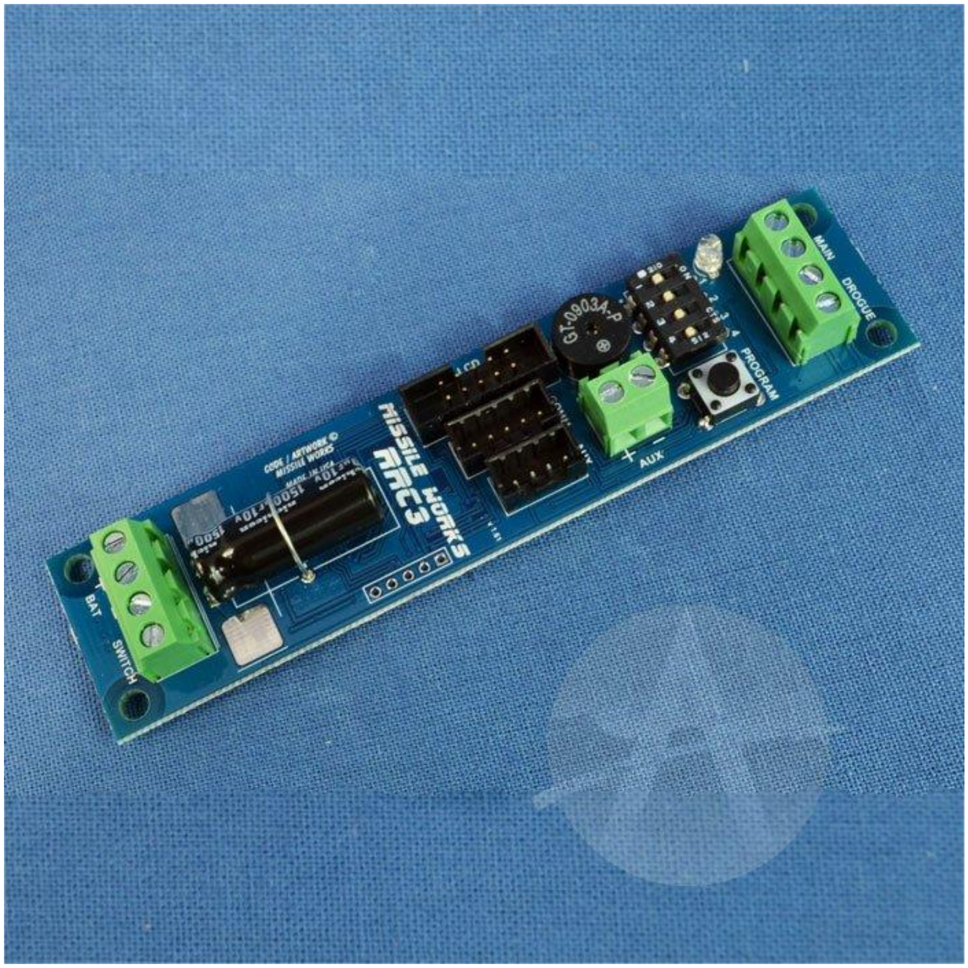

Figure 3-16 Missile Works RRC3 Altimeter

Figure 3-17 Entacore AIM USB 3.0 Altimeter

The PerfectFlite StratoLogger CF and Altus Metrum EasyMini provide wiring and form factor advantages over the RRC3 and Entacore AIM 3.0 altimeters, with one wire per terminal port and a much shorter design. However, the Altus Metrum EasyMini is not currently owned by the team. This makes the Altus Metrum EasyMini a more expensive option.

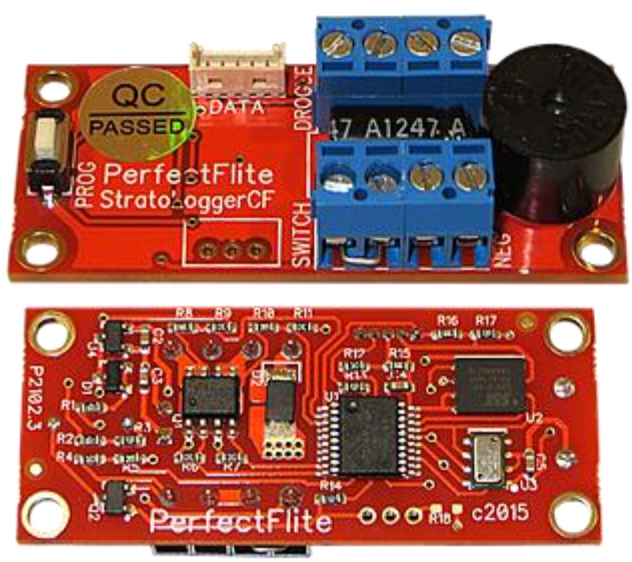

Figure 3-18 PerfectFlite StratoLogger CF Altimeter

Figure 3-19 Altus Metrum EasyMini Altimeter

3.2.2.3 Avionics Sled Alternatives

The avionics (AV) sled must mount in a secure fashion the tracking device, altimeters, arming switches, and power sources for all recovery electronics. All these components must remain secured during all stages of flight at all orientations and under accelerations of up to or exceeding 15 gees. The AV sled must fit smoothly within the AV coupler section. The recovery electronics must also be easily accessible during launch day assembly. Ease of assembly at the launch field and in the lab will be given a high priority since errors in assembly and wiring of the AV sled have the potential for significant launch vehicle failures.

AV sled materials considered include plywood, aluminum, 3D printed ABS plastic, and fiberglass over an aluminum frame. All these materials allow for precise manufacturing, with different levels of difficulty. Laser cutting and conventional machining processes are available for plywood, aluminum, and the fiberglass over frame construction. The ABS plastic is as indicated additively manufactured, and portions of the fiberglass design would have to be cured in a mold or layup.

Aluminum requires either a high-powered laser cutter or access to a machine shop to accurately manufacture. Higher power laser cutters present greater safety concerns, and conventional machining would require out-of-house manufacturing. However, aluminum also presents the most precise and durable option for AV sled material as the only alternative capable of having threads tapped into the AV sled material.

Plywood is considerably lighter than aluminum and is very easy to design and manufacture with a “jigsaw” design approach and a low power laser cutter. It is durable enough to mount components via through bolting and can be easily assembled using wood glue or epoxy. This is possible since the plywood is thick enough to provide adequate surface area for adhesive bonding without excessive weight.

The 3D printed ABS plastic is the safest and simplest to manufacture but is the least durable of the material choices and the most difficult to mount electronics to. Screw holes of the size typical of electronics mounting hardware are difficult to manufacture using 3D printing, and boring operations on 3D printed parts run the risk of easily cracking or damaging the AV sled.

Fiberglass is the most difficult material to manufacture, and the lengthy curing process combined with the difficult of working with it after curing makes it a comparatively undesirable alternative for the purposes of the AV sled.

Two basic sled layouts have been considered, a conventional lengthwise sled and a stacked disk configuration. The lengthwise sled is mounted to threaded rods, with the main sled body lying flat on the plane between the two rods. The stacked disk bulkhead also uses threaded rods, but the sled sections are stacked on the roads normal to the rods. These disks would then be wired together as needed. The stacked disk design presents a more efficient use of volume than the lengthwise design, at the cost of more difficult assembly at the launch field. A stacked disk design allows for easier separation of recovery electronics, and therefore easier mitigation of RF interference from any onboard transmitters.

3.2.2.4 Block Schematic of Recovery Electronics and Proof of Redundancy

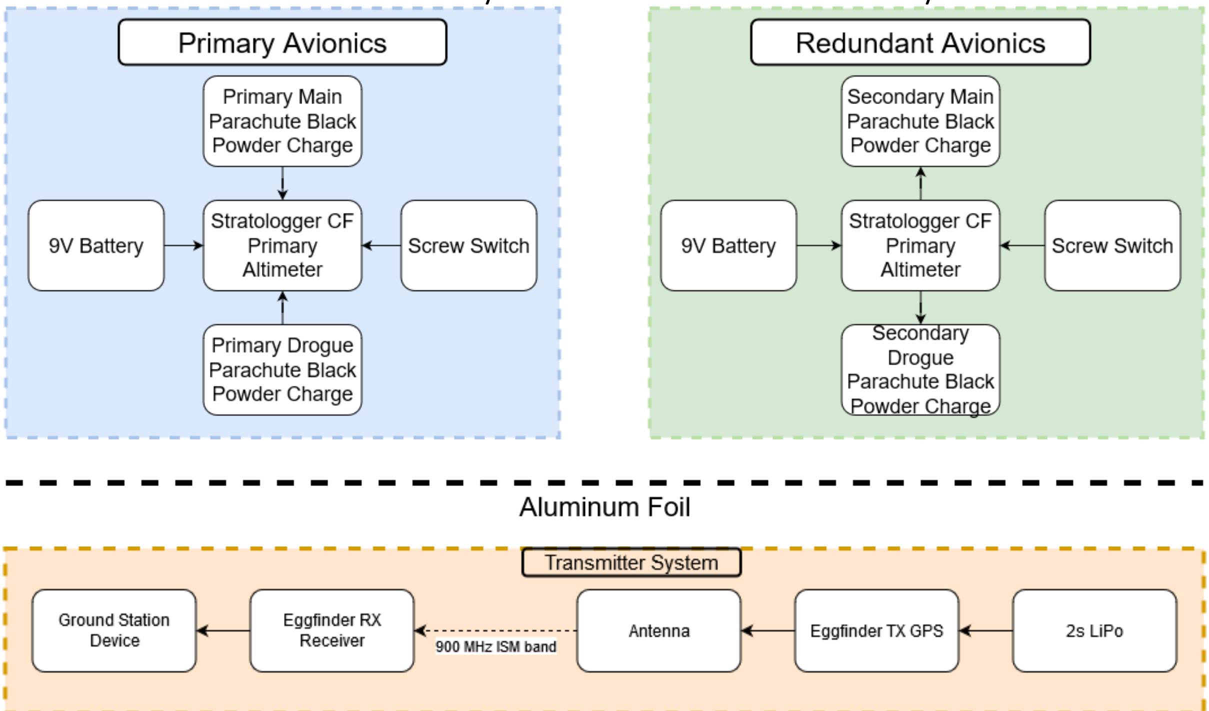

Figure 3-20 AV Bay Electronics Diagram

The diagram above shows the components housed within the AV bay and the wiring connections between them, as well as ground station components that are communicating with the AV bay via radio. The dashed blue box illustrates the components comprising the primary altimeter system. The primary altimeter system will contain the competition altimeter which will record the team's official apogee, as well as all components necessary for the required recovery events. The altimeter has four pairs of terminal blocks that are connected to a 9V battery, a screw switch, and two E-matches. One E-match is inserted into the drogue black powder ejection charge and triggered at apogee, while the other is inserted into the main black powder ejection charge and triggered at 675 ft. An unused 9V battery will be inserted before each flight to ensure sufficient power to detonate both ejection charges and record data following the design maximum pad wait. The screw switches are activated using a Phillips-head screwdriver through holes in the AV bay, and once tightened are locked in the "ON" position and cannot be unintentionally disengaged.

The redundant or secondary altimeter system is shown in the dashed green box is almost identical to the primary altimeter system. The only differences are in the ejection charge timing and sizing. The drogue parachute ejection charge is programmed to detonate one second after apogee and the main parachute ejection charge is programmed to detonate at 650 ft.

Both primary and secondary avionics systems are capable of independently recovering the launch vehicle. Since they are not connected in any way, as is evidenced in the above block diagram, this creates a redundancy in the system.

Should any one component of either system fail, the other system will function in its place allowing for a safe recovery.

3.2.2.5 Avionics Bay Sampling Holes

For the barometric altimeters to properly detect the ambient pressure, the AV bay must remain pressurized to the ambient pressure for the duration of the flight. To this end, sampling ports are drilled through the AV bay to equalize the pressure as the launch vehicle ascends and descends. Should these holes be too large, the freestream velocity will impact the readings and the altimeters will be reading the total pressure as opposed to the static pressure. Too small, and the AV bay will fail to equalize. As the current leading design makes use of two PerfectFlite StratoLogger CF altimeters, the methodology for sizing the sampling holes is taken from the manufacturer's recommendation. The recommended formula for sizing the sample ports is as below, where P is the diameter of the holes, L is the compartment length, and D is the compartment diameter.

$$P = 0.0008LD^2 \tag{8}$$

This approach assumes four holes are drilled at 90° from each other around the circumference of the launch vehicle. These ports will be located on the band of body tube bonded to the AV bay coupler, with two of the holes allowing access to the screw switches to arm the recovery avionics.

3.2.2.6 Drogue Parachute Alternatives

Descent under the drogue parachute is constrained primarily by the descent velocity of the launch vehicle. Excessive drogue descent velocity leads to a greater deceleration during main parachute deployment. Severe opening shock has the potential to cause structural damage to the airframe or disrupt payload deployment events. Section 3.3.9 discusses opening shock in greater detail. Conversely, should the launch vehicle descent too slowly under drogue, descent times and wind drift become excessive. The terminal velocity of the launch vehicle under a parachute can be calculated via the formula below.

$$V = \sqrt{\frac{2gm}{AC_D\rho}} \tag{9}$$

This gives the descent velocity where m is the burnout mass of the launch vehicle, g is the gravitational acceleration, rho is the air density, A is the area of the parachute, and C_D is the drag coefficient of the parachute. It is assumed that the drag contribution of the body sections is negligible.

Table 3-8 Drogue Parachute Comparison Chart

Parachute	Drag Coefficient	Descent Velocity	Scent Time from Apogee to Main Deployment	Wind Drift from Apogee to Main Deployment (20 mph
Fruity Chutes 12 inch Classic Elliptical	1.34	181.3 ft/s	20.9 s	614.5 ft
Fruity Chutes 15 inch Classic Elliptical	1.37	143.3 ft/s	26.5 s	777.3 ft
Fruity Chutes 18 inch Classic Elliptical	1.43	117.0 ft/s	32.5 s	953.3 ft
Fruity Chutes 24 inch Classic Elliptical	1.47	86.4 ft/s	43.9 s	1289.0 ft
Fruity Chutes 24 inch Compact Elliptical	1.41	88.5 ft/s	42.9 s	1259.4 ft

Table 3-8 above shows a comparison of drogue parachute alternatives between 15 and 30 inches in diameter. In order to mitigate the potential for damage to the launch vehicle and payload, TDR 3.3 specifies that the launch vehicle shall not exceed a drogue descent velocity of 125 ft/s. Under this requirement, the 24" Classic Elliptical, the 24" Compact Elliptical, and the 18" Classic Elliptical are suitable for use as the drogue parachute since they do not exceed the drogue descent velocity restrictions.

3.2.2.7 Main Parachute Alternatives

Main parachute selection is constrained by limits on section kinetic energy on landing, descent time, and wind drift. All these requirements are a function of velocity, hence the most important parameter in the selection of the main parachute is again the descent velocity. The launch vehicle descent under main parachute is calculated using the same terminal velocity formula as was used for the drogue parachute in the preceding section. Here again the drag produced by the launch vehicle sections is considered negligible.

As the LOPSIDED-POS will be jettisoned from the launch vehicle during flight, the descent system shall be capable of landing in a safe manner should the payload fail to successfully separate from the recovery harness per TDR 3.10. Therefore, calculations are performed for both scenarios, with the LOPSIDED-POS successfully separating and remaining attached to the recovery harness.

Table 3-9 Main Parachute Comparison Chart – LOPSIDED-POS Attached

Parachute	Drag Coefficient	Landing Velocity	Maximum Section Kinetic Energy	Descent Time from Main Deployment	Wind Drift Under Main (20 mph)
Fruity Chutes 96 inch Iris UltraCompact	2.09	18.1 ft/s	68.3 ft-lbf	37.2 s	1091.2 ft
Fruity Chutes 120 inch Iris Ultra Standard	2.11	14.5 ft/s	55.2 ft-lbf	46.7 s	1369.9 ft

Table 3-10 Main Parachute Comparison Chart – LOPSIDED-POS Separated

Parachute	Drag Coefficient	Landing Velocity	Maximum Section Kinetic Energy	Descent Time from Main Deployment	Wind Drift Under Main (20 mph)
Fruity Chutes 84 inch Iris UltraCompact	2.13	17.9 ft/s	66.5 ft-lbf	36.4 s	1067.7 ft
Fruity Chutes 84 inch Iris Ultra Standard	2.13	17.9 ft/s	66.5 ft-lbf	36.4 s	1067.7 ft
Fruity Chutes 120 inch Iris UltraCompact	2.11	12.6 ft/s	33.1 ft-lbf	51.7 s	1516.5 ft
Rocketman 144 inch Pro-X	0.82	16.8 ft/s	58.8ft-lbf	38.8 s	1138.1 ft

Three different styles of parachute are presented in the tables above, with each being the smallest example of its type able to meet the kinetic energy requirements. The descent rate was calculated as described above, and the maximum section kinetic energy is calculated as detailed in Section 3.3.6 for nominal deployment of the LOPSIDED-POS. Descent time and wind drift calculations are also included as calculated in Sections 3.3.7 and 3.3.8. This gives a comprehensive picture of the design factors influencing the selection of the main parachute. All parachute alternatives in the table above satisfy requirement NASA 3.3, with the Pro-X having a maximum section kinetic energy of 60.2 ft-lbf and the two Fruity Chutes parachutes having a maximum section kinetic energy of 68.2 ft-lbf. The main disadvantage of the Pro-X is the parachute's packed dimensions of 4" D x 15" L which approach the limits of the main parachute bay. This excessively constricts the volume available for the black powder charges to expand into. While the Fruity Chutes parachutes do have an increased kinetic energy at landing, they offer a decreased wind drift and descent time from main parachute opening.

3.2.2.8 Payload Parachute Alternatives

Selection of the payload parachute under which the LOPSIDED-POS will descend is dependent on kinetic energy at landing and wind drift requirements, as the payload recovery is exempt from the restrictions on descent time per NASA 3.11. As laid out in Section 3.2.1, the payload is jettisoned from the main parachute recovery harness at 500 ft. After this it descends to 200 ft under the furled payload parachute, before being slowed to landing following release of the parachute retention device. An analysis of the drag of the furled parachute is required to fully assess the feasibility of using the furled parachute as a streamer. Should it be determined that excessive opening shock will result from using the furled parachute as a streamer, a small pilot parachute will be used with the payload parachute to slow the LOPSIDED-POS during the first stage of its independent descent. The table below presents descent parameters for the payload parachute alternatives.

Table 3-11 Payload Parachute Comparison Chart

Parachute	Drag Coefficient	Landing Velocity	Maximum Section Kinetic Energy	Wind Drift from Apogee (20 mph)
Fruity Chutes 36 inch Classic Elliptical	1.43	28.5 ft/s	123.1 ft-lbf	2062.8 ft
Fruity Chutes 48 inch Classic Elliptical	1.44	21.3 ft/s	68.7 ft-lbf	2167.4 ft
Fruity Chutes 60 inch Iris UltraCompact	2.16	13.9 ft/s	29.3 ft-lbf	2386.9 ft

Of the three payload parachute alternatives presented, the 36” Classic Elliptical is unable to be used as it exceeds the kinetic energy requirement set forth by NASA 3.3. Of the two remaining parachutes, it is relevant for the purposes of analyzing the LOPSIDED-POS descent to reference manufacturer data provided for the purpose of calculating the drag coefficient. The 60” Iris UltraCompact is rated for a descent velocity of 20 ft/s with a mass of 19 lbs attached. Since this reference mass is almost double the payload weight, the concern of excessive drift presents itself for the 60” Iris UltraCompact parachute. The 48” Classic Elliptical is better sized for the payload and boasts a smaller packing volume while still remaining within the kinetic energy limits set forth by NASA 3.3.

3.2.2.9 Parachute Deployment and Protection Alternatives

Two different methods of shielding parachutes from ejection gasses were considered. The first method is to wrap the folded parachute in a Nomex cloth, which is heat resistant and will shield the parachute from the hot ejection gasses. Alternatively, a Nomex deployment bag could be utilized. A deployment bag has the advantages of more fully containing the parachute during flight, providing an organized fold for the shroud lines, and reducing the importance of proper parachute folding technique for successful parachute deployment. Use of deployment bags is limited by the size of available deployment bags on the market, with most deployment bags sold for larger parachutes and not for parachutes of the size of a typical drogue parachute.

3.2.2.10 Shock Cord Selection

During descent, lengths of shock cord will be used to tether the launch vehicle sections together. These shock cords must be strong enough to withstand the force of decelerating body sections during parachute deployment and parachute placement must be such that body sections cannot impact each other during descent. As detailed in Section 3.3.9, the maximum deceleration experienced by the launch vehicle will be 11 g's at main deployment. This translates to a maximum load experienced by a length of shock cord of 383.2 lbf if the entire launch vehicle and LOPSIDED-POS were suspended by the shock cord. Both 1 inch nylon webbing and 5/8 inch tubular Kevlar webbing are capable of withstanding loads in excess of 4000 lbf. The 5/8 inch tubular Kevlar shock cord is advantageous as it has a higher load rating of 6600 lbf and large quantities of this shock cord are already owned by the team. Kevlar has better heat resistance than Nylon, which is beneficial for applications where the Kevlar is exposed to hot ejection gasses.

Typically shock cords are 3 to 5 times longer than the length of the launch vehicle. The leading launch vehicle design has an overall length of 106 inches, the shock cord must therefore measure between 27 and 44 feet long. The team owns two 40 ft long shock cords made of 5/8 inch Kevlar, one of the suggested material alternatives. As this is at the longer end of the allowable length range, care must be taken in the parachute placement process to reduce the distance that body sections fall before being caught under parachute. During main parachute deployment the body sections may accelerate beyond the drogue descent velocity, increasing the deceleration during this event and therefore increasing the stresses on the airframe and recovery harness. Parachute placement along the shock cord can be determined based on the desired configuration of the launch vehicle body sections as it descends.

During drogue descent, the forward section of the launch vehicle must descend above the fin can since it contains the main parachute. Should the forward section descend below the fin can, the fin can could impact and foul the main parachute during its deployment. This scenario would result in an unacceptably fast main parachute descent velocity. In order to achieve this orientation, a constraint is imposed on the drogue parachute placement that the distance from the tip of the nose cone to the drogue parachute attachment point should be less than the distance from the drogue attachment point to the top of the fin can with at least a foot of clearance between the two. This assumes that both body sections hang vertically under the drogue parachute. The valid placement range for the drogue parachute attachment point is between 25% and 37% down the shock cord from the AV shock cord attachment point. The drogue parachute will be placed 1/3 of the way between the AV bay and the fin can, at 160 inches from the AV shock cord attachment point and 320 inches from the fin can attachment point. This provides a balance between maintaining body section clearance and reducing the distance either body section must fall during parachute deployment. The selected location provides a clearance between body sections of 3.8 ft.

Under the main parachute, parachute placement must be considered with respect to the LOPSIDED-POS deployment. It is therefore desirable to have the LOPSIDED-POS as far from any body section as possible to reduce the chance of it impacting a parachute or body section following separation from the main parachute recovery harness. To this end, the nose cone will be situated between the midsection and fin can. The drogue parachute will tend to pull the midsection and fin can sideways, away from the nose cone. Since the LOPSIDED-POS will be secured between the main parachute and midsection, it will be able to descend through the gap between the nose cone and the midsection and fin can tethered assembly. To facilitate this, the main parachute attachment point has been selected to be 320 inches from the nose cone attachment point corresponding to 2/3 of the way along the shock cord from the nose cone attachment point. This configuration allows for 7.2 ft of clearance between the nose cone and midsection, sufficient room to attach the LOPSIDED-POS. A diagram for the shock cord spacing for both parachutes is provided below.

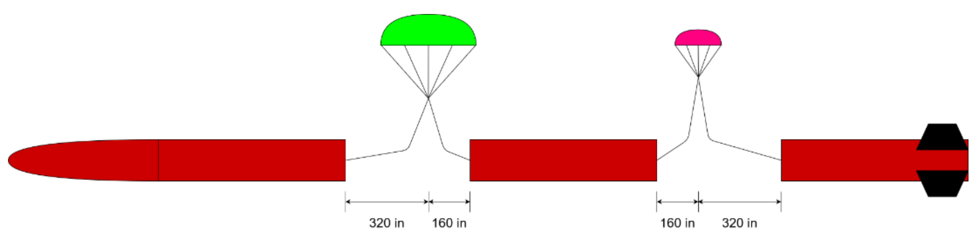

Figure 3-21 Leading Parachute Placement

3.2.2.11 Ejection Charge Sizing

Black powder ejection charge masses are determined by calculating the mass of black powder required to reach a known cavity pressure sufficient to break the shear pins holding the body sections together. It is assumed that the cavity volume is constant for the duration of this process, which is a valid assumption given the rapid burn rate of black powder. First, the volume of the cavity that the black powder will detonate into is determined. It is assumed that the drogue parachute and attached shock cord occupy negligible volume, while the main parachute and payload volumes are subtracted. With the volume determined, a desired pressure of 10 psi is calculated for. This will produce sufficient force to separate body sections without damaging the airframe or recovery systems. For a desired pressure and volume, the mass of black powder required can be calculated using the below equation.

$$PV = mRT \tag{10}$$

Here R is the gas constant for the black powder combustion products, T is the mixture temperature of the black powder and existing gas in the cavity, P is the target pressure, V is the cavity volume, and m is the mass of black powder. The combustion gas properties are known constants for the 4f black powder used for the ejection charges. Solving for m yields the required mass of black powder.

To complete the redundant recovery systems discussed earlier in Section 3.2.2.4, a secondary black powder charge is utilized. If the primary black powder charge of the calculated mass fails to separate the body sections during flight, the secondary charge will be 0.5 grams larger. This increase in mass will force separation if the first charge detonates but does not fully separate the body sections, without being so large that over pressurization occurs due to the increase in pressure.

According to the previous equation, for a 6 inch internal diameter launch vehicle and a cavity length of 6.125 inches, a black powder charge of 1 gram will produce a cavity pressure of 11.2 psi and an ejection force of 316.2 lbf. For the main ejection charge, a cavity length of 47 inches minus the volume occupied by the main parachute and payload gives an ejection charge of 3.2 grams producing a cavity pressure of 10.2 psi. These values represent the primary charge masses, applying the secondary charge mass increases gives a 1.5 gram drogue secondary mass and a 3.7 gram main secondary mass.

In accordance with requirement NASA 3.2, a ground ejection test of both primary charges will be conducted. This test will experimentally verify that both the main and drogue black powder ejection charges are correctly sized and loaded into the launch vehicle. Further, it allows the launch vehicle assembly team to practice and check the procedure for loading the energetic material onboard the rocket. Should the black powder ejection charges fail to successfully separate the body sections during testing, the ejection charge mass will be corrected and the test repeated.

3.2.3 Recovery Leading Design

Holding the core of the launch vehicle's recovery system is the AV bay, which will contain the AV sled. The AV sled will be constructed from laser cut aircraft grade birch plywood, using a jigsaw puzzle design of interlocking teeth. Plywood is more durable than 3D printed ABS plastic while being lighter than aluminum and easier to manufacture than fiberglass. Laser cutting allows for rapid prototyping and quick design iterations at a low cost. The lengthwise configuration that will be used for the AV sled allows sufficient room for all required components to be securely mounted in a configuration that is easy to access and wire. A sheet of aluminum foil will be placed between the altimeters and GPS transmitter to reduce the risk of interference.

The drogue parachute selected for the leading design is a Fruity Chutes 18” Compact Elliptical parachute, which has been selected for its low descent velocity. Smaller parachutes provide an excessive descent velocity with the potential for damage to the launch vehicle airframe during the deceleration of main parachute deployment. The 18” Compact Elliptical descends at 117.0 ft/s which is less than the 125 ft/s drogue descent velocity limit imposed by TDR 3.3. Since the descent velocity of the 18” Compact Elliptical is greater than the 86.6 ft/s of descent velocity calculated for the 24” Classic Elliptical, the Compact Elliptical design will be used as the increased descent velocity correlates to decreases in descent time and wind drift. Additionally, the 24” Classic Elliptical parachute is incapable of meeting the wind drift requirements with the leading main parachute alternative. These characteristics form the most desirable combination for the drogue parachute and hence this is the leading candidate. The drogue parachute will be protected from hot ejection gasses with a Nomex sheet wrapped around the folded parachute and secured to the recovery harness.

The main parachute selected for the leading design is a Fruity Chutes 120” Iris UltraCompact parachute. Although a Fruity Chutes 84” Iris Ultra Standard parachute provides more favorable wind drift and descent time characteristics while meeting the on-design kinetic energy requirements, in the scenario that the payload does not separate from the recovery harness the kinetic energy requirements are exceeded. The 120” Iris UltraCompact with the selected leading drogue parachute alternative is capable of meeting the wind drift and descent time requirements for the selected main parachute deployment altitude while still sufficiently slowing the launch vehicle such that kinetic energy requirements are met for potential off-design landing configurations with the payload attached to the main parachute recovery harness or still inside the payload bay. The main parachute will be protected from hot ejection gasses using a Nomex deployment bag.

The payload parachute selected to recover the LOPSIDED-POS is the Fruity Chutes 48” Classic Elliptical. This parachute provides a sufficiently slowed landing velocity of 20.5 ft/s and is within the kinetic energy limitations set forth by NASA 3.3. A larger parachute would be too large for the mass of the LOPSIDED-POS, leading to excessive wind drift for the deployment altitude. Smaller parachutes fail to sufficiently slow the LOPSIDED-POS, thus failing to meet kinetic energy restrictions. Using the furled parachute as a streamer is selected over a pilot chute since it reduces the number of parachutes in the system leading to a simpler recovery harness and reduced risk of recovery system failure.

Recovery events will be controlled by a pair of PerfectFlite StratoLogger CF altimeters mounted on the AV sled and housed in the AV bay. These altimeters provide a balance of compact form factor, adaptable flight profile programming, favorable human factors considerations during wiring of the AV sled, and proven reliability. Two fully separate altimeter systems provide an independent, redundant method of firing the primary and secondary sets of black powder ejection charges. Tracking of the launch vehicle will be provided via an Eggfinder TX/RX GPS transmitter system. The Eggfinder RX Bluetooth Dongle will be connected to a field recovery team members’ laptop or phone to display the launch vehicle location in real time.

3.3 Mission Performance Predictions

3.3.1 Launch Day Target Altitude

Based on simulations ran in RockSim, the team has determined the target apogee for the launch vehicle to be 4473 feet.

3.3.2 Flight Profile Simulations

The team relied on flight simulation data from RockSim to determine the characteristics of the launch vehicle. Simulations were based on a 5° cant, a 12-foot launch rail, and wind conditions ranging from 3 to 14.9 mph. Twenty simulations were run and their apogees averaged to calculate the team's declared apogee for launch day. Figure 3-22 shows one of the simulated flight profiles. The launch vehicle's rail exit velocity is 73.75 ft/s. The launch vehicle reaches a maximum Mach number of 0.488 during flight, well under the Mach 1 limit imposed by requirement NASA 2.22.6.

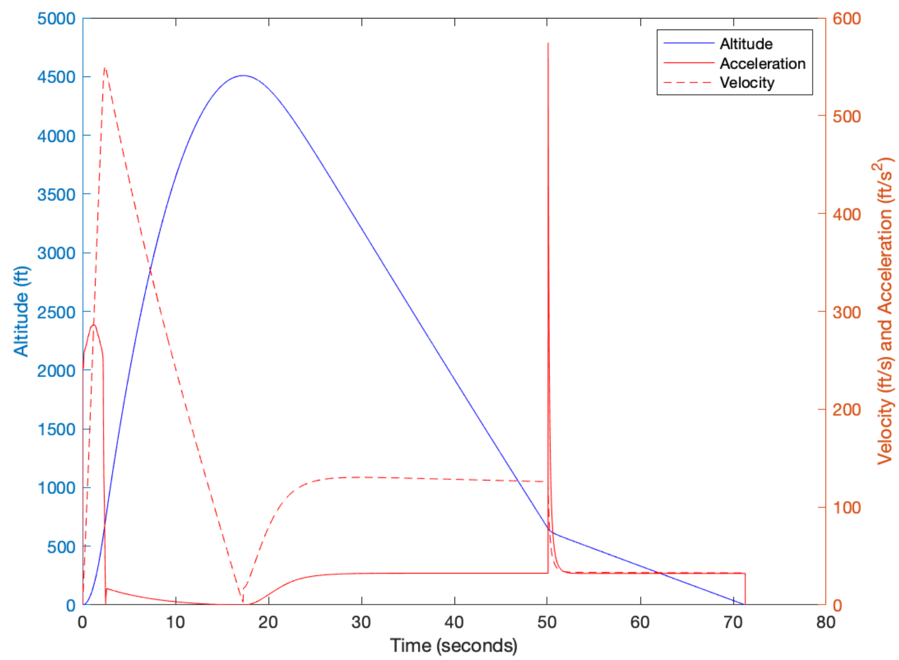

Figure 3-22 Launch Vehicle Flight Profile

Given the inherent uncertainty in both the wind speeds on launch day and the final weight of the constructed payload, a tolerance study was conducted in RockSim. The figure below shows the predicted apogee of the launch vehicle given different potential weights of LOPSIDED and different varying windspeeds on launch day. Any final LOPSIDED weight within 7.5% of the predicted weight will result in an apogee within 100 feet of the declared apogee. Wind speeds above 14.9 MPH were not simulated for this study as, for safety reasons, the team has decided not to launch the rocket in wind speeds at or above 15 MPH.

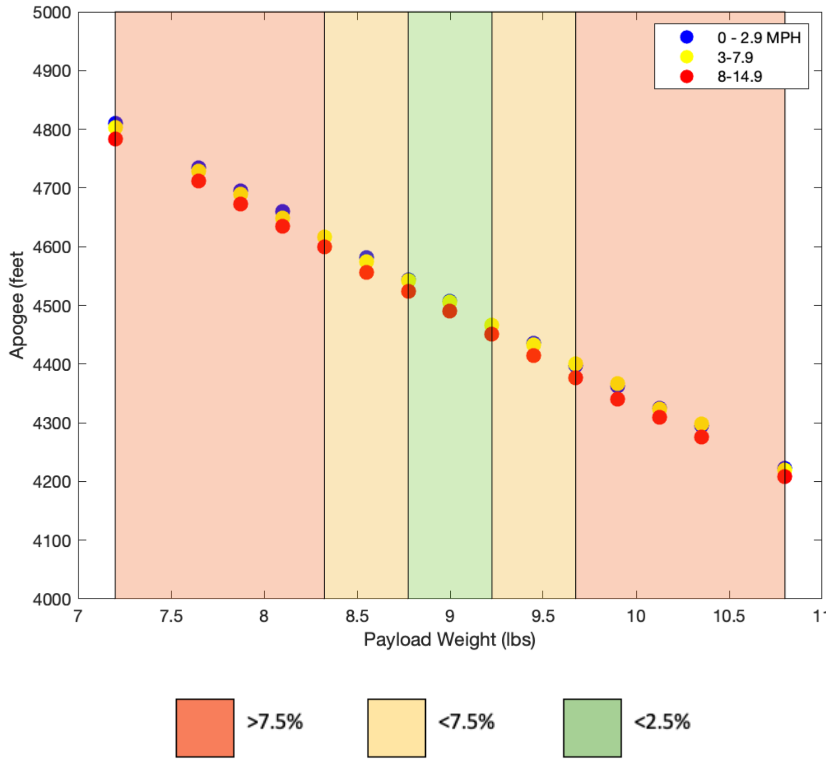

Figure 3-23 Weight of Payload vs. Apogee

3.3.3 Altitude Verification

Several measurements from the rockets are necessary to perform hand calculations to calculate the apogee of launch vehicle and can be found in Table 3-12. The first step to altitude verification is to calculate the wind resistance coefficient, *k*, using the density of air, ρ, the reference area of the launch vehicle, *A*, and the C_D of the launch of vehicle. Additionally, in order to simplify the final equation, the variables q and x are solved using average thrust, *T*, mass of the rocket, M, and gravity, *g*.

Table 3-12 Variable Definition for Altitude Verification

Quantity	Variable	Value	Units
Density	ρ	1.225	Kg/m^3
Radius	R	0.078	m
Coefficient of Drag	C_D	0.292	N/A
Average Thrust	T	1,567.8	Newtons
Mass	M	20.07	Kg
Gravity	g	9.81	m/s^2
Burn Time	t	2.4	s

$$A = \pi R^2 = 19.113 * 10^{-3}\ m^2$$

$$k = 0.5\rho C_D\, A = 3.418 * 10^{-3}\frac{kg}{m}$$

$$q = \sqrt{\frac{T - Mg}{k}} = 633.314\frac{m^2}{s^2}$$

$$x = \frac{2kq}{M} = 0.216\frac{m}{s^2}$$

With x and q calculated, the maximum velocity of the launch vehicle, v_{max}, can be calculated:

$$v_{max} = q\frac{1 - e^{-xt}}{1 + e^{-xt}} = 160.57\frac{m}{s}$$

Using v_{max}, the height upon motor burnout can be calculated:

$$h_{burnout} = -\frac{M}{2k}ln\left(\frac{T - Mg - kv_{max}{}^2}{T - Mg}\right) = 195.067m$$

The height gained during coast was calculated next:

$$h_{coast} = \frac{M}{2k}ln\left(\frac{Mg + kv_{max}{}^2}{Mg}\right) = 1085.217\mathrm{m}$$

The maximum height can be calculated by adding together the two previous values:

$$h_{max} = h_{burnout} + h_{coast} = 1280.284m = 4200.4\,feet$$

Table 3-13 Altitude Prediction Results

Method	Result
RockSim	4470 feet
Algebraic	4200 feet

The algebraic method yields a result within 6% of the RockSim calculations and, given how simple the method this, the team has decided that this is within an acceptable margin of error.

3.3.4 Stability Margin Simulation

According to RockSim, the leading design of the launch vehicle has an initial stability margin of 2.20 calipers and a stability margin of 2.38 upon departure of the launch vehicle. The stability margin of the launch vehicle over the entirety of its flight can be seen in the figure below.

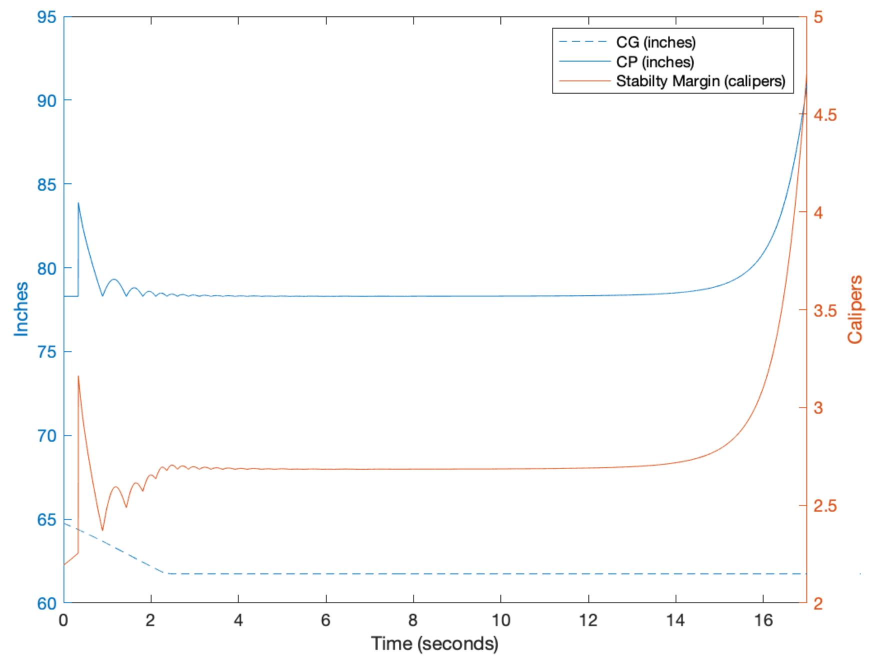

Figure 3-24 Stability Margin Simulation Results

Additionally, the Barrowman's method was used to calculate the initial stability margin in order to verify the results the team received from RockSim simulations. Barrowman's method is a simple series of algebraic equations that is popular in rocketry and is used to identify the center of pressure of the launch vehicle.

The first value that must be found is the arm length of the nose cone, X_N, which for an ogive nose cone has a linear relationship with the length of the nose cone, L_N. The leading design for the nosecone of the launch vehicle is 24 inches, which gives:

$$X_N = 0.466 \cdot L_N = 11.184 \; inches$$

The next value to be calculated is the sweep angle. The sweep angle is a simple trigonometric relationship between the fin semi-span, S, and the fin sweep length when measured parallel to the launch vehicle body, X_N. The leading design has S=6.03 inches and X_R=3.5 inches.

$$\theta = 90° - \tan^{-1}\frac{S}{X_R} = 30.13°$$

The sweep angle allows for the calculation of the fin mid-chord line length, L_F. Other values which are needed to calculate L_F are chord tip length, C_T, and chord root length, C_R. The leading design has C_T = 3.5 inches and C_R = 10.5 inches.

$$L_F = \sqrt{S^2 + \left(\frac{1}{2}C_T - \frac{1}{2}C_R + \frac{S}{\tan\theta}\right)} = 6.58 \; inches$$

With L_F calculated, the coefficient for the fins, C_F, can be calculated using the radius of the launch vehicle, *R,* and the number of fins, *N.* The leading design has $R = 3$ inches and $N = 3$.

$$C_F = \left(1 + \frac{R}{S+R}\right)\left(\frac{4N\left(\frac{S}{2*R}\right)^2}{1+\sqrt{1+\left(\frac{2L_F}{C_R+C_T}\right)^2}}\right) = 6.806$$

In order for the arm length of the fins to be calculated, the distance from the tip of the nose cone to the fin root chord leading edge, X_B, must be used. The leading design has an $X_B = 95.25$ inches.

$$X_F = X_B + \frac{X_R}{3}\frac{C_R + 2C_T}{C_R + C_T} + \frac{1}{6}\left(C_R + C_T - \frac{C_R C_T}{C_R + C_T}\right) = 98.6\ inches$$

Given that $C_N = 2$, with these values, the center of pressure is calculated:

$$X_{CP} = \frac{C_N X_N + C_F X_F}{C_N + C_F} = 78.46\ inches$$

Using the center of gravity from Rocksim, $X_{CG} = 64.7$ inches, the stability margin can be calculated. The stability margin from both the Barrowman method and RockSim simulations can be found in the table below. There's a 3.93% difference between the two calculation methods, but both stay above the NASA requirement of 2.0 calipers at launch.

Table 3-14 Stability Margin Calculations

Computation Method	Result (calipers)
Barrowman	$S_{M,B} = \frac{X_{CP}-X_{CG}}{R}$ =2.29
RockSim	2.20

A summary of the variables, and the values of those variables, used in the Barrowman method can also be found below.

Table 3-15 Stability Variable Definitions

L_F – Mid-chord line length	X_F – Fin arm length	X_R – Fin sweep length measured parallel to launch vehicle body	X_B – nose cone tip to fin root chord leading edge
L_N – Length of nose cone	C_N – Nose cone coefficient	X_N – Nose cone arm length	θ – Sweep angle
R – Radius of launch vehicle	C_R – Root chord length	C_T – Tip chord length	
S – Fin semi-span	N – Number of fins	C_F – Fin coefficient	

Table 3-16 Measured Stability Variable Values

Variable	Input Value	Units
C_N	2	N/A
L_N	24	Inches
R	3	Inches
S	6.03	Inches
N	4	N/A
C_R	10.5	Inches
C_T	3.5	Inches
X_B	95.25	Inches
X_R	3.5	Inches
CG	64.83	Inches

Table 3-17 Calculated Stability Variable Values

Variable	Output Value	Units
X_N	11.184	Inches
θ	30.13°	Degrees
L_F	6.58	Inches
C_F	6.806	N/A
X_F	98.6	Inches
X_{CP}	78.46	Inches
$S_{M,B}$	2.29	Calipers

3.3.5 Stability Margin Tolerance Study

Due to uncertainties in the final manufactured weight and CG of the payload, the team determined it would be necessary to conduct a tolerance study on how these uncertainties would affect the stability margin of the launch vehicle.

In order to perform this study, data was taken from RockSim manually while the weight and CG of the payload were varied. The weight was varied in 0.225lb increments up to a threshold of ±1.35lb, which is 15% of the leading design weight of the payload. Additional values were evaluated at ±20% of leading design weight to account for edge cases. The CG of the payload, as measured from the front of the payload bay, was varied in 0.5 inch increments up to a threshold of ±2 inches, which is 11.11% of the leading design CG for the payload. The results of the tolerance study can be seen in the figure below.

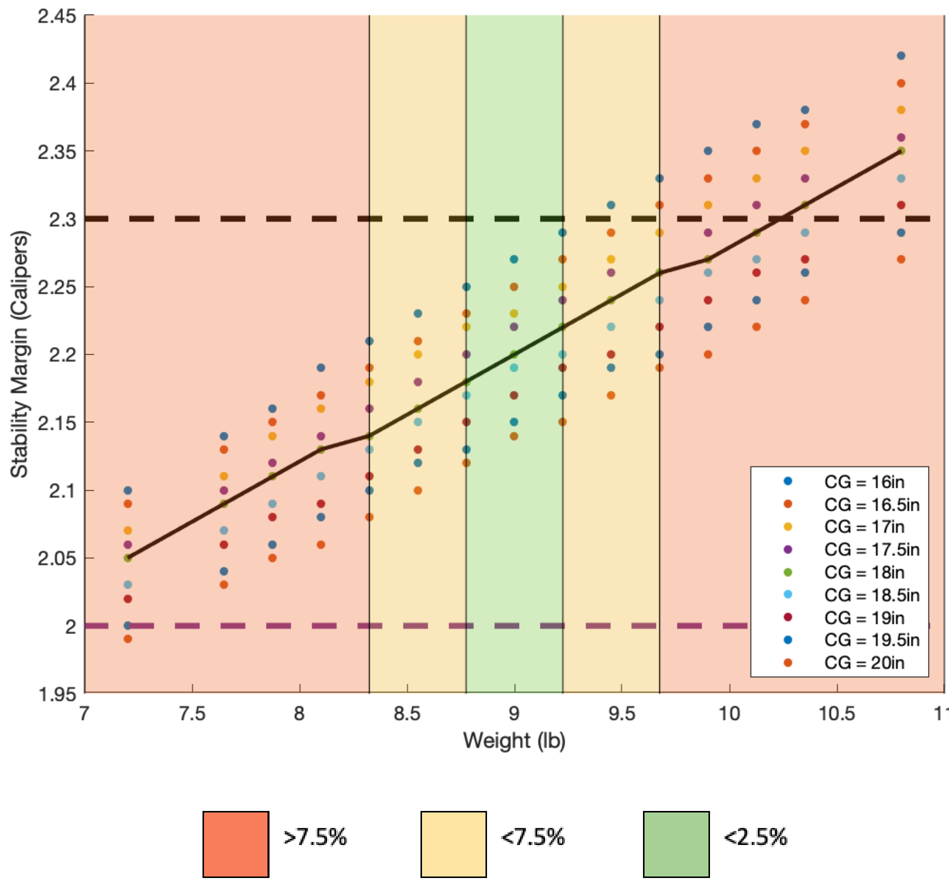

Figure 3-25 Stability Tolerance Study

The lower bound of a stability margin of 2.0 calipers is defined by NASA and the upper bound of 2.3 calipers is defined by TDR 2.4 to reduce the chances of weathercocking after departure from the launch rail. The leading design CG of the payload is indicated by the black line, making it easy to tell which data points are above or below the leading design. What the tolerance study reveals is that the payload is skewing towards the heavy end of the acceptable stability margin envelope; there is more room to lower the weight of the payload than there is to increase it. The tolerance study also reveals that while the CG of the payload does affect stability, the weight of the payload has a much greater effect. Both of these insights will be helpful when further iterating on the design of the payload and the launch vehicle itself.

3.3.6 Kinetic Energy at Landing

Kinetic energy can be calculated using the below equation, where E is the kinetic energy of the object, m is its mass, and V is its velocity.

$$E = \frac{1}{2}mV^2 \tag{11}$$

For each body section, the velocity of the body section for which it would have 75 ft-lbf of kinetic energy at landing has been tabulated below. The smallest of these velocities can be identified and serves to constrain the selection of a main parachute.

Table 3-18 Body Section Maximum Descent Velocity

Section	Mass	Mass (LOPSIDED-POS)	Descent Velocity	Descent Velocity (LOPSIDED-POS)
Nosecone	0.225315 slugs	0.528335 slugs	25.8 ft/s	16.8 ft/s
Midsection	0.334088 slugs	0.334088 slugs	21.2 ft/s	21.2 ft/s
Fin can	0.414601 slugs	0.414601 slugs	19.0 ft/s	19.0 ft/s

Using this data, the Fruity Chutes 120" Iris UltraCompact parachute has been selected as the main parachute as described in Section 3.2.3. The descent velocity of this parachute is calculated for the vehicle mass both with and without the LOPSIDED-POS attached to the launch vehicle as detailed in Section 3.2.2.7. The kinetic energy of each body section at landing has been calculated and tabulated below.

Table 3-19 Body Section Kinetic Energy at Landing

Section	Mass	Descent Velocity (LOPSIDED-POS)	Descent Velocity	Kinetic Energy (LOPSIDED-POS)	Kinetic Energy
Nosecone	0.225315 slugs	14.5 ft/s	12.6 ft/s	23.6 ft-lbf	18.0 ft-lbf
Nosecone w/ LOPSIDED-POS	0.528335 slugs	14.5 ft/s	N/A	55.2 ft-lbf	N/A
LOPSIDED-POS	0.30304 slugs	14.5 ft/s	N/A	31.7 ft-lbf	N/A
Midsection	0.334088 slugs	14.5 ft/s	12.6 ft/s	34.9 ft-lbf	26.6 ft-lbf
Fin can	0.414601 slugs	14.5 ft/s	12.6 ft/s	43.3 ft-lbf	33.1 ft-lbf

With a successful separation of the LOPSIDED-POS, the heaviest section of the launch vehicle will descend with a kinetic energy of 33.1 ft-lbf. If the LOPSIDED-POS successfully exits the payload bay but remains attached to the recovery harness, the descent velocity will increase with landing mass and the maximum body section kinetic energy becomes 43.3 ft-lbf. Should the LOPSIDED-POS remain in the payload bay all the way to descent, the maximum section kinetic energy becomes 55.2 ft-lbf. All three of the maximum body section kinetic energy predictions calculated for the leading main parachute alternative are within the kinetic energy requirements set forth by NASA 3.3, therefore the 120" Iris UltraCompact is an effective selection for the main parachute.

By squaring the terminal velocity equation given in Section 3.2.2.6, it can be shown that the velocity squared term in the kinetic energy equation scales linearly with mass. Should all body section masses be increased proportionally, kinetic energy will scale quadratically with mass. To this end, the total mass of the launch vehicle could increase by 10.75% and still meet kinetic energy requirements. This buffer accounts for any increases in mass due to manufacturing, changing mission or payload requirements, or model inaccuracies.

3.3.6.1 Alternative Calculation Method

RockSim simulations performed to date using the leading launch vehicle configuration have also been used to calculate kinetic energy at landing. Since RockSim does not treat sections as independent entities during descent calculating the mass and kinetic energy within RockSim would prove difficult and require significant workarounds. Therefore, the descent velocity at landing is used along with the section mass estimates to determine the kinetic energy at landing.

Table 3-20 Body Section Kinetic Energy at Landing – RockSim Calculations

Section	Mass	Descent Velocity (LOPSIDED-POS)	Kinetic Energy (LOPSIDED-POS)
Nosecone w/ LOPSIDED-POS	0.528335 slugs	14.7 ft/s	57.1 ft-lbf
Midsection	0.334088 slugs	14.7 ft/s	36.1 ft-lbf
Fin can	0.414601 slugs	14.7 ft/s	44.8 ft-lbf

Table 3-20 lists the section kinetic energy at landing based on RockSim simulations. Here the descent velocity is found to be 14.7 ft/s, which is slightly higher than the descent velocity calculated by hand. This corresponds to slight increases in kinetic energy with a maximum of 57.1 ft-lbf, 1.9 ft-lbf more than as calculated by hand. This continues to allow for a sufficient margin of launch vehicle mass increase without exceeding the kinetic energy limits set forth by requirement NASA 3.3. This difference in descent velocity is due to differences in how RockSim models the parachute. The RockSim model accounts for the spill hole in calculating the parachute drag, while the hand calculations calculate a drag coefficient based on empirical data provided by the manufacturer directly. Despite calculation differences, the difference in final values is within acceptable ranges to confirm the validity of these calculations.

3.3.7 Descent Time Calculations

Descent time calculations are performed as a step in the wind drift calculations and are detailed in the following section.

3.3.8 Wind Drift Calculations

Wind drift calculations rely on several key assumptions. First, it is assumed that the rocket travels vertically to the predicted apogee, deploys the drogue parachute and immediately begins descending at the drogue parachute’s terminal velocity. The same assumption is applied to the main parachute, where it is assumed that the launch vehicle immediately decelerates to the main parachute terminal velocity at 675 ft, and to the payload deployment where it is assumed that when the LOPSIDED-POS separates from the main parachute recovery harness the launch vehicle assembly immediately decelerates with the reduction in weight. From apogee to landing, the launch vehicle and separated LOPSIDED-POS are assumed to travel in a single direction at a constant speed equal to the wind condition.

Although this calculation approach is not wholly accurate to the actual wind drift performance, it is a reasonable worst-case approximation. The assumption of the launch vehicle instantly reaching terminal velocity after parachute deployment means that the modeled vehicle descent time will be a few seconds shorter than the actual vehicle descent time.

The distances between apogee, main deployment, and landing are known as is the descent velocity of the launch vehicle during each stage of the recovery process. The descent time can therefore be easily calculated as below where t is the total descent time, z_d is the drogue deployment altitude, z_{m-} is the main deployment altitude, z_p is the LOPSIDED-POS deployment altitude, v_d is the descent velocity under drogue, v_{m1} is the descent velocity under main with the LOPSIDED-POS attached, and v_{m2} is the descent rate under main following separation of the LOPSIDED-POS.

$$t = \frac{(z_d - z_m)}{v_d} + \frac{(z_m - z_p)}{v_{m1}} + \frac{(z_p)}{v_{m2}} \quad (12)$$

Leading design parachutes and an apogee of 4473 ft gives a descent time of 84.7 seconds using the equation above. Multiplying this descent time by a given wind speed gives the distance traveled across the ground by the launch vehicle constantly sustaining these wind speeds, as seen in the table below.

Table 3-21 Wind Drift and Descent Time

Speed	Apogee	Descent Time	Drift Distance
0 mph	4473 ft	84.2 s	0 ft
5 mph	4473 ft	84.2 s	617.5 ft
10 mph	4473 ft	84.2 s	1234.9 ft
15 mph	4473 ft	84.2 s	1852.4 ft
20 mph	4473 ft	84.2 s	2469.9 ft

At the maximum permissible wind speed for launch of 20 mph, the launch vehicle's estimated drift distance is 2484.5 ft, only 15.5 ft shy of the maximum allowable value by requirement NASA 3.10. Since this model constitutes the worst-case scenario for wind drift, this proximity to the limit is deemed acceptable. While this does not allow any margin of error for reductions in launch vehicle weight, ballast may be added as detailed in Section 3.1.4.4 without exceeding recovery system kinetic energy limits to increase descent speed and reduce wind drift.

3.3.8.1 Alternate Calculation Method

Wind drift and descent time have also been calculated using RockSim. The wind was modeled as a constant speed, with wind speed being the only variable changed between simulations. The total descent time was backed out by subtracting the time to apogee from the total flight time. Wind drift distance was also calculated by taking the difference of the range at apogee and the range at landing.

Table 3-22 Wind Drift and Descent Time – RockSim Calculations

Speed	Apogee	Descent Time	Drift Distance
0 mph	4447 ft	75.4 s	0 ft
5 mph	4474 ft	78.2 s	573.5 ft
10 mph	4481 ft	77.9 s	1142.5 ft
15 mph	4466 ft	78.9 s	1735.8 ft
20 mph	4426 ft	77.9 s	2285.1 ft

Beyond the trends in drift distance, which are very similar to those computed using hand calculations, the RockSim calculations show the effects of wind speed on descent time and, with both first increasing then decreasing as the wind speed increases. The RockSim values are roughly 92% of the hand calculations, which is sufficient to be considered similar. This difference is most likely due to differences in how the parachute is modeled. The RockSim model accounts for changes in apogee and models the parachute deployment in a non-instant manner, thus reducing the descent time of the launch vehicle.

3.3.9 Parachute Opening Shock Calculations

In order to model the stresses that will be applied to the shock cords and recovery harness attachment points, it is important to approximate the deceleration of the launch vehicle and the time it takes the parachute to inflate. A rule of thumb for parachute inflation time is that it takes a time approximately equal to the time it takes air to travel from the edge to the center of the furled parachute. This is given in the equation below, where t_i is the inflation time of the parachute, r is the parachute radius, and V is the descent velocity prior to parachute deployment. The coefficient of 8 is taken from Ludtke for a cloth parachute.

$$t_i = \frac{8 * r}{V} \tag{13}$$

From this and the difference in terminal velocities between the leading main and drogue parachute alternatives, the equation below gives the deceleration of the launch vehicle at main parachute deployment where a is the deceleration, v_d is the drogue descent velocity, v_m is the main descent velocity, and t_i is the parachute inflation time.

$$a = \frac{v_d - v_m}{t_i} \tag{14}$$

For a total launch vehicle mass with the LOPSIDED-POS included of 1.526 slugs, the main parachute opening shock is computed to be 458 lbf using the deceleration value as computed above. The individual loading experienced by each section is given in the table below.

Table 3-23 Body Section Opening Shock Calculations

Section	Mass	Opening Shock
Forward Section	0.528355 slugs	158 lbf
Midsection and Fin Can	0.748689 slugs	225 lbf

4. Payload Criteria

4.1 Payload Objective

The objective of the payload mission is to design, construct, and launch a planetary lander within a high-powered launch vehicle. This lander will achieve an upright orientation and be able to self-level within five degrees of horizontal. The initial and final angles relative to horizontal will be recorded. After leveling is complete, the payload will capture a 360-degree panoramic photo of the landing site and transmit this photograph back to the team's ground station.

4.2 Payload Success Criteria

Table 4-1 Payload Success Criteria

Success Level	Payload Aspect	Safety Aspect
Complete Success	LOPSIDED successfully lands in its upright configuration; the self-leveling procedure is completed within 5 degrees of horizontal; the POS captures and transmits at least one 360-degree image of the landing site, which is received by the team.	No injuries are inflicted on individuals present during the execution of mission requirements.
Partial Success	LOPSIDED successfully lands in its upright configuration; the self-leveling procedure fails to level the vehicle within 5 degrees of horizontal, or the leveling system is impeded in some way; the POS captures and transmits at least one 360-degree image of the landing site, which is received by the team.	One or more close calls involving individuals present during the execution of mission requirements, but no injuries occur.
Partial Failure	LOPSIDED fails to land upright; self-leveling cannot be attempted; the POS is still able to capture and transmit photos, but the images do not properly encapsulate the landing site; damage to LOPSIDED's landing gear/leveling system can be repaired.	Minor injuries inflicted on individuals present during the execution of mission requirements.
Complete Failure	LOPSIDED's deployment system fails, leading to an unrecoverable payload; no images are captured or transmitted.	Major injuries inflicted on individuals present during the execution of mission requirements.

4.3 Alternative Payload Designs

4.3.1 Alternate LOPSIDED Designs

LOPSIDED's main goal is to provide the conditions for POS to capture a usable 360-degree panorama. This requires the LOPSIDED to house the POS within its chassis and to re-orient that chassis using its leveling system. The main design components of the LOPSIDED can therefore be broken into its leveling system and its accommodating chassis design.

4.3.1.1 Leveling System

The leveling system largely determines the chassis shape as its requirements can vary between internal or external and the components needed to drive its mechanisms. This not only changes the chassis design but also the internal volume designated for other electrical components that the chassis will house.

4.3.1.1(a) Varying Leg Attachment Height

This design consists of each of the four landing legs attaching to a sled that slots into the payload chassis. This sled would be driven by a stepper motor and lead screw to convert the motion from rotational to linear. These components will be stored mostly inside the chassis. Varying the leg attachment points can alter the orientation of the body.

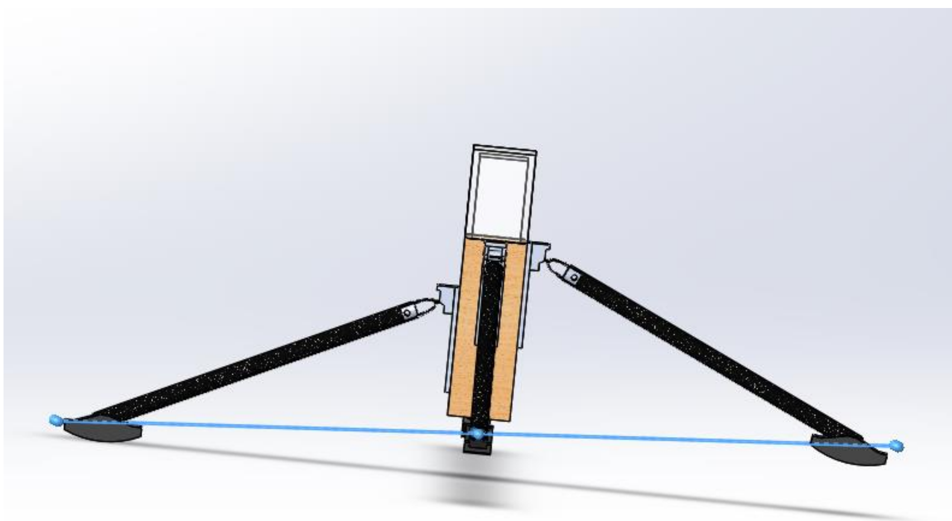

Figure 4-1 LOPSIDED Leg Tilt Demonstration

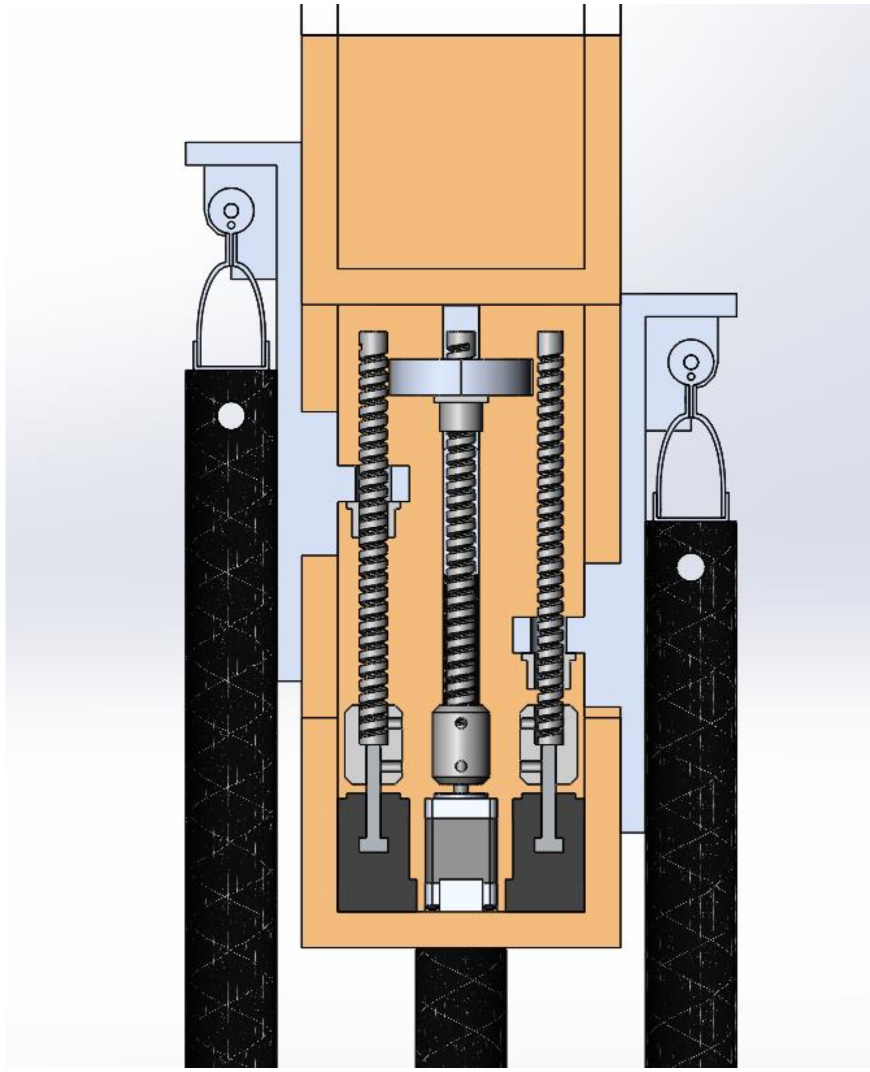

Figure 4-2 Leg Sled Attachments

Two main issues with this design are that it requires a large percentage of the chassis' internal volume for its components and that it does not offer a large re-orientation range. This design takes up as much space as it does, shown in Figure 4-2, because the leg sleds would need to travel a far enough distance in order to make a large enough change in the orientation angle. With this much internal volume being designated to the leveling system, there would not be enough room to fit the POS and controller for the leveling system as well as batteries and other components that would be needed. This design offers a maximum tilt of around 4 degrees. Given the 5-degree tolerance, this means that the lander can only land on a surface with a maximum grade of 9 degrees. As defined in the derived requirements, the payload must be able to tilt a minimum of 15 degrees of its neutral position.

4.3.1.1(b) 2-Axis Vertical and Horizontal Rotation about Support Structure

This design centers around using 2 axes that the chassis can rotate about. A static support structure consisting of a ring that the four legs connect to will interface with the body in such a way to allow these degrees of freedom.

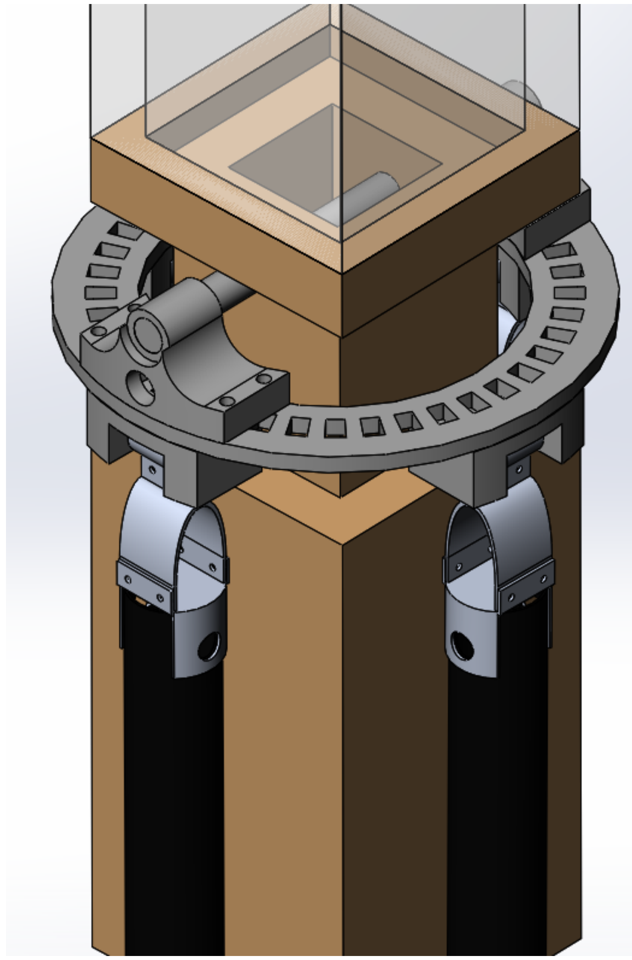

Figure 4-3 2-Axis Vertical and Horizontal Rotation Mechanism

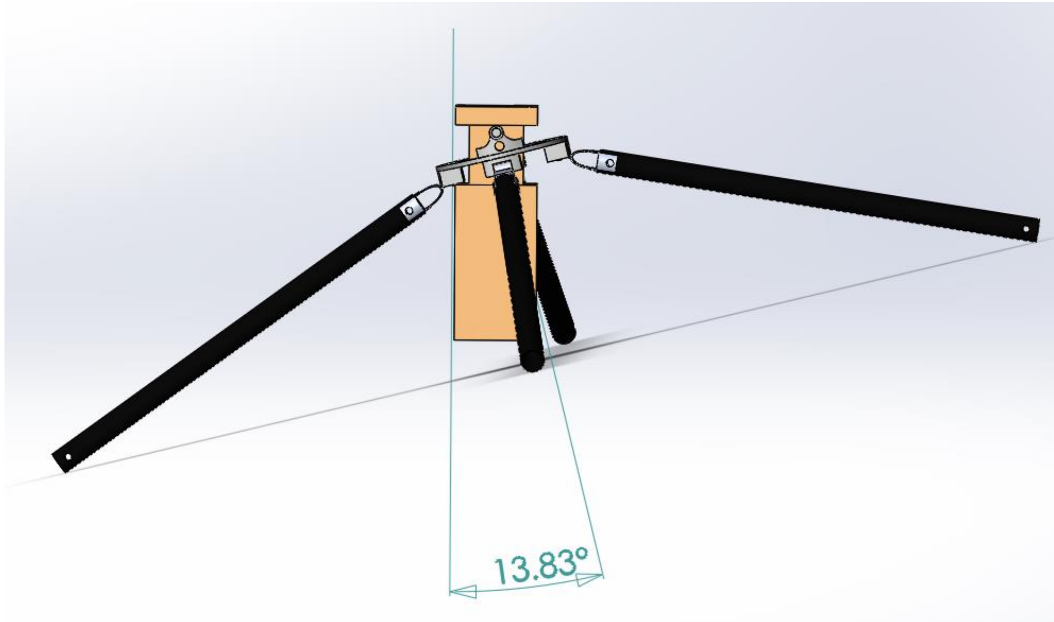

Figure 4-4 2-Axis Vertical and Horizontal Rotation Tilt Range

As shown in Figure 4-3 one axis will be a rod that passes through the body about which it can rotate. The second axis is the vertical axis. The horizontal rod that passes through the body will connect to two mounting brackets. These brackets will interface with the support structures ring which will have slots for gear teeth to interface with. A gear, driven by a motor, will move these brackets along the ring, rotating the body about the vertical axis. The tilt across the horizontal rod will also be controlled by a motor inside the chassis.

As Figure 4-4 shows, the maximum tilt that this design can offer for re-orientation is around 14 degrees. That number can be slightly increased to match the design requirement by increasing the cutout used in this rudimentary model.

A problem that this design has is that it would be difficult to actuate, specifically about the vertical axis. Fitting a stepper motor in this design would be a challenge as finding one small enough to not interfere with other parts means that the chosen motor would not be able to provide enough torque to drive the rotation. The amount of torque could be reduced by lowering the center of gravity of the chassis to beneath the axes.

4.3.1.1(c) 2-Axis Horizontal Rotation about Support Structure

An alternative to the other 2-axis design is using 2 horizontal axes using two concentric rings (an inner ring connecting to the chassis and an outer ring connecting to the inner ring) and operate much like a gyroscope. This design would avoid the challenge of fitting motors into the system by relying entirely on the force due to gravity to orient the chassis.

This is the leading LOPSIDED design and will be discussed further in Section 4.4.1.

4.3.1.2 Chassis

The chassis design needs to accommodate for the payload integration and deployment system, the POS, and for its own leveling system. Across all designs, the chassis will allot a volume for the POS with clear walls so that the camera modules can see through the chassis. It will also have a mounting block on the top so that the integration and deployment system can attach. The chassis design will mainly vary according to the needs of the leveling system.

4.3.1.2(a) Uniform Body – High internal Volume

This chassis design consists of shelled-out rectangular prism section designated for interfacing with the leveling system and housing the electronics. This design allows for a maximum internal volume.

4.3.1.2(b) Cutaway – Better Leveling Range

The design that was chosen for the leading LOPSIDED design is a type of cutaway geometry. This design prioritizes range of tilt over internal volume. This works by having a narrow section in the body that will allow the leveling system to tilt the chassis more without collisions. This design is optimal for systems that require less internal space.

4.3.2 Alternate Payload Integration and Deployment Designs

4.3.2.1 Payload Deployment Alternative Designs

The payload deployment system enables LOPSIDED to exit the launch vehicle and proceed to the landing process. It is important to ensure that LOPSIDED exits immediately at the 500 ft AGL as it will need to deploy its own parachute with enough time for a safe landing.

4.3.2.1(a) ARRD Integration Design

The Advanced Retention Release Device (ARRD) is being used as the device that, after being activated at a certain altitude, separates LOPSIDED from the main parachute recovery harness. The altitude is determined using a StratoLogger CF altimeter, independent from the launch vehicle recovery system. The StratoLogger will send an electric signal through an e-match to ignite the ARRD black powder. The ARRD has been attached to the top of the payload as seen in Figure 4-5 using the ARRD housing which consist of LOPSIDED's ceiling and an extra wooden platform on which the ARRD will screw onto on the bottom. The ARRD will separate into two pieces due to the ignition of a small black powder charge at a lower altitude of 500 ft where it is certain that the payload has exited the payload bay.

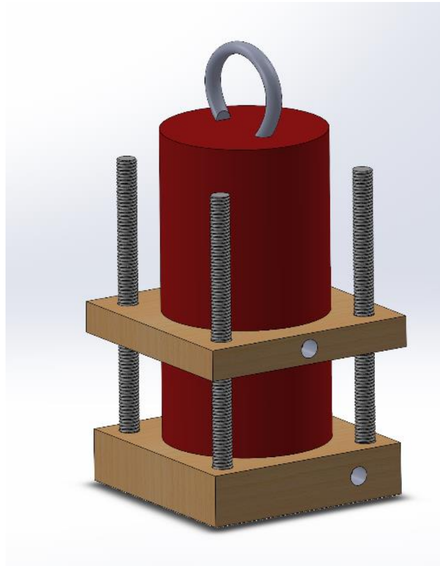

Figure 4-5 ARRD Housing

4.3.2.1(b) Payload Bay Disengage Design

The payload bay disengage design has the payload bay being able to open using hinges and let go of LOPSIDED at a given altitude. The payload bay is held closed by a long and strong enough pin-locking mechanism where it would disengage once the main parachute is released. The disengaging mechanism opens the payload bay as part of the payload bay is sectioned and held together with hinges. The doors open and the payload then has more space to exit the payload bay as it is pulled by its own parachute. Once the payload bay opens, LOPSIDED would be unlatched from the payload pay bulkhead. The problem with this design is that it influences the structural integrity of the main vehicle as the payload bay must be sectioned and latches need to be attached to the doors.

4.3.2.2 Payload Retention Alternative Designs

The payload retention system is important as it serves to hold LOPSIDED in place throughout the main vehicle launch up to the deployment of LOPSIDED. The retention system is designed with the idea of keeping LOPSIDED from considerable movement as this could increase the probability of impacts and possible damage of the electronics and POS carried inside LOPSIDED. There are three modes of movement which are to be minimized throughout the launch: radial, axial, and rotational. Radial movement is described as the movement of LOPSIDED along the radius from the center of the main vehicle towards the walls of the payload bay. Axial movement is the movement of LOPSIDED along the length of the main vehicle, up or down the payload bay. Rotational movement is the rotation of LOPSIDED where the center of rotation is the axis along the main vehicle's length.

4.3.2.2(a) Electronic Retention Device

The electronic design for retention consists of a rotary latch tied to the bottom of LOPSIDED using a chord to pull on LOPSIDED from above during ascent. The latch is attached to the bulkhead using L-brackets. As shown in Figure 4-6, an altimeter is required behind the payload bulkhead in order to actuate the latch at a desired altitude. The latch prevents LOPSIDED from moving down the rocket in the axial direction. In addition, there are 4 shear pins that will break at the moment the main parachute pulls on the payload with a force of about 250 lbf. Once the shear pins break and the latch had actuated to let go of LOPSIDED at apogee, the main parachute will pull on LOPSIDED to bring it out. The main parachute shock cord will be tied to the ARRD in addition to the payload bay bulkhead. LOPSIDED will exit aft of payload bay pulled by main parachute attached to the ARRD. This design offers an easier way to separate the payload bay from midsection where the main parachute is. In addition, it maximizes the space inside the payload bay so LOPSIDED can be larger in diameter or cross-section. However, the electronics may fail; therefore, the team shall ensure that the electronics work as intended through testing before launch.

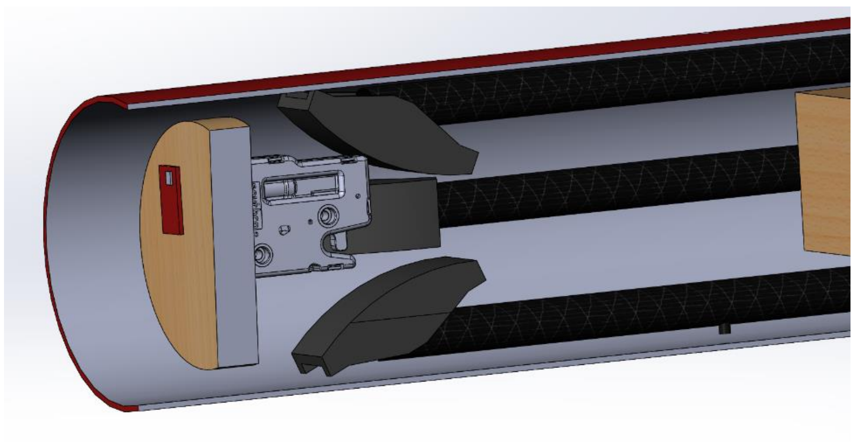

Figure 4-6 Electronic Retention Design using Altimeter and Electronic Rotary Latch

4.3.2.2(b) Mechanical Retention Device

The mechanical retention design consists of attaching rollers to LOPSIDED's body so that they serve as the contact points between LOPSIDED and the payload bay walls. The rollers then roll on the aluminum rails. The rails then keep the rollers on track limiting the rollers to only move up and down the payload bay (axial movement) and preventing LOPSIDED from rotating inside the payload bay. There is a rail lock that holds LOPSIDED in place during ascent as they prevent LOPSIDED to move towards fin can. This retention design also requires LOPSIDED to be pulled by ARRD but with this design, the ARRD is tied to the nosecone instead of main parachute as it was for electronic design. This mechanical design has no need to rely on electronics other than the avionics to separate the nosecone and the ARRD. However, this retention design adds another level of complexity as the nosecone needs to be separated from the body. In addition, the mechanical design reduces the space inside the payload bay due to the rollers and rails which means the payload itself needs to be smaller.

As shown in Table 4-2, the electronic retention device outscored the mechanical retention device due to the design being lighter in weight in addition to taking no space from LOPSIDED inside the payload bay. Although the mechanical design is more reliable as there is no dependance on electronics and possible electrical failures, the implementation of this design would require LOPSIDED to exit towards the nosecone. Having LOPSIDED exit forward of the rocket would require an additional level of complexity as the nosecone and payload bay needed to be sectioned with an additional avionics system. Therefore, the electronic retention design was selected for the payload integration leading design.

Table 4-2 Payload Retention Design Criteria

Criteria	Criteria Weight	Mechanical	Electronic
Space Occupancy	0.5	2	5
Weight	0.15	2	3
Reliability	0.2	4	3
Complexity	0.15	3	4
Total	1.00	2.55	4.15

4.3.2.3 Post-landing Parachute Release Alternative Designs

The team has decided to attach a parachute to LOPSIDED for safe landing. The parachute poses more challenges such as covering POS and dragging LOPSIDED if wind were to inflate the parachute after LOPSIDED had landed. Due to these complications, LOPSIDED requires a way to release the parachute upon landing. As shown in Figure 4-7 where the payload is pulled out by the main parachute in step one and then LOPSIDED deploys its own parachute in step two. Once LOPSIDED lands, the parachute is required to detach from LOPSIDED. There are two main devices being considered to detach the parachute post-landing: electronic rotary latch or electromechanical lock.

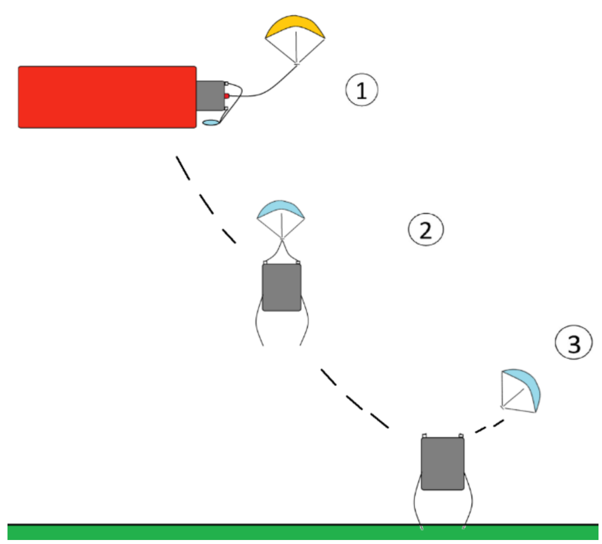

Figure 4-7 Post-Landing Parachute Release Process

4.3.2.3(a) Electromechanical Lock

The first option for the post-landing parachute release system is the electromechanical lock. The selected device is the 3510LM Cabinet Lock from the lock company, dormakaba. This device is capable of withstanding 250lbf with its compact design. The dimensions of the lock are 1.65inx1.89inx0.81in. This device requires a 12V power supply for the drawing current of 250mA. This device weighs 0.22lbs each. The electromechanical lock was selected for the leading design for its compact size, light weight, and holding force. The plan for implementation is to have the parachute attach to the upper (smaller) piece while the lower actuator piece is attached to the ARRD housing. The lock will be actuated electronically using an accelerometer to send a signal as soon as LOPSIDED lands.

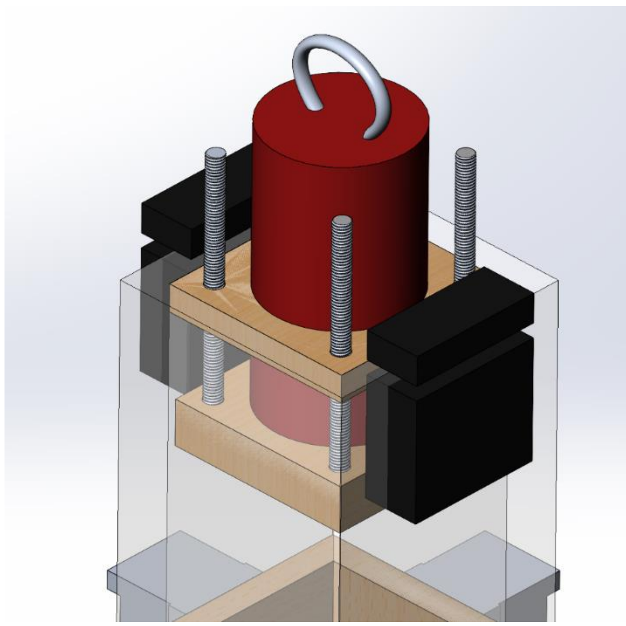

Figure 4-8 Electromechanical Lock Attached to ARRD Housing

4.3.2.3(b) Electronic Rotary Latch

The second alternative to a post-landing parachute release system implements two electronic rotary latches. Figure 4-9 shows how the electronic rotary latches are attached to the ARRD housing, where the pull from the parachute translates to the metal screws for extra support. The rotary latch selected as an option is the R4-EM 5 & 7 Series Electronic Rotary Latch from Southco. This R4-EM series can hold a large holding force of 993lbf which is more than enough for the 200lbf estimate maximum load now the LOPSIDED's parachute opens. The rotary latch has dimensions of 2.74inx2.74inx0.83in. The rotary latch requires a power supply of 12V. The device current draw is 600mA per latch. The electronic rotary latch size is slightly larger compared to the electromechanical lock; in addition, the latch is heavier than the electromechanical lock by 0.403lbs. The weight is an important characteristic for the parachute release device since the devices are attached at the top of LOPSIDED and can influence the center of gravity useful for the gravity-induced leveling.

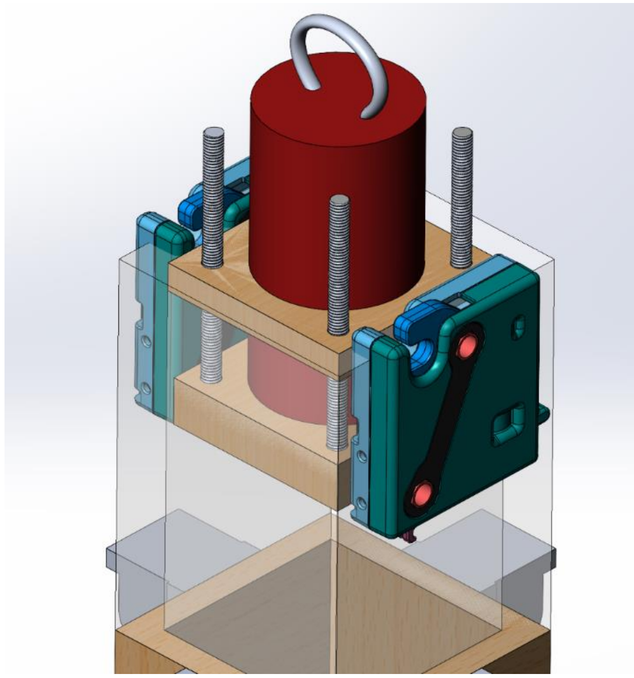

Figure 4-9 Electronic Rotary Latch Attached to ARRD Housing

As shown in Table 4-3, the rotary latch was outscored by the electromechanical lock for the given criteria of weight, maximum loading, and size. One of the main factors for the rotary lock selection is the weight of the device. For LOPSIDED leveling purposes, the parachute release device is required to be as light as possible. However, the rotary latch has a greater holding force than the electromechanical lock by a difference of about 700lbf.

Table 4-3 Parachute Release System Design Criteria

Criteria	Criteria Weight	Electronic Latch	Electromechanical Lock
Weight	0.6	2	4
Max Loading	0.2	5	2
Size	0.2	3	4
Total	1.00	2.8	3.6

4.3.3 Alternate POS Designs

The POS will be used to capture and transmit 360-degree images of the payload landing site to fulfill NASA requirements 4.2 and 4.3.4.1-4.3.4.4. The major components of POS design consist of the imaging components responsible for capturing the 360-degree photograph, a transmitter for sending the photograph, and the on-board computer for controlling these components.

4.3.3.1(a) Single 360-Degree Camera

One possible method of image capture is the use of a single camera specifically designed to capture 360-degree images. There are multiple 360-degree camera models available for purchase from a variety of reliable manufacturers. Select models are specifically designed for use in high-intensity activities and extreme environmental conditions. These designs would provide the robustness necessary for the imaging components of the POS to survive launch, and the deployment and landing of LOPSIDED. These cameras are also contained within individual housings – only a single unit would need to be installed, which would make integration into the POS straightforward.

Higher-quality units are expensive compared to the cost of other POS components, but there are 360-degree cameras available for less than 100 dollars with positive reviews from multiple consumers. However, most 360-degree camera modules are designed to interface with manufacturer-specific software, usually in the form of a smartphone or computer application. These apps host a wide variety of imaging and editing features but would not provide the flexibility necessary to fulfill mission requirements. These cameras are not designed to easily interface with a single-board computer (SBC) such as a Raspberry Pi, which would make software integration difficult.

4.3.3.1(b) Single Actuated Raspberry Pi Camera Module

The decision was made early on to use a Raspberry Pi for operating the POS. Multiple companies manufacture camera modules specifically for use with Raspberry Pi boards and utilizing these cameras would greatly increase chances of mission success. By actuating a single camera module about its vertical axis, multiple images can be captured and spliced together to form a single 360-degree image. This solution would be very inexpensive – standard Raspberry Pi camera modules sell for 30 dollars from most retailers, and the team has a variety of servo motors in stock for these types of applications.

The actuation of the camera, while not inherently complicated, adds a level of unnecessary complexity to the design. A central, free-spinning camera in the upper section of the payload would be more difficult to integrate with LOPSIDED's self-leveling system. If the motor were to fail, the POS would only be able to capture a fraction of an image.

4.3.3.1(c) Two 180-Degree Raspberry Pi Camera Modules

Multiple stationary camera modules are capable of capturing a 360-degree image, given a sufficiently wide horizontal field of view (HFOV). There are Raspberry Pi camera modules, such as the Arducam Fisheye Camera, with lenses capable of capturing a HFOV over 180 degrees. Specifically, the Arducam Fisheye Cameras have a HFOV of 194 degrees. With these cameras, a design was proposed consisting of two modules mounted 180 degrees from each other, within a transparent section of LOPSIDED. Since the two cameras cover over 360 degrees between them, this design is the simplest way to achieve a full panoramic photograph. The leading POS design has been modified slightly to include four camera modules instead of two. This design is detailed more in Section 4.4.3.

4.4 Leading Payload Design

4.4.1 Leading LOPSIDED Design

The leading LOPSIDED design consists of a 2-axis (both horizontal) gyroscopic leveling system driven by the force due to gravity. In order to record an initial orientation measurement upon landing, solenoid latches will be used to lock the axes from rotating. Two solenoid latches for each axis (four total) will be used for redundancy in maintaining the LOPSIDED's neutral position.

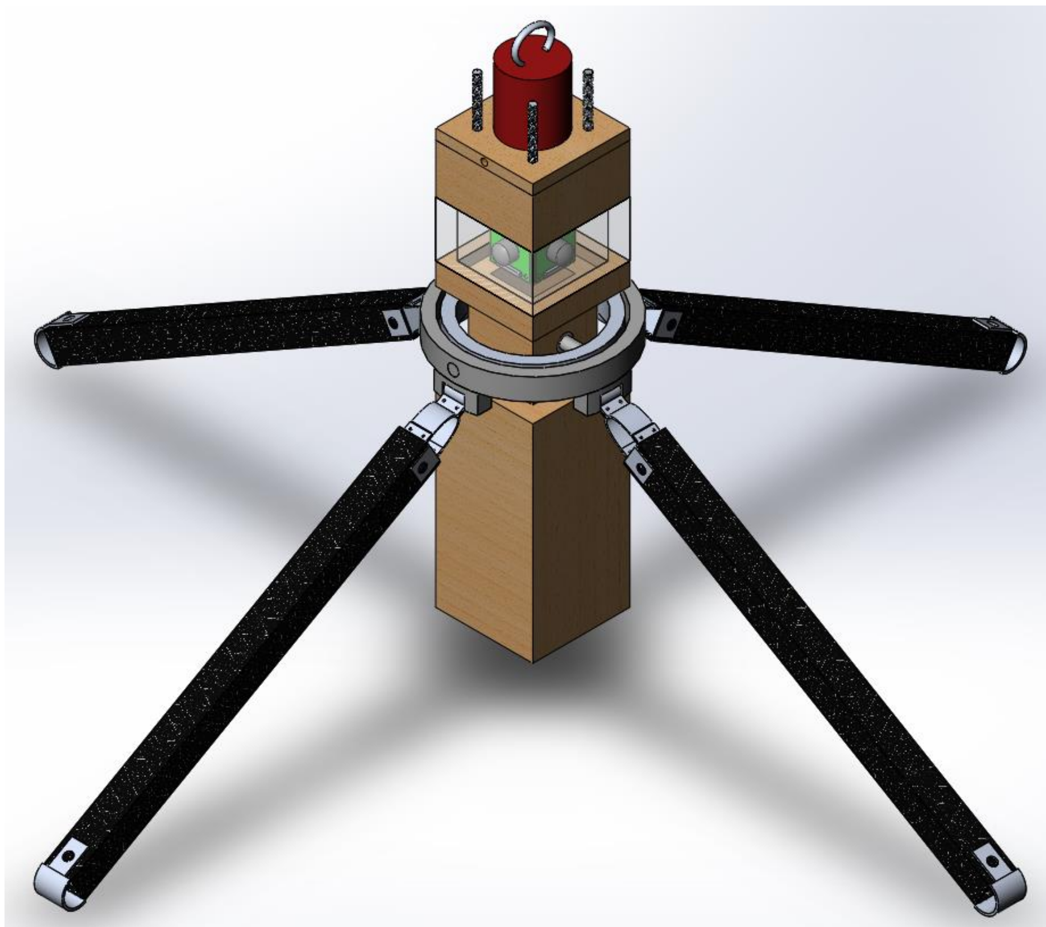

Figure 4-10 LOPSIDED in Deployed Configuration

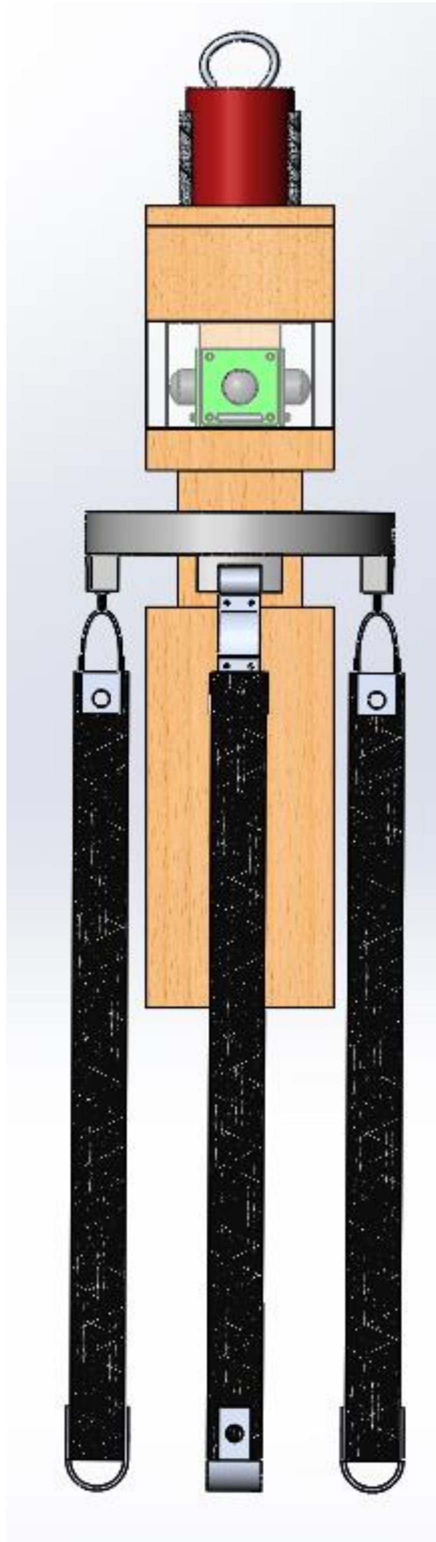

Figure 4-11 LOPSIDED in Stowed Configuration

The chassis will feature a cutout for the leveling system's rings to tilt further before occluding with the chassis and therefore extending the range of tilt to the required 15 degrees off the neutral axis. The chassis will also feature a clear acrylic section so that the POS will have an unobstructed view of the environment surrounding the payload. Four bolt locations are designated at the top of the payload for the payload integration and deployment system to interface with. The rest of the chassis will consist mainly of laser-cut plywood as its structure.

The leading design features an estimated minimum of an 18.5° tilt and a maximum of a 22° tilt shown in Figure 4-12. This allows for a landing site grade of 23.5-27° grade depending on the landing orientation.

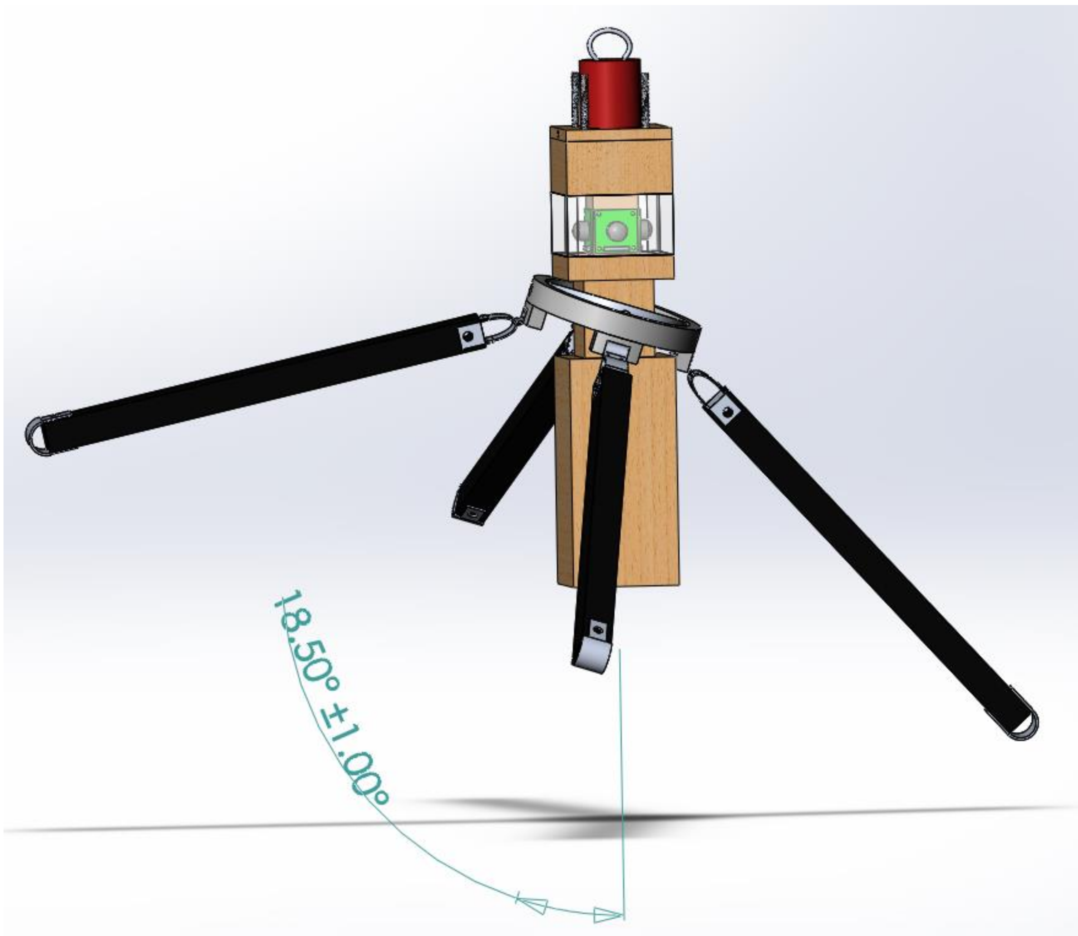

Figure 4-12 LOPSIDED Maximum Tilt Angle

The legs will be 1”x1” shelled-out carbon fiber rods. The feet and the brackets that attach the legs to the outer leveling ring will be constructed from bending sheets of aluminum and joined by rivets. At the connection between the legs and the outer leveling ring there will be an axis driven by a torsional spring. The spring will attempt the flay the legs out until they are stopped. Inside the payload bay, the legs will be stopped by the walls of the payload bay. Once deployed, the legs will rotate out until stopped by a set screw determining the leg’s angle with respect to the outer leveling ring.

Figure 4-13 LOPSIDED Levelling and Support System

The inner and outer rings will be manufactured from 0.787” thick sheets of aluminum and will be joined by 0.394” aluminum rods and press-fitted ball bearings. The solenoids locking these in place will be located on the inner ring over the inner rod interfacing with the outer ring as well as inside the chassis interfacing with the inner rod. Two will be placed symmetrically for each axis of rotation for redundancy

The chassis will separate into three sections: the upper section housing the POS’s cameras and the integration equipment, the middle section consisting of the cutaway section that the leveling system’s inner rod will pass through, and a lower section that will be devoted to any electronics that can be fitted there.

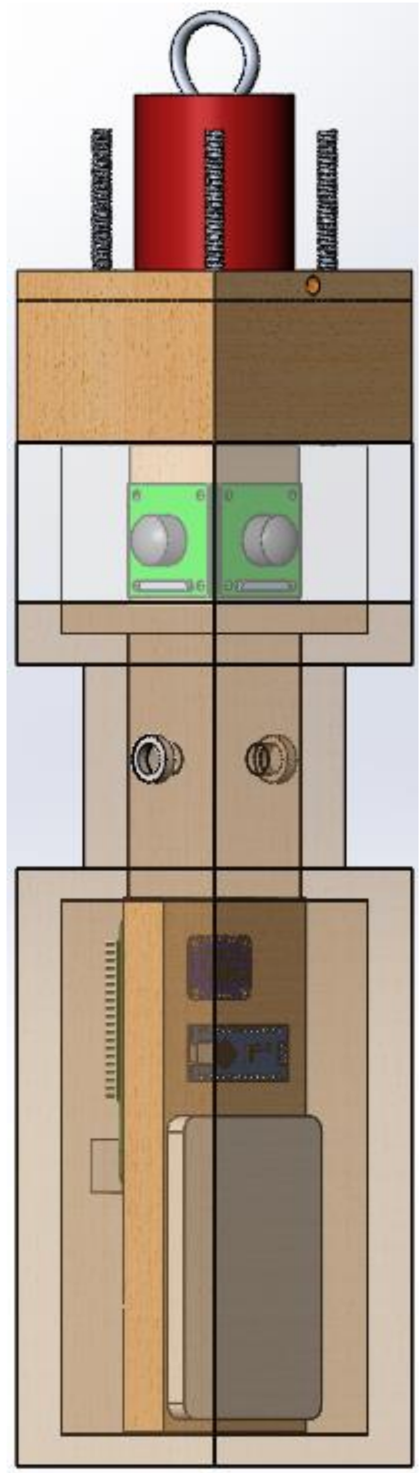

Figure 4-14 LOPSIDED Internal View

The lower section of the chassis is dedicated to electronics as well as a ballast to lower the chassis' center of gravity. A sled will fit into this lower section so that the team can easily access the electronics in the small and confined lower section of the chassis. The sled and ballast will be positioned so that the CG of the chassis and its components will be centered. This will make it so that the leveling due to gravity will result in a perfectly level system given that it is within the leveling system's range of rotation.

Aside from the ballast in the lower electronics chassis section, the leveling system will require a Programmable Logic Controller (PLC), the system will use an Arduino Nano programmed in some variant of C, a 9-axis accelerometer (separate from the one used in the POS for redundancy), a battery to power the system, and wires that connect to the solenoid latches.

LOPSIDED will use the 9-axis accelerometer to determine when it lands. Upon landing, an initial orientation measurement will be recorded. The solenoid latches will then be released allowing the leveling system dynamics to operate. After the chassis stabilizes, an additional and final orientation measurement will be recorded. All measurements and signals will be recorded from the 9-axis accelerometer.

4.4.2 Leading Payload Integration and Deployment Design

The leading payload integration design includes the electronic retention design, the ARRD deployment design, and the electromechanical lock for the post-landing parachute release design. The electronic retention design includes the electronic rotary latch signaled to actuate by a StratoLogger CF altimeter, which is placed on the back of the payload bay bulkhead, at apogee. This retention system requires LOPSIDED to be pointing towards the back of the rocket as the latch attaches to the bottom of LOPSIDED. At 675 ft, the main parachute will deploy and pull on the top of LOPSIDED at the ARRD. LOPSIDED will exit the payload bay while being attached to the main parachute shock chord. At 500 ft, LOPSIDED will separate from the shock chord using the ARRD signaled by a second altimeter placed inside LOPSIDED. A Jolly Logic chute release will deploy LOPSIDED's parachute at 200 ft. Once LOPSIDED lands, an accelerometer, the Adafruit BNO05, will signal the electromechanical lock to disengage and release LOPSIDED's parachute. Table 4-4 shows the total weight of the electronics included inside the payload bay as part of the integration system.

Table 4-4 Payload Bay Integration Electronics Weight

Payload Bay	Weight (lbs)
Altimeter (StratoLogger CF)	0.024
Electronic Rotary Latch	0.623
9V Battery	0.081
Total	0.729

As part of integration, LOPSIDED will carry four main electronic components for deployment, the ARRD, the Jolly Logic chute release, the GPS tracker, and the parachute electronic locks. A selection for the power source of this design is still pending where some ideas include using separate 9V for the electronic components, shared 9V batteries, or a 12V battery for most or all the electronic components inside LOPSIDED. A compact battery with enough capacity can be put together using a battery pack with NiCd Rechargeable Cell 1.2V batteries with 700mAh or NiMH Rechargeable cell 1.2V 2200mAh.

These battery packs can weigh around 1.12lbs and provide enough voltage for the electromechanical lock in addition to the other electronic components if implementing the use of voltage regulators to transform 12V to 9V or another lower voltage required. A battery pack dimension can be 3inx1.2inx2in (LxWxH) for the NiCd battery pack option, for example. Another option is to use a LiPo battery with the right voltage and capacity for all the electronics. A LiPo battery can weigh around 1lbs; however, the size is greater than the battery pack making it possible for the LiPo battery to not fit inside LOPSIDED. The StratoLogger CF altimeter was selected as the team already owns multiple of this type. Similarly, the accelerometer Adafruit BNO05 was already owned by the previous year team and it is proven to work for payload location. Table 4-5 shows the total weight of the electronic components mentioned.

Table 4-5 LOPSIDED Integration Electronics Weight

Payload	Weight (lbs)
Altimeter (StratoLogger CF)	0.02
Tracker (BigRedBee 900)	0.06
Electromechanical Lock (x2)	0.44
12V Battery	1.12
ARRD	0.34
Accelerometer (Adafruit BNO05)	0.06
Total	2.04

4.4.3 Leading POS Design

The leading POS design consists of four Arducam Fisheye Raspberry Pi camera modules. The camera modules are arranged coaxially about the vertical axis of LOPSIDED and are housed within a clear section of LOPSIDED above the pivoting axis of the self-leveling system. The cameras will be attached to the four outward faces of a wooden mounting block, which will be attached to the top surface of LOPSIDED via mounting brackets.

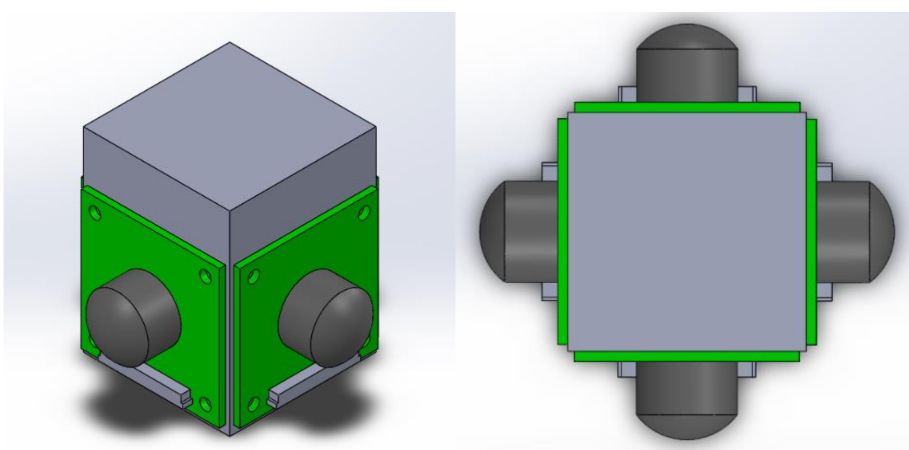

Figure 4-15 Isometric and Top View of Camera Mounting Block

The cameras will be connected to a Raspberry Pi model 3B+. Standard Raspberry Pi boards only have one Camera Serial Interface (CSI) port, so an Arducam Multi-Camera Adapter

Board will be used. This board houses four additional CSI ports, and attaches to the Pi's built-in CSI port via a short ribbon cable. A 9-axis accelerometer and a 433MHz radio transceiver will also be connected via the Pi's GPIO pins. The accelerometer will be used to detect the landing and self-leveling of LOPSIDED. Once leveling has been detected, the image capture sequence will be initiated. The adapter board can only activate one camera at a time. Because of this, four images – one from each camera – will be taken individually, in rapid succession. This will be executed using a Python script installed on the Raspberry Pi. Once the images have been captured, another Python script will be used to splice the necessary images into two separate panoramic images. These images will then be transmitted to the team's ground station via a 433 MHz transmitter.

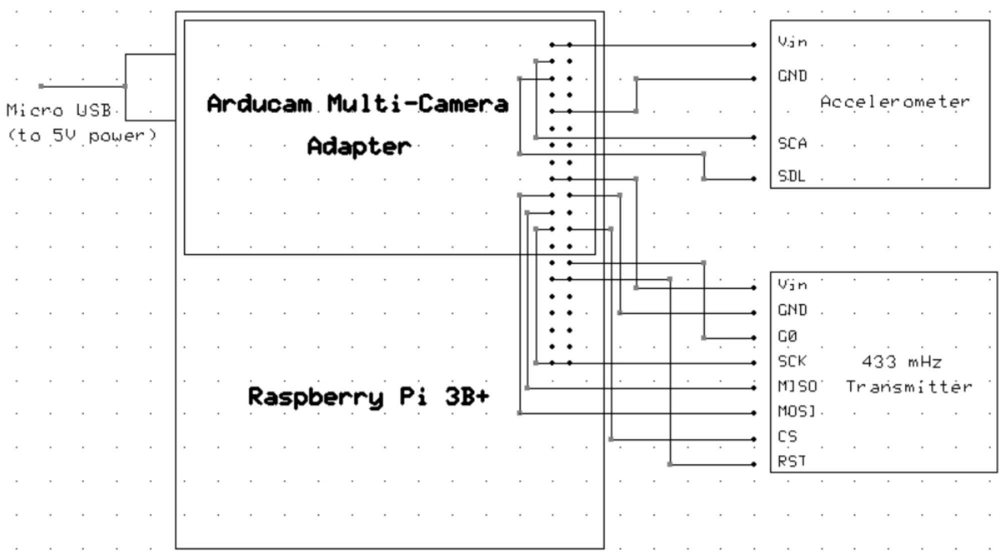

Figure 4-16 Basic POS Electronics Schematic

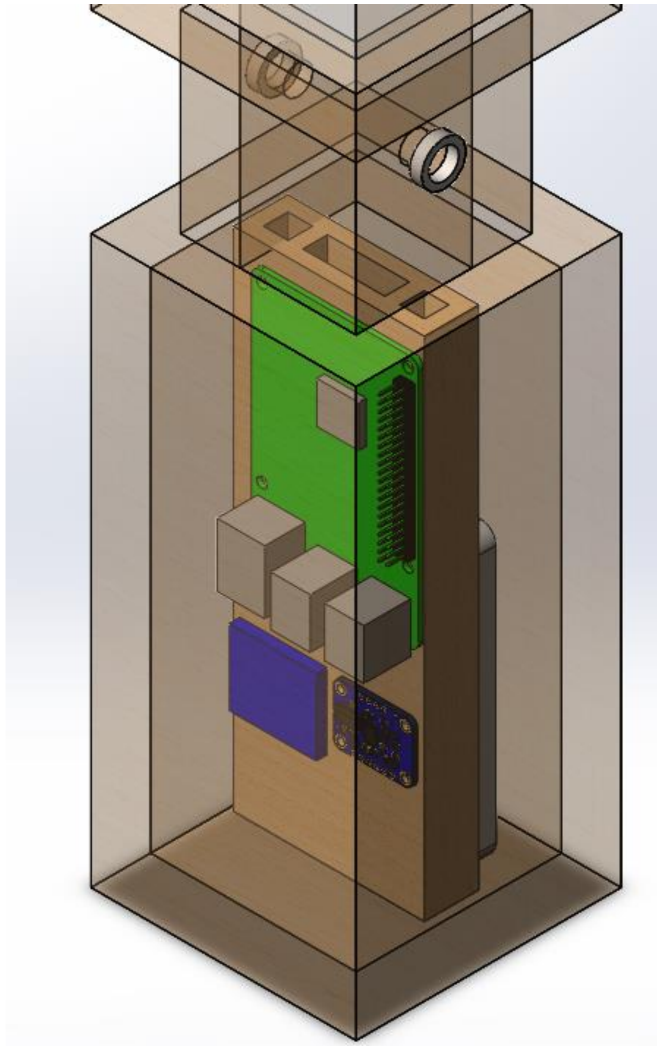

Figure 4-17 POS Electronics Within LOPSIDED

Excluding the camera modules, all POS electronics are housed within the lower section of LOPSIDED, located below the axis of the self-leveling system. This space will also include the electronics necessary for LOPSIDED's self-leveling system, detailed in Section 4.4.1. This helps to lower LOPSIDED's CG and by extension, improve the performance of the self-leveling system. The cameras are connected to the Raspberry Pi via 24" flexible ribbon cables and will be routed through the interior of the payload. The transmitter will be positioned low within the payload body to ensure there is room for the attached antenna.

The POS components and their weights are given in Table 4-6.

Table 4-6 POS Component Weights

Component	Quantity	Total Mass (lb)
Arducam Fisheye Raspberry Pi Camera	4	0.088
24" Ribbon Cable	4	0.079

Raspberry Pi 3B+	1	0.110
Arducam Multi-Camera Adapter Board	1	0.040
Koulomb 5V, 2500 mAh battery	1	0.130
Adafruit RFM69HCW Transceiver Radio Bonnet - 433 MHz	1	0.024
Adafruit Triple-Axis Accelerometer	1	0.003
Additional mounting hardware	~	0.20
Total weight		**0.67**

The POS will utilize Slow-Scan Television (SSTV) as the method of image transmission. This is a communication method common in amateur HAM radio, which allows static images to be transmitted via radio waves over long distances. This is achieved by converting images into WAV audio files and transmitting this audio data over HF, VHF, or UHF radio frequencies. On the receiving end, a personal computer equipped with SSTV software, such as MMSSTV, can be used to demodulate the incoming signals. With SSTV software, a PC's sound card can be used as a modem to achieve this. The resolution of the received images will be reduced, but the required power for this transmission method is low, and the range is very high.

Some components of the POS will be tested within the subscale launch vehicle. This includes a Raspberry Pi 3B+, an Arducam Fisheye camera, an accelerometer, and a 5V battery. The camera will be mounted to the inner surface of the payload bay with a 3D printed housing, and will take a series of images during and after flight. These images will provide data on the effectiveness of this camera model after enduring flight conditions.

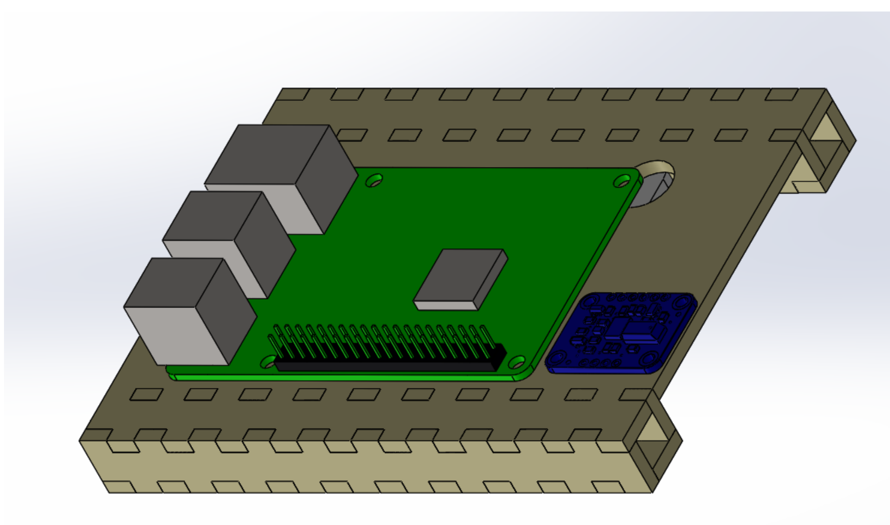

Figure 4-18 Subscale Payload Sled

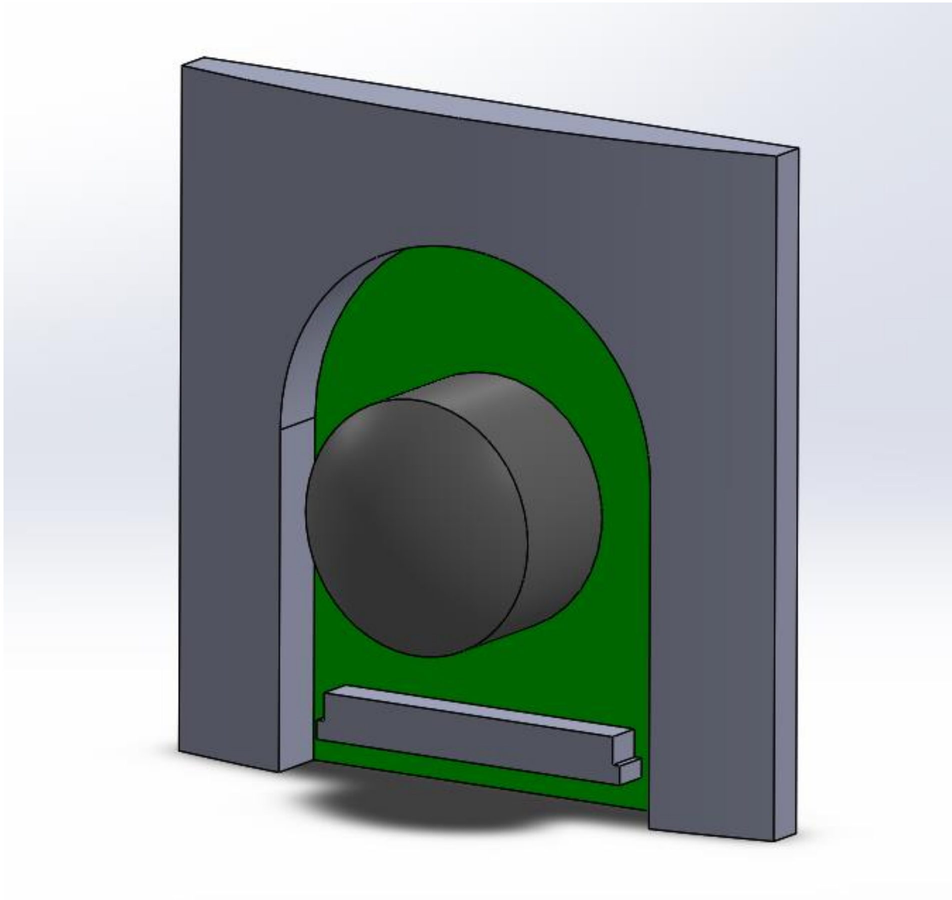

Figure 4-19 Subscale Payload Camera Mount

5. Safety

5.1 Safety Officer

The team safety officer for the 2021 Student Launch competition is Frances McBride. As safety officer, Frances is responsible for ensuring an overall level of safety within the project. This includes collaborating with design teams to ensure safety of all subsystems and develop hazard mitigations, being present for all fabrication activities, being present for launch day activities, and fostering a safety culture within the team. Frances is also responsible for meeting all requirements detailed in handbook requirement NASA 5.3.

5.2 Hazard Classification

In order to better classify risks and hazards associated with the project, the team has developed a likelihood-severity (LS) method of hazard classification. This classification system is detailed in Table 5-1 below.

Table 5-1 Likelihood-Severity (LS) Classifications

		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

These LS classifications are used to determine the importance of mitigations of their associated hazards. Any hazard with a severity of at least 3 or likelihood of at least C must be mitigated. The definitions of each severity level are provided in Table 5-2 below.

Table 5-2 Severity Definitions

Hazard Type	1 (Low Risk)	2 (Medium Risk)	3 (High Risk)	4 (Severe Risk)
Personnel	No personnel injury.	Any personnel injury is treatable with first aid.	Moderate personnel injuries, manageable with launch field first aid.	Severe personnel injuries requiring hospitalization.
Launch Vehicle	Any damage to launch vehicle is reversible.	Any damage to launch vehicle is repairable.	Damage to launch vehicle is repairable, but not to original condition.	Damage to launch vehicle is irreparable.
Mission	Mission success.	Partial mission failure, successful flight.	Partial mission failure, partially successful flight.	Complete mission failure, unsuccessful flight

Each hazard is also given a label for unique identification. These labels follow the format "System.Hazard Category.Number." Systems are defined by one or two-letter codes, shown in Table 5-3 below. Hazard categories are listed as horizontal bars within the FMEA table separating different sections. Their abbreviations are different for each system. The number is used to differentiate hazards with the same system and category.

Table 5-3 Hazard Label System Codes

Code	System Represented
A	Aerodynamics
E	Environmental
P	Payload
Pe	Personnel
R	Recovery
S	Structures

5.3 Personnel Hazard Analysis

The team's personnel hazard analysis is defined in Table 5-4 below.

Table 5-4 Personnel Hazard Analysis

Label	Hazard	Cause	Effect	LS	Mitigation
Hazard to Skin and Soft Tissue					
Pe.S.1	Slips, trips, and falls	Uneven launch field conditions	(1) Ligament sprain (2) Bruising (3) Skin abrasion	3C	(1) Running shall be strictly forbidden on launch day (2) Those recovering launch vehicle are ONLY trained, experienced, pre-assigned personnel (3) Recovery personnel shall be instructed to wear closed toe walking shoes on launch day
Pe.S.2	Contact with large, airborne shrapnel	Catastrophe at takeoff	Scrape Deep scratch	2C	Aerotech motors are chosen for their low likelihood of

					factory defects leading to CATO
		On-ground explosion of motor/black power			(1) Personnel are required by the RSO to maintain a minimum distance away from the launch pad (2) Altimeters shall be armed and igniters shall be attached only when launch vehicle is on launch pad and ready for flight
		Ballistic descent of payload or airframe shrapnel			See payload, recovery, and structures FMEAs for full mitigations
Pe.S.3	Contact with small, airborne particulates	Sanding, cutting, or drilling into brittle/granular materials			(1) Protective eye and face equipment shall be provided to personnel working with power tools (2) Dremel blades are inspected for defects prior to use
Pe.S.4	Exposure to uncured epoxy fluid	Working with liquid epoxy	(1) Skin irritation (2) Rash	2C	Nitrile gloves shall be provided to personnel working with uncured epoxy or volatile organic compounds
Pe.S.5	Exposure to chemical fumes	Working with volatile organic compounds		3A	
Hazard to Bones					

Pe.B.1	Personnel contact with large airborne shrapnel	Pre-flight motor/black powder ignition Catastrophe at takeoff	Bone fracture requiring immediate medical attention	4B	(1) Personnel are required by the RSO to maintain a minimum distance away from the launch pad (2) Aerotech motors are chosen for their low likelihood of factory defects leading to CATO
Pe.B.2	Slips, trips, and falls	Uneven ground conditions	(1) Bone bruise (2) Minor bone fracture (3) Ligament sprain	4C	(1) Running shall be strictly forbidden on launch day (2) Those recovering launch vehicle are ONLY trained, experienced, pre-assigned personnel (3) Recovery personnel shall be instructed to wear closed toe walking shoes on launch day
Hazard to Eyes					
P.E.1	Exposure to fumes	Working with uncured epoxy	Eye irritation	2B	(1) Protective eye equipment shall be provided to personnel working with uncured epoxy (2) In the event of eye irritation, eye wash stations shall be made accessible to all personnel

P.E.2	Eye contact with metallic, wood, or plastic shrapnel	Sanding Material failure while drilling Premature, on-ground black powder or motor ignition	(1) Eye abrasion (2) Blindness	3C	(1) Protective eye equipment shall be provided to personnel working with potentially airborne material (2) Personnel shall be instructed in checklists to stand away from any potential explosives
Hazard to Limbs					
Pe.L.1	Contact with explosive gases Contact with large, airborne shrapnel	On ground motor/black powder ignition	(1) Limb loss (2) Severe injury warranting amputation	4A	(1) Personnel shall be instructed in checklists to stand away from any potential explosives (2) Altimeters shall remain disarmed until launch vehicle is on launch rail, ready for flight
Pe.L.2	Personnel contact with ballistic launch vehicle components	Ballistic descent (R.A.2)	(1) Limb loss (2) Severe injury warranting amputation	4A	(1) See recovery FMEAs for mitigations through design (2) Personnel shall be instructed to remain still until launch vehicle components are confirmed by the RSO to have landed
Pe.L.3	Finger snag in bit of power tool	Wearing gloves while working with power tools	Finger loss	4B	(1) Gloves are located at the far end of the lab from desktop power tools and in a separate cabinet from

					hand-held power tools (2) Safety presentations are conducted that detail proper procedure prior to use of a power tool
Hazard to Respiratory System					
Pe.R.1	Exposure to fumes	Working with uncured epoxy	(1) Lung irritation (2) Difficulty breathing	2D	(1) Particulate masks shall be provided to personnel working with epoxy, paint, and chemicals from the flame cabinet (2) In cases where epoxy cannot be applied outside, an oxygen monitor is in use to prevent irritation/difficulty breathing (3) No facilities shall exist indoors without ventilation; thus, all painting occurs outdoors
		Working with uncovered paint			
		Off-gassed chemicals stored in flame cabinet			
Pe.R.2	Exposure to particulates	Sanding, drilling, and/or cutting brittle or granular materials		3D	Particulate masks shall be provided to personnel working with power tools
Hazard to Head					

Pe.H.1	Impact with ballistic launch vehicle sections	Shock cord breakage or disconnection	(1) Concussion (2) Memory loss (3) Brain injury (4) Skull fracture	4B	(1) Kevlar shock cord shall be used to withstand, at least, the maximum expected load (3660 lbf) (2) Any bulkhead-shock cord systems shall have a factor of safety of at least 1.5
		No/late parachute deployment			(1) Adequately sized pressure ports are drilled during construction (2) 4-40 shear pins shall be used
	Impact with ballistic payload	Premature payload parachute ejection		4A	(1) Payload and recovery altimeters shall be tested before flight (2) Personnel shall maintain a minimum distance away from the launch site as maintained by the RSO
		Premature payload ejection from launch vehicle			
	Impact with large, airborne shrapnel (payload or launch vehicle components)	Catastrophe at takeoff		4A	Aerotech motors are chosen for their low likelihood of factory defects leading to CATO
		Premature on-ground motor/black powder ignition			(1) Personnel shall be instructed in checklists to stand away from any potential explosives

					(2) Altimeters shall remain disarmed until launch vehicle is on launch rail, ready for flight
		Shock cord shear; shock cord disconnection from bulkhead			(1) Kevlar shock cord shall be used to withstand, at least, the maximum expected load (3660 lbf) (2) Any bulkhead-shock cord systems shall have a factor of safety of at least 1.5

5.4 Failure Modes and Effects Analysis (FMEA)

The following tables detail the potential failure modes of each project subsystem, their effects, their severity, and the steps taken to mitigate these hazards.

Table 5-5 Structures FMEA

Label	Hazard	Cause	Effect	LS	Mitigation
Fin Hazard					
S.F.1	Fin structural damage (cracks/shears)	Freestream velocity approaching transonic values; fin flutter	(1) Flight path diverted (2) Failure to reach target apogee (3) Launch vehicle enters nose-over-over-tail spin	4B	Flight velocity simulations have been performed in RockSim – no part of the launch vehicle nears transonic speeds
S.F.2	Fin delamination from airframe	Insufficient time for a complete fillet cure Gaps in fillet epoxy	(1) Flight path diverted (2) Target apogee not reached	4A	Epoxy fillets are given at least 24 hours to fully cure before flight

			(3) Launch vehicle enters nose-over-tail spin		
Airframe Hazard					
S.A.1	Airframe cracking/rupture	Fin loss; launch vehicle enters into nose-over-tail spin	(1) Premature black powder detonation due to rapid pressure change (2) Loss of inner components	4A	See S.F.1 and S.F.2
		Loose inner payload components during flight			An electronic latch secures the payload in place prior to ejection
		Excessive internal stresses			(1) G12 Fiberglass, a strong and durable material, is chosen for the fullscale airframe (2) Prior to launch, ejection tests shall be performed to confirm that the minimum necessary black powder amount is used
S.A.2	Shear pin failure to shear	Insufficient black powder charge	Ballistic launch vehicle descent	4B	(1) Black powder mass is calculated using the ideal gas law (2) Ejection tests shall be performed prior to launch to confirm appropriate black powder mass
		Excessive shearpin diameter			(1) 4-40 shear pins are used (2) Shear pin stress tests shall be performed prior to

					launching the launch vehicle
S.A.3	Premature shearpin shear	Premature black powder detonation	(1) Potential airframe exposure to burning motor (2) Ballistic payload descent (3) Failure to reach target apogee	3B	Pressure ports are drilled into airframe
		Insufficient shearpin diameter			(1) 4-40 shear pins are used (2) Shear pin stress tests shall be performed prior to launching the launch vehicle
S.A.4	Airframe exposure to burning motor	Premature section separation	(1) Inability of airframe to withstand flight forces; complete disintegration (2) Nosecone plastic melting	4A	See S.A.3 Mitigations
S.A.5	LV section collision	Excessive shock cord length	Airframe cracking/rupture	2B	Shock cord length for drogue is 3.8 ft; shock cord length for main is 7.2 ft
		Insufficient shock cord length			
S.A.6	Elevated pressure inside LV	Insufficient pressure ports	(1) Airframe rupture (2) Inability of black powder to detonate	4B	Pressure ports are drilled into airframe
Hazard to Bulkheads					
S.B.1	Bulkhead delamination	Gaps in epoxy layers	(1) Inner LV component structural damage	2A	(1) Bulkheads shall be designed with a factor of safety of at least 1.5

		Excessive forces from bolts	(2) Airframe cracking/shear (3) Motor separation from airframe		(2) Bulkhead stress/delamination tests shall be performed
S.B.2	Bulkhead cracking/ripping	Excessive force from bolts	(1) Airframe cracking/shear (2) Parachute separation from LV	2A	U-bolts shall be chosen over eyebolts for their superior load distribution
		Insufficient bulkhead thickness			Bulkheads and their U-bolts shall have a FOS greater than or equal to 1.5

Table 5-6 Aerodynamics/Propulsion FMEA

Label	Hazard	Cause	Effect	LS	Mitigation
Stability Hazard					
A.S.1	Launch vehicle weather cock into wind gust	Overstability (Stability margin ≥ 2.5)	Diverted flight path; target apogee not reached	2C	(1) CP, CG, and stability margin calculated through RockSim prior to flight (2) CG, weight, and stability margin observed manually directly prior to flight
A.S.2	Launch vehicle diversion away from wind gust	Understability (Stability margin ≤ 2.0)		3C	
A.S.3	Fin flutter	Transonic freestream velocity around fins	(1) Fin structural damage (2) Loss of fins (3) Launch vehicle enters nose-over-over-tail spin	4A	Flight velocity simulations have been performed in RockSim – no part of the launch vehicle nears transonic speeds
Motor Hazard					

A.M.1	Motor retention ring ejection	Structural failure of retention ring	Catastrophe at Takeoff (CATO)	4A	Aerotech motor casings and retention rings are chosen for their strength and durability
A.M.2	Uneven pressure buildup inside of motor	Gap/bubble in motor propellant grain Clogged nozzle Holes/cracks in motor casing		4B	Aerotech motors are chosen for their low likelihood of factory defects leading to CATO and motor failures
A.M.3	Rapid change in stability margin	Difference between in-flight and theoretical thrust curve of L1520T	Diverted flight path; target apogee not reached	2B	
A.M.4	Absence of igniter ignition	High humidity	Absence of motor ignition; inability to fly	4C	The team shall not launch in high humidity
		Faulty igniters			Extra igniters are supplied by the team on launch day

Table 5-7 Recovery FMEA

Label	Hazard	Cause	Effect	LS	Mitigation
Hazard to/from Avionics					
R.A.1	Avionics exposure to ejection gases	Gap between AV bulkheads and body tube	(1) Electronics damage/ destruction (2) Failure of altimeter to detonate second charge; LV lands with excessive kinetic energy	3A	Bulkhead-airframe contacts filleted on both sides; bulkheads confirmed to be flush with airframe
R.A.2	Dual deploy with main at apogee	Crossed main and drogue signal wires	Large wind drift; inability to recover rocket	4C	AV Bay wiring will be confirmed by at least four team
R.A.3	No parachute deployment		(1) Ballistic descent	4B	

			(2) Personnel injury		members prior to final assembly
Hazard to Parachutes/Shock Cord					
R.P.1	Shock cord disconnection	U-bolt shear	(1) Ballistic descent of at least one launch vehicle section (2) Excessive kinetic energy upon landing	4B	Bulkheads and their connecting pieces shall be designed with a factor of safety of 1.5 or greater
		U-bolt disconnection			
		Bulkhead wood crack, shear		4B	
		Bulkhead delamination			
		Shock cord rip		4A	Kevlar shock cord is chosen to withstand the maximum expected force on shock cord, 3660 lbf
R.P.2	Parachute rip/hole/tear	Contact with explosive black powder	Partial parachute deployment	2D	Fireproof Nomex cloth is wrapped around parachute to insulate it from ejection gases
R.P.3	Partial parachute deployment	Parachute rip/hole/tear	(1) Excessive kinetic energy upon landing (2) Personnel injury	4B	
		Recovery harness + parachute entanglement			
		Impact of body section with parachute			Shock cord length for drogue is 3.8 ft; shock cord length for main is 7.2 ft
R.P.4	Late parachute deployment	Delayed e-match burning	Structural damage due to excessive force on shock cords	3B	Strips of flammable paper towels are placed on top of black powder to aid in combustion
		Delayed black powder detonation			
R.P.5	Premature parachute deployment	Incorrect altimeter pressure readings	(1) Parachute deployment during ascent (2) Recovery harness/airframe	4A	Pressure ports are drilled in airframe to allow for ambient pressure

			structural damage		changes to be detected
		Excessive flight forces			(1) 4-40 Shear pins are used to secure body sections (2) Shear pin stress tests shall be performed prior to launch
Hazard to/from Black Powder					
R.BP.1	Lack of shear pin breakage	Shear pins of excessive diameter/strength	(1) Ballistic descent; excessive kinetic energy upon landing (2) Personnel injury	4C	(1) 4-40 Shear pins are used to secure body sections (2) Shear pin stress tests shall be performed prior to launch
		Insufficient black powder charges			(1) Black powder mass is calculated using the ideal gas law (2) Ejection tests shall be performed prior to launch to confirm appropriate black powder mass
R.BP.2	Premature section separation	Premature shear pin break during ascent	(1) Body tube zippering from shock cord (2) Diverted flight path resulting in low final apogee	3B	(1) 4-40 Shear pins are used to secure body sections (2) Shear pin stress tests shall be performed prior to launch

			(3) Premature payload ejection		
R.BP.3	Premature shear pin break during ascent	Shear pins of insufficient diameter or strength Shear pin ejection	Premature section separation	3B	(1) 4-40 Shear pins are used to secure body sections (2) Shear pin stress tests shall be performed prior to launch
R.BP.4	High wind drift	Premature section separation during descent	Inability to communicate with payload or launch vehicle GPS	3B	Pressure ports are drilled into airframe
R.BP.5	Excessive pressure in AV bay	Sympathetic detonation of primary and secondary black powder charges	(1) Hoop stress on body tube (2) Structural damage (See S.A.1)	4A	Bulkheads separate primary and secondary black powder charges such that they fire away from one another

Table 5-8 Payload FMEA

Label	Hazard	Cause	Effect	LS	Mitigation
Payload Structure Hazard					
P.S.1	Pre-separation retention latch disengagement	Premature payload altimeter signal	(1) LOPSIDED essential power/ communication cord disconnection (2) LOPSIDED/POS cracks, chips, and breaks	2B	Payload retention latch test is scheduled to be completed prior to flight; design mitigations pending
P.S.2	LV ground touchdown while retaining payload in payload bay	Retention latch failure to open	(1) Cracks, chips, and breaks in payload structure from loads imparted to system by	2B	Payload retention latch test is scheduled to be completed prior to flight;

			payload latch and rail (2) Destruction of LOPSIDED's supportive rollers and rails		design mitigations pending
P.S.3	Payload-parachute connection shear/break	Excessive load from payload parachute	(1) Ballistic descent of payload (2) Irreparable payload structural damage	4C	**DESIGN OPTIONS:** Area around connection is reinforced with composite material LOPSIDED-POS is made out of carbon fiber such that it can withstand the approximately 112 lbf of maximum load
Payload Retention Hazard					
P.R.1	**RAIL-ROLLER DESIGN:** Retention latch failure to open	Retention latch power cord disconnection	Payload retention structural failure (see F.P.S.2)	2B	Payload wires shall be chosen such that their length allows some slack
P.R.2	**RAIL ONLY DESIGN:** Payload angled such that frictional forces from payload retention rail prevent motion	Insufficient grease on payload retention rail	Untimely/absent payload ejection	3B	Grease is applied to payload rail prior to flight; this is noted in future checklists
P.R.3	Excess in-flight load imparted to payload-parachute connection	Parachute movement during flight	Structural failure (cracks, breaks) in payload-parachute connection	3C	**DESIGN OPTIONS:** Area around connection is reinforced with

					composite material LOPSIDED-POS is made out of carbon fiber such that it can withstand the 112 lbf of maximum load
P.R.4	Payload parachute attachment at landing	Electronic latch failure	(1) Payload abrasion against ground from parachute drag (2) Inability of leveling system to right POS; no clear picture obtained	2B	The electronic latches shall be tested for their ability to disconnect the payload parachute prior to flight
Imaging System Hazard					
P.I.1	In-flight camera dislodgement	LV structural failure (F.S.A.1, F.S.B.1&2)	(1) Obstructed field of view (2) No clear picture obtained	4A	A powered camera module shall fly on the sub-scale model to test its ability to withstand flight forces
		Inability of camera to withstand flight forces		3A	
P.I.2	Abrasion to camera surface	LV structural failure (F.S.A.1, F.S.B.1&2)	No clear picture obtained	3B	(1) Payload bay is confirmed to be clean prior to launch (2) LOPSIDED will right the payload such that no contact is made between the lenses and the ground
		Parachute movement during flight (see F.P.R.3)			

P.I.3	In-flight camera cable disconnection	Pre-separation retention latch disengagement	Inability for POS to capture/transmit image	3B	All wiring is chosen to be long enough for there to be slack
	In-flight Raspberry Pi USB power cord disconnection				
	In-flight transmitter wire disconnection				
	In-flight accelerometer wire disconnection				
P.I.4	Payload landing outside of transmitter range	Wind drift	Poor signal quality of photo transmission	2B	See R.A.2 Mitigation
		Recovery failure (See R.A.2 and R.BP.1)			
	RF interference between POS transmitter and GPS transmitter	POS and GPS transmitters sharing a frequency		2A	POS and GPS transmitters shall operate on different frequencies
P.I.5	Battery disconnection	Excessive loads from main parachute (see F.P.R.3)	Loss of power to POS	3B	(1) A piece of electrical tape secures the lead to the battery (2) Wire lengths shall be chosen such that slack is present in each wire
	Insufficient battery capacity	Unanticipated in-flight current draw		3B	All batteries used for launch shall be tested with a multimeter prior to launch

5.5 Environmental Hazard Analysis

The team's environmental hazard analysis is shown in Table 5-9 below.

Table 5-9 Environmental Hazard Analysis

Label	Hazard	Cause	Effect	LS	Mitigation
Hazard to Environment					
E.1	LV touchdown in surrounding trees	Large wind gusts contributing to wind drift	Damage to trees and local wildlife	N/A	(1) The team shall not launch in wind conditions exceeding 20 mph (2) The team shall not cut down or damage trees/wildlife in any way
E.2	Catastrophe at takeoff	Cracks in motor casing Holes/bubbles in propellant grain	(1) Fire around launch field (2) Wildfire risk	N/A	(1) Aerotech motors are chosen for their low likelihood of defects contributing to CATO (2) A fire extinguisher is made available to personnel by the RSO
E.3	Explosion on touchdown	Late black powder charge	(1) Fire at recovery site (2) Wildfire risk	N/A	A fire extinguisher, provided by the team, shall be brought to the recovery site
E.4	Payload/avionics battery leakage	Puncture during flight	Consumption of volatile chemicals by wildlife	N/A	Batteries are shielded from environmental hazards through protective casing
E.5		Catastrophe at takeoff	Consumption of inedible	N/A	As much shrapnel is

	Abandonment of irretrievable launch vehicle shrapnel	Catastrophic payload failure	materials by wildlife		collected as possible by members of the recovery team
E.6	High-energy impact of launch vehicle with launch field	Ballistic descent of launch vehicle components	Structural damage to launch field; ruts and pits created by launch vehicle	N/A	See R.A.2, R.A.3, and R.BP.1 mitigations
Hazard from Environment					
E.7	LV touchdown in surrounding trees	Large wind gusts contributing to wind drift	(1) Launch vehicle structural damage (2) Inability to recover rocket	4B	The team shall not launch in wind conditions exceeding 20 mph
E.8	Waterlogged launch vehicle sections	Launch vehicle touchdown in irrigation ditch	(1) Waterlogged payload electronics (2) Inability to repair damaged components without significant effort	4C	**DESIGN OPTION (Unconfirmed):** Moisture sensors will be used to shut down payload electronics upon contact with water
E.9	LOPSIDED lopsided touchdown	Launch field ruts, ditches, and dips	Inability of LOPSIDED to right the system; no clear picture obtained	3D	**DESIGN OPTION (Unconfirmed):** LOPSIDED will be able to right the system at steep touchdown angles
E.10	Live black powder charges on touchdown	High humidity	BP explosion prompted by personnel-generated pressure changes	4A	(1) The team shall not launch in high humidity (2) Altimeters shall be disarmed by recovery

					personnel before any other recovery step
E.11	Waterlogged black powder	Rain	No black powder ignition; ballistic descent of launch vehicle	4B	The launch vehicle shall not be launched in rain
	Electronics rust		Loss of electronics	2B	
	Electronics short during flight		No black powder ignition; ballistic descent of launch vehicle No payload ignition	4B	
	Body tube saturation		Airframe structural damage	4B	

6. Project Plan

6.1 Requirements Verification

6.1.1 NASA Handbook Requirements

Table 6-1 below is the requirements verification matrix for the NASA handbook requirements.

Table 6-1 NASA Requirement Verification Matrix

Req No.	Shall Statement	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
NASA 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 will 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 new and original work.	Inspection	Project Management	Not Verified	The team plans on using all original work from students to complete the project.

NASA 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, including 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.
NASA 1.3	Foreign National (FN) team members SHALL be identified by the Preliminary Design Review (PDR) and may or may not have access to certain activities during Launch Week due to security restrictions. In addition, FN's may be separated from their team during certain activities on site at Marshall Space Flight Center.	Foreign National (FN) team members are identified and reported in the PDR milestone document.	Inspection	Project Management	Verified	There are no Foreign National team members.
NASA 1.4	The team SHALL identify all team members who plan to attend Launch Week activites by the Critical Design Review (CDR).	All team members attending launch week activities are identified and reported in the CDR milestone document.	Inspection	Project Management	Not Verified	TBD

NASA 1.4.1	Team members attending competion SHALL include students actively engaged in the project throughout the entire year.	The project management team identifies and selects members actively engaged in the project throughout the year to attend competion.	Inspection	Project Management	Not Verified	TBD
NASA 1.4.2	Team members attending competion SHALL include one mentor (see requirement 1.13).	The project management team invites the mentors listed in Section 1.1.2 to attend competition.	Inspection	Project Management	Not Verified	TBD
NASA 1.4.3	Team members attending competition SHALL include no more than two adult educators.	The project management team invites the team's one adult educator to attend competition.	Inspection	Project Management	Not Verified	TBD
NASA 1.5	The team SHALL engage a minimum of 200 participants in educational, hands-on science, technology, engineering, and mathematics (STEM) activities. These activities can be conducted in-person or virtually. To satisfy this requirement, all events must occur between project acceptance and the FRR due date. The STEM Engagement Activity Report SHALL be submitted via email within two weeks of the completion of each event.	The outreach lead implements STEM engagement plans with K-12 student groups throughout the project lifecycle. The outreach lead submits a STEM Engagement Activity Report via email within two weeks of the completion of each event.	Inspection	Project Management	Not Verified	TBD

NASA 1.6	The team SHALL establish a social media presence to inform the public about team activities.	The webmaster and social media lead coordinate to develop an educational and engaging social media presence on platforms including, but not limited to: the club website, Facebook, Instagram, and Twitter	Inspection	Project Management	Verified	Team social media information has been sent to the NASA project management team.
NASA 1.7	The team 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 will be sufficient.	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 plans to email all deliverables to the NASA project management team by the deadline specified in the handbook.
NASA 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.

NASA 1.9	In every report, the team 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	A Table of Contents is included on page ii of this document.
NASA 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	Page numbers are listed at the bottom of each page of this document.
NASA 1.11	The team SHALL provide any computer equipment necessary to perform a video teleconference with the review panel.	Each team member participating in the video teleconference acquires the necessary equipment for them to perform a video teleconference with the review panel.	Inspection	Project Management	Not Verified	The team plans on providing their own computer equipment necessary to participate in video teleconferences with the review panel.
NASA 1.12	The team SHALL be required to use the launch pads provided by Student Launch's launch services provider.	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 this design.	Inspection	Aerodynamics; Structures	Not Verified	The team plans on equipping the launch vehicle with rail buttons suitable for use on a 12-foot 1515 launch rail.

NASA 1.13	Each team SHALL identify a "mentor."	The team lead identifies qualified community members to mentor team members.	Inspection	Project Management	Verified	See Section 1.1.2 for mentor listing and contact information.
NASA 1.14	Each team 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 for time spent on this milestone.
NASA 2.1	The vehicle SHALL deliver the payload to an apogee altitude between 3,500 and 5,500 feet above ground level (AGL).	The aerodynamics lead designs a launch vehicle to reach an apogee between 3,500 and 5,500 feet AGL. The team then constructs the vehicle as designed and the launch vehicle flies between 3,500 at 5,500 feet AGL.	Analysis; Demonstration	Aerodynamics	Not verified	See Section 3.3.1 for apogee predictions.
NASA 2.2	The team SHALL identify their target altitude goal at the PDR milestone.	The aerodynamics lead declares the team's target altitude goal in the PDR milestone report.	Inspection	Aerodynamics	Verified	See Section 1.2.1 for target altitude identification.
NASA 2.3	The vehicle SHALL carry one commercially available, barometric altimeter for recording the official altitude used in determining the Altitude Award winner.	The recovery lead designates one onboard altimeter to record the official altitude used in determining the Altitude Award winner.	Inspection	Recovery	Not verified	See Section 3.2.3 for leading altimeter selection.

NASA 2.4	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 structures and recovery leads design the launch vehicle such that it is capable of being recovered with minimal damage and launched again on the same day without repairs or modifications.	Demonstration	Recovery; Structures	Not verified	See Section 3.2.3 for the leading recovery design.
NASA 2.5	The launch vehicle SHALL have a maximum of four (4) independent sections.	The aerodynamics and recovery subsystem leads design a launch vehicle that has fewer than four (4) independent sections.	Inspection	Aerodynamics ; Recovery	Verified	See Section 3.1.4 for the leading launch vehicle design.
NASA 2.5.1	Coupler/airframe shoulders which are located at in-flight separation points SHALL be at least 1 body diameter in length.	The aerodynamics lead designs the airframe such that couplers/shoulders at in-flight separation points are at least 1 body diameter in length	Inspection	Aerodynamics	Not verified	See Section 3.1.4 for the leading launch vehicle design.
NASA 2.5.2	Nosecone shoulders which are located at in-flight separation points SHALL be at least 1/2 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	Not verified	See Section 3.1.4 for the leading launch vehicle design.

NASA 2.6	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 less than two (2) hours.	Demonstration	Project Management; Safety	Not verified	TBD
NASA 2.7	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.	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.	Demonstration	Project Management; Safety	Not verified	TBD
NASA 2.8	The launch vehicle SHALL be capable of being launched by a standard 12-volt direct current firing system.	The project management and safety teams select a motor ignitor capable of being ignited from a 12-volt direct current firing system.	Demonstration	Project Management; Safety	Not verified	TBD
NASA 2.9	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 limit the launch vehicle such that no external circuitry or ground support equipment is required for launch.	Demonstration	Project Management; Safety	Not verified	The current launch vehicle design does not require external circuitry for launch.

NASA 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 3.1.5 for the leading motor selection.
NASA 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.	Inspection	Aerodynamics	Not verified	See Section 3.1.5 for the leading motor selection.
NASA 2.10.2	Any motor change after CDR SHALL be approved by the NASA Range Safety Officer (RSO).	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
NASA 2.11	The launch vehicle SHALL be limited to a single stage.	The aerodynamics lead designs the launch vehicle such that it only utilizes a single stage.	Inspection	Aerodynamics	Not verified	See Section 3.1.4 for the leading launch vehicle design.
NASA 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 selects a motor that does not exceed 5,120 Newton-seconds of total impulse.	Inspection	Aerodynamics	Not verified	See Section 3.1.5 for the leading motor selection.

NASA 2.13	Pressure vessels on the vehicle SHALL be approved by the RSO.	The structures lead provides the necessary data on any onboard pressure vessels to the NASA RSO and home field RSO.	Inspection	Structures	Not verified	No pressure vessels are included in the leading design.
NASA 2.13.1	The minimum factor of safety (Burst or Ultimate pressure versus Max Expected Operating Pressure) for pressure vessels on the vehicle SHALL be 4:1 with supporting design documentation included in all milestone reviews.	The structures lead includes design documentation supporting a factor of safety of 4:1 for any pressure vessel on the launch vehicle in each milestone report.	Analysis; Inspection	Structures	Not verified	No pressure vessels are included in the leading design.
NASA 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 any onboard pressure vessels such that they 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.	Analysis; Inspection	Structures	Not verified	No pressure vessels are included in the leading design.

NASA 2.13.3	The full pedigree of any pressure vessel on the launch vehicle 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	No pressure vessels are included in the leading design.
NASA 2.14	The launch vehicle SHALL have a minimum static stability margin of 2.0 at the point of rail exit.	The aerodynamics lead designs the launch vehicle such that it has a static stability margin of at least 2.0 at the point of rail exit.	Analysis	Aerodynamics	Verified	See Section 3.3.4 for stability calculations for the leading launch vehicle design.
NASA 2.15	Any structural protuberance on the rocket SHALL be located aft of the burnout center of gravity. Camera housings will be exempted, provided the team can show that the housing(s) causes minimal aerodynamic effect on the rocket's stability	The aerodynamics lead designs the launch vehicle such that there are no structural protuberances forward 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 3.1.4 for the leading launch vehicle design.

NASA 2.16	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 minimum velocity of 52 fps is achieved by rail exit.	Analysis	Aerodynamics	Verified	See Section 3.3.2 for performance calculations for the leading launch vehicle design.
NASA 2.17	The team SHALL successfully launch and recover a subscale model of their rocket prior to CDR. Subscale flight data SHALL be reported at the CDR milestone.	The team launches and recovers a subscale model of the launch vehicle. The team reports subscale flight data in the CDR milestone report.	Demonstration	Project Management	Not verified	See Section 6.4 for the project plan timeline.
NASA 2.17.1	The subscale model SHALL 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
NASA 2.17.2	The subscale model SHALL carry an altimeter capable of recording the model's apogee altitude.	The recovery lead installs an altimeter capable of recording the subscale launch vehicle's apogee altitude in the subscale launch vehicle.	Inspection	Recovery	Not verified	The team plans on including an altimeter capable of recording the subscale model's apogee altitude onboard the subscale model.
NASA 2.17.3	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 and built specifically for this year's project.	Inspection	Project Management	Not verified	The team is currently constructing a new subscale rocket specifically built for this year's project.

NASA 2.17.4	Proof of a successful flight SHALL be supplied in the CDR report.	The team supplies proof of a successful subscale flight in the CDR milestone report.	Inspection	Project Management	Not verified	TBD
NASA 2.18.1	Vehicle Demonstration Flight - All teams SHALL successfully launch and recover their full-scale rocket prior to FRR in its final flight configuration.	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 the project plan timeline.
NASA 2.18.1.1	The vehicle and recovery system SHALL function as designed during the VDF.	No anomalies are detected in the performance of the launch vehicle and its recovery system during the VDF.	Demonstration	Project Management	Not verified	TBD
NASA 2.18.1.2	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, designed and build specifically for this year's project.	Inspection	Project Management	Not verified	TBD
NASA 2.18.1.3.1	If the payload is not flown on the VDF, mass simulators SHALL be used to simulate the payload mass.	If the payload is not flown on the VDF, the structures lead installs mass simulators to simulate the payload mass.	Inspection	Structures	Not verified	TBD

NASA 2.18.1.3.2	Payload mass simulators SHALL be located in the same approximate location on the rocket as the missing payload mass.	If the payload is not flown on the VDF, the structures lead install mass simulators in the same approximate location of the missing payload mass.	Inspection	Structures	Not verified	TBD
NASA 2.18.1.4	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
NASA 2.18.1.5	Teams SHALL fly the Launch Day motor for the Vehicle Demonstration Flight.	The aerodynamics lead installs the Launch Day motor for the VDF.	Inspection	Aerodynamics	Not verified	TBD
NASA 2.18.1.6	The vehicle SHALL be flown in its fully ballasted configuration during the full-scale test flight.	The structures lead installs all required ballast for the VDF.	Inspection	Structures	Not verified	TBD
NASA 2.18.1.7	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).	Following a successful VDF, the project management team does not allow modification of the launch vehicle or any of its components without approval from the NASA RSO.	Inspection	Project Management	Not verified	TBD

NASA 2.18.1.8	Proof of a successful flight SHALL be supplied in the FRR report. Altimeter data output is required to meet this requirement.	The recovery lead includes altimeter data from the VDF in the FRR milestone report.	Inspection	Recovery	Not verified	TBD
NASA 2.18.1.9	Vehicle Demonstration flights SHALL be completed by the FRR submission deadline. 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	See Section 6.4 for the project plan timeline.
NASA 2.18.2	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 on Launch Day.	The team completes the PDF prior to the PDF deadline using the same rocket to be flown on Launch Day.	Inspection	Project Management	Not verified	See Section 6.4 for the project plan timeline.
NASA 2.18.2.1	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 remains fully retained until the point of intended deployment with each retention mechanism functioning as designed and not sustaining damage requiring repair during the PDF.	Demonstration	Payload Integration	Not verified	TBD

NASA 2.18.2.2	The payload flown SHALL be the final, active version.	The payload flown on the PDF is the final, active version of the payload.	Inspection	Project Management	Not verified	TBD
NASA 2.18.2.4	Payload Demonstration Flights SHALL be completed by the FRR Addendum deadline.	The PDF is completed by the FRR Addendum deadline.	Inspection	Project Management	Not verified	See Section 6.4 for the project plan timeline.
NASA 2.19	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
NASA 2.19.1	If a re-flight is necessary, the team SHALL submit the FRR Addendum by the FRR Addendum deadline.	The team lead submits the FRR Addendum by the FRR Addendum deadline.	Inspection	Project Management	Not verified	TBD
NASA 2.19.2	The team SHALL successfully execute a PDF to fly a final competition launch.	The project management team manages the schedule such that a PDF is successfully completed by the FRR Addendum deadline.	Demonstration	Project Management	Not verified	TBD

NASA 2.20	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 team lead places their contact information on the rocket airframe and any section of the vehicle that is not tethered to the main airframe in a manner that allows this information to be retrieved without opening or separating the vehicle.	Inspection	Project Management	Not verified	TBD
NASA 2.21	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 safety team ensures all Lithium Polymer batteries are sufficiently protected from ground impact and are marked appropriately.	Analysis; Inspection	Safety	Not verified	TBD
NASA 2.22.1	The launch vehicle SHALL not utilize forward firing motors.	The aerodynamics lead designs the launch vehicle such that it does not utilize forward firing motors.	Inspection	Aerodynamics	Not verified	The leading launch vehicle design does not include forward firing motors.
NASA 2.22.2	The launch vehicle SHALL not utilize motors that expel titanium sponges (Sparky, Skidmark, Metal-Storm, etc.)	The aerodynamics lead selects a motor that does not expel titanium sponges.	Analysis; Inspection	Aerodynamics	Not verified	See Section 3.1.5 for the leading motor selection.
NASA 2.22.3	The launch vehicle SHALL not utilize hybrid motors.	The aerodynamics lead selects a motor which uses exclusively APCP.	Analysis; Inspection	Aerodynamics	Not verified	See Section 3.1.5 for the leading motor selection.

NASA 2.22.4	The launch vehicle SHALL not utilize a cluster of motors.	The aerodynamics lead selects a single motor only for use in the launch vehicle.	Analysis; Inspection	Aerodynamics	Not verified	See Section 3.1.5 for the leading motor selection.
NASA 2.22.5	The launch vehicle SHALL not utilize friction fitting for motors.	The structures lead installs a motor retention system that does not use friction fitting.	Inspection	Structures	Not verified	See Section 3.1.5 for the leading launch vehicle design.
NASA 2.22.6	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 flight.	Analysis	Aerodynamics	Verified	See Section 3.3.2 for the leading launch vehicle performance predictions.
NASA 2.22.7	Vehicle ballast SHALL not exceed 10% of the total unballasted weight of the rocket as it would sit on the pad	The aerodynamics lead designs the launch vehicle such that it does not require ballast exceeding 10% of the total unballasted weight of the launch vehicle.	Analysis; Inspection	Aerodynamics	Not verified	See Section 3.1.4 for the leading launch vehicle design. No ballast is currently planned for the leading launch vehicle design.
NASA 2.22.8	Transmissions from onboard transmitters, which are active at any point prior to landing, SHALL not exceed 250 mW of power (per transmitter).	The safety team verifies all transmitters activated prior to landing are not capable of transmissions exceeding 250 mW of power per transmitter.	Analysis	Safety	Not verified	TBD

NASA 2.22.9	Transmitters SHALL not create excessive interference. Teams SHALL utilize unique frequencies, hand-shake/passcode systems, or other means to mitigate interference caused to or received from other teams.	The safety team verifies no transmitter creates excessive interference. The safety team enforces the usage of unique frequencies to mitigate interference with other teams.	Analysis; Demonstration	Safety	Not verified	TBD
NASA 2.22.10	Excessive and/or dense metal SHALL not be utilized in the construction of the vehicle.	The structures lead minimizes the amount of metal onboard the launch vehicle.	Inspection	Structures	Not verified	The leading design includes metal only for threaded rods for the AV bay, and on the payload levelling system.
NASA 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.	The recovery lead designs a dual-deployment recovery system.	Demonstration	Recovery	Not verified	See Section 3.2.1 for description of recovery events.
NASA 3.1.1	The main parachute SHALL be deployed no lower than 500 feet.	The recovery lead designs the recovery system such that the main parachute deploys no lower than 500 feet.	Demonstration	Recovery	Not verified	See Section 3.2.1 for description of recovery events.
NASA 3.1.2	The apogee event SHALL contain a delay of no more than 2 seconds.	The recovery lead designs the recovery system such that the apogee event has a delay of no more than 2 seconds.	Demonstration	Recovery	Not verified	See Section 3.2.1 for description of recovery events.

NASA 3.1.3	Motor ejection SHALL not be used for primary or secondary deployment.	The recovery lead designs a recovery system that does not utilize motor ejection.	Inspection	Recovery	Not verified	See Section 3.2.3 for leading recovery system design.
NASA 3.2	The 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 performs a ground ejection test for each electronically initiated recovery event prior to the initial flights of the subscale and full scale launch vehicles.	Demonstration	Recovery	Not verified	The team plans on completing ground ejection testing prior to the subscale and full-scale launches.
NASA 3.3	Each independent section of the launch vehicle SHALL have a maximum kinetic energy of 75 ft-lbf at landing.	The recovery lead designs a recovery system that results in no launch vehicle section having a kinetic energy greater than 75 ft-lbf at landing.	Analysis	Recovery	Not verified	See Section 3.3.6 for leading kinetic energy calculations.
NASA 3.4	The recovery system SHALL contain redundant, commercially available altimeters.	The recovery lead includes at least two independent commercially available altimeters in the recovery system.	Inspection	Recovery	Not verified	See Section 3.2.3 for leading altimeter selection.
NASA 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 the recovery system such that each altimeter has a dedicated power supply of commercially available batteries.	Inspection	Recovery	Not verified	See Section 3.2.2.4 for leading avionics system design.

NASA 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 installs dedicated mechanical arming switches accessible from the exterior of the launch vehicle airframe for each altimeter.	Inspection	Recovery	Not verified	See Section 3.2.2.4 for leading avionics system design.
NASA 3.7	Each arming switch SHALL be capable of being locked in the ON position for launch.	The recovery lead selects mechanical arming switches capable of being locked in the ON position.	Inspection	Recovery	Not verified	See Section 3.2.2.4 for leading avionics system design.
NASA 3.8	The recovery system electrical circuits SHALL be completely independent of any payload electrical circuits.	The recovery lead designs the recovery system so that its electrical circuits are completely independent of any payload electrical circuits.	Inspection	Recovery	Not verified	See Section 3.2.2.4 for leading avionics system design.
NASA 3.9	Removable shear pins SHALL be used for both the main parachute compartment and the drogue parachute compartment.	The recovery lead designs the recovery system to use removable shear pins for the main parachute compartment and drogue parachute compartment.	Inspection	Recovery	Not verified	See Section 3.2.1 for leading recovery system description.
NASA 3.10	The recovery area SHALL be limited to a 2,500 ft. radius from the launch pads.	The recovery lead selects parachutes that prevent the launch vehicle from drifting	Analysis; Demonstration	Recovery	Not verified	See Section 3.3.8 for leading wind drift calculations.

		more than 2,500 ft. from the launch pads.				
NASA 3.11	Descent time of the launch vehicle SHALL be limited to 90 seconds (apogee to touch down).	The recovery lead selects parachutes that allow the launch vehicle to touch down within 90 seconds of reaching apogee.	Analysis; Demonstration	Recovery	Not verified	See Section 3.3.7 for leading descent time calculations.
NASA 3.12	An electronic 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 selects and installs an electronic tracking device capable of transmitting the position of the launch vehicle or any independent section to a ground receiver.	Inspection; Demonstration	Recovery	Not verified	See Section 3.2.2.1 for leading tracking device selection.
NASA 3.12.1	Any rocket section or payload component, which lands untethered to the launch vehicle, SHALL contain an active electronic tracking device.	The recovery lead installs an electronic tracking device in any launch vehicle section or payload component which lands untethered to the launch vehicle.	Inspection	Recovery	Not verified	See Section 3.2.2.1 for leading launch vehicle tracking device selection. See Section 4.4.2 for leading payload tracking device selection.
NASA 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 the recovery system such that it is not affected by other on-board electronic devices.	Demonstration	Recovery	Not verified	See Section 3.2.3 for leading avionics system design.

NASA 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 which houses the recovery system altimeters in a physically separate compartment within the launch vehicle.	Inspection	Recovery	Not verified	See Section 3.2.3 for leading avionics system design.
NASA 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 and installs shielding for the recovery system electronics from all onboard transmitting devices.	Inspection	Recovery	Not verified	See Section 3.2.3 for leading avionics system design.
NASA 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 and installs shielding for the recovery system electronics from all devices which may generate magnetic waves.	Inspection	Recovery	Not verified	See Section 3.2.3 for leading avionics system design.
NASA 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 and installs shielding for the recovery system electronics from all devices which may adversely affect the proper operation of the recovery system electronics.	Inspection	Recovery	Not verified	See Section 3.2.3 for leading avionics system design.

NASA 4.2	The team SHALL design a planetary landing system to be launched in a high-power rocket. The lander system SHALL be capable of being jettisoned from the rocket during descent, landing in an upright configuration or autonomously uprighting after landing. The system SHALL self-level within a five-degree tolerance from vertical. After autonomously uprighting and self-leveling, it SHALL take a 360-degree panoramic photo of the landing site and transmit the photo to the team.	The payload team designs a planetary landing system to be launched in a high-powered rocket. The payload is capable of being jettisoned from the launch vehicle during descent, landing in an upright configuration or autonomously uprighting after landing, self-levelling within a five-degree tolerance from vertical, and taking a 360-degree panoramic photo of the landing site and transmitting the photo to the team.	Demonstration	Payload vehicle; Payload integration; Payload imaging	Not verified	See Section 4.4 for leading payload design.
NASA 4.3.1	The landing system SHALL be completely jettisoned from the rocket at an altitude between 500 and 1,000 ft. AGL. The landing system SHALL land within the external borders of the launch field. The landing system SHALL not be tethered to the launch vehicle upon landing.	The payload integration lead designs the payload such that it jettisons from the launch vehicle between 500 and 1,000 AGL, lands within the external border of the launch field, and is not tethered to the launch vehicle.	Demonstration	Payload integration	Not verified	See Section 4.4.2 for leading payload jettison design.

NASA 4.3.2	The landing system SHALL land in an upright orientation or SHALL be capable of reorienting itself to an upright configuration after landing. Any system designed to reorient the lander SHALL be completely autonomous.	The payload vehicle lead designs the payload such that it lands in an upright position or reorients itself to an upright configuration after landing using a completely autonomous system.	Demonstration	Payload vehicle	Not verified	See Section 4.4.1 for leading payload lander design.
NASA 4.3.3	The landing system SHALL self-level to within a five-degree tolerance from vertical.	The payload vehicle lead designs the payload such that it is capable of self-levelling within a five-degree tolerance from vertical.	Demonstration	Payload vehicle	Not verified	See Section 4.4.1 for leading payload levelling system design.
NASA 4.3.3.1	Any system designed to level the lander SHALL be completely autonomous.	The payload vehicle lead designs a payload levelling system that is completely autonomous.	Demonstration	Payload vehicle	Not verified	See Section 4.4.1 for leading payload levelling system design.
NASA 4.3.3.2	The landing system SHALL record the initial angle after landing, relative to vertical, as well as the final angle, after reorientation and self-levelling. This data SHALL be reported in the Post Launch Assessment Report (PLAR).	The payload vehicle lead designs a payload levelling system which records the initial angle after landing as well as the final angle relative to vertical. The payload vehicle lead reports this data in the PLAR.	Demonstration	Payload vehicle	Not verified	See Section 4.4.1 for leading payload levelling system design.

NASA 4.3.4	Upon completion of reorientation and self-levelling, the lander SHALL produce a 360-degree panoramic image of the landing site and transmit it to the team.	The payload imaging lead designs an imaging system capable of producing a 360-degree panoramic image and transmitting it to the team following self-levelling of the payload vehicle.	Demonstration	Payload imaging	Not verified	See Section 4.4.3 for leading imaging system design.
NASA 4.3.4.1	The hardware receiving the image SHALL be located within the team's assigned prep area or the designated viewing area.	The payload imaging lead selects a ground station capable of receiving the image and being located within the team's prep area or designated viewing area.	Demonstration	Payload imaging	Not verified	See Section 4.4.3 for leading imaging system design.
NASA 4.3.4.2	Only transmitters that were onboard the vehicle during launch SHALL be permitted to operate outside of the viewing or prep areas.	The team does not operate transmitters outside the viewing or prep areas that were not onboard the vehicle during launch.	Demonstration	Payload imaging	Not verified	See Section 4.4.3 for leading transmitter system design.

NASA 4.3.4.3	Onboard payload transmitters SHALL be limited to 250 mW of RF power while onboard the launch vehicle but may operate at a higher RF power after landing on the planetary surface. Transmitters operating at higher power SHALL be approved by NASA during the design process.	The payload imaging lead selects onboard transmitters limited to 250 mW of RF power while onboard the launch vehicle. The payload imaging lead receives approval from NASA for operating transmitters outside of the launch vehicle at higher power.	Inspection	Payload imaging	Not verified	See Section 4.4.3 for leading image transmitter system design.
NASA 4.3.4.4	The image SHALL be included in the team's PLAR.	The payload imaging lead includes the captured image in the PLAR.	Inspection	Payload imaging	Not verified	TBD
NASA 4.4.1	Black powder and/or similar energetics SHALL only be used for deployment of in-flight recovery systems.	The payload integration lead designs the payload recovery system such that any energetics are only utilized in-flight.	Demonstration	Payload integration	Not verified	See Section 4.4.2 for leading payload deployment design.
NASA 4.4.2	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	The leading payload design complies with all FAA and NAR rules and regulations.

NASA 4.4.3	Any experiment element that is jettisoned, except for planetary lander experiments, during the recovery phase SHALL receive real-time RSO permission prior to initiating the jettison event.	The payload integration lead receives real-time RSO permission prior to initiating the jettison of any experiment element except for the planetary lander.	Demonstration	Payload integration	Not verified	The team does not plan on including a secondary payload experiment.
NASA 4.4.4	Unmanned aircraft system (UAS) payloads, if designed to be deployed during descent, SHALL be tethered to the vehicle with a remotely controlled release mechanism until the RSO has given permission to release the UAS.	The payload integration lead designs any UAS payload to be tethered to the launch vehicle with a remotely controlled release mechanism until the RSO gives permission to release the UAS.	Demonstration	Payload integration	Not verified	The leading payload design is not a UAS.
NASA 4.4.5	Teams flying UASs SHALL abide by all applicable FAA regulations, including the FAA's Special Rule for Model Aircraft (Public Law 112-95 Section 336).	The safety team verifies any UAS payload's compliance with all applicable FAA regulations.	Demonstration	Safety	Not verified	The leading payload design is not a UAS.
NASA 4.4.6	Any UAS weighing more than .55 lbs. SHALL be registered with the FAA and the registration number marked on the vehicle.	The payload vehicle lead registers any UAS weighing more than .55 lbs with the FAA and marks the registration number of the vehicle.	Demonstration	Payload vehicle	Not verified	The leading payload design is not a UAS.

NASA 5.1	The team SHALL use a launch and safety checklist. The final checklists SHALL be included in the FRR report and used during the Launch Readiness Review (LRR) and any Launch Day operations.	The safety team creates a launch and safety checklist. The safety team includes these checklists in the FRR milestone report and verifies their use during LRR and Launch Day operations.	Inspection; Demonstration	Safety	Not verified	TBD
NASA 5.2	Each team SHALL identify a student safety officer who will be responsible for all items in requirements section 5.3.	A safety officer is identified in each milestone report.	Inspection	Project management	Verified	See Section 5.1 for safety officer identification.
NASA 5.3.1.1	The safety officer SHALL monitor team activities with an emphasis on safety during the design of the launch vehicle and payload.	The safety officer is present for and engages in at least half of all launch vehicle and payload design meetings.	Inspection	Safety	Not verified	TBD
NASA 5.3.1.2	The safety officer SHALL monitor team activities with an emphasis on safety during the construction of the launch vehicle and payload components.	The safety officer is present and engaged for all launch vehicle and payload construction meetings.	Inspection	Safety	Not verified	TBD
NASA 5.3.1.3	The safety officer SHALL monitor team activities with an emphasis on safety during the assembly of the launch vehicle and payload.	The safety officer is present and engaged during the assembly of the launch vehicle and payload.	Inspection	Safety	Not verified	TBD

NASA 5.3.1.4	The safety officer SHALL monitor team activities with an emphasis on safety during the ground testing of the launch vehicle and payload.	The safety officer is present and engaged for all ground tests of the launch vehicle and payload.	Inspection	Safety	Not verified	TBD
NASA 5.3.1.5	The safety officer SHALL monitor team activities with an emphasis on safety during the subscale launch test(s).	The safety officer is present and engaged at any subscale launch test.	Inspection	Safety	Not verified	The safety officer plans to attend the subscale launch test.
NASA 5.3.1.6	The safety officer SHALL monitor team activities with an emphasis on safety during the full-scale launch test(s).	The safety officer is present and engaged at any full-scale launch test.	Inspection	Safety	Not verified	TBD
NASA 5.3.1.7	The safety officer SHALL monitor team activities with an emphasis on safety during Launch Day.	The safety officer is present and engaged at Launch Day.	Inspection	Safety	Not verified	TBD
NASA 5.3.1.8	The safety officer SHALL monitor team activities with an emphasis on safety during recovery activities.	The safety officer is present and engaged for all recovery activities.	Inspection	Safety	Not verified	TBD
NASA 5.3.1.9	The safety officer SHALL monitor team activities with an emphasis on safety during STEM engagement activities.	The safety officer is present and engaged for at least half of all STEM engagement activities.	Inspection	Safety	Not verified	TBD

NASA 5.3.2	The safety officer SHALL implement procedures developed by the team for construction, assembly, launch, and recovery activities.	The safety officer develops a safety plan for construction, assembly, launch, and recovery activities, and verifies team adherence to this plan.	Demonstration	Safety	Not verified	TBD
NASA 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 safety officer manages and maintains current revisions of all hazard analyses, procedures, and MSDS/chemical inventory data.	Demonstration	Safety	Not verified	TBD
NASA 5.3.4	The safety officer SHALL assist in the writing and development of the team's hazard analyses, failure modes analyses, and procedures.	The safety officer leads in the writing and development of the team's hazard analyses, failure modes analyses, and procedures.	Demonstration	Safety	Not verified	The safety officer has written the team's hazard analysis and failure mode analysis in Section 5.
NASA 5.4	During test flights, the team SHALL abide by the rules and guidance of the local rocketry club's RSO. Teams SHALL communicate their intentions to the local club's President or Prefect and RSO before attending any NAR or TRA launch.	The safety officer communicates the team's intentions to the local club's President or Prefect and RSO prior to attending any NAR or TRA launch. The safety officer verifies team adherence to the rules and guidance of the local club's RSO.	Demonstration	Safety	Not verified	The team will abide by rules and guidance of the local rocketry club's RSO. The team has communicated its intentions to the local club's Prefect.

NASA 5.5	Teams SHALL abide by all rules set forth by the FAA.	The safety officer verifies the team adheres to all rules set forth by the FAA.	Demonstration	Safety	Not verified	The team currently complies with all FAA regulations.

6.1.2 Team-Derived Requirements

Table 6-2 shows the requirements verification matrix for team-derived requirements.

Table 6-2 Team-Derived Requirement Verification Matrix

Req No.	Shall Statement	Justification	Success Criteria	Verification Method	Subsystem Allocation	Status	Status Description
TDR 2.1	The launch vehicle airframe SHALL be water resistant.	The team's home launch field in Bayboro, NC has several large irrigation ditches that are typically filled with water. A water-resistant airframe will help reduce potential damage in the possible outcome of the launch vehicle landing in water.	The airframe is not damaged or deformed upon exposure to water.	Inspection, Analysis	Structures	Not verified	See Section 3.1.4.1 for details on material selection.

TDR 2.2	All critical components of the launch vehicle SHALL be designed with a minimum factor of safety of 1.5.	This will ensure that assumptions made within analysis or in a scenario with higher than expected loading will not cause unpredicted failure during flight. A factor of safety of 1.5 has been deemed sufficient to account for unexpected loading, while allowing components to remain lightweight.	The factor of safety for each critical component is reported in documentation.	Analysis, Test	Structures	Not verified	Factor of safety is determined through simulations and through testing. See section 3.1.4.10 for details on simulations. Additional testing and analysis will be presented in future documentation.
TDR 2.3	The launch vehicle SHALL be no larger than 6 inches in diameter.	Limiting the size of the launch vehicle makes it safer and easier to manipulate on the field.	The diameter of the airframe is not larger than 6 inches.	Inspection	Aerodynamics, Structures	Not verified	See Section 1.2.3 for launch vehicle size.
TDR 2.4	The launch vehicle SHALL have a stability margin between 2.0 and 2.3 at launch.	The minimum static stability margin as specified in NASA 2.14 is 2.0. The team is imposing a maximum static stability margin of 2.3 to reduce the risk of weathercocking.	The static stability margin of the launch vehicle at launch is between 2.0 and 2.3.	Analysis, Inspection	Aerodynamics	Not verified	See Section 3.3.4 for current stability predictions.

TDR 3.1	The secondary black powder charges SHALL be of greater mass than the primary black powder charges.	In the event the primary black powder charge fails to cause section separation, a larger charge will have a better chance of successful separation.	The secondary black powder charges when weighed are of greater mass than the primary black powder charges.	Inspection, Test	Recovery	Not verified	See Section 3.2.2.11 for black powder charge sizing calculations.
TDR 3.2	The launch vehicles blast caps SHALL be exposed and accessible.	Accessible energetic materials containers will allow for safer loading of energetic materials.	The designed Avionics Bay has easily accessible blast caps.	Inspection	Structures; Recovery	Not verified	See Section 3.2.3 for leading Avionics Bay design.
TDR 3.3	Drogue descent velocity SHALL be less than 125 fps.	This speed will minimize main parachute opening shock.	Calculations for launch vehicle velocity under drogue parachute are less than 125 fps.	Analysis	Recovery	Not verified	See Section 3.2.2.6 for preliminary descent velocity calculations.
TDR 3.4	The black powder ejection charges SHALL produce at least 10 psi.	This 10-psi target will produce sufficient force in most cases to shear a reasonable number of shear pins.	Black powder ejection charge calculations indicate a pressure of 10 psi will be reached.	Analysis	Recovery	Not verified	See Section 3.2.2.11 for black powder charge sizing calculations.

TDR 3.5	The launch vehicle SHALL use U-bolts for all recovery harness attachment points.	U-bolts reduce the chance of a single point of failure as they distribute the opening shock across the bulkhead.	The load bearing bulkheads have recovery attachment points that use U-bolts.	Inspection	Recovery	Not verified	See Section 3.1.4.2 for anchor selection.
TDR 3.6	The launch vehicle SHALL use threaded quick-links for all recovery harness attachment points.	Thread quick-links reduce the likelihood of recovery harness detachment due to flight forces.	The recovery harness will be attached at all points by threaded quick-links.	Inspection	Recovery	Not verified	See Section 3.2.2.10 for recovery component selection.
TDR 3.7	A deployment bag SHALL be utilized for packing of the main parachute.	Deployment bags reduce the likelihood of parachute damage from hot ejection charge gasses and reduce reliance on the folding technique for proper parachute deployment.	A deployment bag will be utilized to pack the main parachute.	Inspection	Recovery	Not verified	See Section 3.2.2.9 for recovery component selection.
TDR 3.8	The onboard altimeters SHALL use one 9V battery for each flight.	StratoLogger CF Altimeters use one 9V battery each.	The avionics sled is designed to hold two 9V batteries, one for each altimeter.	Inspection	Recovery	Not verified	See Section 3.2.2.4 for avionics electrical design.

TDR 3.9	A fresh 9V battery SHALL be selected for both onboard altimeters before each flight.	The system must be capable of remaining powered on for prolonged periods of time.	Each 9V battery placed on the avionics sled before flight must measure greater than 9V before flight.	Inspection	Recovery	Not verified	TBD
TDR 3.10	The recovery system SHALL be capable of recovering the launch vehicle within NASA Handbook requirements should the LOPSIDED-POS fail to separate from the launch vehicle.	The system must be robust enough to accommodate potential mission failures.	Descent calculations are within limits even should the payload fail to separate.	Analysis	Recovery	Not verified	See Section 3.3.6 for kinetic energy calculations.
TDR 4.1	The POS SHALL consist of multiple camera modules.	Multiple camera modules provide redundancy in the event of one camera failing.	Four Raspberry Pi camera modules are installed in the payload vehicle.	Inspection	Payload Imaging	Not verified	See Section 4.4.3 for description of current POS design.

TDR 4.2	POS cameras SHALL be protected from physical damage associated with launch, deployment, and landing while having a proper field of view for image capture.	If the camera modules are damaged, the ability of the payload to capture 360-degree images will be compromised.	The top section of LOPSIDED allows cameras to be housed with significant protection and field of view.	Analysis	Payload Imaging	Not verified	See Section 4.4.3 for description of current POS design.
TDR 4.3	POS electronics SHALL fit in a 69 x 69 x 190 mm volume.	The payload vehicle size is constrained by the 6-inch launch vehicle diameter as well as hardware for the self-leveling system.	The POS electronics, not including the camera modules, successfully fit within LOPSIDED.	Inspection	Payload Imaging	Not verified	See Section 4.4.3 for description of current POS design.
TDR 4.4	POS electronics, excluding the camera modules, SHALL be contained below the pivoting axis of LOPSIDED's leveling system.	For the proposed leveling system to work, LOPSIDED must have an overall CG below the pivoting axis.	LOPSIDED will be able to self-level after all POS components have been installed.	Demonstration	Payload Imaging	Not verified	See Section 4.4.3 for description of current POS design.
TDR 4.5	LOPSIDED SHALL fit within a 6-inch diameter constraint.	This allows the LOPSIDED to fit within the Launch Vehicle's payload bay.	LOPSIDED will be able to fit within the launch vehicle's payload bay.	Inspection	Payload Vehicle	Not verified	See Section 4.4.1 for description of current LOPSIDED design.

TDR 4.6	LOPSIDED SHALL remain locked in its neutral position until after landing and taking its initial orientation measurement.	The payload orientation system must remain locked in order to not damage the launch vehicle, interfere with any deployment mechanisms, and allow for an initial orientation measurement to be recorded.	The gimbal-based leveling system is not allowed to rotate until the initial orientation measurement is recorded.	Test	Payload Vehicle	Not verified	See Section 4.4.1 for description of current LOPSIDED design.
TDR 4.7	LOPSIDED's legs SHALL deploy immediately after exiting the payload bay.	This will expand the payload's footprint from its stowed configuration, increasing its stability. By having this event happen immediately, this decreases the risk of the legs not deploying in time for landing.	LOPSIDED's legs deploy as soon as they are no longer constrained by the payload bay.	Test	Payload Vehicle	Not verified	See Section 4.4.1 for description of current LOPSIDED design.

TDR 4.8	LOPSIDED's gravity-assisted self-levelling system SHALL allow the body to reorient within ~15 degrees of its neutral orientation.	Given the requirement of being within 5 degrees of vertical, the design allows for the body to move 15 degrees, allowing LOPSIDED to land on grades of up to 20 degrees in any direction and still complete the mission.	LOPSIDED has at least a 15-degree range of motion about either axis.	Test	Payload Vehicle	Not verified	See Section 4.4.1 for description of current LOPSIDED design.
TDR 4.9	LOPSIDED SHALL have an allotted volume of 69x69x69 cubic mm for the POS camera frame to fit within.	This will allow the POS to operate fully without any obstruction and will interface with the rest of LOPSIDED.	LOPSIDED has a 69x69x69 mm internal volume of air.	Analysis	Payload Vehicle	Not verified	See section 4.4.1 for description of current LOPSIDED design.
TDR 4.10	LOPSIDED SHALL have a designated ARRD mounting point at the top.	This will interface with the integration subsystem so that any forces it may receive from that system will not damage the payload and will allow for the parachutes to detach.	LOPSIDED has four screw rods at its top for the ARRD to be mounted to.	Analysis	Payload Vehicle	Not verified	See Section 4.4.1 for description of current LOPSIDED design.

TDR 4.11	LOPSIDED SHALL have a center of gravity beneath its pivoting axes.	This will allow the levelling system to utilize the force due to gravity while not requiring any motors to operate.	The LOPSIDED CG is below its pivoting axes.	Analysis	Payload Vehicle	Not verified	See Section 4.4.1 for description of current LOPSIDED design.
TDR 4.12	LOPSIDED-POS SHALL weigh no more than 10.5 lbs.	Maintaining a standard payload weight serves for a stable location of center of gravity for launch vehicle stability purposes.	LOPSIDED-POS has a total weight of 10.5 lbs. or less.	Inspection	Payload Vehicle, Payload Integration, Payload Imaging	Not verified	See Section 4.4 for payload design.
TDR 4.13	LOPSIDED SHALL exit within 3 seconds of deployment during jettison.	Successful deployment of the payload vehicle depends on the payload being able to slide out of the payload bay.	LOPSIDED is completely removed from the payload bay within 3 seconds of deployment.	Test	Payload Integration	Not verified	See Section 4.4.2 for payload integration design.

6.2 Budget

Table 6-3 below details the year-long budget for the 2020-2021 NASA Student Launch competition.

Table 6-3 2020-2021 NASA Student Launch Competition Budget

	Item	Quantity	Price per Unit	Item Total
Subscale Structure	Aerotech I435T-14A motor	2	$56.00	$112.00
	Aero Pack 38mm Retainer	1	$27.00	$27.00
	Motor Casing	1	$340.00	(Already own) $0.00
	38mm G12 Airframe, Motor Tube	1	$64.00	$64.00
	4" Phenolic Airframe, 3 Slots	1	$33.50	$33.50
	4" Phenolic Airframe	2	$26.00	$52.00
	4" Phenolic Coupler	4	$21.00	$84.00
	Plastic 4" 4:1 Ogive Nosecone	1	$23.00	$23.00
	Domestic Birch Plywood 1/8"x2x2	6	$14.82	$88.92
	Rail Buttons	4	$2.50	$10.00
	U-Bolts	4	$1.00	$4.00
	Blast Caps	4	$2.50	$10.00
	Terminal Blocks	3	$7.00	$21.00
	Paint	1	$100.00	$100.00
	Key Switches	2	$12.00	$24.00
	Subtotal:			**$541.42**
Full-Scale Structure	6" G12 Airframe, Full Length (60"), 3 Slots	1	$264.00	$264.00
	6" G12 Airframe, Half Length (30")	1	$114.00	$114.00
	3"/75mm G12 Motor Tube, 22" length	1	$37.00	$37.00
	6” G12 Coupler 14” Length	1	$70.00	$70.00
	6" G12 Coupler 12" Length	1	$60.00	$60.00
	6" Fiberglass 4:1 Ogive Fiberglass Nosecone	1	$149.95	$149.95
	Domestic Birch Plywood 1/8"x2x2	8	$14.82	$118.56
	Aerotech 75/3840 Motor Case	1	$360.00	(Already own) $0.00
	75 mm Motor Retainer	1	$72.00	$72.00
	Rail Buttons	4	$2.50	$10.00
	U-Bolts	4	$1.00	$4.00
	Aerotech 1390G motor	2	$224.00	$448.00
	Aerotech 75mm Forward Seal Disk	1	$37.50	$37.50
	Blast Caps	4	$2.50	$10.00
	Terminal Blocks	3	$7.00	$21.00
	Paint	1	$150.00	$150.00
	Key Switches	2	$12.00	$24.00
	Subtotal:			**$1,590.01**

Payload	Arducam Camera w/fisheye lens	4	$29.00	$116.00
	Rasberrry Pi multi-camera adapter board	1	$50.00	$50.00
	Rasberrry Pi model 3	1	$35.00	$35.00
	5V battery supply	1	$40.00	$40.00
	Accelerometer	1	$8.00	$8.00
	915 MHz transceiver	2	$6.00	$12.00
	200 mm T8 Lead Screw	4	$10.99	$43.96
	1" OD carbon fiber tube 24" long	4	$39.99	$159.96
	Stepper motor	4	19.95	79.8
	Aluminum sheet metal	1	21.48	21.48
	1/4" thick acrylic Sheet 24x48	1	55.37	55.37
	Arduino uno	1	17.6	55.37
	Accelerometer	1	$15.95	$15.95
	Stratologger CF Altimeter	2	$69.95	(Already Own) $0.00
	ARRD	1	$119	$119
	Threaded Rods	3	$6.88	$20.64
	U-bolt	1	$2.66	$2.66
	T-track	2	$33.50	$67.00
	Southco Latch	2	$62.50	$125.00
	Wheels	1	$10.59	$10.59
	Ball bearing	4	$12.96	$51.84
	Subtotal:			**$1,089.7**
Recovery and Avionics	Standoffs	1	$10.99	$10.99
	Iris Ultra 120" Parachute	1	$541.97	(Already own) $0.00
	Stainless Steel Quick Links	14	$1.97	$27.58
	5/8 inch Kevlar Shock Cord (yard)	26.7	$6.35	$169.55
	Black powder	1	$17.95	$17.95
	E-matches	1	$80.25	$80.25
	Shear Pins	1	$1.00	$1.00
	StratoLogger CF	2	$49.46	(Already Own) $0.00
	Classic Elliptical 18" Parachute	1	$57.17	(Already Own) $0.00
	6" Deployment Bag	1	$46.23	$46.23
	18" Nomex Cloth	1	$24.00	$24.00
	Eggfinder TX Transmitter	1	$70.00	(Already own) $0.00
	Eggfinder TX Reciever	1	$55.00	(Already own) $0.00
	4" Deployment Bag	1	$43.00	$43.00
	13" Nomex Cloth	1	$16.00	$16.00
	Subtotal:			**$436.55**

Miscell.	Epoxy Resin	2	$86.71	$173.42
	Epoxy Hardener	2	$45.91	$91.82
	Nuts (box)	1	$5.50	$5.50
	Screws (box)	1	$5.00	$5.00
	Washers	1	$5.00	$5.00
	Wire	1	$13.00	$13.00
	Zip Ties	1	$11.00	$11.00
	3M Electrical Tape	4	$8.00	$32.00
	9V Batteries	2	$14.00	$28.00
	Wood Glue	2	$3.00	$6.00
	Rubber Bands	1	$5.00	$5.00
	Paper Towels	1	$25.00	$25.00
	Battery Connectors	3	$5.00	$15.00
	Shipping			$1,000.00
	Incidentals (replacement tools, hardware, safety equipment)			$1,500.00
	Subtotal:			**$2,915.74**
Travel	Student Hotel Rooms – 4 nights (# rooms)	4	$791.70	$3,166.80
	Mentor Hotel Rooms – 4 nights (# rooms)	3	$1,178.10	$3,534.30
	Van Rentals (# cars)	2	$198.00	$396.00
	Gas (Miles)	1144	$0.60	$686.40
	Subtotal:			**$7,783.50**
Promotion	T-Shirts	40	$14.00	$560.00
	Polos	30	$25.00	$750.00
	Stickers/Pens	500	$0.37	$185.00
	Banner	1	$250.00	$250.00
	Subtotal:			**$1,745.00**
Total Expenses:				**$16,101.84**

Figure 6-1 below shows a budget breakdown for the 2020-2021 competition cycle.

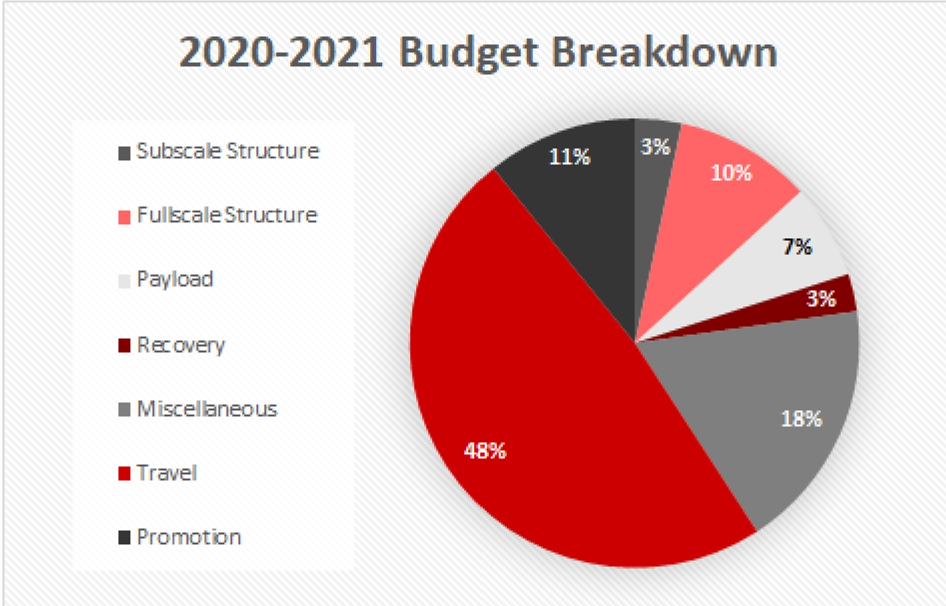

Figure 6-1 2020-2021 Budget Breakdown Chart

6.3 Funding Plan

HPRC receives all funds from multiple NC State University organization and North Carolina Space Grant (NCSG). Below is a breakdown of the team's current funding sources.

The NC State University Student Government Association's Appropriations Committee is responsible for distributing university funds to campus organizations. Each semester the application process consists a proposal, presentation, and an in-person interview. During the 2019-2020 academic year, HPRC received a total of $2,100: $1,030 in the fall semester and $1,070 in the spring semester. A request for $750 has been placed for the current fall semester and $1,350 will be requested in the spring semester, assuming that the Appropriations Committee budget will remain the same.

Educational and Technology Fee is an NC State University fund that allocates funding for academic enhancement through student organizations. Their funding of about $1,500 will primarily pay for the team's faculty advisors travel costs.

Student and mentor travel costs will be covered by NC State's College of Engineering Enhancement Funds. These funds come from a pool of money dedicated to supporting engineering extracurriculars at NC State. The total travel cost for University affiliated attendees comes to an estimated $5,500.

In addition to funding through NC State organizations, North Carolina Space Grant will provide a large amount of monetary support to the club. 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 on the amount awarded, which will likely be the full amount requested. These funds will be available for use starting November 2020.

In the past, HPRC has held sponsorships with Collins Aerospace, Jolly Logic, and more. The team is currently seeking out new sponsorships and reaching out to past sponsors. The team hopes to gain a couple thousand dollars more in funding from various companies.

These totals are listed in Table 6-4 below, which compares the projected costs and incoming grants for the 2020-2021 school year.

Table 6-4 Projected Funding for 2020-2021 Competition

Organization	Fall Semester Amount	Spring Amount	School Year Total
Engineering Technology Fee	-	-	$1,500.00
SGA Appropriations	$750.00	$1,350.00	$2,100.00
Sponsorships	-	-	$2,000
NC Space Grant	-	-	$5,000.00
College of Engineering	-	-	$5,500.00
Total Funding:			**$16,100.00**
Total Expenses:			**$16,101.84**
Difference:			**$1.84**

6.4 Project Timeline

Table 6-5 below shows a tabulated schedule for the 2021 Student Launch Project.

Table 6-5 2021 NASA Student Launch Schedule

Event/Task	Start Date	End Date/Submission
Request for Proposal Released	August 19, 2020	N/A
Proposal	August 19, 2020	September 21, 2020, 3 p.m. CDT
Preliminary Design Review (PDR) Q&A	October 07, 2020	N/A
PDR	October 07, 2020	November 02, 2020, 8 a.m. CST
PDR Team Teleconferences	November 03, 2020	November 22, 2020
Subscale Launch Opportunity	November 21, 2020	November 22, 2020
Critical Design Review (CDR) Q&A	November 23, 2020	N/A
CDR	November 23, 2020	January 04, 2021, 8 a.m. CST
CDR Team Teleconferences	January 07, 2021	January 26, 2021
Flight Readiness Review (FRR) Q&A	January 27, 2021	N/A
FRR	January 27, 2021	March 08, 2021, 8 a.m. CST
Full Scale Launch Opportunity	February 20, 2021	February 21, 2021
FRR Team Teleconferences	March 11, 2021	March 29, 2021

Launch Window for Teams Not Travelling to Launch Week	March 30, 2021	May 02, 2021
Post-Launch Assessment Review (PLAR) – Teams Not Travelling to Launch Week	March 30, 2021	14 Days After Launch
Launch Week Q&A	March 31, 2021	N/A
Team Travel to Huntsville, AL (if applicable)	April 07, 2021	N/A
Launch Readiness Review (LRR)	April 07, 2021	April 08, 2021
Launch Week Activities	April 08, 2021	April 09, 2021
Launch Day	April 10, 2021	N/A
Backup Launch Day	April 11, 2021	N/A
Post-Launch Assessment Review (PLAR) – Teams Travelling to Launch Week	April 12, 2021	April 27, 2021, 8 a.m. CDT

Figure 6-2 below show this same schedule in Program Evaluation and Review Technique (PERT), with additional details for each step.

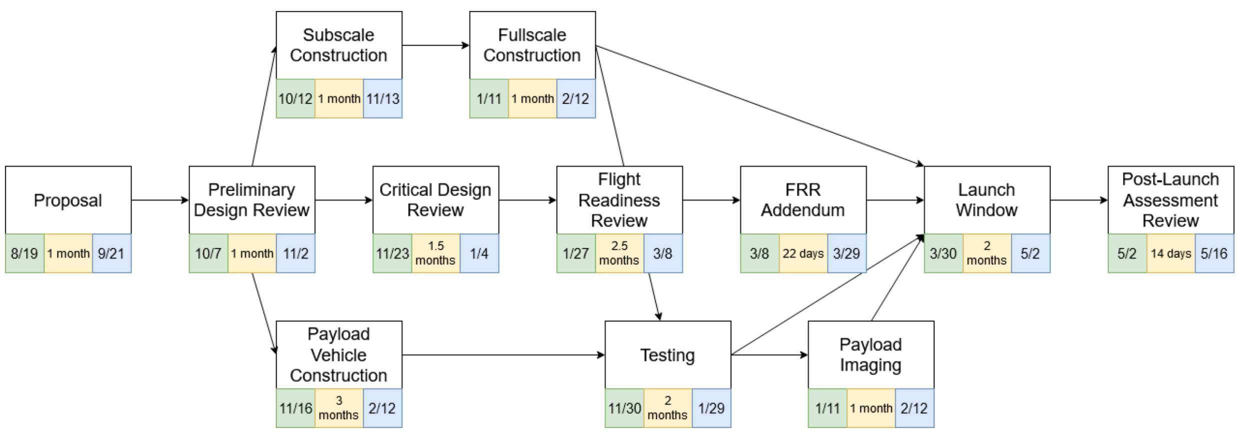

Figure 6-2 2021 Student Launch Project Timeline - PERT

Figure 6-3 below is the construction schedule for the subscale launch vehicle, showing the planned tasks to be completed on each day of the construction process. A similar plan will be developed for the full scale launch vehicle and payload construction.

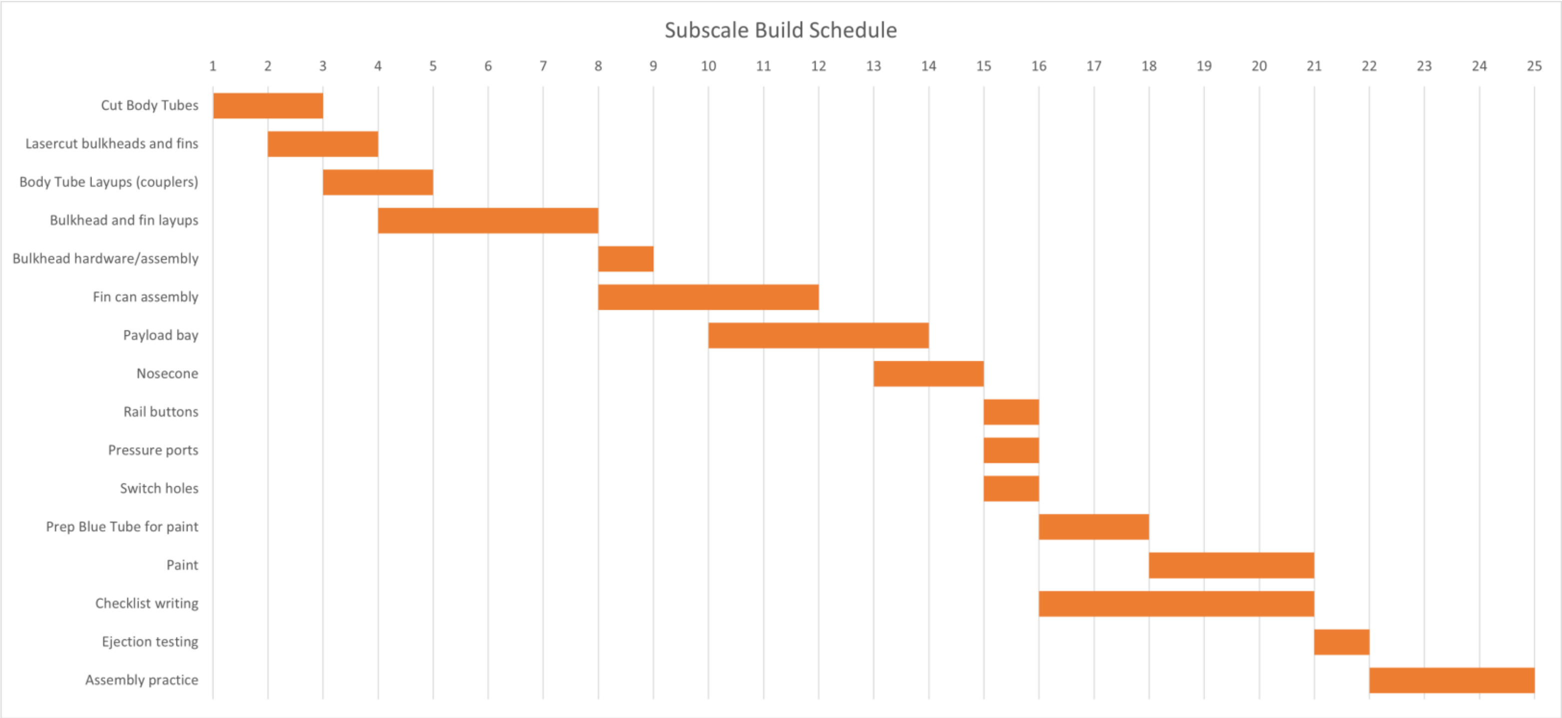

Figure 6-3 Subscale Construction Schedule

Figure 6-4 below shows the same information in Critical Path Analysis (CPA) form, which will also be applicable to the launch vehicle. The critical path, or the set of tasks that have the highest importance of being completed on time, are highlighted in red.

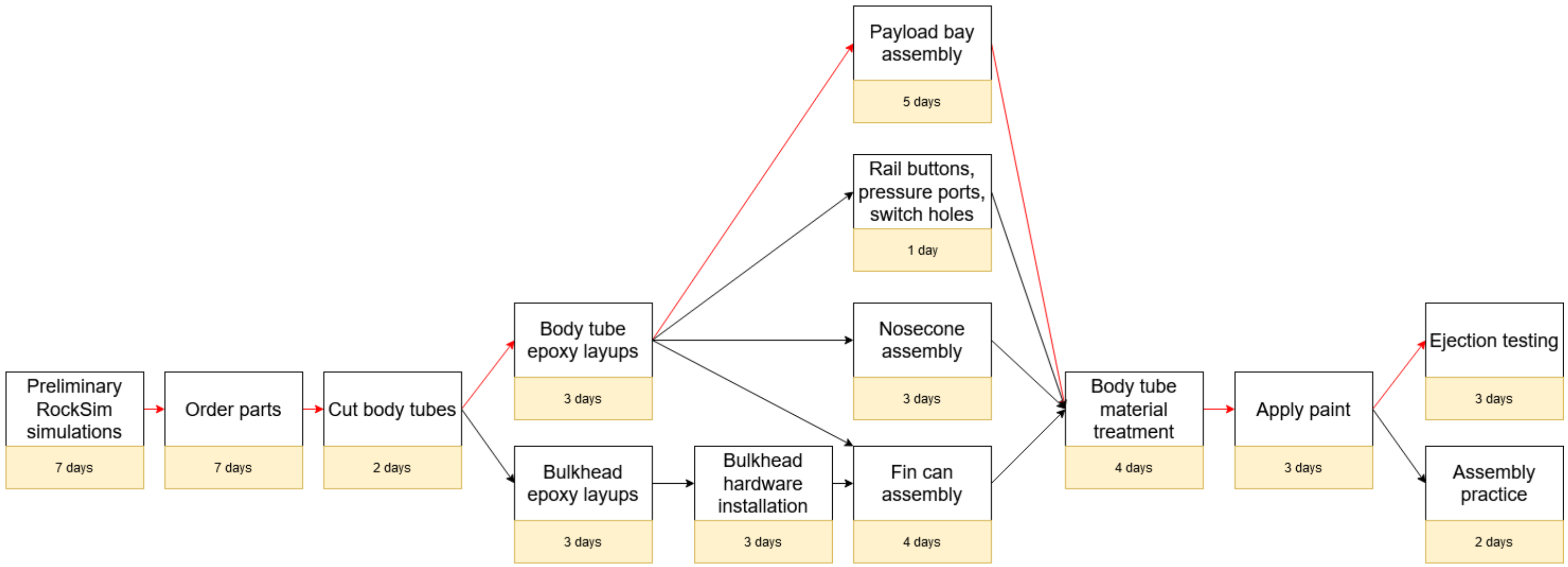

Figure 6-4 Launch Vehicle Construction Critical Path Analysis

Figure 6-5 below shows a similar CPA for the construction of the main competition payload.

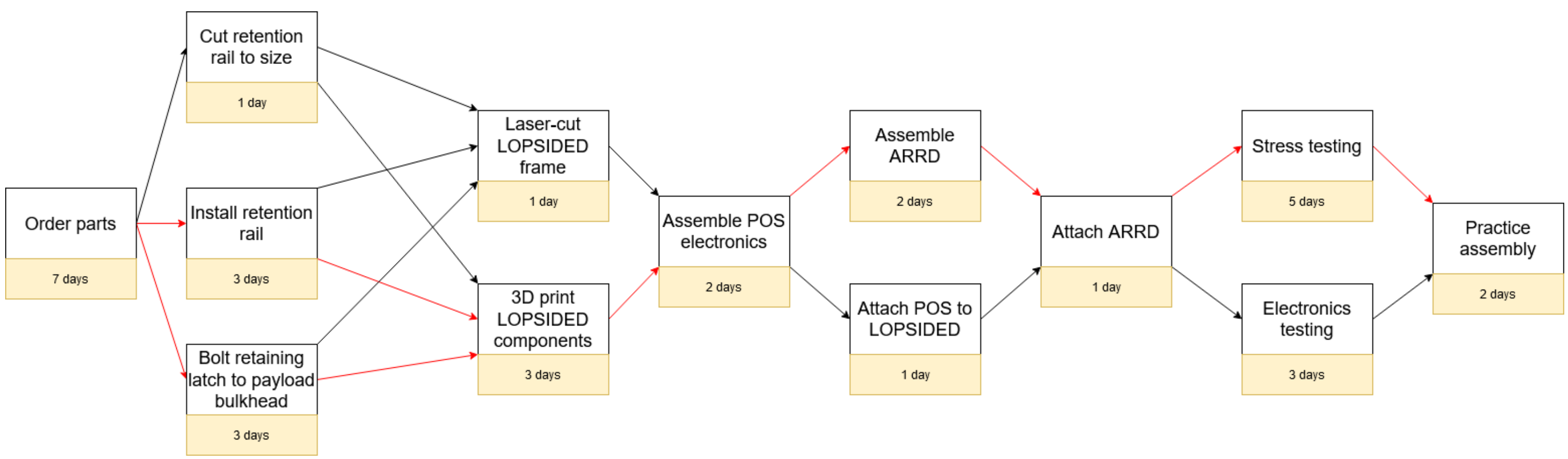

Figure 6-5 Payload Construction Critical Path Analysis