NC STATE UNIVERSITY

Tacho Lycos
2020 NASA Student Launch
Preliminary Design Review

High-Powered Rocketry Club at NC State University
911 Oval Drive
Raleigh, NC 27695

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Common Abbreviations & Nomenclature

AGL	=	above ground level
APCP	=	ammonium perchlorate composite propellant
AV	=	Avionics
BP	=	black powder
CDR	=	Critical Design Review
CG	=	center of gravity
CP	=	center of pressure
FAA	=	Federal Aviation Administration
FMEA	=	failure modes and effects analysis
FN	=	foreign national
FRR	=	Flight Readiness Review
HPR	=	High Power Rocketry
HPRC	=	High-Powered Rocketry Club
L3CC	=	Level 3 Certification Committee (NAR)
LCO	=	Launch Control Officer
LRR	=	Launch Readiness Review
MAE	=	Mechanical & Aerospace Engineering Department
MSDS	=	Material Safety Data Sheet
MSFC	=	Marshall Space Flight Center
NAR	=	National Association of Rocketry
NASA	=	National Aeronautics and Space Administration
NCSU	=	North Carolina State University
NFPA	=	National Fire Protection Association
PDR	=	Preliminary Design Review
PLAR	=	Post-Launch Assessment Review
PPE	=	personal protective equipment
RSO	=	Range Safety Officer
SL	=	Student Launch
SLI	=	Student Launch Initiative
SICCU	=	Simulated Ice Collection and Containment Unit
STEM	=	Science, Technology, Engineering, and Mathematics
TAP	=	Technical Advisory Panel (TRA)
TBD	=	To Be Determined
TDR	=	Team Derived Requirement
TRA	=	Tripoli Rocketry Association
UAV	=	Unmanned Aerial Vehicle

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1.1 Team Summary

1.1.1 Team Name and Mailing Address

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1.2 Launch Vehicle Summary

1.2.1 Size and Mass

The current leading launch vehicle design is 107.5 inches long with a diameter of 6 inches. The mass of the leading launch vehicle design with the leading motor option is 44.7 lbs.

1.2.2 Preliminary Motor Choice

The current leading motor option is the Aerotech L1520T-PS. See section 3.1.5.

1.2.3 Official Target Altitude

The official target altitude is 4420 feet. See section 3.3.1.

1.2.4 Recovery System

The current leading design uses a duel-deployment recovery system using two PerfectFlite StratologgerCF altimeters, a Fruity Chutes 24 in Compact Elliptical drogue parachute deployed at apogee, and a Fruity Chutes 120 in Iris Ultra Compact main parachute deployed at 500 feet altitude.

1.3 Payload Summary

1.3.1 Bilateral Uptake Rover for Regolith Ice Transport Operations (BURRITO)

The payload has been designated BURRITO, an acronym that stands for Bilateral Uptake Rover for Regolith Ice Transport Operations. The BURRITO rover will deploy from the payload bay of the launch vehicle on landing. Deployment will be completed by means of a pusher plate moving along two threaded rods, which will push the rover out of the payload bay. The rover itself consists of two parallel drive wheels powered by electric motors and mounted to a plywood chassis, as well as a stabilization wheel that deploys following removal from the payload bay. The rover will be radio-controlled by an operator who will drive it away from the payload bay toward a simulated ice recovery site. There, BURRITO will deploy its SICCU(Simulated Ice Collection and Containment Unit) system, a set of two scoops that will pick up a 10 mL sample of simulated ice. Once the sample has been scooped up, the operator will drive the rover away from the recovery site by 10 linear feet.

2. Changes Since Proposal

Table 2-1, below lists all changes made to the full scale launch vehicle since proposal submission.

Table 2-1 Launch Vehicle Changes Since Proposal

Description of Change	Reason for Change
Changed the nose cone from a 5.5:1 Von Karman to a 5:1 Ogive	The main factor behind this change was the cost. The Von Karman fiberglass nose cone is more than 6 times as expensive as a plastic Ogive for the subscale launch vehicle and the difference between the aerodynamics of the nosecones is minimal. Thus, it was determined that the subscale would use a 5:1 Ogive which required the team to change the nose cone on the full scale as well.
Primary motor choice changed from the AeroTech L1390G to the L1520T	The L1520T motor lowers the apogee of the launch vehicle and allows the recovery team to meet the 90 second decent timeline.
Engine Bay Airframe was extended by 1 inch	It was discovered that the RockSim software only includes the length of the grains when loading in a motor for launch simulations. This misunderstanding resulted in a motor tube that was 1 inch shorter than the actual motor tube. Fixing this issue also required the airframe to be extended by an inch.
The leading main parachute is now a 120-inch Iris UltraCompact	Changes in nosecone section mass required a slower descent rate in order to comply with the kinetic energy at landing requirements. With an increased total mass, the 120-inch Iris UltraCompact descends fast enough to meet all descent requirements at safe kinetic energy values.
The main parachute is now deployed using a piston ejection system	Shielding of payload electronics from black powder detonations required a more robust solution, for which the piston ejection system was proposed, simultaneously shielding recovery equipment and the payload.
Secondary black powder charges are now 0.2 grams larger than the primary charge	After performing preliminary black powder sizing, an increase of 0.5 grams for the secondary charge was determined to be too large, especially for the drogue charge, as this would double the mass of powder used.
The leading tracking device is now the QRP Labs LightAPRS GPS tracker	The team desired a more robust tracking solution, for which the LightAPRS has been selected due to its customizability and interface capabilities. The LightAPRS can provide GPS and telemetry data, compared to the BeeLine's pure directional tracking.

Table 2-2, below lists all changes made to the payload since proposal submission.

Table 2-2 Payload Changes Since Proposal

Description of Change	Reason for Change
The Sample Collection Method was changed from two drums rotating in opposite directions to two hemicylindrical scoops with arms.	The scoops needed to be changed from a drum because of concerns about leakage and clearance.
Wheel treads changed from standard rubber treads to plaction treads.	Plaction treads conform better to the surface they sit on, generating more traction.
Power Transmission: The gear box has been changed from a three-gear system to a five-gear system.	Radial supports for rover were deemed necessary, which were not possible with the three-gear system, which required the main plate to have a circular geometry. The three-gear system also had a less than ideal gear ratio which would drop than transmittable torque by a factor of 3.75. The five-gear system only drops the torque output by a factor of 1.5.
Retention: Changed from solenoid latch to gear rack locking mechanism	Increased forces during main deployment require high rated locking mechanisms. High strength solenoid latches take up too much surface area
Payload supports: Addition of radial supports along with the aft centering ring.	The radial supports will help restrict radial motion during flight and will keep the rover centered to prevent any interference during deployment
Landing detection redundancy: Elimination of altimeter as a third redundancy	The deployment mechanism will communicate with the SLI team before deploying, which already acts as a redundancy for landing detection. Keeping the altimeter as a third redundancy will adds unnecessary potential for failure.

Table 2-3, below, lists all changes made to the project plan since proposal submission.

Table 2-3 Project Plan Changes Since Proposal

Description of Change	Reason for Change
Team Derived Requirement Verification Matrix added.	Required per NASA guidelines

3. Vehicle Criteria

3.1 Selection, Design, and Rationale of Launch Vehicle

3.1.1 Launch Vehicle Mission Statement

The mission of the launch vehicle is to safely reach the declared target apogee and return to the ground in a reusable condition. The mission of the launch vehicle is also to support the mission of the payload by delivering it to the simulated lunar ice sample collection area.

3.1.2 Launch Vehicle Mission Success Criteria

Mission success is first defined as compliance with both the NASA SL requirements in Table 6-2 and the team derived requirements in Table 6-3. Success is further defined as follows in Table 3-1.

Table 3-1 Launch Vehicle Mission Success Criteria

Level of Success	Definition
Complete Success	Launch vehicle recoverable Nominal launch vehicle takeoff and descent Rover is undamaged and fully functional following the flight of the launch vehicle Launch operations can be repeated the same day
Partial Success	Launch vehicle repairable Successful launch vehicle takeoff and descent Rover is repairable following the flight of the launch vehicle
Partial Failure	Launch vehicle repairable Successful launch vehicle takeoff and unsuccessful descent Rover is repairable following the flight of the launch vehicle
Complete Failure	Launch vehicle unrecoverable Rover unrecoverable

3.1.3 Launch Vehicle Alternative Designs

All potential designs of the launch vehicle will consist of five modules. Starting from the aft end of the launch vehicle. The components are the fin can, the avionics bay, the main chute bay, the payload bay, and the nose cone. Additionally, the planned diameter of the body tube is 6 inches.

3.1.3.1 Material

Choosing the correct airframe material is extremely important to ensure the success of the launch vehicle. If the airframe were to fail, then all other aspects of the launch vehicle would fail with it. The airframe material not only has to endure exterior aerodynamic forces, but also must withstand the internal forces of the motor and other attached components. From the proposal: two main materials alternatives were narrowed down, Blue Tube 2.0 and G12 Fiberglass. These two materials proved to have the most promising characteristics.

There are few simplified equations used when estimating the aerodynamic forces on the launch vehicle. Mainly:

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

Where F_D is the drag force, ρ is the density of the air, V is the velocity of the launch vehicle, C_D is the coefficient of drag, and A is the frontal area of the launch vehicle. The V and C_D are pulled from RockSim simulations. Note that this equation assumes a 0° angle of attack.

Additionally, non-aerodynamic forces need to be accounted for. This comes in the form of inertial forces, which are calculated as follows:

$$F_I = ma_R$$

Where m is the mass of the launch vehicle, and a_R is the peak acceleration of the launch vehicle. These two forces combined give the largest compressive force on the launch vehicle:

$$F_C = F_D + F_I$$

These values will be calculated individually for Blue Tube 2.0 and G12 Fiberglass. This is because the materials have different weights, which then leads to different maximum accelerations and maximum velocities.

All values, along with their sources are listed below:

Table 3-2 Force Calculations for G12 Fiberglass

	Value	Source
m	44.4 lbm	RockSim
C_D	0.355	RockSim
A	0.207 ft^2	RockSim
a_R	289.57 ft/s^2	RockSim
V	546.134 ft/s	RockSim
ρ	0.0765 lbm/ft^3	Constant
F_D	**26 lbf**	Formula
FI	**399.28 lbf**	Formula
F_C	**425 lbf**	Formula

Table 3-3 Force Calculations for Blue Tube

	Value	Source
m	43.2 Ibm	RockSim
C_D	1.5	RockSim
A	0.207 ft^2	RockSim
a_R	289.56 ft/s^2	RockSim
V	561.9 ft/s	RockSim
ρ	.0765 ibm/ft^3	Constant
F_D	**77.64 Ibf**	**Formula**
FI	**388.48 Ibf**	**Formula**

3.1.3.1(a) Blue Tube 2.0

Blue Tube is an airframe material produced by Always Ready Rocketry. Blue Tube is a vulcanized cellulose fiber that is a stronger alternative to cellulose fiber. It features high impact and shatter resistance compared to phenolic. According to Always Ready Rocketry, the ultimate compressive load of Blue Tube is about 3000 Ibf. Current predictions land the maximum in-flight compressive load to be 466.12 Ibf. Blue Tube is more than capable of enduring in-flight loads, as well as having a factor of safety of 6.4. With this factor of safety, Blue Tube is an enticing option as it is significantly cheaper and lighter than more durable alternatives such as fiberglass and carbon fiber. However, these advantages also come with drawbacks. Since Blue Tube is still a paper derivative, it is susceptible to water damage without proper treatment. Additionally, the manufacturing of Blue Tube leaves it with winding grooves that run down the length of the body tube. These will need to be filled and smoothed down to allow for better airflow over the launch vehicle during flight.

3.1.3.1(b) G12 Fiberglass

G12 fiberglass is a fiber filament wound airframe. The angles at which the filaments are wound typically range between 30° and 45°. G12 fiberglass is an extremely strong material that can take significant compressive force as well as impact forces. Additionally, it is significantly stronger than Blue Tube and will be able to easily handle the 425 Ibf of compressive forces predicted during flight. The main drawback of fiberglass is that it is significantly heavier than Blue Tube and significantly more expensive. G12 Fiberglass has a density of 1.07 oz/in^3 while Blue Tube has a density of 0.751 oz/in^3, and G12 fiberglass body tubes cost, on average, $2 more per inch than Blue Tube body tubes. However, there are distinct advantages to utilizing a non-paper-based body tube, the most significant of which is the moisture resistance of G12 fiberglass. Using fiberglass would reduce the potential for failure when operating under humid or wet conditions, as these conditions would not compromise the structural integrity of the material. Additionally, fiberglass will not fray or unravel like Blue Tube can do, and thus is more reliable material to use during construction.

3.1.3.2 Nosecone Bulkhead

3.1.3.2(a) Removable

The nosecone bulkhead is required to mount any required ballast to fine-tune the stability margin. Additionally, it serves as one of the anchor points for the forward recovery harness which attaches to the main parachute. The removable bulkhead will be constructed out of aircraft-grade birch plywood. A removable bulkhead configuration consists of a tightly fitting bulkhead secured to the nosecone structure by way of bolted connections. This allows for the removal of the bulkhead for the addition or removal of ballast as needed during the design and testing process. Bolted designs do lead to stress concentrations in the airframe material and are therefore slightly weaker than properly filleted epoxied joints. This is not considered to be a significant factor in the selection of bulkhead attachment provided that the joint is designed to minimize the risk of tear-through during the main parachute opening shock and sufficiently strong bolts are used.

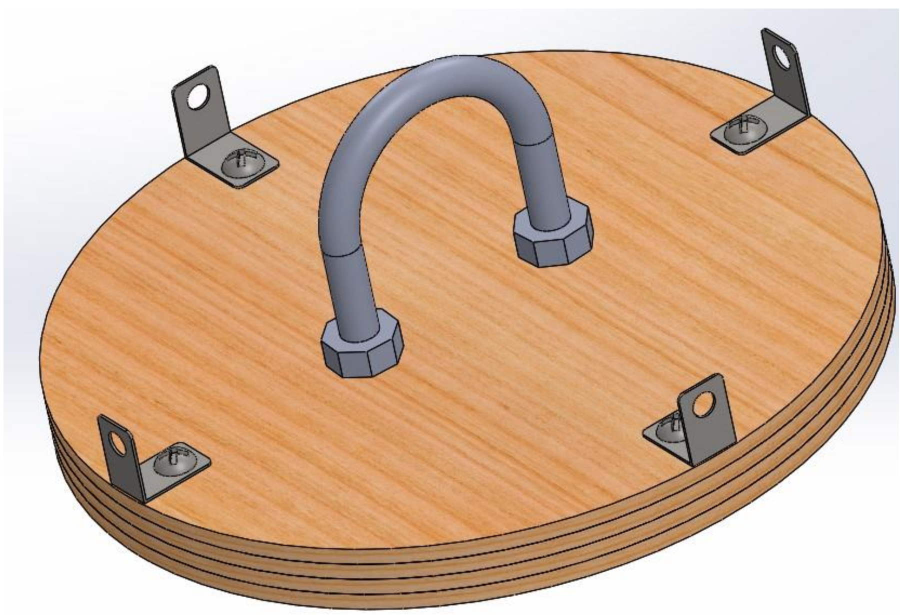


Figure 3-1 Removable Bulkhead

Figure 3-2 Nosecone with Removable Bulkhead

3.1.3.2(b) Non-Removable Bulkhead

A non-removable bulkhead is constructed of aircraft-grade birch plywood, that is epoxied into the nosecone and cannot be removed without cutting or chemical dissolving of the epoxy material. This method supplies a strong joint and does not require drilling through the airframe of the launch vehicle. The primary disadvantage of this approach is the lack of ability to remove the bulkhead in the event that ballast adjustments need to be performed.

3.1.3.3 Nosecone Selection

Commercial availability of large diameter fiberglass nosecones is limited. With this in mind, the launch vehicle is constrained to three alternative nose cone designs.

3.1.3.3(a) Conical

Conical nosecones are extremely popular among hobby rocketeers. When determining the nosecone for the full-scale launch vehicle, the team opted to not select the conical nose cone as the flight of the launch vehicle would be entirely subsonic per NASA 2.22.7 and thus it would not benefit from a conical nosecone.

Figure 3-3 Conical Nosecone

3.1.3.3(b) Von Karman

The Von Karman was determined to be the ideal nosecone for the launch vehicle. This type of geometry is a special case ogive that is mathematically derived have the minimal amount of drag per unit length and diameter. This geometry was initially selected to be the nosecone of choice for the full-scale launch vehicle. This type of geometry is difficult to obtain in the commercial market, however some suppliers offer it.

Figure 3-4 Von Karman Nosecone

3.1.3.3(c) Ogive

Ogive nosecones are another popular choice among hobbyists. The ogive geometry is a blanket term as several types of ogive geometries exist; however, most commercial sellers tend to sell tangent ogive nose cones. The ogive was initially the second consideration for the full-scale launch vehicle. However, due to budgeting limitations with the subscale launch vehicle, the team decided to purchase a 5:1 ogive for the subscale. Since the subscale must be an aerodynamically similar but scaled version of the full scale, the nose cone on the full-scale launch vehicle thus had to change to a 5:1 ogive.

Figure 3-5 Ogive Nosecone

3.1.3.4 Anchor Selection

3.1.3.4(a) U-bolt

U-bolts provide a secure connection between bulkheads and quick links. U-bolts require two points of contact for a proper mount, which slows construction and reduces time to uninstall. However, this additional time is fairly insignificant. The two points of contact provide two points of failure for the U-bolt. Additionally, the total force is split evenly between the points, reducing the stress on each point. However, this increased reliability comes with an increased profile and weight of the U-Bolt.

3.1.3.4(b) Eye-Bolt

Eye-bolts provide a quick connection between bulkheads and quick links. Eye-bolts only require a single point of contact to mount; this allows for easy installation and easy removal is needed. Additionally, an eye-bolt has a low profile and low mass, so it can be installed to heavier and more compact sections without causing major interference. The drawback to eye-bolts is that they only feature a single point of contact to the bulkhead. This translates into a single point of failure, so if this single connection fails then it leads to a failure of a major portion of the launch vehicle recovery system.

3.1.3.5 Fin Configuration

3.1.3.5(a) Three-Fin Configuration

Three-fin configuration is a popular choice among hobby rocketeers for several reasons, primarily since three-fin configurations are largely a simpler fin design to manufacture. By having to cut fewer fin slots, there exists a lower chance of making errors in the location or size of the cuts. Additionally, the fins typically have rectangular extensions that extend from the chord root; this extension is secured between the body tube and the motor tube of the launch vehicle using epoxy fillets. Clearance in this area of the launch vehicle can be fairly tight, which may make the assembly of the fin difficult. With fewer fins comes the benefit of using fewer fillets, which makes the creation of the fin-can simpler. Additionally, having fewer fins results in less weight on the aft end of the launch vehicle. A lower weight in turn helps to keep the center of gravity closer to the tip of the nosecone which increases the stability margin of the launch vehicle.

A disadvantage of this design is the non-axisymmetric aerodynamic loading profile experienced by the aft end of the launch vehicle. During a large crosswind gust, the angle of attack that the fin experiences could vary largely from what fin would experience if freestream was the only flow component acting on the fin. This gust condition combined with the non-axisymmetric loading profile could result in the vehicle rolling to an equilibrium state and a change in apogee may result from this.

3.1.3.5(b) Four-Fin Configuration

Issues of roll instability can be remedied by utilizing a four-fin configuration which will implement two axes of symmetry. Additional fins will also aid in shifting the center of pressure further aft to increase stability. When compared to three-fins, the four-fin configuration allows for a smaller planform area to achieve the same center of pressure location.

It must also be noted that fins that are too small may cause the launch vehicle to weathercock immediately off the launch rail. A four-fin configuration was used on this launch vehicle and the static margin was in excess of 2.0, however, due to the small planform area of the fins the launch vehicle experienced severe weathercocking once it left the launch rail. Fortunately, the launch vehicle managed to have a successful flight. Because of this experience, the team this year is making an effort to have relatively large fins to ensure that this won't be an issue.

3.1.3.6 Avionics Bay

3.1.3.6(a) Integrated

The integrated AV Bay design, consists of a length of body tube containing the AV Bay, drogue parachute, and main parachute. The AV bay sled is mounted on two threaded rods, which run between two bulkheads at the ends of the AV bay. One bulkhead is epoxied to the airframe, while the other is bolted onto the threaded rods, allowing it to be removed. Additionally, a hatch is cut in the side of the airframe, allowing access to the AV bay while the AV sled is installed in the airframe. The hatch is secured to each bulkhead at two points, one at each corner of the hatch. Two blast caps are secured to the outside of each bulkhead, along with a U-Bolt used to secure the AV bay to the shock cord of the associated parachute.

This design allows for easy access to the AV bay even when the launch vehicle is fully assembled, allowing for any last-minute adjustments to be performed easily. However, the gap in the airframe for the hatch causes airflow disruptions and load transfer disruptions in the fiberglass airframe, reducing its strength as compared to a solid fiberglass airframe. Additionally, loading black powder into the blast caps as well as attaching the shock cord to the AV bay requires reaching into long sections of body tube. This adds difficulty to the assembly process and increases the chances for an error to be made.

3.1.3.6(b) Modular

The modular AV Bay design, shown in Figure 3-6, consists of a length of coupler capped by a bulkhead on both ends. The inner layers of each bulkhead are at the inner diameter of the coupler, while the outer layers are at the outer diameter of the coupler. This allows the bulkheads to function as plugs to isolate the inside of the AV bay from the pressure and temperature changes caused by the black powder charges. A thin section of body tube near the center of the coupler section gives space for pressure sampling ports and access to the switches that will arm the altimeters. Two threaded rods run between the two bulkheads, on which the AV sled is mounted. The outside of each bulkhead has two blast caps secured to it, as well as a U-Bolt used to connect to the shock cord of the associated parachute.

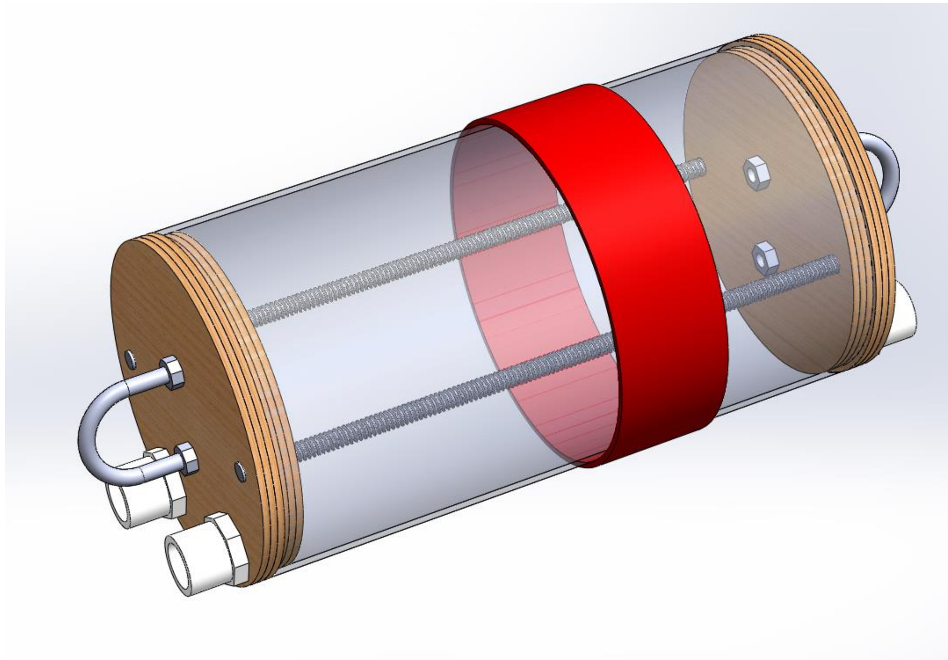


Figure 3-6 Modular AV Bay

This design makes the blast caps much more accessible for loading black powder charges during the assembly process. Additionally, it allows for AV bay assembly to occur separately from the rest of the launch vehicle. This reduces assembly time by allowing for multiple assembly tasks to be performed in parallel. However, this design makes the AV sled inaccessible once the launch vehicle has been assembled. Any adjustments to the altimeters will require disassembly of the launch vehicle.

3.1.4 Launch Vehicle Leading Design

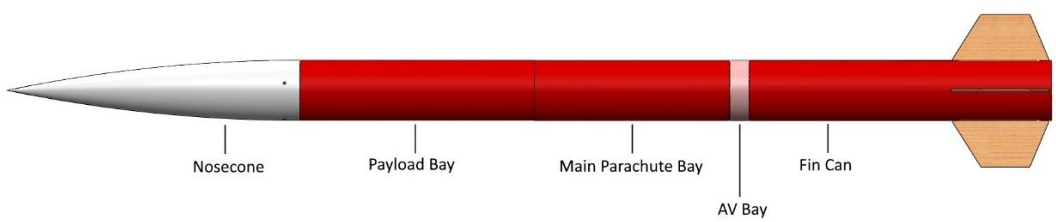


Figure 3-7 Leading Vehicle Layout

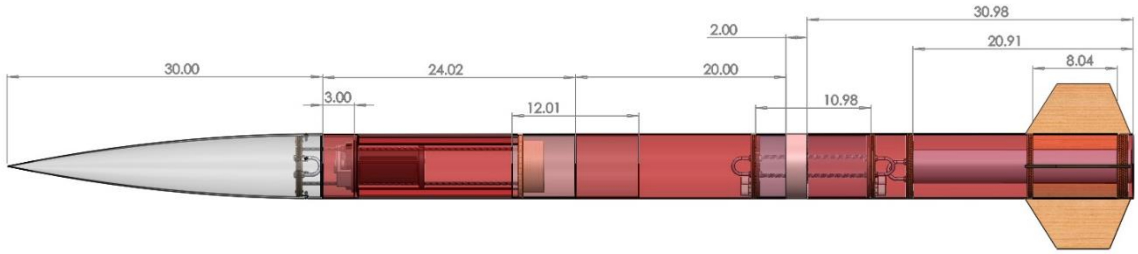


Figure 3-8 Leading Vehicle Dimensions

3.1.4.1 Material Selection

The leading choice for airframe body material is G12 fiberglass. It is critical to have an airframe that is resistant to moisture, especially due to unpredictable launch field conditions in North Carolina. While Blue Tube can be sealed against moisture, this adds time to the fabrication process and is a fabrication step no members have performed before. The sealing process results in a high chance for errors, which is not acceptable for this part of fabrication. Additionally, unlike Blue Tube, there is no risk of material delamination or fraying during fabrication or flight. Fiberglass is heavier and more expensive than Blue Tube, however its increased durability is worth the increased weight and cost. With fiberglass, the leading configuration has a mass of 44.7 lbs, which will not cause any issues with reaching the target apogee or with recovery system requirements.

3.1.4.2 Nose Cone

The nose cone design is shown in Figure 3-9 with a bulkhead and U-bolt mounted in it. The bulkhead will be attached to a shock cord which will be routed through the payload bay and attached to the main parachute. The bulkhead will be recessed 2 inches into the nose cone from the edge of the coupler.

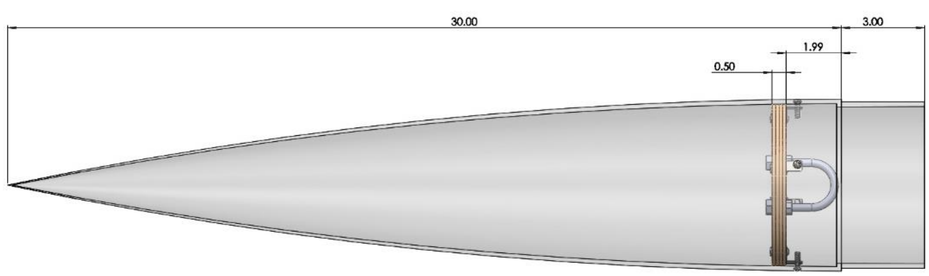


Figure 3-9 Nosecone Dimensions

This design will make the fabrication more difficult as it will require the bulkhead diameter to sit flushed with the contours of the inner wall of the nose cone. However, the added benefits to payload accessibility are deemed to be worth the added manufacturing time.

The nose cone geometry is a 5:1 ogive. This geometry was chosen out of commercial availability as well as pricing. In addition to this, the large inner diameter of this nose cone provides a larger interior space in which some payload deployment electronics may be stored. Current estimates on this deployment device predict that this void space will be utilized once it is integrated into the launch vehicle.

The estimated mass of the nose cone is predicted to be 6.08 Ibs.

3.1.4.3 Nose Cone Bulkhead

A removable bulkhead design was selected in the event of a slightly over/under stable final configuration. The removable nature of the connection is not expected to compromise the structural integrity of the vehicle during launch or recovery. This bulkhead shall be designed to withstand the opening shock of the parachute plus a sufficient factor of safety to account for unexpected forces. The bulkhead will be connected to the nose cone through four L-brackets that will sit equidistant around the perimeter of the bulkhead. Holes will be drilled into the nose cone to match the positions of the L-brackets. The bulkhead will then be supported by bolts inserted through the nose cone and the L-brackets.

3.1.4.4 Anchoring

For all anchoring points, a U-Bolt will be used. The additional weight is well worth the extra reliability gained from having two points of failure. Additionally, analysis has shown that this extra weight will not have any major effects on the stability of the launch vehicle.

3.1.4.5 Payload Bay

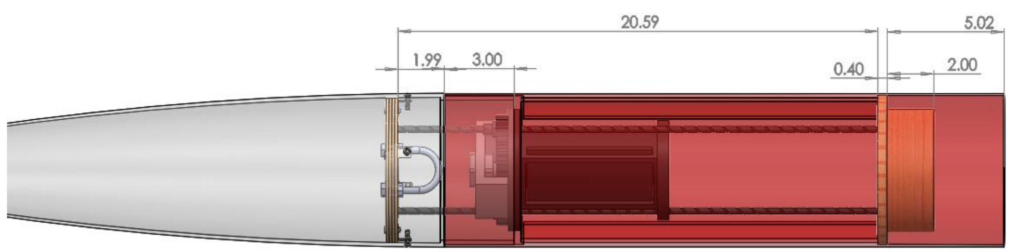


Figure 3-10 Payload Dimensions

Our chosen payload location will be forward as opposed to aft. By having the payload be in the forward section of the launch vehicle, the mass of the motor will be offset keeping the CG will forward of the CP, increasing the stability of the launch vehicle. Along with this, there will be more space available in the forward section of the launch vehicle: allowing for more flexibility with the payload design.

Additionally, having the payload forward would separate it from the fin can. The forward location is significant as these are the two heaviest sections of the launch vehicle. This leads to the fact that they both have the highest kinetic energy during flight. During recovery, it is beneficial to have these sections separated so that a single parachute does not need to compensate for their weight. This separation leads to a lower kinetic energy per parachute and allows for a faster descent velocity, as the descent velocities are restricted by requirement NASA 3.3.

Without the rover, the payload bay is estimated to weigh 10.6 lbs.

3.1.4.6 Main Parachute Bay

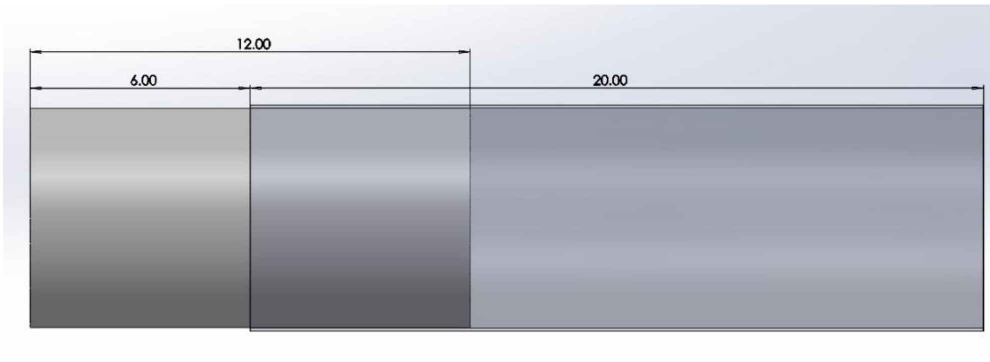


Figure 3-11 Leading Parachute Bay Dimensions

The main parachute will be located between the payload bay in the forward section and the AV bay in the midsection. The parachute bay will be screwed into the AV bay on the short end of the AV bay coupler after all avionics wiring has been completed on the field but before fully assembling the vehicle in launch day configuration. When fully assembled, there will be about 20 inches between the AV bay bulkhead and the payload bay centering ring. This length was determined to be necessary based on recovery system needs. The recovery system will be using a piston ejection system as detailed in section 3.2.2.9. The leading main parachute has a packing volume of 5” D x 7” L. Along with most of the shock cord in the system, the recovery equipment can be estimated as occupying 10 of the 20 inches allocated to it. The remaining 10 inches are left open so that the piston may rest at a distance away from the black powder that will reduce wear and damage over multiple launches, as well as producing a more even pressure curve over time, preventing damage to the airframe.

Without the main parachute, the parachute bay is estimated to weigh 3.5 Ibs.

3.1.4.7 AV Bay

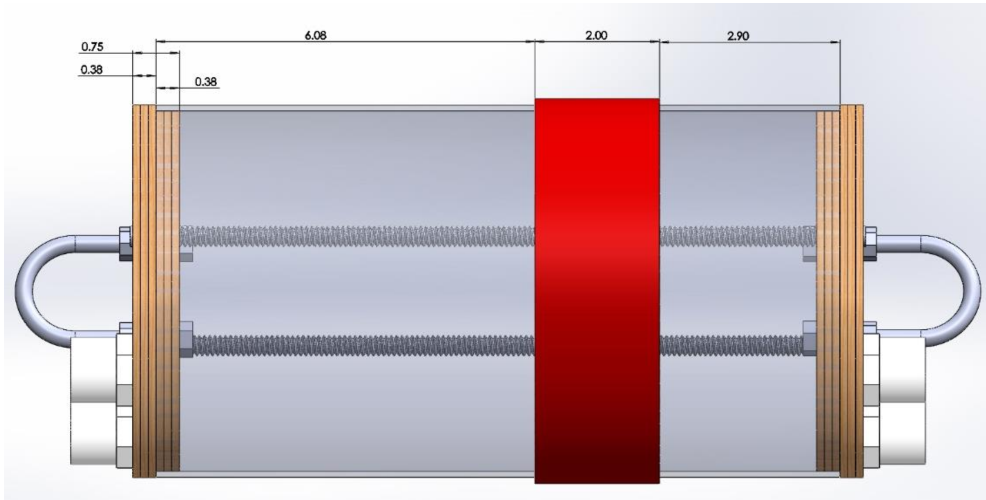


Figure 3-12 Leading AV Bay Dimensions

The final AV bay design will be a modular AV bay. This is the most advantageous design and will be the easiest for the team to implement. Being able to easily remove the AV sled will be invaluable, as it will allow for easy construction and also allow for easy access to the AV sled on launch days. Being able to have access is extremely important as the electronics on the AV sled are constantly being tweaked and improved throughout the launch vehicle's construction.

Without the avionics sled and black powder charges, the AV bay is estimated to weigh 6 lbs.

3.1.4.8 Fin Can

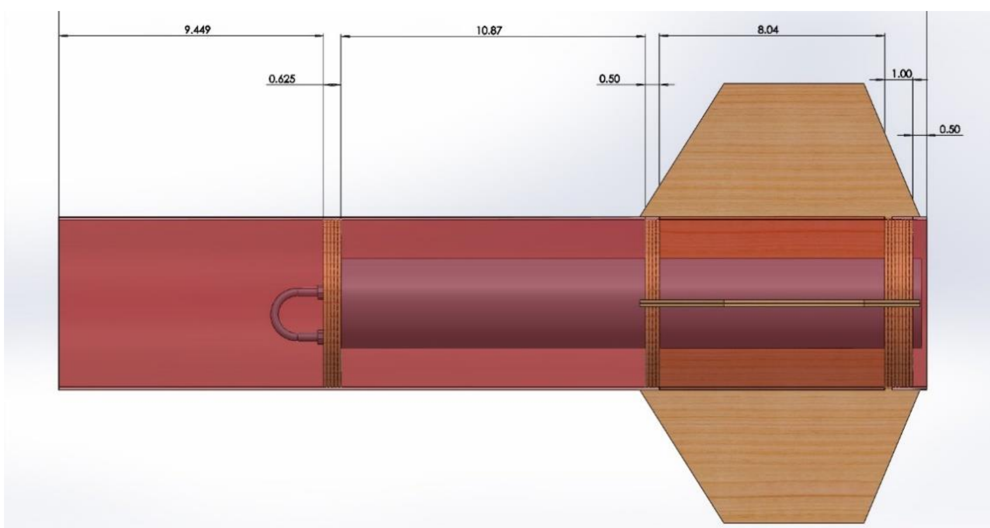


Figure 3-13 Leading Fin Can Dimensions

The fin can supports both the fins and the motor tube. The motor tube is held in place by three bulkheads. The majority of the thrust loads will be supported by the forward-most bulkhead and aft-most centering rings. The primary purpose of the center centering ring is for motor tube and fin alignment. The forward-most bulkhead will have two centering rings attached its aft end. This assists motor tube alignment as the motor tube can fit inside of these centering rings. This bulkhead will also feature a U-bolt for anchoring a quick link. The aft-most centering rings are designed to be thicker than the other bulkheads as this centering ring will carry the majority of the thrust load, after being split between itself and the forward-most bulkhead. The drogue chute will be housed in the forward section of the fin can. It will be attached to the U-bolt of the forward-most bulkhead.

Without the motor tube, the estimated mass of this assembly is 7.442 lbs.

3.1.4.9 Fin Configuration

The team will use a four-fin configuration for the launch vehicle. This design is chosen such that there will be 2 axes of symmetry for the aerodynamics loads that the fins will experience. Additionally, the fins will have a trapezoidal planform area. The leading-edge sweep of this geometry allows the center of pressure of the launch vehicle to be placed further aft. The large trailing edge sweep helps mitigate severe damage to the fins if the launch vehicle touches down fin can first. The fins will also be made of two separate 0.125" thick aircraft grade birch plywood cutouts that will be sandwiched together using a thin layer of epoxy resin, resulting in a fin thickness of 0.25". This is done to mitigate any fin flutter that these fins may experience during the launch vehicles ascent.

3.1.4.10 Bulkhead Sizing

Presently, all bulkhead sizing on the launch vehicle are approximations. This is to avoid performing lengthy calculations on a launch vehicle whose properties are currently fluid. Once the design is solidified, then hand calculations and FEA analysis will be performed to obtain the optimal bulkhead thickness for each component of the launch vehicle. However, the current estimations are based on past successful designs and will still be mostly accurate.

3.1.4.10(a) Hand Calculations

When hand calculations are performed, they will be based off formulation derived in Roarks Formula for Stress and Strain[1]. This book contains a variety of formulas for calculating deflections, stresses, and strains over a flat plate. While there is not an equation listed specifically for forces applied over 2 separate points, as would be created by a U Bolt, this can be approximated by a singular circular force over the center of where the U-bolt would be placed. This model is helped by the fact that a flange will be placed on either side of the U-bolt, which will help to more evenly

[1] Budynas, Richard G. *Roarks Formulas for Stress and Strain*. McGraw-Hill, 2002.

distribute the force over an area. The following equation is used to find the moments of bulkheads with a centered U-Bolt:

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

Where, v is the Poisson's ratio of birch plywood, W is the applied load, a is the distance from the edge of the bulkhead to the loading area, r is the radius of the bulkhead, and r_0' is a function of the radius of the loading circle, r_o. The function is represented here:

$$r_0^2 = \sqrt{1.6r_0^2 + t^2} - 0.675t$$

Where t is the thickness of the bulkhead.

In cases where the U-bolt is not centered and the bulkhead is solid, the following equation is used to find the maximum moment:

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

Where a is the radius of the bulkhead and p is the distance between the center of the bulkhead and the area where the load is being applied. The formulations become significantly more complex when centering rings are mixed within solid bulkheads, as is the case with the aft-most fin can bulkhead. Hand calculations become far more imprecise at this point and it is much better to rely on computer simulations such as ANSYS FEA.

3.1.5 Motor Alternatives

The club is currently in possession of several RMS-75/3840 motor casings that can house the 850W, 1150R, 1390G, and 1520T motors from AeroTech. As the team did not want to purchase additional motor casings it was decided that one of these motors would be the primary choice for propulsion for the launch vehicle. Historically the club has used the 1150R and throughout the years have experienced unreliable performance characteristics from this motor. Last year it was determined that this motor would no longer be used in the club. This year the team has decided to uphold this decision to not use the 1150R. Thus, the 850W, 1390G, and 1520T are the primary motors that were being considered for the launch vehicle.

The team currently possesses several RMS-75/3840 motor casings that can house the 850W, 1150R, 1390G, and 1520T motors from AeroTech. As the team did not want to purchase additional motor casings, it was decided that one of these motors would be the primary choice for the launch vehicle propulsion. Historically, the team has used the 1150R and has experienced unreliable performance characteristics from it. Last year, the team determined that this motor would no longer be used in the team's launch vehicle. The team has decided to uphold this decision to not use the 1150R. Thus, the 850W, 1390G, and 1520T are the primary motors that were considered for the launch vehicle.

One of the largest difficulties the team experienced last year was severe weathercocking of the vehicle immediately upon rail exit. This was believed to be caused by two primary reasons: inadequate fin sizing and the L1150R not accelerating the launch vehicle to a large enough velocity when the vehicle left the launch rail. As such, this year the team has decided to choose a rocket motor that provides a large amount of thrust at the beginning of its ignition and continue this large force output throughout the duration of the boosted flight phase. exit. Due to the latter reason, the team has chosen a motor that provides a high initial thrust and maintain this large output through the duration of the motor burn. Additionally, the team has selected a powerful motor but with a short burn time. The purpose of this decision is to ensure that the launch vehicle achieves a large velocity quickly, bringing the center of pressure further aft and ensuring the vehicle maintains a stability margin of at least 2.0 calipers. The thrust curves, shown below, were examined to identify which of the three potential motors satisfy these requirements. Due to the naming convention of the motors, the units for these plots have not been changed to English units.

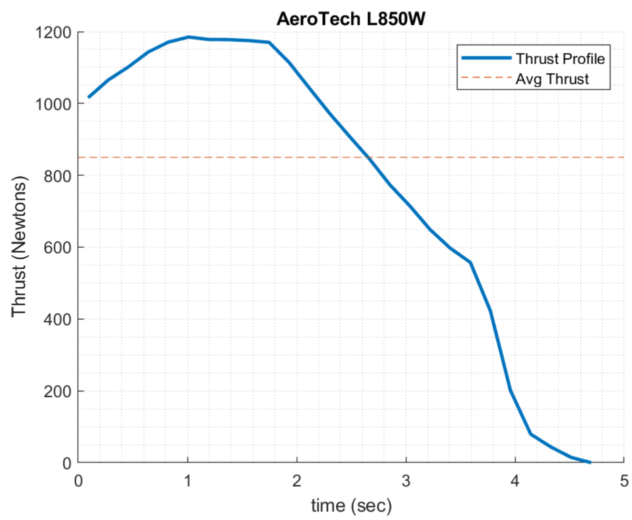


Figure 3-14 Aerotech L850W Thrust Curve

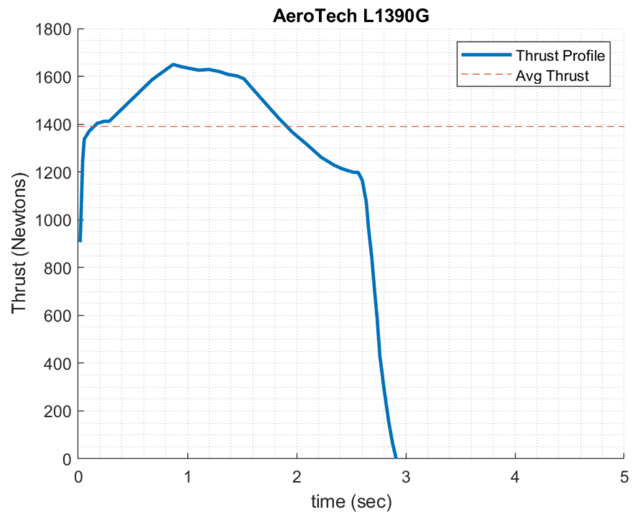


Figure 3-15 Aerotech L1390G Thrust Curve

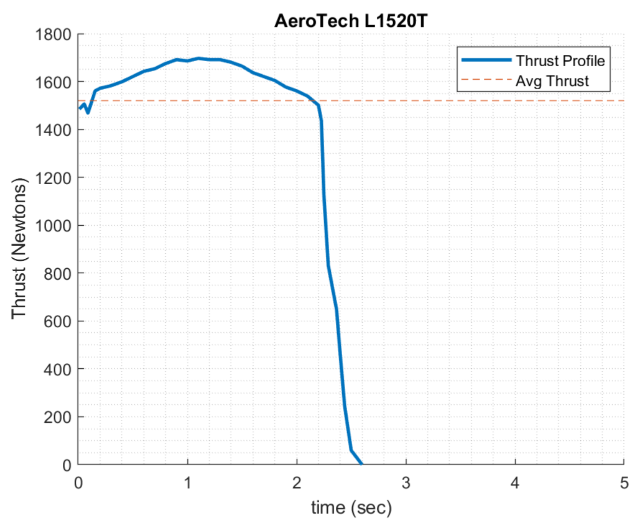


Figure 3-16 Aerotech L1520T Thrust Curve

By briefly inspecting the thrust curves above it becomes obvious that the 850W is not an optimal design choice for the requirements listed above. Namely, this motor has much longer burn time compared to the other two motors and thus the rocket would achieve its maximum stability much later in the flight trajectory.

The 1390G appears to be a worthwhile candidate as it adequately satisfies the requirements listed previously. However, the 1520T appears to be the winning choice as it meets these requirements perfectly.

Simulations conducted in RockSim indicated that all three of these motors resulted in apogees within the 3,500 to 5,500 apogee envelope. These simulations showed that the 1390G resulted in the highest apogee which caused issues for the descent time of the rocket. Most calculations showed that the decent time from apogee would be greater than 90 seconds which is in excess of the limit. The 850W and 1520T gave apogees significantly lower than 1390G and the recovery team found that because of this lower apogee the descent time would fall beneath the 90 second requirement. Motor alternative data is shown below.

Table 3-4 Motor Data

Motor	Total Weight (g)	Total Weight (lb)	Total Impulse (N)	Total Impulse (lb)	Burn Time (s)
L850W	3,742.00	8.2	3646.2	819.7	4.4
L1390G	3,879.00	8.6	3949.0	887.8	2.6
L1520T	3,651.40	8	3715.9	835.4	2.4

From everything listed here, it has been determined that the current leading design choice for the launch vehicles motor is the AeroTech L1520T. This motor will give the launch vehicle a thrust to weight ratio of 7.88.

3.2 Recovery Subsystem

3.2.1 Description of Recovery Events

Once the launch vehicle is mounted on the launch rail and elevated into launch position, both the primary and secondary altimeters will be armed using a screwdriver to close the connected switches. Before the go ahead is given for launch, the output sequence of the altimeters will be listened to and recorded, verifying both altimeters are properly programmed for the mission, have suitable battery voltage, and continuity to the pyrotechnic charges that initiate events. Once proper altimeter function is verified, launch procedure will proceed.

At or shortly after apogee both onboard altimeters will detect apogee and begin initiating drogue parachute deployment. The main altimeter will immediately send a current through an attached E-match, which will detonate a black powder charge in-between the fin can and central AV bay sections of the launch vehicle. The black powder charge sits in a small PVC cap, with the E-match resting on the bottom and small amount of paper towel wadding resting on top. The cap is sealed with electrical tape to prevent leakage of black powder and maintain proximity between the E-match and black powder. The calculations and exact charge sizes are described in detail in section 3.2.2.9. This detonation will separate the two sections which are tethered by a length of Kevlar shock cord and deploy the drogue parachute, attached to a loop on the shock cord. One second after detecting apogee, the secondary altimeter will send a current to a second black powder charge, ensuring redundancy in case the primary charge does not go off or separate the body sections. The secondary charge is delayed in order to prevent over pressurizing the body section if both charges go off at the same time.

With the drogue parachute deployed, the launch vehicle will descend at a constant rate, which is not safe for landing but slow enough to allow deployment of an additional parachute closer to the ground that will not damage the launch vehicle with high acceleration. Upon reaching an altitude of 500 ft, the main altimeter will detect this and send a current to a pyrotechnic charge in the compartment between the AV bay and nosecone, separating them. This charge is constructed and wired identically to the charges in the drogue compartment, except for being connected to different terminals on the altimeter. The charges are also sized differently, as is detailed in section 3.2.2.9. Just as with the drogue parachute, the secondary altimeter will set off another charge at an altitude of 450 ft in order to provide redundancy in the system. Once the body sections separate, again tethered by Kevlar shock cord, a pilot chute will inflate and remove the main parachute from its deployment bag. The pilot chute is attached to the bottom loop of the deployment bag, while the inner loop of the deployment bag is attached to the inseam loop at the top of the main parachute. The main parachute itself is attached to a loop in the shock cord. The vehicle will then descend in this configuration until touching down on the ground.

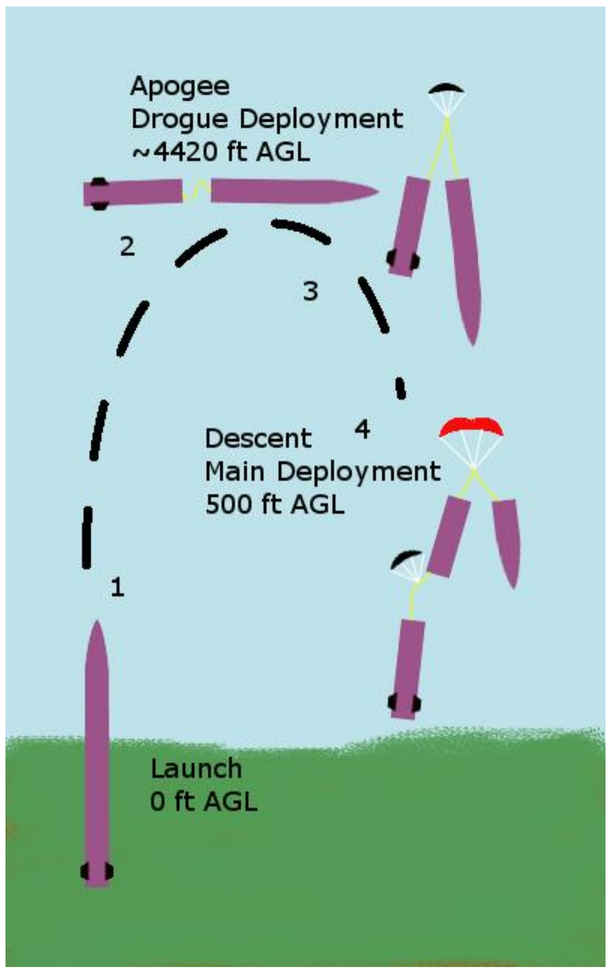


Figure 3-17 Recovery Events Diagram

All body sections which are designed to separate during recovery will be held together with 4 nylon shear pins. These shear pins will be capable of holding the launch vehicle together under its own weight, i.e. when suspending it by the nosecone in the upright position or the fin can when inverted. Black powder charges are sized to shear these pins as well as overcome any frictional forces between the body sections holding them together.

After landing, retrieval of the launch vehicle will be performed using a combination of visible tracking during descent and GPS locating via an RF GPS tracker located in the AV Bay. RF tracking will be performed with a Yagi directional antenna and a handheld radio unit.

3.2.2 Recovery Alternative Designs

3.2.2.1 Avionics Sled Alternatives

The avionics sled must house two altimeters, batteries, switches, and a GPS unit and its respective battery. It must house all these components securely at all orientations under accelerations upwards of 15 gees. It must also be capable of fitting within the coupler section. The electronics must also be able to be detached in order to perform tests on them.

Sled materials to consider include plywood, aluminum, and 3D printed plastic. All materials allow for precision manufacturing, with laser cutting being suitable to plywood and aluminum, while plastic is obviously additively manufactured. While aluminum can be laser cut, it requires a higher power laser, and with stronger lasers come greater safety concerns. 3D printing is much safer in comparison, but the printing and manufacturing process is much longer and prone to errors.

All the materials are strong enough to withstand the flight forces experienced, but the materials hold up to manufacturing differently. Aluminum and plywood can be machined without issue, allowing for easy screw mounting of components. Plastic on the other hand does not hold screw holes well, making component mounting unreliable, which fails to meet a key requirement for the sled. Between aluminum and plywood, plywood is much lighter, saving on weight. Plywood is also thick enough to provide suitable surface area for adhering layers to each other, allowing for 3D sleds from purely laser cut sheets.

Two basic sled designs have been considered, a traditional lengthwise sled and a stacked disk style sled. The lengthwise sled uses threaded rods to keep it in place, with the sled lying flat on the plane between the two rods. The stacked disk concept involves having several removable bulkheads with components mounted on both sides. These bulkheads would then be wired together as necessary. This design would allow for use of more space, but is much harder to manufacture and assemble, given the difficulty of wiring between the bulkheads within the coupler. It also creates separate compartments in the launch vehicle, making it easier to reduce RF interference between altimeters and tracking devices. Additionally, a subscale version of this sled would have very limited space per bulkhead, making it an unfavorable choice. Designing the lengthwise sled is more involved, but actual launch day assembly is much easier.

The sled will also house two 9V batteries and screw switches as shown in section 3.2.2.4. The screw switches will be at elevated off the main sled surface and set at an angle such that the screws are orthogonal to each other and aligned with two of the pressure sampling holes, detailed further in section 3.2.2.5. In order to mount the screw switches at the required angles, a laser cut structure will be constructed with angled plywood pieces through which the mounting screws and standoffs may be attached. A laser cut divided compartment will house the batteries, with the sides being permanently assembled and having a slot on one side through which the wires of the battery cap come through. The top of the compartment is removable and screws into permanently mounted nuts in the inner cavity of the sled. The compartment will be sized to minimize the range of movement of the batteries within, reducing potential damage and ensuring the batteries maintain proper connection throughout the flight.

3.2.2.2 Tracker Electronics Alternatives

Three methods of tracking the launch vehicle during descent and after landing were considered. Two of these systems, the QRP Labs LightAPRS + WSPR tracker and the BigRedBee BeeLine transmitter operate on amateur radio frequencies while the third, the BigRedBee BRB900, utilizes the commercial 900 MHz band. All three of these systems are designed for the tracking of rockets, RC aircraft, or similar vehicles.

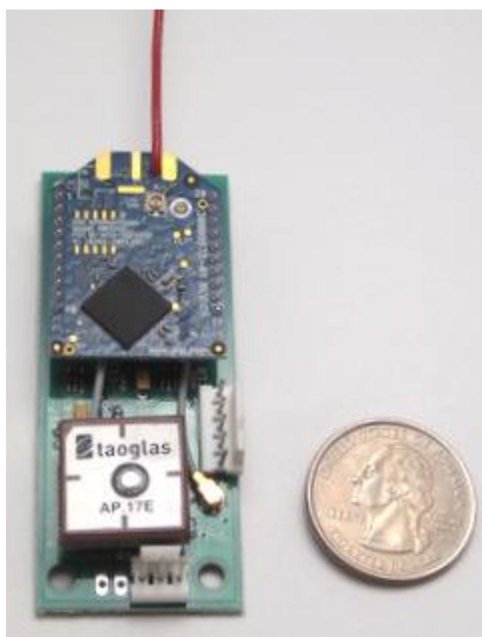


Figure 3-18 BigRedBee BRB900 GPS Tracker

The BigRedBee BRB900 is the simplest of the three tracking systems discussed. Functioning as an all-in-one package, it consists of a 900 MHz handheld receiver, a commercially available 900 MHz transmitter unit mounted to a microcontroller board alongside a GPS antenna, and a single cell LiPo battery. The transmitter makes use of a basic wire antenna to save cost and space, and the receiver uses a standard monopole antenna. The downsides of this system include limited range and frequency availability, a lack of directional tracking in the event of loss of GPS signal, and experience with poor durability and loss of signal during past use of this device.

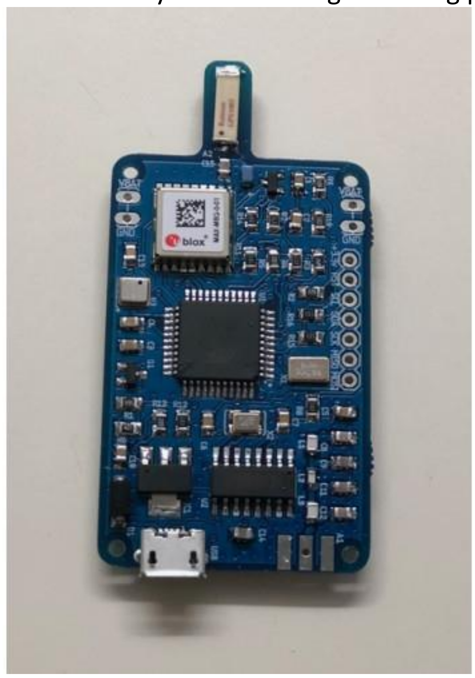


Figure 3-19 QRP Labs LightAPRS GPS Tracker

The LightAPRS by QRP Labs is the most robust and most expensive of the three trackers considered. It is an open source and makes use of an ATmega architecture which allows for ease of reprogramming to meet mission requirements and interface with other telemetry systems. It transmits in the very high frequency (VHF) 134-174 MHz band, allowing for APRS data to be received by standard amateur radio equipment. This in turn may be relayed to a commercial GPS, a custom Raspberry Pi based ground station, laptop, or tablet to allow for real time decoding and plotting of the GPS data transmitted. The device also collects basic telemetry data including temperature, pressure, and altitude in addition to GPS tracking. The main obstacle to use of this device is the requirement for club members to have an amateur radio certification in order to operate the device and the involved ground station setup that would be required.

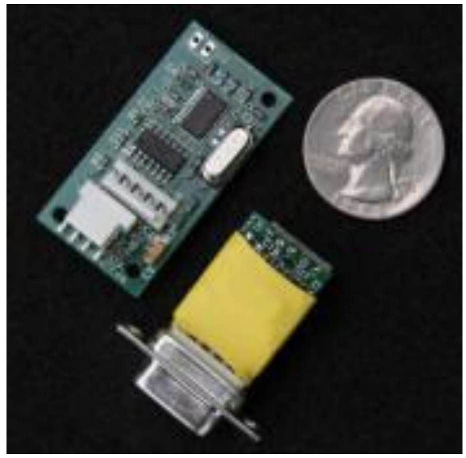


Figure 3-20 BigRedBee BeeLine 70 cm transmitter

Finally, the BRB BeeLine transmitter represents a transmitter only option for tracking the launch vehicle. The BeeLine transmits on the 70cm ham band, which allows for a wide range of frequency selection, thus avoiding conflict with other teams' tracking systems. This device does not contain a GPS unit, rather sending out a tracking signal that the operator then homes in on using a handheld radio receiver and a directional antenna. This offers the advantage of simplicity and low cost, though still requires an amateur radio license much like the LightAPRS. Manual tracking through signal intensity is a relatively unreliable method of determining vehicle location in comparison to real-time GPS data.

3.2.2.3 Recovery Electronics Alternatives

Four altimeters were considered for the recovery system based on commercial availability and dual deployment capability. The altimeters considered were the Missile Works RRC3 "Sport", Entacore AIM USB 3.0, PerfectFlite StratoLoggerCF, and AtlusMetrum EasyMini.

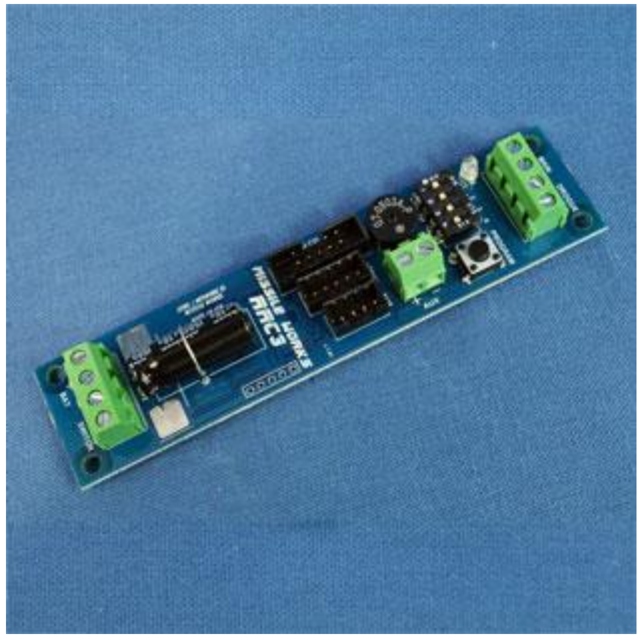


Figure 3-21 Missile Works RRC3 "Sport" Altimeter

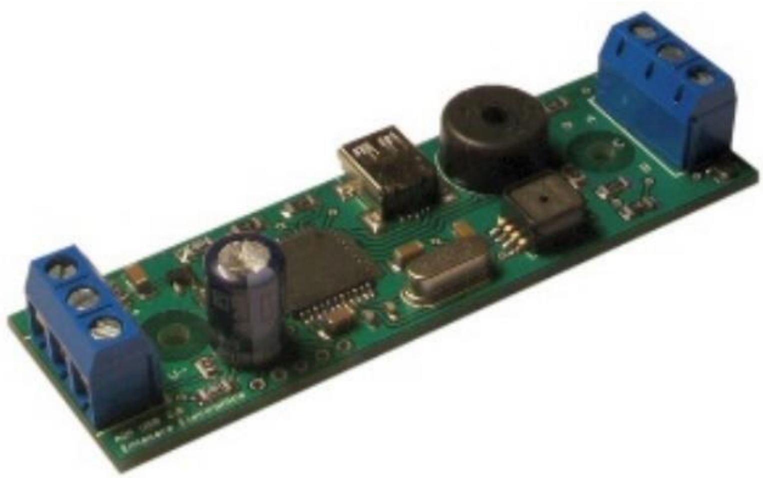


Figure 3-22 Entacore AIM USB 3.0 Altimeter

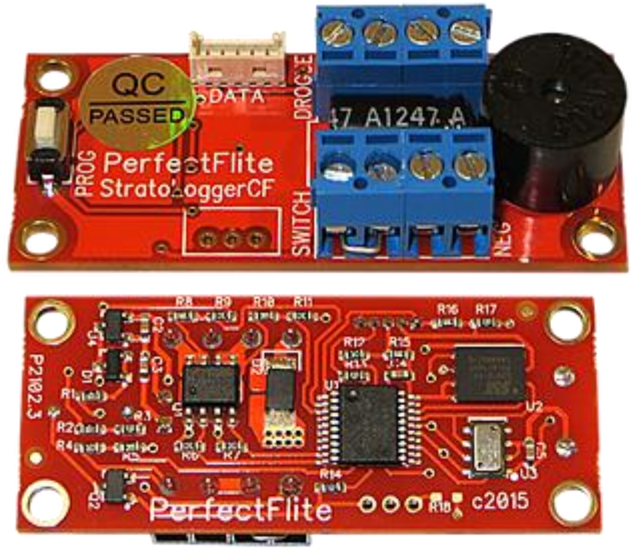


Figure 3-23 PerfectFlite StratologgerCF Altimeter

Figure 3-24 AtlusMetrum EasyMini Altimeter

Table 3-5 Altimeter Comparison Chart

Altimeter	RRC3 Sport	Entacore AIM	StratoLoggerCF	EasyMini
Main Deployment Variability	300 to 3000 ft increments of 100 ft	100 to 9999 ft increments of 1 ft	100 to 9999 ft increments of 1 ft	Any
Delay after apogee	0 to 30 sec, increments of 1	Yes	0 to 5 sec	Yes
Max Altitude	40,000 ft MSL	38,615 ft MSL	100000 ft MSL	100000 ft MSL
Minimum Apogee	100 to 300 ft	N/A	100 AGL	N/A
Altitude 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, temperature, voltage	Altitude, velocity, acceleration, temperature, voltage
Additional Comments		Shared Ground Pin, no switch pins		Reported Quiet Speaker

Table 3-5, above, details the altimeter choices under consideration using information collected from manufacturer and supplier documentation. All the altimeters considered here are dual deploy altimeters capable of controlling the entire recovery process. The parameters of greatest importance are form factor, ease of wiring, measurement precision, and main deployment altitude variability. The Entacore AIM 3.0 and the Missile Works RRC3 "Sport" altimeters both share a long, narrow form factor taking up a large amount of space on the avionics sled. Wiring of the Entacore AIM 3.0 is also difficult. A common ground pin and lack of a dedicated switch circuit increase the chance for assembly errors during wiring of the avionics sled. In particular, the presence of two wires in a single screw terminal causes less secure connections than if only a single wire is present, increasing the likelihood of failure of both charges for a single altimeter. This design is not acceptable to the team. The limited main parachute deployment options of the Missile Works RRC3 "Sport" make it unacceptable for use as a secondary altimeter which would require the ability to set the main deployment altitude to an altitude of 450 ft.

The StratoLoggerCF and EasyMini provide both a wiring and form factor advantage over the RRC3 Sport and the Entacore AIM 3.0. This allows for a more compact design of the avionics sled and greater flexibility in mounting electronic components to provide ease of assembly. The StratoLoggerCF meets the TDR 3.3 that specifies preference for use of equipment currently owned by the team.

The human-factors and reliability issues arising from the design of the Entacore AIM 3.0 make it less suitable as an alternative altimeter. The Missile Works RRC3 is less desirable than the EasyMini or the StratoLoggerCF by way of form factor. Team derived requirements favor the use of StratoLoggerCF altimeters for both primary and secondary altimeter based on adaptability, ease of use, and compact layout.

3.2.2.4 Electrical Schematic of Recovery System and Proof of Redundancy

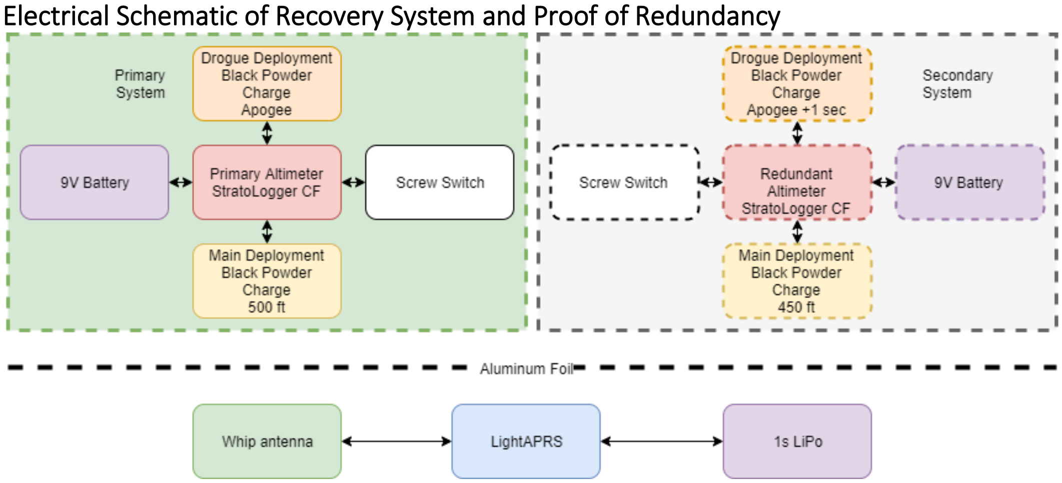


Figure 3-25 AV Bay Recovery Electronics Diagram

The diagram above illustrates the components housed within the AV bay and how all of them are wired. The dashed green box contains all the components of the primary altimeter recovery system. This system contains the competition altimeter which will record the team's official apogee, as well as all the components necessary for the required recovery events. The altimeter has 4 pairs of terminal screws that are connected to a 9V battery, a screw switch, and 2 E-matches. One E-match is in the drogue compartment and triggered at apogee, while the other is in the main chute compartment and triggered at 500 ft. A fresh, tested battery is installed with each flight to ensure the altimeter has enough power to detonate both charges and record flight data.

The dashed grey box houses the secondary altimeter recovery system, which is nearly identical to the primary system, with the black powder charges being the only difference. The drogue charge is programmed to detonate one second after apogee, while the main charge is set to detonate at 450 ft. Both systems are independently capable of safely recovering the launch vehicle, and as they are not connected in any fashion this creates a redundancy in the system. If any one component is to fail, the other system will activate in its place and allow for safe recovery.

Also housed in the AV bay is the LightAPRS, used to track and locate the launch vehicle. The transmitter is connected to a single cell LiPo battery and a whip antenna in order to send and receive signals. In order to satisfy requirement NASA 3.13, the transmitter and antenna will be located on the opposite side of the AV sled, with aluminum foil installed between the two to minimize RF interference.

3.2.2.5 Avionics Bay Sampling Holes

Altimeters detect altitude using barometric pressure of the surrounding air, making properly pressurizing the AV bay key to its successful operation. To do so, holes are drilled in the vehicle body in order to equalize the pressure inside and outside the AV bay. These holes must be precisely sized, as holes that are too small will prevent proper equalization, while holes that are too large will allow airflow into the compartment, compromising aerodynamics and measuring total pressure rather than static pressure. Samplings holes will be sized according to the recommendations of the altimeter manufacturer. The current leading design utilizes two StratoLoggerCF altimeters, for which PerfectFlite recommends sizing the sampling holes according to the following equation, where P is the diameter of the sample holes, L is the length of the compartment, and D is the diameter of the compartment:

$$P = .0008 * L * D^2$$

This equation assumes 4 holes are drilled at 90° from each other at the same axial distance. This method is recommended for 6 inch diameter launch vehicles as opposed to a single sample port. These ports will be located along the band of body tube assigned to the AV Bay, with two of the holes also serving as the holes through which screw switches will be activated.

3.2.2.6 Drogue Parachute Alternatives

The deciding factor in selecting a drogue parachute is the rate at which the vehicle descends under it. If the vehicle descends too fast under drogue, the vehicle experiences extreme accelerations at main deployment that could cause structural damage to the launch vehicle, even potentially severing the connection between a section and the recovery system. If the vehicle descends too slow under drogue, wind drift distance and recovery times become too great. Vehicle descent speed under parachute can be calculated using the terminal velocity formula below.

$$v = \sqrt{\frac{2\,g\,m}{A\,C_D\,\rho}}$$

In the equation above, m is the burnout mass of the launch vehicle, g is the acceleration due to gravity, ρ is the density of air, A is the area of the parachute, and C_D is the coefficient of drag of the parachute. While the area and drag of the vehicle sections themselves do contribute to the velocity at which they fall, their contribution is negligible in comparison to the parachute.

Table 3-6 Drogue Parachute Comparison Chart

Parachute	Descent Velocity	Descent Time from Apogee to Main Deployment	Wind Drift from Apogee to Main Deploy (20 mph)
Fruity Chutes 15 inch Classic Elliptical	138.7 ft/s	28.3 s	829.8 ft
Fruity Chutes 18 inch Classic Elliptical	113.3 ft/s	34.6 s	1016.3 ft
Fruity Chutes 24 inch Classic Elliptical	83.7 ft/s	46.9 s	1376.1 ft
Fruity Chutes 24 inch Compact Elliptical	85.6 ft/s	45.8 s	1344.5 ft

Table 3-6 above compares drogue parachute alternatives in the 15” to 24” diameter range. In order to minimize damage to the vehicle and its payload, the team has decided that the launch vehicle shall not descend under drogue at a rate greater than 100 ft/s. With this requirement in place, the 15” and 18” parachutes are not suitable for use as drogue, descending at rates of 138.7 ft/s and 113.3 ft/s respectively. Both of the 24” parachutes inspected descend at an acceptable rate, though the Compact Elliptical parachute descends at a slightly faster rate, reducing the time from apogee to main deployment and the distance which the launch vehicle will travel due to drift in that time.

3.2.2.7 Main Parachute Alternatives

Selection of the main parachute must consider both the speed at which the launch vehicle descends from apogee and the maximum kinetic energy of all tethered sections. Given that both requirements are a function of velocity, comparison of main parachutes is again based entirely on the descent rate which they produce. The descent rate under the main parachute can be calculated using the same terminal velocity formula as for the drogue, in the above section. In this calculation, both the launch vehicle sections' and the drogue parachute's drag are small enough in comparison to be neglected.

Table 3-7 Main Parachute Comparison Chart

Parachute	Descent Velocity	Maximum Section Kinetic Energy	Descent Time from Main Deployment	Wind Drift Under Main (20 mph)
Fruity Chutes 120-inch Iris UltraCompact	14. ft/s	53.1 ft-lbf	35.7 s	1047.9 ft
Fruity Chutes 120-inch Iris Ultra Standard	14. ft/s	52.9 ft-lbf	35.8 s	1050.5 ft
Sky Angle CERT 3 XXL	13.6 ft/s	50.3 ft-lbf	36.7 s	1077.1 ft
168-inch Rocketman Pro-X	15.8 ft/s	67.9 ft-lbf	31.6 s	927.3 ft

Four parachutes of different designs were compared in the table above, with each being the smallest parachute of that design meeting the kinetic energy requirements. The descent rate was calculated for each parachute as detailed above, and the maximum section kinetic energy was calculated according to section 3.3.2. Descent time and wind drift calculations were also included in order to gauge the effect the main parachute would have on these parameters. Inspecting the Maximum Section Kinetic Energy column shows that all of the parachutes compared satisfy requirement NASA 3.3, with the Pro-X having a kinetic energy of 68 ft-lbf, while the other 3 have kinetic energies in the low 50's. Of the four parachutes compared, the Pro-X has the fastest descent rate, giving it the most favorable descent time and wind drift characteristics, as well as reducing acceleration at main deployment. A major drawback of the Pro-X is the volume it occupies in the launch vehicle, with the manufacturer giving packing dimensions of 4" D x 14" L. While the main parachute bay is spacious enough to fit such a parachute, occupying this much volume may modify black powder calculations, in addition to being more difficult to fold and pack into the launch vehicle on the field. Of the remaining three parachutes, the Iris UltraCompact has the most favorable characteristics. While the kinetic energy and descent time calculations are relatively accurate and directly applicable to reality, wind drift calculations make a few assumptions that limit their predictive accuracy on the field, detailed further in section 3.3.4. For these reasons, the higher wind drift value of the Iris UltraCompact compared to the Pro-X is not immediately disqualifying.

3.2.2.8 Shock Cord Selection

Shock Cord must be strong enough to withstand the force of accelerating the body sections during chute deployment and spaced properly to prevent body sections from impacting each other during descent. For the current projected mass and parachute selection, the maximum acceleration experienced by the launch vehicle is 39 g's at main deployment. If the entire body were to be supported by a shock cord, it would have to withstand approximately 1500 lbs of force. ½ inch nylon and Kevlar webbing both support over 2000 lbs, so these sizes are suitable for the launch vehicle. Nylon is much cheaper than Kevlar, but is not particularly heat resistant, while Kevlar has very little issue with heat. In addition to this, the team already possesses multiple lengths of 5/8" tubular Kevlar suitable for the recovery system.

Conventionally shock cords are about 3 to 5 times longer than the length of the vehicle they support. With a length of 107" this would suggest a shock cord somewhere between 27 and 45 ft. The team possesses two shock cords that are 40 ft long, putting them on the longer end of the spectrum. Longer shock cords result in sections falling further before being caught under parachute. In the case of drogue parachute, this means the body section may reach a velocity greater than the rate of descent under drogue, increasing the acceleration it experiences at that point. During main deployment, longer shock cords can delay the slowing of the body sections as they become taut. Given the length of the body sections, the placement of parachutes along the shock cord can be determined based on the desired configuration of the body as it descends.

During the drogue phase, the upper section of the launch vehicle should descend above the fin can, as this upper section still holds the main parachute. If the fin can were to descend above the upper section, it could foul the main parachute during deployment and result in an unsafely fast descent. In order to achieve this, the distance between the tip of the nosecone and the connection point of the drogue should be less than the distance from the drogue connection point to the opening of the fin can, assuming both sections hang directly downwards from this point. Preliminary sizing suggests the drogue may attached anywhere between the AV shock cord attachment point to 45% down the shock cord from this point. Given this, the drogue parachute will be placed about 1/3 of the way between the AV bay and fin can, closer to the AV bay. This distributes the length somewhat evenly so one section does not experience significantly more acceleration than the other, while still giving a clearance of ~5 ft between the nosecone and fin can.

During the main phase, it is preferable to have the nosecone descend above the other two sections, as this results an entirely vertical arrangement, rather than having the midsection above the others and having the fin can and nose cone occupy the same space. Preliminary sizing suggests a similar sizing window for the main parachute, and so the location has been selected similarly. The main parachute will be attached about 1/3 of the way between the nosecone and midsection, closer to the nosecone. A diagram for the shock cord spacing for both parachutes is provided below.

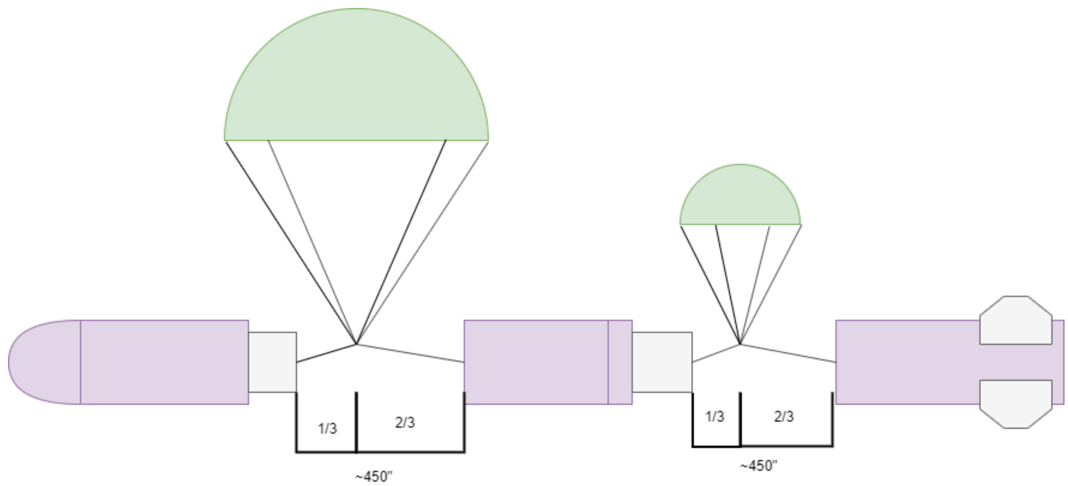


Figure 3-26 Shock Cord Spacing

3.2.2.9 Ejection Charge Sizing

Black powder sizing is determined by finding the force or pressure necessary to separate body sections, and then calculating the mass of black powder necessary to achieve this. First the volume of the sections to be separated is found. In making these calculations, the volume of the drogue parachute and its recovery harness is ignored, while the length of the body tube the main parachute and its harness occupy are removed from the calculation due to utilizing a piston ejection system. A desired pressure of 10 psi will ensure separation of body sections without damage to the airframe or recovery systems, though this may be lowered for the main parachute section as piston ejection systems generally require lower pressures. Given a desired pressure and volume, the mass of black powder required can be calculated using the following equation:

$$PV = mRT$$

Here R is the gas constant of black powder gas, and T is the combustion temperature of black powder. These are known quantities for the 4F black powder used in the ejection charge. Rearranging the equation to solve for m gives the mass of black powder necessary.

In order to provide redundancy in the recovery system, a secondary charge will be activated after the primary charge for each event. This charge will be 0.2 grams larger, to force separation if the first charge was detonated but did not fully separate the body sections. Separation may also fail to occur if the first charge does not detonate; in this case 0.2 grams is small enough of a difference that over pressurization is unlikely to occur if the primary charge would not also cause it.

In order to shield the drogue parachute from the detonation of black powder charges, it will be wrapped in an appropriately sized Nomex sheet. This Nomex sheet is flame resistant, protecting the easily melted nylon of the parachute fabric and shroud lines. The parachute, once folded, will be placed on the sheet, and wrapped in a fashion resembling a burrito to enclose the parachute within. A single quick link will fasten both the drogue's attachment point and the Nomex sheet to the shock cord. This Nomex sheet is not bound around the parachute, so once the body sections separate the sheet will unfurl and allow the drogue to inflate.

The main parachute will use a different system to protect it, as the payload bay beyond it must also be shielded from the effects of the black powder charge. To do so, a piston ejection system will be utilized to deploy the main parachute. A bulkhead with a slot in it is permanently affixed to a length of coupler, this is the "piston". The shock cord is fed through the slot in the bulkhead and affixed at a distance along the shock cord. On one side of the piston, a quick link attaches the shock cord to the AV bay, right next to the main black powder charges. On the other side of the piston is the main parachute, its attachment point, and the nosecone attachment point. When the main black powder charge detonates, the pressurized area is between the black powder charge and the piston. When pressurized, the piston begins expanding towards the nose cone, until eventually forcing the nose cone to separate. In this way the flames and ejection gases will be contained within a separate section, preventing damage to the main parachute or payload electronics.

According to the previous equation, for a 6-inch diameter launch vehicle and cavity 3 inches in length, 10 psi will be produced by 0.5 grams of black powder. Thus, the drogue black powder charges will be 0.5 grams for the primary charge and 0.7 grams for the secondary charge. For a 20-inch-long cavity, 10 psi will be produced by 2.9 grams of black powder, so the primary main charge will be 2.9 grams, and the secondary main charge will be 3.1 grams.

In accordance with requirement NASA 3.2, ground ejection tests will be performed before every flight to verify the charges are appropriately sized and constructed. These tests will be especially important for the main parachute due to the team's lack of familiarity with piston ejection systems.

3.2.3 Recovery Leading Design

The heart of the launch vehicle's recovery system is the AV bay, which will house a flat, lengthwise sled suspended between two threaded rods, constructed out of laser cut plywood. The lengthwise design provides adequate space for all necessary components on a singular removable platform. It shall be constructed out of laser cut plywood to allow for precision manufacturing at an affordable weight. The plywood holds up well to machining and sanding as necessary compared to a plastic solution, while much lighter than an equivalent aluminum sled. Laser cutting allows for rapid prototyping at low time and material cost.

The drogue selected for the leading design is a Fruity Chutes 24” Compact Elliptical Parachute, which has been selected primarily due to a balance in its descent speed and wind drift/descent time characteristics. Smaller parachutes descend at a rate that the team has deemed too high for main deployment, being over 100 ft/s. The 24” Compact Elliptical descends at a rate of 85.6 ft/s with current mass predictions, which is slow enough to safely deploy the main parachute. It is also slightly faster than the Classic Elliptical of the same size, giving it the edge in wind drift and descent time considerations and making it the leading candidate.

The main parachute selected for the leading design is the Fruity Chutes 120” Iris UltraCompact. The Rocketman 168” Pro-X is the only parachute of those compared with more favorable descent speeds and wind drift but is not favored due to its large packing volume. At 14 feet in diameter, the Pro-X would be particularly difficult to fold on the field if necessary and has projected packing dimensions of 14” x 4” diameter. In addition to this, the team already owns a 120” Iris UltraCompact, allowing for that budget to be reallocated elsewhere.

Avionics used will consist of two independent, redundant altimeters that will fire the primary and secondary sets of black powder charges. Both altimeters will be a PerfectFlite StratoLoggerCF selected for small form factor, flight profile customizability, and to improve human factors considerations during wiring of the avionics bay. Tracking will be provided via a QRP Labs LightAPRS radio tracker, transmitting on amateur VHF frequencies. This device will interface with a handheld radio receiver owned by a club member and provide real-time tracking of the launch vehicle throughout the flight. The radio data will be fed into a ground station capable of decoding the audio packets and plotting the GPS and other telemetry data during the flight. A directional antenna will be fitted to the ground station to aid in locating the launch vehicle in the event of loss of GPS signal.

3.3 Mission Performance Predictions

3.3.1 Launch Day Target Altitude

Based on RockSim data, the team has determined that the target altitude for the full-scale launch vehicle will be 4,420 feet.

3.3.2 Flight Profile Simulations

The team relied on flight simulation results to determine the flight characteristics of the full-scale launch. Figure 3-27 shows the result from one of these simulated tests for which the launch conditions included a 12-foot launch rail at a 5° angle from vertical and a slight breeze (8.0 – 14.9 mph).

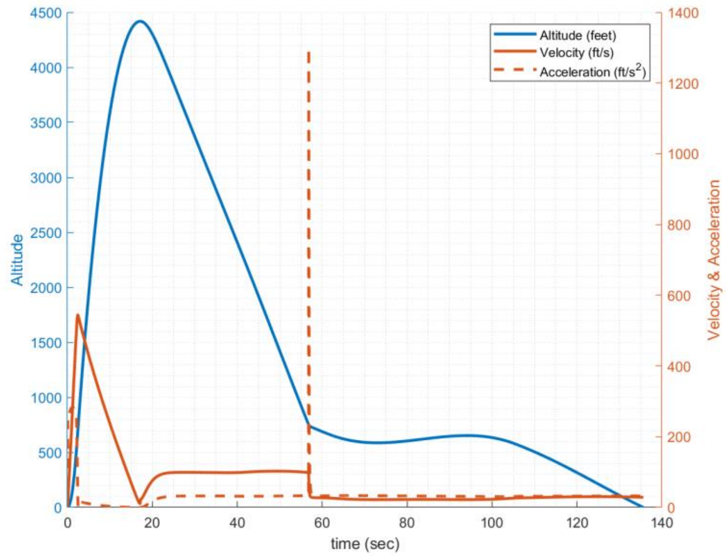


Figure 3-27 Flight Profile

Several apogee simulations were then conducted in RockSim on the leading design to determine a reasonable range of target altitudes. A tolerance study was achieved from this data.

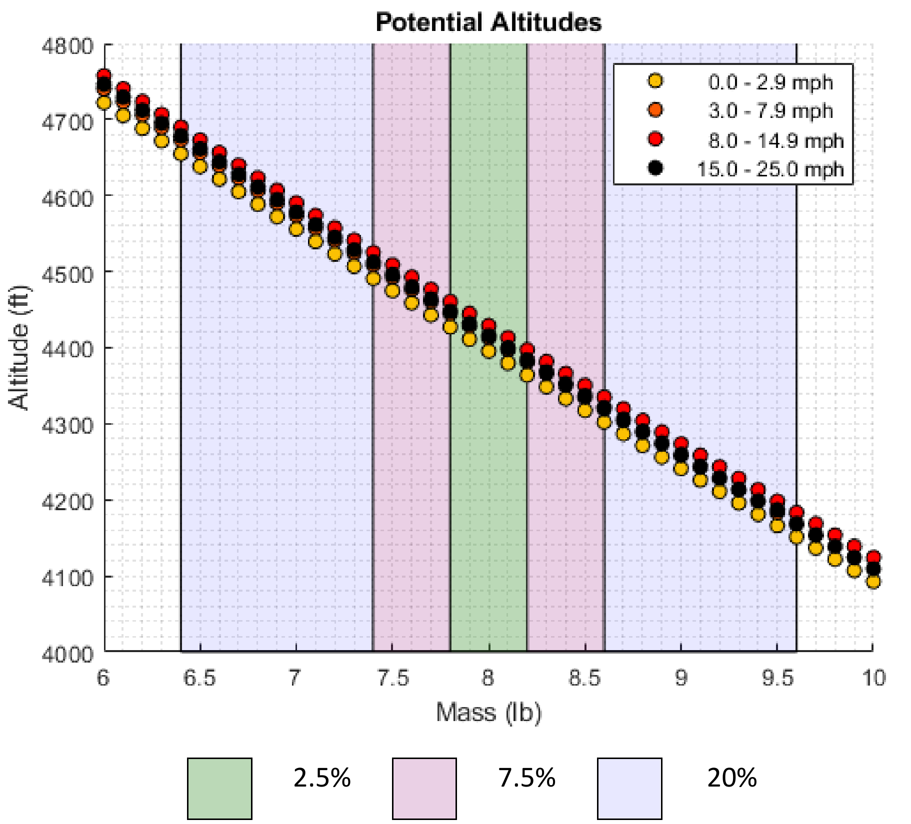


Figure 3-28 Percentage of Mass Uncertainty vs Apogee

This figure shows the predicted apogees from RockSim using the AeroTech L1520T. These simulations ran through several total payload masses and windspeed conditions at the launch site. Although the current leading design budgets a payload mass of 8 pounds, there exists obvious uncertainties as to what the exact payload mass will be. Bounds are marked on the plot to identify uncertainties in this payload mass to highlight potential apogees. Within a 7.5% uncertainty of the total payload mass (i.e. the payload mass could range from 7.4 – 8.6 pounds), the corresponding apogee should fall within the range of 4300 – 4500 feet.

Because the mean wind speed in Huntsville Alabama on April 4th is 8.4 mph, the data points for windspeeds of 3.0 – 7.9 and 8.0 – 14.9 mph were isolated. The corresponding apogees with the design condition of 8 pounds ranged from 4,415 to 4,430 feet. Since the 8.4 mph data point lies about 45% of the way through the 3.0 – 14.9 mph range, the difference in apogees were weighted similarly. Thus, the 15 feet difference between these two apogees was then multiplied by 45% and added to the lower apogee. This resulted in an apogee of 4,421.75 feet. For simplicity, this value is rounded to the nearest tens place value, and the apogee was estimated at 4,420 feet.

Table 3-8 Launch Simulation Parameters

Parameter	Assumption	Justification
Launch Rail Angle	5 degrees	Handbook Requirement 1.12
Launch Rail Length	96 in	Handbook Requirement 1.12
Wind Speed	8.4 mph	Mean at launch site
Launch Direction	Into Wind	Prevailing wind at launch site

Individual section weights are included in section 3.1. Motor thrust curves are shown in section 3.1.5.

3.3.3 Altitude Verification

A secondary method to calculate the altitude was performed to verify that an altitude of 4,420 feet was reasonable. A simple set of algebraic expressions were evaluated and can be seen within this section.

First, preliminary values are needed to begin these calculations:

Table 3-9 Preliminary Values to Algebraically Solve for Apogee

Quantity	Variable	Value	Units
Mass	M	20.27	kg
Frontal Area	A	0.00048	m^2
Gravitational Accel.	g	9.81	$\mathrm{m/s}^2$
Total Impulse	I	3715.9	$\mathrm{N \cdot s}$
Avg Thrust	T	1567.8	N
Burn time	t	2.4	s
Air density	ρ	1.225	$\mathrm{kg/m}^3$
Drag coefficient	C_D	0.35	n/a

Using these values, the first calculation solved for the wind resistance:

$$K = \frac{1}{2} * \rho * C_D * A = 0.000103\ \mathrm{kg/m}$$

Next, the maximum velocity was calculated:

$$v_{max} = \sqrt{\frac{T - M * g}{k}} * \frac{1 - \exp(-xt)}{1 + \exp(-xt)} = 160\ \mathrm{m/s}$$

From this, the altitude of the launch vehicle once the motor has burned out was calculated:

$$h_{boost} = -\frac{M}{2k} * \ln\left(\frac{T - Mg - kv^2}{T - Mg}\right) = 190\ \mathrm{m}$$

Next, the altitude that the launch vehicle will achieve from coast was calculated:

$$h_{coast} = \frac{M}{2k} * \ln\left(\frac{Mg + kv^2}{Mg}\right) = 1296\ \mathrm{m}$$

Finally, from summing the altitudes calculated, the total altitude (or apogee) was calculated:

$$h_{total} = h_{boost} + h_{coast} = 1485\ \mathrm{m} \ \Rightarrow\ 4872\ \mathrm{ft}$$

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

Table 3-10 Apogee Validation Table

Method	Result	Comparison
RockSim	4,420	$\%_{\mathrm{diff}} = 9.71\ \%$
Algebraic	4,872	

Although a percent difference of 10% is significant, it was reasonable considering how simple algebraic analysis was. The 4,872 feet result does not vary wildly from the 4,420 foot result obtained from the more rigorous calculations done by RockSim. From this, it was concluded that the apogee of 4,420 feet is reasonable.

3.3.4 Stability Margin Simulation

Simulations in RockSim indicated that the stability margin of the launch vehicle at rail exit will be 2.44 calipers. However, the simulation is designed for an operating Mach number of 0.3, the average Mach number that the launch vehicle will experience during its ascent. For this Mach number, the simulation resulted in a stability margin of 2.19 calipers. In order to verify the accuracy of this result a validation method utilizing separate calculations was required. One of the most popular sets of equations in rocketry is known as Barrowman's method which involves a series of simple algebraic equations that identify the center of pressure of the launch vehicle. This result can then be used to calculate the stability margin of the launch vehicle.

Firstly, the arm length for any ogive nose, X_N, is a linear function of the length of the nose cone L_N. For the current leading launch vehicle design, this length is 30 inches, thus:

$$X_N = 0.466 * L_N = 13.38$$

Next, the sweep angle of the fins is calculated. For this calculation the fin semi-span length, S, is needed along with the fin sweep length measured parallel to the launch vehicle body. X_R. For the current leading launch vehicle $S = 4.75$ $X_R = 3.0$ inches.

$$\theta = 90° - \tan^{-1}\left(\frac{S}{X_R}\right) = 32.28°$$

This angle value then allowed for the computation of fin mid-chord line length. The important values in this equation were the chord root length, C_R, and the chord tip length, C_T. For the current leading vehicle design, these values are $C_R = 10$ inches and $C_T = 5$ inches.

$$L_F = \sqrt{S^2 + \left(\frac{1}{2}C_T - \frac{1}{2}C_R + \frac{S}{\tan\theta}\right)^2} = 6.91 \text{ inches}$$

Next, the coefficient for the fins was calculated. This value depended on the radius of the launch vehicle body, R, and the number of fins, N. For the current leading vehicle design, these values are $R = 6$ inches and $N = 4$.

$$C_F = \left(1 + \frac{R}{S + R}\right)\left(\frac{4N\left(\frac{S}{R * 2}\right)^2}{1 + \sqrt{1 + \left(\frac{2L_F}{C_R + C_T}\right)^2}}\right) = 5.89$$

The arm length of the fins were also computed. This calculation was dependent on the distance from the nose cone tip to the fin root chord leading edge, X_B. For the current leading vehicle design, this value is $X_B = 96.5$ inches.

$$X_F = X_B + \frac{X_R}{3}\frac{C_R + 2C_T}{(C_R + C_T)} + \frac{1}{6}\left[(C_R + C_T) - \frac{C_R C_T}{(C_R + C_T)}\right] = 99.78 \text{ inches}$$

Finally, from these values the center of pressure was calculated:

$$X_{CP} = \frac{(C_N X_N + C_F X_F)}{C_N + C_F} = 77.89 \text{ inches}$$

The center of gravity location was obtained from RockSim :

$$X_{CG} = 64.89 \text{ inches}$$

With this information the stability margin of the launch vehicle can now be calculated and compared to the stability margin given by RockSim:

Table 3-11 Stability Margin Verification Table

Computation Method	Result	Comparison
Barrowman	$S_{M_1} = \frac{X_{CP} - X_{CG}}{2 * R} = 2.17$	$\%_{\text{diff}} = 0.917\%$
RockSim	$S_{M_2} = 2.19$	

Table 3-11 shows that Barrowman's equations provided a remarkably similar stability margin compared RockSim's calculations. With a percentage difference of less than 1%, it was concluded that the result for the stability margin was precise. On page 51 a summary of the input values required for computation of the Barrowman's equations are listed along with the corresponding outputs from the calculations.

A summary of the input and output values from these equations along with a variable definition list and a sketch showing variable locations is provided on the following page.

For further clarity of the stability margin, data is obtained from RockSim for the center of pressure, center of gravity and the resulting stability margin starting from motor ignition to apogee. This data was then plotted using MatLab and can be seen in Figure 3-29 below:

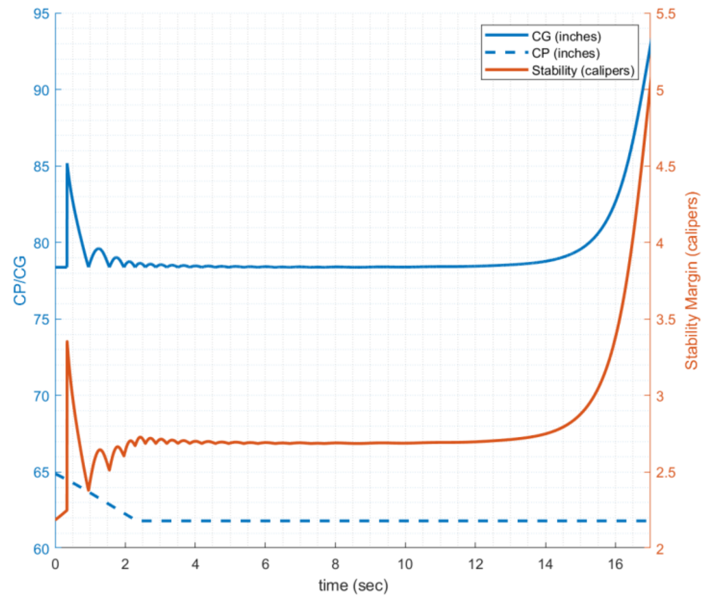


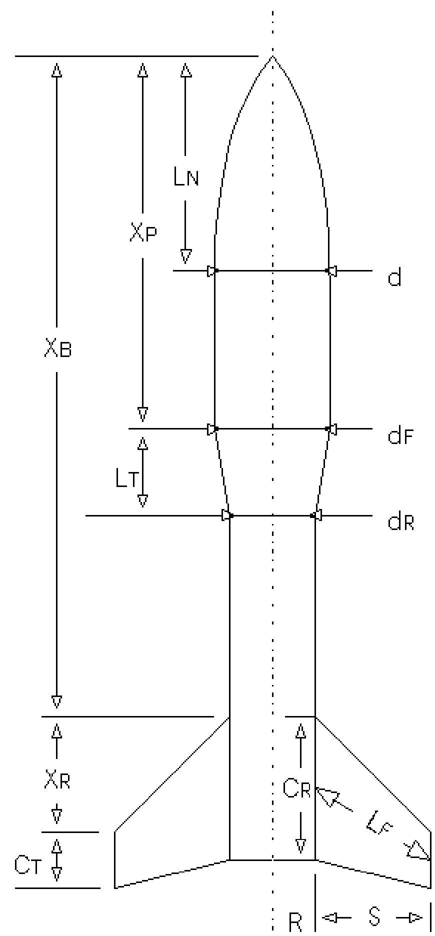
Figure 3-29 Stability Margin Data

Variable Definitions

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

Variable	Input Value	Units
C_N	2	n/a
L_N	30	inches
R	3	inches
S	4.75	inches
N	4	n/a
C_R	10	inches
C_T	5	inches
X_B	96.5	inches
X_R	3	inches
CG	64.89	inches

Variable	Output Value	Units
X_N	13.38	inches
θ	32.38 °	degrees
L_F	6.91	inches
C_F	5.89	n/a
X_F	99.78	inches
X_{CP}	77.89	inches
$\boldsymbol{S_{M_1}}$	**2.17**	**calibers**



3.3.1 Stability Margin Tolerance Study

The current leading design is to have a payload with a total mass of 8 pounds, and the center of gravity of this payload is to be located 9.5 inches behind the forward most edge of the payload bay section of the launch vehicle. However, the team understands that precise measurements of these values are effectively impossible. For this reason, a tolerance study was conducted to determine the range of acceptable mass and center of gravity locations for the payload for the launch vehicle to maintain the stability within a nominal region.

3.3.1.1 Optimal Stability Margin Definition

A stability margin of at least 2.0 upon launch rail exit is required to compete. Therefore, this 2.0 limit defined a hard minimum for the stability margin of the launch vehicle. In addition to this requirement, the team decided to set a soft minimum and soft maximum for the stability margin of the launch vehicle. The term "soft" is used because the intent is for these lower and upper bounds to change and become closer together as simulations improve and manufacturing of the fullscale launch vehicle begins. As long as the stability margin of the launch vehicle falls within these bounds, the stability will be considered nominal by the team.

The team defined a soft minimum stability margin of 2.05 and a soft maximum of 2.3. The minimum bound was chosen such that the team has some room for errors in the simulation data or difficulties with manufacturing the launch vehicle itself. This was done solely as a precautionary measure to ensure that the stability will meet the minimum requirement of 2.0. The upper bound was chosen because the team did not wish to create an over stable launch vehicle. A minimum requirement of 2.0 is considered large for the hobbyist community and launch vehicles with a stability in excess of 2.0 are often deemed too stable. Hobbyists do not intentionally design this large of a stability margin because it could prevent the vehicle from being able to dynamically reach equilibrium if a disturbance unfavorably angles the nose of the vehicle. This is particularly important because the team will not be implementing a means of pitch correction into the vehicle's design. Therefore, an upper limit was also put in place to prevent the launch vehicle from being overly stable.

3.3.1.2 Tolerance Study Data Acquisition

For the tolerance study, over 2,000 data points were manually collected from RockSim. These data points identify stability margins as the mass of the payload changes and as the center of mass changes. For this study, a single center of mass location was held constant as the mass values were ranged from a lower bound to an upper bound in 0.1-pound increments. Stability margins were collected at every mass interval tested. Once this data was gathered, the center of mass location was moved by 0.5 inches, and the process was again repeated. For the sake of thoroughness, the mass values were ranged from 6.0 – 10.0 pounds, $\pm$ 2 pounds from design, and the center of mass locations were ranged from 5.5 – 13.5 pounds, $\pm$ 4 inches from design.

3.3.1.3 Tolerance Study Results

Once the data points were gathered, they were plotted in Figure 3-30.

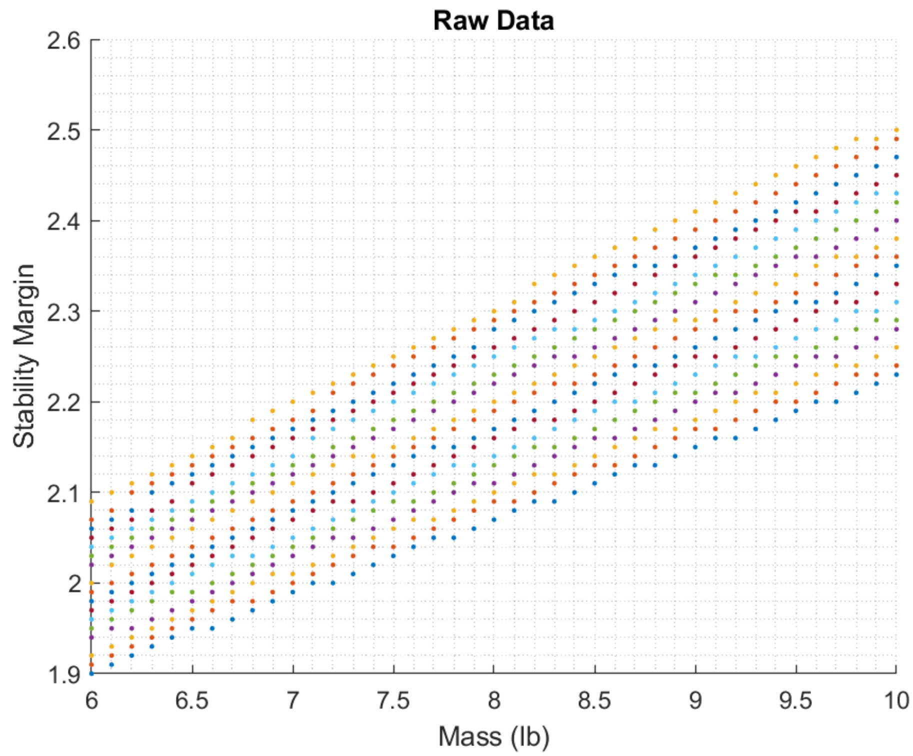


Figure 3-30 Raw Data from Tolerance Study

In the plot above each color shown represents a difference center of mass location that was tested. By inspection it is noted that a linear trend is present in this data collected. To simplify this plot, polynomial expressions with a degree of one were created for each center of mass location tested. Coefficients of determination, r^2, were calculated for each polynomial as well to verify the accuracy of the linear fit. For conciseness, the tabulated data of polynomial expressions and the corresponding coefficients of determination are not listed in this report. However, the lowest r^2 value from these linear polynomial fits was $r^2 = 0.9982$ and the max was $r^2 = 0.9994$, indicating a strong linear fit for the data recorded. Further examination was done to determine whether a pattern was present in the coefficient values of the expressions. Such a pattern could potentially allow for a single function to generate a polynomial expression for any location tested. However, it was ultimately concluded that no pattern exists between the coefficients of the expressions. Nonetheless, the series of expressions from the raw data is presented below in Figure 3-31. This figure also includes the soft minimum, maximum bounds, and the absolute minimum of 2.0.

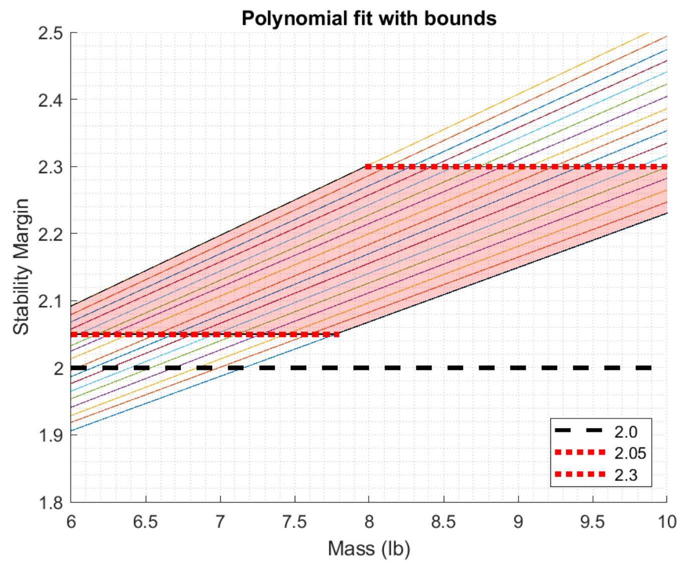


Figure 3-31 Polynomial Fits with Optimal Range Identified

Figure 3-31 indicates that the design for the payload mass and center of mass location could vary significantly and the launch vehicle would still maintain a nominal stability margin. The intent of this plot was to help guide the team in identifying how much variation can occur with the total mass of the payload. From this information, it can be determined how the payload should be mounted within the payload bay to have a center of mass location that results in an acceptable stability margin. For example, if the final payload mass was 20% greater than expected it would result in a mass of 9.6 pounds. This figure shows that the center of mass should be placed somewhere between 10 – 13.5 inches from the forward most edge of the payload bay in order to maintain a stability margin between 2.2 – 2.3 calipers.

3.3.2 Kinetic Energy at Landing

Kinetic Energy can be calculated using the following equation, where KE is the kinetic energy of the object, m is its mass, and V is its velocity:

$$KE = \frac{1}{2}mV^2$$

For each body section, the velocity of the body section for which it has 75 ft-lbs of kinetic energy has been tabulated in the table below. By calculating this velocity, the smallest value can be identified and used as a constraint when selecting main parachutes.

Table 3-12 Maximum Descent Velocity

Section	Mass	Maximum Descent Velocity
Nosecone	.5426 slugs	16.6 ft/s
Midsection	.3588 slugs	20.4 ft/s
Fin can	.2953 slugs	22.5 ft/s

Using this data, the Fruity Chutes 120” Iris UltraCompact parachute has been selected as the main parachute as detailed in section 3.2.3. Using this parachute and its corresponding descent velocity, explained in more detail in section 3.2.2.7, the kinetic energy of each body section at landing has been calculated and tabulated below.

Table 3-13 Kinetic Energy at Landing

Section	Mass	Main Velocity	Kinetic Energy at Landing
Nosecone	.5426 slugs	14.0 ft/s	53.1 ft-lbs
Midsection	.3588 slugs	14.0 ft/s	35.1 ft-lbs
Fin can	.2953 slugs	14.0 ft/s	28.9 ft-lbs

With the heaviest section of the launch vehicle descending with a kinetic energy of 53.1 ft-lbs, the vehicle has a large margin between this value and the maximum as given by requirement NASA 3.3. By squaring the terminal velocity equation, it can be shown that the velocity squared component of kinetic energy scales linearly with mass. If all body sections were to be proportionally increased, this means kinetic energy scales quadratically with mass. Given this relation, the total mass of the launch vehicle could increase by 18% while still meeting kinetic energy limitations. This buffer safely accounts for any differences in mass due to manufacturing, changing mission or payload requirements, or model inaccuracies.

3.3.2.1 Alternative Calculation Method

Kinetic energy has also been calculated using RockSim simulations. Multiple simulations were run at varying conditions to determine the velocity of the launch vehicle at landing. Using the same section mass estimations, the kinetic energy of each section was calculated with the same equation used above. RockSim does not treat sections as separate entities during descent, so calculating the mass and kinetic energy directly within RockSim would require significant workarounds. For this reason, the descent velocity at landing was deemed to be the end of RockSim’s duties in kinetic energy calculations.

Table 3-14 RockSim Kinetic Energy at Landing

Section	Mass	Main Velocity	Kinetic Energy at Landing
Nosecone	.5426 slugs	14.7 ft/s	58.6 ft-lbs
Midsection	.3588 slugs	14.7 ft/s	38.8 ft-lbs
Fin can	.2953 slugs	14.7 ft/s	31.9 ft-lbs

Table 3-14 above tabulates the kinetic energy at landing based on RockSim simulations. Under various conditions landing velocity was found to be 14.7 ft/s, slightly faster than as calculated by hand. As a result, the kinetic energy of each section also increased, with the heaviest section having a kinetic energy of 58.6 ft-lbs, 5.5 ft-lbs greater than as calculated by hand. While a smaller margin, this calculation of the descent rate still allows up to a 13% increase in mass assuming the quadratic relation between mass and kinetic energy holds. The reason for this difference in descent rates and kinetic energy is likely due to the way the parachute is modeled in RockSim compared to the hand calculations. The RockSim model accounts for the spill hole and uses an assigned coefficient of drag, while hand calculations back out the coefficient of drag from the ratings given by the manufacturer. Despite these differences, the difference in the final values is within acceptable ranges.

3.3.3 Descent Time Calculations

Descent time has been calculated as a necessary component of the wind drift calculations and is detailed further in the following section.

3.3.4 Wind Drift Calculations

Wind drift calculations were done by assuming that the rocket traveled straight up to the projected apogee, deployed its drogue parachute and immediately started descending at terminal velocity under drogue, and then deployed the main parachute at 500 ft and immediately accelerated to terminal velocity under the main parachute. From apogee to landing, the rocket is assumed to travel in a single direction at constant speed equal to the wind condition. While not wholly accurate to actual performance, this model covers the time between recovery events well. Assuming the rocket immediately reaches terminal velocity at apogee will reduce descent time by a few seconds, while assuming it immediately reaches terminal velocity at main loses a second or two, as the main parachute takes a second or so to fully deploy and accelerate the rocket. Assuming the rocket travels at wind speed in a single direction is also a worst-case scenario.

Given the distance between apogee, main deployment, and landing and the rate at which the rocket is falling between these locations the descent time of the rocket can be simply calculated as such, where t is the descent time, z_d is apogee, z_m is main deployment altitude, v_d is descent rate under drogue, and v_m is descent rate under main:

$$t = \frac{z_d - z_m}{v_d} + \frac{z_m}{v_m}$$

Leading design parachutes along with an apogee of 4425 and main deploy altitude of 500 give a descent time of 82 seconds using the equation above. Multiplying this descent time by the speed of a given wind condition gives the distance traveled by the rocket constantly sustaining these winds, as seen in Table 3-15.

Table 3-15 Wind Drift and Descent Time

Wind Speed	Apogee	Descent Time	Drift Distance
0 mph	4425 ft AGL	82 s	0 ft
5 mph	4425 ft AGL	82 s	598 ft
10 mph	4425 ft AGL	82 s	1196 ft
15 mph	4425 ft AGL	82 s	1794 ft
20 mph	4425 ft AGL	82 s	2392 ft

At 20 miles per hour, the rocket's estimated drift distance is 2392 ft, just 108 ft less than the maximum allowable value by requirement NASA 3.10. Given that the wind drift calculation assumes drift to be linearly correlated to the descent time, the descent time of a maximum wind drift distance can be backed out to be 85.2 seconds. This means the rocket does not have much margin for error if it is lighter than currently modeled. With allowable weight increases detailed in section 3.3.2, ballast may be added as necessary to increase descent speed and reduce wind drift while staying within allowable kinetic energy limits.

3.3.4.1 Alternative Calculation Method

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

Table 3-16 RockSim Wind Drift and Descent Time

Wind Speed	Apogee	Descent Time	Drift Distance
0 mph	4376 ft AGL	76 s	0 ft
5 mph	4425 ft AGL	76 s	556 ft
10 mph	4337 ft AGL	75 s	1098 ft
15 mph	4286 ft AGL	75 s	1649 ft
20 mph	4211 ft AGL	74 s	2176 ft

In addition to the general trend of drift distance, the RockSim drift table also shows how wind speed affects apogee and descent time, with both dropping as wind speed increases. Comparing the hand calculations and RockSim calculations, both drift distance and descent time for the RockSim calculations appear to be approximately 90% of hand calculation values. The difference in values could be a function of the parachute as modeled in RockSim not lining up perfectly with what has been used in hand calculations. These differences could also be due to RockSim making less assumptions, or more educated ones than those used in the hand calculations. Regardless of the source, 90% is still similar, and both methods return data that suggests the rocket recovery will meet all required descent requirements.

4. Payload Criteria

4.1 Payload Mission Statement

The payload mission is the collection of a suitable sample size of simulated lunar ice from a location near the launch vehicle landing site by a remotely controlled rover following the deployment of said rover from the launch vehicle. Following sample collection, the rover is to move a set distance from the sample collection area as per the requirements of the NASA Student Launch Competition.

4.2 Payload Success Criteria

Payload success is defined firstly by the requirements set forth by the NASA Student Launch Competition as defined in section 6.1 as well as the team derived requirements set forth by the team derived requirements defined in section 6.2. Additional success criteria as well as levels of success and failure are further defined below in Table 4-1.

Table 4-1 Payload Success Criteria

Level	Project Aspect	Human Aspect
Complete Success	Successful sample recovery, rover movement, and payload deployment Recoverable rover Unimpeded payload deployment and operation	No near misses and/or injuries to team members and/or spectators related to operational or non-operational factors
Partial Success	Successful sample recovery, rover movement, and payload deployment Rover repairable Payload deployment or operation impeded or otherwise delayed during operation	Near miss incidents involving team members and/or spectator related to operational or non-operational factors
Partial Failure	Failure to recover required sample Payload fails to deploy Repairable damage to rover or deployment Rover repairable	Minor team member and/or spectator injuries treatable with basic first aid due to operational or non-operational factors
Complete Failure	Deployment system fails in a manner that leads to severe airframe damage Unrecoverable rover	Severe team member and/or spectator injury due to operational or non-operational factors

4.3 Payload Alternative Designs

4.3.1 Bilateral Uptake Rover for Regolith Ice Transport Operations (BURRITO)

BURRITO's systems can be categorized into the chassis, the traction system, the drivetrain, and the controls hardware. Each of these systems must be selected to achieve the desired range and speed for the rover while avoiding the addition of failure modes.

4.3.1.1 Chassis Design

The design of the rover chassis determines how its wheels are attached, how much space is allocated to electronic and mechanical components, and how efficiently the rocket's available payload volume is used for the vehicle. The following are chassis design options considered for the rover:

4.3.1.1(a) Four Wheels and Tank

A chassis utilizing four wheels or tank treads is very stable; once driving, a rover with this design would be difficult to flip on its side or upside down. However, if this rover design were placed in the rocket's payload bay, an additional system would be needed to ensure that, once the bay landed, the rover would drive out upright. This system would introduce another mode of failure that could be avoided with a different chassis design. Additionally, a four-wheeled or tank chassis would not use the payload volume efficiently. Being a relatively flat, rectangular design, this chassis type would not effectively use the volume above or below it in the cylindrical payload bay. Using the volume below the chassis for components would interfere with ground clearance once the rover deployed, as the wheels would have a relatively small diameter. Using the volume above the chassis for components would present the issue of placing the rover's center of mass too high, which the righting mechanism of the payload bay would need to counteract.

4.3.1.1(b) Two Coaxial Wheels

A chassis using two wheels placed on the same axis would appear similar to a chariot and use much larger wheel diameters than the four-wheeled chassis. Assuming the chassis body fit within the diameter of the two wheels, and the friction on the wheels' treads was high enough, the body could rotate about the wheels' axis, eliminating the need for a separate righting system following deployment from the payload bay. Motion about the wheels' axis could be stopped using an arm extended from the chassis, allowing for forward motion. A two-wheeled rover would have the advantage of using the payload bay diameter as the diameter of its wheels, making them relatively large. This would open more of the bay's volume to rover components without sacrificing ground clearance or stability. The design is difficult to stabilize, however; should one of the two wheels encounter an obstacle, the disturbance could tip the whole body forward, which may lead to a situation where the rover becomes stuck or spills whatever ice sample it carried. Additionally, in a field where ruts from soil tilling exist, the rover could become easily stuck if its long axis aligned with the direction of the rut;

much of the motor's power would go into rotating the chassis rather than driving it up the slope.

Below is a table summarizing the characteristics of the different chassis design options, ranked by their characteristics. A higher number for a characteristic indicates better performance, where numbers can range from 0 to 5. "Complexity" refers to the amount of additional systems or moving parts that a chassis design may add to the deployment system, where a lower score indicates higher complexity. "Stability" refers to how resistant a rover with the given design is to falling over due to outside disturbances. "Clearance" refers the magnitude of the wheel diameter (clearance from the ground) a design can accommodate. "Volume" corresponds to how much space the design can accommodate for electronic components and tools. Each of the characteristics are weighted based on importance. Having low complexity was deemed the most important characteristic, as minimizing the failure modes for deployment would better ensure mission success. Stability was weighted slightly lower; though one chassis could be more naturally unstable, there are a variety of ways to stabilize it at the cost of simplicity. Clearance was ranked next, since the terrain at the competition site was not expected to be extremely rugged. Finally, volume was ranked last since the payload bay could be stretched if more space was needed for components, and because electronics can be made relatively compact to start with. Each design has an overall score that is a weighted average of the different characteristic scores; the highest scoring material is best suited to the use case. For the chassis design choice, the two coaxial wheels option performs the best.

Table 4-2 Wheel Base Design Option Evaluation Table

	Weighting	4W / Tank	2 Coax W
Complexity	0.4	1	4
Stability	0.3	5	2
Clearance	0.2	2	5
Volume	0.1	3	4
Weighted Average	-	3	3.6

4.3.1.2 Chassis Material

The primary limitations to the chassis material are weight and strength. A lighter chassis will free up more of the mass budget for other components of the rover, but the strength of the material should not decrease to the point the chassis breaks. The following covers the advantages and disadvantages of some possible chassis materials.

4.3.1.2(a) Aluminum

Aluminum plates cut to shape by a CNC machine could provide very high strength, and cutouts of the material could reduce the weight of the parts. Alternatively, the extruded aluminum parts that make up standard robotics kits could be used, resulting in less optimization for weight. However, aluminum would still be heavier than either of the other two options covered here; the amount of strength it could provide may prove unnecessary, as explained in the paragraph on plywood. Additionally, if new holes need to be drilled into the structure for component attachment, aluminum proves harder to drill into when compared to other material options.

4.3.1.2(b) 3D Printed Plastics

3D printed plastics provide a unique advantage over other materials; they can be printed in customized shapes with as much optimization the team needs. Manufacturing could be simplified by printing the entire structure as a single piece, only requiring nuts and bolts to attach components. However, for the rover chassis to survive the loads of launch, the plastic would likely be printed at maximum density, requiring very long print times. Errors in printing or small changes in the rover's design could force the team to start a new print from scratch, delaying the time to complete a working rover.

4.3.1.2(c) Plywood

Plywood is a material that can be found in much of the structure of high-powered rockets, often making up bulkheads and avionics bays. As such, it has been proven that this material can withstand the forces of launch; at the same time, it is light enough to be used extensively in high powered rockets. Manufacturing is relatively easy, as a laser cutter can be used to form puzzle-like pieces that can then be epoxied or glued together. Modification by drilling or cutting is also low-effort, and holes can be cut in the structure to reduce weight. Plywood may be the weakest (structurally) of the options considered here, but given its existing success in high powered rocketry, it may be an ideal choice for a light and sufficiently strong rover chassis.

Below is a table summarizing the characteristics of the different chassis material options, ranked by their characteristics. This table follows the same rules as the one seen in the chassis design section. "Mass" refers to the relative mass of a material, where a higher number correlates to lower mass (e.g. better performance). Having low mass was deemed the most important characteristic, as minimizing the structure mass would allow other components to use up the saved mass. "Strength" refers to the relative strength of a material. Strength was weighted slightly lower than mass; though the chassis must withstand launch loads, it was deemed unlikely that any of the materials would break. "Adaptability" refers to how easily the structure can be altered after its construction is complete. Adaptability was ranked next since additions to the structure would not necessarily require drilling or cutting into the material. "Familiarity" corresponds to how familiar HPRC is with using the material based on previous projects. Familiarity was ranked last since all three materials are relatively easy to work with, even when prior experience is lacking. Each material has an overall score that is a weighted average of the different characteristic scores; the highest scoring material is best suited to the use case. For the chassis material choice, plywood shows the best performance.

Table 4-3 Chassis Material Selection Evaluation Table

	Weighting	Aluminum	Plastics	Plywood
Mass	0.4	2	3	4
Strength	0.35	5	4	3
Adaptability	0.25	1	3	5
Familiarity	0.1	2	4	5
Weighted Average	-	3	3.75	4.4

4.3.1.3 Traction

The traction system will affect how easily the rover navigates terrain, as well as how much the rover weighs. The following are traction options considered for the rover. Unlike the chassis, which should be custom-built to the team's specifications, wheels and other traction systems are widely available in a variety of sturdy designs. No manufacturing will be needed here, and thus it is not discussed.

4.3.1.3(a) Fixed Radius Wheels

Standard wheels with no additional moving parts on them provide high traction due to their treads and are relatively lightweight due to their simple construction. Given the right chassis or suspension design, they can be good at climbing over obstacles. When it comes to deploying a standard wheel from the rocket payload bay, however, complications can arise. If the rover has a tight fit with the inside of the payload bay, the friction of the wheels' treads could hinder the rover from driving out. In the case of a coaxial two-wheeled design, the wheels may be perpendicular to the deployment direction, increasing resistance to movement substantially. Additionally, since a wheeled rover would have few points of contact with the ground, the rover could become stuck on a hump or in a ditch without enough traction to escape.

4.3.1.3(b) Tank Treads

Due to their many points of ground contact, tank treads have very high traction and avoid the possibility of getting stuck on rough terrain. Using such a system also eliminates the need for an independent steering mechanism that would be required for a standard rover with wheels. However, tank treads are relatively heavy when compared to standard wheels; they would also take up substantial volume in the payload bay. There is also the possibility that the treads could break apart or slip off their sprockets in extreme situations.

4.3.1.3(c) Mecanum Wheels

Mecanum wheels are specially designed with independent rollers in a corkscrew configuration around their radius. This allows them to provide traction when a vehicle drives forward but can also allow a rover to drive sideways when spun in opposing directions. This type of wheel would be specifically advantageous for the coaxial two-wheeled rover design; by spinning in opposite directions in the payload tube, these wheels could allow the rover to deploy with little resistance. However, due to their rollers, Mecanum wheels have relatively low traction and are more prone to breaking compared to a simple wheel.

Below is a table summarizing the characteristics of the different traction options, ranked by their characteristics. This table uses the same rules as the one seen in the chassis design section. “Traction” refers to how much grip a design has on the ground; this was placed at highest priority, since the rover should have maximum grip to overcome any rough or inclined terrain it may encounter. “Volume” refers to the amount of space a traction design takes up in the payload bay; this was considered the next highest priority due to the importance of saving space for other components. “Mass” refers to how much the traction system will impact the rovers’ mass budget; this was placed third in importance, since the traction system is relatively lightweight when compared to the drivetrain battery and motors. “Complexity” corresponds to the amount of moving parts which introduce potential failure modes that could hinder the rover’s mission; this was ranked lowest in importance since failures of such systems are rare as long as the system is installed correctly. Each traction system has an overall score that is a weighted average of the different characteristic scores; the highest scoring traction system is best suited to the rover. For the chassis material choice, the standard wheels show the best performance.

Table 4-4 Traction Design Options Evaluation Table

	Weighting	Wheels	Tank Tread	Mecanum
Traction	0.5	4	5	1
Volume	0.3	2	1	3
Mass	0.1	4	2	3
Complexity	0.1	5	3	2
Weighted Average		3.5	3.3	1.9

4.3.1.4 Drivetrain

The drivetrain, which consists of motors and a battery, will take up the largest portion of the rover’s weight; optimization is critical. The following sections cover the design considerations for motors and batteries. As batteries and motors exist on a wide spectrum of sizes and power outputs, the following sections do not present distinct design options other than the number of motors used. The tradeoffs between power and weight of the drivetrain cannot be tabulated, as the motors and batteries do not vary significantly basic design, but rather in how much output they can provide.

4.3.1.4(a) Number of Motors

The number of motors used to propel the rover will affect how much power is available to overcome terrain obstacles, as well as the weight of the vehicle. A single motor could be used to provide all driving power, while gearing transfers the motor's power to the traction system. The advantage of a single motor lies in its light weight. Motors are generally heavy and using as few as possible frees up the rover's mass budget to other components, such as the chassis. However, should the single motor fail to run, no backup source of propulsion would be available to complete the rover's mission. If the motor does work, it may still fail to move the rover if the rover encounters a steep incline or large obstacle. Additionally, if only one drive motor is used with the coaxial two-wheeled chassis design, it would become challenging to implement steering when both wheels are powered by the same motor.

Using two or more motors could eliminate both the lack of redundancy and lack of power but come at the cost of weight. Doubling a component that already takes up a significant fraction of the rover's mass budget would force other components to lose weight. Additionally, in the circumstance of the coaxial two-wheeled chassis design, a failure of one of the two motors would prevent the rover from travelling anywhere; it would become stuck in a perpetual circle when only one wheel can be powered.

4.3.1.4(b) Motor Selection

Given the desired average speed of the rover, the rover's weight, the characteristics of the traction system, and a few other factors, the required torque and rotational speed of the drive motors can be roughly calculated. Based on these calculations, motors can be selected from a wide variety of manufacturers, while additionally considering their weight and power requirements. The motors' voltage and current requirements then determine what batteries can be used to power them. Again, due to the wide variety of motors available, the options cannot be succinctly tabulated. However, it should be noted that HPRC has decided to not use stepper motors and will rely on brushed motors instead. This decision was due to the difficulties encountered with a critical stepper motor in the previous years' Student Launch competition, where the motor's torque would not remain consistent during operation. As such, brushed motors were deemed more reliable and will be used for the rover's drivetrain.

4.3.1.4(c) Battery Selection

Depending on the motors selected, the battery needs to be the correct voltage to operate the motors and hold enough capacity to power the motors over the maximum possible distance. Additionally, the same batteries will need to power other electronic components on the rover, such as the control system. Batteries will make up a significant portion of the rover's weight, but the impact can be mitigated by purchasing batteries intended for aerial vehicles. These batteries are weight-optimized and still hold plenty of capacity to run a ground vehicle, which requires less power than an aerial vehicle. For RC applications, most batteries are lithium-polymer rechargeable batteries; other variants are not suited to lightweight, high power applications.

4.3.1.5 Controls

The design of the rover's controls system is essentially predefined, as countless examples of remote-controlled robots already exist. All electronic components would ultimately be linked to a central control board which would receive inputs from the driver and translate them into signals for the rover's motors and servos. To receive the signal, a radio antenna and receiver would detect inputs from the driver and send the signal to the control board. The control board would interpret these signals using onboard code and send instructions to the drive motor controllers or sample collection servo controllers. These smaller controllers, in turn, would directly control movement of the rover motors and servos. No autonomy would be involved in this process, as this was not a requirement set forth in the competition. The following sections discuss the characteristics of the components of this control system.

4.3.1.5(a) Radio Frequency

Most RC applications use a 2.4 GHz frequency for sending control signals to a ground vehicle or UAV. As such, most commercially available RC components use this frequency, and higher frequencies are less commonly found. HPRC's collection of radio technology is centered around 2.4 GHz signals, and since the rover does not need to operate at a long distance from the driver, this frequency has enough range to be used on the rover.

4.3.1.5(b) Control Board

There are two primary variants of control board that could be used on the rover. The first is a microcontroller, which consists of one chip that performs all programmed tasks in real time. Microcontrollers are best suited for simple applications where heavy computing power is not needed; additionally, they operate at very low power levels. They are often applied to situations where direct communication between the controller and controlled hardware is needed.

Microprocessors, on the other hand, must interact with supporting electronics and operate using a preinstalled operating system. Microprocessors can handle complex problems and programs, unlike microcontrollers; as such, they also require more power. Rather than directly communicating with hardware, microprocessors must use abstraction to send signals to outboard motors or sensors.

With either system, motor controllers must be connected to the primary controller, since the primary controller itself cannot supply all the power needed to run the motor. The motor controller takes signals from the main controller and allows current to flow from the battery into the motors to run them at the commanded speed.

4.3.1.6 Programming Language

The program used on the controller depends on what the manufacturer designed the control board to use; however, many controllers feature compatibility with a variety of languages. Python is relatively easy for those who have used MATLAB before, but not tailored toward the control of hardware. Some controllers feature their own variant of C, which increases compatibility with the controller, but requires the user to spend time learning the variations from regular C. In any case, there are many online resources for popular controllers applied to simple problems. As the rover is relatively simple, the choice of language is entirely up to whichever control board manufacturer is selected and HPRC's own capacity to learn a new programming language.

4.3.2 Sample Collection

4.3.2.1 Collection System Geometry

4.3.2.1(a) Two-Scoop Geometry

The first method of collecting the sample material utilizes two opposing scoops. The scoops would be the shape of a hollow cylinder cut in half to be semicircular. These scoops would be mounted to servo arms on the underside of the chassis of the rover. From the time of launch to reaching the sample collection area, the arms would be in the stowed position, with both scoops' open ends facing downwards at the outer edge of the chassis. When the collection area is reached, the rover will roll onto the material and then activate the collection mechanism. Once activated, the scoops will rotate inwards, collecting material at the lowest point on their path, and ending their movement with the scoops facing upwards, flush with the bottom of the chassis, and meeting each other in the middle. Figure 4-1, below, shows the initial configuration of the two-scoop design.

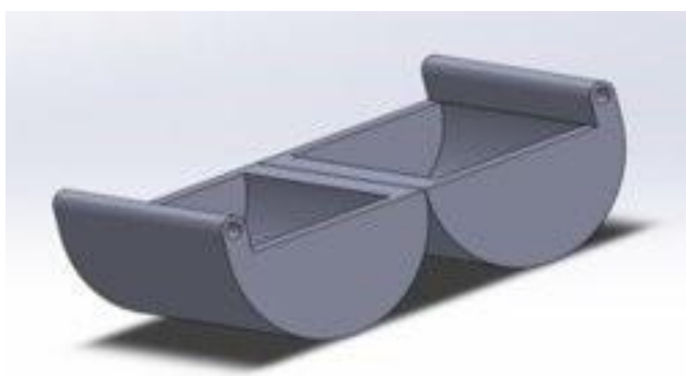


Figure 4-1 Initial design of two-scoop collection method

The scoops shall be large enough that a single activation of the scoops will collect at least 10 mL of sample material, but in the case this does not happen, the scoops may be reset and perform the operation again. One disadvantage to this approach is that verifying enough material has been collected is difficult. A possible solution to this problem is to use a transparent material for the end caps of the scoops and marking them with volumetric measurements like a graduated cylinder. This would allow visual inspection to confirm that an appropriate amount of material has been collected. Due to the scoops resting flush against the chassis at their final position, material should remain within the scoops with minimal chances of losing material as the rover leaves the operational area. The second iteration of this method was created to better accommodate the clearance by improving the reach of the collection system.

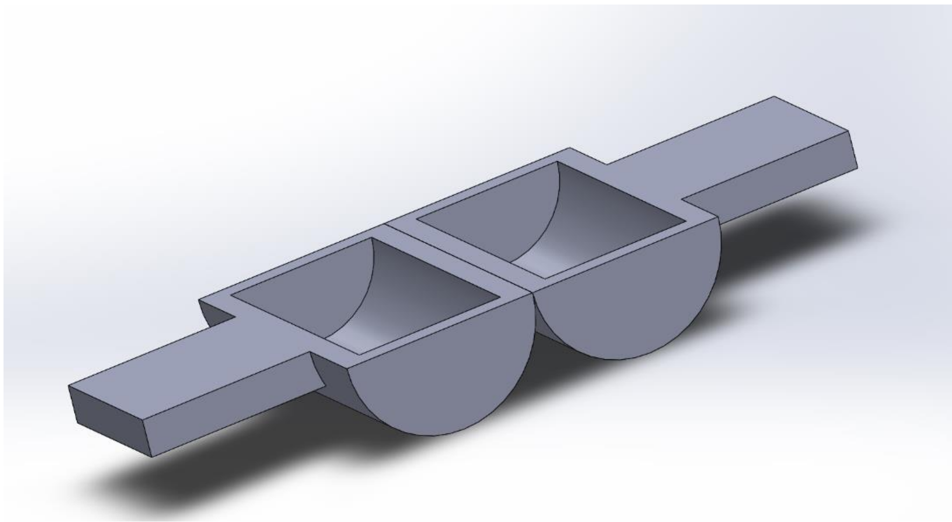


Figure 4-2 Current design of two-scoop collection method

4.3.2.1(b) Rotating Drum

The second method utilizes two rotating drums with an array of scoops to collect material. The two drums would be mounted on arms such that they can be stored up against the bottom of the chassis while in transport, then lowered to the ground after reaching the collection area. The drums would be continually rotated with the scoops on the exterior feeding into the internal drum. This methodology allows for continual collection of material but loses material as it spins. As the scoops reach the bottom of the rotation, material may fall back out through the hole by which new material may enter. By tuning the size of the scoops and the rate of rotation an equilibrium of retained material can be found experimentally. Once the material has been collected, the arms will return the drums to the stored position and leave the collection area. Unfortunately, this method does not have a way of sealing the drum once collection has stopped, so transport may result in loss of material, especially over rough terrain. Figure 4-3, below, shows a section view of the rotating drum design.

Figure 4-3 Rotating Drum design, section view

4.3.2.1(c) Collection Method Evaluation

In evaluating which collection method to use, complexity and volume were the most important considerations as seen in Table 4-5 below. Volume will greatly affect clearance due to being mounted on the underside of the chassis, so smaller solutions will result in more consistent navigation of the rover and is subsequently a heavily weighted factor. Here, complexity is a judgement of the perceived possible sources of error or failure in the design. With more sources of failure present in a design, more resources must be dedicated to ensuring proper assembly and operation of the design. By reducing the complexity of a system, each failure mode can have more resources dedicated to it resulting in a more robust solution, giving it its heavy weighting in the evaluation table below. Mass is an important consideration for any part of the rocket, though the mass of the sample collector itself is relatively small regardless of which solution is chosen. Familiarity is based off if HPRC has used similar solutions before. While familiarity with a solution is not required, it does help the team execute on a solution quickly and accurately. Both familiarity and cost were less stressed because both solutions would be new and the cost of each would depend largely on other design choices and less on which sample collector was chosen.

Table 4-5 Sample Collection Method Evaluation Table

	Weighting	Two Scoop	Drum
Mass	0.2	5	5
Cost	0.1	3	4
Volume	0.3	4	4
Complexity	0.3	4	3
Familiarity	0.1	5	4
Weighted Average	-	4.2	3.9

4.3.2.2 Component Selection

4.3.2.2(a) Material Selection

A material being considered for the scoops or drum to be made of is Polyvinyl Chloride (PVC). The distinct advantages of PVC are that it is strong at the relevant scale of the sample collection, it is affordable, it has a short iteration process, and commercially available PVC components closely model the geometry and size needed for a drum or a scoop. The disadvantage of PVC is that despite convenient geometry both collection methods would necessitate multi-step assembly and finishing to create a viable sample collection device.

Another material being considered to fashion the sample collection apparatus is 3D Printed ABS Plastic. The advantage of being able to reliably replicate the exact geometry denoted in a computer-aided-design (CAD) File means that less finishing and assembly is required. A drawback inherent to 3D printing is that the products are highly anisotropic. This consideration is especially pertinent to the sample collection solutions because both depend on rotation of the component. Because of the rotation the forces experienced by each of the components would be oriented in different directions throughout the different phases of deployment and operation.

Table 4-6 Sample Collection Material Evaluation Table

	Weighting	PVC	3D Printing
Mass	0.2	5	5
Cost	0.1	3	4
Volume	0.1	4	4
Complexity	0.3	4	3
Familiarity	0.3	5	4
Weighted Average	-	4.2	3.9

Complexity and familiarity are the most heavily considered factors. Complexity, as defined by the scale of how many steps are necessary to implement, is important because manufacturing will play a central role in design validation. Familiarity, as defined by if HPRC has used the technology before, is also key in ensuring a smooth manufacturing process in terms of utilizing delegation. Mass is important to a degree, but the materials are similar enough that it would not create a large disparity. Volume and cost were considered the least because the volume should be identical if the design is executed correctly and the relative cost of these items relative to the other electronics needed for Sample Collection is minimal.

4.3.2.2(b) Motor versus Servo Selection

Motors present a robust power delivery option for sample collection and could support a wide number of methods. The advantages of a motor are motors are multi-rotational which would be a necessity for the drum collection method, they provide a large amount of torque to the system which would be a benefit in rigorous digging, and if maintained properly would have a sizable lifespan which would help justify purchasing costs to HPRC. The biggest disadvantage of motors is the weight. With the amount of material needed to be collected and the scale of the rover motors may be too heavy and unnecessarily powerful for the task required.

Servos offer a unique formfactor but a similar function to motors. While a respectable amount of torque can be produced from a servo the weight of the device is substantially smaller. The drawbacks of a servo are that the range of motion is not multi-rotational, this would be an issue for the drum solution but for the scoop solution servos would still be viable. Another difference between servos and motors is that the revolutions per minute (rpm) of servo are usually quite fewer than that of a motor. Because the scope of how many scooping attempts is reasonable, having a high rpm is not a priority. Additionally, an rpm that is too high risks sample being ejected from the sample collection device.

Table 4-7 Sample Collection Power Delivery Evaluation Table

	Weighting	Motor	Servo
Mass	0.25	2	3
Cost	0.125	1	1
Volume	0.25	3	4
Complexity	0.25	3	4
Familiarity	0.125	2	2
Weighted Average	-	2.375	3.125

Mass and volume are stressed because they play a significant role in the viability of the rover, in terms of clearance and center of mass. Complexity, as determined by the amount of coding and control measures needed to implement the power solution, is important because a useful part that can't be implemented well or improperly is a liability. Cost and familiarity, as defined by HPRC using the technology previously, aren't as stressed because since new parts are being bought regardless not much of an advantage would be gained by either solution

4.3.2.2(c) Mechanical Complexity

Mechanical complexity is an important aspect to consider when selecting designs. Additional complexity creates more modes of failure and makes manufacturing more difficult and less reliable. In this respect the two-scoop design is much easier to create and replicate. The scoops are half cylinders, easily created with either a pre-shaped material like PVC or additively manufactured. The operation has few moving parts, with deployment, digging, and storage all being handled by a single servo motor for each scoop. Scale is also easy to visualize for the two-scoop design, with each scoop likely being about an inch in diameter.

The rotating drum on the other hand has several complexities that would make its manufacturing difficult. The greater number of scoops could mean that could result in misalignments and errors in construction. These could cause unexpected torque profiles that could affect operation. 3D printing the drum and scoops would avoid this issue, but the overhanging nature of the scoops may make the printing process difficult. Additionally, the scale of the drum is hard to visualize and therefore implement. With the length of the drum being constrained by size of the wheels of the rover, as this is the dimension along which they will be oriented, the drums may only be about 3 inches long. If this is the case, then the scoops may be quite small and difficult to manufacture. Operation is more involved as well, with at least two motors necessary for each drum. One for deployment and one for rotation.

4.3.3 Payload Integration

4.3.3.1 Background

As defined by the NASA SLI handbook, the main requirement of the payload integration system is to retain the payload during flight. As this retention of the payload is a safety concern, it is prioritized higher than the actual functioning requirement of the integration system. The payload and aerodynamics team have decided that to obtain the required 2.0 stability margin, the payload and deployment components of this challenge should cumulatively weigh no more than eight pounds. The payload team has divided this weight by allotting six pounds to payload and two pounds to integration. The following sections will discuss tradeoffs in designs for the deployment mechanisms, controls and retention. Throughout the design process, an emphasis will be placed on optimizing retention, mass, and power consumption of components.

4.3.3.2 Linear Actuated Pushing Plate

All alternative designs for payload deployment are centered around a pushing plate that is driven down screws, or a linear actuated design. The main plate will house the power transmission, controls and retention component; therefore, the surface area of each design must be large enough to contain all the deployment systems components. The payload bay is a section of the rocket that will separate during flight; therefore, a full body diameter length coupler will be placed at the aft end of the payload bay. In order to push the payload clear of this coupler section, a pushing plate must extrude from the main plate by six inches. The material of the pushing plate will either by ABS plastic or any other lightweight material. The full dimensions, both axial and radial are displayed in the following two figures.

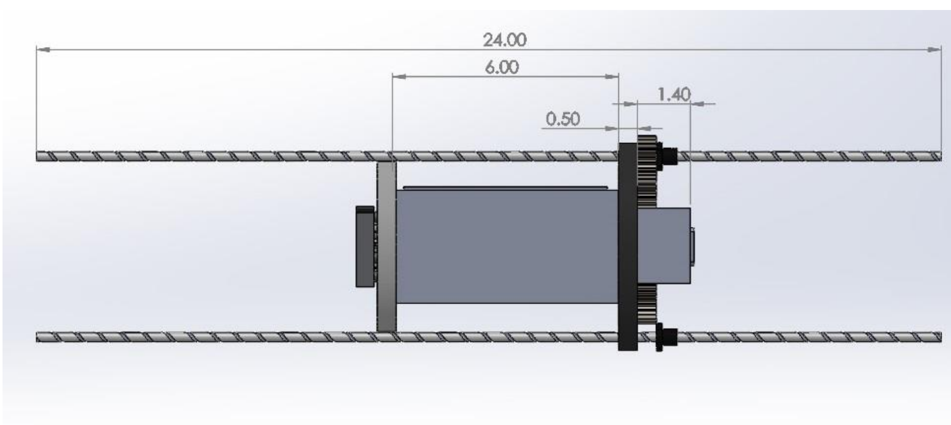


Figure 4-4 Axial Dimensions of Payload Deployment System

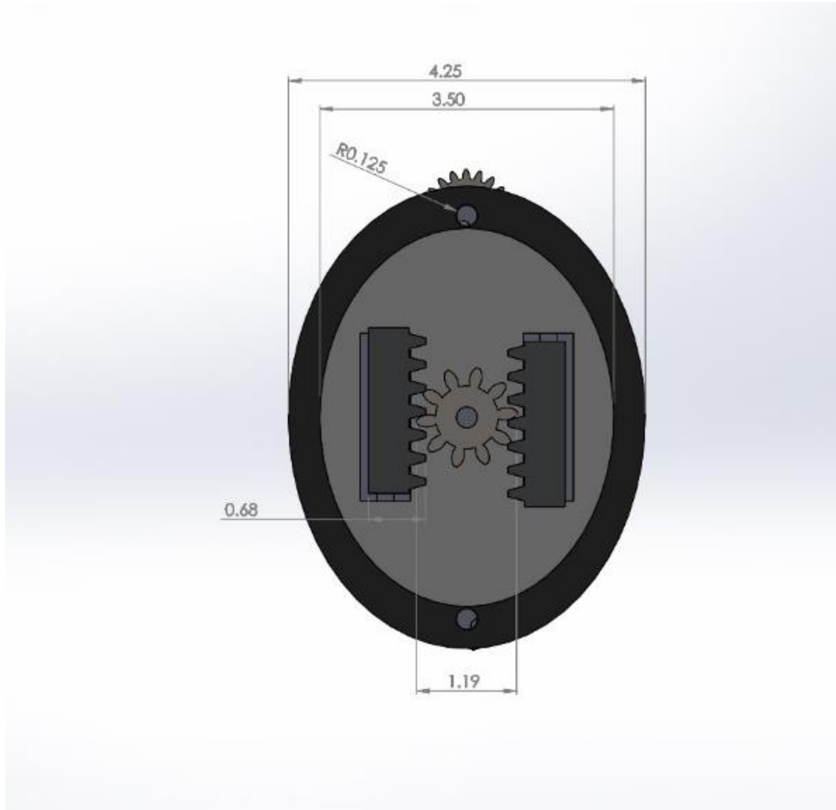


Figure 4-5 Radial dimensions of payload deployment system

4.3.3.3 Power Transmission

4.3.3.3(a) Pulley Drive System

The initial design for power transmission was a pulley driven system. This system would operate by use of various pulleys, bearings, and a timing belt. As the belt and pulleys do not require much surface area, this would allow for the main pushing plate to take on an oval geometry and allow for additional radial supports to be placed between the interior body tube and the payload. These supports would mitigate radial stress on both the payload and deployment mechanisms as well as providing friction to assist in the payload retention. A drawback to the pulley system is its reliability. There is potential for the belt to slip or break during flight and the deployment system would be rendered useless. However, this can be mitigated by using a timing belt that is reinforced with material such as fiber glass or Kevlar. A fiberglass reinforced timing belt is rated for 350,000psi tensile strength dropping the potential for breaking to nearly zero.

4.3.3.3(b) Three Gear System

Pursuing more reliable routes, a gear driven design may be desired. The initial gear train design consisted of a drive gear connected to the motor and two exterior gears holding the nuts for the actuation within the bore hole of the gears. The reliability of this system is primarily dependent on the interface between gear components and if a gear is damaged, the system will most likely be unable to deploy the payload. The gear system is simplistic and reliable; however, the three-gear design does consume more surface area and forces the main plate's geometry to take on a circular shape. With the circular plate geometry, there would not be enough room to implement the radial supports as mentioned in the pulley system. The exterior gears in this system are restricted to being approximately one inch in diameter, or they will protrude off the main plate and cause interference with the rest of the system. This creates a second problem for the three-gear system, where the gear ratio between the exterior gears and drive gear is approximately 3.5, causing a significant reduction in torque.

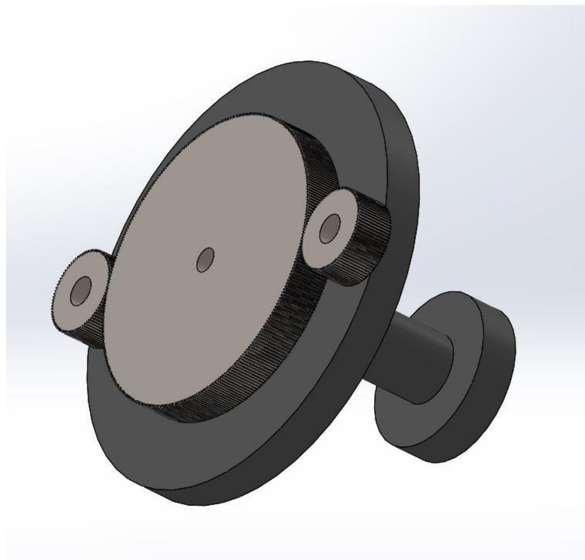


Figure 4-6 Three gear power transmission design

4.3.3.3(c) Five Gear System

Increasing the complexity of the gear design but maintaining the reliability of having a gear driven system, a five-gear system could be implemented. This five-gear system addresses both concerns of torque reduction and main plate geometry. The five-gear system reduces the drive gear (largest gear) from approximately 3.75 inches in diameter to 1.6 inches in diameter. The reduction in the drive gear results in a manageable stepdown in torque of approximately 1.5 opposed to the initial 3.5 reduction in the three-gear system. The reduction in size also allows for the desired oval geometry of the main plate to be kept, meaning that the radial supports can still be implemented. For both the three and five gear transmission system, the material of gears needs to be considered for reliability. Nylon and acetyl gears should not be used due to the potential of damage to gear teeth. If the interfacing between the teeth becomes damaged, the payload will not be able to deploy. For this reason, only metal gears will be considered, and aluminum becomes the most ideal due to its low density.

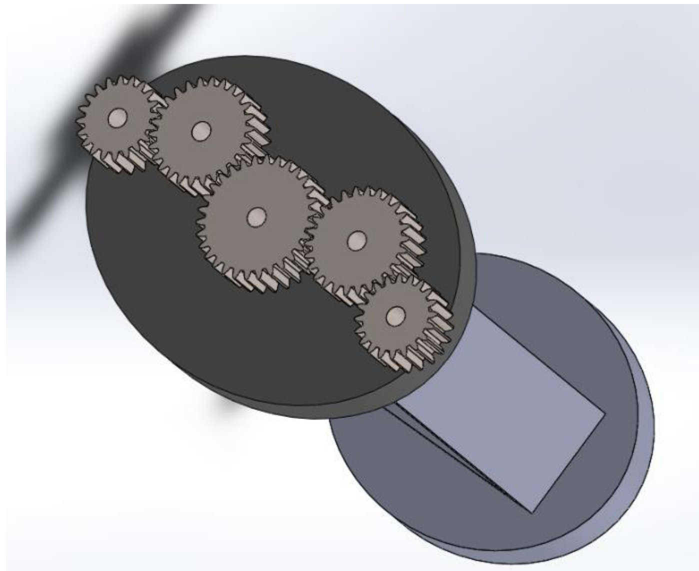


Figure 4-7 Five gear train power transmission design

Based upon the three designs, a general analysis was conducted on a preliminary parts list for each of the designs shown in Table 4-8. The analysis includes the net weight and cost of all components, geometry of the main plate, as well as the effect of the design on the output Torque. From this preliminary parts list analysis, a trade study of each of the designs was conducted to summarize the findings and is displayed in Table 4-8. The system reliability and ability to include radial supports (main plate geometry) were the highest weight components as design prioritization is based upon retention (supports) and actual functionality. The mass is also amongst the top three design parameters as the integration weight budget is only two pounds. It should be noted that the values obtained for masses and costs are general marketed values. Specifically, for gears the masses and costs were obtained using values for aluminum gears.

The table below shows power transmission preliminary parts.

Table 4-8 Power transmission preliminary parts

	Components	Amount	Weight (lbs.)	Cost	Affected Torque	Pushing Plate Geometry
Pulley	Pulleys	5	0.065	$12.50	Motor Output-friction components	Oval
	Bearings	6	0.06	$25.14		
	Timing Belt	1	0.005	$10.00		
	Motor	1	0.09375	$12.00		
	Total		0.2238	$59.64		
3 Gear	Drive Gear	1	0.09375	$25	3.49 Reduction	Circle
	Exterior Gear	2	0.025	$25.98		
	Mounting Rod	3	0.110001	$5.00		
	Motor	1	0.09375	$12.00		
	Total		0.3225	$55.98		
5 Gear	Drive Gear	1	0.028125	$10.99	1.45 Reduction	Oval
	Mid Gear	2	0.037	$21.98		
	Exterior Gear	2	0.025	$25.98		
	Mounting Rod	5	0.1833335	$5.00		
	Motor	1	0.09375	$12.00		
	Total		0.5341	$69.95		

Table 4-9 Trade Analysis on the power transmission design alternatives

	Weighting	Pulley	3-Gear	5-Gear
Reliability	0.3	3	5	4
Mass	0.2	4	4	3
Main Plate Geometry	0.25	5	1	5
Cost	0.1	4	4	3
Torque Reduction	0.15	5	2	3.5
Weighted Average	-	4.1	3.25	4.175

4.3.3.4 Actuator Components

This section has a relatively straight forward comparison between design alternatives and will compare the efficiency differences between lead screws and ball screws as well as the tradeoffs between using two screws or a screw and a guide rod. The primary weight component of this section comes from the screws themselves, so the mass difference between each of these components is negligible and will not be considered. For all the power transmission alternatives, the screws (or screw and guide rod) will be fed through either the exterior pullies or threaded into a nut which will be secured into both exterior gears.

4.3.3.4(a) Lead Screw

Depending on the material of the screw and nut, the friction coefficient for a lead screw ranges from 0.06 with a bronze nut and bronze screw to 0.25 for a cast iron nut and steel screw, according to amesweb. The most commercially available lead screws on the market are manufactured from steel and a coefficient of approximately 0.15 can be assumed for a preliminary comparison.

4.3.3.4(b) Ball Screw

The primary difference between lead screws and ball screws are that ball screws utilize spheres in the interface between the driving nut and the screw, which significantly reduces the friction generated during actuation. Common coefficients of friction for a ball screw were found to be approximately 0.004, which is greater than a magnitude less than the best coefficient of friction for a lead screw. This friction coefficient was obtained from a ball screw product sheet from tech.thk. The increase in efficiency does come at the expense of cost of materials as ball screws do cost more than a lead screw.

4.3.3.4(c) Number of Screws

Another consideration in the actuation design is whether to use a single screw and a guide rod, or to implement two screws. Regardless of the screw material, a 20 inch screw will add considerable weight to the design. Utilizing a plastic guide rod would significantly reduce the mass of this component of the design, but can add potential failures. If the guide rod bends, or if the torque application is too uneven, the deployment sequence may halt causing the motor to stall. If the mass budget can afford two screws, the reliability of the system would be improved. Using two screws would provide a symmetric application of torque and allow for smooth, reliable deployment.

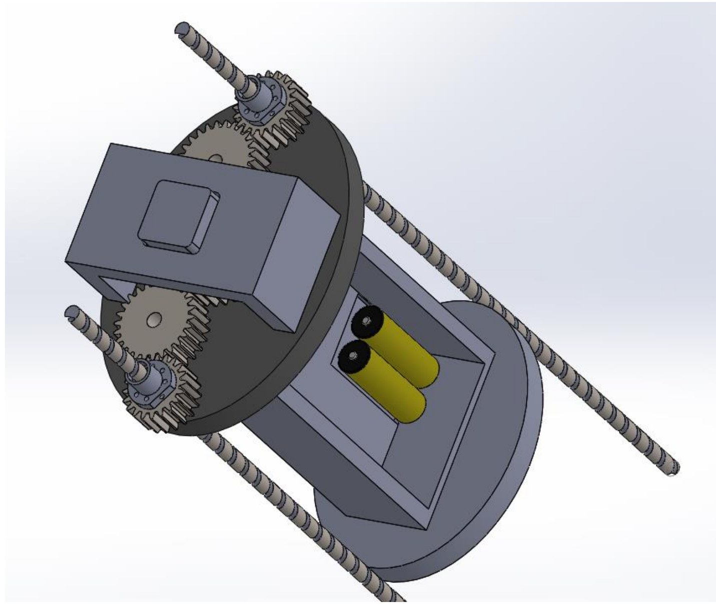


Figure 4-8 Gear assembly and Motor Table

4.3.3.5 Motor Selection

The motor comparison is also straight forward as the motor selection will primarily be based off its ability to provide the minimum torque requirements and its mass. Using the following two equations for required torque which were obtained from amesweb, a preliminary torque analysis was conducted and a minimum torque requirement of 68.8 oz-in was determined.

$$T_R = \frac{Fd_m}{2}\left(\frac{l + \pi f d_m \sec(\alpha)}{\pi d_m + fl\sec(\alpha)}\right) + T_c$$

Where T_R = required torque, F = load, d_m = mean thread diameter, f = coefficient of thread friction, α = half thread angle, and T_c = frictional torque of thrust collar.

$$T_C = \frac{Ff_c d_c}{2}$$

Where f_c = collar friction coefficient, d_c = mean collar diameter.

Stepper motors provide an irregular torque profile and as reliability is prioritized, this motor route was quickly eliminated for potential selection. Both brushless and brushed DC motors offer the desired constant torque profile. In general, brushless motors are more compact, have higher efficiency and less heat generation, typically designed for high RPM low torque applications. However, upon preliminary research, both brushless and brushed motors offer torques that exceed the minimum torque requirement. The difference in size between the two alternative appears to be negligible as well as their masses. Either a brushed or brushless motor can and will be implemented into the overall design.

The motor will need be fixed into place and will be done so by a table placed over the power transmission as shown in Figure 4-8. Both brushed and brushless motors in this torque range have lengths from 2.5 to 3.0 inches long, in order to meet the payload team's derived requirement of the deployment system being no longer than 10 inches, this motor shaft could be a right-angle shaft to minimize overall length.

4.3.3.6 Controls

Another requirement of the payload team is that the deployment system should demonstrate autonomy. In order to reach this requirement, feedback is required to determine at which stage (launch, falling under drogue, falling under main, land) of flight the rocket is in. A microcontroller will also be required to interpret the feedback and control the system. The only consideration for the microcontroller board are the number of pins it can support, otherwise any Arduino Uno or Raspberry Pi will be enough for any design.

4.3.3.6(a) Landing Detection

An internal measurement unit, comprised of an accelerometer, gyroscope, and magnetometers, would be able to give full orientation information at every stage. However, as the stages of flight are determined off acceleration changes primarily in the translational coordinate system rather than rotational, only an accelerometer is necessary to determine the stages. To keep the battery mass at a minimum, power requirements need to be kept at a minimum. Therefore, only a 3-axis accelerometer will be used to save power. The range for the accelerometer must also be determined. RockSim was able to conduct a dynamic analysis of the flight of the rocket. The result was that the rocket would experience the largest acceleration change during deployment of the main parachute, at 28.6g's. This means that the accelerometer should be capable of detecting acceleration changes up to a minimum of 30g's.

The system was intended to operate completely autonomously. The initial design was to include an altimeter as a redundancy. However, as deployment is contingent upon the Range Safety Officers approval, the redundancy will come in the form of user input as discussed in the following Data Transmission section. Including the altimeter, this presents three forms of redundancy which is not necessary and only adds potential failure modes to the control's aspect of this design. Due to the increased risk of failure, the altimeter will be eliminated moving forwards in the design process.

The success of this system will be dependent on whether the code running this system correctly deals with the sensors input. The rocketry club has yet to gather acceleration data from an accelerometer during flight and as such, there are no records from previous years to analyze such data. Therefore, the acceleration feedback is the largest uncertainty in this controls system. In order to appropriately account for the four acceleration changes as well as the thresholds to wake the accelerometer from a power saving mode, an accelerometer will be placed on the subscale rocket. This data will be written to an SD card and analyzed post-launch, giving the necessary data to properly code the controls for full scale flight as well as future use within the club. A model of the electronics setup for acceleration data acquisition is displayed below in Figure 4-9.

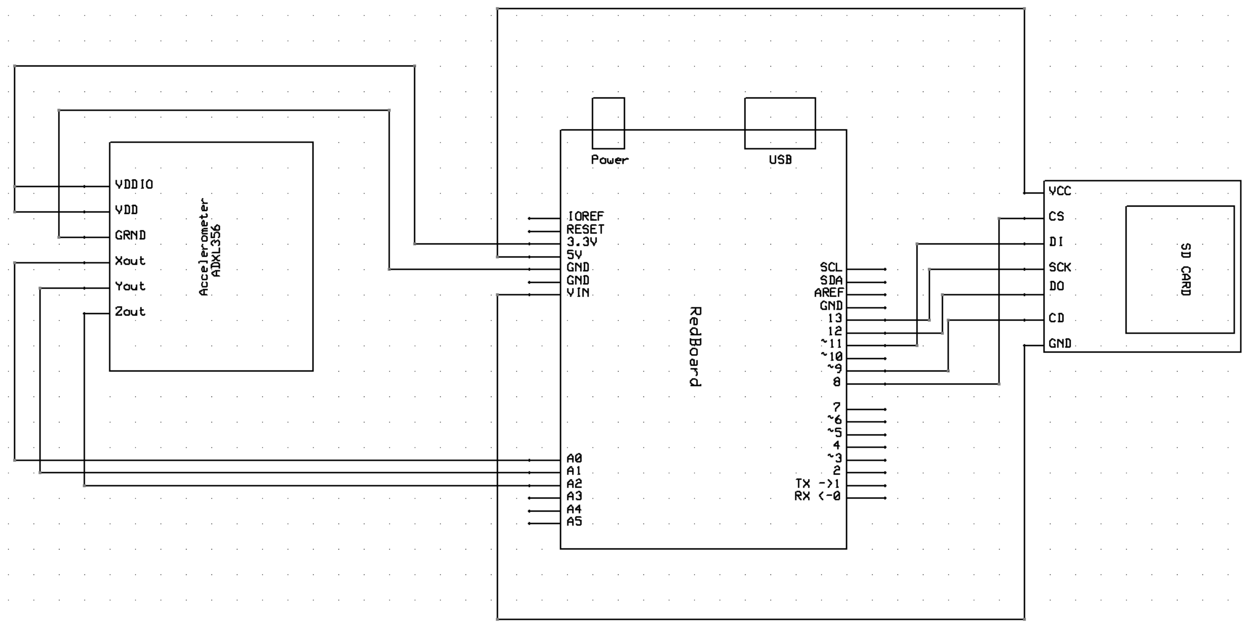


Figure 4-9 Model of Pin Connections from Arduino Uno to accelerometer and SD Card writer

4.3.3.6(b) Data Transmission

All systems should include redundancy when the option is present, and this comes in the form of user input. As SLI teams will have to wait for the Range Safety Officer's approval before deployment, the deployment system will contact the team once the feedback indicates it is in the landing stage. The team will confirm deployment when appropriate and the deployment system must receive this confirmation before initiating the deployment sequence. There are two main components to consider in selecting an appropriate RF transmitter device. The most important of which is the range. A worst-case scenario is that the rocket lands on the edge of the field and is approximately 2,500 feet from the team, so the device must be able to transmit data up to 2,500 feet. The second is whether the device is a transceiver, or if it requires both transmitting and receiving components. If both components are required, it will take up additional power, surface area within the electronics bay, and pins on the microcontroller. The following trade study presented in Table 4-10 shows an analysis on potential radio transmission modules. A later study will include a comparison of whether these modules are compatible with frequency hopping before making a final selection. Frequency hopping will protect from unwanted interference and noise, which could pose to be an issue if other SLI teams have similar components. It is also important to note that most of the modules presented in Table 4-10 operate at 433MHz, which is a restricted band. The rocketry team does have members that hold a radio transmitting license and the team will be able to operate on this band.

Table 4-10 Trade Analysis on the radio transmission modules

	Weighting	NRF24	LoRa	CC1101	HC-12	433 MHz
Range	0.4	2.5	5	1.5	5	1
Transceiver	0.2	5	5	5	5	1
Power	0.15	5	3	3	5	5
Pins	0.1	1	1	5	5	2
Dimension	0.1	2	3	5	5	1
Speed	0.05	4	3	5	4	1
Weighted Average	-	3.25	4	3.2	4.95	1.7

A more detailed version of this trade study contains specific values that will be utilized later for an overall power budget, designing the placement of components within the electronics housing, and contains the maximum output power of each module. Per NASA 2.22.9, the maximum power output of all modules considered is below 250mW.

Table 4-11 Trade Analysis on the radio transmission modules

Module	Range	Transceiver	Current	Voltage	Power Output	Pins	Dimensions	Speed (bps)	Cost
NRF24 2.4Ghz	700	yes	32uA-11mA	3.3	100mW	7	25.5x40.7	250000	2$
LoRa 433Mhz	6000	yes	100mA	3.3	100mW	7	24x25mm	115200	4
CC1101 433Mhz	500	yes	30mA	3.3	100mW	4	19x17mm	500000	7
HC-12 433Mhz	1800	yes	80uA-100mA	3.2-5	100mW	4	27.8x14.4	250000	5
433Mhz	150	No	200uA	5	25.1mW	4	43.2x10mm	4800	4

4.3.3.6(c) Housing

As all components are contained within the pushing plate mechanism, the most ergonomic use of this space would be to house the electronics within the six-inch extrusion from the main plate to the plate that latches to the rover. All electronics will be placed within foam padding to reduce the vibrations on these fragile components.

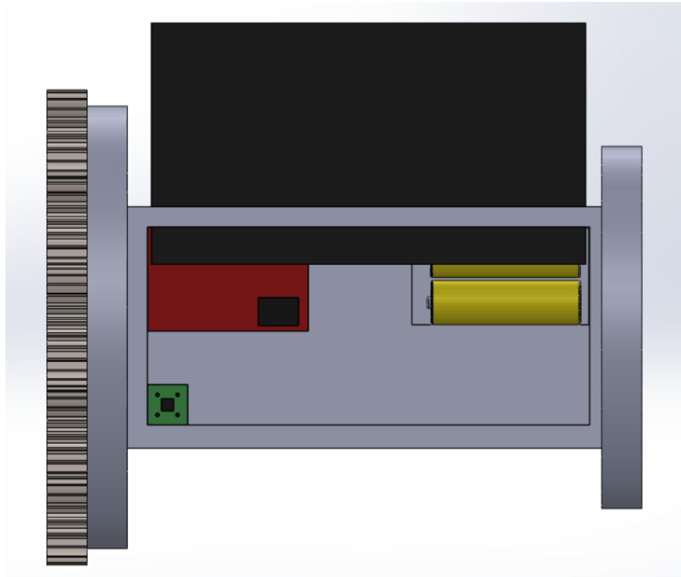


Figure 4-10 Electronics Housing design

4.3.3.7 Retention

The retention of the payload is the most important aspect of this design. As the RockSim dynamic analysis shown in Figure 3-27 shows a maximum acceleration change of 40 g's, the retention mechanism needs to be rated to withstand peak forces at a minimum of 240 lbf, assuming a rover weight of six pounds. For the rocket to experience the predicted 40 gees in the RockSim analysis, the recovery team calculated that the parachute would have to open in approximately 1/20th of a second. Therefore, the rocket will most likely not experience the drastic change in acceleration and a safety factor of 1.5, or a peak rating of 60 gees (360 lbf) is acceptable to ensure that the retention system will not be ripped out of the payload bay.

The drag on the rocket produced from the parachute would cause the payload bay's body tube to descend at a slower rate than the payload itself, so it is unlikely that the payload would be expelled from the bay and become ballistic in the event that the retention mechanisms breaks.

However, the deployment sequence relies on the rover staying latched and in the odd circumstance that the parachute becomes tangled and the rockets orientation is flipped during descent, the latch shall be selected to withstand the forces felt within flight and a factor of safety.

The retention mechanism is modeled after a vault door. As shown in Figure 4-11 it is comprised of a central gear that's teeth are interlaced with two gear racks mounted to a guiding track. As the motor driven gear rotates, the gear racks can move within their respective guiding tracks. The motion of the locking bars is tangent to the rotation of the gear at the gear, locking bar interface.

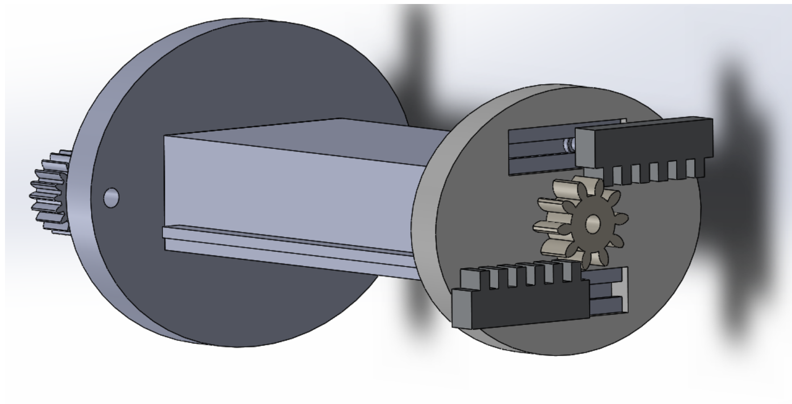


Figure 4-11 Payload Retention Mechanism in the Locked State

Upon pre-launch assembly, the rover will be secured to the deployment mechanism by metal extrusions from the forward wheel of the rover as seen in Figure 4-12. The gear will be driven until the locking bars are fully extended through wheels extrusions, effectively securing the rover to the deployment mechanism. After the deployment sequence has completed, the motor will rotate opposite of the assembly procedure. The gear racks will be pulled back in their tracks to a central position and free of their connection to the wheel, as shown in Figure 4-12.

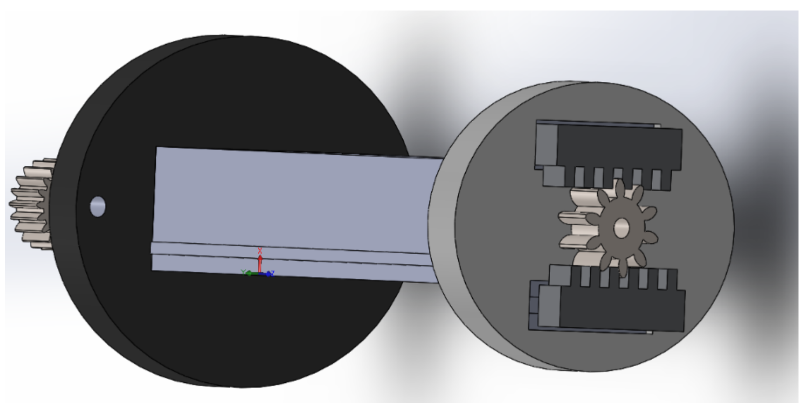


Figure 4-12 Payload Retention Mechanism in the Unlocked State

The guiding tracks will be embedded into the aft pushing plate, securing the locking bars to the deployment mechanism. The main failure point of this design is if the guiding tracks are ripped out of the pushing plate. In order to prevent this, a stress analysis will be performed on the interface between the guiding tracks and the pusher plate. This analysis will give insight into the proper thickness of the pushing plate, guiding tracks and the material both should be made from in order to successfully retain the locking mechanism.

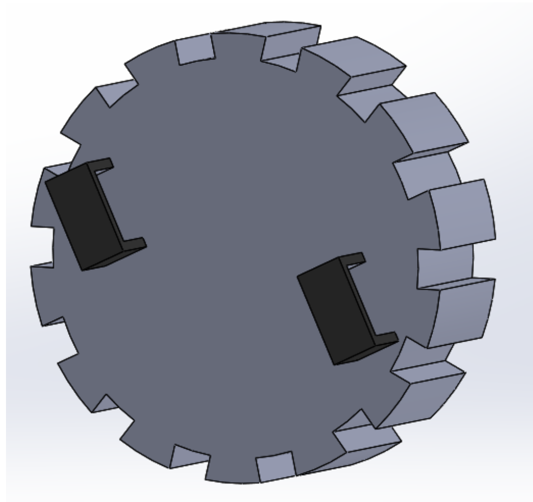


Figure 4-13 Payload Retention Interface on Forward Rover Wheel

As an alternative to the locking mechanism explained above, the club owns a SouthCo R4-EM electronic rotary latch. This latch is rated for 1522 lbf peak loading, which gives a safety factor of greater than 6. This mechanism would be rather simple to implement and would minimize the power requirements. However, the club has had poor experiences with these latches in the past and at 0.625 pounds, the addition of the R4-EM would exceed the allotted weight budget for the payload integration system.

Aside from the locking mechanism, the rover will also need to be supported radially within the payload bay. If the power transmission design allows for an oval shaped main plate, radial supports will be placed from the interior body tube to the rover as shown in Figure 4-14. These radial supports will restrict the rover in one degree of freedom, reduce the stress placed upon the latch and help to guide the rover out of the payload bay during deployment. The rover's aft wheel will also be placed within an extrusion from the aft centering ring. This extrusion will add additional support for the latch as well as keep the rover centered within the payload bay.

Table 4-12 Analysis of Retention Methods

	Part	Amount	density (lb/in^3)	Component weight (lb)	Total Weight (lb)	Tensile Strength (lbf)
Gear Rack Mechanism	Pinion Gear	1	N/A	0.015625	0.426	>= 360
	Gear Rack	2	0.284	0.316875		
	Motor	1	N/A	0.09375		
R4-EM	-	1	-	-	0.625	1522

One design consideration that could reduce the force on the latch is a larger drogue parachute. This would increase the acceleration change felt at apogee.

However, as the rocket approaches apogee, its momentum decreases to zero and a larger drogue parachute would not cause extreme changes in acceleration at this stage in flight. This consideration can only be implemented if the descent time is well below the limit (90 seconds) and the team can afford to trade additional descent time for lower forces in flight.

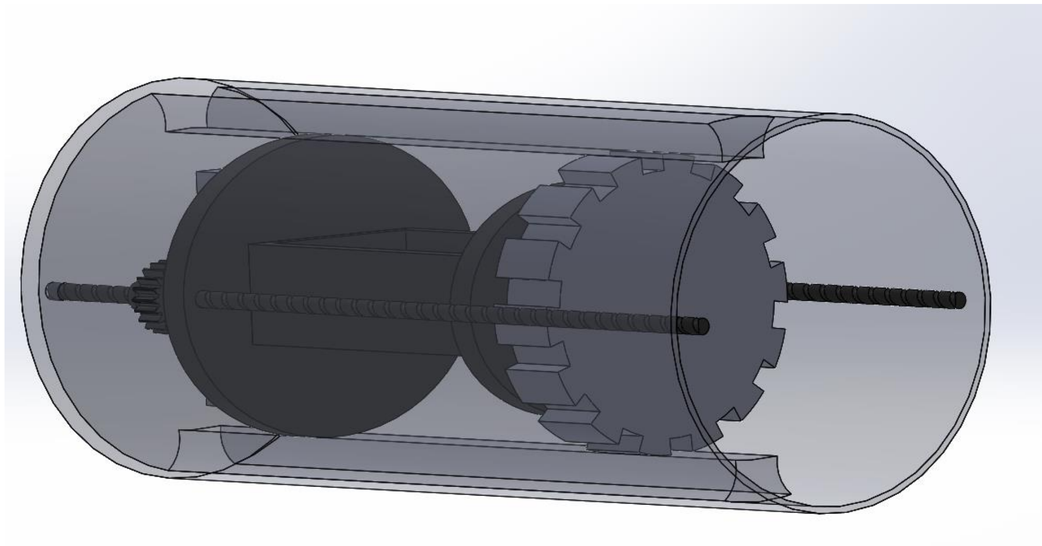


Figure 4-14 Payload Integration System with Radial Supports

Aside from failure modes of each component that were discussed in each respective section, the final failure mode of this system is regarding friction between the rubber wheels and the radial supports as well as the aft end extrusion. The motor selected will include a large factor of safety to overcome any frictional forces. However, one way to reduce these frictional forces is to wrap each wheel with fabric. The fabric would have a far less coefficient of friction allowing the wheels to smoothly slide across the supports and out of the aft extrusion support.

4.3.4 Payload Leading Design

4.3.4.1 BURRITO

The selected baseline design for the payload vehicle is a rover with two coaxial, independently motorized wheels utilizing a plywood body and standard wheels. The control system consists of a microcontroller communicating between a 2.4 GHz radio receiver and the rover's motor controllers and servos. To maintain stability, a spring-loaded arm with a caster wheel is deployed on the rover's back end such that, when the rover drives forward, the caster wheel prevents the body from rolling backward. A model of the baseline design can be seen below. The first image shows the stowed configuration of the rover; the second image shows the deployed configuration.

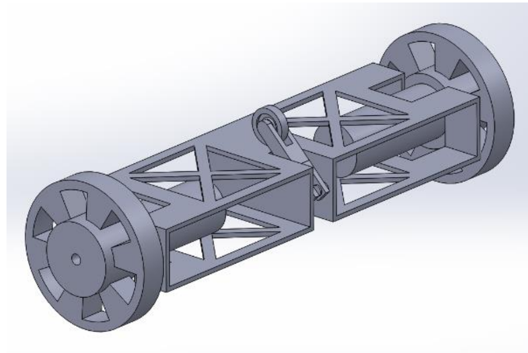


Figure 4-15 Spring-loaded caster wheel stowed

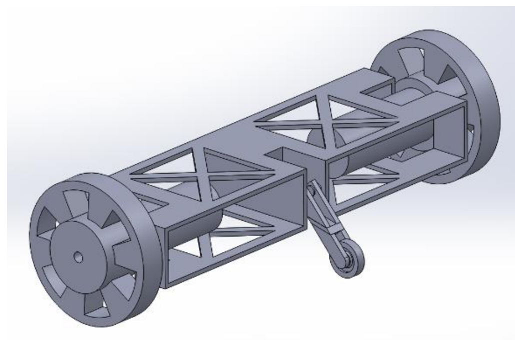


Figure 4-16 Spring-loaded caster wheel deployed

The two-wheeled design will be used since the righting systems needed in the payload bay for a four-wheeled or tank-treaded rover would be complex. Orienting the chassis would be necessary to deploy such rover types, and these mechanisms would need sensors to determine how far to rotate the chassis. Failure of any part of such a righting mechanism would prevent the rover from driving. By contrast, the two-wheeled design could deploy in any rotation and correct its orientation outside the payload bay by rotating the chassis about the wheels' axis. Such simplicity is the primary reason for selecting the two-wheeled design; other reasons include a greater volume available for components on the rover and greater ground clearance. Due to the low stability of the two-wheeled coaxial design, the caster wheel on a spring-deployed arm will be used to counteract this disadvantage of the selected chassis.

Plywood will be the chassis material, since HPRC already possesses a successfully flown avionics bay similar in size and appearance to the rover that is constructed out of plywood pieces. This existing example demonstrates that a plywood rover chassis could theoretically withstand launch loads equal to those experienced by the avionics bay. Additionally, the benefits of an easily cut and modified material factor into the selection of plywood.

The standard wheels are selected for the traction system, since tank treads would not be compatible with the selected chassis, and Mecanum wheels lack traction.

The standard wheels are also the simplest of the traction choices and are thus very unlikely to fail.

The wheels are independently powered by their own motors, since implementing a single motor and steering with a two-wheeled design would be very challenging. Using two motors allows for a simple drive system where spinning one motor faster than the other can produce turning motion. The exact battery and motor selection will be confirmed based on prototyping of the rover, since all inefficiencies and effects of terrain cannot be easily accounted for in preliminary calculations. It is certain that brushed motors will be used, as they do not possess the irregular torque profiles of stepper motors (as stated in the Payload Design Options section). Also, since lithium-polymer batteries are almost universally used in RC applications, the rover will use them as well.

As for the control systems, the radio system will operate on a 2.4 GHz frequency so that commercially bought components can be easily found. The central control will be via a microcontroller, since the rover is a very simple system that does not require the processing power of a microprocessor. This microcontroller will be programmed in whichever compatible language is most straightforward, since most HPRC members lack advanced programming experience.

4.3.4.1(a) Rover Prototype

Once the leading alternatives were selected, work began on a prototype of BURRITO to test the compatibility of different components and discover any problems with the design. The first system to be constructed was a preliminary control system that served to demonstrate how to connect a microcontroller to the different components of the rover. The following is a schematic of the controls prototype.

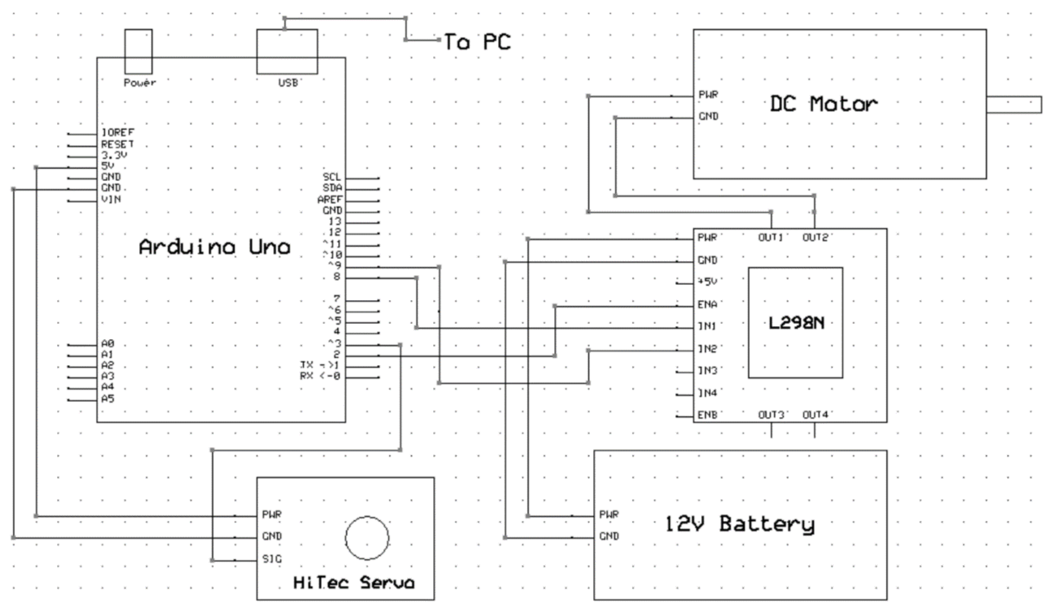


Figure 4-3 Electrical schematic of prototype control system

An Arduino Uno was selected for the microcontroller, as the C-based language it uses allows access to a wide range of libraries that can easily be implemented, even by inexperienced programmers. The two motors for the rover that were selected are planetary gear DC motors from ActoBotics that are capable of a 313 rpm rotational speed. Combined with 4.25-inch diameter wheels, these motors could allow the rover to achieve its target speed of 3 mph. An L298N electronic speed controller was selected to regulate power to the DC motors, as the chip is widely used and compatible with the Arduino Uno. A 12V battery was selected for a power supply, since each of the drive motors on the prototype need to run at 12V.

The controls prototype successfully rotated both the DC motor and the servo, demonstrating that the Arduino Uno microcontroller could be programmed to command these components. What remains to be determined is how the radio receiver connects and communicates with the Arduino Uno, as well as how the Arduino Uno ought to be programmed to interpret the receiver’s signals.

The following is a list of masses for the components that will be used to build a prototype BURRITO capable of driving. The drivetrain system takes up most of the mass budget for the rover, and thus its contribution should be carefully monitored in the event that the components are changed.

Table 4-13 List of Prototype Rover Components By Mass

Product Name	Mass (kg)
(2x) 313 RPM HD Premium Planetary Gear Motor	0.66
Tenergy NiMH Battery Pack 12V 2000mAh	0.25
(2x) 4" Plaction Wheel w/ Wedgetop Tread	0.44
Plywood Body	0.20
(2x) Blank Hub with 0.125 in. Dimple	0.14
All control components	0.07
2 Pcs Fixed Metal Top Plate 1" Diameter Rigid Caster Wheel	0.01
TOTAL	**1.77**

The following figures show the dimensions of the prototype payload, which is designed to fit within the payload bay of the full-scale rocket. The dimensions are expressed in inches.

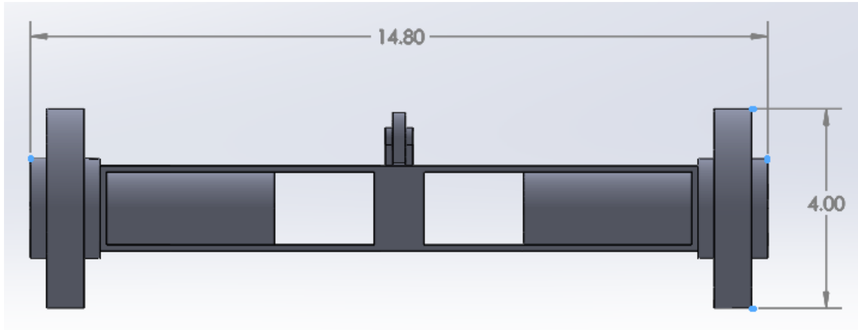


Figure 4-3 Front view of prototype BURRITO

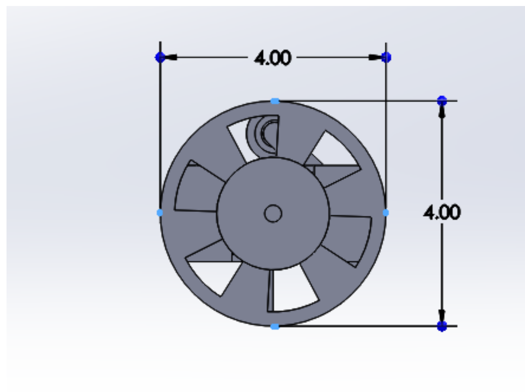


Figure 4-17 Side View of prototype BURRITO

4.3.4.2 Sample Collection

The leading design moving forward for the sample collection is going to be the two-scoop design because of the numerous advantages that it presents. The fact that the team can remove an intermediary arm from the design means less electrical components are necessary. By removing the arm, it also removes another system that would need calibration or could contribute to mission failure. Additionally, the manufacturing of the two-scoop design is far less involved than that of the drum design as it requires less modifications to pre-existing parts. The leading choice of material for the sample collection device is PVC. The advantage of getting exact geometry out of 3D printing while helpful becomes inconsequential when the simpler geometry of collection method is chosen (Two Scoop Design). Furthermore, since the commercially available geometry matches as well as it does, PVC's advantages of strength and faster iteration time become very helpful. With respect to power delivery the servo is the leading design choice because of the weight savings and the two-scoop Design does not necessitate a larger than 360° range of motion. Based off the scope of the task a servo would be suitable for the collection of 10mL of simulated lunar ice.

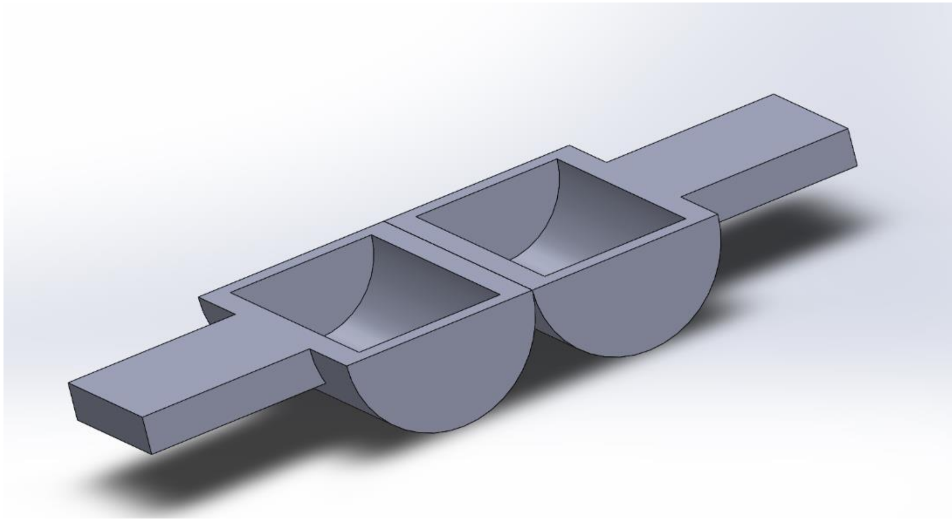


Figure 4-18 Current Leading Sample Collection Design

4.3.4.3 Payload Integration

The current leading designs for the payload deployment system will be discussed in this section. Due to a new payload team requirement of additional supports for the rover within flight, the only potential power transmission designs that can be considered are the belt driven system or the five-gear system. Regardless of reinforced timing belts being commercially available, the team has selected the five-gear transmission system due to the potential of the timing belt slipping. The actuation components will consist of two screws instead of a screw and a guide rod. The reason for this is the desire to keep the system and force application as symmetric as possible to prevent any interference during the deployment sequence. Ideally, the deployment system will utilize ball screws due to their low coefficient of friction. However, upon preliminary research, the cost of ball screws is exponentially higher than that of lead screws. Further research will be conducted to see if there are any cheaper options for ball screws, but the preliminary required torque calculations utilized coefficients of friction for lead screw designs and the system is not contingent upon the low friction that ball screws offer.

The controls of the system have been reduced from the proposal and as the SLI team acts as a redundancy for landing detection by transmitting a confirmation to the deployment system, the use of an altimeter has been eliminated. Currently, the feedback the system receives is from a +/- 40 gees accelerometer (ADXL356) and the SLI team's confirmation. To transmit the data, based upon power requirements, surface area of components and range the team has narrowed down the transceiver module selection to either the LoRa module or the HC-12. Both modules are capable of transmitting data within the required range (up to 2500 feet). However, the LoRa module is capable of frequency hopping and is more desirable to protect from interference from other SLI teams.

Therefore, the team will be utilizing two LoRa modules to communicate with the deployment system.

Due to the weight of the R4-EM, the team has selected to prototype the gear rack retention system. Upon further analysis in ANSYS, the exact materials and dimensions of the gear rack and pusher plate will be selected to meet the minimum retention requirement of 360 lbf. This 360lbf minimum includes the current max acceleration change of 40 gees from the RockSim simulation as well as a 1.5 safety factor. As the retention system will not be rated nearly as high as the R4-EM (1522 lbf tensile strength), it has a 30% reduction in weight, preventing the net weight of the system from exceeding the allotted weight budget. As previously mentioned, the payload will also be supported by radial extrusions spanning the payload bay, as well as an extrusion from the aft centering ring that will prevent any cantilever effects from introducing unnecessary stress at the interface of the locking mechanism and the rover wheel.

5. Safety

Safety engineering is an essential part of Tacho Lycos' team structure, and as such, the team takes active steps, detailed below, to minimize personnel injury and maximize mission success. Using FMEA tables, fault tree analysis, personnel safety training, and a newly created lab handbook, the team seeks to create an environment with no worry of personnel injury or partial/total mission failure.

5.1 Safety Officer

Frances McBride is the safety officer for the 2019-2020 competition. Frances' responsibilities include collaborating with design teams to ensure mission success and personnel safety, educating the team on safe lab and launch practices, and optimizing lab design for increased safety. Frances or an individual trained by Frances as a stand-in safety officer is present for all team meetings, launches, and fabrication sessions for NASA Student Launch.

Responsibilities:

- Establish safety culture in club
    - Via frequent discussions on safety, training days, and regular safety meetings.
- Provide redundant safety and checklist approval during launches
    - Every checklist item past 1A hazard level has a signed safety officer confirmation.
- Be present for all build days
    - Frances or another highly trained stand-in safety officer will monitor all build activities.
- Develop verifiable strategies to minimize personnel and mission damage
    - Including, but not limited to creation of lab handbook of condensed MSDS and how-to-use for all tools, labels on all power tools, and safety training for every lab space user.
- Collaborate with design teams to prevent failure through improving designs
    - Designs analyzed for redundancy and likelihood to succeed.

The safety officer is responsible for maintaining and enforcing the contents of the club safety handbook. This handbook includes condensed safety data sheets, instructional guides for tool usage, instructions for safe handling of hazardous materials, and guidelines for hazardous manufacturing procedures.

Additionally, the safety officer is responsible for creating supplemental failure mode and hazard analysis using Fault Tree Analysis (FTA). FTA helps the team deductively determine mitigations for hazards encountered during the project. FTA for the recovery/avionics, structures, and payload subsystem teams are detailed below:

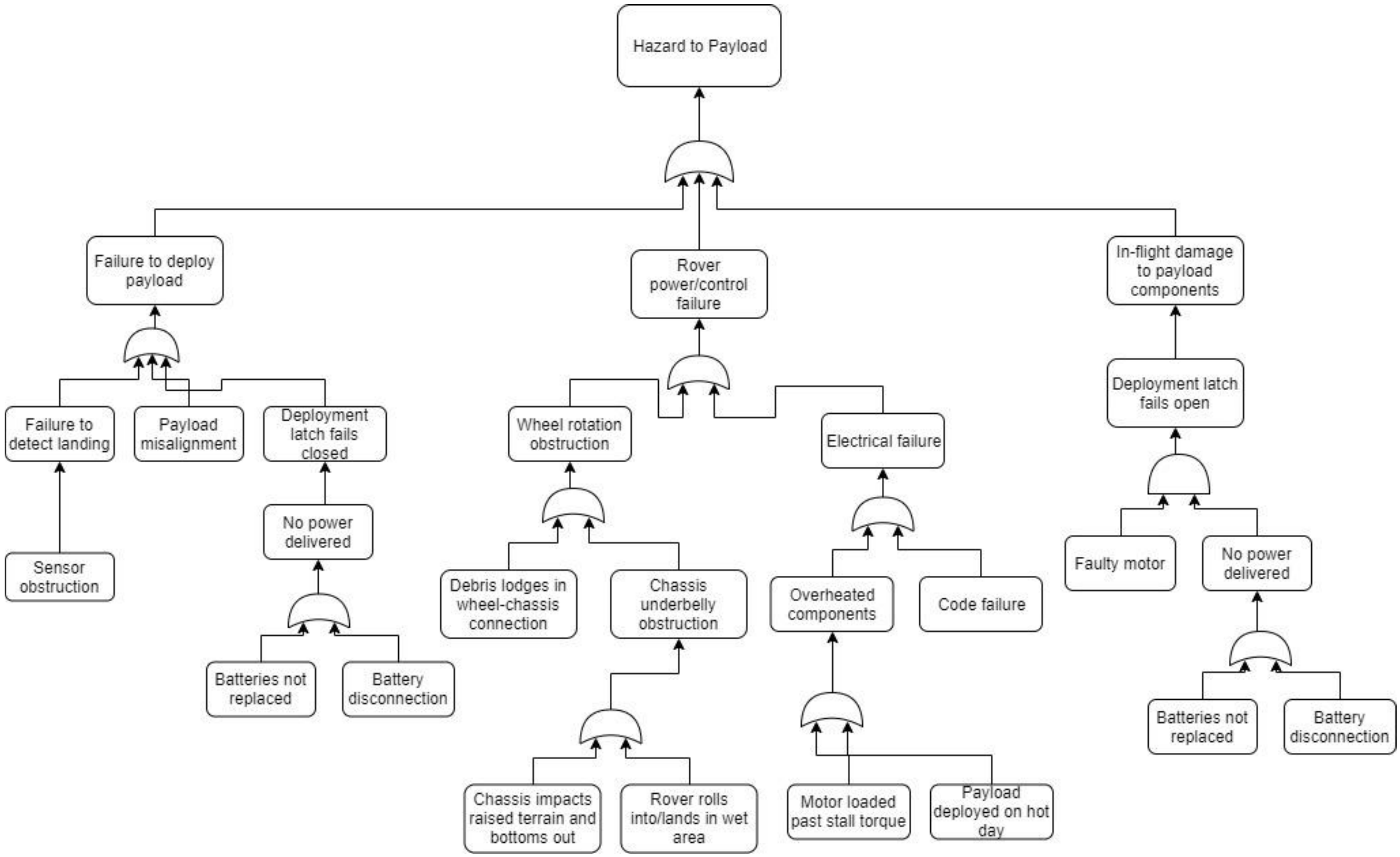


Figure 5-1 Payload Fault Tree Analysis

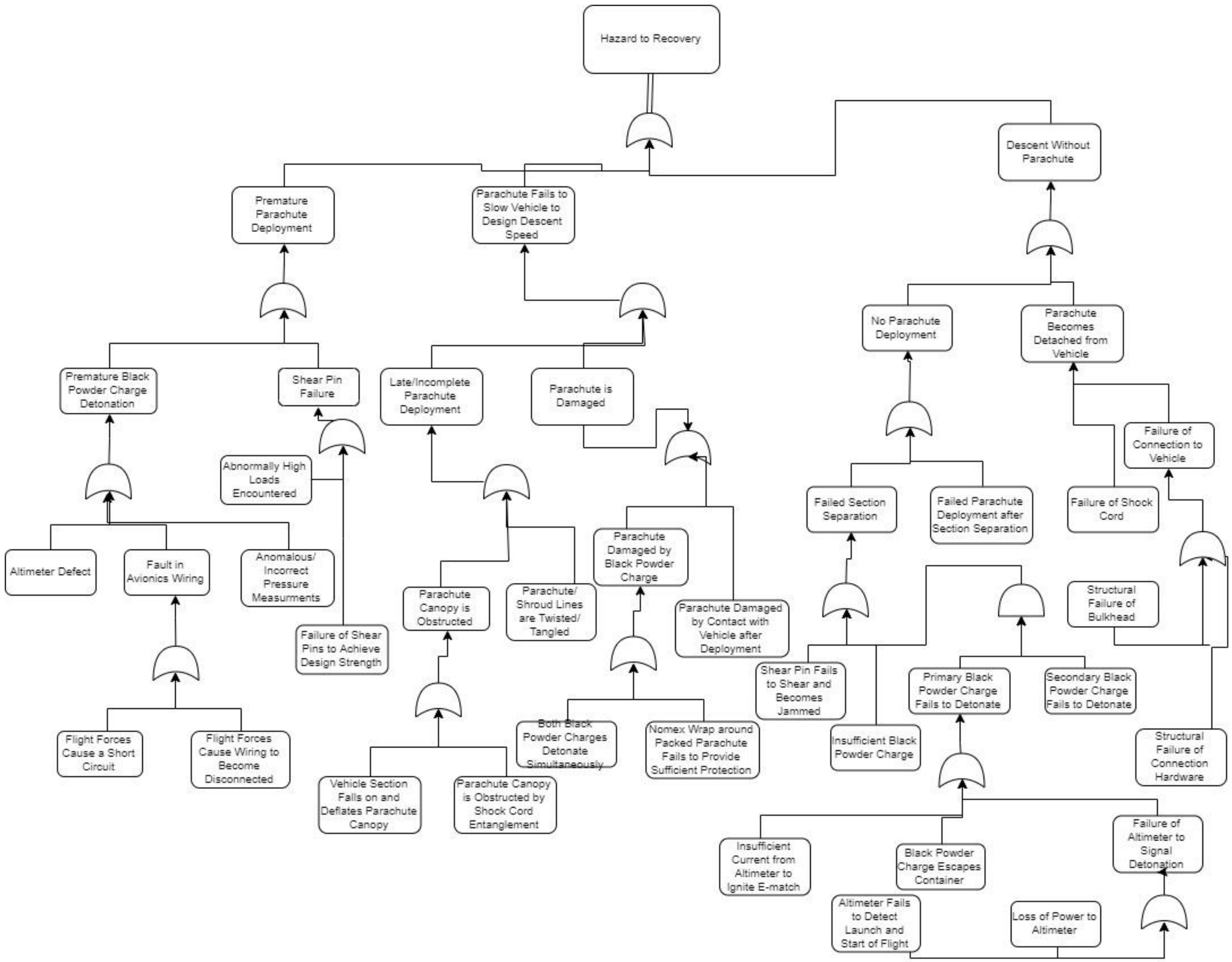


Figure 5-2 Recovery Fault Tree Analysis

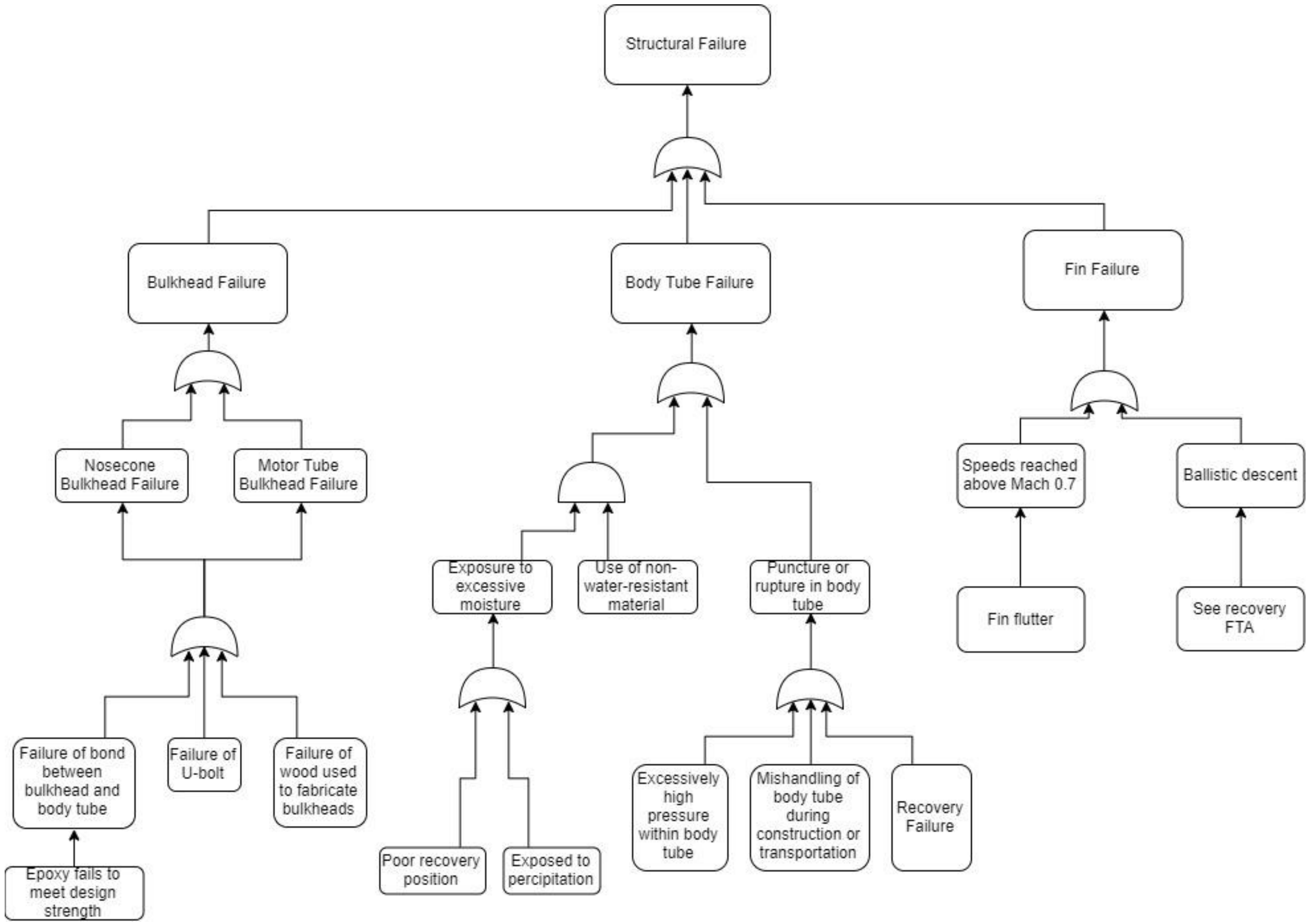


Figure 5-3 Structural Fault Tree Analysis

5.2 Risk Analysis

Table 5-1, below, defines levels of severity as associated with likeliness of occurrence.

Table 5-1 Hazard and Likelihood Classifications

		Level of Severity			
		1 Low Risk	2 Medium Risk	3 High Risk	4 Severe Risk
Likeliness 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

Hazard and likelihood classifications are used to determine the importance of mitigation of a system component. Any component with a hazard level of 3+ or likelihood of C+ is required to be mitigated in some form to decrease the likelihood of the failure occurring and the severity of the outcome. Classifications are detailed below:

Hazard Level 1:

- No personnel injury occurs
- Damage done to launch vehicle is reversible
- Mission success

Hazard Level 2:

- Any personnel injury can be treated with minimal first aid
- Damage done to launch vehicle is repairable
- Partial mission failure; successful flight

Hazard Level 3:

- Moderate personnel injury; manageable with launch field first aid

- Damage done to launch vehicle is repairable but in poorer condition than before flight
- Partial mission failure; partially successful flight

Hazard Level 4:

- Personnel severely injured or killed; hospitalization is required
- Damage done to launch vehicle is irreparable
- Total mission failure; catastrophe

Table 5-2 Project Risk Analysis

Risk	Likelihood	Impact	Mitigation	Mitigation Cost
Overspending	Low	Low	Sticking to a strict budget	Fundraising may become more necessary
Delays	Medium	Medium	Rigid and realistic time management	Loss in flexibility
Lack of resources	Low	High	Clearly communicated needs of materials	Unexpected costs and subsequent delays
Improper Design	Medium	High	Cooperative review of the rockets systems on a regular basis	Need for an expensive and time-consuming redesign late in the competition
Motor Failure	Low	High	Buy reputable and quality motors	An increase in motor costs
Power Supply Failure	Low	High	Using a large power supply	Rocket has a higher weight
Structural Failure	Low	High	Rigorous review of design and testing	Delays due to testing

5.3 Personnel Hazard Analysis

Table 5-3, below, shows identified hazards to personnel throughout the course of the project.

Table 5-3 Personnel Hazard Analysis

Hazard	Causal Factors	Effects	LS Rating	Mitigation
Lack of visibility	Cloud cover	Ground crew and spectators at risk of descending rocket components	4C	In compliance with NAR Safety Code section 6, launches shall not occur during or following adverse weather or field conditions. The RSO has the final say on the status of launches and the team will follow the guidelines set by the RSO.
Team member/spectator tripping	Slippery ground surfaces from heavy precipitation	Injury of recovery sub-team members in the process of retrieving the separated rocket components	2B	
	Non-level ground surfaces		2A	
Unpredictable rocket flight and descent paths	High velocity crosswinds	Ground crew and spectators at risk of descending rocket components. Injury of recovery sub-team members in the process of retrieving separated rocket components	4C	
	Severe humidity			
Live black powder charges within the rocket after descent	Severe humidity affecting altimeter calibration	Injury of recovery sub-team members in the process of retrieving the separated rocket components	4A	Redundant altimeters will be used to ensure all charges detonate. Members will avoid generation of static and will wear PPE to protect against flames.
Inhalation, ingestion, or contact with black powder	Improper or no personal protective equipment used	According to GOEX BP SDS: Skin and respiratory irritation	2C	Members working with black powder will wear nitrile gloves, safety glasses, and face masks.

Inhalation, ingestion, or contact with large airborne particulates	Uncontrolled sanding or use of Dremel	Eye or skin abrasion	3B	A vacuum will be secured near the manufacturing surface to collect hazardous particulates.
Inhalation, ingestion, or contact with epoxy resin or hardener	Working with epoxy, insufficient ventilation	According to 206 SDS: Lung irritation and lightheadedness of team members working with/around epoxy	2B	An oxygen monitor will be used to determine when additional ventilation is necessary.
Machining accident	Improper use of power tools	Serious injury and bodily harm	4A	All team members shall be trained in the use of a power tool before use; use of proper PPE (no gloves) will be enforced
Fire and/or explosion	Premature ignition of black powder	According to GOEX BP SDS: Severe burns	4B	The minimal amount of black powder necessary to ensure full separation of vehicle components will be used; members will be equipped with proper PPE; black powder will be kept away from sources of static and heat; only necessary personnel will be present for black powder installation
	Misuse of motor components during assembly	According to Estes SDS: Severe burns, injury from shrapnel impact	4A	Only NAR and Tripoli officials will assemble and work with motor components. Static and

				excessive heat will be minimized during transport and storage.
Contact with soldering iron, hot solder, or solder fumes	Working with solder and soldering tools	Mild to moderate burns	3C	Members working with solder wear protective face masks and are instructed how to properly hold the soldering iron to eliminate burns.
		Lung irritation	2B	

5.4 Failure Modes and Effects Analysis (FMEA)

Failure mode and effect analysis (FMEA) tables allow clear identification of failure possibilities, failure causes, and their effect on the project. They also assign a likelihood-severity (LS) rating, identify how the causes will be mitigated, and preliminary verification steps. When all possible failures are identified the team can effectively mitigate such possibilities which will minimize personnel injuries and flight failures. The tables are organized into six columns: mission component, hazard type, causes, effects, likely-severity rating, and mitigation.

5.4.1 Recovery

Table 5-4 Recovery FMEA Tables

Component	Hazard	Cause	Effect	LS Rating	Mitigation
Shear pins	Delayed/lack of shear pin breakage	Shear pins of excessive diameter	Ballistic descent; excessive kinetic energy upon landing	4A	Shear pins used will be 2-56, in the range for ability to break.
Shear pins	Delayed/lack of shear pin breakage	Insufficient black powder charges	Ballistic descent; excessive kinetic energy upon landing	4B	Black powder will be calculated with equations from and will be tested via ejection testing well before launch.
Shear pins	Delayed/lack of shear pin breakage	Shear pin ejection	Premature section separation; final apogee lower than expected	3A	Holes drilled to accommodate shear pins will be the same diameter as shear pins to ensure snug fit; tape will be used to secure loose shear pins.
Shear pins	Premature shear pin breakage	Shear pins of insufficient diameter	Premature section separation; final apogee lower than expected	3A	Shear pins used will be 2-56, in the range for ability to break.

		Premature black powder detonation		3A	Altimeters will be tested for functionality using pressure vessel prior to installation on the launch vehicle.
Chute assembly (parachutes, shock chords, U bolts)	Shock chord disconnection	U-Bolt shear	Ballistic descent; excessive kinetic energy upon landing	4A	Shear tests will be performed on bulkheads and U-bolts.
		U-Bolt disconnection			
		Bulkhead crack			See section 3.1.4 for bulkhead analysis
		Bulkhead separation			
	Limited parachute deployment	Shock chord tangling	Excessive kinetic energy upon landing	3A	At least 3 team members including the recovery lead and the safety officer will be present for the folding and installation of shock chords.
		Parachute collapse			
	Late parachute deployment	Delayed E-match burning	Large opening shock; damage to airframe in the form of "zippering"; potential causal factor for shock cord disconnection	3B	Altimeters will be tested before launch; black powder and E-match assembly shall be inspected by the safety officer and recovery lead during assembly to ensure proper packing.
		Defective altimeter			
		Delayed black powder detonation			
	No parachute deployment	Lack of section separation		4A	Battery voltage will be checked prior to

		No power supplied from battery to altimeter	Ballistic descent; excessive kinetic energy upon landing		battery installation and again when altimeters are armed.
	False apogee detection	Incorrect pressure readings	Parachute deployment during ascent; damage to airframe or recovery harness	2C	Altimeters will be tested at least 3 times prior to launch: in the lab, during dry runs of checklist items, and directly before launch.
		Defective altimeter			
	“Zippered” body tube	Shock chord imparts excessive force to airframe	Complete airframe failure; inability to relaunch rocket	4B	Fiberglass will be used in the full-scale airframe; additional bulkheads will be added to reinforced couplers for extra support against zippering.
Altimeters	Overpressure in parachute bay	Detonation of primary charge causes detonation of secondary charge	Airframe and recovery harness damage	4A	Place black powder charges on opposite sides of the bulkhead to reduce the chance of sympathetic detonation.
		Altimeter error			
	Avionics exposed to black powder charge	AV bulkhead did not provide proper seal	Electronics damage or destruction; altimeter failure in flight	3C	Test seal prior to launch.
Recovery	Poor GPS Signal	Signal interference	Inability to locate rocket in the event that visual tracking is insufficient	3C	GPS signal will be confirmed before inserting the AV sled into the rocket, and
		Insufficient range capabilities of GPS			

					again before taking the rocket to the launch pad.

5.4.2 Structures

Table 5-5 Structures FMEA Tables

	Hazard	Cause	Effect	LS Rating	Mitigation
Bulkheads	Nosecone bulkhead failure	Improper epoxy application	Nose cone enters a ballistic descent during recovery	3B	Ensure that bulkhead is manufactured and installed properly.
	Motor tube bulkhead failure	Improper epoxy application	Motor separates from launch vehicle	4B	Ensure that bulkhead is manufactured and installed properly.
	AV bay bulkhead failure	Excessive loading during recovery events	Ballistic descent of all body section not attached to main parachute shock cord	4B	All bulkheads will be designed such that they can withstand the loads caused by recovery events with a factor of safety of 2 or higher.
Airframe	Elevated pressure within body tube	Undrilled or too small pressure ports	Airframe rupture	4A	All pressure ports will be sized and located to optimize airflow.
	Launch Vehicle exposure to excessive moisture	Adverse weather, bad recovery position	Weakens structural integrity of launch vehicle	3C	Use materials that are moisture resistant in launch vehicle construction, do not launch in rain.
	Motor separation from motor tube	Excessive loading on epoxy	Dangerous and unpredictable flight	4A	Epoxy resin to hardener ratio will be 1:5 to meet load specifications.
	Ballistic Impact	Recovery failure		2B	Examine fins prior to flight; ensure recovery

			Airframe, fins, and/or nosecone rupture		system is sufficient to land rocket at desired speed.
	Launch vehicle sections colliding during descent	Shock cord improperly sized		3B	Shock chord will be sized such that body sections will be sufficiently distributed along recovery harness.

5.4.3 Payload

Table 5-6 Payload FMEA Tables

Component	Hazard	Cause	Effect	LS Rating	Mitigation
Rover (chassis, drivetrain, control electronics)	Rover movement impeded by terrain	Impacting raised terrain thus bottoming out the rover	Rover unable to move; inability to complete mission	3D	The rover will be built with a chassis high enough above the ground so that field bumps present little issue.
		Rover becomes entangled with debris on field	Rover movement impeded; inability to complete mission or severe delay in mission completion	4B	See section 5.5
		Landing in or entering an irrigation ditch	In the event of rover inability to exit ditch; inability to complete mission		
		Drivetrain becomes embedded in simulated ice material	Rover movement impeded, rover unable to transport material required distance	3C	Rotating components will be shielded from simulated lunar ice and debris.

	Damage to rover control system and circuitry	Exposure of rover to significant amounts of water	Shorting of circuitry leading to loss of rover control and permanent damage to affected components	2C	See section 5.5
		Overheating from environmental or operational factors	Mission delay, possibility of mission failure in the event of severe or prolonged thermal exposure	2B	All rover components will be tested for their ability to withstand the mission's voltage and current demands.
		Damage to wiring connections during launch, landing, or rover operations	Partial rover failure resulting in partial or full inability to complete mission. Shock hazard for recovery personnel	2B	A system to secure the rover during flight will be redundant and immobile until deployment.
	Rover software failure	Edge cases	Unpredictable program response, crashing of control system	2C	Code will be debugged and tested prior to flight testing.
		Improperly written or implemented software	Erratic functionality or no functionality of rover during attempted operation	2C	
	Rover flipped during operation	Sudden stopping of the rover; uneven terrain encountered	Rover unable to continue movement, unable to continue mission	2C	Instruct rover operator of this risk; develop braking procedure to mitigate this hazard
	Damage to rover chassis	Loads during launch, landing, or deployment exceed design loads	Rover is unable to be reused without significant repair; unable to complete mission	2B	The rover chassis will be made of treated plywood and the rover itself will be securely fastened during flight.
	Rover drivetrain jams	Debris lodged in drivetrain elements	Rover motion impeded; unable to continue	2C	Chassis and wheel connection will be shielded

			mission or delay in mission completion		by a piece sitting flush to the rover.
		Motor stall torque reached during driving		2B	The motor will have a higher stall torque than peak operating torque during driving.
	Separation of rover power transmission components (wheels, stabilizing arm, etc.)	Terrain impacts exceeding maximum design loading criteria	Rover damaged; unable to continue mission	2A	During rover fabrication, multiple team members will confirm securement of rover components.
		Improper securing of attachment points	Rover unintentionally disassembled; unable to continue mission	2A	
	Payload separation from rocket during recovery sequence	Failure of payload retention system combined with unusual attitude of the forward recovery section	Ballistic landing of payload; hazard to personnel or bystanders	4A	The latch will be designed to retain greatest anticipated load with a safety factor of 2.
Payload deployment/ retention	Failure to detect landing	Flight sequence thresholds not detected	Payload deployment sequence not initiated	2B	The accelerometer will be on-board the subscale to acquire acceleration data. This will allow for appropriate threshold levels to be set on the microcontroller.
	Failure to initiate deployment operation	Payload deployment system wiring fault	Payload does not deploy	2B	The deployment mechanism's wiring and programming will be
		Programming fault			

					verified by pre-flight testing.
	Deployment motor stall	Payload misalignment	Damage to or overheating of deployment motor; inability to complete payload deployment	2B	Radial supports in payload bay will guide the rover.
		Excessive drivetrain friction			Rover wheels will be loosely wrapped in fabric.
	In-flight failure of payload retention latch	In-flight loads exceed expected design loads	Payload unsecured within rocket; potential for damage to airframe and payload, potential for payload separation leading to ballistic payload descent	4A	Push buttons at aft end of payload bay will activate latch unlocking motor.
	Slipping or binding of deployment system power train	Loss of drivetrain alignment	Inability to drive deployment mechanism; inability to deploy payload	2B	Power transmission mechanism will be placed under housing.
		Lodging of foreign debris in drivetrain			
	Failure of the latch to actuate during payload deployment sequence	Control system fails to actuate latch	Payload trapped in rocket; unable to continue mission	2B	Latch actuation will be tested prior to launch.
		Binding or debris in latch mechanism			Latch will be sufficiently powerful to dislodge any debris.
Lunar Ice Collection System	Insufficient battery longevity for mission completion	Incorrect battery sizing	Rover loses power before mission completion	3A	Battery drainage tests will be performed to ensure power supply system meets pad and flight time requirements.
		Unanticipated current draw			Components will utilize standby mode when not in operation to conserve battery capacity.

	Scoops fail to deploy	Software error	Scoops are unable to collect sample, resulting mission failure	2B	Software will be tested to remove bugs.
	Excessive loading on scoop	Signal error	Scoop breakage	2B	Wiring and avionics will be tested for continuity.
		Mechanical failure			Clearance of gearing will be ensured before activation.
		Unexpected Ground Resistance			Scoop prototypes will be tested in sample site analog prior to launch.
	Inadequate sample collected	Mechanical failure of the scoop mechanism	Scoops are unable to collect sufficient material for analysis resulting in mission failure	2B	The scoop is designed to withstand varied loading conditions.
		Incorrect design volume			The scoop will have more than 10 mL internal volume for required sample size.
	Damage to scoop during deployment	Mechanical blockage; excessive loading on scoop	Rover unable to collect sample	2B	The scoop will not be deployed if doing so would damage the payload or launch vehicle.

5.4.4 Aerodynamics and Propulsion

Table 5-7 Aerodynamics and Propulsion FMEA Tables

Component	Hazard	Cause	Effect	LS Rating	Mitigation
Trajectory	Undesirable flight trajectory during flight immediately after rail exit	Unexpected large gust of wind	Apogee change, potential loss of launch vehicle control, potential loss of launch vehicle if severe enough	4A	Ensure stability meets minimum requirements of 2.0 at rail exit; Determine wind gust

					speeds and do not launch in sustained wind over 20 mph.
	Undesirable glide trajectory after parachute deployment	Partial/complete failure of parachute deployment, unexpected large gust of wind, weight overestimation	Vehicle will land farther away from launch site, vehicle retrieval becomes complicated, payload is unable to deploy	2C	Ensure that the vehicle weight and parachute selected are compatible; Determine wind gust speeds and do not launch in sustained gusts of anything greater than 20 mph.
	Unintended Thrust Curve	Poorly loading the solid propellent into the motor casing	Unintended flight trajectory and a potentially less than ideal static margin of the vehicle	2C	Motor assembly carried out under supervision of experienced Tripoli L3 mentors. Motor will be examined beforehand for external flaws
		Motor nozzle failure			
Motor	Motor failure during burn	Fuel grain defect	Loss of launch vehicle	4A	Using Aerotech motors, known to be reliable motors; Assembly supervised by Tripoli mentors experienced with experimental motors.
		Incorrect motor assembly			
	Motor fails to ignite	Faulty igniter	Launch vehicle fails to leave launch rail	1C	The team will bring multiple igniters to the launch site and replace the faulty
		High humidity			

					igniter using a contingency checklist. Determine humidity and do not launch in high humidity.
	Fin flutter	Mach number of launch vehicle greater than 0.7	Fin structure damage and alteration of aerodynamic stability	4A	Mach number will be verified in RockSim to ensure peak Mach less than 0.7. Fin assembly will be tested beforehand for proper angles and secure attachment to fin can.
		Improper fin installation			

5.5 Environmental Hazard Analysis

The team has performed a preliminary environmental hazard analysis on the risks to the environment due to the project and vice versa.

Hazard	Cause	Effect	LS Rating	Mitigation
Risks to Environment Due to Project				
Fire at launchpad	Motor ignition and exhaust	Ignition of launch field	3C	A metal plate is used at the base of the launch rail to deflect all flames away from the dry grass of the field.
Fire at recovery site	Overheated electronics		3B	Electronics will be chosen on their ability to withstand the current and voltage requirements of the project.
	Leftover ignited black powder		4A	See section 5.3
Disposal of nonbiodegradable waste	Unrecovered launch vehicle components	Harm to local wildlife and waterways	2C	Recovery teams are sent out to assess damage and recover all system components when safe to do so.

	Waste from launch preparation abandoned			All team members participate in post-launch cleanup to eliminate littering.
Rocket landing in trees or other natural features	Recovery system failure	Damage to natural features	2B	See section 5.3
Chemical exposure	Battery leakage	Pollution to launch area	1A	Batteries are secured to an enclosed AV Bay, ensuring that if leakage occurs, nothing escapes.
Risks to Project Due to Environment				
Waterlogged altimeters	Water Exposure	Altimeters will become inoperable and must be replaced	2B	The launch vehicle will be directed away from water features
Waterlogged payload electronics		Payload electronics will become inoperable and must be replaced	2B	
Saturated body tube		Fundamental rocket structure compromised; inability to relaunch rocket	4A	Full-scale body tube will be constructed of water-resistant fiberglass
Payload component rust		Payload components will become damaged and must be replaced	3A	Payload chassis will be treated to be water resistant and use non-rusting materials; any water collected will be cleaned immediately after recovery
Unintended launch trajectory	High wind speed	Failure of vehicle to reach intended apogee	2C	See section 5.4.1
Increased recovery distance		High drift rate during descent	2C	The team will not launch in winds over 20 mph

6. Project Plan

6.1 NASA Requirements Verification Matrix

Table 6-1, below, defines the fields present in Table 6-2, NASA Requirements Verification Matrix.

Table 6-1 Field Definition for Requirements Verification Matrix

Field	Definition
Req No.	Identification number associated with requirement.
Shall Statement	Statement of what the product shall accomplish.
Success Criteria	Description of how the team will define a met requirement.
Verification Method	How the requirement is verified. By Inspection, Demonstration, Test, or Analysis.
Subsystem Allocation	Indicates which subsystems are responsible for meeting the requirement.
Results	Results of the verification.

Table 6-2, below, shows the NASA defined requirements for the NASA Student Launch project.

Table 6-2 NASA Requirements Verification Matrix

Req No.	Shall Statement	Success Criteria	Verification Method	Subsystem Allocation	Status
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	The students of the High-Powered Rocketry Club at NC State design and implement a solution to the requirements listed in this section.	Inspection	Project Management	All documents and analysis has been executed by student team members.

	submit new work. Excessive use of pas work will merit penalties.				
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, consisting of the team lead, vice president, treasurer, coordination lead, safety officer, outreach lead, web administrator, and social media lead will manage the project planning tasks listed in this requirement.	Inspection	Project Management	See section 6 for the 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.	The team lead will identify and report Foreign National (FN) team members by November 1, 2019 with the submission of the PDR milestone document.	Inspection	Project Management	No FN team members are attending competition.
NASA 1.4	The team SHALL identify all team members attending launch week activities by the Critical Design Review (CDR).	The team lead will identify and report team members attending launch week activities by January 10, 2020 with the submission of the CDR milestone document.	Inspection	Project Management	Competition attending team members will be identified by CDR milestone delivery.
NASA 1.4.1	Team members attending competition SHALL include students actively engaged in the project throughout the entire year.	The project management team will identify actively engaged team members to attend launch week activities.	Inspection	Project Management	Student's participation in the project will determine who

					attends the competition.
NASA1.4.2	Team members SHALL include one mentor (see requirement 1.13).	The team lead will invite the mentors listed in section 1.1.2 to attend launch week activities.	Inspection	Project Management	Alan Whitmore and Jim Livingston will attend launch week activities.
NASA1.4.3	Team members SHALL include no more than two adult educators.	The team lead will invite the team's adult educator to attend launch week activities.	Inspection	Project Management	Dr. Felix Ewere is serving as the team's adult educator.
NASA1.5	The team SHALL engage a minimum of 200 participants in educational, hands-on science, technology, engineering, and mathematics (STEM) activities, as defined in the STEM Engagement Activity Report, by FRR.	The outreach lead will identify K-12 student groups to implement STEM engagement plans with throughout the project lifecycle.	Inspection	Project Management	Outreach forms will be sent to the appropriate NASA project management team member.
NASA1.6	The team SHALL establish a social media presence to inform the public about team activities.	The web administrator and social media lead will cooperate to develop an engaging and educational social media presence on various platforms including, but not limited to: club website, Facebook, Instagram, and Twitter.	Inspection	Project Management	The team's social media lead will maintain the social media accounts.
NASA1.7	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,	The team lead will send all completed documents to the NASA project management team prior to each deadline. In the event that the deliverable is too large, the web administrator will post the	Inspection	Project Management	The team lead emails each deliverable to the appropriate NASA project management team member.

	inclusion of a link to download the file will be sufficient.	document on the team's website and the team lead will send the NASA project management team a link to the document.			
NASA1.8	All deliverables SHALL be in PDF format.	The team lead will convert all deliverables to PDF format prior to submission.	Inspection	Project Management	This report is submitted in PDF format.
NASA1.9	In every report, the team SHALL provide a table of contents including major sections and their respective sub-sections.	The team lead will manage the Table of Contents in each milestone report.	Inspection	Project Management	See the Table of Contents at the start of this document.
NASA1.10	In every report, the team SHALL include the page number at the bottom of the page.	The team lead will identify the page numbers in each milestone report.	Inspection	Project Management	Page numbers are included in the bottom right corner of each page.
NASA1.11	The team SHALL provide any computer equipment necessary to perform a video teleconference with the review panel.	The team lead will acquire the necessary equipment to communicate with the NASA project management team through teleconference.	Inspection	Project Management	The team lead organizes required equipment for teleconferencing.
NASA1.12	The team SHALL use the launch pads provided by Student Launch's launch services provider.	The aerodynamics lead will design a launch vehicle to be launched from either an 8 foot 1010 rail or a 12 foot 1515 rail. The Structures lead will fabricate said launch vehicle.	Inspection	Aerodynamics; Structures	See section 3.1.
NASA1.13	The team SHALL identify a "mentor."	The team lead will identify community members qualified to mentor team members through the design process.	Inspection	Project Management	See section 1.1.2 for mentor listing and contact information.

NASA2.1	The launch vehicle SHALL deliver the payload to an apogee altitude between 3,500 and 5,500 feet above ground level (AGL).	The aerodynamics subsystem team designs a rocket to launch between 3,500 and 5,500 feet AGL. The team then constructs the vehicle as designed and the launch vehicle flies between 3,500 and 5,500 feet AGL.	Analysis; Demonstration	Aerodynamics	See section 3.3.2 for current altitude predictions.
NASA2.2	The team SHALL identify the target altitude goal at the PDR milestone.	The aerodynamics subsystem team reports the altitude goal in the PDR milestone report and is sent to the NASA project management team by November 1, 2019.	Inspection	Aerodynamics	See section 3.3.1 for official target altitude.
NASA2.3	The launch vehicle SHALL carry one commercially available, barometric altimeter for recording the official altitude.	The recovery subsystem team chooses a commercially available, barometric altimeter to be used in the launch vehicle.	Inspection	Recovery	See section 3.2.3 with the leading design choice for altimeter.
NASA2.4	The launch vehicle SHALL be designed to be recoverable and reusable.	The recovery subsystem team designs a recovery harness system that will allow the launch vehicle to be recovered upon ground impact with minimal damage.	Demonstration	Recovery	See section 3.2.3 with the leading recovery device design.
NASA2.5	The launch vehicle SHALL have a maximum of four (4) independent sections.	The aerodynamics and recovery subsystem teams design a launch vehicle that has fewer than four (4) independent sections.	Inspection	Aerodynamics; Recovery	See section 3.2.3 with the leading recovery device design.

NASA2.5.1	Couplers which are located at in-flight separation points SHALL be at least one (1) body diameter in length.	The aerodynamics subsystem team designs a rocket with couplers at in-flight separation points at least one body diameter in length. The structures subsystem team construct the couplers in the correct lengths.	Inspection	Aerodynamics; Structures	See section 3.1.4 with the leading launch vehicle dimensions.
NASA2.5.2	Nosecone shoulders which are located at in-flight separation points SHALL be at least 1/2 body diameter in length.	The aerodynamics subsystem team designs a rocket with nosecone shoulders at in-flight separation points at least 1/2 body diameter in length. The structures subsystem team construct the couplers in the correct lengths.	Inspection	Aerodynamics; Structures	See section 3.1.4 with the leading launch vehicle dimensions.
NASA2.6	The launch vehicle SHALL be capable of being prepared for flight at the launch site within two (2) hours of the time the Federal Aviation Administration flight waiver opens.	The project management and safety teams develop launch day checklists that can be executed in under two (2) hours.	Demonstration	Project Management; Safety	Checklists included in the CDR milestone review will be executable within 2 hours.
NASA2.7	The launch vehicle and payload SHALL be capable of remaining in launch-ready configuration on the pad for a minimum of two (2) hours without losing the functionality of any critical on-board components.	The project management and safety teams monitor the selected power supplies for each on-board component and test to verify functionality after over two (2) hours.	Demonstration	Project Management; Safety	See flysheet for estimated pad stay time.

NASA2.8	The launch vehicle SHALL be capable of being launched by a standard 12-volt direct current firing system.	The project management and safety teams choose the motor ignitor that can be ignited from a 12-volt direct current firing system.	Demonstration	Project Management; Safety	See section 3.1.4 for current launch vehicle design.
NASA2.9	The launch vehicle SHALL require no external circuitry or special ground support equipment to initiate launch.	The project management and safety teams limit the launch vehicle such that it has no external circuitry or ground support equipment.	Demonstration	Project Management; Safety	See section 3.1.4 for current launch vehicle design.
NASA2.10	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 team selects a solid motor propulsion system.	Inspection	Aerodynamics	See Section 3.1.5 with the leading motor selection.
NASA2.10.1	Final motor choices SHALL be declared by the Critical Design Review (CDR).	The aerodynamics team selects and reports the final motor choice by January 10, 2020.	Inspection	Aerodynamics	The motor will be chosen prior to CDR delivery.
NASA2.10.2	Any motor change after CDR SHALL be approved by the NASA Range Safety Officer (RSO) and SHALL only be approved if the change is for the sole purpose of increasing the safety margin.	The aerodynamics team selects and reports the final motor choice by January 10, 2020.	Demonstration	Aerodynamics	The motor will be chosen prior to CDR delivery.
NASA2.11	The launch vehicle SHALL be limited to a single stage.	The aerodynamics team designs a launch vehicle with a single stage.	Inspection	Aerodynamics	The motor will be chosen prior to CDR delivery.

NASA2.12	The total impulse provided by a College or University launch vehicle SHALL not exceed 5,120 Newton-seconds (L-class).	The aerodynamics team chooses an L-class motor for the full-scale launch vehicle.	Inspection	Aerodynamics	The motor will be chosen prior to CDR delivery.
NASA2.13	Pressure vessels on the vehicle SHALL be approved by the RSO.	The structures lead provides the necessary information on any on-board pressure vessels to the NASA RSO and home field RSO.	Inspection	Structures	The current design does not include pressure vessels.
NASA2.13.1	Pressure vessels on the vehicle SHALL have a minimum factor of safety (Burst or Ultimate pressure versus Max Expected Operating Pressure) of 4:1 with supporting design documentation included in all milestone reviews.	The structures lead provides the necessary information on any on-board pressure vessels to the NASA RSO and home field RSO.	Analysis	Structures	The current design does not include pressure vessels.
NASA2.13.2	Pressure vessels on the vehicle 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 provides the necessary information on any on-board pressure vessels to the NASA RSO and home field RSO.	Analysis	Structures	The current design does not include pressure vessels.
NASA2.13.3	Pressure vessels on the vehicle SHALL be described, including the application for which the tank was designed and the history of the tank.	The structures lead provides the necessary information on any on-board pressure vessels to the NASA RSO and home field RSO.	Inspection	Structures	The current design does not include pressure vessels.
NASA2.14	The launch vehicle SHALL have a minimum static stability margin of 2.0 at the point of rail exit.	The aerodynamics team designs a launch vehicle	Analysis; Inspection	Aerodynamics	See section 3.1.4 for current launch vehicle design.

		with a minimum static stability margin of 2.0.			
NASA2.15	Any structural protuberance on the rocket SHALL be located aft of the burnout center of gravity.	The aerodynamics team designs a launch vehicles with all structural protuberances aft of the burnout center of gravity. The structures team verifies that all structural protuberances are aft of the burnout center of gravity upon construction.	Analysis; Inspection	Aerodynamics; Structures	See section 3.1.4 for current launch vehicle design.
NASA2.16	The launch vehicle SHALL accelerate to a minimum velocity of 52 fps at rail exit.	The aerodynamics team designs a launch vehicle with a minimum velocity of 52 fps at rail exit.	Analysis	Aerodynamics	See section 3.1.4 for current launch vehicle design.
NASA2.17	The team SHALL successfully launch and recover a subscale model of the rocket prior to CDR.	The structures team leads the construction of the subscale model of the launch vehicle. The project management and safety teams lead the launch of the subscale model of the launch vehicle before January 10, 2020.	Demonstration	Project Management; Safety; Structures	The subscale launch is scheduled for November 16, 2019.
NASA2.17.1	A full-scale model SHALL not be used as the subscale model.	The project management team verifies the subscale model is a different size than the full scale launch vehicle.	Inspection	Project Management; Safety	The subscale launch is scheduled for November 16, 2019.
NASA2.17.2	The subscale model SHALL carry an altimeter capable of recording the model's apogee altitude.	The recovery team chooses an altimeter to record the subscale model's altitude.	Demonstration	Recovery	The subscale launch is scheduled for November 16, 2019.

NASA2.17.3	The subscale model SHALL be a newly constructed rocket, designed and built specifically for this year's project.	The project management team acquires the materials necessary to construct a new launch vehicle for this year's project.	Inspection	Project Management	The subscale launch is scheduled for November 16, 2019.
NASA2.17.4	Proof of a successful flight SHALL be supplied in the CDR report.	The recovery team provides altimeter data from the subscale launch in the CDR by January 10, 2020.	Inspection	Recovery	The subscale launch is scheduled for November 16, 2019.
NASA2.18	The team SHALL execute demonstration flights of the launch vehicle and payload.	The project management team holds to the schedule for the team to be able to launch demonstrations flights for both the vehicle and payload.	Demonstration	Project Management	The vehicle demonstration flight is scheduled for February 22, 2020.
NASA2.18.1	The team SHALL successfully launch and recover their full-scale rocket prior to FRR in its final flight configuration.	The project management team holds to the schedule for the team to be able to launch demonstrations flights for both the vehicle and payload.	Demonstration	Project Management	The vehicle demonstration flight is scheduled for February 22, 2020.
NASA2.18.1.1	During the Vehicle Demonstration Flight (VDF) the vehicle and recovery system SHALL function as designed.	The launch vehicle specific subsystem teams design and construct the launch vehicle as written and the systems function as designed.	Demonstration	Project Management; Safety; Recovery; Structures; Aerodynamics	The vehicle demonstration flight is scheduled for February 22, 2020.
NASA2.18.1.2	The full-scale rocket SHALL be a newly constructed rocket, designed and built specifically for this year's project.	The project management team acquires the materials necessary to construct a new launch vehicle for this year's project.	Inspection	Project Management	The vehicle demonstration flight is scheduled for February 22, 2020.
NASA2.18.1.3.1	If the payload is not flown during the VDF, mass simulators SHALL	The payload team chooses mass simulators to fly in the	Inspection	Payload	The vehicle demonstration

	be used to simulated the payload mass.	vehicle demonstration flight if the payload is not ready for launch.			flight is scheduled for February 22, 2020.
NASA2.18.1.3.2	If the payload is not flown during the VDF, mass simulators SHALL be located in the same approximate location on the rocket as the missing payload mass.	The payload team attaches mass simulators in the same approximate location on the launch vehicle as the missing payload mass.	Inspection	Payload	The vehicle demonstration flight is scheduled for February 22, 2020.
NASA2.18.1.4	If the payload affects the external surfaces of the rocket or manages the total energy of the vehicle, those systems SHALL be active during the full-scale VDF.	The payload team has external components of the payload prepared for the vehicle demonstration flight.	Inspection	Payload; Aerodynamics	The vehicle demonstration flight is scheduled for February 22, 2020.
NASA2.18.1.5	The team SHALL fly the launch day motor during the VDF.	The safety and aerodynamics team work alongside the team's mentors to acquire and use the motor on launch day.	Inspection	Safety; Aerodynamics	The vehicle demonstration flight is scheduled for February 22, 2020.
NASA2.18.1.6	The vehicle SHALL be flown in its fully ballasted configuration during the full-scale test flight.	The aerodynamics team decides on the final ballasting configuration. The structures team constructs the designed ballasting configuration.	Inspection	Aerodynamics; Structures	The vehicle demonstration flight is scheduled for February 22, 2020.
NASA2.18.1.7	The launch vehicle or any of its components SHALL not be modified without the concurrence of the NASA RSO.	If any modifications are necessary after the demonstration flights, the safety and structures team communicate with the NASA RSO prior to making modifications.	Inspection	Safety; Structures	The vehicle demonstration flight is scheduled for February 22, 2020.

NASA2.18.1.8	The team SHALL provide proof of a successful flight in the FRR report.	The recovery team reports the altimeter data from the demonstration flights in the FRR by March 02, 2020.	Inspection	Recovery	The vehicle demonstration flight is scheduled for February 22, 2020.
NASA2.18.1.9	The team SHALL complete the VDF before March 02, 2020.	The project management team holds to the team's schedule and turns in the FRR milestone report by March 02, 2020.	Inspection	Project Management	The vehicle demonstration flight is scheduled for February 22, 2020.
NASA2.18.2	The team SHALL successfully launch and recovery the full-scale rocket containing the completed payload prior to March 23, 2020.	The project management team holds to the team's schedule and launches the payload demonstration flight prior to March 23, 2020.	Inspection	Project Management	The payload demonstration flight is scheduled for February 22, 2020.
NASA2.18.2.1	During the Payload Demonstration Flight (PDF), the payload SHALL be fully retained until the intended point of deployment.	The payload and safety teams design a fail-safe retention system and demonstrate its performance prior to flight.	Demonstration	Safety; Payload	The payload demonstration flight is scheduled for February 22, 2020.
NASA2.18.2.2	The payload flown during the PDF SHALL be the final, active version.	The payload team completes the construction of all payload systems prior to the payload demonstration flight.	Inspection	Payload	The payload demonstration flight is scheduled for February 22, 2020.
NASA2.18.2.4	The PDF SHALL be completed by March 23, 2020.	The project management team manages the schedule such that the payload demonstration flight is complete prior to March 23, 2020.	Inspection	Project Management	The payload demonstration flight is scheduled for February 22, 2020.

NASA2.19	If a reflight is necessary, the FRR Addendum SHALL be submitted by March 23, 2020.	The project management team manages the schedule such that any reflight is complete prior to March 23, 2020.	Inspection	Project Management	No reflight is currently necessary.
NASA2.19.1	If a reflight is necessary, the FRR Addendum SHALL be submitted by March 23, 2020.	The project management team manages the schedule such that the FRR Addendum is complete prior to March 23, 2020.	Inspection	Project Management	No reflight is currently necessary.
NASA2.19.2	The team SHALL successfully execute a PDF to be allowed to fly the payload at competition.	The project management team manages the schedule such that the payload demonstration flight is complete prior to March 23, 2020.	Demonstration	Project Management	The payload demonstration flight is scheduled for February 22, 2020.
NASA2.19.3	The team SHALL not fly the payload at competition if the PDF was unsuccessful.	The project management team manages the schedule such that the payload demonstration flight is complete prior to March 23, 2020.	Demonstration	Project Management	The payload demonstration flight is scheduled for February 22, 2020.
NASA2.20	The team SHALL mark each independent launch vehicle component with the team's launch day contact information.	The project management team marks each independent section of the rocket with the team’s contact information.	Inspection	Project Management	The team lead's contact information will be written on each section of the launch vehicle.
NASA2.21	The team SHALL sufficiently protect all Lithium Polymer batteries from ground impact. The team SHALL mark all Lithium Polymer batteries with brightly colored, clearly marked	The payload team designs a retention system for all LiPo batteries in the payload that protects the battery from impact. The safety team will inspect and test the aforementioned design.	Analysis; Inspection	Safety; Payload	All LiPo batteries will be protected from ground impact.

NASA2.22.1	The launch vehicle SHALL not utilize forward canards.	The aerodynamics team designs a launch vehicle that does not utilize forward canards.	Inspection	Aerodynamics; Structures	See section 3.1.4 for current launch vehicle design.
NASA2.22.2	The launch vehicle SHALL not utilize forward firing motors.	The aerodynamics team designs a launch vehicle that does not utilize forward firing motors.	Inspection	Aerodynamics	See section 3.1.4 for current launch vehicle design.
NASA2.22.3	The launch vehicle SHALL not utilize motors that expel titanium sponges.	The aerodynamics team designs a launch vehicle that does not utilize motors that expel titanium sponges.	Inspection	Aerodynamics	See section 3.1.4 for current launch vehicle design.
NASA2.22.4	The launch vehicle SHALL not utilize hybrid motors.	The aerodynamics team designs a launch vehicle that does not utilize hybrid motors.	Inspection	Aerodynamics	See section 3.1.4 for current launch vehicle design.
NASA2.22.5	The launch vehicle SHALL not utilize a cluster of motors.	The aerodynamics team designs a launch vehicle that does not utilize clustered motors.	Inspection	Aerodynamics	See section 3.1.4 for current launch vehicle design.
NASA2.22.6	The launch vehicle SHALL not utilize friction fitting for motors.	The aerodynamics team designs a launch vehicle that does not utilize friction fitted motors.	Inspection	Aerodynamics	See section 3.1.4 for current launch vehicle design.
NASA2.22.7	The launch vehicle SHALL not exceed Mach 1 at any point during flight.	The aerodynamics team designs a launch vehicle that does not exceed Mach 1 during flight.	Analysis	Aerodynamics	See section 3.1.4 for current launch vehicle design.
NASA2.22.8	Vehicle ballast SHALL not exceed 10% of the total unballasted weight of the launch vehicle.	The aerodynamics team designs the final ballasting configuration. The structures team	Analysis; Inspection	Aerodynamics; Structures	See section 3.1.4 for current launch vehicle design.

		implements the designed ballast configuration.			
NASA2.22.9	Transmissions from onboard transmitters SHALL not exceed 250 mW of power per transmitter.	The recovery and payload teams choose transmitters that do not exceed 250 mW of power per transmitter.	Analysis	Recovery; Payload	TBD
NASA2.22.10	Transmitters SHALL not create excessive interference.	The recovery and payload teams choose transmitters with minimal interference.	Analysis	Recovery; Payload	TBD
NASA2.22.11	The team SHALL not use excessive and/or dense metal in the construction of the launch vehicle.	The structures and payload teams choose materials that use minimal amounts of dense metal.	Inspection	Structures; Payload	See section 3.1.4 for current launch vehicle design.
NASA3.1	The launch vehicle SHALL stage the deployment of its recovery devices.	The recovery team designs a dual deployment recovery system.	Demonstration	Recovery	See Section 3.2.3 for recovery system design.
NASA3.1.1	The main parachute SHALL be deployed no lower than 500 feet.	The recovery team designs a main parachute deployment event no lower than 500 feet.	Demonstration	Recovery	See Section 3.2.3 for recovery system design.
NASA3.1.2	The apogee event SHALL not have a delay longer than 2 seconds.	The recovery team designs a drogue parachute deployment event with an apogee delay of no more than 2 seconds.	Demonstration	Recovery	See Section 3.2.3 for recovery system design.
NASA3.1.3	The launch vehicle SHALL not utilize motor ejection.	The recovery and aerodynamics teams design an ejection system that does not use motor ejection.	Demonstration	Aerodynamics; Recovery	See Section 3.2.3 for recovery system design.
NASA3.2	The team SHALL perform a successful ground ejection test for both the drogue and main	The recovery and safety teams demonstrate the performance of the launch	Demonstration	Safety; Recovery	Ground ejection testing will occur prior to each launch.

	parachutes before both the subscale and full-scale launches.	vehicle's ejection system prior to each launch.			
NASA3.3	Each independent section of the launch vehicle SHALL not exceed a maximum kinetic energy of 75 ft-lbf at landing.	The recovery team designs a launch vehicle that does not exceed a kinetic energy of 75 ft-lbf.	Analysis	Recovery	See Section 3.2.3 for recovery system design.
NASA3.4	The recovery system SHALL contain redundant, commercially available altimeters.	The recovery team designs a redundant recovery electronic system that utilizes commercially available altimeters.	Inspection	Safety; Recovery	See Section 3.2.3 for recovery system design.
NASA3.5	Each altimeter SHALL have a dedicated power supply, and all recovery electronics SHALL be powered by commercially available batteries.	The recovery team designs a redundant recovery electronic system such that both the primary and redundant altimeters are powered by different commercially available batteries.	Inspection	Recovery	See Section 3.2.3 for recovery system design.
NASA3.6	Each altimeter SHALL be armed by a dedicated mechanical arming switch that is accessible from the exterior of the launch vehicle airframe when the rocket is in the launch configuration on the launch pad.	The recovery team designs a redundant recovery electronic system such that both the primary and redundant altimeters are turned on by a mechanical arming switch.	Demonstration	Safety; Recovery	See Section 3.2.3 for recovery system design.
NASA3.7	Each arming switch SHALL be capable of being locked in the ON position for launch.	The recovery team designs a redundant recovery electronic system such that both the arming switches can be locked in the ON position during launch.	Demonstration	Safety; Recovery	See Section 3.2.3 for recovery system design.

NASA3.8	The electronic components of the recovery system SHALL be completely independent of any payload electrical circuits.	The recovery team designs a redundant recovery electronic system such that all recovery electronics are independent of other on-board electronics.	Demonstration	Safety; Recovery; Payload	See Section 3.2.3 for recovery system design.
NASA3.9	Removable shear pins SHALL be used for both the main parachute compartment and the drogue parachute compartment.	The recovery team uses removable shear pins at in-flight separation points.	Inspection	Recovery	See Section 3.2.3 for recovery system design.
NASA3.10	The launch vehicle SHALL not drift more than 2,500 feet radius from the launch pad.	The recovery team designs a recovery system that results in a drift of no more than 2,500 feet from the launch pad. The vehicle demonstration flight results in a drift radius of no more than 2,500 feet.	Analysis; Demonstration	Recovery	See Section 3.2.3 for recovery system design.
NASA3.11	The launch vehicle SHALL make ground impact within 90 seconds after apogee.	The recovery team designs a recovery system that results in ground impact within 90 seconds of apogee.	Analysis; Demonstration	Recovery	See Section 3.2.3 for recovery system design.
NASA3.12	The team SHALL use an electronic tracking device to transmit the position of the tethered vehicle or any independent section to a ground receiver.	The recovery team chooses an electronic tracking device to transmit the position of the launch vehicle.	Demonstration	Recovery	See Section 3.2.3 for recovery system design.
NASA3.12.1	Each untethered section of the rocket SHALL have its own electronic tracking device.	The recovery team chooses an electronic tracking device to transmit the position of the launch vehicle.	Demonstration	Recovery	See Section 3.2.3 for recovery system design.

NASA3.12.2	The electronic tracking device SHALL be fully functional during the official flight on launch day.	The recovery team chooses an electronic tracking device to transmit the position of the launch vehicle.	Demonstration	Recovery	See Section 3.2.3 for recovery system design.
NASA3.13	The recovery system electronics SHALL not be adversely affected by other on-board electronic devices.	The recovery team designs a redundant recovery electronic system such that all recovery electronics are independent of other on-board electronics.	Demonstration	Safety; Recovery; Payload	See Section 3.2.3 for recovery system design.
NASA3.13.1	The recovery altimeters SHALL be located in a separate compartments with the vehicle from any other radio frequency transmitting and/or magnetic wave producing device.	The recovery team designs an avionics bay that is separate from other on-board, transmitting electronics.	Demonstration	Safety; Recovery	See Section 3.2.3 for recovery system design.
NASA3.13.2	The recovery system electronics SHALL be shielded from all onboard transmitting devices.	The recovery team designs an avionics bay that is separate from other on-board, transmitting electronics.	Demonstration	Safety; Recovery	See Section 3.2.3 for recovery system design.
NASA3.13.3	The recovery system electronics SHALL be shielded from all onboard devices which may generate magnetic waves.	The recovery team designs an avionics bay that is separate from other on-board, transmitting electronics.	Demonstration	Safety; Recovery	See Section 3.2.3 for recovery system design.
NASA3.13.4	The recovery system electronics SHALL be shielded from any other onboard electronic devices which may adversely affect the proper operation of the recovery system electronics.	The recovery team designs an avionics bay that is separate from other on-board, transmitting electronics.	Demonstration	Safety; Recovery	See Section 3.2.3 for recovery system design.

NASA4.2	The team SHALL design a system capable of being launched in a high power rocket, landing safely, and recovering simulated lunar ice.	The project management team organizes each of the subsystem teams and works to integrate each subsystem with each other.	Demonstration	Project Management	See section 4.3.4 for current payload design.
NASA4.3.1	The launch vehicle SHALL be launched from the NASA-designated launch area using the provided Launch pad.	The aerodynamics team will design a launch vehicle to be launched from either an 8 foot 1010 rail or a 12 foot 1515 rail. The structures team will fabricate said launch vehicle.	Inspection	Aerodynamics; Structures	See section 3.1.4 for current launch vehicle design.
NASA4.3.2	The team SHALL recover a lunar ice sample from one of five recovery areas.	The payload team designs a lunar ice recovery vehicle that will collect an ice sample from one of the recovery areas.	Demonstration	Payload	See section 4.3.4 for current payload design.
NASA4.3.3	The payload SHALL recover a lunar ice sample of a minimum of 10 milliliters.	The payload team designs a lunar ice recovery vehicle that is capable of storing 10 milliliters of simulated lunar ice.	Demonstration	Payload	See section 4.3.4 for current payload design.
NASA4.3.4	The payload SHALL transport the stored sample 10 linear feet from the recovery site.	The payload team designs a lunar ice recovery vehicle that can travel 10 linear feet after collecting the sample of lunar ice.	Demonstration	Payload	See section 4.3.4 for current payload design.
NASA4.3.5	The team SHALL abide by all FAA and NAR rules and regulations.	The payload team designs a lunar ice recovery vehicle alongside the safety team that abides by all FAA and NAR rules.	Demonstration	Safety; Payload	See section 4.3.4 for current payload design.

NASA4.3.6	The team SHALL not deploy the payload via black powder charges after ground impact.	The payload team designs a deployment system that does not utilize black powder charges after ground impact.	Inspection	Safety; Payload	See section 4.3.4 for current payload design.
NASA4.3.7	The payload SHALL be fully retained until it is deployed as designed.	The payload team designs a payload retention system that functions as designed.	Demonstration	Safety; Payload	See section 4.3.4 for current payload design.
NASA4.3.7.1	The team SHALL design a mechanical retention system.	The payload team designs a mechanical payload retention system.	Inspection	Safety; Payload	See section 4.3.4 for current payload design.
NASA4.3.7.2	The retention system SHALL be designed to successfully endure flight forces.	The payload team designs a payload retention system designed to withstand flight forces.	Analysis	Safety; Payload	See section 4.3.4 for current payload design.
NASA4.3.7.3	The retention system SHALL be a fail-safe design.	The payload team designs a fail-safe payload retention system.	Demonstration	Safety; Payload	See section 4.3.4 for current payload design.
NASA4.3.7.4	The retention system SHALL not exclusively use shear pins as a method of retention.	The payload team designs a payload retention system that does not use shear pins exclusively.	Inspection	Payload	See section 4.3.4 for current payload design.
NASA4.4.1	If jettisoned during the recovery phase, the payload SHALL receive real-time RSO permission prior to initiating the jettison event.	The safety team does not jettison the payload until real-time RSO permission is granted.	Demonstration	Safety; Payload	See section 4.3.4 for current payload design.
NASA4.4.2	If jettisoned during the recovery phase and if the payload is a UAV, the payload SHALL be tethered to the launch vehicle until the RSO has given permission to release the UAV.	The safety team does not jettison the payload until real-time RSO permission is granted.	Demonstration	Safety; Payload	See section 4.3.4 for current payload design.
NASA4.4.3	If a UAV is chosen as the payload vehicle, the team SHALL abide by	The safety team holds the payload design accountable	Inspection	Safety; Payload	The team has chosen to not

	all FAA regulations for model aircraft.	for all FAA rules and regulations if the payload is a UAV.			pursue a UAV payload.
NASA4.4.4	If a UAV is chosen as the payload vehicle and weighs more than 0.55 pounds, the UAV SHALL be registered with the FAA.	The safety team holds the payload design accountable for all FAA rules and regulations if the payload is a UAV.	Inspection	Safety; Payload	The team has chosen to not pursue a UAV payload.
NASA5.1	The team SHALL use a launch and safety checklist.	The project management and safety teams write a launch and safety checklist. The launch and safety checklist is included in the CDR milestone report.	Inspection	Project Management; Safety	The launch and safety checklist will be included in CDR and FRR documentation.
NASA5.2	The team SHALL identify a student safety officer.	The student safety officer is identified in each milestone report.	Inspection	Safety	See section 5.1 with safety officer information.
NASA5.3.1.1	The student safety officer SHALL, monitor team activities with an emphasis on safety during the design of vehicle and payload.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety; Structures; Payload	The student safety officer, Frances McBride, monitors team activities.
NASA5.3.1.2	The student safety officer SHALL, monitor team activities with an emphasis on safety during the construction of vehicle and payload components.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety; Structures; Payload	The student safety officer, Frances McBride, monitors team activities.
NASA5.3.1.3	The student safety officer SHALL, monitor team activities with an emphasis on safety during the assembly of vehicle and payload.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Project Management; Safety	The student safety officer, Frances McBride, monitors team activities.

NASA5.3.1.4	The student safety officer SHALL, monitor team activities with an emphasis on safety during the ground testing of vehicle and payload.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Project Management; Safety	The student safety officer, Frances McBride, monitors team activities.
NASA5.3.1.5	The student safety officer SHALL, monitor team activities with an emphasis on safety during the subscale test launch.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Project Management; Safety	The student safety officer, Frances McBride, monitors team activities.
NASA5.3.1.6	The student safety officer SHALL, monitor team activities with an emphasis on safety during the Full-scale test launch.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Project Management; Safety	The student safety officer, Frances McBride, monitors team activities.
NASA5.3.1.7	The student safety officer SHALL, monitor team activities with an emphasis on safety during the competition launch day.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Project Management; Safety	The student safety officer, Frances McBride, monitors team activities.
NASA5.3.1.8	The student safety officer SHALL, monitor team activities with an emphasis on safety during the recovery activities.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Safety; Recovery	The student safety officer, Frances McBride, monitors team activities.
NASA5.3.1.9	The student safety officer SHALL, monitor team activities with an emphasis on safety during STEM engagement activities.	The student safety officer monitors team activities and ensures team members are practicing proper safety techniques.	Demonstration	Project Management; Safety	The student safety officer, Frances McBride, monitors team activities.
NASA5.3.2	The student safety officer SHALL implement procedures developed by the team for	The project management and safety teams write a launch and safety checklist that encompasses the	Demonstration	Project Management; Safety	The launch and safety checklist will be included in CDR

	construction, assembly, launch and recovery activities.	assembly, launch and recovery of the launch vehicle.			and FRR documentation.
NASA5.3.3	The student safety officer SHALL manage and maintain current revisions of the team's hazard analyses, failure mode analyses, procedures, and MSDS/chemical inventory data.	The safety team manages all safety documentation for the team.	Inspection	Safety	See section 5 for safety analysis.
NASA5.3.4	The student safety officer SHALL assist in the writing and development of the team's hazard analyses, failure mode analyses, and procedures.	The safety team manages all safety documentation for the team.	Inspection	Safety	See section 5 for safety analysis.
NASA5.4	The team SHALL abide by the rules and guidance of the local rocketry club's RSO during test flights.	The safety team ensures all regulations from the local rocketry club are followed.	Demonstration	Safety	The team will abide by all RSO guidelines.
NASA5.5	The team SHALL abide by all rules set forth by the FAA.	The safety team ensures all rules from the FAA are followed.	Demonstration	Safety	The team will abide by all FAA guidelines.

6.2 Team Derived Requirements Verification Matrix

Table 6-3, below, defines the fields present in Table 6-3, Team Derived Requirements Verification Matrix.

Table 6-3 Derived Requirements Field Definition

Field	Definition
Req No.	Identification number associated with requirement.
Shall Statement	Statement of what the product shall accomplish.
Justification	Why the requirement is necessary.
Success Criteria	Description of how the team will define a met requirement.
Verification Method	How the requirement is verified. By Inspection, Demonstration, Test, or Analysis.
Subsystem Allocation	Indicates which subsystems are responsible for meeting the requirement.
Results	Results of the verification.

Table 6-4, below, shows the derived requirements for the NASA Student Launch project.

Table 6-4 Derived Requirements Verification Matrix

Req No.	Shall Statement	Justification	Success Criteria	Verification Method	Subsystem Allocation	Status
TDR 1.1	The team SHALL meet weekly to discuss project goals and progress.	Holding weekly general body meetings will allow for better delegation of tasks and educate club members on what is necessary to execute this project.	The project management team hosts weekly meetings to update team members on the project progress and delegate tasks.	Demonstration	Project Management	General body meetings are scheduled every Thursday at 7:30 pm. Additional subsystem team meetings are scheduled throughout the week, each week.
TDR 1.2	The team SHALL engage over 750 K-12 students before the FRR milestone report.	NASA SL requires an outreach goal of 200 K-12 students. HPRC strives to exceed expectations regarding sharing interest in rocketry.	The project management team organizes enough K-12 outreach events to exceed 750 participants.	Demonstration	Project Management	The team has currently completed outreach with 44 students.

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 the launch vehicle is prone to landing in. A water resistant airframe will reduce the potential damage in this event.	The airframe is not damaged nor deformed by exposure to water.	Analysis; Demonstration	Structures	See section 3.1.4.1 for current material selection
TDR 2.2	The launch vehicle SHALL not exceed a velocity of Mach 0.7.	At speeds over Mach 0.7, the launch vehicle enters transonic flight. Fin flutter become much more likely with this increase in velocity which increases risk for fin damage.	The launch vehicle does not exceed Mach 0.7 during flight.	Analysis	Aerodynamics	See flysheet for additional information.
TDR 2.3	The launch vehicle SHALL utilize a motor compatible with a motor casing already in the team's possession.	Not purchasing a motor casing allows the team the budgetary freedom to pursue more innovative payload designs.	A motor casing for the full scale launch vehicle is not purchased.	Inspection	Project Management; Aerodynamics	See section 6.3.1 for line item budget
TDR 2.4	The launch vehicle SHALL reach an apogee between 4300 ft and 4550 ft.	This range represents the range relative to a target apogee of 4090ft. This target was chosen in the interest of cost and weight-savings and is therefore a conservative apogee.	The apogee falls within 4300 ft and 4550 ft on competition launch day. All analysis reported prior to competition state that apogee is reached between these altitudes.	Analysis	Aerodynamics	See section 3.3 for preliminary apogee analysis

TDR 2.5	The launch vehicle SHALL have a static stability margin between 2.0 and 2.30 upon rail exit.	To meet the requirement NASA x.xx, a stability margin of 2.0 is necessary. The maximum value of 2.3 was selected because excessive stability margin causes undesirable weathervaning in high winds.	The calculated launch vehicle's static stability margin lies within the range of 2.00 and 2.30.	Analysis	Aerodynamics	See section 3.3 for preliminary static margin analysis.
TDR 2.6	The launch vehicle's blast caps SHALL be exposed and accessible.	Accessible energetics allow for safer and easier installation of black powder charges.	The designed Avionics Bay has easily accessible blast caps.	Inspection	Structures; Recovery	See section 3.1.4.7 for current AV Bay design.
TDR 2.7	All critical components of the launch vehicle SHALL be designed with a minimum factor of safety of 2.	This will ensure that assumptions made in analysis or reasonably higher than expected loading will not cause unpredicted failure during flight.	The factor of safety of each flight critical component is reported in documentation.	Analysis	Structures	Further analysis present in future documentation.
TDR 2.8	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 the vehicle at the field.	The diameter of the launch vehicle is not larger than 6 inches.	Inspection	Aerodynamics; Structures	See section 3.1.4 for full scale design.
TDR 3.1	The onboard altimeters SHALL use one 9V battery each.	StratoLogger altimeters use one 9V battery each.	The avionics sled is designed to hold two 9V batteries. One for each altimeter.	Inspection	Recovery	See section 3.2.3 for avionics design

TDR 3.2	Drogue descent velocity SHALL be less than 100 fps.	This speed will minimize the deployment shock at main deployment.	Calculations for descent velocity under drogue parachute are less than 100 fps.	Analysis	Recovery	See section 3.3 for preliminary descent velocities.
TDR 3.3	The launch vehicle SHALL use recovery devices that the team owns.	This restriction on the recovery section of the launch vehicle enables a better distributed budget for diversified development of flight systems.	The designed recovery system does not require the team to purchase new recovery devices.	Inspection	Recovery	See section 3.2.3 for preliminary recovery devices.
TDR 3.4	The black powder ejection charges SHALL produce at least a 15 psi pressure.	The 15 psi is a standard for most ejection charges in hobby rocketry.	Ejection charge calculations indicate a pressure of 15 psi is reached.	Analysis	Recovery	See section 3.2.2.9 for preliminary black powder charge calculations.
TDR 3.5	The launch vehicle SHALL use U-Bolts for all shock cord attachments.	U-Bolts reduce the chance of a single point of failure.	The load bearing bulkheads have recovery connection points that use u-bolts.	Inspection	Recovery	See section 3.1.4 for bulkhead design.
TDR 4.1	The payload SHALL have a combined weight of no more than 9 lbs.	To maintain kinetic energy requirements in NASA x.x the weight of the payload is limited.	The payload weighs less than 9 lbs.	Inspection	Aerodynamics; Payload	See section 4.3.4 for current payload design.
TDR 4.1.1	The payload vehicle SHALL have a weight of no more than 6 lbs.	To maintain kinetic energy requirements in NASA x.x the weight of the payload is limited.	The payload vehicle weighs less than 6 lbs.	Inspection	Aerodynamics; Payload	See section 4.3.4 for current payload design.
TDR 4.1.2	The payload integration system SHALL	To maintain kinetic energy requirements in	The payload integration system	Inspection	Aerodynamics; Payload	See section 4.3.4 for current payload design.

	have a weight of no more than 2.5 lbs.	NASA x.x the weight of the payload is limited.	weighs less than 2.5 lbs.			
TDR 4.2	The payload vehicle SHALL drive at a speed of at least 3 mph.	The average human walking speed is 3 mph. The rover must keep up with the payload operator walking speed.	The payload vehicle consistently drives at a speed of 3 mph.	Analysis; Test	Payload	See section 4.3.4 for current payload design.
TDR 4.3	The payload vehicle SHALL cover a range of at least 2000 feet.	If the launch vehicle were to land at a maximum win drift distance of XXXX, the rover would have to travel at most 2000 feet.	The payload vehicle is capable of driving over 2000 feet at one time.	Analysis; Test	Payload	See section 4.3.4 for current payload design.
TDR 4.4	The payload vehicle SHALL have a diameter of less than 4.25 inches.	The launch vehicle itself is already constrained to a size of 6 inches. Space also needs to be reserved for retention and deployment mechanisms.	The payload vehicle fits within the payload integration system.	Inspection	Payload	See section 4.3.4 for current payload design.
TDR 4.5	The payload vehicle SHALL resist getting stuck on terrain.	The rover is subject to rough terrain at the field in Huntsville, AL. It is important to design a payload that can operate in adverse conditions as well as favorable.	The payload vehicle maintains traveling configuration over rough terrain and inclines. The design team creates a mechanism to maintain this configuration.	Test	Payload	See section 4.3.4 for current payload design.

TDR 4.6	The payload vehicle SHALL resist getting stuck during deployment.	The rover is subject to rough terrain at the field in Huntsville, AL. It is important to design a payload that can operate in adverse conditions as well as favorable.	The payload vehicle is expelled from the launch vehicle evenly and as designed.	Test	Payload	See section 4.3.4 for current payload design.
TDR 4.7	The payload vehicle SHALL be radially supported within the body tube.	The rover is subject to all flight forces. To limit movement during flight, the rover must be radially supported.	The payload integration system radially supports the payload vehicle such that the rover is not a cantilever upon deployment.	Inspection	Payload	See section 4.3.4 for current payload design.
TDR 4.8	The payload integration system SHALL be a maximum of 10 inches long.	Limiting the length of the payload integration in turn limits the length of the payload bay itself. This will contribute to a favorable static stability margin.	The payload integration system is less than 10 inches long.	Inspection	Payload	See section 4.3.4 for current payload design.
TDR 4.9	The sample collection system SHALL contain at least 15 mL of simulated lunar ice.	Collecting a larger sample than necessary will ensure that more than 10 mL are collected for competition.	The payload can collect and store 15 mL of simulated ice.	Test	Payload	See section 4.3.4 for current payload design.
TDR 4.10	The sample collection system SHALL contain the sample in a closed	A closed compartment will rescued the likelihood of the sample spilling during travel.	The payload seals the sample of collected simulated ice.	Demonstration	Payload	See section 4.3.4 for current payload design.

	compartment when stowed for travel.					
TDR 4.11	The sample collection system SHALL not impede rover mobility when stowed.	The sample collection cannot inhibit travel after the sample is collected.	The payload is capable of traveling 10 feet after sample collection.	Demonstration	Payload	See section 4.3.4 for current payload design.
TDR 4.12	The sample collection system SHALL use the same power source as the rover.	This reduces the weight of the rover itself as only one battery is necessary.	The rover has one battery to drive both the motors and the sample collection system.	Inspection	Payload	See section 4.3.4 for current payload design.
TDR 4.13	The sample collection system SHALL deploy only after operator input.	The sample collection system will interface with the ground to collect the simulated ice. If this is deployed too early, it can impact the rover's travel.	The sample collection system is failsafe and is only deployable after user input.	Demonstration	Payload	See section 4.3.4 for current payload design.
TDR 5.1	Team members SHALL be trained on power tools prior to use.	Tool training ensures that each team member is aware of how each tool works and what precautions to take prior to use.	Team members receive tool training prior to power tool use.	Inspection	Safety	Safety officer, Frances McBride, held a safety seminar and tools workshop at the team's 10/24/19 general body meeting.
TDR 5.2	A designated safety officer SHALL be present at all fabrication activities	The safety officer specializes in identifying hazardous situations. Their presence at fabrication	The designated safety officer is present and signs in with design lead at each fabrication session.	Inspection	Safety	Safety officer, Frances McBride, has been present at each fabrication session to date.

	requiring hazardous materials or power tools.	events allows them to help prevent these situations from occurring.				
TDR 5.3	A designated safety officer SHALL be present at all launch day activities.	The safety officer specializes in identifying hazardous situations. Their presence at launch day events allows them to help prevent these situations from occurring.	The designated safety officer is present and signs in with each required checklist procedure.	Inspection	Safety	Safety officer identification and check in signatures are to be included at the start of each launch day checklist.
TDR 5.4	Launch vehicle hazards that present risk in the red zone SHALL be mitigated to lower likelihood and/or severity by full scale launch date.	Fewer launch vehicle hazards in the red zone is desirable for mission success.	There are no launch vehicle hazards in the red zone at the full scale launch date.	Inspection	Safety	Hazards will be mitigated prior to full scale launch.

6.3 Financing

6.3.1 Budget

Table 6-5, below details the year-long budget for the 2019-2020 competition year.

Table 6-5 2019-2020 Competition Budget

	Item	Quantity	Price per Unit	Item Total
Subscale Structure	Aerotech J570W-14A	2	$70.00	$140.00
	Aero Pack 38mm Retainer	1	$27.00	$27.00
	Motor Casing	1	$340.00	$340.00
	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:			**$993.42**
Full-Scale Structure	6" G12 Airframe, Half Length (30"), 3 Slots	1	$80.00	$80.00
	6" G12 Airframe, Full Length (60")	1	$228.00	$228.00
	3" G12 Airframe, Half Length (30"), Motor Tube	1	$50.00	$50.00
	6" G12 Coupler 12" Length	2	$60.00	$120.00
	6" Fiberglass 5:1 Ogive Fiberglass Nosecone	1	$94.95	$94.95
	Domestic Birch Plywood 1/8"x2x2	8	$14.82	$118.56
	Aerotech 75/3840 Motor Case	1	$360.00	$360.00
	75 mm Motor Retainer	1	$72.00	$72.00
	Rail Buttons	4	$2.50	$10.00
	U-Bolts	4	$1.00	$4.00
	Aerotech L1520T-PS	2	$161.00	$322.00
	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
	Screw Switches	2	$12.00	$24.00
	Poster Printing (feet)	4	$10.00	$40.00
	Subtotal:			**$1,742.01**

Payload	1 inch PVC Pipe	1	$5.98	$5.98
	1 /2 inch PVC pipe	40	$0.49	$19.60
	1 inch PVC cap	8	$0.83	$6.64
	PVC cement	1	$8.98	$8.98
	Drum Motor	2	$19.99	$39.98
	Encoder Connector	2	$0.99	$1.98
	Bore Gear	2	$9.99	$19.98
	Stratologger CF altimeter	1	$61.06	$61.06
	ADXL 354 multi axis Accelerometer	1	$47.39	$47.39
	Limit Switches	1	$7.99	$7.99
	77 oz-in DC Motor	1	$24.99	$24.99
	ESP32 Feather Board	1	$19.95	$19.95
	3.75 in Aluminum Spur	1	$39.72	$39.72
	1.25 in Aluminum Spur	2	$18.49	$36.98
	400mmx8mm Lead Screw	2	$21.55	$ 43.10
	6V Solenoid Lock Latch	1	$12.80	$12.80
	Drive Motor	2	$39.99	$79.98
	Lipo Battery	1	$33.99	$33.99
	4’’ Traction Wheel	2	$15.99	$31.98
	Wheel Hub	2	$2.99	$5.98
	Master Controller	1	$19.95	$19.95
	Motor Controller	1	$99.99	$99.99
	Arm Servo	1	$69.99	$69.99
	Rover Structure	1	$74.29	$74.29
	Caster Wheel	1	$4.36	$4.36
	Deployment/Suspension Spring	1	$11.27	$11.27
	Radio Receiver	1	$33.99	$33.99
	Servo Controller	1	$19.99	$19.99
	Radio Antenna	1	$54.99	$54.99
	Subtotal:			**$998.23**
Recovery and Avionics	Iris Ultra 96" Compact Parachute	1	$433.89	$433.89
	18" Elliptical Parachute	1	$57.17	$57.17
	Stainless Steel Quick Links	14	$1.97	$27.58
	5/8” Kevlar Shock Chord (yard)	20	$4.34	$86.80
	1 /4” Kevlar Shock Chord (yard)	15	$3.69	$54.75
	Black Powder	1	$17.95	$17.95
	E-Matches	1	$80.25	$80.25

	Shear Pins	1	$1.00	$1.00
	StratoLogger CF Altimeter	4	$49.46	$197.84
	6" Deployment Bag	1	$49.45	$49.45
	18" Nomex Cloth	1	$24.00	$24.00
	BeeLine Radio Transmitter	1	$59.00	$59.00
	4" Deployment Bag	1	$43.00	$43.00
	13" Nomex Cloth	1	$16.00	$16.00
	Iris Ultra Elliptical 24" Compact Parachute	1	$64.00	$64.00
	Iris Ultra Compact Elliptical 60" Parachute	1	$241.88	$241.88
	Subtotal:			**$1,454.56**
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			$1,200.00
	Incidentals (replacement tools, hardware, safety equipment)			$1,500.00
	Subtotal:			**$3,115.74**
Travel	Student Hotel Rooms (# rooms)	4	$791.70	$3,166.80
	Mentor Hotel Rooms (# 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	500	$0.37	$185.00
	Banner	1	$250.00	$250.00
	Subtotal:			**$1,745.00**
Total Expenses:				**$17,832.40**

6.3.2 Funding Plan

HPRC gets all it's funding from multiple NC State University organization and North Carolina Space Grant (NCSG).

The NC State University Student Government Association's Appropriations Committee is responsible for distributing university funds to campus organizations. The application process is similar to the Engineers' Council with a proposal, presentation, and an in-person interview. In the 2018-2019 academic year, HPRC received a total of $2,160: $640 in the fall semester and $1,520 in the spring semester. A request for $2,000 has been placed for the current fall semester and the same amount will be requested in the spring semester, assuming that the Appropriations Committee budget will remain the same.

Engineering and Technology Fee is an NC State University fund that allocates funding for academic enhancement through student organizations. Their funding will primarily pay for the faculty advisor's 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 $5,500.

In addition to funding through NC State organizations, the 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 2019.

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 the team's past sponsors. The team hopes to get at least $4,500 more in funding from various companies.

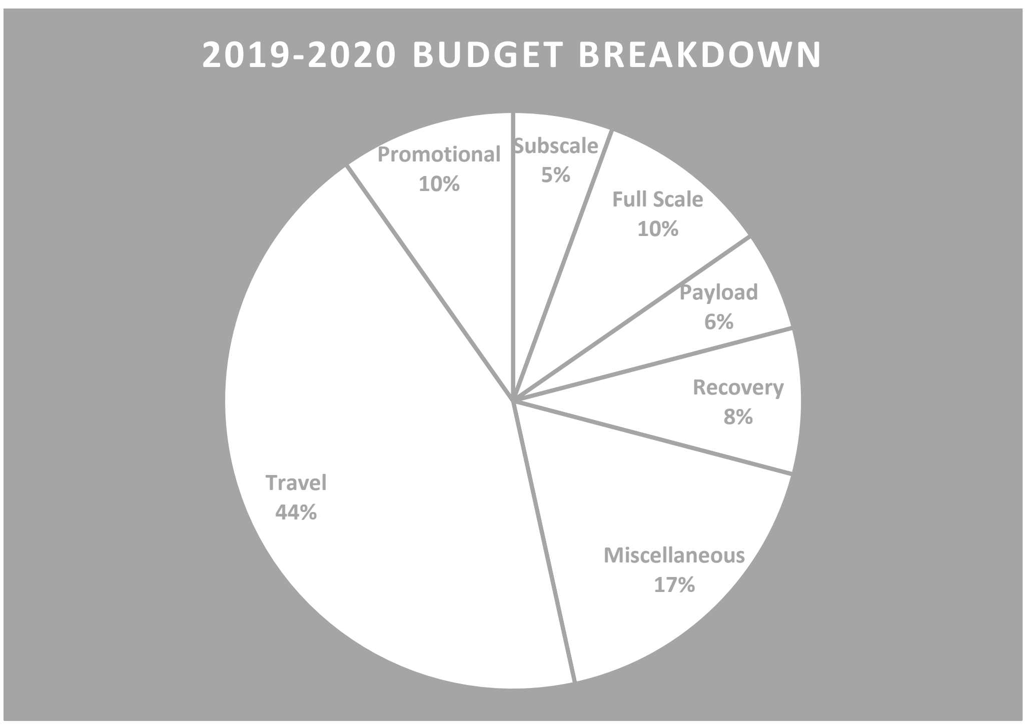
These totals are listed in Table 6-6, below, which compares the projected costs and incoming grants for the 2019-2020 school year.

Table 6-6 2019-2020 Funding Sources

Organization	Fall Semester Amount	Spring Semester Amount	School Year Total
Engineering Technology Fee	-	-	$1,500.00
SGA Appropriations	$900.00	$900.00	$1,800.00
Sponsorships	-	-	$3,600.00
NC Space Grant	-	-	$5,000.00
College of Engineering	-	-	$6,000.00
Total Funding:			**$17,900.00**
Total Expenses:			**$17,832.40**
Difference:			**$68.00**

Figure 6-1, below shows a budget breakdown for the 2019-2020 competition cycle.

Figure 6-1 2019-2020 Budget Breakdown Chart



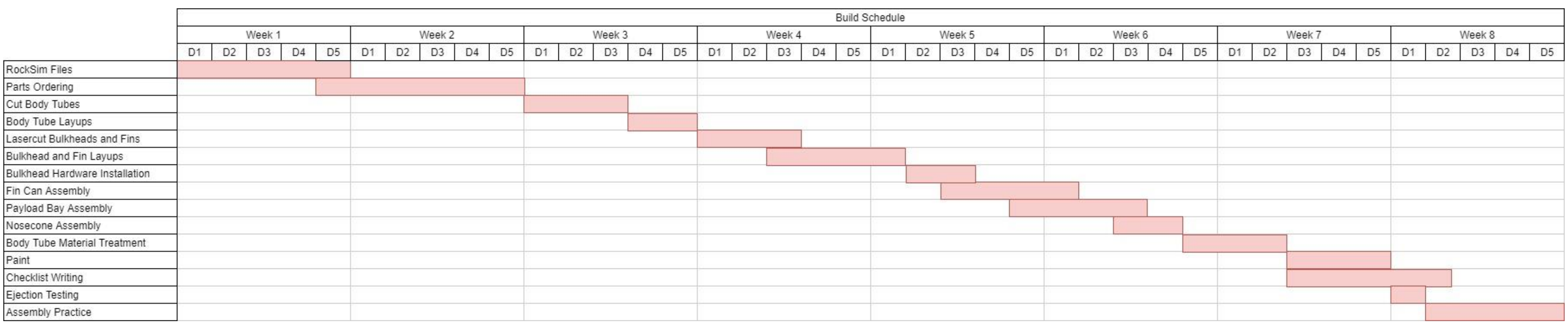
6.4 Project Timelines

The figure below shows a project timeline for all builds, events, and documents affiliated with the 2019-2020 project. The dates here are mostly fixed based on the due dates outlined in the NASA SL handbook. Documents are generally active between the date of the Q&A session for that document and its due date. Builds are generally active for the 6-8 weeks prior to the desired launch window.

Figure 6-2 2020 Overall Project Timeline

The example build schedule below gives an idea of the order of steps taken for a build and the relative time it takes to complete each task. This build schedule is applicable to both the full scale and subscale builds, with possible minor differences due to payload housing and integration. Steps with some amount of overlap generally indicate steps which are guided by another but can be worked on simultaneously. Steps without overlap require the previous step be completed before work can begin on the following step.

Figure 6-3 Example Build Schedule



The Funding Cycle Gantt Chart below indicates the periods in which funding sources are available to the team. Funding received from a source will be spent in the window indicated by the Gantt Chart, either due to requirements defined by the funding source or due to the team determining that revenue source will likely be consumed by project demands by that point.

Figure 6-4 Funding Cycle Gantt Chart