NC STATE UNIVERSITY

Tacho Lycos

2026 NASA Student Launch

Preliminary Design Review

High-Powered Rocketry Club at NC State University
1840 Entrepreneur Drive
Raleigh, NC 27606

November 3, 2025

Common Abbreviations and Nomenclature

AIAA	=	American Institute of Aeronautics and Astronautics
APCP	=	Ammonium Perchlorate Composite Propellant
AV	=	Avionics
AVAB	=	Avionics and Air Brakes Bay
CAD	=	Computer Aided Design
CATO	=	Catastrophe at Take Off
CDR	=	Critical Design Review
CFD	=	Computational Fluid Dynamics
CG	=	Center of Gravity
CNC	=	Computer Numerical Control
CP	=	Center of Pressure
ECE	=	Electrical and Computer Engineering
ETF	=	Educational and Technology Fee
EYE	=	Engineer Your Experience
FAA	=	Federal Aviation Administration
FMEA	=	Failure Modes and Effects Analysis
FRR	=	Flight Readiness Review
GPS	=	Global Positioning System
GUI	=	Graphical User Interface
HAUS	=	Habitat for Agricultural Utilization Study
HPRC	=	High-Powered Rocketry Club
ID	=	Inner Diameter
IMU	=	Inertial Measurement Unit
LED	=	Light Emitting Diode
LiPo	=	Lithium Polymer
LRR	=	Launch Readiness Review
LS	=	Likelihood Severity
MAE	=	Mechanical & Aerospace Engineering
NAR	=	National Association of Rocketry
NASA	=	National Aeronautics and Space Administration
NCSG	=	North Carolina Space Grant
NCSU	=	North Carolina State University
NFPA	=	National Fire Protection Association
NPK	=	Nitrogen, Phosphorus, and Potassium
OD	=	Outer Diameter
PCB	=	Printed Circuit Board
PDR	=	Preliminary Design Review
PDF	=	Payload Demonstration Flight
PEM	=	Penn and Manufacturing Corp.
PETG	=	Polyethylene Terephthalate Glycol
PLA	=	Polylactic Acid
PLAR	=	Post-Launch Assessment Review
PPE	=	Personal Protective Equipment
RF	=	Radio Frequency
RSO	=	Range Safety Officer
RVM	=	Requirement Verification Matrices
SDS	=	Safety Data Sheets
SGA	=	Student Government Association
SL	=	Student Launch
STEM	=	Science, Technology, Engineering, and Mathematics
TAP	=	Technical Advisory Panel
TRA	=	Tripoli Rocketry Association
UHMWPE	=	Ultra-high-molecular-weight polyethylene
VDF	=	Vehicle Demonstration Flight

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1 Summary of Report

1.1 Team Summary

Table 1.1: Team Information and Social Media

Information Required	Details	Social Media Platform	Details
Team Name	Tacho Lycos	**Twitter**	@ncsurocketry
Mailing Address	1840 Entrepreneur Drive, Raleigh, NC 27606	**Facebook**	@tacholycos
Name of Mentor	Jim Livingston	**Instagram**	@ncsurocketry
TRA Number	02204	**TikTok**	@ncsurocketry
Certification Level	TRA Level 3 Certification	**LinkedIn**	Tacho Lycos
Email	livingston@ec.rr.com	**Email**	rocketry-org@ncsu.edu
Phone Number	(910) 612-5858	**Website**	ncsurocketry.org
The team will launch in Huntsville during Launch Week.			

1.2 Launch Vehicle Summary

Table 1.2: Vehicle Motors, Sections, and Recovery System

Motor Choice						
Motor	**Name**					
Primary Motor Choice	L1390G					
Secondary Motor Choice	L1520					
Vehicle Section Breakdown						
Section	**Length**	**Dry Mass**	**Wet Mass**	**Ballasted**	**Burnout**	**Landing**
Nosecone / Payload Bay	36.6 (in)	8.20 (lbs)	8.20 (lbs)	8.20 (lbs)	8.20 (lbs)	8.20 (lbs)
Main Bay	33.5 (in)	7.82 (lbs)	7.82 (lbs)	7.82 (lbs)	7.82 (lbs)	7.82 (lbs)
Drogue Bay / Fin Can	48.5 (in)	11.44 (lbs)	19.99 (lbs)	19.99 (lbs)	15.64 (lbs)	15.64 (lbs)
Recovery System						
Parachute	**Specification**			**Descent**	**Main**	**Backup**
Main Parachute	Fruity Chutes 96 in. Iris Ultra Compact parachute			15.76 (fps)	2.30 (g) of BP	3.70 (g) of BP
Drogue Parachute	18 in. in-house fabricated elliptical parachute			100.23 (fps)	2.40(g) of BP	3.90(g) of BP
Altimeter Details						
Brand	**Model**		**Main Deployment**		**Drogue Deployment**	
Silicdyne	Fluctus		500 (ft)		Apogee	
Altus Metrum	EasyMini		550 (ft)		Apogee + 1s	

1.3 Payload Summary

A self-righting lander will deploy from the nosecone after the Launch Vehiclehas landed. Roughly cylindrical in shape, this lander will be mounted on rails and secured with a latch while inside the rocket. After landing, a lead screw mechanism will push the lander along its tracks and out of the Launch Vehicle. The latch will release and the lander will be completely separate from the Launch Vehicle and on its side. Then, the lander will deploy four legs to right itself. These legs will be hinged to the base of the lander and attached with struts to a collar that extends with another lead screw. Once upright, the lander deploys an auger to collect soil. This auger is rotated and extended at the same time, allowing it to drill effectively into the soil. When fully extended, the auger will continue to spin, forcing soil into the lander. This soil will be directed by fixed elements into a collection chamber. A 7-in-1 soil sampler will be positioned so that its sensors are covered by the funneled soil. Finally, the soil sensor can collect the required pH, electrical conductivity, and nitrogen measurements.

The Air Brakes payload will be located near the top of the fin can section. This payload will, increase the reference area of the for greater drag for altitude control to reach a target altitude. It uses a Bang Bang control scheme with custom software, an inertial measurement unit(IMU), Servo, and Raspberry Pi to gather data, which is processed to get an accurate prediction of the peak altitude. After motor burnout, if the target apogee is lower than the predicted peak altitude, Air Brakes will deploy 4 symmetric fins, if not, the Air Brakes fins will retract.

2 Changes Made Since Proposal

2.1 Changes Made To Vehicle Criteria

Table 2.1: Changes Made to Vehicle

Change Description	Justification	Affected Subsystem(s)
Vehicle Mass reduced from 40.9 (lbs) to 36.0 (lbs).	Reduction of mass as a result of material changing from balsa wood to a lighter Nomex core for bulkheads and centering rings.	Structures, Aerodynamics, & Recovery
The primary altimeter was changed from a PerfectFlite StrattologgerCF to a Silicdyne Fluctus. The separate Eggfinder Mini GPS was removed.	The Fluctus integrates GPS tracking, which consolidates functions and reduces mass and size in the avionics bay by eliminating the need for a separate GPS unit.	Recovery
The main parachute was changed from a 120 (in) toroidal parachute to a 96 (in) Iris Ultra Compact	The 120 in chute was too large and would have caused the descent time and drift distance to exceed NASA limits. The 96 in chute meets all requirements and keeps the landing kinetic energy low enough to qualify for bonus points	Recovery
Main parachute bay length was increased from 24.0 (in) to 28.0 (in).	Increased parachute bay allows for more room for the main parachute and main shock cord.	Structures & Recovery
Drogue parachute bay length was shortened from 24.0 (in) to 23.0 (in).	The 23 (in) was determined to be the optimized volume to house the 18 (in) drogue parachute and the 28.50 (ft) shock cord.	Recovery & Structures
Fin can was shortened from 28.0 (in) to 25.5 (in).	The 25.5 (in) length was the final dimension required to house all internal components and properly interface with the 9 (in) Air Brakes bay.	Recovery & Structures
Backup separation charges changed from adding 0.5 oz to the main charge to utilizing a factor of safety of 2.	Adding 0.5 oz of black powder to a primary charge is arbitrary and does not account for variations in actual deployment conditions; using a factor of safety ensures reliability.	Recovery & Structures

2.2 Changes Made to Payload Criteria

Table 2.2: Changes Made to Payload

Change Description	Justification	Affected Subsystem(s)
Redesigning the apogee prediction method for Air Brakes.	The current apogee prediction is reliable to within a 4 percent error of the set apogee but is still not precise. Switching to a Runge-Kutta-4 method based on in flight data will benefit accuracy and make the Air Brakes software independent of the vehicles' configuration	Aerodynamics

The payload team selected one of its two proposed concepts. The design is now a Self-Righting Lander that deploys on the ground after landing.	The Mounted Nosecone design had severe space constraints on the drill length. The Ground-Deploying Lander was chosen because it solved this issue and also reduced complexity by removing the need for a separate payload parachute and associated RSO approval for an in-air deployment.	Structures, Recovery, & Payload
Payload system changed from a single Raspberry Pi 5 to a Raspberry Pi 4B for the main lander, an Arduino Nano to control the nosecone ejection system, and 2 STM32F405 microcontrollers to manage the sensors.	The Arduino provides a cost and size effective way to manage the ejection mechanism. The STM32F405 is used to handle all real-time sensor communication, the Raspberry Pi 4B is used for processing.	Payload Team
The leg deployment mechanism was finalized to the link and collar system.	The collar system was decided based off of manufacturability, torque, mechanical complexity, reliability, cost, and weight.	Payload Structures & Payload Electronics

2.3 Changes Made to Project Plan

Table 2.3: Changes Made to Project Plan

Change Description	Justification	Affected Subsystem(s)
Finalized a more in-depth development timeline for all subteams.	To ensure the Project Management Subteam can monitor weekly progress, and ensure VDF and PDF flights are successful and on time.	Structures, Recovery, Payload,Project Management, and Aerodynamics
Moved the painting of Subscale to occur one week after the November 1st Subscale launch	To allow for more time to dry run components of the Launch Vehicle during the week before launch	Structures, Recovery, Payload, Project Management, and Aerodynamics
Added a combined budget and sponsorship section to the funding plan	The Team has started more fundraisers this year, and plans on raising $1000 each semester	Project Management
Date for PDR submittal was pushed back a week, but not the writing timeline	NASA updated the PDR submittal window, but the Team still wanted to complete their documentation early	Structures, Recovery, Payload, Project Management, and Aerodynamics.

3 Vehicle

3.1 Mission Statement

The primary mission of the launch vehicle is to deliver the payload to the declared apogee and return it to the ground within specified kinetic energy and descent time requirements. The vehicle's secondary objective is to house the Air Brakes system, the system which improves the accuracy of the launch vehicle's apogee in relation to the targeted declared apogee. The launch vehicle will be designed to enable a complete execution of the payload mission requirements with re-usability, reliability, and safety. Various criteria will be utilized to determine the launch vehicle's mission success in Table 3.1.

Table 3.1: Vehicle Success Criteria

Success Level	Vehicle Criteria
Success	· The launch vehicle exhibits nominal performance during powered flight and coast phases · Stage separation and recovery system deployment occur as intended at the designated altitudes · The vehicle achieves an apogee within $\pm$200 (ft) of the declared target altitude · Landing occurs within the designated recovery area and within the allotted recovery timeframe · The vehicle and payload sustain no structural or electrical damage · All vehicle sections are successfully located and recovered · The launch vehicle remains fully operational and capable of same-day relaunch
Partial Success	· Powered flight and coast phases proceed nominally · Separation and recovery system deployment occur at the intended altitudes · The vehicle achieves an apogee between 4,000 ft and 6,000 (ft) · The vehicle experiences only minor landing damage and can be repaired and reflown within the same day.
Partial Failure	· Powered flight and coast phases proceed nominally · The vehicle fails to achieve an apogee within the 4,000-6,000 (ft) range. · Recovery system deployment is incomplete, tangled, or otherwise impaired · The vehicle or payload sustains damage preventing same-day reflight. · The vehicle lands outside the designated recovery area or in an inaccessible or hazardous location (e.g., tree, water, power line).
Failure	· The launch vehicle fails to leave the launch rail · A catastrophic motor failure (CATO) occurs · Premature separation occurs during powered flight or coast phase · The vehicle fails to exceed an altitude of 3,500 ft · The recovery system fails to separate or deploy · Any other incident results in significant structural damage or total loss of the launch vehicle or primary payload.

3.2 Alternative Launch Vehicle Designs

3.2.1 Multi-Purpose Avionics and Air Brakes Bay

A multi-purpose avionics and air brakes housing was considered for vehicle design. A combined bay would result in one less airframe section and one less coupler section in the Launch Vehicle. The design provides a more condensed, lighter, and shorter Launch Vehicle, requiring a lower total impulse to achieve the desired apogee. A multi-purpose section was not chosen due to the high integration complexity between the air brakes and the avionics bay. Open ports for Air Brakes fins require the complete seal of the bay between the avionics to ensure there is no differential pressure growth, potentially causing early recovery event deployment. Combined sections additionally require a multitude of wires to travel between the forward and aft bulkhead for ejection charges, requiring additional complex wire management.

3.2.2 Airframe Materials

S2 Fiberglass

Hand-rolled S2 fiberglass airframes provide strengths similar to standard-modulus and standard-strength carbon fiber strands, higher density, but remove the radio opaqueness. Using a tightly woven weave with higher silicon content fibers like those found in US Composites Style 6781 will provide high workability, with a sufficiently thick weave which can be layered to meet strength requirements for a 6 (in) inner diameter (ID) airframe with multiple layers [3]. S2 fiberglass will be the material of choice for all airframe tubing as per a feasibility study completed in Table 3.2.

Carbon Fiber

Carbon fiber provides the highest strength-to-weight ratio for easily available composite materials. Carbon fiber will not be used in airframe tubing components due to its radio-opaque, electrically conductive fibers. There will be three avionics sections in the that all require transmission, which would necessitate the need to create external body antennas.

Paper-Derived

Paper-derived airframe components will not be used for their high tolerances. Variances in temperature and humidity at the team's local launch field between construction and launch conditions have led to compressed airframe components with bad fitments or motors that do not fit in the motor mount tubing. Paper-derived components provide a high strength for their relatively low densities, but environmental conditions remove paper-derived components as a potential choice for vehicle construction.

Criteria	Weight	Fiberglass	Carbon Fiber	Paper-Derived
Strength	5	4	5	2
Weight	3	3	4	5
RF Transparency	5	4	1	4
Complexity	3	2	2	4
Climate Resistance	4	5	5	1
Unweighted Totals		18	17	16
Weighted Totals		**75**	68	61

Table 3.2: Pugh Matrix

3.2.3 Bulkhead, Centering Ring, and Fin Materials

The primary airframe components: fins, centering rings, and bulkheads will be manufactured using a common core material and laminate process. This will reduce the need for additional purchases and testing methods. Customization of component-level strength will take place by adjusting the fabric material between carbon fiber and S2 fiberglass, along with the number of plies on the faces of the material. A feasibility study was conduced to determine the effectiveness of the materials for the leading vehicle design 3.3.

Honeycomb Nomex Sandwich Composite

Utilizing ultra-low-density honeycomb Aramid-Nomex core material allows for the material to be obtained in large sheets without the need to bond multiple widths together like long-grain balsa wood. Honeycomb Nomex core material achieves low tolerances compared to grained materials. A notable downside of honeycomb core construction is the bond between the face sheet laminates and the thin surfaces. A high-peel strength epoxy or film adhesive is the best choice for the small fillets between the laminate and the honeycomb structure. Honeycomb Nomex core materials are impact-resistant and moisture-resistant due to the phenolic resin coating [14]. Honeycomb cores with cut or sanded edges need to be filled with epoxy or include bonded fairings to provide a smooth edge surface for fins. Laminates fabricated in-house with honeycomb Nomex core material have a calculated 55% weight savings compared to a plate manufactured of pure composite laminate.

Long-Grain Balsa Wood Sandwich Composite

Long-grain balsa wood core provides a lightweight, low-cost core material for face sheet bonding. Compared to end-grain balsa wood, long-grain is more available in the 1/8 (in) regime, despite the lower shear and compressive strengths. Long-grain balsa wood does not experience difficult bonding conditions due to the large, continuous surface area. Face-sheets and sealed edges allow for creating moisture resistance. Long-grain balsa sheets come in widths up to 4 (in) before the need to glue sheets of balsa together to form a larger core surface for laminate bonding.

Baltic Birch Plywood

Baltic birch plywood is a structural wood material with low void content when compared to standard-grade construction plywood. Baltic birch plywood comes in large sheets for easy manufacturing and milling via ShopBot CNC routing for rapid construction. Baltic birch plywood is low-cost with no need for composite face sheets, by increasing the thickness by up to twice that of sandwich core material.

Composite Laminate

Composite laminates provide a high-strength, higher-density option for airframe components. Composite laminates have much simpler processes compared to layups with core materials due to the lower part count, and benefit from higher inter-material bonds with no difference in surface bonds. Composite laminates have less ambiguity in surface properties due to the Composite Laminate Theory calculations available for quasi-isotropic property generation. Laminates are much higher in density than plywood or cored counterparts, with much of the inner plies providing little benefit to flexural properties.

3.2.4 Three-Point Bending Tests

To experimentally compare the properties of material choices for the subscale and full-scale vehicle, sample coupons were prepared for three-point bending tests. Three-point bending tests provide helpful information to aid in the comparison of numerical models to experiments due to significant variation in as-produced parts. Six different material types were compared with three coupons for each test to prove the repeatability and invariability of the manufactured parts. The material choices and configurations were designed to mimic those found in subscale and full-scale components for bulkhead, centering ring, and fin construction. Different variations of balsa or honeycomb Nomex sandwich cores with fiberglass and carbon fiber face sheets, and Baltic birch plywood sheets were used. All produced parts were used as measured dimensions for stress, strain, and modulus calculations to account for slight variations in the manufactured dimensions of the sample coupons.

Experimental Setup

A variation of ASTM D7264 was applied as the three-point bending test setup. Coupons were cut to an average length of 5.75 (in), and 1.00 (in) wide with variations in thickness from 0.125 to 0.185 (in). The samples were straddled with 5.00 (in) between the centers of 0.375 (in) diameter rigid mounts with a 0.375 (in) diameter load application. The samples were loaded at 1.00 (in) of deflection per minute for a maximum of 1.50 (in) of deflection.

Figure 3.1: Three-Point Bending Test Setup

Materials

All composite layups utilized the US Composites 635 Thin 2:1 Slow laminating epoxy system. Layups were vacuum bagged with peel-ply sheets on opposing sides of all laminates, with a vacuum between 25 and 27 inHg applied to all layups. Following the manufacturer's recommended 24-hour cure at room temperature, samples received a modified post-cure schedule with three hours at 212°F.

$$V_f = \frac{\frac{m_{fiber}}{\rho_{fiber}}}{\frac{m_{fiber}}{\rho_{fiber}} + \frac{m_{final} - m_{fiber}}{\rho_{epoxy}}} \quad (1)$$

Using known starting masses and densities of components along with the final masses, the fiber volume fraction for the coupons was between 50 and 55% (1). Variations in fiber volume fraction can be accounted for with excess weave strands displaced from the layup during the epoxy application process.

Sample Sets

1. The first set of coupons was constructed from 0.125 (in) thick end-grain balsa wood with two layers of 3K 5.7 oz/yd^2 carbon fiber plain weave on each side. Both layers were placed with matching 0/90 orientation, with the 0-degree weave following the length of the coupon. The orientation was followed for all coupon layups.
2. The second set of coupons was constructed from 0.125 (in) thick end-grain balsa wood with three layers of 6K 5.7 oz/yd^2 carbon fiber plain weave on each side. The balsa wood used for sets 2 and 3 used low-density balsa compared

to set 1 (0.00397 lb/in^3 vs 0.0635 lb/in^3). The balsa wood for sets 2 and 3, on average, measured 0.119 (in) instead of the advertised 0.125 (in).

3. The third set of coupons was constructed from 0.125 (in) thick end-grain balsa wood with three layers of 6.0 oz/yd^2 E-glass fiberglass with plain weave on each side.
4. The fourth set of coupons was cut from 0.125 (in) thick Baltic birch wood with no additional layups applied to the core material. This set is meant to serve as the baseline due to the high availability and low prices compared to composite materials and sandwich layup processes. The wood fibers were cut to follow the length of the coupon, maximizing bending.
5. The fifth set of coupons utilized honeycomb Nomex sandwich core material with 0.125 (in) thickness. The material was chosen for its high impact strength and ultra-low density compared to even low-density balsa wood (.00200 lb/in^3 vs .00397 lb/in^3). Set five matches set three in the layup construction.
6. The sixth set of coupons utilized honeycomb Nomex sandwich core material with 0.125 (in) thickness. Set six matches set two in the layup construction.

Results

All coupons within each set experienced the same break type. In addition, all sets demonstrated repeatability with a ±7.5% flexural strength tolerance and ±2.5% flexural modulus. Set one and two experienced core shear failures. It was expected that set two would exhibit a higher strength due to the extra layer of carbon fiber on each side of the layup; however, the much lower density of the balsa wood resulted in set two experiencing a similar strength to set one. Set three experienced a compressive laminate failure on the face sheet in addition to a shear core failure. It is anticipated that the set would have benefited significantly from a higher-density core material. The baseline set four, of Baltic birch, experienced a bending failure instead of shear, such as sets one, two, and three. The bending failure allowed the birch wood to experience a breaking strength rivaling the carbon fiber layups. The honeycomb core carbon fiber composite surpassed all the other sets with the highest breaking strength. The fiberglass face-sheeted honeycomb cores additionally outperformed their balsa wood counterparts. For component strength calculations, the strength of fiberglass will use a higher stress calculated component. The higher stress is due to the usage of higher density and higher quality fiberglass. 8.9 oz/yd^2 S2 fiberglass yields a 42% higher density and 25% higher fiber strength than 6.0 oz/yd^2 E-glass fiberglass. Due to the higher strength and density, a S2 fiberglass model is anticipated to have broken at 17.0 ksi.

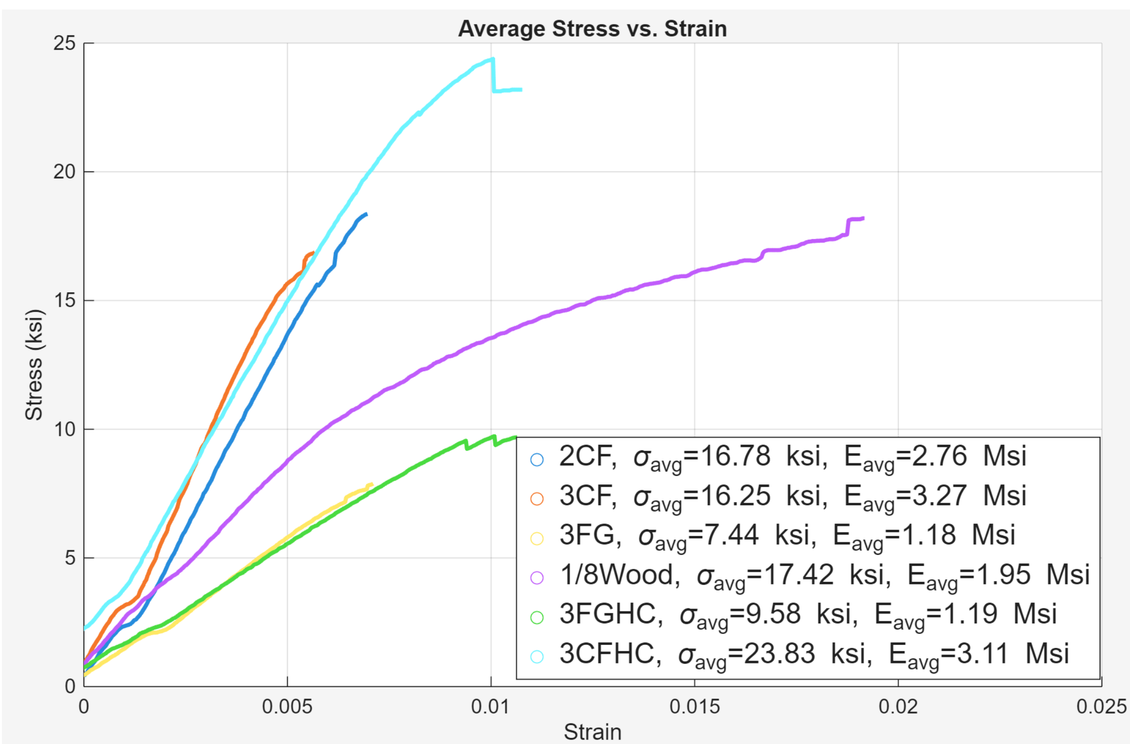


Figure 3.2: Three-Point Bending Test Stress vs. Strain Plot

Criteria	Weight	Composite Balsa	Plywood	Composite Nomex Core	Composite Plate
Weight	4	4	4	5	2
Strength	5	3	2	4	5
Complexity	3	2	5	2	3
Material Dimensionality	2	2	2	4	5
Unweighted Totals		12	14	16	15
Weighted Totals		41	45	**54**	52

Table 3.3: Pugh Matrix.

3.2.5 Tube Compression Test

Since composite parts have lower compressive ratings, parts are more likely to break in compression due to flexural forces on bulkheads, fins, and the airframe tubing. An experimental test was developed using 6.0 (in) long tubes with 1.400 (in) outer diameters and a .06 (in) wall thickness. The tubes were compressed at .05 (in) per minute. Both tubes were cut from the same composite layup consisting of multiple layers of 6.0 oz/yd^2 E-glass fiberglass and US Composites laminating resin. Tubes approximated a 40% fiber volume fraction. In both tests, we see a slow ramp in stress due to minor deformations from an imperfectly flat surface interface on the compressed surfaces. On the first tube, we also observe a fracture occurring before the material increases to a higher strength. It is assumed that minor cracking takes place on the non-flat surfaces. After the major strength break, the material begins radial deformation with no buckling as compression continues.

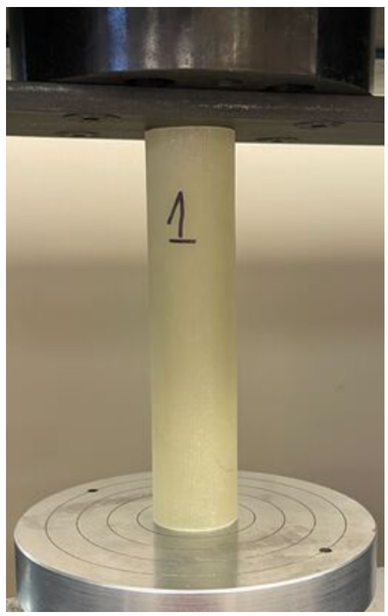


Figure 3.3: Compressive Test Setup

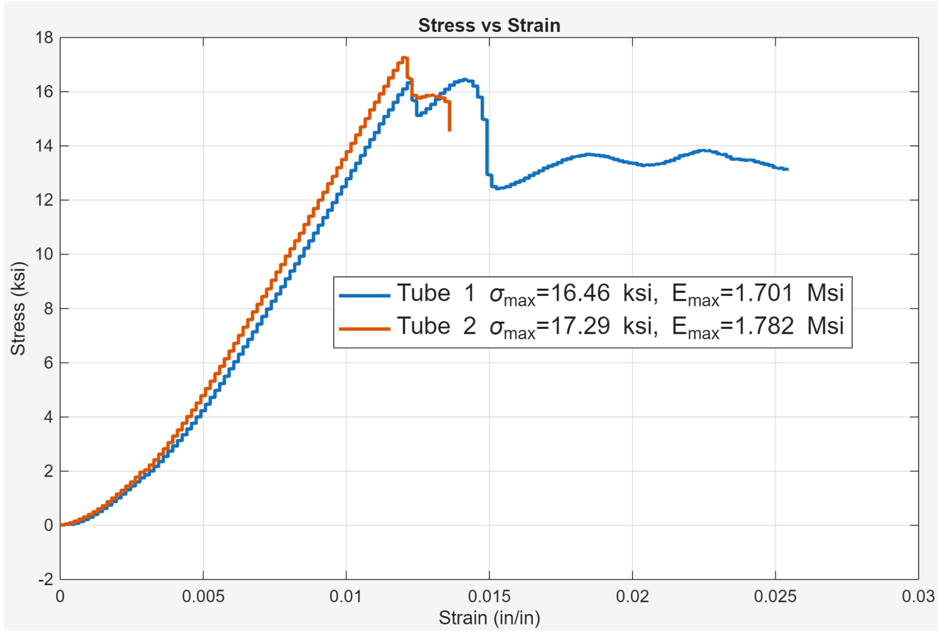


Figure 3.4: Compressive Test Stress vs. Strain Plot

3.2.6 Fin Fairing

Honeycomb Nomex core fins require the edges to be sealed for aerodynamic components due to the exposed, low-strength, and blunt-faced core material. A compression-molded co-bonded design is the leading design for its high customization and high-strength isotropic material properties.

Co-Cured Fairing

A co-cured design would utilize pre-manufactured wooden half-round dowels to provide a rounded edge pre-placed around the edges of the core material to be bonded to the carbon fiber laminate. Bonds also form as epoxy fillets formed between the core and the half-round. The benefits of co-cured profiles include lower manufacturing time with only a single layup rather than multiple.

Co-Bonded Fairing

Co-bonding the edges after the initial laminate curing cycle provides the ability to create custom aerodynamic profiles. Using carbon fiber strand cutoffs in various orientations for an isotropic material. The male composite laminate would receive surface preparation at the leading edges before bonding. A female mold would be 3D printed with mold release spray applied to model the aerodynamic surface. The female mold would provide an aerodynamic edge along with an overlap onto the fin laminate for ample bonding area. Once surfaces are ready, a mixture of laminating epoxy and carbon strands will be mixed at a 1:1 mass ratio before being pushed into the mold until full. The filled mold will then be clamped to the fin, and excess strands will be removed until the cure cycle completes. Following the cure cycle completion, the female mold will be removed.

3.3 Leading Vehicle Design

3.3.1 Vehicle Overview

The leading launch vehicle is designed with 6.12 (in) fiberglass airframe body tubing with a total vehicle length of 111 (in) from the tip of the blunted nosecone to the aft of the motor retainer, Figure 3.5 and 3.6. The integrated launch vehicle wet mass is 36 lbs, utilizing an Aerotech 75/3840 reloadable motor system with an L1390G motor. The launch vehicle includes six sections from the forward to aft end: the nosecone and payload bay, the main parachute bay, the avionics bay, the drogue parachute bay, the air brakes bay, and the fin can.

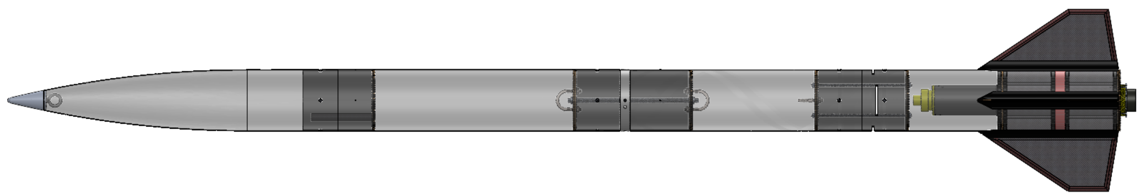


Figure 3.5: Assembled launch vehicle

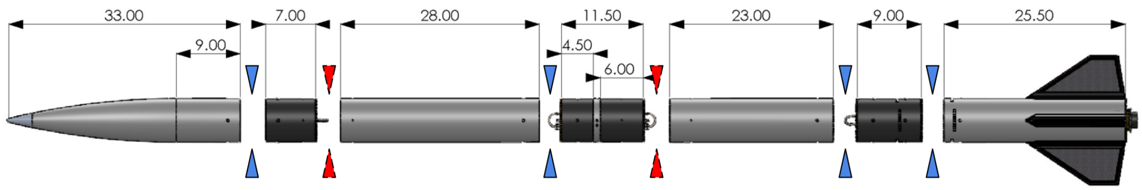


Figure 3.6: Dimensioned Launch Vehiclesections (in). In-flight separation points (red and dashed). Non-in-flight separation points (blue and solid).

The vehicle design includes two in-flight separation points and four non-in-flight separation points. The in-flight separation points are retained with 4-40 nylon shear pins until energetic deployment. The non-in-flight separation points are retained with Penn Engineering & Manufacturing Corp. (PEM) nuts press-fit into the coupler body tubing with countersunk stainless steel screws. The PEM nuts are additionally epoxied to provide additional reinforcement during shock loadings. Energetic charges are located forward and aft of the centrally located avionics bay. Two vehicle rail guides will be fixed to the vehicle, the aft one will be mounted with an additional PEM nut in the fin can and a forward rail guide mounted to the avionics forward PEM nut. All vehicle sections are tethered with shock cord during recovery events.

3.3.2 Airframe Tubing

Vehicle airframe and coupler tubing will be manufactured in-house utilizing 8.9 oz/yd^2 S2 fiberglass cloth with satin weave. Fiberglass will be laminated with US Composites 635 Slow epoxy resin over a mandrel. Airframe tubing will utilize six layers of impregnated cloth, with seven layers for coupler tubing.

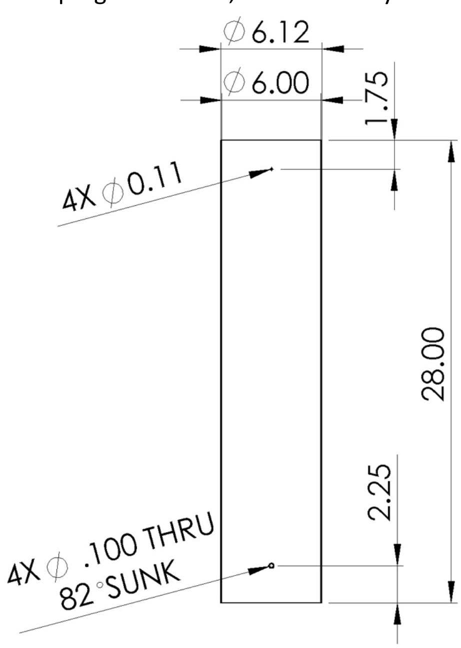


Figure 3.7: Forward airframe tube (in).

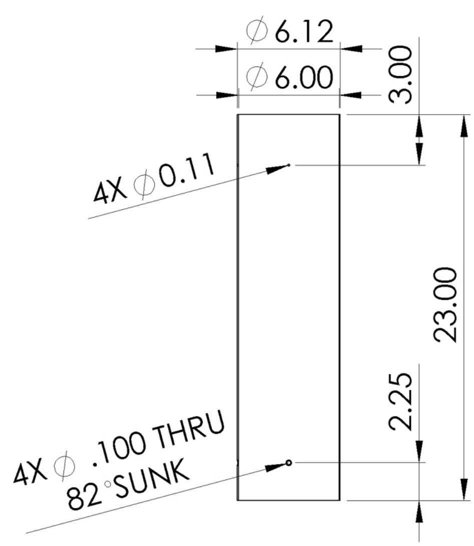


Figure 3.8: Aft airframe tube (in).

Airframe Tubing Preliminary Force Analysis

Flight compressive loadings from Table 3.7 will be combined with bending aerodynamic lateral loads to determine the body tube thickness for the airframes. Richard Nakka's Experimental Rocketry website guide for Rocket Body Design Considerations will be followed to determine the combined compressive forces affecting body tubing from bending and axial loads

[7]. The primary design and input parameters are provided in Table 3.4.

Table 3.4: Vehicle body tubing input parameters

$\alpha(°)$	$v(\frac{ft}{s})$	$\rho(\frac{slug}{ft^3})$	$G-force$	$r_{OD}(in)$	$t_{wall}(in)$	$F_T(lbf)$	$m_{rocket}(lbm)$	$\sigma_{laminate}(psi)$
10.0	693	0.00238	9.00	3.0582	0.0582	370.9	36	16000
$C_t(in)$	$C_r(in)$	$b(in)$	$Sweep(in)$	N_{fins}	$CG(in)$	C_D	$L_{nosecone}(in)$	$L_{LeadingEdge}(in)$
4.00	14.0	12.0	9.50	4	67.5	0.559	23.5	50

The calculations provided use the max loading conditions for a vehicle at a high angle of attack using the vehicle's peak thrust, velocity, and fully loaded mass. Loads from inertia, lateral aerodynamic loads, and drag are combined to a factor of safety of 3.30. The high factor of safety is to account for high impact loads with the potential for hard-packed dirt landings and potentially asphalt.

Using the determined airframe body thickness, the coupler thickness can be calculated by matching the section modulus of the coupler and the airframe tubing 2. Due to the set thickness of each layer of S2 fiberglass, an additional layer of fiberglass will be added, setting the thickness of the coupler beyond the calculated difference to match the section modulus.

$$t_{coupler} = \frac{d_i}{2} - (\frac{d_i^4}{8} - \frac{d_o^4}{16})^{\frac{1}{4}} = \frac{6.00}{2} - (\frac{6.00^4}{8} - \frac{6.116^4}{16})^{\frac{1}{4}} = 0.0618(in) \tag{2}$$

Stress concentrations develop at the drilled holes in the airframe, with higher concentration for anisotropic composite laminates compared to isotropic materials [4]. Utilizing the countersunk holes for PEM nuts with the outer diameter (OD) of the countersink for conservative estimates, we can determine the additional stress experienced by the airframe tubing 3.

$$\sigma_{concentration} = \sigma\sqrt{2(\sqrt{\frac{E_x}{E_y}} - \nu) + \frac{E_x}{G}} \tag{3}$$

Integrating the stress concentration into Nakka's guide for airframe tube thickness provides a new 1.50 factor of safety. The new loading still provides an ample factor of safety to account for unanticipated flight conditions, conservative loading applications allow for an expected factor of safety much higher than what was calculated.

3.3.3 Nosecone/Payload Bay

The nosecone bay is a multi-purpose aerodynamic vehicle fairing and payload housing, Figure 3.9 and 3.10. The nosecone will be constructed with six layers of tubular fiberglass sleeve to conform to the surface without the need to cut gores with varying seam lengths. The nosecone will use a machined 6061 aluminum nosecone tip with a washer to keep the tip in place. The nosecone tip will also serve as a mounting point for an eye bolt and the forward main recovery retention. The nosecone will feature an extended length airframe diameter portion to provide ample volume to the payload bay. The nosecone is an approximate 4:1 length to diameter ratio, the measured construction will be less due to a blunted nosecone tip with a .25 (in) radius. The nosecone tip can be utilized with an extended threaded rod coupling for forward ballast placement with aluminum pucks should significant changes in vehicle mass occur.

The nosecone coupler will feature an aft in-flight separation point and a forward non-in-flight separation point. The coupler will be manufactured with multiple layers of fiberglass cloth.

The nosecone bulkhead will be retained via the payload system for removability upon landing, Figure 3.11. On the inner diameter of the nosecone coupler tubing, a carbon fiber tube will be epoxied to the inside of the tubing to provide a routing point for the shock cord. The shock cord will additionally route through the nosecone bulkhead. The bulkhead will be manufactured using the same method and face sheet thickness as the avionics bulkheads in 10.

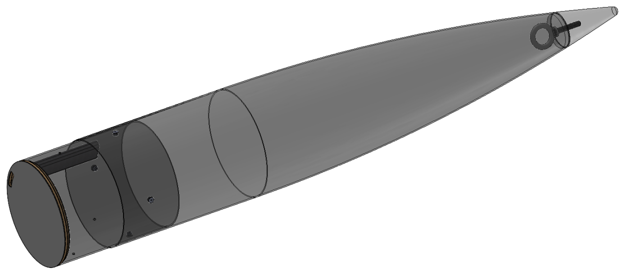


Figure 3.9: nosecone bay.

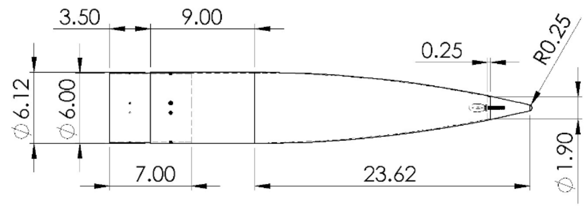


Figure 3.10: nosecone assembly dimensioned drawing (in).

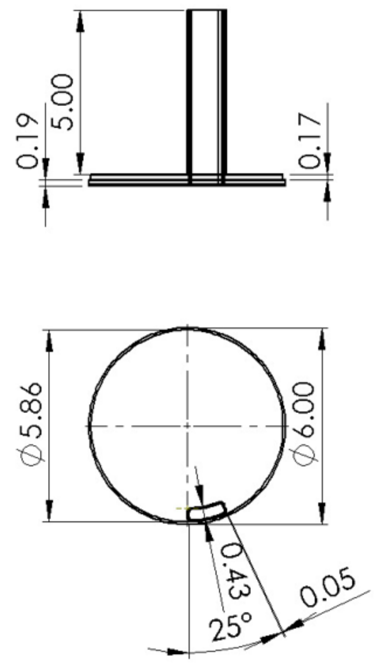


Figure 3.11: Payload bulkhead (in).

3.3.4 Avionics Bay

The avionics bay houses the avionics controlling energetic recovery deployment events, vehicle tracking, and altitude measurement, Figure 3.12 and 3.13. The avionics bay includes recovery attachment points for the forward end of the drogue shock cord and the aft end of the main shock cord. The recovery attachment points and energetic charges are mounted with bulkheads that additionally protect the avionics from black powder charges. The avionics bay is housed in coupler tubing, with an epoxied on the switchband for pull-pin and altimeter vent holes.

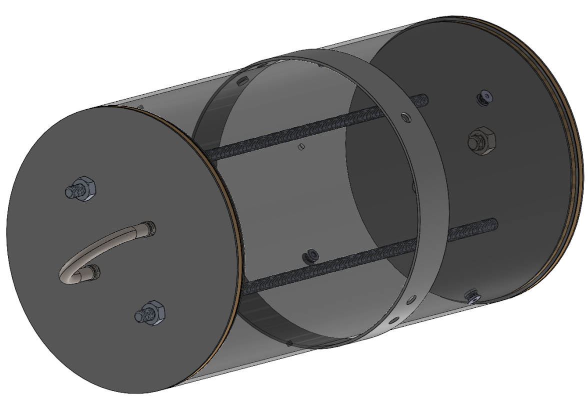


Figure 3.12: Avionics bay assembly.

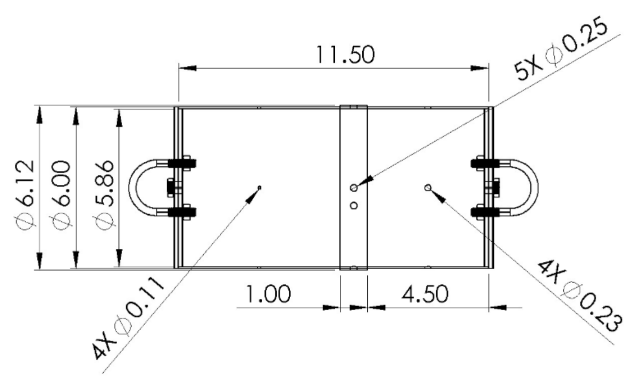


Figure 3.13: Avionics dimensioned drawing (in).

All vehicle bulkheads shall be designed to withstand the minimum shock loading expected to be 260 (lbf). All bulkheads will use 1/8 (in) thick honeycomb Nomex core sandwich composites, with variations in face sheet laminate thickness and material. The air brake bay and nosecone bulkheads will use carbon fiber face sheets to allow for thinner face sheets and lower-density materials to match the strength requirements. The avionics bay bulkheads will use S2 fiberglass face sheets for their proximity to radio frequency (RF)-emitting components. Bulkheads will be a stepped design with a larger and smaller diameter honeycomb Nomex core. The core materials will be bonded with a singular face sheet of composite laminate with face sheets on the exterior faces with their respective cloth material.

The avionics bay and air brakes bay bulkheads will include two pairs of drilled holes for the threaded rods and U-bolts. Additional access holes will be drilled for Wago lever connectors. The threaded rods will connect the forward and aft avionics bulkhead with nuts to provide shock load distribution and mounting points for the avionics assembly.

Bulkhead Preliminary Force Analysis

Roark's Formulas for Stress and Strain (7th Edition), Table 11.2, Case 1e, will be used to calculate the minimum thickness for a flat circular plate of constant thickness, Figure 3.14 [16]. Case 1e is applied to a circular flat plate with a fixed outer edge, free inner edge, central hole, and applied loading at a known distance. To simplify the component for calculation, it is assumed that the bulkhead is of constant diameter using the inner stepped bulkhead diameter. The shock loading will be converted to a shear loading over the circumference of the middle of a washer to be applied at the center of the bulkhead. Threaded rod attachment points will be disregarded. The simplifications taken will provide a conservative load estimate.

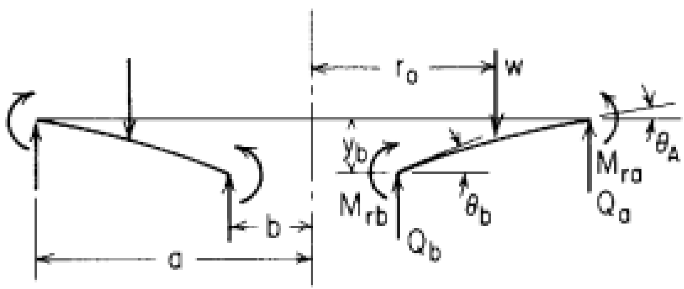


Figure 3.14: Roark's Formulas for Stress and Strain Table 11.2, Case 1.

Equations (4), (5), (6), and (7) are combined to calculate the shear and moment loading at the outside edge of the circular plate (8) and (9).

$$L_6 = \frac{r_0}{4a}\left[\left(\frac{r_0}{a}\right)^2 - 1 + 2\ln\left(\frac{a}{r_0}\right)\right] \tag{4}$$

$$L_9 = \frac{r_0}{a}\left[\frac{(1+\nu)}{2}\ln\left(\frac{a}{r_0}\right) + \frac{(1-\nu)}{4}\left(1-\left(\frac{r_0}{a}\right)^2\right)\right] \tag{5}$$

$$C_4 = \frac{1}{2}\left[(1+\nu)\frac{b}{a} + (1-\nu)\frac{a}{b}\right] \tag{6}$$

$$C_7 = \frac{1}{2}(1-\nu^2)\left(\frac{a}{b} - \frac{b}{a}\right) \tag{7}$$

$$Q_a = -\frac{wr_0}{a} \tag{8}$$

$$M_{r_a} = -wa\left(L_9 - \frac{C_7 L_6}{C_4}\right) \tag{9}$$

Following the moment and shear loading, the minimum bulkhead thickness can be calculated using experimentally determined material strengths. Table 3.5 provides the inputs, and Table 3.6 provides the outputs for both fiberglass and carbon fiber bulkhead configurations. The calculated 260 (lbf) shock loading 3.8.8 will be used as the loading. A 3.0 factor of safety will be included to account for the low-risk ballistic recovery deployment shock loading.

$$t_{bulkhead} = \sqrt{\frac{6M_{ra}}{\sigma_{laminate}}} \tag{10}$$

Table 3.5: Bulkhead thickness input parameters

$a(in)$	$b(in)$	$r_0(in)$	$w(\frac{lbf}{in})$	ν	$\sigma_{CF}(psi)$	$\sigma_{FG}(psi)$
2.93	0.1875	0.200	621	0.330	23800	17000

Table 3.6: Bulkhead thickness output parameters

Bulkhead	M_{ra} **($\frac{\text{in-lbf}}{\text{in}}$)**	Q_a **($\frac{\text{lbf}}{\text{in}}$)**	t_{bulkhead} **(in)**
Carbon Fiber	-64.0	-42.4	0.118
S2 Fiberglass	-64.0	-42.4	0.150

The avionics and nosecone bulkheads will be constructed using stepped plates of composite sandwich material 3.15. The bulkhead order from the aft to forward end of the rocket: 4 layers S2 8.9 oz/yd^2 fiberglass, $\frac{1}{8}$ (in) honeycomb core, 1 layers fiberglass, $\frac{1}{8}$ (in) honeycomb core, 4 layers fiberglass. The air brakes bay will utilize a laminate structure, but with 3 layers of 6 oz/yd^2 3K carbon fiber on the outer faces, and carbon fiber on the interior face.

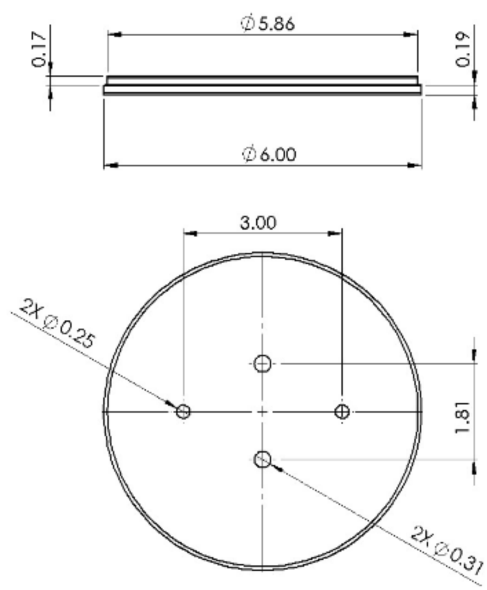


Figure 3.15: Avionics bay bulkhead (in).

3.4 Air Brakes Bay

The air brakes bay is similar in construction to the avionics bay, but without a switchband Figure 3.16 and 3.17. The bay will feature stepped bulkheads constructed using honeycomb Nomex core and carbon fiber face sheets, the number of layers is described in 10 with preliminary force calculations. The avionics bay couples the middle and fin can airframe tubing with PEM nuts and countersunk screws at non-in-flight separation points. Recovery load distribution is maintained with a u-bolt on the forward bulkhead, with an aluminum threaded rod connecting the forward and aft bay bulkheads with nuts. The aft end of the air brakes bay threaded rods can be extended as potential aft ballast placement points for mechanical fixture.

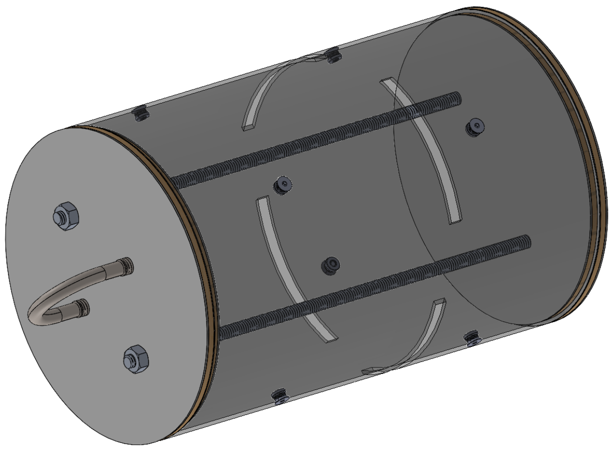


Figure 3.16: Air brakes bay assembly.

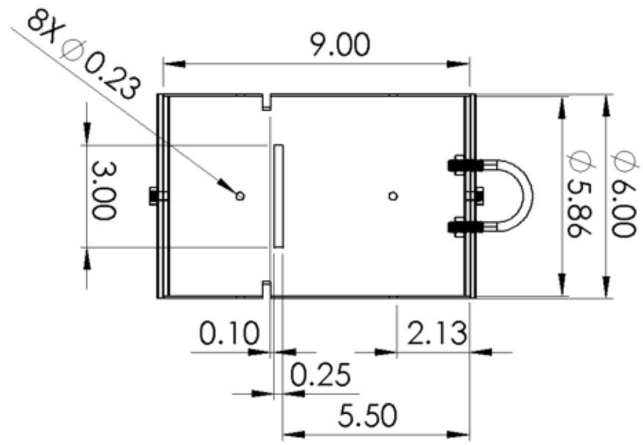


Figure 3.17: Air brakes dimensioned drawing (in).

3.4.1 Fin Can

The vehicle's fins are permanently bonded to the airframe body tubing in the fin can. The fin can contains the centering rings, thrust plate, motor retainer, motor tube, fins, and motor assembly, Figure 3.19. The fins are epoxied using external and internal fillets with through-wall construction attached to the central motor tube, Figure 3.18. The fins are fixed axially using the forward, aft, and middle centering rings. The middle centering ring additionally provides radial alignment using evenly spaced slots that interface with the fins. The aft centering ring is stepped to interface with the aft end of the airframe body tubing.

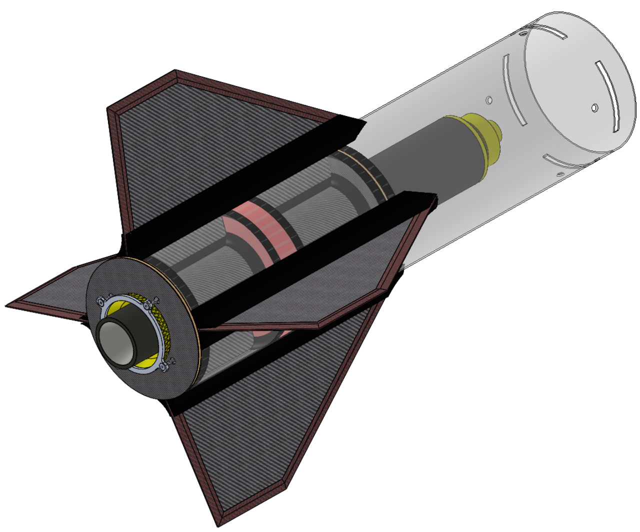


Figure 3.18: Fin can with fillets.

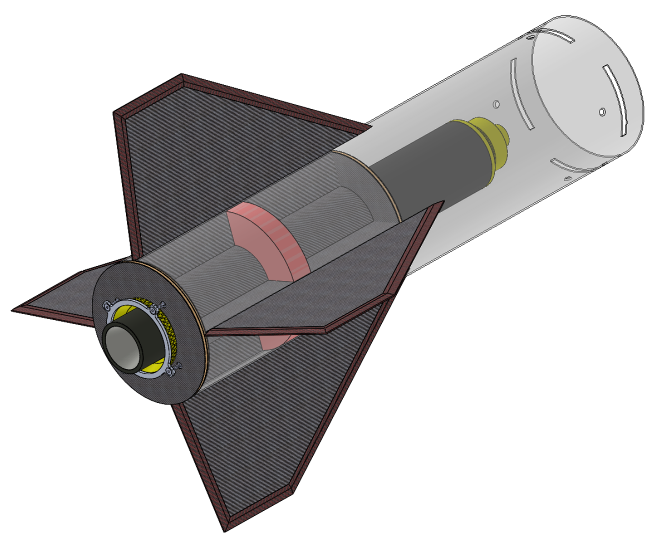


Figure 3.19: Fin can with no fillets.

Fins

Fins are constructed using $\frac{1}{8}$ (in) honeycomb Nomex core with carbon fiber face sheets, Figure 3.20. The fins will use molded fin fairings. The fins will use four plies of carbon fiber with symmetric [0/+45/90/-45]s weave orientations. The alternating weave orientation provides the majority of fibers following the direction of the individual fin center of pressure, where the loading will be centered during flight.

Future experiments will determine the shear modulus of the manufactured fins to calculate the flutter velocity of the fins. If flutter velocity experiments are shown to have a factor of safety less than 1.5, additional face sheets will be considered for the fin design.

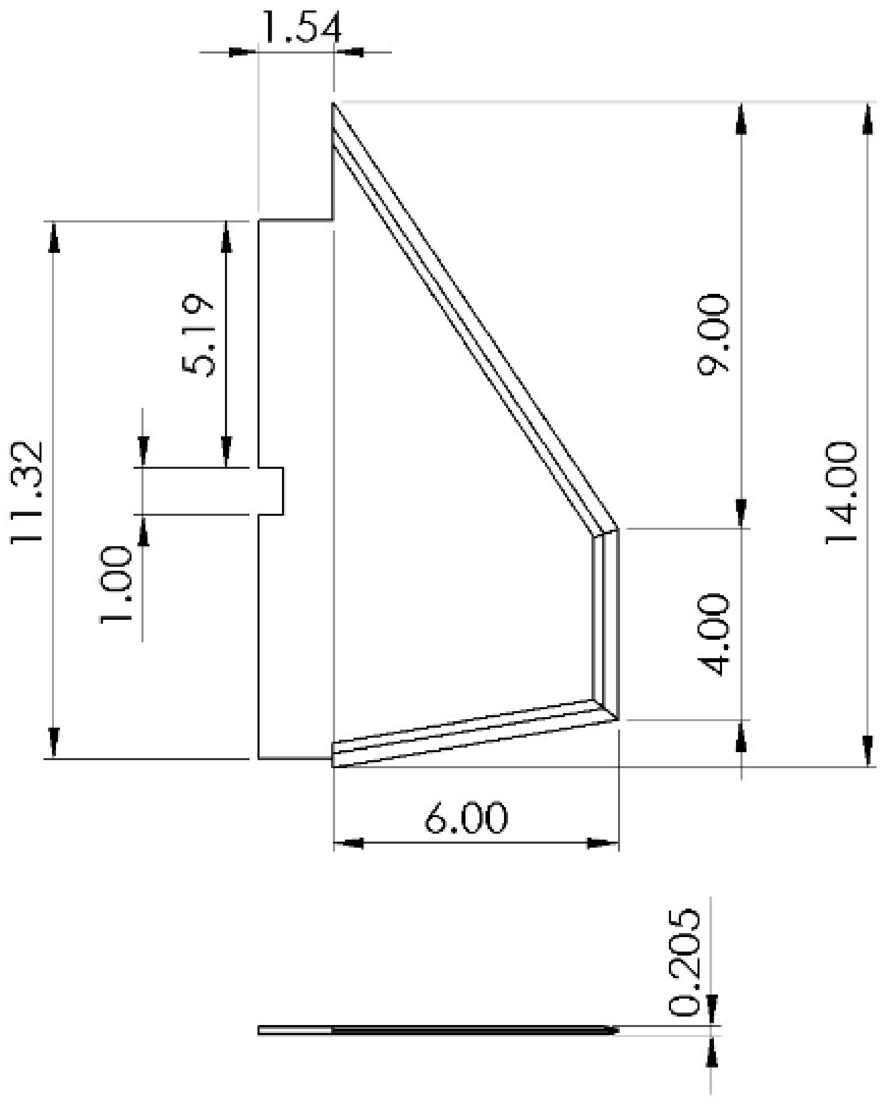


Figure 3.20: Fin (in).

Fin Can Tubing

The motor tube provides a centrally located mount for the fins to epoxy internally, and for the motor to slot into 3.21. The motor tube will be constructed with four plies of carbon fiber for a light construction due to the high compressive strength of carbon fiber for the compressive flight loads.

The fin can airframe tube provides an aerodynamic fairing for the fin structure and a coupling joint for the air brakes assembly 3.22. The fin can tube includes slots for the fin can to mount through. The slots are cut through the aft end of the tube and filled in once the fin assembly is bonded into the fin can tube.

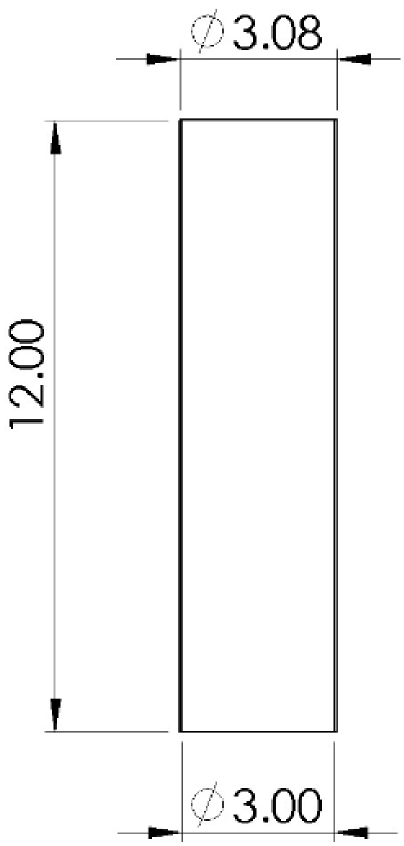


Figure 3.21: Motor tube (in).

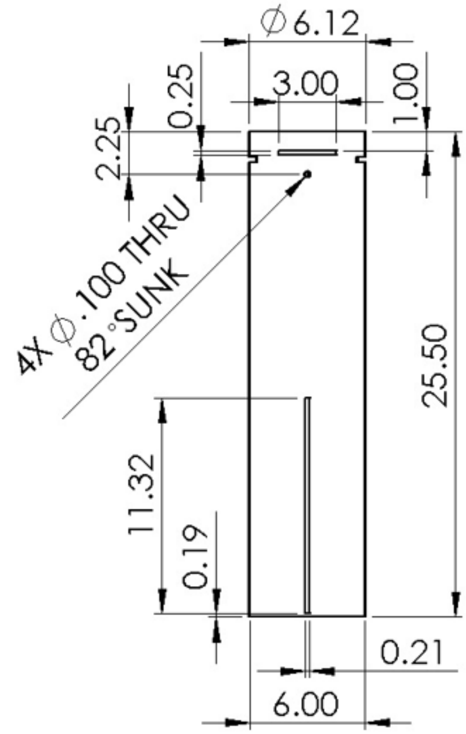


Figure 3.22: Fin can tube (in).

Centering Ring

The forward centering ring and aft thrust plate will be constructed of 1/8 (in) thick honeycomb Nomex core sandwich with carbon fiber composites. The thrust plate is a stepped configuration with construction matching the bulkheads. The middle centering ring will be 3D printed from PLA with low infill, Figure 3.24. Due to PLA's low elastic modulus and the high elastic modulus of the carbon fiber sandwich laminates, it is assumed that the centering ring does not experience significant loading.

Centering Ring Preliminary Force Analysis

The maximum compressive force will be used to calculate the anticipated loading applied to the centering rings from thrust and drag loadings. The maximum loading will be split between both the forward and aft centering rings. Loading and stress calculations will determine the face sheet thickness. To provide a conservative estimate, the loading will use the peak motor thrust and peak drag thrust. The reference area will additionally include fully extended air brake fins. Table 3.7 is provided to include the axial loading from peak thrust and drag forces. The peak axial compressive force experienced by the Launch Vehicleis calculated to be 482.4 (lbf).

Table 3.7: Vehicle Compressive Forces

Variable	Name	Value	Source
ρ	Air Density	0.00238 $\frac{slug}{ft^3}$	Standard Atmosphere at Sea Level
v	Max Velocity	693 $\frac{ft}{s}$	OpenRocket Simulation
A	Reference Area	0.357 ft^2	Calculated
C_D	Drag Coefficient	0.559	Aerodynamics Simulations
F_T	Peak Thrust	370.9 lbf	Motor Data Sheet
F_D	Peak Drag	111.5 lbf	Calculated

Roark's formula case 1e will be applied to determine the centering ring thickness, similar to 10. The minimum epoxy thickness that fixes the centering ring to the airframe body tubing and motor mount tubing will additionally be calculated. Table 3.8 provides the inputs, and Table 3.9 provides the outputs for both centering ring thickness and epoxy fillet coverage.

$$t_{epoxy} = \sqrt{\frac{6M_{ra}}{\sigma_{epoxy}}} \tag{11}$$

$$t_{CR} = \sqrt{\frac{6M_{ra}}{\sigma_{laminate}}} \tag{12}$$

Table 3.8: Centering ring and epoxy thickness input parameters

$a(in)$	$b(in)$	$r_0(in)$	$w(\frac{\text{lbf}}{\text{in}})$	ν	$\sigma_{CR}(psi)$	$\sigma_{epoxy}(psi)$
3.00	1.54	1.56	75	0.330	23800	9000

Table 3.9: Centering ring and epoxy thickness output parameters

Bulkhead	$M_{ra}(\frac{\text{in-lbf}}{\text{in}})$	$Q_a(\frac{\text{lbf}}{\text{in}})$	$t_{CR}(in)$
Centering Ring	-54.2	-38.4	0.107
Epoxy	-54.2	-38.4	0.255

The forward centering ring will be constructed of 3 layers of 6 oz/yd^2 3K carbon fiber fabric on each side of the $\frac{1}{8}$ (in) honeycomb Nomex core material for a .185 (in) total composite sandwich thickness. The aft thrust plate and centering ring combination will utilize a stepped layout with the order from the aft to forward end of the rocket: 3 layers carbon fiber, $\frac{1}{8}$ (in) honeycomb core, 2 layers carbon fiber, $\frac{1}{8}$ (in) honeycomb core, 3 layers carbon fiber, Figure 3.23. The aft thrust plate includes four socket holes for PEM nuts to be press-fit and epoxied into the bulkheads. The PEM nuts are for screws to connect the motor retainer and rocket.

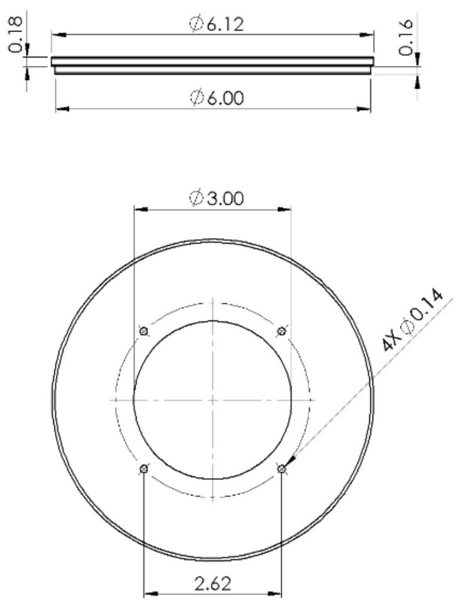


Figure 3.23: Thrust Plate (in).

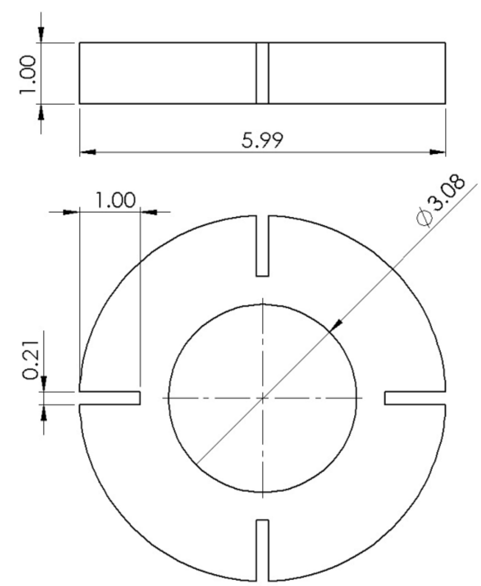


Figure 3.24: Middle Centering Ring (in).

Motor Retainer

The motor retainer will be manufactured from a flat plate of $\frac{1}{8}$ (in) 6061 aluminum Figure 3.25. The retainer will mount to the aft thrust plate using 6-32 countersunk screws in PEM nuts. The motor retainer loading includes the weight of the fully loaded motor, thus, a high factor of safety.

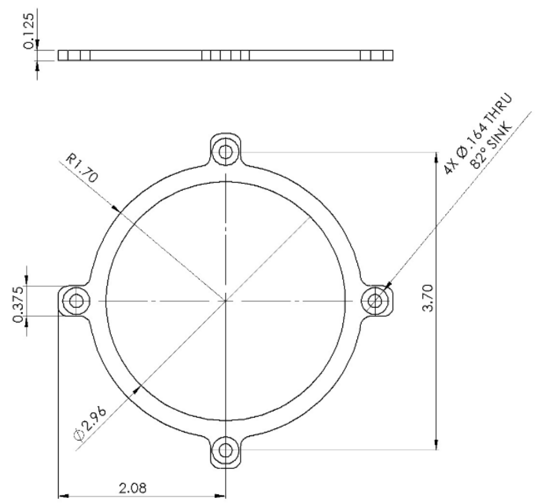


Figure 3.25: Motor retainer (in).

Fillets

Fillets applied to the fins on the internal and external construction will include a 10% fin radius relative to the fin's root chord for the internal fillets, and a 6% radius on the external aerodynamic surface. The larger internal fillet is due to the increased load with a larger bending moment arm. The 6% external fillet is guided for an effective aerodynamic surface with low turbulence between 4-8% [5].

3.5 Tail Cone Consideration

Tail Cone

A Tail Cone, in vehicle design, is used to reduce the drag force around the tail end of the vehicle. This is done by adding a smooth contour on the aft end of the vehicle, promoting airflow to stay attached to the body. This delays the onset of turbulence at the aft end of the vehicle, thereby decreasing the drag force experienced. The launch vehicle utilizes an Air Brakes system to intentionally increase the drag force. Considering that a tail cone is counterintuitive to this aspect of the vehicle design, its benefits would not yield any noticeable results based on the club's past experience. Therefore, a tail cone will not be employed for the launch vehicle's design.

3.6 Launch Vehicle Weight Estimates

Table 3.10: Summary Weight Estimate

Section	Weight [lbs.]
nosecone/Payload Bay	8.20
Main Parachute Bay	4.58
Avionics Bay	3.24
Drogue Parachute Bay	3.05
Air Brakes Bay	4.02
Fin Can	12.92
Launch Vehicle Total	**36.00**

/n

Table 3.11: Section Weight Estimates

nosecone/Payload Bay	
Component	Weight [lbs.]
Machined Tip	0.15
nosecone Body	1.98
nosecone Coupler	0.69
Payload	5.00
Bulkhead	0.18
Eyebolt	0.20
Section Subtotal	**8.20**

Main Parachute Bay	
Component	Weight [lbs.]
Main Parachute	1.25
Shock Cord	0.96
Soft Links	0.06
Airframe	2.31
Section Subtotal	**4.58**

Avionics Bay	
Component	Weight [lbs.]
Avionics & Threaded Rods	1.32
2 x U-bolts	0.40
2 x Bulkheads	0.36
Airframe	1.16
Section Subtotal	**3.24**

Drogue Parachute Bay	
Component	Weight [lbs.]
Drogue Parachute	0.125
Shock Cord	0.96
Soft Links	0.06
Coupler	1.90
Section Subtotal	**3.05**

Air Brakes Bay	
Component	Weight [lbs.]
Air Brakes & Threaded Rods	2.646
U-bolt	0.20
2 x Bulkheads	0.32
Coupler	0.85
Section Subtotal	**4.02**

Fin Can	
Component	Weight [lbs.]
Fins	1.05
Epoxy Fillets	0.55
Centering Rings and Thrust Plate	0.27
Motor Retainer	0.13
Motor Tube	0.27
Airframe	2.11
Loaded Motor	8.55
Section Subtotal	**12.92**

3.7 Motor Selection

Due to design constraints of the Air Brakes coupler outlined in section 3.4.1 nestling in 4.5" deep into the top of the Fin Can, motor options are limited. The only two motors meeting the NASA requirements 2.7, 2.8, 2.9, and fit into the space constraints, are the AeroTech L1390G and AeroTech L1520T. An igniter that is capable of being set off using a 12-volt direct firing system will be employed satisfying NASA requirement 2.6. Moreover each of the motors are not oriented forward facing, are not Sparky motors, and not a hybrid motor satisfying NASA requirements 2.20.1, 2.20.2, 2.20.3, and 2.20.4 since only one motor will be used on launch day.

Table 3.12: Motor Specifications and Details

Motor	Propellant Mass (slug)	Total Mass (slug)	Total Impulse (lbf·s)	Average Thrust (lbf)	Maximum Thrust (lbf)	Burn Time (s)	Casing	Length (in)
L1390G	0.1351	0.2657	887.77	312.48	370.89	2.6	RMS-75/3840	20.86
L1520T	0.1270	0.2501	835.37	352.46	396.85	2.4	RMS-75/3840	20.39

Outlined above in Table 3.12 is all details and specifications relevant to the launch vehicle. Between the two motors, they are similar in many aspects except their burn times, average, and peak thrust values.

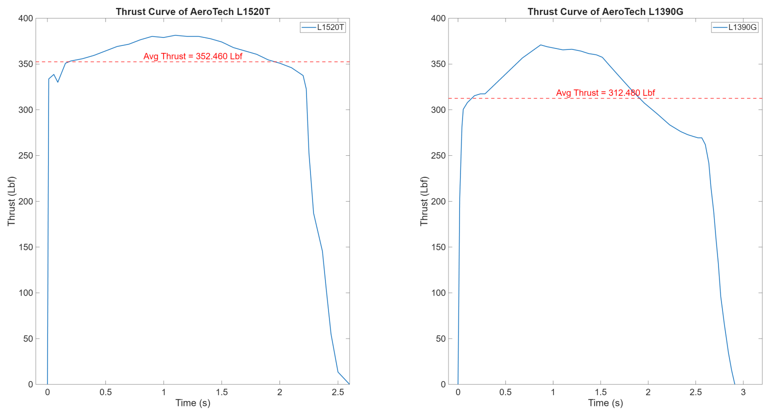


Figure 3.26: Thrust Curves of Motor Choices

As shown in figure 3.26, the thrust curves of the two motors could not be more different. The L1520T produces a smoother burn rate over the burn time, and also delivering a greater average thrust at 352.46 lbf. The main issues with this motor is the faster burn time and higher overall thrust produced. This decreases the time available for the Air Brakes control system to gather the flight profile necessary, coupled with the higher acceleration, in G's, experienced throwing off measurements taken by the IMU. Both of these are further outlined in section 5.3.3. The L1390G meets the NASA requirements 2.12, 2.14, 2.15, with a max acceleration of 9.56 G's and a total burn time of 2.6 seconds.

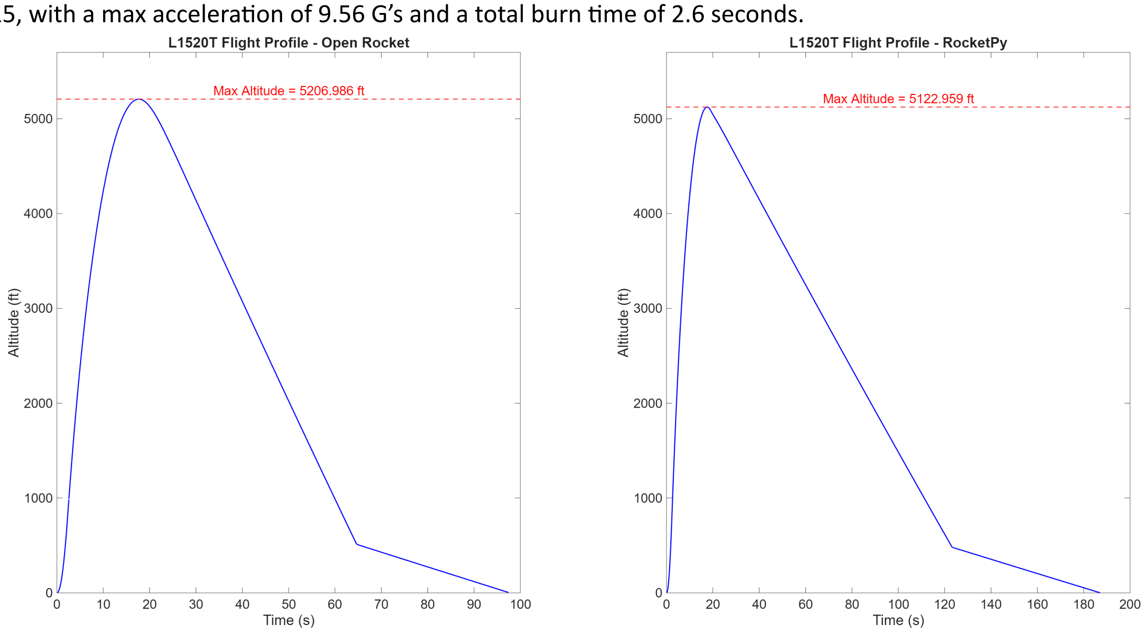


Figure 3.27: Flight Profile of the AeroTech L1520T

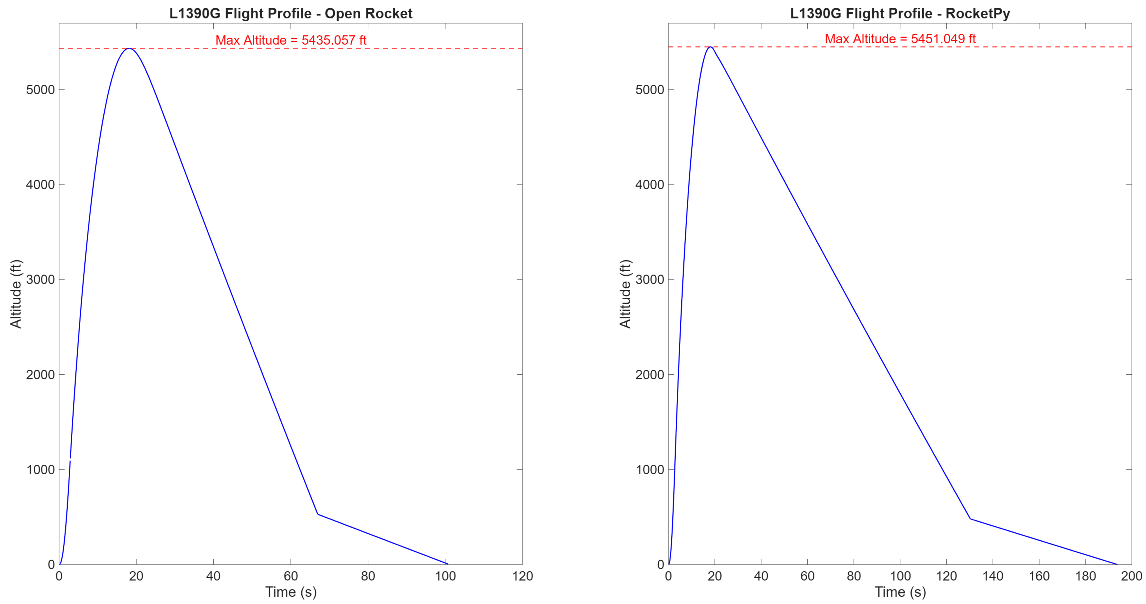


Figure 3.28: Flight Profile of the AeroTech L1390G

Further shown in figures 3.27 and 3.28 are the flight profiles of each motor through two different software suites, OpenRocket on the left most plot and RocketPy on the right. Both of the simulation suites show that the L1390G will carry the launch vehicle to a greater apogee, around 5435 ft, which ensures that the Air Brakes System will have altitude to work with to bring the vehicle to the target apogee.

Criteria	Weight	L1390G	L1520T
Length	2	6	7
Average Thrust	1	7	9
Burn Time	3	9	5
Total Impulse	1	7	6
Apogee	2	8	5
Acceleration	1	8	6
Unweighted Totals		44	37
Weighted Totals		**77**	55

Table 3.13: Motors Pugh Matrix.

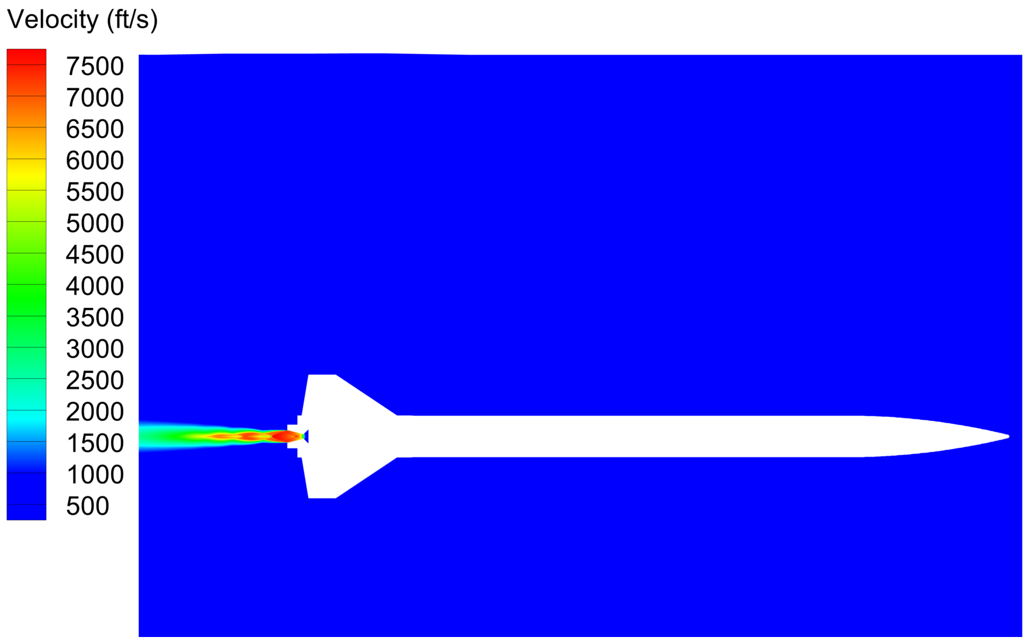


Figure 3.29: AeroTech L1390G Firing at Average Thrust

Lastly, figure 3.29 shows the outlet Mach Number from the AeroTech L1390G when producing average thrust. Moreover table 3.13 highlights the decision matrix for the chosen motor. In all, with all the factors discussed above, the chosen motor for the launch vehicle is the AeroTech L1390G. This will enable a launch rail exit velocity of 78.5 (fps) and a thrust to weight ratio of $8.58 : 1$. The backup motor for the launch vehicle will be the AeroTech L1520T.

3.8 Alternative Recovery Components

The recovery system is designed to operate through two distinct recovery events: one for drogue parachute deployment and one for main parachute deployment.

The first recovery event occurs at apogee when the primary altimeter initiates the primary drogue black powder charge on the aft end of the Avionics Bay (AV Bay). One second later, the secondary altimeter triggers the secondary drogue black powder charge in the same location, providing system redundancy. The ignition of these black powder charges produces a pressure buildup within the Drogue Bay that shears the connecting pins between the AV Bay and Drogue Bay. This separation allows the drogue parachute to deploy, stabilizing the launch vehicle during the initial descent phase.

The second recovery event occurs at 550 (ft) above ground level. At this point, the primary altimeter activates the primary main black powder charge on the forward end of the AV Bay. At 500 (ft), the secondary altimeter fires the secondary main black powder charge on the same end, again ensuring redundant deployment. The resulting pressure buildup in the Main Parachute Bay breaks the shear pins connecting the Main Parachute Bay to the nosecone, causing the sections to separate and allowing the main parachute to deploy. The vehicle then continues descending under the main parachute until touchdown.

The AV Bay serves as the central hub for recovery system components and is designed to keep all subsystems organized, compact, and structurally sound. This ensures consistent performance, ease of integration, and reliable recovery operations.

Important system design considerations include the selection and configuration of the following components: altimeters, trackers, batteries, switches, drogue and main parachutes, parachute protection systems, deployment mechanisms, shock cord material and length, AV sled material, ejection charge containment, and structural attachment hardware.

3.8.1 Altimeters

Altimeters serve as the primary control units within the recovery system, continuously measuring the launch vehicle's altitude throughout flight and initiating parachute deployment at predetermined recovery events. To ensure system reliability and compliance with redundancy requirements, the recovery system incorporates two dual-deploy altimeters operating on fully independent electrical circuits.

In accordance with NASA Requirement 3.3, all candidate altimeters are commercially available barometric units specifically designed for automated initiation of recovery events in high-power rocketry applications. Each selected device is capable of recording and storing flight data, including altitude, velocity, and time, for the full duration of the mission.

Altimeter selection criteria were based on an evaluation of size, measurement accuracy, reliability, power consumption, cost-effectiveness, and available features. Table 3.14 provides a summary of the key parameters considered in the decision-making process.

Table 3.14: Altimeter Options

Altimeter	Size		Power	Cost	Owned	Sensors		Features		
	Length	Width				Barometer	Accelerometer	GPS	Tele	Radio
Silicdyne Fluctus	4.30”	1.00”	3.7V - 10V	$400	✓	✓	✓	✓	✓	✓
Eggtimer Quasar	5.50”	1.09”	7.4V	$100	✓	✓		✓	✓	✓
Altus Metrum EasyMini	1.50”	0.80”	3.7V - 12V	$80	✓	✓				
PerfectFlite StratologgerCF	2.00”	0.84”	4V - 16V 9V Nominal	$70	✓	✓				

Silicdyne Fluctus

As seen in Figure 3.30, the Silicdyne Fluctus is a compact and versatile flight computer. It has integrated GPS, dual 3-axis accelerometers, and a gyrometer for precise attitude and velocity measurements. The incorporated GPS eliminates the need for a separate tracking device, reducing the required AV Bay mass and size. It supports Bluetooth and long-range radio communication with a ground station for real-time telemetry, remote control, and configuration. The Fluctus has a high altitude limit and Mach-tolerant apogee detection. The onboard display, light-emitting diodes (LED), and buzzer provide clear feedback during setup and flight. The potential limitations of the Fluctus are its size, being 4.30(in) long, and it’s lofty price. The size requirement is well within the Team’s Subscale and Full-scale AV Bay designs. Additionally, the club already owns this device, making it a significantly more accessible option. While a Fluctus has never been used by the North Carolina State University High-Powered Rocketry Team, the device has extensive positive reviews and a reliable track record.

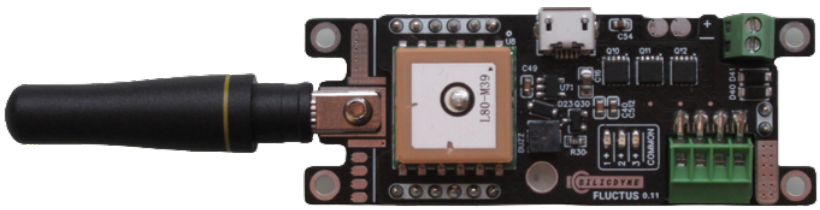


Figure 3.30: Silicdyne Fluctus

Eggtimer Quasar

The Eggtimer Quasar, shown in Figure 3.31, is a WiFi-enabled dual-deployment altimeter that allows users to program, arm, disarm, and retrieve flight data wirelessly, eliminating the need for physical cables. A key advantage of the Quasar is its integrated GPS functionality, enabling it to serve both as an altimeter and as the primary flight tracker, thereby reducing the number of required avionics components. The primary limitation of the Eggtimer Quasar is its physical size. With an overall length of 5.50 inches, the Quasar requires an avionics (AV) Bay of at least equal length for proper integration. The length, however, is well within the available space of both the Team’s Subscale and Full-scale AV Bay designs.

Figure 3.31: Eggtimer Quasar

Altus Metrum EasyMini

The Altus Metrum EasyMini, shown in Figure 3.32, is a strong candidate for dual-deployment applications due to its compact form and consistent performance. Its small size enables efficient integration within constrained avionics bay configurations, and its operation on a single-cell LiPo battery allows for the use of a smaller power source, further reducing mass and space requirements. The team currently owns an EasyMini, making it an accessible and cost-effective option. The North Carolina State University High-Powered Rocketry Team has successfully employed the EasyMini on a NASA Student Launch competition vehicle, demonstrating its reliability and consistent performance in flight.

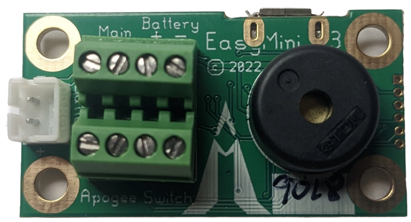


Figure 3.32: Altus Metrum EasyMini

PerfectFlite Stratologger

The PerfectFlite StratologgerCF, as seen in Figure 3.33, is a compact and reliable altimeter widely respected within the hobby rocketry community for its balance of performance and affordability. The team has extensive experience with StratologgerCF units, reinforcing confidence in their consistent functionality. At 2.00(in) long, this altimeter represents the second smallest option under consideration and is already owned by the Team, eliminating additional procurement costs for the current project. A notable limitation is the device's reliance on a 9V alkaline battery as recommended by the manufacturer, in contrast to the lithium polymer batteries preferred for other units, which may impact integration flexibility.

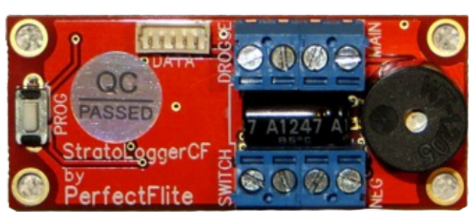


Figure 3.33: PerfectFlite StratologgerCF

Leading Altimeters

Currently, the leading design includes one Silicdyne Fluctus as the primary altimeter and one Altus Metrum EasyMini as the secondary altimeter. The Fluctus is designated as the primary altimeter due to its advanced functionality, precision, and robust system integration capabilities that exceed those of other devices considered. It combines multiple sensors, including dual high-resolution accelerometers, a gyrometer, GPS, and a barometric pressure sensor, to ensure accurate flight data and fault-tolerant apogee detection. The Fluctus also provides high-rate data logging, Bluetooth and radio telemetry, and real-time communication with a ground station, offering complete monitoring and post-flight analysis. These capabilities establish the Fluctus as a technically superior and reliable primary altimeter, while the Altus Metrum EasyMini serves as a simplified, redundant secondary system.

The EasyMini is a great choice for a backup altimeter due to its simplicity, compact design, and proven reliability in dual-deployment operations. It utilizes a high-precision barometric sensor to control parachute ejection events accurately while maintaining a streamlined configuration that simplifies setup. The device's lightweight construction and minimal power requirements make it easy to integrate into nearly any avionics bay without significant modifications. With a well-established track record in competition rocketry, the EasyMini provides dependable performance, making it a trustworthy and efficient secondary system to complement the more advanced primary altimeter.

Criteria	Weight	Silicdyne Fluctus	Eggtimer Quasar	Altus Metrum EasyMini	PerfectFlite StratologgerCF
Size	2	8	7	10	9
Power	1	10	10	10	10
Cost	1	7	8	9	10
Owned	2	10	10	10	10
Sensors	2	10	5	5	5
Features	2	10	10	0	0
Unweighted Totals		55	50	44	44
Weighted Totals		**93**	84	69	68

Table 3.15: Altimeters Pugh Matrix.

3.8.2 Tracking Devices

Per NASA Requirement 3.12, a GPS tracking device will be installed in the launch vehicle and will transmit the location of the tethered vehicle and any independent sections. All sections of the launch vehicle will remain tethered for the duration of the descent, thus one GPS tracker will be installed in the launch vehicle. The tracker will be installed in the AV Bay, alongside all other avionics components. Since the recovery area of the launch vehicle is limited to 2500 (ft), the tracker must be able to transmit across a minimum of 5000 (ft) Table 3.16 compares the Team's tracking options. Important factors considered in the comparison include the size, transmission frequency and range, cost, and accessibility of the device.

Table 3.16: Tracking Options

Tracker	Size		Transmitter Frequency	Range	Cost	Owned
	Length	Width				
Silicdyne Fluctus	4.30"	1.00"	900 MHz	6.00 miles	$400	✓
Eggtimer Quasar	5.50"	1.09"	900 MHz	6.00 miles	$100	✓
Eggfinder Mini	7.00"	3.25"	900 MHz	1.50 miles	$75	✓

Leading Tracker

As discussed in Sections 3.8.1 and 3.8.1, both the Fluctus and the Quasar can function as GPS trackers in addition to their altimeter capabilities. By selecting flight computers that serve dual roles as both altimeters and GPS trackers, the design benefits from significant mass and size reductions. Consolidating these functions into a single device reduces the total number of electronic modules, simplifying internal organization and wiring while minimizing potential points of failure. Additionally, since all three tracker options operate on the same 900 MHz frequency band, no HAM license is required for operation. Each satisfies the minimum range requirement of 5000 (ft), with the Fluctus offering the greatest range, up to 6 miles, making it the most capable option for maintaining consistent telemetry throughout flights. Consequently, the Fluctus emerges as the leading choice for a GPS tracker in this launch system.

3.8.3 Altimeter Arming Devices

Each altimeter installed in the AV Bay will be armed using a mechanical switch, accessible from the exterior of the launch vehicle, and be capable of being locked in the "on" position for the duration of the flight, satisfying NASA Requirement 3.5 and 3.6. The two switches being considered for implementation into the AV Bay are the Lab Rat Rocketry Double Pull Pin Switch and the Missile Works Screw Switch.

Lab Rat Rocketry Double Pull Pin Switch

The double pull pin switch assembly, showing in Figure 3.34, integrates two micro switches, a pin guide, and an actuating pin that interfaces with the active surfaces of the switches. When the pin is inserted, pressure on the switch actuators keeps both circuits in an open state. Removing the pin releases the actuators, closing each circuit and thus powering the connected devices. The design allows a single pin to simultaneously control both switches, enabling independent operation of two circuits while maintaining a unified mechanical interface. One drawback of this configuration is the need for the switch assembly to be mounted directly inline with the Switchband of the AV Bay. Positioning the switch elsewhere would prevent removal and assembly of the AV Bay without unintentionally arming the connected devices. Despite this limitation, the double pull pin system offers a significant convenience advantage. Once installed, the pin can easily be removed on the launch pad, providing a reliable and simple method for arming the avionics system immediately before flight.

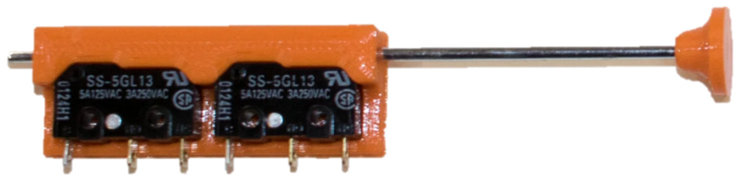


Figure 3.34: Lab Rat Rocketry Double Pull Pin Switch

Missile Works Screw Switch

The screw switch, shown in Figure 3.35, comprises a printed circuit board (PCB) with a nut fixed on one side and a corresponding conductive pad on the opposite side. Tightening the screw establishes electrical continuity between the nut and the pad, closing the circuit. Once secured, the screw remains fixed, resisting loosening from typical forces experienced during launch. Unlike pull pin switches, screw switches do not require alignment with the Switchband, meaning the AV Bay can be assembled without altering the circuit state. However, implementation of a screw switch requires drilling two access holes in the launch vehicle's airframe to allow screwdriver access. Additionally, tightening screws on the launch pad can be challenging due to limited accessibility.

Figure 3.35: Missile Works Screw Switch

Leading Switch

The leading switch design is using one Lab Rat Rocketry Double Pull Pin Switch to arm both the primary and secondary altimeters. This decision was motivated by the simplicity and convenience of the arming process. Although a single physical device is used, the switch design enables independent control of each altimeter's circuit, ensuring that both can be armed and disarmed independently despite sharing the same pull pin. This configuration balances simplicity with necessary redundancy, providing reliable initiation of dual-deploy recovery systems without complicating launch pad preparations.

3.8.4 Batteries

Batteries for each device will be selected in accordance with the manufacturer recommendation. A 1S or 2S Lithium Polymer (LiPo) battery is recommended for the Silicdyne Fluctus. A 7.4V 800 mAh LiPo battery will be used for the Fluctus. The Fluctus has a maximum power draw of 400 mW, thus the operating time can be calculated using Equation 13,

$$t = \frac{QU}{P} \tag{13}$$

where t is operating time, Q is battery capacity, U is battery voltage, and P is device power. The 7.4V LiPo battery will sustain the Fluctus for 14.8 hours.

The Altus Metrum EasyMini will be supplied power via a 3.7V 500 mAh LiPo battery. With a nominal current draw of 10 mA, the 3.7V LiPo will be able to power the EasyMini for 185 hours. Both battery choices satisfy NASA Requirement

2.2, that devices should remain launch-ready for a minimum of 3 hours without losing functionality or any critical on-board components. All LiPo batteries used for avionics have a nominal discharge temperature range from -4 °F to 140 °F, satisfying team-derived requirement RE 3.

3.8.5 Avionics Sled Materials

All avionics components will be fastened to the AV sled, which is mounted within the AV Bay. The design of the AV sled prioritizes cost efficiency, manufacturability, and structural integrity of the selected material. Previous iterations have utilized birch plywood and various 3D printed filamens as primary construction materials.

3D Printed Filaments

One AV sled material option is printing the sled with Polyethylene Terephthalate Glycol (PETG) or PLA filament. The primary advantage of this method is the ease of modeling and fabrication, eliminating the need for complex jigsaw geometries required in plywood assemblies. Additionally, 3D printing enables the integration of intricate geometries and spatially efficient designs that allow compact component placement. A limitation of this approach is that all hardware mounting features must be incorporated into the computer aided design(CAD) model prior to fabrication, as the infill structure of 3D printed parts typically lacks the density required to securely anchor fasteners for avionics hardware.

Plywood

To fabricate the AV sled from plywood, each panel is laser cut with interlocking jigsaw-style edges that align during assembly. The panels are then bonded along these joints using epoxy adhesive to form a rigid structure. Manufacturing the AV sled from plywood provides several advantages, including high structural strength and quick fabrication time. The material has demonstrated durability, with no recorded failures during launch-induced loading conditions. Additionally, plywood allows for easy modification, such as drilling, to accommodate component mounting.

Leading Sled Material

The current leading design utilizes a 3D printed AV sled fabricated from PLA. PLA was selected over PETG due to its higher rigidity, ease of modeling and printing, and smoother, more consistent surface finish. While PETG offers greater flexibility and impact resistance, PLA provides the stiffness necessary to maintain structural stability and precise component alignment within the AV bay.

3.8.6 Parachute Selection and Sizing

In accordance with NASA requirement 3.2, the maximum allowable kinetic energy of the heaviest independent section of the launch vehicle at landing must not exceed 75 ft-lbf, with additional credit awarded to teams maintaining kinetic energy below 65 ft-lbf. Requirements 3.10 and 3.11 specify that the rocket's drift distance from the launch pads shall not exceed 2,500 ft, and the total descent time from apogee must be limited to 90 seconds. Teams achieving descent times under 80 seconds will receive bonus points. Relevant details regarding these performance metrics are elaborated in Mission Performance Predictions Sections 7.5 through 3.10.9. These criteria were central to the parachute selection process for both drogue and main deployment. Secondary considerations included cost efficiency, with preference given to parachutes already in the team's inventory, as well as packing volume and total system mass.

Drogue Parachute

The main role of the drogue parachute is to decelerate the vehicle to a speed where the sudden force from deploying the main parachute does not cause the shock cord or shroud lines to fail, nor inflict damage on the launch vehicle. Additionally, the drogue assists in stabilizing and properly orienting the vehicle during descent, ensuring reliable main parachute deployment. Table 3.17 summarizes different drogue parachutes options alongside important parameters to help inform design decisions.

Table 3.17: Drogue Parachute Options

Parachute	Drag Coefficient	Descent Velocity	Descent Time: Apogee to Main Deployment	Max Drift Distance: Apogee to Main Deployment	Owned
15" Elliptical	1.5	120.27 fps	33.67 s	987.75 ft	✓
18" Elliptical	1.5	100.23 fps	40.41 s	1185.30 ft	✓
24" Elliptical	1.5	75.17 fps	53.88 s	1580.40 ft	✓
30" Elliptical	1.5	60.14 fps	67.35 s	1975.50 ft	

Meeting the descent velocity criteria is critical to ensure the launch vehicle adheres to descent time limits, described in NASA Requirement 3.11. Excessive descent speed risks inflicting damage on the launch vehicle during main parachute deployment, informing team derived requirement RF 8, requiring the descent velocity under drogue parachute be less than 120 fps. For this reason, the 15 (in) elliptical parachute is ruled out. Although the 30 (in) elliptical chute complies with maximum descent time and drift distance requirements, it is unsuitable for pairing with a main chute that would also meet kinetic energy landing limits of 75 ft-lbf. Consequently, the 18(in) and 24 (in) chutes present themselves as viable drogue parachute candidates. Among these, the 18 (in) chute is preferred as it comfortably meets the Team's maximum descent velocity threshold, thereby providing greater flexibility in selecting a compatible main parachute.

Main Parachute

The main parachute's function is to decelerate the launch vehicle sufficiently to guarantee that ground impact occurs without causing damage, thereby enabling full and successful recovery of the vehicle. Another critical factor in selecting the main parachute is ensuring that, together with the drogue parachute, the vehicle's total descent time and drift distance remain within NASA's prescribed limits. A list of all parachutes considered during the selection process is presented in Table 3.18.

Table 3.18: Main Parachute Options

Parachute	Drag Coefficient	Descent Velocity	Descent Time: from Main Deployment	Kinetic Energy	Max Drift Distance: Apogee to Main Deployment	Owned
Fruity Chutes Iris Ultra 72" Compact	2.2	21.02 fps	26.17 s	99.53 ft-lbf	767.59 ft	
Fruity Chutes Iris Ultra 84" Compact	2.2	17.73 fps	31.01 s	70.86 ft-lbf	909.72 ft	✓
Fruity Chutes Iris Ultra 96" Compact	2.2	15.76 fps	34.89 s	55.99 ft-lbf	1023.45 ft	✓
Fruity Chutes Iris Ultra 120" Compact	2.2	12.61 fps	43.61 s	35.83 ft-lbf	1279.32 ft	✓

The 72 (in) Iris Ultra parachute is deemed unsuitable as it exceeds the maximum allowable kinetic energy limit of 75 ft-lbf required by the competition. All other parachutes sufficiently reduce the vehicle's descent speed to satisfy this criterion. Conversely, the 120 (in) Iris Ultra chute decelerates the vehicle excessively, resulting in descent times and drift distances that surpass acceptable limits, thus disqualifying it as an option. The 84 (in) and 96 (in) Iris Ultra parachutes emerge as the most viable choices, capable of ensuring a safe and timely descent. The 96(in) chute is preferred to the 84 (in), as the 84 (in) chute exceeds the kinetic energy requirement for the bonus points. When paired with an 18(in) drogue parachute, the 96(in) chute maintains the launch vehicle's landing kinetic energy below 65 (ft-lbf) and achieves touchdown within 80 seconds, qualifying for bonus points on both metrics.

Leading Parachutes

The preferred configuration currently involves deploying an 18 (in) in-house fabricated elliptical parachute as the drogue and a 96 (in) Iris Ultra Compact parachute as the main. This selection offers the launch vehicle a total descent time of 75.33 seconds, a maximum drift distance reaching 2209.71 (ft), and a landing kinetic energy of 55.99 ft-lbs. This setup provides the Team with sufficient leeway to accommodate minor changes in vehicle mass while still complying with all safety and competition requirements. Additionally, this combination enables the Team to qualify for bonus points related to the descent time and kinetic energy criteria.

Parachutes will be packed according to reputable rocketry manufacturers' guidelines, per team derived requirement RF 1, ensuring consistent and reliable deployment for each flight.

3.8.7 Parachute Materials

Calendared Ripstop Nylon

Calendared ripstop nylon is a specialized fabric commonly used in high-powered rocketry parachutes due to its superior durability and strength-to-weight ratio. Calendared nylon has a tensile strength of 24.7 lbf/in[1]. Alternatives, such as nylon with a polyurethane coating, have a lower tensile strength of around 18.9 lbf/in[2]. The calendaring process involves passing the nylon fabric through heated rollers, which smooth and compact the material, increasing its tensile strength and reducing porosity. This process improves the fabric's resistance to tearing and increases its stiffness while maintaining flexibility, important properties for parachute canopies subject to high aerodynamic loads. Additionally, the ripstop weave pattern, characterized by thicker reinforcement threads woven at regular intervals, helps prevent tears from propagating across the canopy. These combined properties make calendared ripstop nylon an ideal choice for ensuring reliable deployment, longevity, and safety of parachutes in rocketry applications, where weight savings and material performance are significant.

Figure 3.36 illustrates the distinct weave patterns of three different fabrics, providing a clear comparison of their microscopic structures. When examining ripstop fabric under a microscope, the weave displays sharply defined lines and a consistent, uniform spacing of reinforcing threads. This precision in the pattern stands in contrast to more common fabrics, such as those used in everyday t-shirts, which exhibit a less distinct and irregular texture without the pronounced grid structure found in ripstop. The crispness and organization of the ripstop weave contribute significantly to its durability and resistance to tearing.

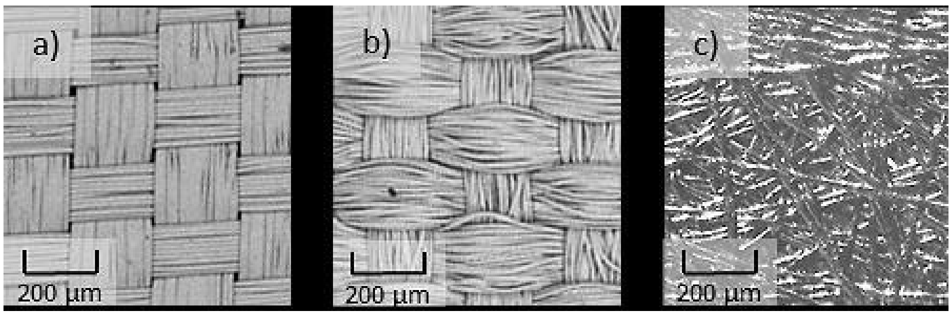


Figure 3.36: a) Ripstop, b) plain weave, and c) nonwoven.[15]

Figure 3.37 depicts the difference between a typical fabric weave and a ripstop weave. In ripstop fabric, thicker yarns are woven at regular intervals ranging from 0.2 to 0.3 inches[15]. These thicker threads, shown in blue, act as reinforcing ribs that possess significantly higher tensile strength compared to the surrounding yarns. This structural arrangement increases the fabric's resistance to tear propagation, effectively preventing small tears from expanding and thus improving the overall durability and reliability of the material.

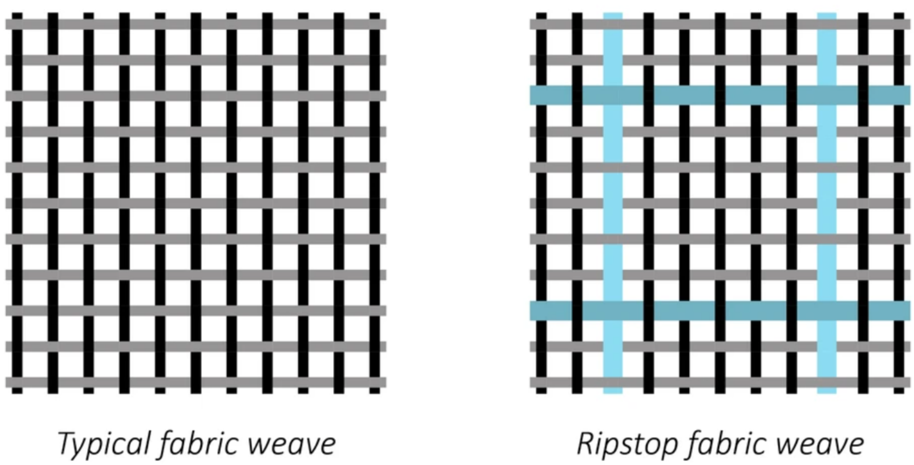


Figure 3.37: Typical fabric weave vs. ripstop fabric weave.[15]

Uncalendared Ripstop Nylon

Uncalendared nylon fabric lacks the additional surface treatment that calendared nylon possesses, which means it retains a softer, more flexible, and generally more porous texture. While this can offer greater breathability, uncalendared nylon is typically less resistant to air permeability and abrasion. In high-powered rocketry parachute applications, calendared nylon is preferred over uncalendared fabric due to its enhanced mechanical properties. The calendaring process compresses the nylon fabric, reducing porosity and increasing tensile strength, tear resistance, and stiffness, qualities that contribute to more predictable and reliable parachute deployment under high dynamic loads. Using calendared nylon increases fabric durability and overall parachute performance, which are critical for ensuring safe and consistent recoveries.

Thread Material

When comparing 100% bonded nylon thread to traditional cotton thread for parachute construction in rocketry, the advantages of nylon are clear. Bonded nylon thread offers significantly higher tensile strength, flexibility, and excellent resistance to abrasion, chemicals, and environmental factors such as moisture and UV exposure. It maintains its performance under the repeated stresses of parachute deployment and heavy loads typical in rocketry applications. In contrast, cotton thread, while soft and easy to handle, is more prone to breaking under tension and degrades quickly when exposed to moisture, abrasion, or prolonged use. Cotton fibers also lack the stretch and durability of nylon, making them unsuitable for situations where reliability and longevity are necessary. For these reasons, bonded nylon thread is strongly preferred over cotton in the construction of rocketry parachutes, ensuring secure seams and the integrity of recovery systems even under harsh conditions.

French Seam

A French seam, shown in Figure 3.38, is a self-enclosed seam construction method where the raw edges of the fabric are wrapped inside the seam, resulting in a neat and tidy finish. The technique involves sewing the wrong sides of the fabric together, trimming the seam allowance, pressing the seam, then folding the fabric right sides together and stitching a second line enclosing the raw edge. While French seams offer moderate strength and aesthetic appeal, they are not the optimal choice for parachute construction. The strength of a French seam is generally less than that of flat-felled used in high-stress textiles because of its uneven stress distribution along the seam line. In parachute applications, minimizing seam bulk and maximizing tensile strength and integrity under dynamic loads are important. Thus, alternate seam techniques such as a flat-felled seam is favored for its superior strength, durability, and resistance to failure under tension.

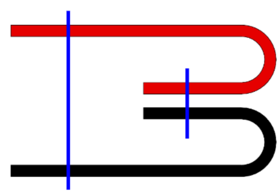


Figure 3.38: French seam.

Flat-Fell Seam Variation

A flat-felled seam is a strong and durable seam where fabric edges are folded and sewn together in such a way that all raw edges are fully enclosed within the seam. Typically, one seam allowance is trimmed narrower and then the wider edge is folded over it before being stitched down, creating a flat, clean finish with two rows of topstitching. This technique produces an exceptionally strong and durable seam capable of withstanding substantial tensile loads and abrasion, making it a favored choice in rocketry applications. For parachutes, flat-felled seams are ideal because they minimize bulk while preventing fabric fraying and seam failure during the high stresses of deployment and descent. The secure, enclosed edges reduce the risk of seam unraveling and distribute loads evenly. The variation of a flat-felled seam, shown in Figure 3.39b is the preferred method, due its simplicity and strength.

(a) Traditional flat-fell seam. (b) Flat-fell seam variation.

Figure 3.39: Flat-fell seam variations.

UHMWPE Shroud Lines

Ultra-high-molecular-weight polyethylene (UHMWPE) is a popular choice for shroud lines in parachutes due to its exceptional strength-to-weight ratio, high tensile strength, and excellent resistance to abrasion and environmental degradation such as UV exposure and moisture. It is incredibly lightweight, which helps reduce the overall mass of the parachute system, improving deployment performance and reliability. One drawback of UHMWPE as a shroud line material is its low coefficient of friction, which makes the fibers slippery and prone to tangling more easily than other materials. This slipperiness leads to difficulties in knotting, as UHMWPE lines tend to slip and untie themselves. Additionally, the low friction among fibers can cause rope layers to twist or distort under load, complicating line management and deployment. While UHMWPE's strength and abrasion resistance are excellent, these handling and knotting challenges require careful design consideration.

Kevlar Shroud Lines

Kevlar is a popular choice for parachute shroud lines due to its excellent abrasion resistance and thermal stability, which help ensure the lines remain intact under the harsh conditions of Launch Vehicledeployment and descent. While Kevlar's tensile strength is lower compared to UHMWPE, it offers enhanced durability and is easier to sew and handle during parachute construction. This ease of sewing can result in more reliable and secure attachment points. Additionally, Kevlar's ability to resist cuts and wear better protects the shrouds from damage caused by sharp edges or rough surfaces. Overall, Kevlar balances strength and practical usability, making it a preferred material in parachute applications where toughness and longevity are important.

Parachute Design

Elliptical parachutes are a popular choice in rocketry due to their efficient fabric usage, compared to hemispherical chutes, and effective aerodynamic performance. By selecting an elliptical parachute with a ratio of $\frac{b}{a} = 0.7$, where b is the vertical axis and a is the horizontal axis, a shape is achieved that balances drag efficiency with material usage. This ratio ensures that the parachute canopy has a flattened ellipsoid form, which produces a slightly higher drag coefficient comparable to hemispherical designs but requires less fabric. This reduction in material not only decreases weight but also packing volume. Additionally, the elliptical shape promotes stable descent characteristics, helping to reduce oscillations and sway. Overall, an elliptical parachute offers a combination of performance and resource conservation for effective vehicle recovery. An elliptical parachute design, where $\frac{b}{a} = 0.7$, is shown is Figure 3.40.

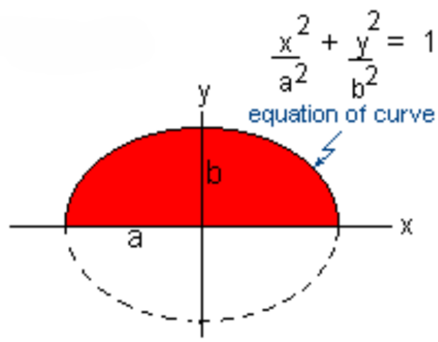


Figure 3.40: Ellitpical parachute design.[6]

Chutemaker is a valuable online tool designed to generate precise gore patterns for parachute construction. It allows users to input specific parameters such as the number of panels, parachute diameter, and seam allowances to create accurate templates for cutting fabric gores. This streamlines the design process by providing ready-to-use patterns that ensure consistent panel sizes for sewing. Using Chutemaker simplifies the transition from design to fabrication, reducing manual calculations and errors. An example of a gore pattern generated by the program is shown in Figure 3.41. Chutemaker is the preferred source for gore templates.

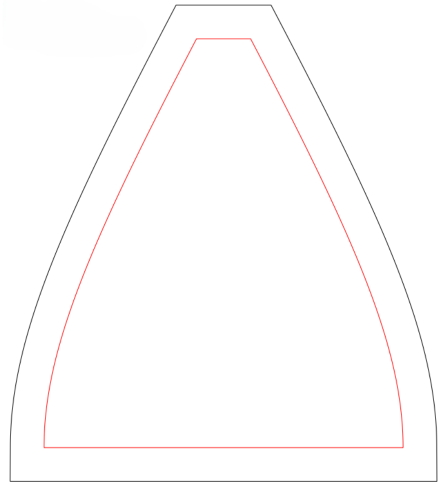


Figure 3.41: Parachute gore pattern.[11]

3.8.8 Shock Cord

Shock cord is a crucial element in maintaining the structural integrity of a launch vehicle during descent by keeping all sections tethered together. It must be sufficiently strong to absorb the forces generated by the rapid change in acceleration during parachute deployment and durable enough to withstand the heat and mechanical stresses caused by ejection charges. The length of the shock cord plays an important role in preventing collisions between vehicle sections during descent. A longer shock cord allows for gradual absorption of separation forces, reducing the risk of damage such as zippering.

A commercial soft link will be used to connect the shock cord to the launch vehicle attachment points. In this setup, the shock cord is attached to the launch vehicle's U-bolt via the soft link, replacing traditional metal quick links. The shock cord itself is secured with a bowline knot to the soft link. The parachute is also connected to the shock cord by a soft link, using an alpine butterfly knot.

Figure 3.42a depicts the descent configuration of the launch vehicle following the deployment of the drogue parachute. In accordance with team derived requirement RD 6, a minimum clearance of 10 (ft) must be maintained between all independent vehicle sections during descent. Specifically, this requirement mandates that when the shock cord is fully extended, the fins on the Fin Can and the upper U-bolt on the AV Bay remain separated by no less than 10 (ft)

In the diagram, D_1 denotes the distance between the AV Bay U-bolt and the drogue parachute connection point, while D_2 represents the distance from the Fin Can U-bolt to the same parachute connection. The chosen parachute connection point is positioned at 20% of the distance along the shock cord from the AV Bay connection, ensuring it is closer to the AV Bay end. Considering these lengths along with the combined length of the forward sections of the launch vehicle, L_{FWS}, and the length of shock cord contained within the Fin Can, D_{FC}, the total drogue shock cord length, L_{drogue}, can be determined using the following system of equations.

$$L_{drogue} = D_1 + D_2 \tag{14}$$

$$D_2 = 4D_1 \tag{15}$$

$$D_2 - D_{FC} = D_1 + L_{FWS} + 120\,\text{in.} \tag{16}$$

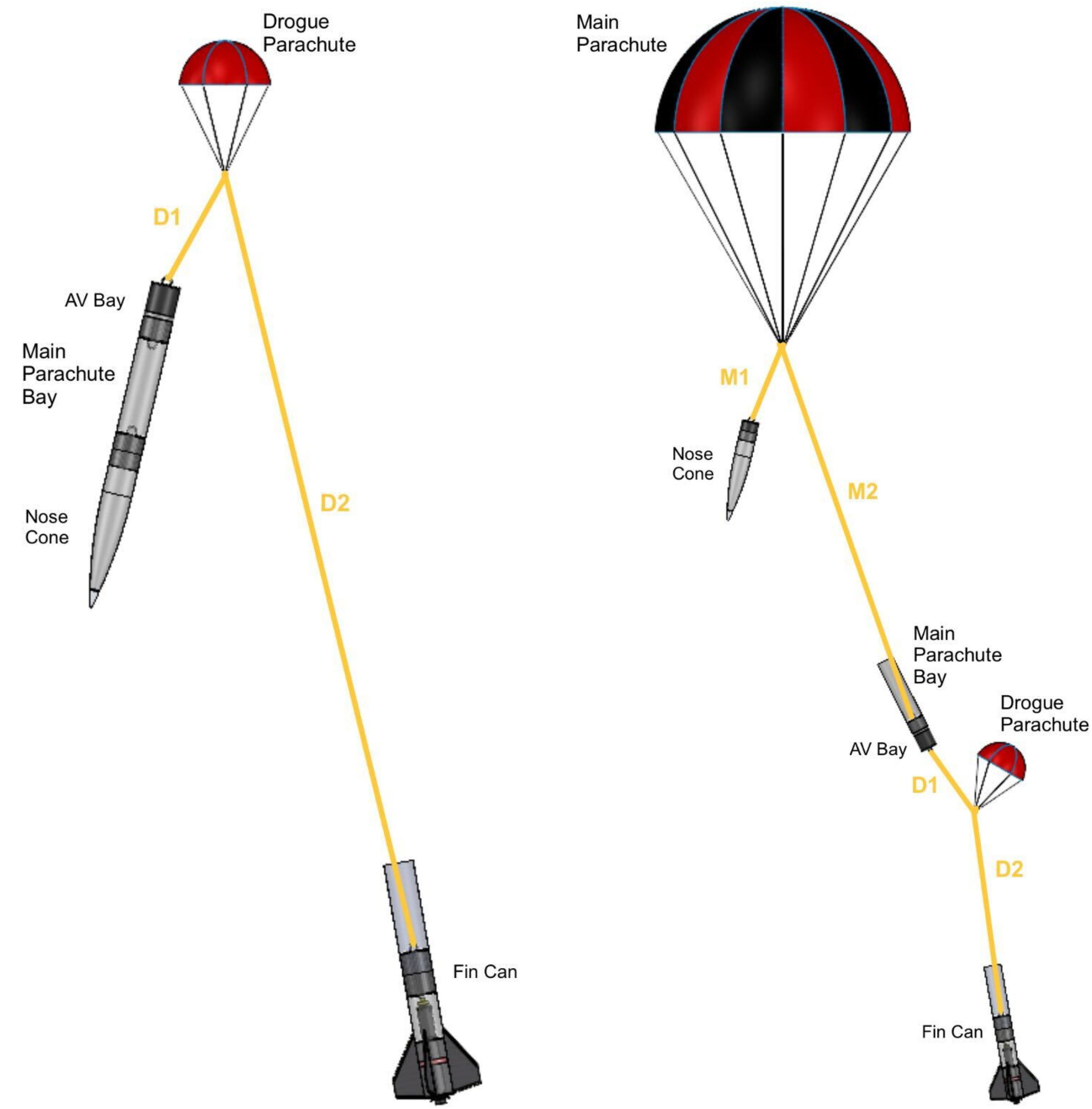
Using a L_{FWS} of 59.50 (in) and a D_{FC} of 23.00 (in), the minimum length of shock cord needed to satisfy all requirements is 28.13 (ft) A shock cord of length 28.50 (ft) will be used for drogue deployment.

Figure 3.42b illustrates the configuration of the launch vehicle during descent following the main parachute separation event. Consistent with the requirements applied during drogue parachute descent, a minimum separation distance of 10 (ft) must be maintained between all vehicle sections. Specifically, this necessitates a minimum spacing of 10 feet between the tip of the nosecone and the upper attachment point of the Main Parachute Bay.

In the figure, M_1 denotes the distance between the nosecone U-bolt and the parachute attachment point, while M_2 represents the distance between the Main Parachute Bay U-bolt and the parachute connection. Analogous to the drogue shock cord configuration, the parachute attachment point is positioned at 20% of the total shock cord length from the nosecone connection. Using these constraints along with the known nosecone length, L_{NC}, and the length of main shock cord contained within the Main Parachute Bay, M_{MPB}, the total main shock cord length, L_{main}, is calculated using the following system of equations.

$$L_{main} = M_1 + M_2 \tag{17}$$
$$M_2 = 4M_1 \tag{18}$$
$$M_2 - M_{MPB} = M_1 + L_{NC} + 120\,\text{in.} \tag{19}$$

Using a nosecone length of 31.50 (in) and the length of shock cord contained within the Main Parachute Bay being 28.00 (in), the required length of shock cord is 24.93 (ft) A length of 25.00 (ft) of shock cord will be used for the main parachute deployment.



(a) Drogue deployment configuration.

(b) Main deployment configuration.

Figure 3.42: Deployment configurations.

Parachute Opening Shock

In order to calculate the maximum opening shock force that will act on the shock cord, the inflation time of the parachute must be obtained. The largest force acting on the launch vehicle will be when the main parachute is deployed, slowing the launch vehicle from 100.15 (fps) to 15.76 (fps) Equation 20 is used to estimate the inflation time,

$$t_{infl} = \frac{nD}{v_d} \tag{20}$$

where t is the time it takes for the parachute to inflate, n is the canopy fill constant, D is the nominal diameter of the parachute, and v_d is the descent velocity at the time of opening. Using an approximated canopy fill constant of 4, a nominal diameter of 96 (in), and a descent velocity of 100.15 (fps), the parachute inflation time is calculated to be 0.3195 seconds. The expected shock force on the shock cord is then calculated using Equation 21,

$$F_{shock} = \frac{m\Delta v}{t_{infl}} \tag{21}$$

where F_{shock} is the shock force, m is the dry mass of the launch vehicle, Δv is the change in velocity, and t_{infl} is the previously calculated inflation time. Using a dry mass of 31.63 lbm, a change in velocity of 84.39 (fps), and an inflation time of 0.3195 seconds, the maximum force the shock cord will experience is 259.42 (lbf).

The team has selected 5/8 (in) Kevlar shock cord for all shock cord applications. Kevlar offers exceptional mechanical strength and outstanding resistance to high temperatures, making it well-suited for the launch vehicle recovery system. The 5/8 (in) Kevlar cord features a maximum rated strength of 6,600 pounds, yielding a factor of safety of 25 relative to expected operational loads. Although this strength exceeds the actual requirements for the expected forces, the team has opted for this material as it is already owned by the Team, eliminating the need for additional procurement or expense.

3.8.9 Ejection Charge Sizing and Calculations

Ejection charges are employed to separate the individual sections of the launch vehicle and initiate parachute deployment. The sequence begins when altimeters activate the e-matches housed within the blast caps at both ends of the AV Bay. Upon ignition, these e-matches light the black powder, which rapidly generates a high pressure environment inside the parachute bays. This elevated pressure produces sufficient force to break the shear pins, thus separating the launch vehicle sections. The pressure required to break the shear pins is calculated using Equation 22,

$$P = \frac{F}{A} \tag{22}$$

where P is pressure, F is the force required to break the shear pins, and A is the cross-sectional area of the launch vehicle. Four 4-40 shear pins will be used to keep the sections of the launch vehicle in place until specified recovery events take place. The shear pins have a known strength of 76 (lbf) Knowing the shear strength of the shear pins and the cross-sectional area of the launch vehicle, 28.27 (in^2), the pressure required to break four shear pins is calculated to be 10.75 (psi). A factor of safety of 1.25 is applied to the mass of the primary ejection charges. The mass of the black powder charge can then be calculated using the ideal gas law,

$$m = \frac{PV}{RT} \tag{23}$$

where P is the pressure required to break the shear pins, V is the volume of the section, R is the gas constant for black powder, and T is the temperature of the gas produced by black powder combustion.

777 FFFg black powder will be used for the ejection charges. This black powder has been selected based on its fine particle size, which promotes quick combustion, cleaner separation events, and minimal residual powder within the system. Thermodynamic parameters for black powder combustion include a gas constant of 22.16 ft-lbf/lbm- $^\circ$R and an adiabatic combustion temperature of 3307 $^\circ$R. Volumetric assessments estimate the available empty space within the main parachute bay, including shock cord and parachute, at 330.26 (in ^{3}), while the corresponding empty volume for the drogue parachute bay is calculated at 350.57 (in ^{3}). In addition to the primary ejection charge, a secondary charge will be incorporated as a redundancy measure for both main and drogue separation events.

In alignment with Team Derived requirement RD 3, the secondary charge mass is determined using a factor of safety of 2 applied to the pressure required for shear pin failure. Table 3.19 summarizes the calculated values for both primary and secondary ejection charges for main and drogue parachute deployments.

Table 3.19: Ejection Charge Sizing

Separation Event	Volume of Section	Primary Charge Mass	Secondary Charge Mass
Drogue Parachute Deployment	350.57 in^3	2.40 g	3.90 g
Main Parachute Deployment	330.26 in^3	2.30 g	3.70 g

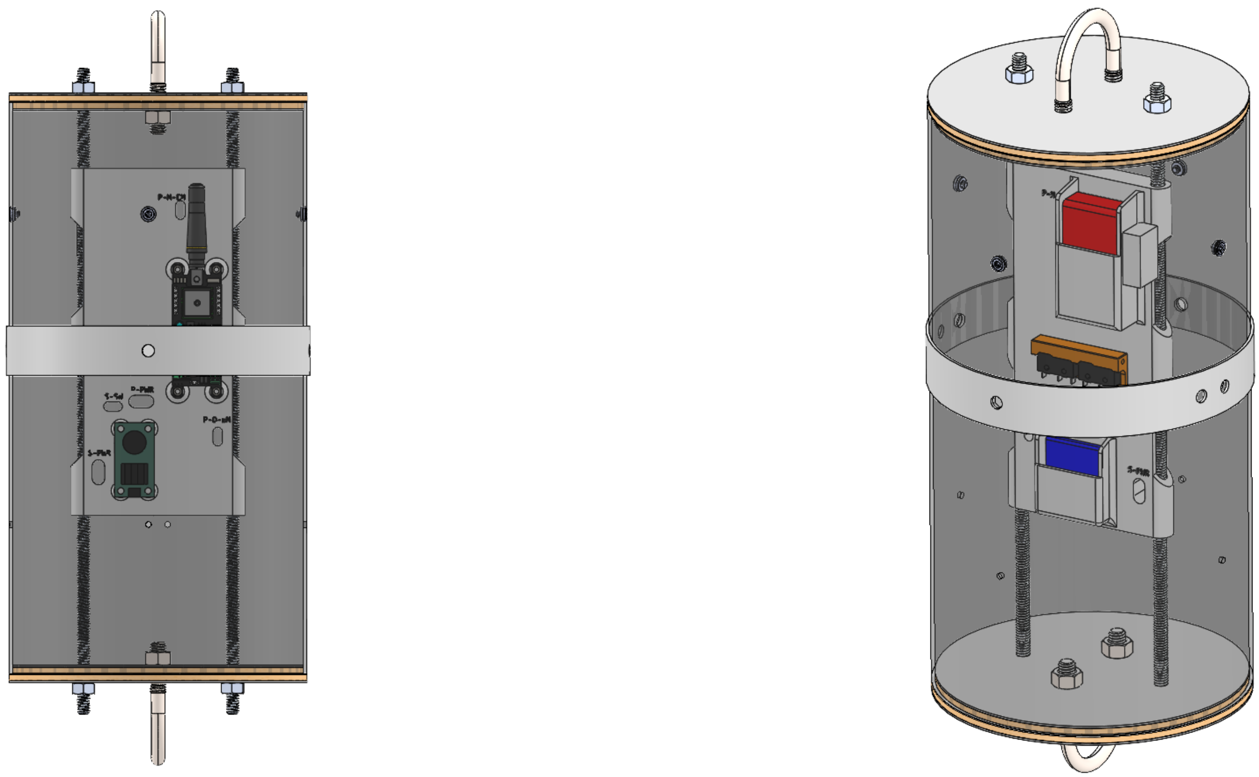
The validity of the calculated ejection charge masses was confirmed using Chuck Pierce's Black Powder Ejection Charge Calculator[10]. Consistent with NASA Requirement 3.1, ground ejection tests are conducted to verify the primary charge sizing prior to launch. These tests involve assembling the Launch Vehiclein its final launch day configuration and simulating separation events on the ground. In cases where the vehicle fails to achieve separation using the calculated charges, increments of 0.2 grams are added to the ejection charge mass. The tests are repeated until successful separation is obtained.

The ejection charge wells will be made of 3D printed PLA. Pressure vessels will not be used for ejection charges. Firewire Electric Matches will be used to ignite the black powder charge.

3.9 Leading Recovery Design

3.9.1 Avionics Bay Design

Figure 3.43 shows all avionics components necessary for the recovery system. The leading design includes one Silicdyne Fluctus acting as the primary altimeter and tracking device, one Altus Metrum EasyMini as the secondary altimeter, one 3.7V 500 mAh LiPo battery, one 7.4V 800 mAh LiPo battery, and one Lab Rat Rocketry Double Pull Pin Switch. Each altimeter is powered by an independent battery on separate circuits.



(a) AV Bay view 1.

(b) AV Bay view 2.

Figure 3.43: AV Bay.

3.9.2 Recovery Electronics

Figure 3.44 illustrates the wiring diagram for the primary altimeter. This circuit operates independently from all other avionics systems and is powered by a 7.4V 800 mAh LiPo battery. The Fluctus incorporates two terminal blocks, which secures wires to establish all electrical connections to the altimeter. The power connection is routed through the pull pin switch to enable controlled activation.

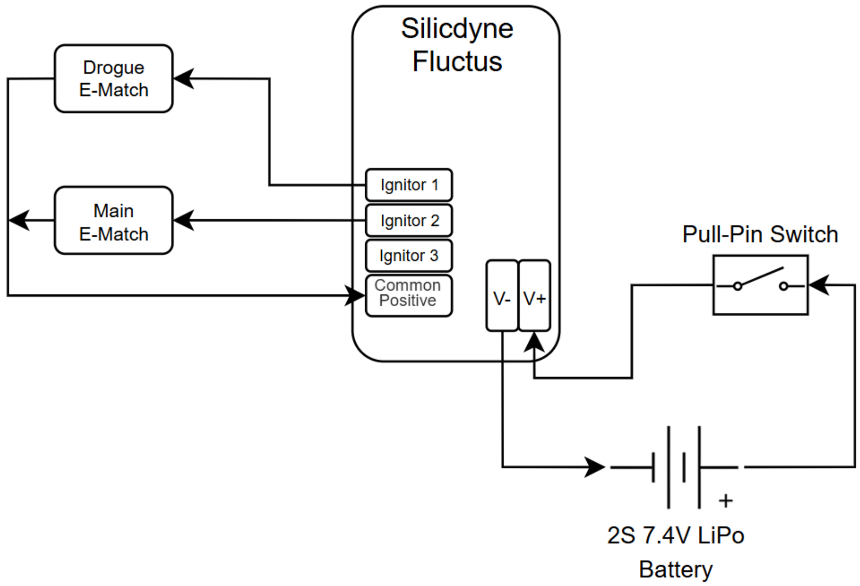


Figure 3.44: Primary altimeter wiring diagram.

Figure 3.45 shows the wiring layout for the secondary altimeter. The Altus Metrum EasyMini altimeter operates on a dedicated 7.4V 800mAh LiPo battery. Similar to the Fluctus configuration, the EasyMini features two terminal blocks that secure the wires, forming all necessary electrical connections to the altimeter. Additionally, one of the pull pin microswitches interfaces directly with the EasyMini's integrated switch.

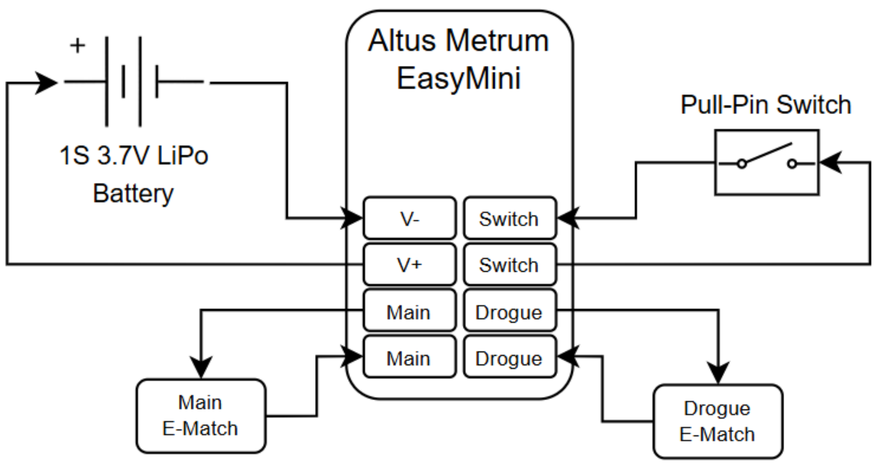


Figure 3.45: Secondary altimeter wiring diagram.

3.9.3 Recovery Events

The vehicle's dual-deployment recovery system initiates its first event at apogee, where the primary altimeter activates a 2.40 g black powder charge within the drogue parachute bay. One second later, the secondary altimeter triggers a redundant 3.90 g black powder charge to ensure reliable separation. This sequence results in the deployment of an 18 (in) elliptical drogue parachute, which is connected between the Fin Can and the Avionics Bay using 28.50 (ft) of 5/8 (in) Kevlar shock cord, as illustrated in Figure 3.42a.

The second recovery event occurs during descent at an altitude of 550 (ft) At this point, the primary altimeter initiates a 2.30 g black powder charge in the main parachute bay. The secondary altimeter follows at 500 (ft), igniting a redundant 3.70 g charge to confirm full separation. The Fruity Chutes Iris Ultra 96 (in) Compact main parachute then deploys, tethered between the nosecone and the Avionics Bay with 25 (ft) of 5/8 (in) Kevlar shock cord, as shown in Figure 3.42b.

From apogee at 4,600 (ft) to main parachute deployment at 550 (ft), the 31.63 lbm launch vehicle descends under drogue deployment for 40.41 seconds at an average velocity of 100.15 fps. Following main parachute deployment, the vehicle continues its descent for 34.89 seconds at 15.76 fps until touchdown. At landing, the heaviest vehicle section has a kinetic energy of 55.99 ft-lbf.

3.9.4 Recovery Launch Preparation

In preparation for launch, the primary and secondary altimeters, as well as the GPS tracker, will be ground tested to ensure appropriate ejection charge size and device functionality. The primary and secondary altimeters will be tested differently, as each has different capabilities and limitations.

The Altus Metrum EasyMini will be tested using a PCB connected to the main and drogue output terminals on the altimeter. Two independent circuits on the board, utilizing different color LEDs in place of e-matches, serve to simulate deployment events. The altimeter and testing board are then placed in a pressure chamber. Pressure is removed from the chamber to simulate launch vehicle ascent, and slowly increased to simulate descent. The EasyMini must be verified to be working properly before it is installed in the Avionics Bay.

The Silicdyne Fluctus will be utilized during ejection testing to verify both the altimeter's functionality and ejection charge sizing. For testing, the Fluctus will be configured to trigger ejection charges via remote command through the Fluctus Control Center software. This remote capability allows the Team to maintain a safe distance from the vehicle and verify that the Fluctus is working as it should.

Altimeters and tracking devices remain unarmed until immediately prior to launch, when the vehicle is vertically positioned on the launch rail, ensuring safety during handling and preparation. Arming is performed before inserting the motor igniter to verify the functionality of the recovery system in case of premature motor ignition. Each altimeter is armed sequentially to distinguish the unique beep sequences, confirming proper arming of individual devices. Once both altimeters are armed and the tracking device has acquired a signal, the system is ready for launch. If any anomalies occur during the activation of recovery electronics, launch operations are paused until full verification of all avionics functionality is established.

3.10 Mission Performance Predictions

The target apogee is tentatively 4600 (ft) With a redesign of the Air Brakes system from the prior year, the launch vehicle can be subjected to further altitude reduction with the increase in surface area of each fin of the Air Brake's assembly. These details are further outlined in section 5. This altitude is derived through countless calculations and simulations. The methodology is highlighted in figure 3.46

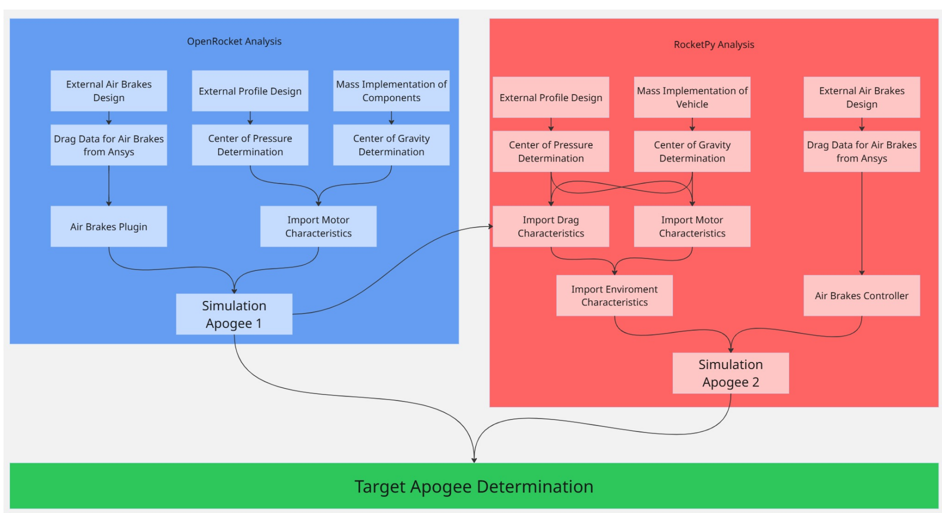


Figure 3.46: Calculation and Simulation Methodology

3.10.1 Preliminary Apogee and Trajectory Simulation

The apogee simulated by both OpenRocket and RocketPy has the launch vehicle reaching an altitude of 5435 (ft) The simulation details and settings are highlight in sections 3.10.3 and 3.10.4. Both of these software utilize psudo-6DOF, Runge-Kutta-4 simulation strategies to simulate the flight trajectory of the launch vehicle. Both offer the same level of control over most, if not all, parts of a given configuration but go about it in starkly different ways. OpenRocket is a graphical user interface (GUI) based software, with a database of many components and items to build any vehicle desired. It also provides user customization through menus to provide and override critical values to provide an accurate simulation. RocketPy on the other hand is a Python Library utilizing classes and objects to define various components of a configuration. Within these objects, values are specified by the user with floating point numbers which allows for a slight boost in accuracy of the simulation.

RocketPy has one distinct advantage over OpenRocket in that it natively supports a custom controller function for the simulation of the Air Brakes system. This was used in the prior year to garner estimates for the altitude reduction that the system can produce. There are some issues with its implementation that conflicts with the system's controller logic, so another route was explored with OpenRocket. OpenRocket allows for the creation of plugins to modify any behavior in the software, including in flight simulation. An idea to create a custom plugin that simulates our custom apogee prediction and control scheme was explored and flushed out. Figure 3.47 depicts the overall methodology for how RocketPy is used.

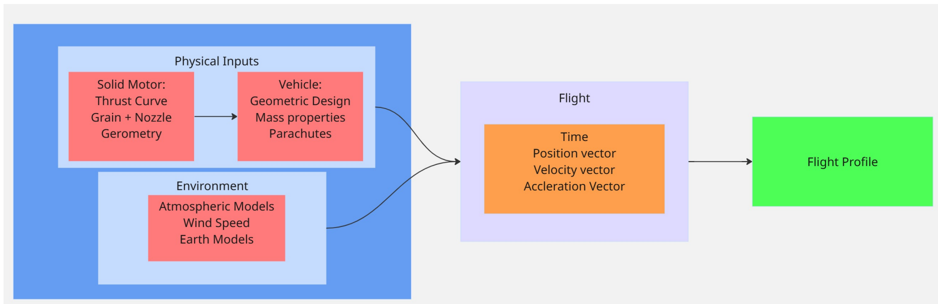


Figure 3.47: RocketPy Simulation Methodology Flow Chat

Overall, both of these software have their pros and cons for use. Therefore both will be used in conjunction to provide estimates for apogee and any other quantity in regards to the launch configuration. Moreover, both methods for simulating the Air Brakes system will be employed to garner an accurate measurement of altitude reduction.

3.10.2 Fin Performance and Alternate Designs

The fins were designed with drag reduction in mind to ensure the launch vehicle will reach the target apogee. As mentioned in section 3.20 the fins will be constructed out of a balsa core and carbon fiber weave epoxied over the core for a lightweight and structurally rigid design. The fins will be 4 in count, to ensure balancing of aerodynamic forces over the vehicle. These will shift the aerodynamic center of pressure, CP, aft of the center of gravity, CG, to ensure a minimum stability of 2.0 on the launch rail, meeting NASA requirement 2.11. The fins were run through computational fluid dynamics (CFD) simulations using ANSYS Fluent to assess their performance. The solver settings are outlined in table 3.20, with a concentration to ensure capturing of the boundary layer that forms over the fin body.

Table 3.20: ANSYS Fluent Solver Settings

Parameter	Option
Mesh Cell Size	9 - 13 million Cells
Turbulence Model	2 equation $\kappa - \omega$ SST
Solver	Steady State Pressure Based

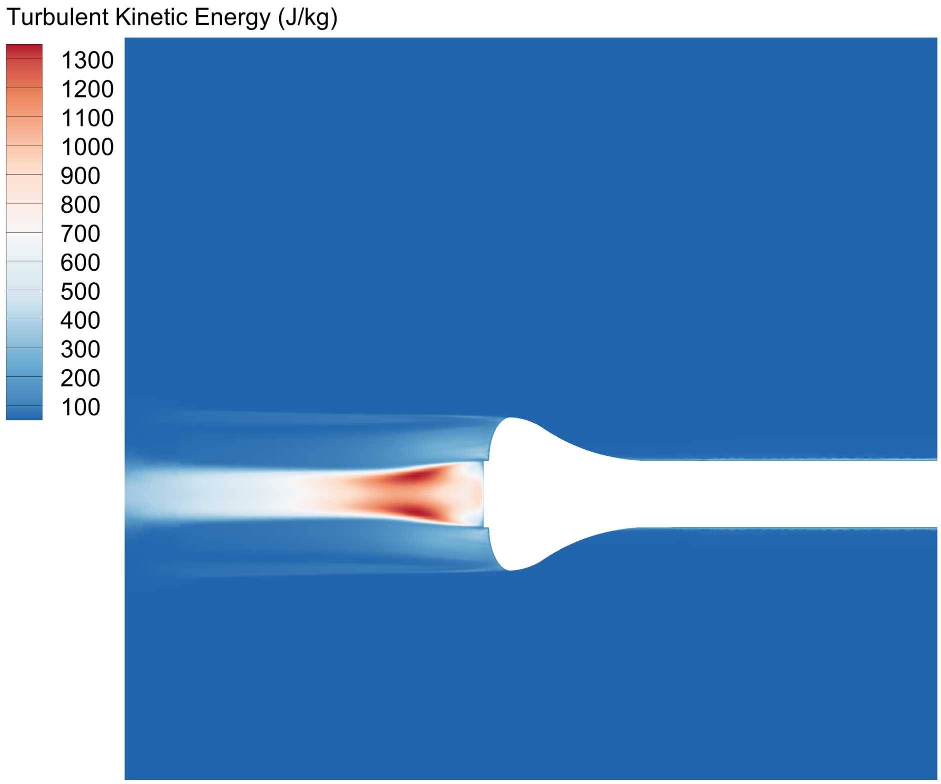


Figure 3.48: Elliptical Delta Fin Turbulence Contour

As shown in figure 3.48 is an Elliptical Delta fin. This design combines the aerodynamic benefits of an elliptical fin where fin tip vortices are reduced therefore reducing overall drag, and resistance to fin flutter is increased. The delta leading edge aids to the stiffness of the fin thereby counteracting the weakness of elliptical fins. The drag force around each fin is 2.248 (lbf) at 619 (fps). One downside is turbulence around the fin is greater along the aft end of the fin. This design was considered for its aerodynamic benefits and new methodology of manufacturing. Each of the fins along with the fin can were going to be 3-D printed into a mold and then wrapped in carbon fiber weave and epoxied together creating a molded fin can. The Senior Design team attempted to aid the structures lead in prototyping but the complex process did not yield results of satisfactory quality for a 6 (in) diameter vehicle. Furthermore due to the complex curves, there would uneven beveling along the leading and trailing edge of each fin leading to induced roll during flight. For those reasons the Senior Design deemed the design unfit for further consideration.

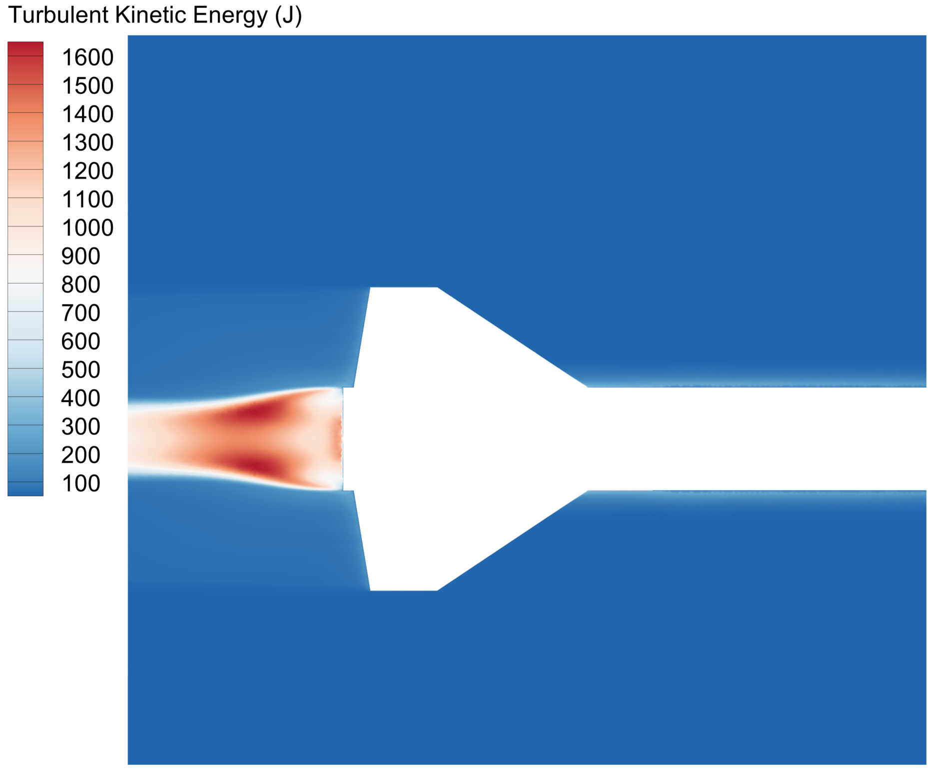


Figure 3.49: Swept Trapezoidal Fin Turbulence Contour

As shown in figure 3.49 the leading design for the fins is a swept trapezoidal fin. Its tip chord and root chord are 4 (in) and 14 (in) respectively. Further dimensions are found in figure 3.20. This design ensures ease of manufacturing compared to the elliptical fin in figure 3.48. The trapezoidal fin allows for a similar turbulence across the trailing edge leading to an efficient design. The drag force around each fin is 5.845 at 619 $\frac{ft}{s}$ (lbf), which is over two times greater than the elliptical delta fin. Standard designs such as a swept delta fin and a symmetrical trapezoidal fin were considered but simulations show these fins will not work. Both fin designs decreased the stability of the launch vehicle and overshot the flight ceiling of 6000 (ft) when updated masses were added into the simulation. Further more, the current fin design allows for the Air Brakes System to take off apogee without impedance on performance or introduce adverse moments on the vehicle. The design moving forward for the Full Scale launch vehicle is depicted in figure 3.49.

3.10.3 Wind Effect On Apogee

Shown in figure 3.28, the Apogee based on a crosswind of 5 mph from OpenRocket is 5435 (ft) To account for a variety of wind conditions on launch day, a sweep of simulations at increasing wind speeds were run. This provides data for a ballast adjustment on launch day if wind speeds are greater than expected.

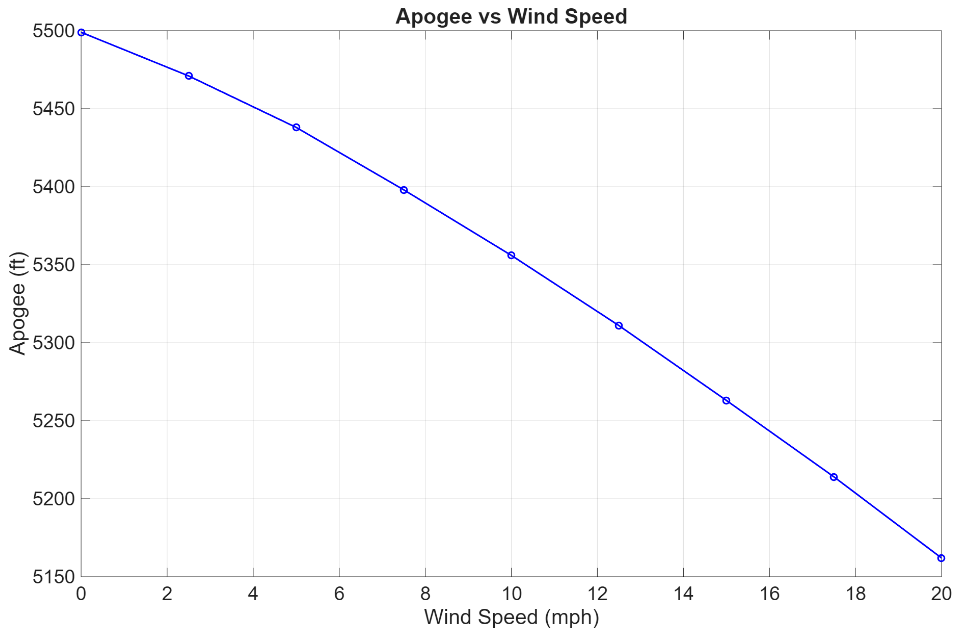


Figure 3.50: Apogee Vs Wind Speed

Figure 3.50 depicts the apogee decrease starting from a wind speed of 0 mph, to the greatest wind speed allotted to launch in at 20 mph.

3.10.4 Stability Margin

Another avenue to consider is the probable weight of the payload. Due to early design considerations the weight of the payload may be subject to change depending on a variety of factors. Another sweep of simulations was done with a set wind speed of 5 mph, with the payload weighing nothing to a maximum of 10 (lbs).

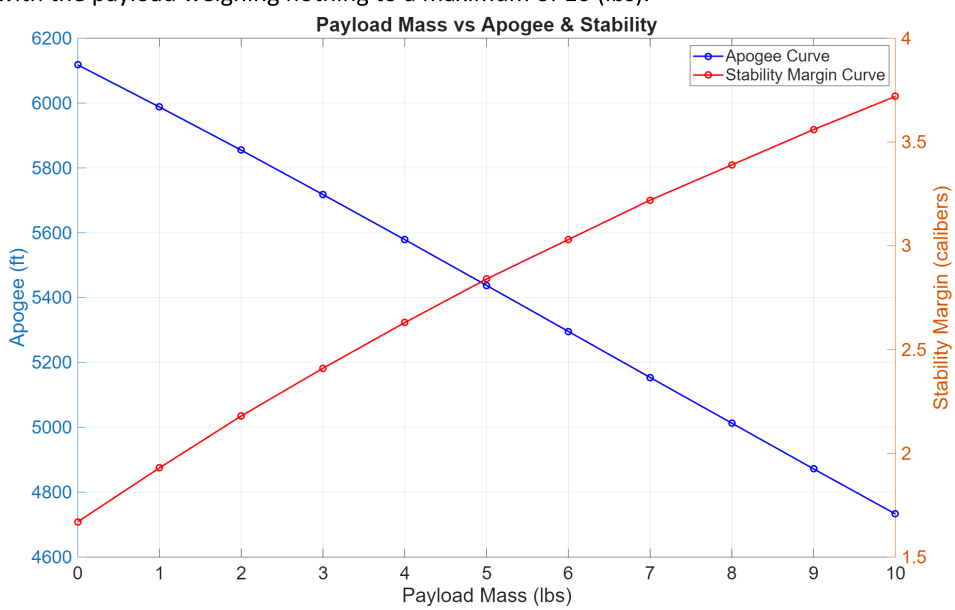


Figure 3.51: Payload mass vs Apogee and Impact on Stability

Figure 3.51 depicts how the apogee and on the launch rail stability will be affected by the change in payload weight. The apogee follows a near linear decrease from a max at 6118 (ft) to 4733 (ft), while the stability follows an increase from a min stability of 1.67 cal to a maximum of 3.72 cal. Based on the intersection point a near ideal payload weight will be 5 lbs which will ensure an apogee of 5435 ft and a stability of 2.84 calibers but there will be margin for the payload weight to increase without affecting the target apogee.

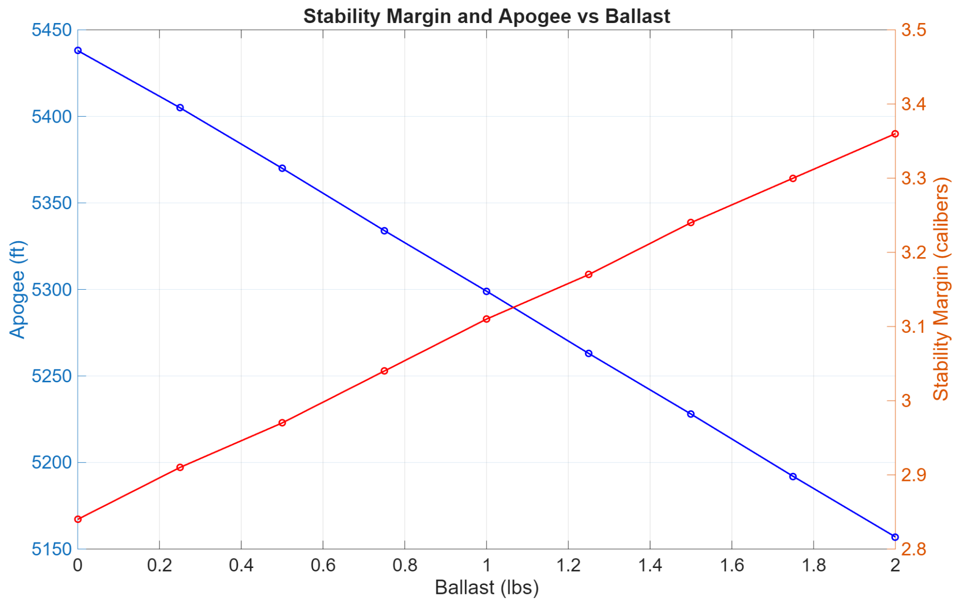


Figure 3.52: Ballast Impact on Apogee

Furthermore, adding ballast will reduce apogee and increase stability. As shown in figure 3.52 apogee linearly decreases from 5435 (ft) down to 5157 (ft) while stability increases from 2.87 cal. to 3.36 cal. For the given configuration and current trajectory analysis there is 0 lbs of ballast.

Flight Configuration

Below figures 3.53 and 3.54 depict the current flight configuration for the launch vehicle. In figure 3.53 depicts all parts currently in the vehicle from the general body and fin shape down to the attachment location of each shock cord. Whereas in 3.54 gives a general shape, length, and motor location of the vehicle.

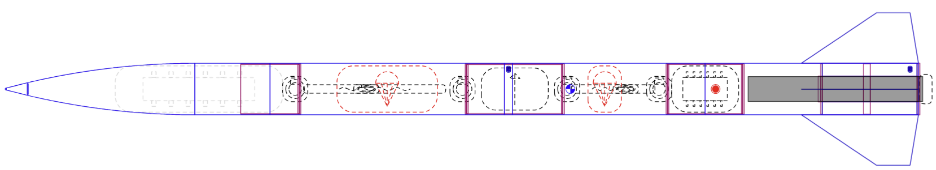


Figure 3.53: Open Rocket Flight Configuration

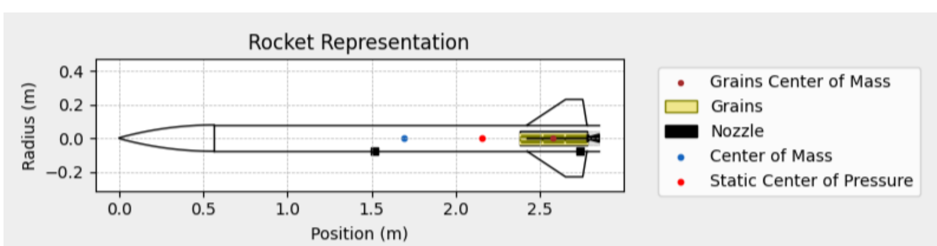


Figure 3.54: RocketPy Flight Configuration

Both simulation software depict and label the CP and CG of the rocket. The values are tabulated in table 3.21.

Table 3.21: Center of Pressure, Center of Gravity, and Stability Margin by Software

Software	Center of Pressure	Center of Gravity	Stability Margin
OpenRocket	84.901 (in)	67.54 (in)	2.84 Calibers
RocketPy	84.803 (in)	66.693 (in)	2.934 Calibers

Verify Stability Calculations

To verify the calculations done via each of the software, the CP, CG locations, and stability can be analytically calculated from the Barrowman equations listed below.

$$C_{N_f} = \left[1 + \frac{R}{S+R}\right]\left[\frac{4\,N\left(\frac{S}{d}\right)^2}{1+\sqrt{1+\left(\frac{2L_f}{C_R+C_T}\right)^2}}\right] \tag{24}$$

$$X_f = X_B + \frac{X_R(C_R+2C_T)}{3\,(C_R+C_T)} + \frac{1}{6}\left[(C_R+C_T) - \frac{C_R C_T}{C_R+C_T}\right] \tag{25}$$

$$X_{CP} = \frac{C_N X_N + C_F X_F}{C_N + C_F} \tag{26}$$

$$SM = \frac{X_{CP} - X_{CG}}{2R} \tag{27}$$

Using equations above, in conjunction with estimates from OpenRocket and measurements, table 3.22 tabulates all values and results.

Table 3.22: Constants and Barrowman Results

Constant	Variable Name	Value	Units
$(C_N)_N$	nosecone Coefficient	2	NA
X_N	nosecone Length Factor	10.555	(in)
R	Body Radius	3.085	(in)
S	Fin Span	6	(in)
N	Number of Fins	4	NA
d	Base of Nose Diameter	6.12	(in)
L_F	Fin Midchord Line Length	9	(in)
C_R	Fin Root Chord Length	14	(in)
C_T	Fin Tip Chord Length	4	(in)
X_B	Nose to Root Chord LE length	95	(in)
X_R	Tail to Root Chord LE length	10.817	(in)
X_{CG}	Center of Gravity (OpenRocket)	67.54	(in)
Equation	**Result**	**Units**	
$(C_N)_f$	8.521545671	NA	
X_f	101.883	in.	
X_{CP}	84.527	in.	
SM	2.776	Calibers	

3.10.5 Flight Profile Simulations

Figure 3.55 shows the launch vehicle's flight profile with vertical velocity, vertical acceleration and altitude with respect to time of flight. Vertical lines are plotted highlighting key events over the vehicle's trajectory with motor burnout, apogee, and deployment of recovery devices.

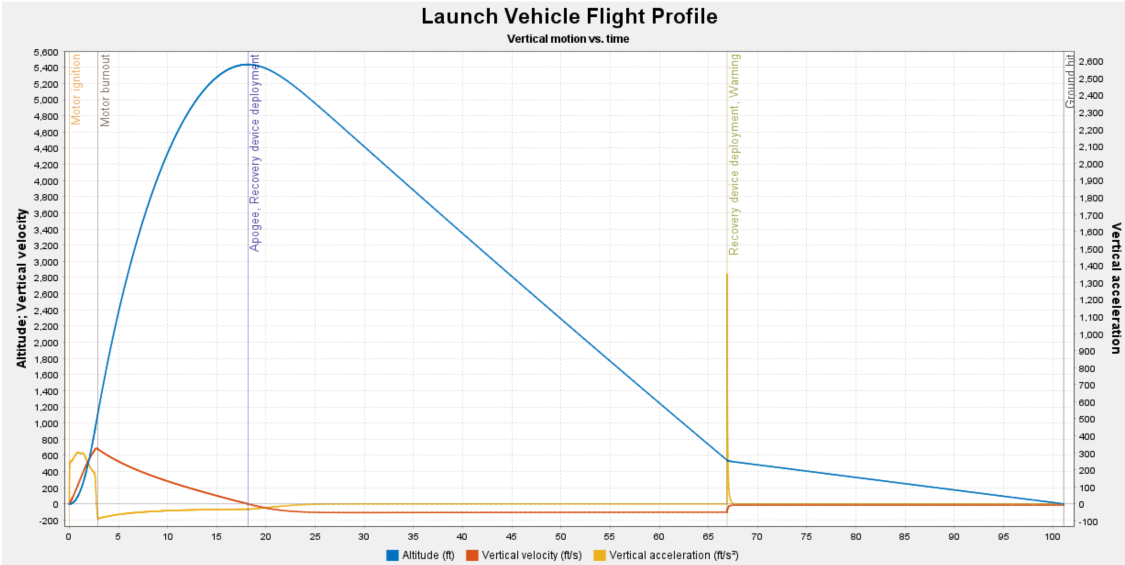


Figure 3.55: Open Rocket Flight Profile

Furthermore, figure 3.56 depicts the 3-Dimensional flight profile using RocketPy. RocketPy easily depicts the flight trajectory with wind drift in 3-D space. This capability will be used further once the vehicle configuration and simulation reaches a CDR maturity.

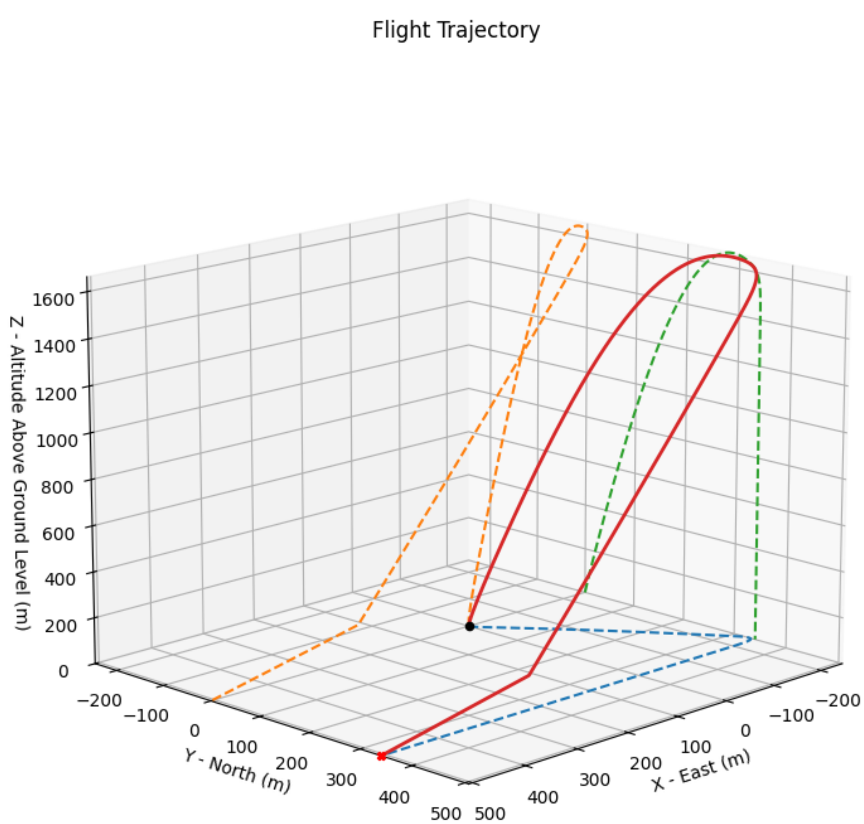


Figure 3.56: 3-dimensional Flight Profile

Verify Calculations

As like the Barrowman equations, the Fehskens-Malewicki equations can be used to analytically validate the apogee reported by both OpenRocket and RocketPy. The equations are listed below. For clarification T is approximated to be the average thrust of the motor, M is the mass of the Launch Vehicleduring liftoff, and C_d is derived from CFD simulations.

The drag force is expressed by the constant K:

$$k = \frac{1}{2}\,\rho\, C_D\, A \tag{28}$$

The relationship between the thrust, drag, and gravity, can be expressed as q:

$$q = \sqrt{\frac{T - Mg}{k}} \tag{29}$$

The relationship the drag force and q per unit mass, is expressed as:

$$x = \frac{2kq}{M} \tag{30}$$

By using Equations 29 and 30, the maximum velocity of the launch vehicle is found through:

$$v_{\text{max}} = q\,\frac{1 - e^{-xt}}{1 + e^{-xt}} \tag{31}$$

At motor burnout drag force and gravity become the main forces acting on the rocket. The altitude at motor burnout can be computed via:

$$Z_{\text{burnout}} = -\frac{M}{2k}\ \ln\left(\frac{T - Mg - kv_{\text{max}}^2}{T - Mg}\right) \tag{32}$$

The total coast distance of the launch vehicle after burnout is calculated with:

$$Z_{\text{coast}} = \frac{M}{2k}\ \ln\left(\frac{Mg + kv_{\text{max}}^2}{Mg}\right) \tag{33}$$

Lastly, the total apogee reached is the sum of equations 32 and 33.

$$Z_{\text{apogee}} = Z_{\text{burnout}} + Z_{\text{coast}} \tag{34}$$

The values, measurements, and results from equations 28 to 34 are depicted in table 3.23

Table 3.23: Constants and Results

Constant	Variable Name	Value	Units
M	Power On Average Mass	1.1169	Slug
m	Power Off Average Mass	0.8545	Slug
g	Gravitational Acceleration	32.1719	ft/s^2
t	Motor Burn Time	2.6	s
T	Average Thrust	312.4844	lbf
ρ	Air Density	0.0764	slug/ft^3
A	Launch Vehicle Frontal Area	0.1963	ft^2
C_d	Drag Coefficient	0.2908	NA
Equation	**Result**	**Units**	
k	0.000730	slug/ft	
q	6623.7386	ft^2/s^2	
x	0.8047	ft/s^2	
v_{max}	622.8066	ft/s	
Z_{burnout}	823.0962	ft	
Z_{coast}	4522.9743	ft	
Z_{apogee}	5346.0705	ft	

3.10.6 Drag During Flight

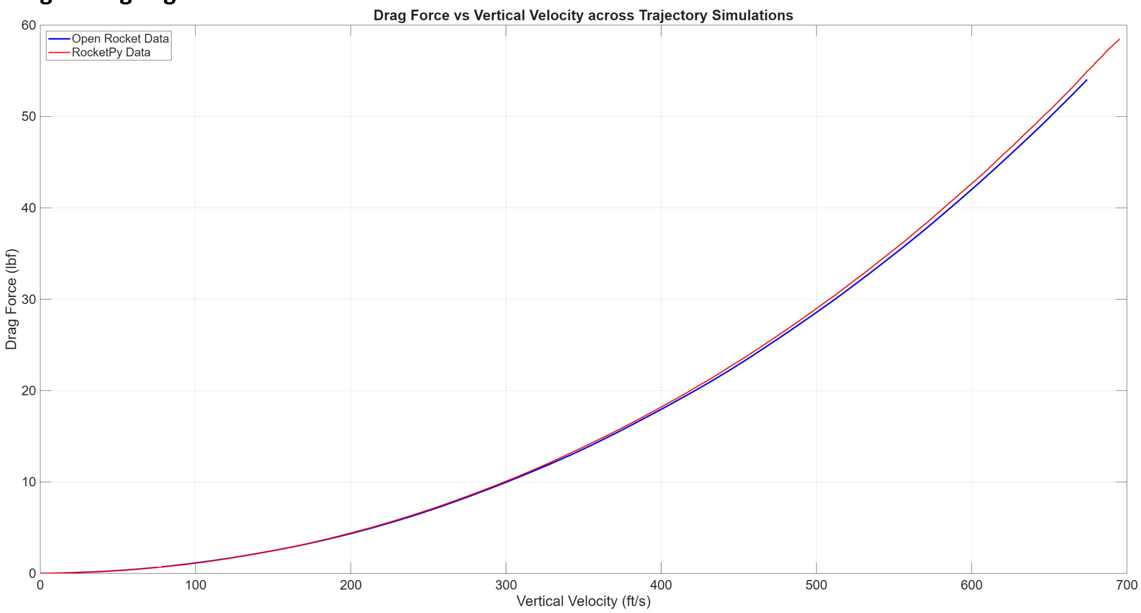


Figure 3.57: Drag curve During Ascent Without Air Brakes

As shown above in figure 3.57 is a curve of the drag force on the vehicle during its ascent phase, without Air Brakes, between simulation software. Both OpenRocket and RocketyPy agree within a 0.293 % error of each other. This ensures that the flight profiles between both software are comparable.

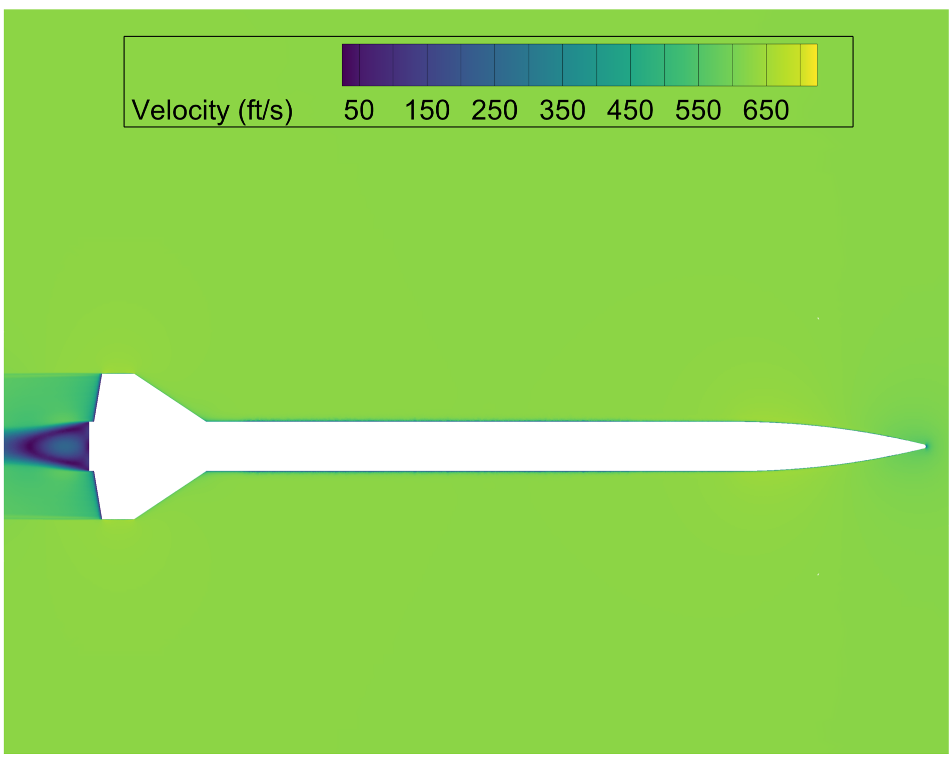


Figure 3.58: Drag Contour Over Launch VehicleThrough Fluent

Shown above in figure 3.58 is a velocity contour profile of the launch vehicle during coast phase at 619 (fps). This further illustrates the low turbulence near the trailing edge of the fin along with a boundary layer being captured all across the vehicle's surface. Furthermore, this ensures CFD simulations calculating a precise estimate of drag force which aids in the simulation of Air Brakes' effect on the flight trajectory of the rocket. Further discussion is done in the next section.

From all analysis in from the prior sections, the simulation settings to be used going forward are tabulated in table 3.24.

Table 3.24: Trajectory Simulation Options

Simulation Parameter	Value	Reasoning
Wind Speed	5–10 mph	Enveloping profile to allow informed ballast adjustments
Launch Rail Cant	5 deg.	NASA rule 1.9
Launch Rail Length	12 ft	NASA rule 1.9

3.10.7 Air Brakes Effect

With the Air Brakes System, its fins will be exposed a maximum wind speed of 675 $\frac{ft}{s}$. This allows the Air Brakes Fins' to induce a greater amount of Drag Force and flight control to the launch vehicle but will need to be rigid. Figure 3.59 shows the Drag values over the entire Air Brakes assembly at either fully retracted or fully deployed. All numbers in the figure were calculated via ANSYS Fluent, and to further ensure accuracy, each data point was tested at the altitude where the velocity would occur during the flight profile found from OpenRocket and RocketPy.

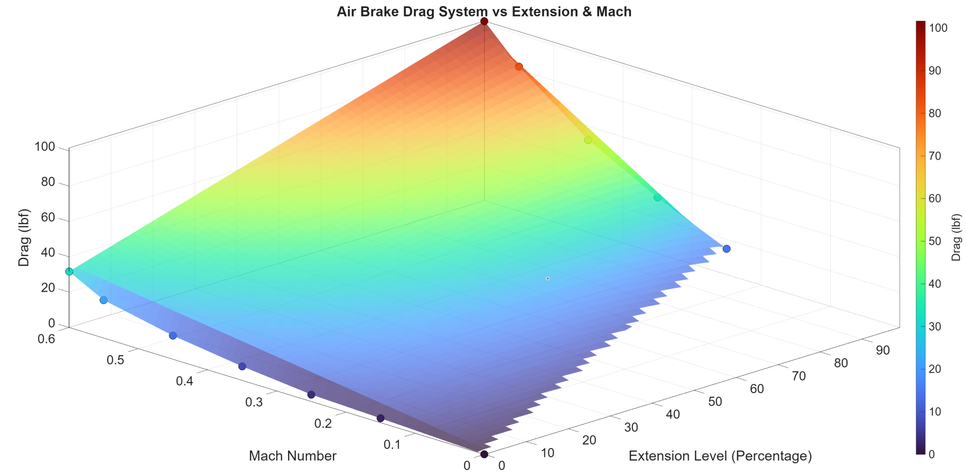


Figure 3.59: 3-dimensional Contour of Drag generated by Air Brakes Fins

Over the system, the maximum drag force experienced by the Air Brakes is 101 (lbf) with the values decreasing as the mach number lowers and the extension level lowers to a minimum of 14.629 (lbf) fully extended. This directly increases the total drag experienced the Launch Vehicleallowing a large apogee reduction. Construction methods are detailed in section 5.3.1 to ensure each fin will be able to withstand fin flutter and meet team derived requirement LVD 8.

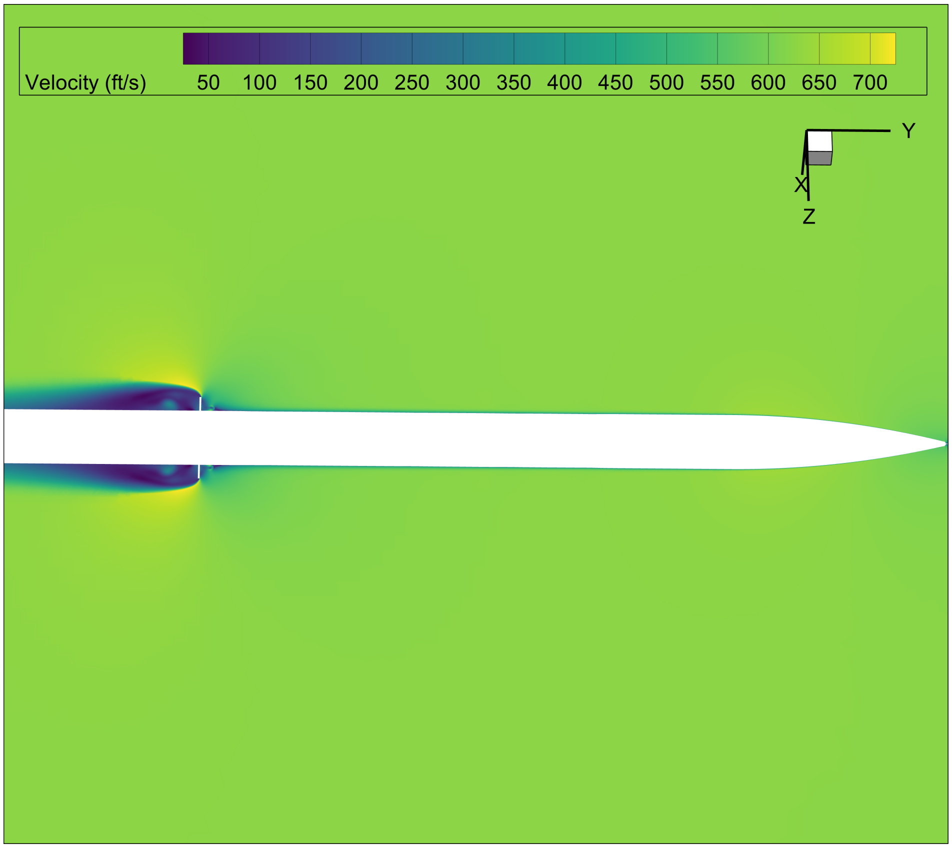


Figure 3.60: Air Brake Deployed Velocity Contour

Shown about in figure 3.60 is a velocity contour of the launch vehicle with Air Brakes fully deployed at 619 $\frac{ft}{s}$. Air flow on the rocket is streamline until hitting the fins, forcing the flow to go around and continue aft. A recirculation zone forms behind each fin illustrating fast moving air keeping air flow that is close to the body trapped, causing a pressure build up and inducing further drag on the vehicle. Furthermore, flow is accelerated near the tips of each fin as a by product of transonic effects. Seeing these phenomenon in the CFD velocity contour, validates the accuracy of the simulation and reliability of results.

Air Brakes Plugin for Open Rocket

To determine the altitude reduction that the Air Brakes assembly can afford the launch vehicle, analysis with Air Brakes deployment must be used. OpenRocket supports the use of custom code to influence any part of the software via a Plugin implementation using Java as the code language. Therefore, custom software was written to interface with OpenRocket's simulation and trajectory analysis. Since there is full control over the software as opposed to RocketPy, the apogee predictor algorithm was translated to Java code and serves as the heart of the software. Further details of the algorithm can be found in section 5.3.5. Figure 3.61 shows the flow diagram for the plugin.

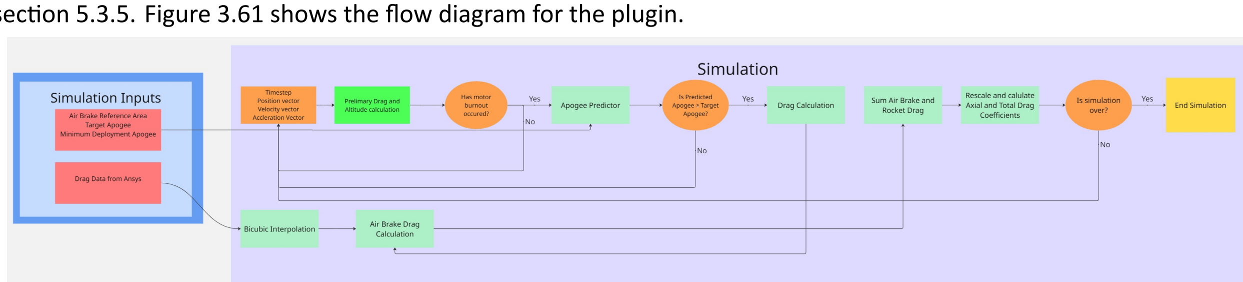


Figure 3.61: Flow Diagram of OpenRocket Plugin for Air Brakes

As shown above, inputs taken in are CFD drag data as a .csv file, total fin area, target apogee and minimum deployment apogee. For the drag, a bi-cubic interpolation is performed on the drag data so that for any given extension level and mach number a drag value can be calculated. For the other inputs, OpenRocket first performs its calculations assuming only the simulation parameters of the vehicle configuration without air brakes for each timestep. Then control logic is run to check if motor burnout has occurred while the apogee predictor is slowly gathering data for a prediction. If motor burnout has not occurred the software lets OpenRocket run the simulation without any modification. Once motor burn out has occurred, the apogee predictor will start running, and outputting a peak altitude. If the control logic detects that the peak altitude is less than the target apogee, the software will simulate air brakes deploying. This is done by calling a method to calculate the drag over Air Brakes and the extracting the drag over the vehicle for its current velocity. This is done as well for the axial drag. Then the drags are added up, and a new drag coefficient is calculated and rescaled by the new reference area to override the simulation. These equations are detailed in equations 35 and 36 for total drag and axial drag respectively.

$$C_d = \frac{2\,F_{\text{total}}}{\rho\,V^2\left(A_{\text{ref,rocket}} + A_{\text{ref,airbrakes}}\right)}. \tag{35}$$

$$C_{d,\text{axial}} = \frac{2\,F_{\text{axial}}}{\rho\,V^2\left(A_{\text{ref,rocket}} + A_{\text{ref,airbrakes}}\right)}. \tag{36}$$

These new drag coefficients are then pass through to the next timestep where the cycle is repeated until the software declares apogee. This method is flushed out for the current Air Brakes system but is subject to change with new code additions and simulation refinements. Looking forward, figure 3.62 shows a curve varying target apogee with a fixed minimum Deployment Height of 1500 (ft)

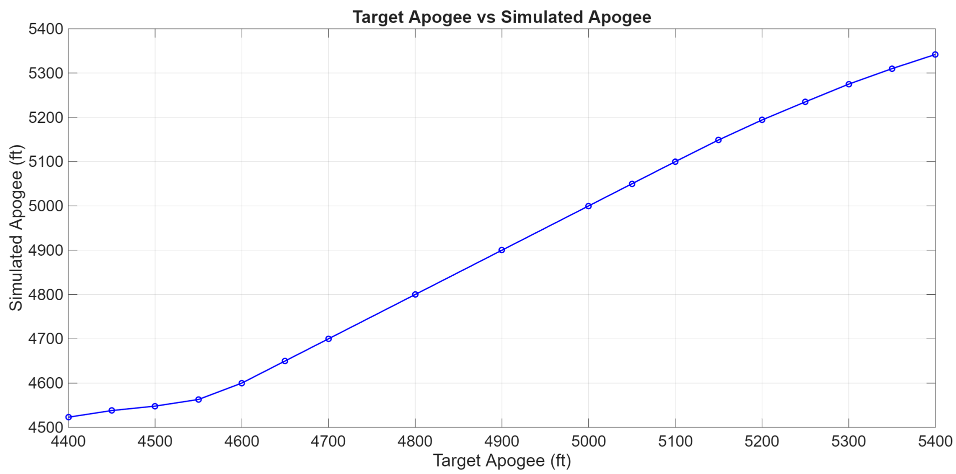


Figure 3.62: Air Brakes Target Apogee vs Simulated Apogee

The minimum deployment height was chosen to ensure the apogee prediction algorithm has ample time to derive a flight profile for the simulation. As shown in figure 3.62, any target apogee under 4600 (ft) and above 5200 (ft), shows over predictions and under predictions respectively of the apogee prediction. This is mainly due to the minimum deployment height were there are over predictions for the lower target apogees with drag being overestimated, and vice versa with drag underestimated at higher target altitudes. Between those two, the reported simulated apogee is virtually identical to the target apogee which bodes strongly for the implementation of the apogee prediction algorithm.

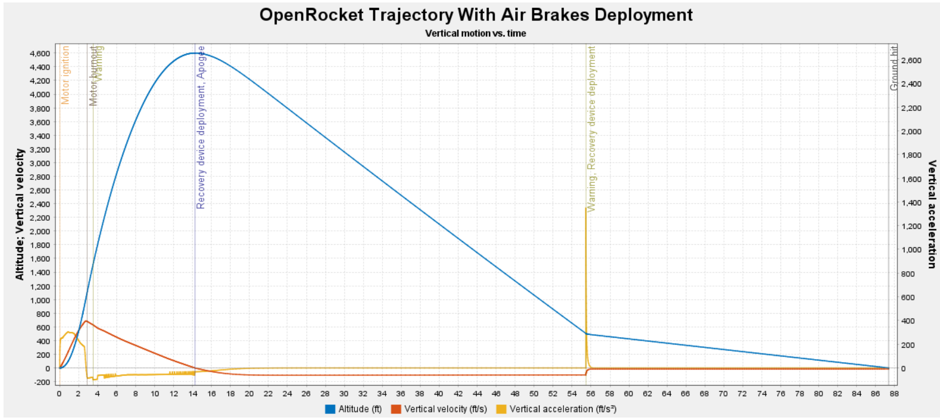


Figure 3.63: OpenRocket Flight Profile with Air Brakes

Further shown in figure 3.63, is the flight entire flight trajectory with Air Brakes deployed with a target apogee of 4600 (ft) The target apogee selected for the launch vehicle is within this band at 4600 (ft)

3.10.8 Kinetic Energy

The predicted descent velocity of the vehicle was calculated using the following formula,

$$v_{desc} = \sqrt{\frac{2mg}{\rho A C_d}} \tag{37}$$

where v_{desc} is the descent velocity, m is the mass of the vehicle, g is gravitational acceleration, ρ is air density, A is the projected area of the parachute, and C_d is the drag coefficient of the parachute.

The selected drogue parachute is a 18 (in) elliptical parachute. This parachute has a projected area of 1.77 (ft^2) and a C_d of 0.75. Given that the dry mass of the launch vehicle is 31.63 (lb), the descent rate under the drogue parachute is calculated to be 100.15 (fps). The main parachute chosen for the vehicle is a 96 (in) toroidal parachute, which has a projected area of 48.71 (ft^2) and a C_d of 2.2. This will result in a main descent rate of 15.76 (fps).

After obtaining the launch vehicle descent velocity, the kinetic energy of the heaviest section can be calculated using the following equation,

$$K = \frac{1}{2} m v_m^2 \tag{38}$$

where K is the maximum kinetic energy, m is the mass of the heaviest section, and v_m is the descent velocity under the main parachute. Using a descent velocity of 15.76 (fps) and a mass of 14.51 lb, the maximum kinetic energy of the launch vehicle is calculated to be 55.99 ft-lbf, satisfying NASA requirement 3.2.

3.10.9 Descent Time

The total descent time of the launch vehicle is calculated using Equation 39,

$$t_d = \frac{r_a - r_m}{v_d} + \frac{r_m}{v_m} \tag{39}$$

where t_d is descent time, r_a is the altitude of apogee, r_m is the main parachute deployment altitude, v_d is the drogue descent velocity, and v_m is the main descent velocity. Assuming the drogue deployment occurs at an apogee of 4600 ((ft)) and the main deployment occurs at 550 ft, this results in a descent time of 75.33 seconds, which satisfies NASA Requirement 3.11.

3.10.10 Drift

Using the total descent time and wind speed, the drift distance of the launch vehicle can be calculated using Equation 40.

$$d_{drift} = v_w t_d \tag{40}$$

Table 3.25 presents the variation in drift distance of the launch vehicle with wind speeds ranging from 0 mph to 20 mph. The analysis assumes the vehicle reaches apogee directly above the launch pad and drifts at a constant rate with the wind. As these calculations are based on a worst-case scenario, specifically, constant wind conditions, the resulting values represent an upper bound on drift distance, satisfying NASA Requirement 3.10.

Table 3.25: Wind Drift Distance

Wind Speed (mph)	Drift Distance (ft)
0	0
5	552.43
10	1104.86
15	1657.28
20	2209.71

4 Payload

4.1 Payload Objectives

The objective of the Habitat for Agricultural Utilization Study, or HAUS, is to secure the STEMnauts and to collect soil measurements. The HAUS must have an atmosphere-isolated compartment that retains the STEMnauts for the entirety of the flight. Additionally, the HAUS will collect and retain a 50 (mL) sample of soil within 15 minutes of landing. This soil will be tested for its pH level, electrical conductivity, and Nitrate-Nitrogen content.

The soil sample will be collected using an auger drill. To ensure this drill is pointed at the ground, a portion of the payload will deploy from the Launch Vehicle after landing and self-right using deploying legs. The soil collected by the auger will be deposited into a container within the payload that contains the soil sampler. The soil sensor will begin taking readings after a set time period and will record these readings using a Raspberry Pi.

4.2 Payload Success Criteria

Table 4.1: 2025-2026 Payload Requirements

Success Level	Payload Aspect	Safety Aspect
Complete Success	> 50 ml of soil is collected AND soil sensor data is retained.	No individuals are harmed during payload operations, and all risks are mitigated.
Partial Success	< 50 ml of soil is collected AND soil sensor data is retained.	No individuals are harmed during payload operations, but some risks are unmitigated.
Partial Failure	< 50 ml of soil is collected, but soil sensor data is not retained, OR > 50ml of soil is collected, but no soil sensor data is retained	Individual(s) receive(s) minor harm by unmitigated risk in payload operations.
Total Failure	No soil is collected, and no soil sensor data is retained	Individual(s) receive(s) major harm by unmitigated risks in payload operations.

4.3 Potential Payload Designs

4.3.1 Primary Structure

Rotating Coupler Section

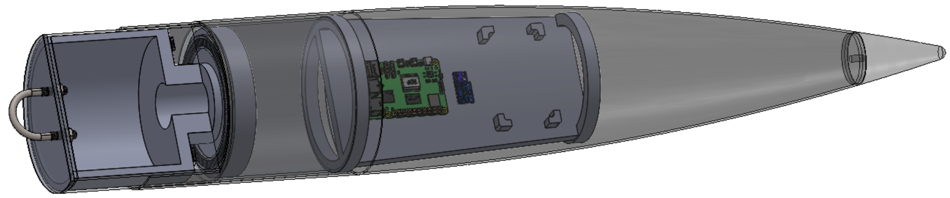


Figure 4.1: Rotating coupler section design.

This design will consist of a section at the end of the nosecone that rotates to orient a drill downwards. The payload structure will be mounted to the inside of the nosecone coupler. Part of the payload will remain inside the nosecone structure, but part will protrude beyond the coupler. While in flight, the whole payload will be contained inside the Launch Vehicle body. When the Launch Vehicle separates for parachute deployment, part of the payload will be exposed. Using the Launch Vehicle separation to reveal part of the payload removes the need for a separate mechanism for payload deployment.

Once the Launch Vehicle lands, the payload will begin the process of soil collection. The part of the payload exposed to the air will be mounted on a set of thrust bearings that allow it to rotate around the axis running the length of the Launch Vehicle . This rotation will be achieved using an electric motor mounted farther forward in the nosecone. An auger drill mechanism inside the payload will be deployed once in the right orientation. An accelerometer will measure the direction of gravity and assign that direction ”down.” The same accelerometer will then measure the current orientation of the payload. This orientation will be measured along the length of the drill. A comparison will be made between the current and desired orientations, converted into motor instructions, and sent to the motor to rotate the payload to this desired orientation.

Now that the drill is pointed towards the ground, it will be deployed. A rack and pinion mechanism will extend the drill out of a hole in the payload structure. Another motor will rotate the drill as it is being extended to best enter the soil. Once the drill has been extended to its full length, the rack and pinion will retract the auger with soil still resting on it. This soil will then be scraped into a canister by a fixed brush inside the payload. The canister will be built around the soil sensor so that the collected soil falls on top of the sensor prongs. With this setup, the sensor does not have to be moved to test the soil, which reduces operational complexity.

The benefits of this design are the reduced complexity associated with the payload being able to remain within the structure of the Launch Vehicle , as well as the large space available for electronics by utilizing the nosecone’s entire volume. However, the drawbacks include the large bearings needed to rotate the payload which will increase weight and cost, as well as the length constraints on the soil collection drill due to its orientation sideways in the Launch Vehicle . In addition, the Launch Vehicle may get dragged along the ground by its main parachute after landing, and if the payload is still attached to the Launch Vehicle during its operation, this may result in damage to the drilling mechanism.

Self-Righting Lander

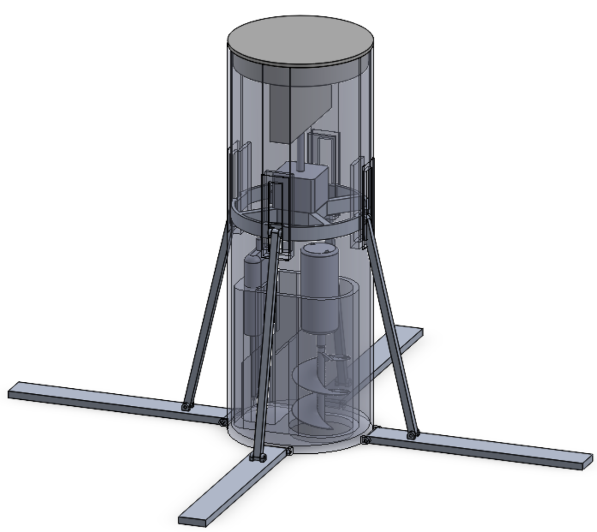


Figure 4.2: Self-righting lander design.

The cylindrical self-righting lander will use four automatically deploying legs to achieve a set orientation. Once in this upright state, a drill will be deployed through a hole in the bottom of the payload.

After landing, it is assumed that the payload will be on its side. The payload may land on either of its ends, but this is unlikely. If there is any roughness on the ground, the payload will fall onto one of its sides. The legs begin flush with the payload walls and will deploy simultaneously after landing is detected. As the legs unfold, at least one will remain in contact with the ground. The force applied will result in the payload lifting itself up. When the legs are fully deployed, the payload will be perpendicular to the ground.

Having a set orientation is beneficial for multiple reasons. With a self-righting design, a drill will only ever have to extend in one direction. Since that direction is known, there is no need for a mechanism to orient the drill. Additionally, the final orientation of the payload is very conducive to a drill device. A cylinder is very easy to lengthen along its center axis, which is the axis around which the drill is oriented. This means the drill can be as long as needed. Conversely, efforts will be made to keep the payload as short as possible. A taller payload results in a larger torque requirement for the legs, and therefore a stronger and more expensive motor requirement.

A circular drill will take up only part of the volume of the payload. The drill itself will take up the very center of the design, but the surrounding area is still available for use. At the bottom of the payload, this space will be used for soil sample containment and collection, as well as for leg hinging mechanisms. Further up, this space will be occupied by electronics.

While the Launch Vehicle is in flight, the payload will be held in place by a retention method. When the payload is ready to be deployed, a retention mechanism will release the payload, and it will be expelled from the Launch Vehicle . Possible designs for this retention method are discussed in Section 4.3.5. If the payload deploys on the ground, as described in Section 4.3.3, a set of tracks will be used to ensure the payload deploys with minimal friction. The mechanism that performs this deployment is discussed in Section 4.3.4.

The maximum diameter of the self-righting lander will be constrained by the Launch Vehicle and several mechanisms on board. The diameter on the inside of the Launch Vehicle body will be the maximum theoretical diameter of the payload, but that does not account for the deployment method and the legs. The tracks on the inside of the Launch Vehicle and their corresponding parts on the payload would decrease the maximum diameter of the payload. The legs will be installed between these tracks, but legs larger than the track system will consume even more space. Efforts will be made to keep the payload diameter at a maximum.

4.3.2 Leg Design

A deploying leg system is required for a self-righting lander payload structure. The payload structure will end up on its side, and these legs will right it. Any leg system will have four legs that fold out after the payload has left the Launch Vehicle . Three legs have too much angular distance between them to ensure that the payload will actually self-right when they deploy, and using five legs adds unnecessary mass. These legs will have to extend past the center of gravity of the payload when retracted in order to have the right application of force and avoid slipping.

Collar

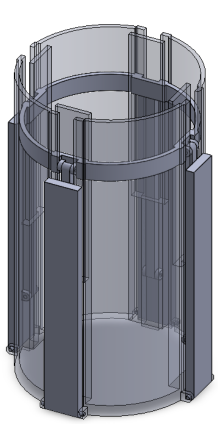


Figure 4.3: Collar system retracted.

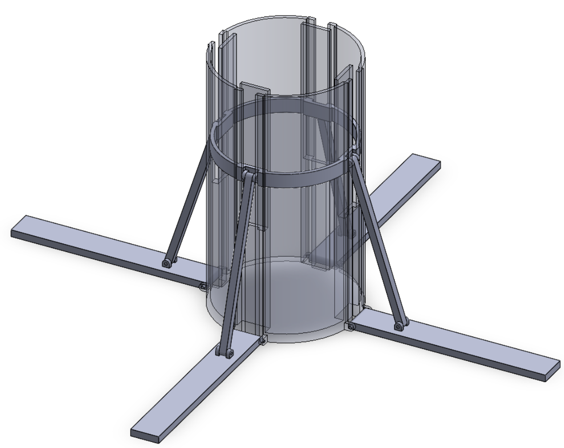


Figure 4.4: Collar system upright.

A collar system will see all four legs attached via linkages to a unified collar that translates along the payload. This collar will slide along tracks built into the side of the payload structure. A linear actuator or rack and pinion will be used to provide the linear motion of the collar.

Through the use of hinging struts, the vertical motion of the collar will be turned into rotational motion for the legs. The connection point of these struts determines the torque applied to the legs. Attachment points closer to the base transmit less torque but allow for a smaller strut length and less actuation distance. An attachment point farther out would transmit more torque but require a longer strut and farther actuation distance.

A main challenge with the collar design is keeping the collar level during actuation. If the payload is on its side resting on one of its legs, that leg will require significantly more torque to deploy than the rest. This in turn produces an uneven force on the collar, which may become unlevel, causing it to get jammed in its tracks and be unable to actuate further. This issue can be remedied by designing the collar such that it is not able to twist while deploying and giving the correct tolerances and geometries so that the likelihood of jamming is reduced.

Four Servos

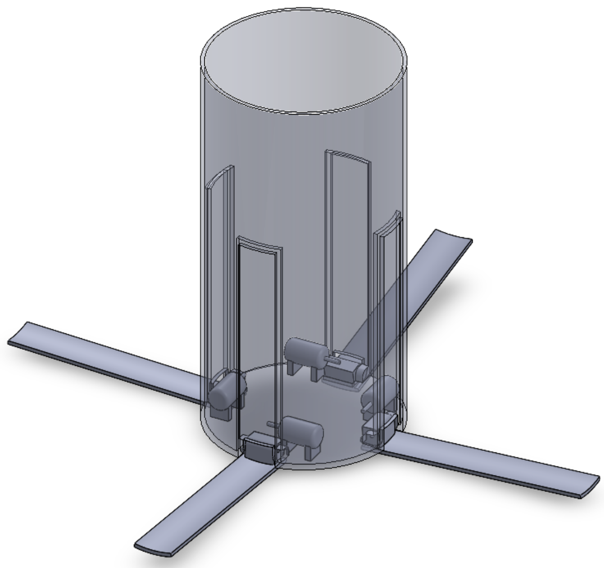


Figure 4.5: Four servo leg design.

In the four servo design, each leg will have its own servo that directly deploys the leg without an in-between mechanism. For the most direct torque transmission, these servos will be mounted to the bottom of the payload structure. The available space would only permit very small servos, such as 9-gram servos.

The legs will have gear teeth built into them that mesh with a gear attached to the servo. A shaft through the middle of the legs will connect them to the payload structure. The base of the legs will be slotted into holes cut out of the main payload structure. The corners on the base will be rounded to minimize the size of the hole needed. The size of the holes is minimized because larger holes create more space for dirt and debris to enter the payload. This has the potential to affect payload electronics and cause a failure.

Having all the servos at the bottom of the payload will create a lower center of gravity, decreasing the required leg length. Conversely, the servos at the bottom will reduce the usable space for the drill. Research also revealed that 9-gram servos are unable to meet the torque requirements of the payload. A 5 (lbm) payload that is 10 (in) tall would require 6 (in) long legs. 5 (lbf) applied at the end of 6 (in) legs requires 30 (ft-lbf) of torque. 9-gram servos can generate a maximum of 2 (ft-lbf) torque, making them unsuitable for this application.

Bevel Gears

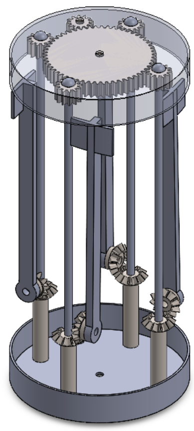


Figure 4.6: Bevel gear legs retracted.

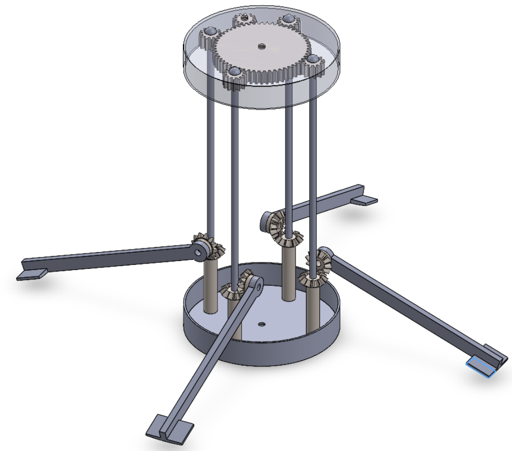


Figure 4.7: Bevel gear legs extended.

In order to keep mechanisms away from the drill area, a large spur gear will rotate four shafts. These shafts will run down the length of the payload and connect to bevel gears at the bottom. The bevel gears will, in turn, rotate the legs.

Using a gear system such as this is an effective way to distribute rotation to multiple outputs with only one input. A motor mounted at the top of the payload will provide input torque, which the central gear in the system will distribute. An output gear will be directly attached to a shaft that connects to a bevel gear towards the bottom of the payload. This bevel gear will mesh with another bevel gear connected directly to the leg.

In this design, the input and output gears are the same size, which provides no torque advantage. One rotation of the motor will create less than one rotation for the center gear, but when the central gear turns the output gear, the ratio will be reversed and the output gear will rotate once. Additionally, the bevel gears provide no torque benefit. Larger output and bevel gears were considered but dismissed due to more difficult fabrication and space constraints.

This design may be hard to assemble due to the challenges associated with mounting the bevel gears so that they mesh properly. Additionally, due to the torque requirements, the gears may skip when rotated instead of transmitting the torque to the next gear. The long shafts running from the top gears to the bottom gears may also be prone to warping due to their length when a torque is applied.

4.3.3 Deployment Condition

During Descent

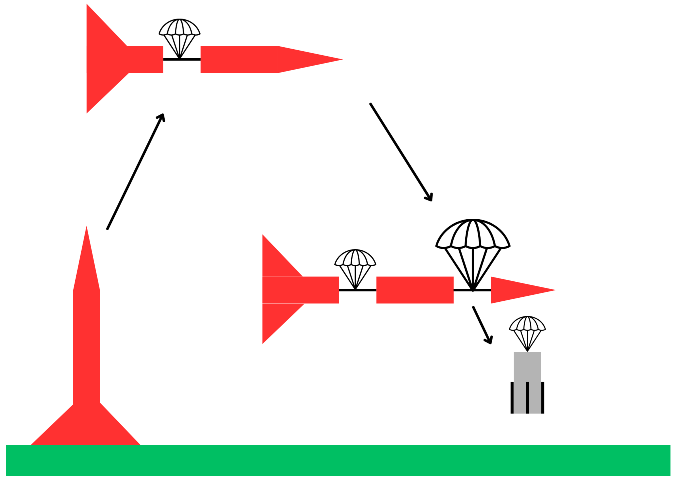


Figure 4.8: Deployment during descent.

Landers traditionally deploy during descent. In-air deployment involves releasing the lander after main parachute deployment. A latch (Section 4.3.5) will be used to retain the payload until it is dropped. The payload will then descend to the ground under its own parachute.

Deploying a payload under a parachute has been done by the club several times before. Additionally, parachutes are used every year to recover the Launch Vehicle . There is a large supply of knowledge surrounding parachutes that will be pulled for this design.

On the other hand, several aspects make air deployment difficult. The primary issue is one of orientation. During descent, the exposed end of the nosecone, where the payload will deploy from, is pointed up. This means gravity cannot be used to separate the lander and nosecone. A deployment mechanism, like those outlined in Section 4.3.4, will have to be used. A dedicated payload bay in another section of the Launch Vehicle will solve this issue, but for weight purposes, the payload must remain in the nosecone. The orientation of the nosecone will also be fixed regardless of the recovery method used for the Launch Vehicle .

Another issue arises once the payload has landed and deployed its legs. After landing, the parachute will still be attached to the lander. High winds will cause the parachute to re-inflate and knock over the payload. This will damage the drill and make the challenge much harder to complete. Several solutions were considered. Cutting the parachute free will completely solve this problem, but results in a loose parachute that is highly likely to be blown away by wind and lost. Reeling in the parachute will mitigate the risk of falling but add great mechanical complexity.

Landed

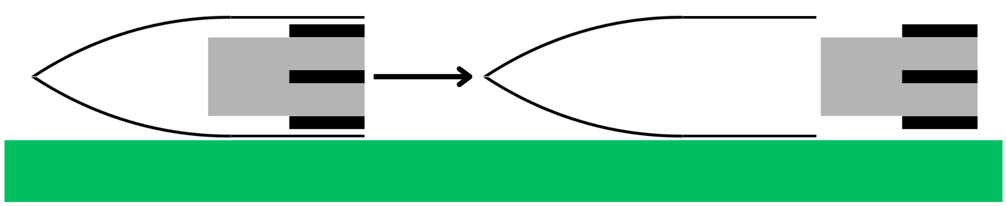


Figure 4.9: Deployment after landing.

The lander will deploy after the Launch Vehicle has landed. The state machine program running on the payload electronics will detect the landed state and activate a mechanism to deploy the lander (Section 4.3.4).

While the lander is in the Launch Vehicle , it will be held in place by its retention mechanism. After landing detection, it will be deployed out of the nosecone by its deployment system. After completing deployment and now that it is completely independent and separate from the Launch Vehicle , the lander will deploy its legs, self-right, and begin soil collection operations.

This design involves significant mechanical complexity. The legs, rails, and pushing mechanism will all have to be designed from scratch. These mechanisms will all add weight and failure points to the payload. However, on-ground deployment reduces the complexity of the Launch Vehicle and payload recovery scheme as a whole, and removes the need for RSO approval before deploying the payload.

4.3.4 Deployment Mechanism

A self-righting lander that deploys on the ground, as described in Section 4.3.3, requires a mechanism to push the lander out of the nosecone.

Lead Screw

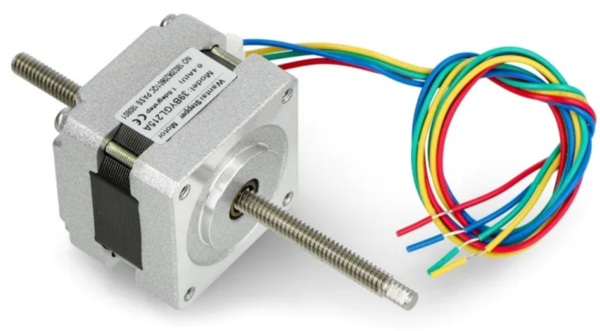


Figure 4.10: Lead screw motor.

A lead screw design will generate a high-torque linear force using a rotational input. To achieve this, a motor fixed to the nosecone spins the screw. As this happens, the screw extends outwards. In this design, a pusher plate attached to the end of the lead screw will transmit this force into the lander, gradually pushing it out of the nosecone.

The main benefits of a lead screw are the high mechanical advantage it provides to the pushing mechanism and the simplicity of the system as a whole. However, this design requires a long nosecone to house the lead screw. The lander must be pushed fully out of the nosecone, which means the lead screw must extend at least the length of the payload. While retracted, the lead screw will take up space in the nosecone equal to the length of the payload. Consequently, the length of the nosecone must be at least twice the length of the payload.

Pulley

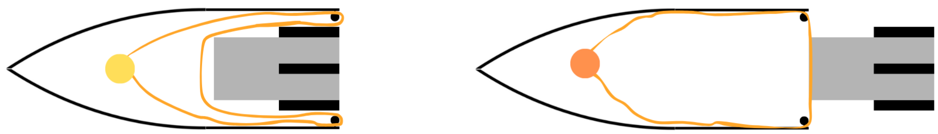


Figure 4.11: Pulley deployment system.

A system using pulleys and rope only requires one unmoving motor to deploy the lander. A rope, shown in orange in Figure 4.11, will be run from a winding motor at the top of the nosecone down to a pulley at the end of the nosecone coupler. The rope will go around this pulley before coming back up and over the forward end of the payload. The rope will loop back to another pulley on the opposite side before returning to the winding motor. By rotating the motor, the length of rope is shortened and a force is applied on top of the payload. The payload will continue to extend until it reaches the pulleys at the nosecone coupler, at which point it will fall out and be free to deploy.

A limiting factor on this design is the inability to push the payload beyond the lip of the nosecone coupler. Even with pulleys mounted directly at the end of the nosecone coupler, the payload can only extend to that point. The system will be designed so that the payload will reliably deploy out of the tracks or retention method used.

Rack and Pinion

Figure 4.12: Rack and pinion.

A rack and pinion is a simple device that transforms rotational motion to linear motion, similar to a lead screw. The pinion (a circular gear) will rotate, which causes the rack (a linear gear) to extend. At the end of the rack will be a pusher plate that will be used to deploy the lander.

This design has low torque, but it makes up for that in simplicity. A single gear will be manufactured or bought and mounted on the end of a generic motor. Similarly, the rack will be bought or manufactured in a common form. The issue of torque will be mitigated by using a high-torque motor. This will involve gearing down a common motor or buying a more expensive motor with this torque. These options add complexity and cost, respectively, both of which are undesirable.

4.3.5 Retention Method

A mechanism is needed to retain the self-righting lander before it is deployed. When the time is right, this mechanism will need to be able to release the payload reliably.

Latch

Figure 4.13: Example of a SouthCo rotary latch.

The payload will use a rotary latch as the retainment method in flight. A rotary latch is a latch that can rotate until it passes a certain point and locks. A release must be pressed to release the latch so that it opens again. Rotary latches are often made of medal and can withstand thousands of pounds of force. The example in Figure 4.13 can withstand 1061 (lbf) before failing while closed. The strength of this latch is more than enough for the forces that will be experienced in flight. Figure 4.13 is designed to be electronically controlled via a computer.

Figure 4.14: Example of a SouthCo manually actuated rotary latch.

To save money, the club will use a manual rotary latch to retain the payload. To release the payload, a non-continuous servo will be utilized to actuate the release mechanism of the latch. This will allow the computer to control the release timing of the payload upon landing. Figure 4.14 shows an example of a manual latch that will be used in the payload retention system.

Hooks

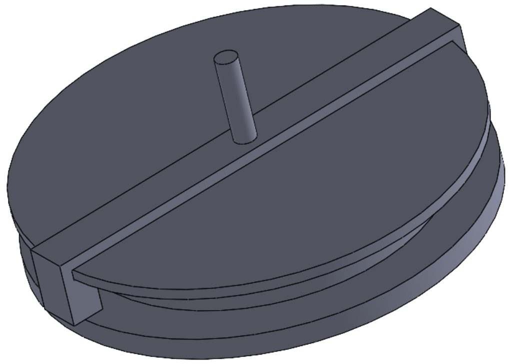


Figure 4.15: Hook retention design.

A groove around the top of the payload will allow for hooks to hold the payload before flight. The hooks will be part of the pusher plate at the end of the lead screw (Section 4.3.4) or rack and pinion (Section 4.3.4) deployment mechanisms. An indent will be manufactured into the payload structure to fit these hooks.

The hooks will be rectangular prisms connected in an ”L” shape perpendicular to the pusher plate. Two hooks will be used, at opposite sides of the pusher plate rim. When the payload is contained on its tracks within the Launch Vehicle , these hooks will prevent movement forward and aft. When the pusher plate is extended, the hooks will aid in pushing the payload out of the Launch Vehicle . Finally, when the payload has completely exited the Launch Vehicle , it will slip out of the hooks. The hooks only constrain the motion of the payload along the long axis of the Launch Vehicle , so as soon as it is free of the tracks, the payload will roll off the hooks and be independent. Even if the hooks end up nearly vertical, external forces like wind or the Launch Vehicle shaking will dislodge the payload.

4.3.6 Drill Design

An auger bit will be used by the payload to collect a dirt sample. The efficiency of this auger will be important to the overall performance of the payload. Two different materials are being considered for the auger and both are being tested for overall performance: carbon steel or 3D printed filament.

Carbon Steel Auger

A commercially sold metal auger will be adapted for use in the payload. The auger is about one foot long out of the box and will be cut down to proper size to allow for integration into the payload. This auger is made from carbon steel with a outer shell of paint. Carbon steel is strong and rigid making it a good option for the drill. This is because the auger bit will be able to withstand any stress applied by the motor or friction from the dirt.

Figure 4.16: Side view of carbon steel auger.

While the steel auger offers high strength, it comes at the cost of a high weight. The full auger weighs in at 0.492 (lbm). High weight will increase the amount of torque needed to lift the payload up during deployment.

During drill testing, the metal auger required a high amount of torque to properly drill into the dirt. This is caused by a dull edge and high friction on the coat of paint on the drill bit. The auger edge was ground down to create a sharp edge, which improved the efficiency of the auger. Friction is an important factor when considering the auger bit because motor space is limited and a high torque motor most likely will not fit in a payload.

Figure 4.17: Sharpened tip of carbon steel auger.

3D Printed Auger

A custom-made 3D printed auger will be designed and produced to proper specifications. The auger is designed to have a high strength-to-weight ratio, with low enough friction that the bit does not experience too much resistance. The 3D printed auger will be manufactured out of PETG filament. This is because when compared to PLA, PETG filament has a higher impact resistance and durability while keeping the weight the same.

Initially, an auger prototype was produced entirely out of PETG material. This design was light, weighing in at 0.05 (lbm) and performed well when tested by hand. The drill was able to penetrate the soil easily because of the sharp edge of the PETG. After the hand tests, the auger was then tested using a hand drill and failed due to high torsion loads. This is most likely because the plane that the layer lines fall on is perpendicular to the axis of rotation of the drill. Layer lines are a common failure point for 3D printed models because it is the area with the weakest bond.

Figure 4.18: PETG auger side view.

Figure 4.19: Failure point on PETG auger.

To reinforce the structure of the drill, the auger bit was redesigned to have a threaded rod screw into the center of the drill bit. This greatly increased the resistance to torsion and bending along the axis of the auger. It also did not come at a high weight cost. The reinforced bit only weighs in at 0.149 (lbm). Drill tests were conducted and the sharp edge of the reinforced auger paired with the rigid structure yielded the best results. The reinforced auger was able to resist the torsional load applied by the soil when drilling and still took advantage of the sharpness of the PETG auger.

Figure 4.20: Reinforced auger side view.

Figure 4.21: Threaded rod interface at top of auger.

4.3.7 Soil Measurement

A soil sensor must be utilized in the payload to be able to measure the required properties of the collected soil. Requirement 4.2 states that at least one of three properties must be measured and recorded, those properties being the nitrogen content, electrical conductivity, and pH. The Team has elected to attempt to measure all three properties so that the maximum amount of possible points can be scored. A commercial soil sensor will be utilized to probe the soil for data. When picking a sensor to use, the main criteria used were the cost, reliability, properties measured, and communication method.

7-in-1 Soil Sensor

This sensor utilizes five probes to measure seven different properties of the soil. The properties measured by the sensor are the temperature, moisture content, electrical conductivity, pH, nitrogen content, phosphorus content, and potassium content. It is optimal that all required properties are measured by one probe, as this can reduce the cost and electronic complexity of the payload. Furthermore, having all needed variables in one probe can free up valuable space for other systems in the payload.

Figure 4.22: 7-in-1 soil sensor.

The probe uses the RS485 communication standard with Modbus communication protocol to communicate with the client. This is standard among most commercial sensors at they are designed to be used at an industrial scale. Most small computers, such as a Raspberry Pi, cannot receive RS485 communication. This issue can simply be solved by adding in a signal converter that will convert the communication signal to UART standard. This can be read by any computer that is being considered for data processing.

A large drawback that is associated with this sensor is the lack of documentation provided on the sensor. Aside from a description of the Modbus communication protocol, there is little to no description of the systems in this device. To mitigate this risk, early testing will be conducted to ensure that the sensor is reliable. Testing will be conducted using subjects with known properties to ensure the accuracy of the sensor.

Two-Sensor Setup

Two sensors with different functionalities will be utilized when testing the soil. One probe is responsible for measuring the pH and electrical conductivity of the soil while the second probe will measure the nitrogen, phosphorus, and potassium content. The sensors are manufactured by DFRobot and require at least 5 volts. Both sensors use RS485 communication standard and communicate using the Modbus protocol.

Figure 4.23: pH and EC sensor.

Figure 4.24: NPK soil sensor.

These sensors each cost less than a probe that measures all the soil properties required, which will reduce replacement cost if one sensor fails at any point in the design process. Although the two probes reduce possible replacement cost, they will also double the space required for the probes in the payload housing. The complexity of the data collection will also increase because calls will need to request data from both sensors to get the required data.

4.4 Feasibility Study

Criteria	Weight	Nosecone Coupler	Self-Righting Lander
Complexity	3	1	-1
Cost	4	-1	-1
Manufacturability	4	-1	0
Reliability	5	0	1
Versatility	5	0	1
Unweighted Totals		-1	0
Weighted Totals		-5	3

Table 4.2: Pugh Matrix of Primary Structures

Criteria	Weight	Sliding Collar	Individual Actuation	Bevel Gear
Manufacturability	4	1	1	-1
Torque	4	1	-1	0
Mechanical Complexity	3	0	1	-1
Reliability	5	0	0	-1
Cost	4	1	-1	-1
Weight	2	1	1	-1
Unweighted Totals		3	1	-5
Weighted Totals		8	3	-16

Table 4.3: Pugh Matrix of Leg Designs

Criteria	Weight	Rack & Pinion	Pulley	Pusher Plate
Available Payload Space	4	1	0	1
Force	4	0	0	1
Mechanical Complexity	3	-1	0	0
Reliability	5	-1	0	1
Unweighted Totals		-1	0	3
Weighted Totals		-4	0	13

Table 4.4: Pugh Matrix of Deployment Mechanisms

Criteria	Weight	Air	Ground
Deployment Reliability	5	0	1
Mechanical Complexity	3	1	-1
Impact After Landing	4	-1	1
Unweighted Totals		0	-1
Weighted Totals		2	6

Table 4.5: Pugh Matrix of Deployment Locations

Criteria	Weight	Latch	Hooks
Reliability	5	1	-1
Simplicity	3	0	1
Weight	2	-1	1
Unweighted Totals		1	-3
Weighted Totals		0	0

Table 4.6: Pugh Matrix of Retention Methods

Criteria	Weight	Carbon Steel	3D Printed
Weight	3	-1	1
Drill Efficiency	5	1	0
Customization	4	-1	1
Unweighted Totals		-1	-2
Weighted Totals		2	7

Table 4.7: Pugh Matrix of Drill Designs

Criteria	Weight	7-in-1 Soil Tester	Separate Sensor
Deployment Reliability	5	0	1
Mechanical Complexity	3	1	-1
Impact after landing	4	-1	1
Unweighted Totals		0	-1
Weighted Totals		2	6

Table 4.8: Pugh Matrix of Soil Sensors

4.5 Leading Payload Design

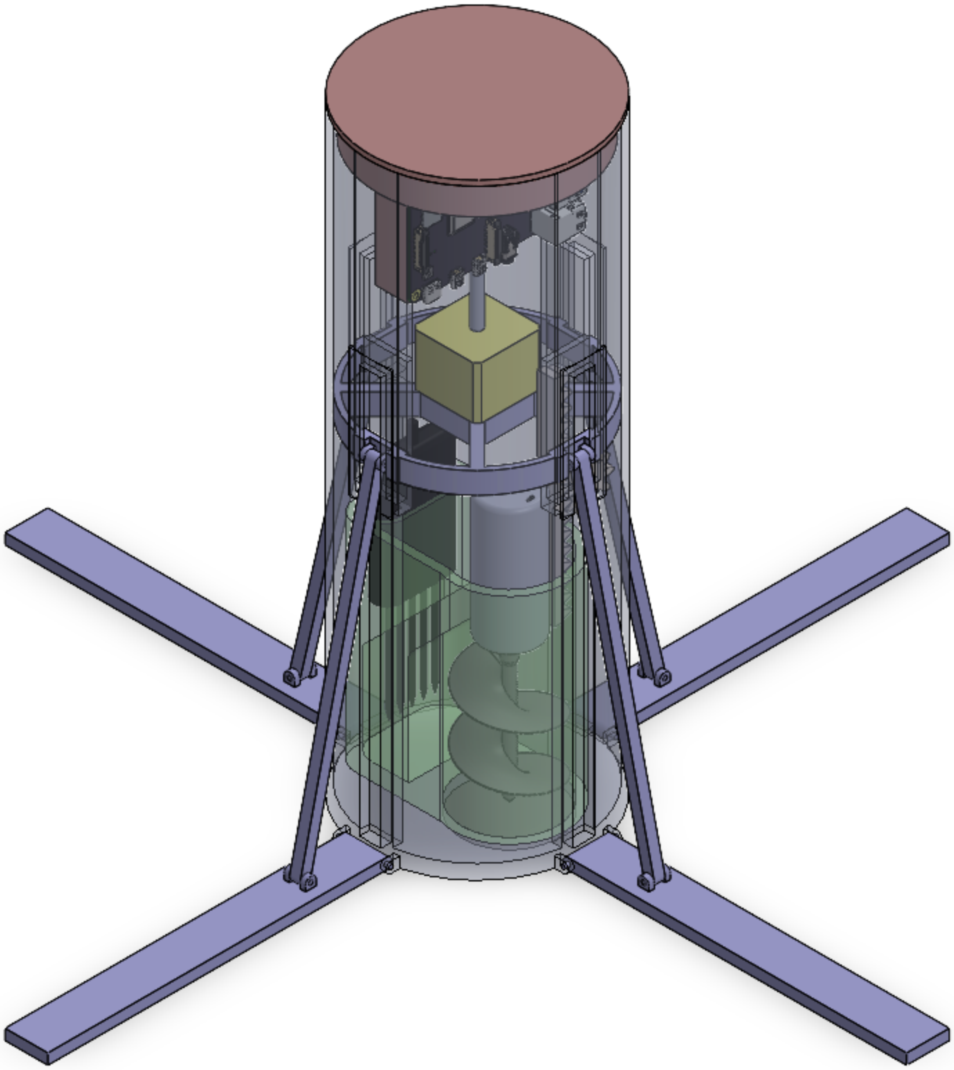


Figure 4.25: CAD model of the payload in its deployed configuration.

The payload will be a self-righting lander that is deployed on the ground. The lander design was chosen mainly due to the space required for the drill mechanism and its actuation, as this design does not have the space constraint that is present in the rotating coupler design. On-ground deployment was selected for its reduced complexity and to remove the need for RSO approval before deployment.

The lander will be a 5 (in) diameter, 14 (in) long cylinder. Inside, it will contain a 3D printed drill, a collar mechanism to deploy the legs, and the required payload electronics. Four legs will be mounted on the outside of the payload. Spaced between the legs will be tracks that connect the payload to the nosecone. The lander will be attached to a lead screw pushing mechanism with a latch at the end most forward in the nosecone, as shown in the retained configuration seen in Figure 4.26. A lead screw motor will be attached to a bulkhead above the pushing mechanism. The lead screw that it drives will extend from the pusher plate with the latch towards the top of the nosecone. The length of this lead screw will be at least the length of the payload and ideally a few inches longer.

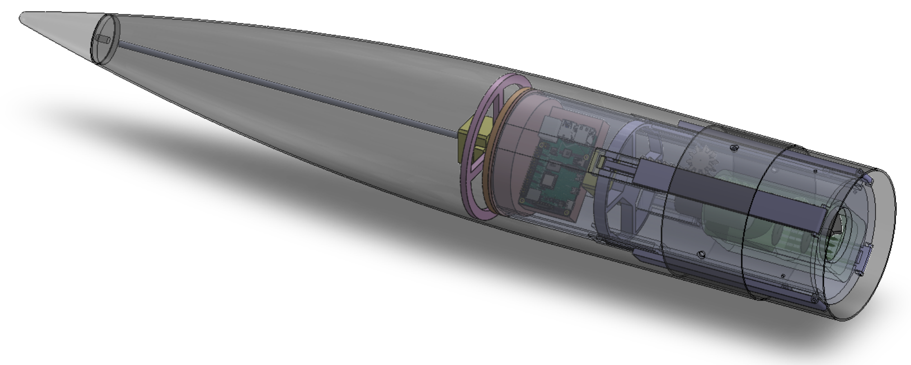


Figure 4.26: Payload retained in the nosecone.

When the Launch Vehicle is fully assembled, the payload will be attached to the latch on the pushing mechanism and mounted to tracks inside the nosecone structure. The lander will remain latched and situated between the tracks for the duration of the Launch Vehicle 's launch and descent. When landing is detected, the pushing mechanism will activate and the retaining latch will release. When the lead screw motor turns, it will drive the pusher plate and lander out of the Launch Vehicle . This motion will be aided by the tracks, which will reduce the risk of the lander getting stuck. When the pushing mechanism is fully extended and the lander is free of the tracks, the lander will be fully separated from the Launch Vehicle , shown in Figure 4.27.

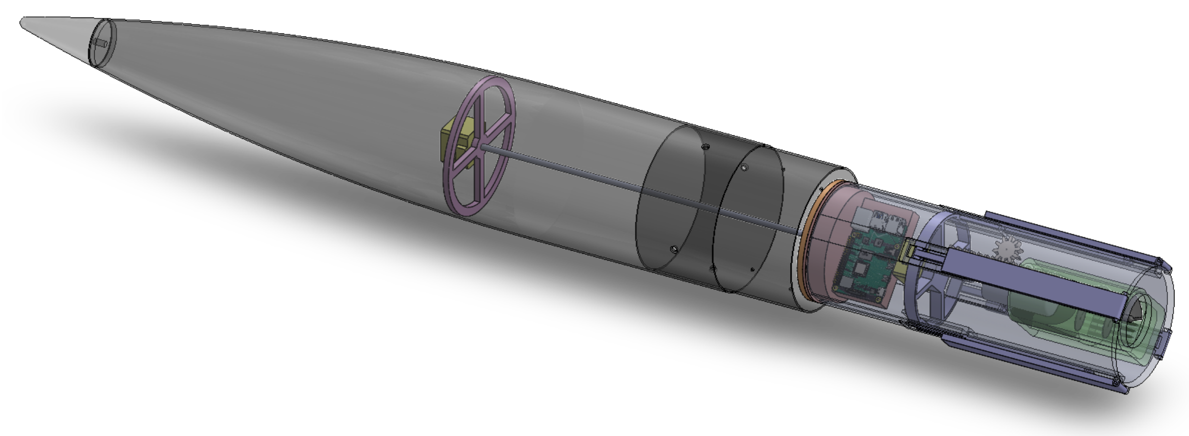


Figure 4.27: Payload after deployment from the nosecone.

Now free, the lander will self-right using its four legs. Another lead screw mechanism inside the lander will drive a collar, which connects to the legs with struts. These legs will unfold from the sides of the lander and push the lander upright, as shown in the transition from Figure 4.28 to Figure 4.29. The lander is assumed to be on its long side due to the orientation in which it was deployed. The legs will reach from the base of the payload past the center of gravity, so as they are unfolded, the payload will be lifted upright. As these legs deploy, two will always be in contact with the ground. The normal force of these two legs deploying will transfer to the payload, which will lead to its self-righting.

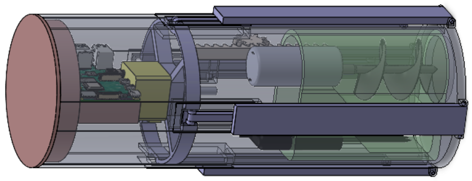


Figure 4.28: Payload horizontal on ground.

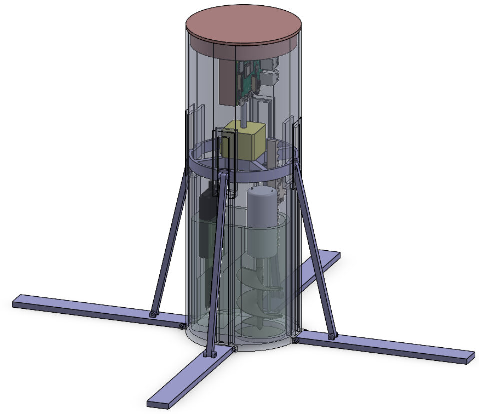


Figure 4.29: Payload after leg deployment.

When all the legs are deployed, the payload will be in a stable condition and can begin drilling. The auger drill will deploy out of the bottom of the payload. A rack and pinion will drive the auger into the soil and another motor will cause it to spin. This will facilitate easy soil collection. When fully extended, the auger will continue to spin. This will drive dirt up the auger and into a container. When the collection is done, the auger will be retracted while still rotating as if it were drilling into the ground. This will continue to bring soil up and will drive even more soil into the container. The soil sensor will already be set within the collection chamber, such that this new soil will fall into place around it. The soil sensor will be powered on and take pH, EC, and NPK readings from the soil around it. Finally, that data will be timestamped and stored on the Raspberry Pi.

A loop will run in the background of the payload code to periodically check the orientation of the lander. If it is detected that the lander has fallen over, the legs will retract and extend once again. This is an unlikely scenario and may indicate larger problems, such as a broken auger, but efforts will be made to mitigate payload failure scenarios.

4.5.1 Major Components

Body

The body of the lander will be manufactured via 3D printing. It will be a 5 (in) diameter and 14 (in) tall cylinder, 1/8 (in) thick. One of the ends will be mostly covered, but the other end will be uncovered. The covered end, designated the bottom of the lander, will have a 2.2 (in) diameter hole slightly offset from the middle. This will allow the 2 (in) diameter auger to be extended when the lander is deployed. Rectangular holes will also be manufactured 7 (in) from the bottom. These holes will provide space for the leg struts to connect to the collar mechanism inside the lander.

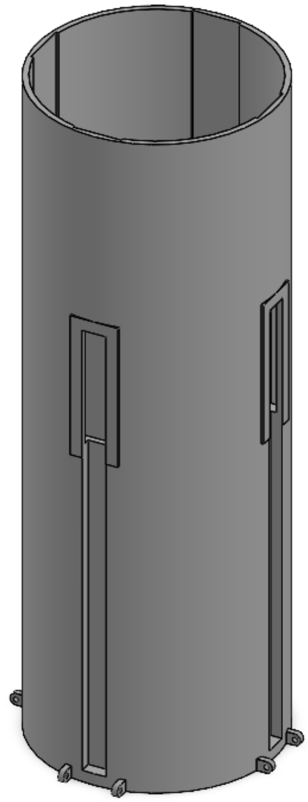


Figure 4.30: Payload main body.

The top of the payload will be capped with a separate 3D printed piece. This piece will have a sled section that extends downward, allowing for the mounting of electronics and batteries. It will also be hollow to serve as the atmosphere-isolated HAUS compartment that will house the STEMnauts during flight. The electronics sled will have a hole cut out to serve as the mounting point for the lead screw of the leg deployment mechanism.

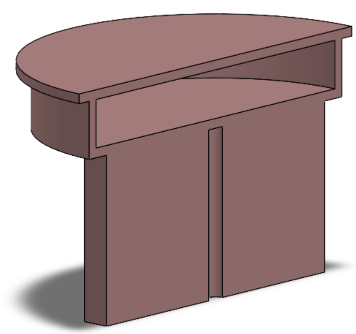


Figure 4.31: Section view of the HAUS STEMnaut compartment and sled.

Leg Deployment System

Four legs will be mounted around the payload. The legs will be 8 (in) long, 1 (in) wide, and 1/4 (in) thick. They will be hinged at one end and attached to the base of the lander. 1.4 (in) from the mounting point will be another hinge where the strut connects to the leg. This 8 (in) strut will run from the leg to a collar which actuates to deploy the legs as shown in Figures 4.32 and 4.33.

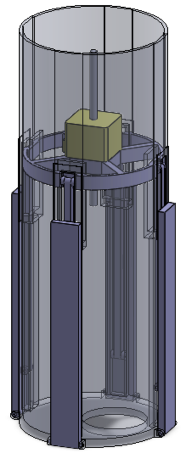


Figure 4.32: Leg system in folded state.

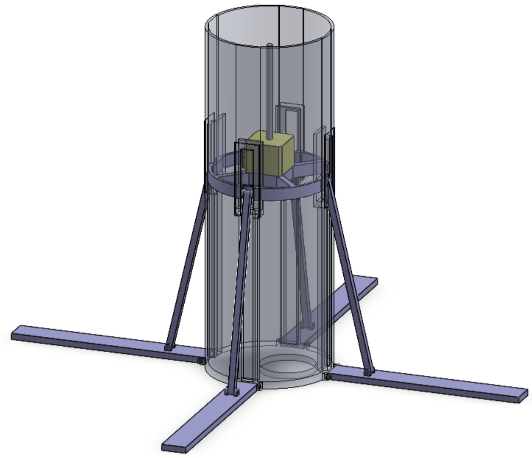


Figure 4.33: Leg system in deployed state.

The collar system was selected out of the leg deployment options due to its combination of torque capability and design simplicity compared to the other design options. The actuating collar will sit just inside the lander structure on a set of rails, and will have cutouts to allow wires to pass through to connect the electronics on the sled to the soil sensor and motors below. A lead screw motor will be mounted on top of the collar and attached to a lead screw that mounts within the electronics sled. Actuating the lead screw motor will move the collar vertically, therefore providing pushing or pulling forces to the linkages that cause the legs to fold in or out depending on the direction the collar is translated.

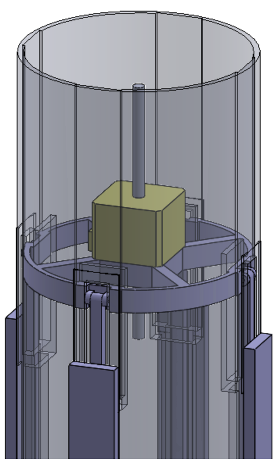


Figure 4.34: Collar location when legs are folded.

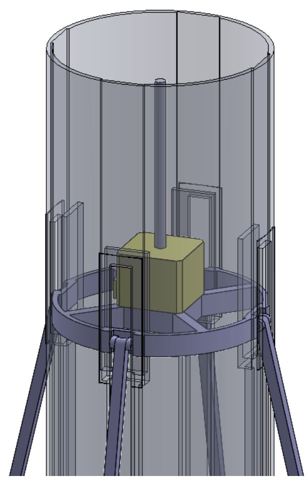


Figure 4.35: Collar location when legs are deployed.

Soil Sampling Container

A unified containment structure will be manufactured to be installed at the bottom of the payload. This structure will have space for the auger, soil sensor, and soil containment. A 2 (in) 3D printed auger attached to a motor will be mounted on a rack and pinion mechanism to move it up and down. While the auger is being actuated linearly, it will also be rotated by the motor. These motions together will allow the auger to easily dig into the soil. When the rack and pinion is fully extended, the auger will continue to rotate. This motion will force the dirt up and into the containment structure. A fixed wall and ramp will direct the dirt into a collection location. The soil sensor will already be mounted with its sensors in this container, and the soil will fall on top of it. After a set time, the auger will begin to be retracted. As the auger is retracted and spins, more soil will be lifted and pushed into the container. This will ensure a solid contact for all the soil sensor probes and good readings for pH, EC, and NPK values.

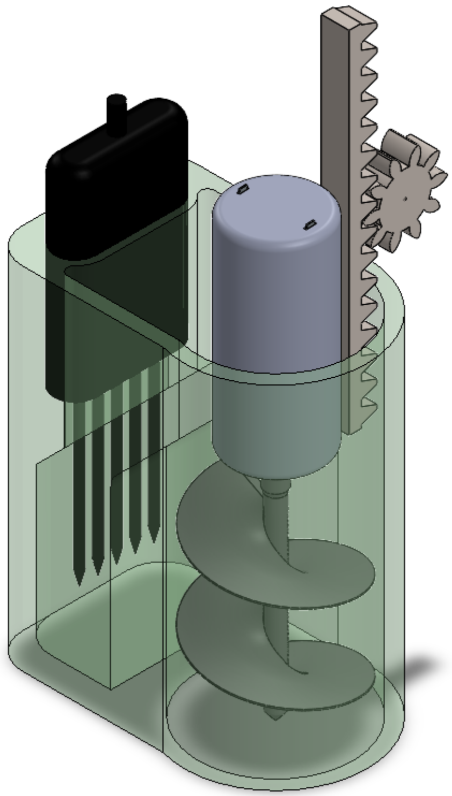


Figure 4.36: Sampling container with auger retracted.

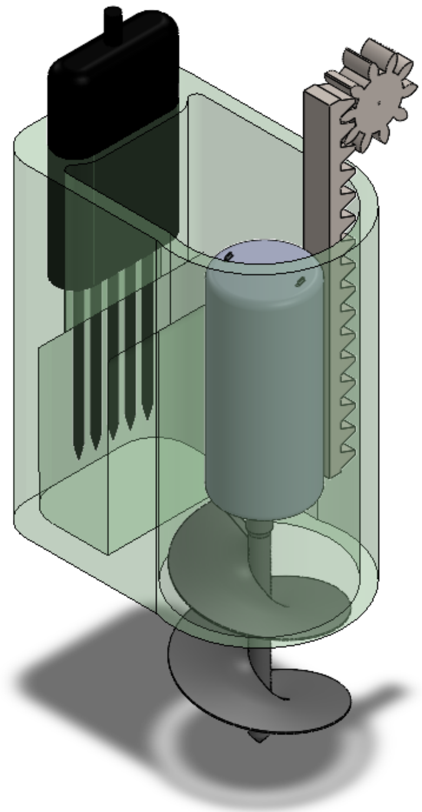


Figure 4.37: Sampling container with auger extended.

Payload Deployment System

To deploy the lander from the nosecone, a lead screw design will be used. This design was chosen for its operational simplicity and ease of manufacturability. At the top of the payload, a U-bolt in the lid will be connected to a latch mechanism. This latch mechanism will be mounted at the end of a threaded rod of length slightly greater than the payload. The threaded rod will run through a lead screw motor mounted to a bulkhead about halfway up the nosecone. Before deployment, the lander will be pushed far up into the nosecone, almost to the bulkhead. When onboard electronics detect that the Launch Vehicle has landed, the latch will be released and the lander will be deployed. This will be achieved by actuating the lead screw motor. The motor will spin and push the threaded rod outwards. The pusher plate at the end of the rod will transmit this force to the lander. The lander will slide on tracks that connect it to the inside of the nosecone. When the lander reaches the end of the tracks, it will be free of the nosecone and resting on the ground, now independent of the Launch Vehicle and ready to deploy its legs.

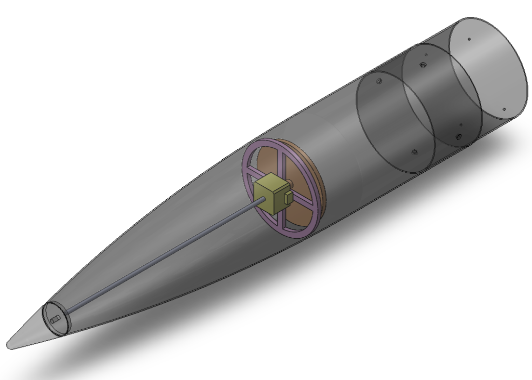


Figure 4.38: Deployment system in retracted state.

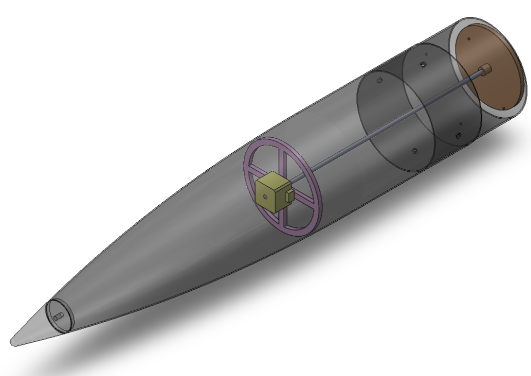


Figure 4.39: Deployment system in extended state.

4.5.2 Component Masses

Component	Amount	Unit Mass (lbm)	Total Mass (lbm)
Reinforced drill bit	1	0.149	0.149
Soil retainer	1	0.283	0.283
Soil sensor	1	0.295	0.295
Signal converter	1	0.004	0.004
Servo for latch	1	0.143	0.143
Nosecone mount	1	0.283	0.283
Raspberry Pi 4	1	0.102	0.102
FIRM	1	0.090	0.090
Planetary motor	1	0.732	0.732
Threaded rods	2	0.091	0.182
Lead screw motor	2	0.363	0.726
Leg	4	0.124	0.496
Linkage	4	0.037	0.148
Collar	1	0.248	0.248
Body	1	1.118	1.118
STEMnaut housing sled	1	0.334	0.334
Assembly hardware	1	0.1	0.1
Total Mass			5.433

Table 4.9: Major Component Masses and Quantities

4.5.3 Electrical Schematics

The payload electronics will be split up into two different systems. One system will be present on the deploying lander and will control all data collection and control. The second system will be present inside the nosecone. This system is responsible for retaining the payload during flight and ejecting the payload after landing.

Electronic Design for Deploying System

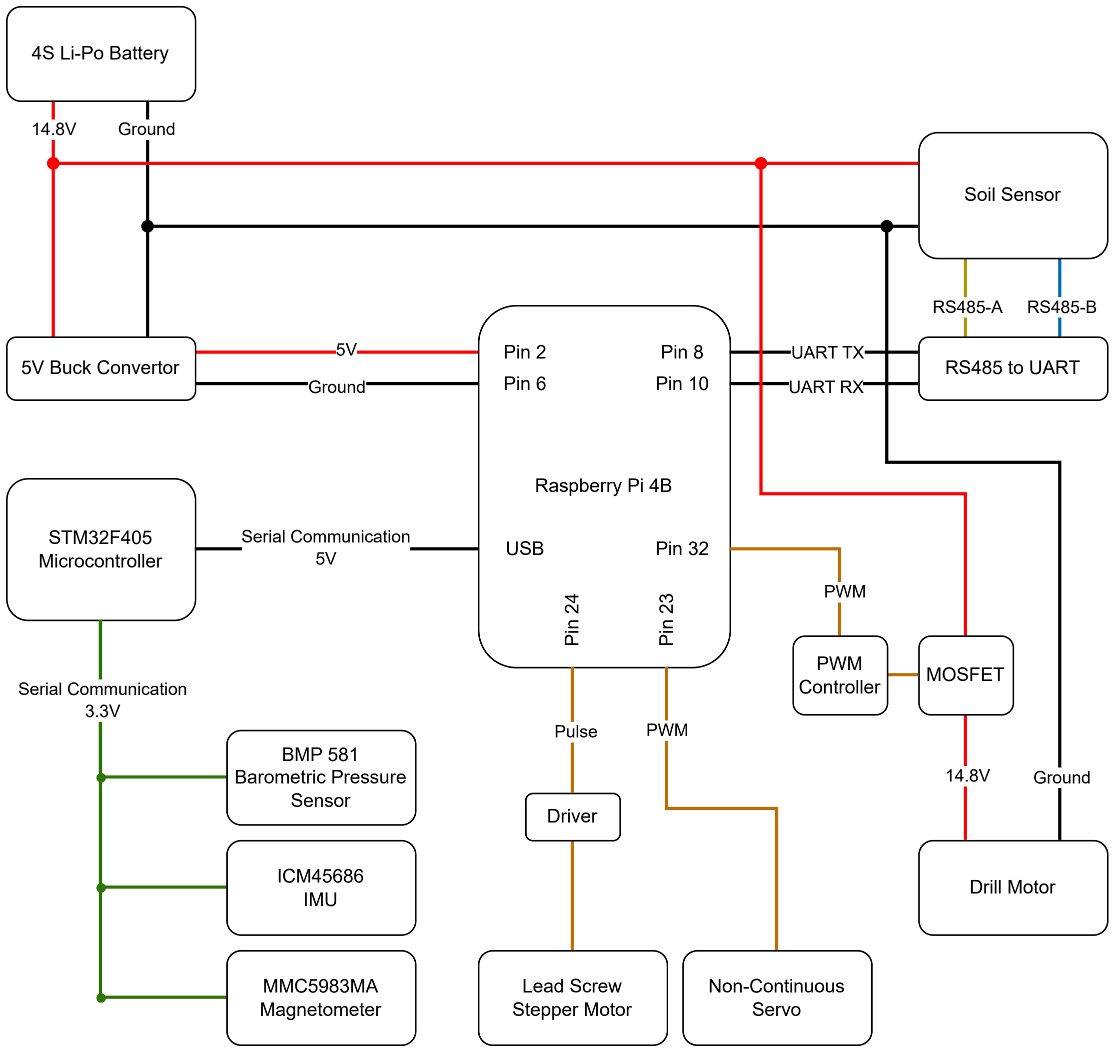


Figure 4.40: Block diagram of electronic design.

As seen in Figure 4.40, all of the electronics on-board the payload are powered by the same 4S lithium-ion polymer battery. Using one battery for all of the electronics maintains a common ground for all of the motors and sensors. A common ground ensures proper operation of all electronics. A 4S LiPo is necessary because of the power requirements of the drill motor and the soil sensor. The soil sensor requires at least 12V of power to function and the drill motor, which is currently a planetary gear motor, requires around 12V to spin at the desired speed.

The payload system will be controlled by a Raspberry Pi 4b. A Raspberry Pi was chosen over an Arduino because of the higher processing power provided by a Raspberry Pi. The Raspberry Pi will handle the input from the sensors, the data processing and logging, and all motor operation and control. The Pi also is designed to work with Python, which is the coding language the payload's software is being written in. The Pi is powered with 5V, so the 14.8V supply will be stepped down to 5V with a voltage regulator as seen in Figure 4.40. All communication through the GPIO pins will be 3.3V.

The payload's sensor array includes three sensors to determine the state of flight and orientation of the payload. According to team derived requirement PF 4, the payload has to be able to detect different states of launch. To find acceleration and rotation speeds during launch, an IMU with a 3-axis accelerometer and gyroscope will be used. This sensor will be able to detect all of the launch forces so that the software can determine what state of flight it is in. The payload will also utilize a barometric pressure sensor to determine the altitude. The altitude will be used to ensure that the flight state determination is working. It will require that certain altitudes be hit in succession so that no payload processes happen prematurely. The final sensor being used in the payload is a magnetometer. The magnetometer will be used to determine the magnetic field vector when landing. That vector paired with the accelerometer data will be used to determine the payload's orientation relative to the ground. The orientation data will be used to determine when to initiate the self-righting and drilling processes. All sensor operation will be handled by a microcontroller that communicates with the Raspberry Pi.

To reduce the size and complexity, the payload will use one soil sensor that is capable of sensing all three of the required properties of the soil. The sensor consists of five metal probes that stick into the sample to measure all of the elements of the soil. The sensor uses Modbus protocol over RS-485 communication. This communication standard is not compatible with the Raspberry Pi, so a converter board will convert the signal to UART. This is compatible with the Pi and will interface with the computer using the two UART communication pins on the Pi.

The payload will utilize three motors to control the moving parts inside the payload. A stepper motor will be used with a lead screw to articulate the legs. The stepper motor draws 0.4A under normal conditions and operates using 12V. The stepper motor will interface with the Raspberry Pi through a stepper motor driver. To control the linear actuation of the drill, a rack and pinion design will be implemented using a non-continuous servo to drive the pinion. The servo will connect directly to the Raspberry Pi as seen in Figure 4.40. The rotation of the drill head will be driven by a planetary gear motor. The motor has a nominal current draw of 0.4A, but the stall current is 20A. To prevent damage in the case of a motor stall, instead of directly controlling the motor with a PWM driver, the driver will instead connect to a MOSFET switch that controls the power transmission directly from the battery. This will protect the PWM driver and the Raspberry Pi from damaging overcurrent.

Electronic Design for Mounted Nosecone System

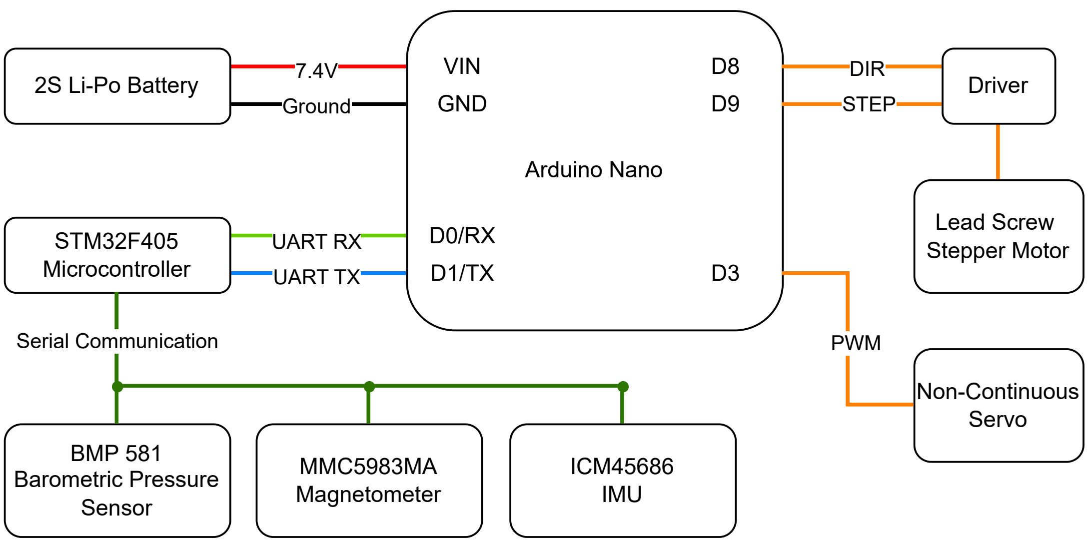


Figure 4.41: Block diagram of nosecone electrical design.

For the payload to be properly ejected, there must be a second electronic system present inside the top of the nosecone to control the pusher plate and the electronic latch that will be present in the nosecone. The processing power required for the operation of the pusher is lower compared to the power required for the main payload systems. To save money and space inside the upper section of the nosecone, a lower power Arduino Nano will be used to control the system in the nosecone. The reason there is less power needed for this system is because there is one less motor and sensor integrated into the system. There is also no need to store all of the data, which reduces necessary processing power.

A 2S LiPo battery will be used to power the electronics in the nosecone. No 5V converter will be needed to power the Arduino because it is able to receive 7-12V of power without issue. The LiPo will supply 7.4V directly to the Arduino and it will also power both motors in the system. All sensors and motors will be connected to a common ground to ensure that all components are operating correctly relative to each other.

There will be two motors used in the nosecone. One will be the stepper motor used for lead screw articulation. The Arduino will communicate with the motor through a stepper motor driver. The lead screw motor will be powered by the 2S LiPo battery. The second motor will be used to actuate the electronic latch release. This motor will be a non-continuous servo because it only needs to articulate a small latch. This does not require a full rotation of the servo horn. A PWM signal will be sent to the servo to push the latch open, releasing the payload upon landing. The sensor array used will be the same as seen in Figure 4.40. The main difference is that the Arduino cannot communicate with the sensors via USB serial connection. Instead, the microcontroller will connect directly to the processor's GPIO pins. The microcontroller will still communicate via serial connection, but it will now use the UART standard.

4.5.4 Manufacturing Methods

3D Printing

Most large parts of the payload, including the body, sample collection tube and chamber, and auger, will be 3D printed using PETG. 3D printing payload structural components allows for creating completely customizable geometries that would otherwise be difficult to fabricate, and allows for control over the accuracy of specific parts. It also allows for rapid iteration of payload design with lower cost than traditional manufacturing methods, giving the option to test fit parts of the payload for tolerance. PETG will be used in the payload fabrication for its ease of printing over other materials such as ABS, and for its higher temperature resistance and better impact resistance than PLA. Parts will be printed on HPRC's Bambu Lab A1 printer, which is located in the lab for use by the team.

Figure 4.42: Bambu Lab A1 3D printer.

Aluminum Parts

Some load-bearing payload components will require greater strength than a 3D printed part can provide. These components will be made of 6061 aluminum alloy, which will be cut out of sheets into the necessary shapes by a waterjet available for use at NCSU. Aluminum was chosen for these parts for its strength and low density, allowing the aluminum parts to have a high strength-to-weight ratio compared to other metals while still being lightweight to keep the payload's overall weight low.

Laser Cutting

The team also has access to laser cutters at NCSU for cutting wooden parts which may be required for bulkheads or other similar components. Laser cutters are preferred for manufacturing of wood parts over hand-cutting methods because of their capability to follow CAD-modeled outlines to precisely cut parts to exact specifications. Commercially available sheets of plywood will be used with the laser cutter for any wooden parts required in the payload, as plywood is available in several different standard thicknesses for various load requirements.

4.5.5 STEMnauts

Figure 4.43: STEMnaut resin duck.

The STEMnauts selected for this mission will be four small resin ducks, approximately 0.7 (in) long, 0.5 (in) wide, and 0.6 (in) tall. They will be housed in an atmosphere-isolated compartment at the top of the lander, and retained in such a way that they are secure and will not move during flight. This will be the club's fourth consecutive year flying these ducks in the SL competition flight as crew and STEMnauts. The specific ducks chosen to fly will be named at a later date.

5 Air Brakes

5.1 Air Brakes Objectives

The Air Brakes system, hereafter referred to as Air Brakes, aims to decrease the maximum altitude that the launch vehicle will reach. In its implementation, it is an active control system with four simultaneously deploying fins, which protrude into the freestream air during the coast phase of the launch vehicle. This increases the reference area to which the rocket is subjected, thereby increasing the effects of pressure drag and consequently decreasing the apogee.

5.2 Air Brakes Success Criteria

As shown in table 5.1 below, the team derived success criteria for Air Brakes.

Table 5.1: Levels of Success and Criteria for Air Brakes

Level of Success	Air Brakes Criteria
Complete Success	Apogee prediction algorithm generates a flight profile that allows the active control system to deploy and retract Air Brakes to reach an altitude within 4% of the target height when overshooting **AND** does not deploy when undershooting the target apogee.
Partial Success	Apogee prediction algorithm generates a flight profile that allows the active control system to deploy and retract Air Brakes to reach an altitude within 15% of the target altitude when overshooting **AND** does not deploy when undershooting the target apogee.
Partial Failure	Apogee prediction algorithm generates a flight profile that allows the control system to deploy and retract Air Brakes to reach an altitude within 25% of the target height **OR** deploy Air Brakes when undershooting the target apogee.
Complete Failure	Apogee prediction algorithm fails to generate a flight profile in the allotted time window **OR** the Air Brakes active control system fails to deploy Air Brakes **OR** the altitude reached is greater than 45% of the target altitude.

5.3 Air Brakes Design

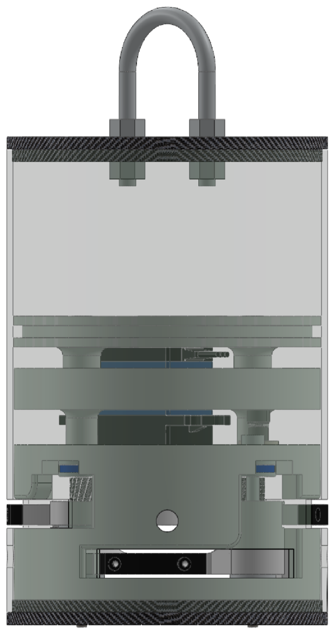


Figure 5.1: Air Brakes full assembly profile view.

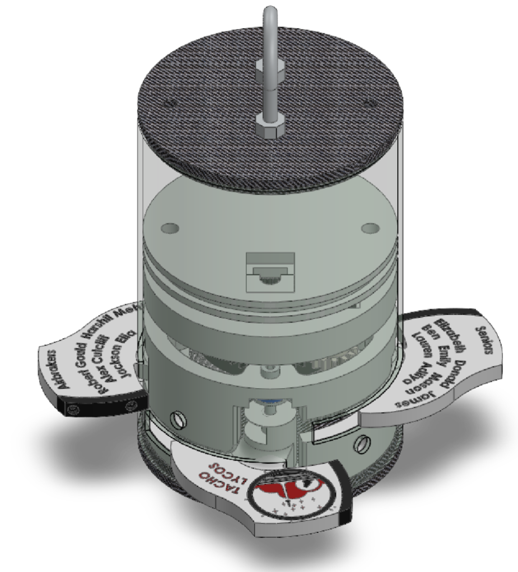


Figure 5.2: Air Brakes full assembly isometric view with fins deployed.

Shown in figure 5.1 is part of the preliminary design for the Air Brakes. The preliminary design consists of 4 simultaneously deploying fins via a central servo motor. The servo is controlled by a Raspberry Pi 5, which runs custom software to determine the current altitude, predicted apogee, and target apogee. The Air Brakes assembly will have the single servo powered by a 4S Lithium Polymer battery, or Li-Po. The entire system is situated in the Fin Can of the launch vehicle, oriented 45 degrees off center of each fin. This is to ensure no turbulent air flow is not affecting the fins. These conditions satisfy team derived requirements AF 1, AF 4, and AD 1. Furthermore, the entire assembly of Air Brakes is planned to be assembled the day before launch day for ease of use, time constraints, and troubleshooting purposes to ensure any hiccups in assembly are caught. This further satisfies team derived requirement AS 1. Each component and subsystem are further detailed in their respective sections.

5.3.1 Manufacturing Methods and Assembly

All the major components will be mostly 3D Printed. 3D Printing offers cheap, fast, and reliable parts through additive manufacturing. The material of choice is PETG due to the high ridgy of the material. Many types of PLA offer the same speed and ease of use for manufacturing but does not offer the proper yield stresses desired. All other components such as threaded rods, nuts, and screws are made of aluminum to offer high stress resistance for repeated use and weight savings for the entire system.

5.3.2 Major Components

The main component of Air Brakes is each fin. They are designed to maximize the area that is impacted in the cross flow. Each fin is structured to be circular to ensure deployability from within the body tube of the launch vehicle and to be retracted together for maximum area per fin. For the full scale launch vehicle each fin is projected to be 5.766 (in^2) and have a total reference area of 23.06 (in^2) Since there are 4 fins in total, there are a total of two pairs of fins which are staggered laterally and rotated away from one another to ensure maximum contact with the airflow. The top fin pair is labeled as tall fins, while the AFT fin pair are labeled short fins. This is also to ensure no adverse moments affect the pitch and yaw of the launch vehicle. A short fin is depicted on the left in figure 5.3 and A tall fin is depicted on the right in figure 5.4.

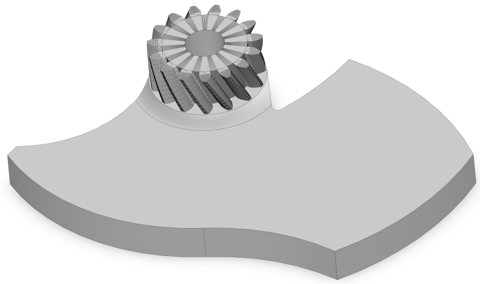


Figure 5.3: Short fin for Air Brakes.

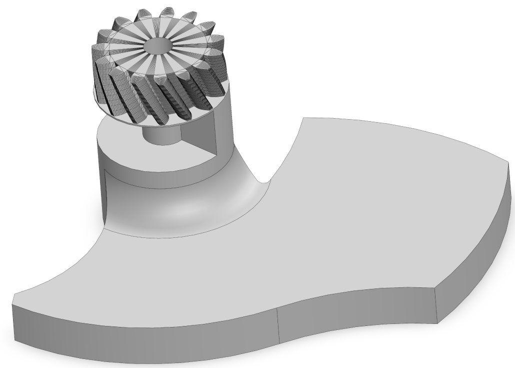


Figure 5.4: Tall fin for Air Brakes.

To drive the fins in rotation simultaneously, the tall fins have a stud with a small helical gear, while the short fins have a recess for the same small helical gear to be inserted. A large central helical gear connects each smaller gear to deploy both fin pairs together. These gears can be viewed in figure 5.5.

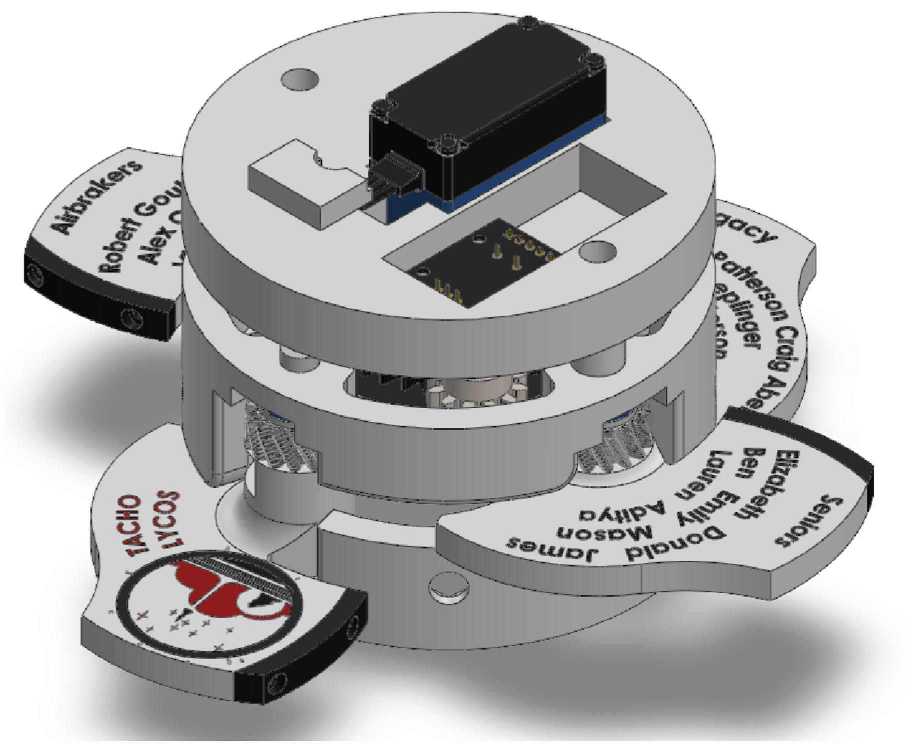


Figure 5.5: Air Brakes full deployed isometric view.

The central gear is situated under the servo motor while the smaller gears are situated in their respective locations. Helical gears were chosen to minimize the total friction of the system thereby, increasing the reliability and decreasing the rotational power needed to deploy the fins. The gear ratio is 52:15 to ensure that the system does not require too much torque to actuate the fins.

Each pair of fins and smaller gears are then mounted to a central housing via threaded rods and screws. The screws are screwed into the tall fins while the threaded rods secure the short fins. In between each housing mount location and fin are thrust bearings to ensure smooth deployment of each fin. Printed to the central helical gear is a straight cut gear, which in conjunction with another small gear, connects directly to the servo shaft to deploy the assembly. For safety, if there is high torque applied or one fin is out of alignment, the gears on the central helical gear will shear to prevent the other fins from deploying. Furthermore, there is a top plate with a groove recessed at the midpoint continuing circumferentially around the plate. This will house a fabricated O-ring to ensure the electronics sled is sealed to prevent internal pressure from affecting barometric pressure readings from the IMU. This meets team derived requirement AD 4.

Another major component of the Air Brakes assembly is the flight computer and electronics to control the actuation of the fins. These systems also measure pressure, acceleration, and other parameters to determine the location and altitude of the launch vehicle. As mentioned earlier, the central computer of the system is a Raspberry Pi 5, shown in figure 5.6 and measurements are taken with the Inertial Measurement Unit, or IMU.

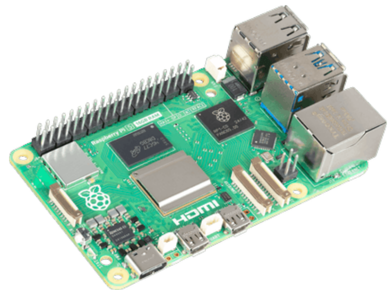


Figure 5.6: Raspberry Pi 5.

The Raspberry Pi 5 is a quad-core ARM based processing unit which has nearly double the processing power of the Raspberry Pi 4. This processing unit provides enough compute power to process incoming data from the IMU, command Air Brakes deployment, and run the apogee prediction algorithm at the same time. It also provides a ubiquitous amount of GPIO pins for sensor connections, power inputs, and data outputs. Moreover, the GPIO pins allow for the Raspberry Pi 5 to be powered via a Li-Po battery rather than the standard USB-C connector on the board.

These components are mounted on a printed sled which contains space for both the aforementioned parts, the Li-Po battery, all wiring needed for all the electrical components, and pull pin switches to ensure safety. Figure 5.7 depicts the current design concept for the sled. This will be redesigned to accommodate any future changes needed.

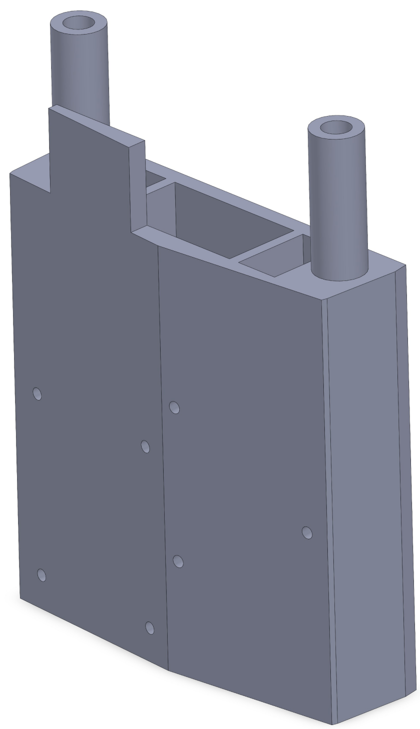


Figure 5.7: Preliminary Air Brake sled design.

Furthermore, figure 5.8 shows the Air Brakes fins fully retracted and figure 5.9 shows the fins fully deployed, both from a top down view.

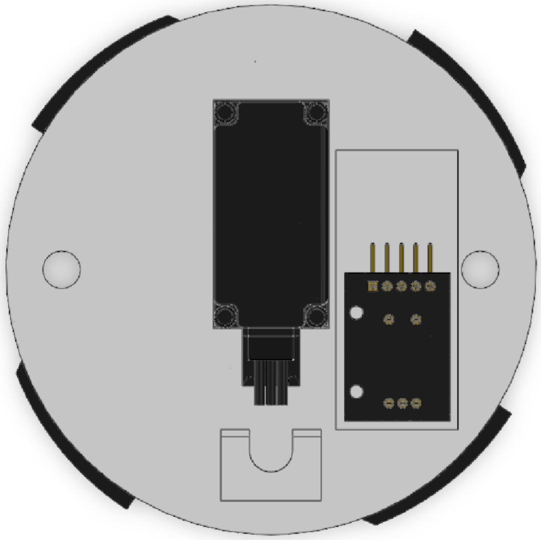


Figure 5.8: Air Brakes full retracted top down view.

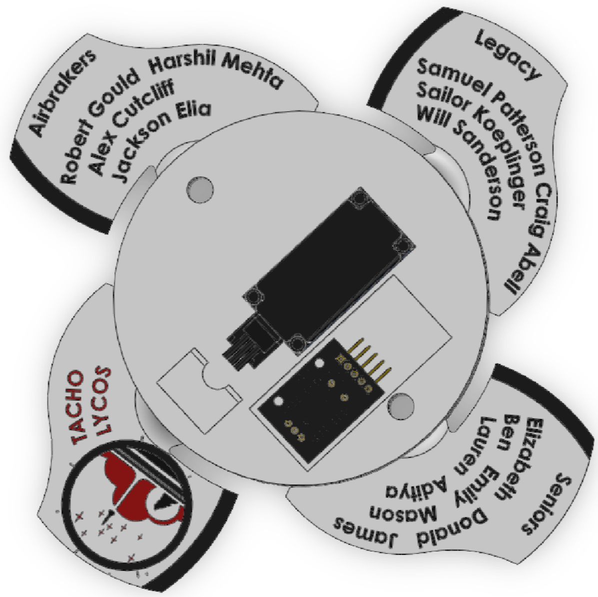


Figure 5.9: Air Brakes full deployed top down view.

5.3.3 Electrical Schematics

Primary power is supplied through one 5200 mAH 2S Li-Po battery which supplies 7.4V to the entire system. This battery allows for the proper power delivery for both the Raspberry Pi and single servo for their operating voltages. Moreover, the capacity is sufficient for the duration of the flight based on prior competition years. A capacitor connected in parallel to the output of the voltage regular to smooth out input voltage. This aids in brownout protection for the Raspberry Pi if the servo reaches its stall voltage. A camera will be housed in nose cone of the rocket to monitor the deployment of the Air Brakes. This ensures that there is a verification of deployment when all other methods in software fail due to data corruption or data loss. The camera also allows confirmation of non-deployment if the apogee prediction reads that the peak altitude is not greater than the target apogee. This meets team derived requirement AF 3.

The circuit is managed via LM2596 buck converters which stabilize power from the battery to all components connected in the circuit. Lastly, 2 pullpin switches are used to arm the Air Brakes system while the vehicle is on the launch rail, meeting team derived requirement AS 3. All these systems join together to allow for safe operation and function of the Air Brakes active control system. The leading circuit design is shown in figure 5.10

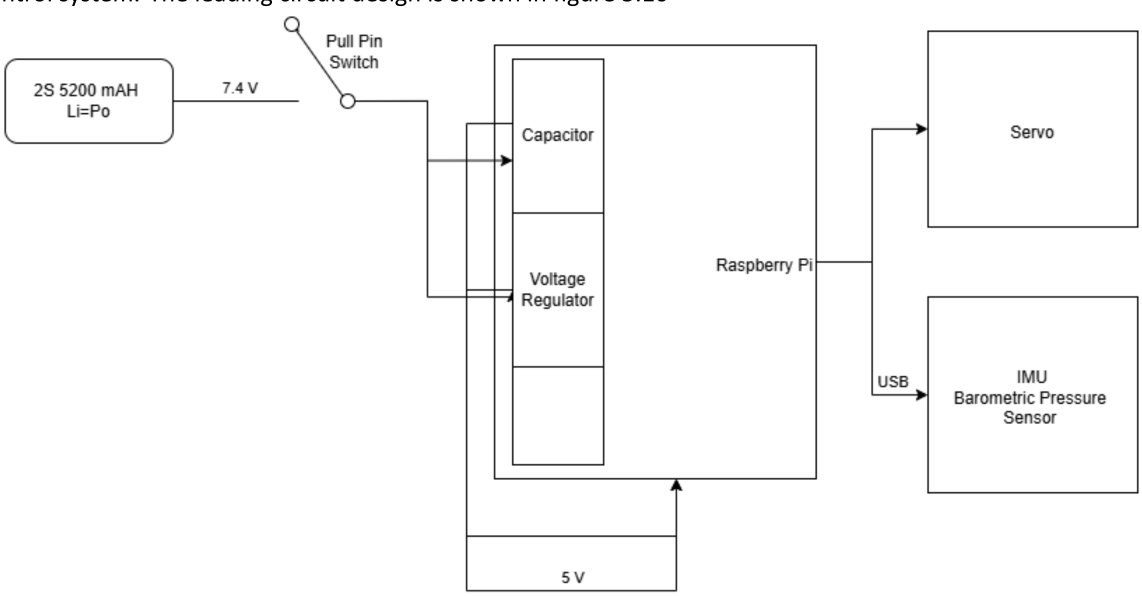


Figure 5.10: Block diagram for the Air Brakes system.

5.3.4 Air Brakes Software and Control Scheme

Control Scheme

The chosen scheme for the Air Brakes is a bang-bang control scheme. A bang-bang control scheme has only two options, either on or off. This allows for only two states that the Air Brakes may be in. There are many advantages to this approach over a traditional PID based system. First and foremost, the tuning of constants for a PID system requires many launches to be reliable, and PID constants will only be constant for a given flight configuration. This is simply not feasible for the Launch Vehicle given time and budget constraints. Secondly, a bang-bang control system is easier to program with similar performance and accuracy to that of PID system. Lastly, the differences between a PID system and bang-bang control scheme can be made up for with an accurate apogee prediction algorithm, which is outlined in section 5.3.5. This also satisfies team derived requirement AD 2.

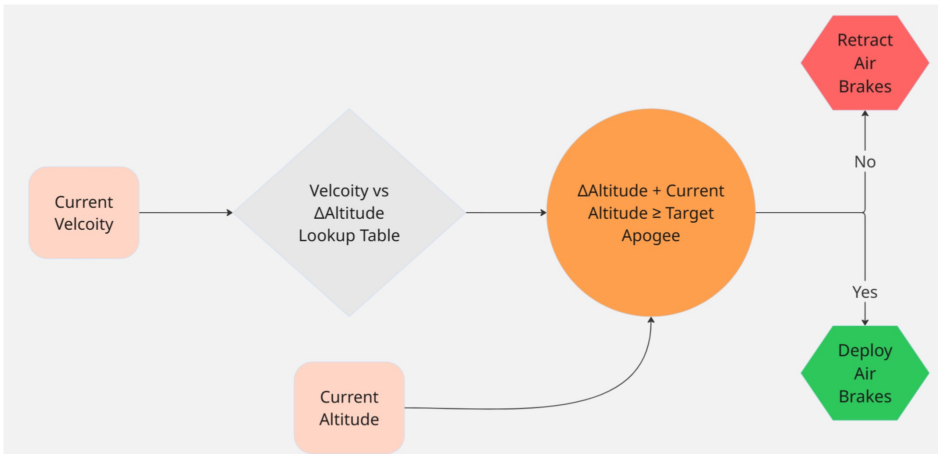


Figure 5.11: Control logic flow diagram.

Figure 5.11 illustrates the control logic used by the Air Brakes system. Each cycle, the current velocity and altitude are measured. The current velocity is fed into a look up table, which logs the first two seconds of flight after motor burn, then transitions into a reference point for the rest of the flight. Then both are fed into an apogee predictor which calculates the peak altitude and compares it to the target apogee. If peak altitude is greater than the target apogee, the Air Brakes' fins will deploy, if it is not, the fins will retract.

States for Software

The core software is written in Python and consists of a finite state machine structure that keeps track of various stages of flight. This ensures that the Air Brakes fins only extend in the coast phase and meet team derived requirement AD 2. To prevent failure of the system, there are many checks implemented in the code base to ensure each stage, also known as state, is abided by. Each state that the Air Brakes detects is described below in their respective section.

Standby State

The software boots up in the standby state. This zeros out critical measurements such as pressure, altitude and acceleration. A rolling buffer window is employed to discard redundant and unnecessary data while the vehicle idles on the launch rail. It also allows for only 6 seconds before motor ignition is recorded. This state continues until the proper checks pass for the transition into Motor Burn. This also satisfies team derived requirement AS 2, since Standby State will be defaulted to unless any of criteria below are met to switch states.

Motor Burn State

The system transitions into Motor Burn, when the speed of the rocket exceeds 32.81 $\frac{ft}{s}$. This is also triggered if the IMU measures an altitude above 32.81 ft. The data in this state is unreliable due to data noise produced by the motor burring. This data is logged, and conditions are monitored until Coast begins. This satisfies team derived requirement AF 2.

Coast State

The transition between Coast and Motor Burn occurs when the flight computer reads that the current velocity is less than the maximum velocity of the motor. This logic holds due to max velocity only occurring while the motor is burning, producing the max thrust that the launch vehicle will see. Coast can also be manually set in software as a time delay after motor burnout since the burn time is known before the time of launch. During this state, the software will take 1-2 seconds to gather data for the apogee prediction algorithm to predict the rocket's apogee. As Coast continues, the apogee prediction will be updated at a frequency of 500 Hz. Moreover, this state runs the control algorithm to deploy Air Brakes if the predicted altitude is greater than target apogee, and retract the fins if the predicted altitude is less than target apogee.

Free Fall State

Once the rocket reaches the maximum altitude, the system will run a check to detect if the current altitude has fallen by 1% of apogee. If the criteria is hit, the state calls the Air Brake's fins to retract fully.

Landing State

Lastly, the Landing State starts when the altitude measured reaches close to zero. The flight computer commands another buffer window of 10 seconds to log all final data and ensure that all motion of the vehicle has ceased. Then the program will be shut down, providing a safety measure not to use excess power, and all flight data is saved for post-analysis.

5.3.5 Air Brakes Apogee Prediction

The apogee prediction algorithm takes in quaternions, XYZ accelerations, pressure altitude, the current rocket state, and the current timestep. While in Standby, the system uses quaternions and the Earth's gravity as a reference vector to determine the orientation of the launch vehicle. It then orients the IMU to this frame of reference, which garners initial state angles on the launch rail. The frame of reference has the Z-axis perpendicular to the Earth. While in Coast, the XYZ accelerations are oriented to the same frame of reference as the IMU. This allows the Z-acceleration component to be extracted and subtracted from acceleration since gravity is accounted for in the measurements.

$$A(1 - Bt)^4 \tag{41}$$

Equation 41, is used for a curve fit for the acceleration data to be extrapolated close to zero. Apogee occurs when the z-axis acceleration reaches zero. Figure 5.12 shows the curve fit plotted against z-axis acceleration.

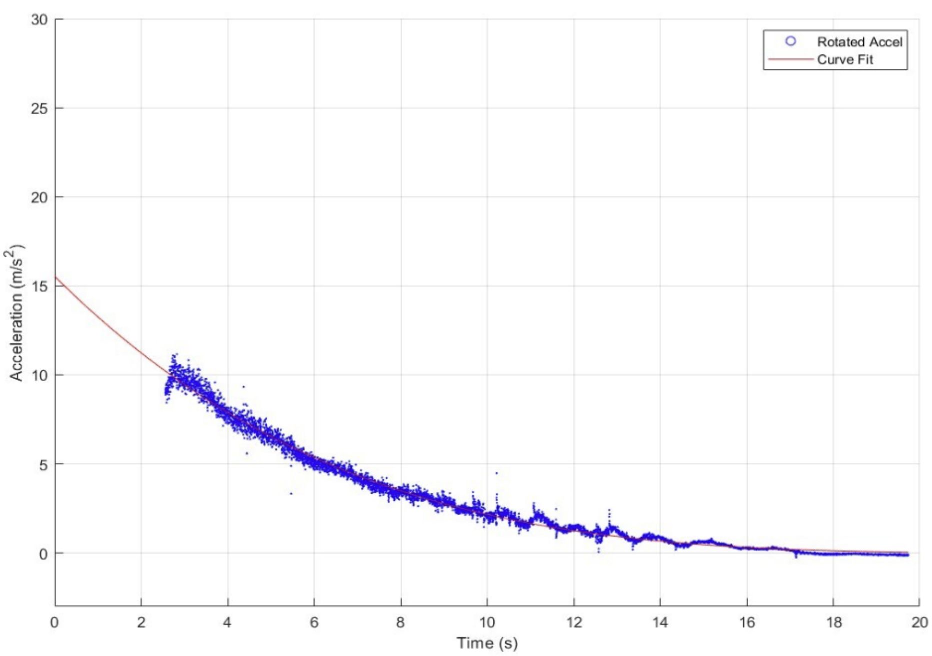


Figure 5.12: Acceleration data with curve fit.

From t= 2s to t= 4s the acceleration data starts to become less noisy, and predictions from data and the curve fit only have a difference of 1%. This is due to apogee prediction running only after motor burnout. The data is very noisy due to the thrust and high vibrations from liftoff. The altitude is calculated from a double integration of the predicted acceleration curve.

5.3.6 Component Masses

Table 5.2: Air Brakes System Component Weights

Component	Weight (oz)
Sled	2.617
IMU + Cord	3.291
Raspberry Pi 5	1.647
Fin Housing	1.524
O-Ring Plate	1.097
Tall Fin Pair	1.258
Short Fin Pair	1.13
Capacitor + Wiring	1.877
Additional Sensors	0.112
Battery	8.843
Camera	0.11
Pullpin Switches	0.268
Aluminum Threaded Rods	5.39
Thrust Bearing Assemblies	0.848
	+
Total Weight	30.012

6 Safety

6.1 Students Responsible

6.1.1 Safety Officer

Aidan McCloskey serves as the elected Safety Officer for the 2025–26 competition year. His primary responsibility is to foster a strong culture of safety within the club. This includes ensuring that all laboratory safety procedures are consistently followed. He is also tasked with maintaining the Rocketry Lab's first aid kit and personal protective equipment (PPE), keeping them fully stocked throughout the year. At the start of the school year, Aidan provided members with a lab safety quiz to communicate all safety protocols. Additional responsibilities include supervising all ejection testing, enforcing the use of launch day checklists for all club rockets, promoting safety practices during personal launches, and conducting a safety briefing before any team-attended launches.

6.1.2 Safety Team

Aidan will also serve as the leader of the Safety Team, which consists of members trained in safety protocols, launch procedures, and overall safety practices. He will conduct monthly Safety Team meetings. During launches, team members will help manage launch day checklists for WolfWorks Experimental launches. Additionally, Safety Team members are responsible for ensuring all safety protocols are followed during design, fabrication, storage, and launch activities in the event that Aidan is unable to attend.

6.1.3 Integration Lead

The Integration Lead, James Garmon, will oversee all safety documentation. James is responsible for maintaining and organizing all required Safety Data Sheets (SDS), Failure Mode and Effects Analyses (FMEA), and other relevant safety records. Aidan will collaborate with James to assist in preparing documentation, ensuring NASA Requirement 5.2 is fulfilled.

6.2 Safety Documentation

For safety documentation, Likelihood-Severity (LS) matrices were employed to visualize the effectiveness of mitigation efforts. Each identified hazard was evaluated based on its likelihood of occurrence and severity of consequence, both before and after mitigation. The design and layout of these LS matrices were guided by the Goddard Space Flight Center Guideline for Failure Modes and Effects Analysis and Risk Assessment [8].

To illustrate the progression of hazard mitigation efforts, a ranking key is used that categorizes each hazard by its severity and likelihood. Table 6.1 details the severity scale, ranging from Level 1 (lowest severity) to Level 4 (highest severity). Table 6.2 details the likelihood scale, where Level A is the least probable and Level D the most probable.

Table 6.1: Level of Severity Key

Level of Severity	Mission	Launch Vehicle	Personnel	Environment
(1) Negligible Harm	Negligible impact on mission objectives	Negligible damage	Personnel unaffected	No damage
(2) Minor Harm	Minor, reversible impact on mission objectives	Minor reversible damage	Minor injuries, can be treated with basic first aid	Minor reversible damage
(3) Moderate Harm	Partial loss or delay of mission objectives	Major reversible damage or minor irreversible damage	Moderate injuries requiring intensive first aid or professional medical care	Major reversible damage or minor irreversible damage
(4) Major Harm	Mission failure or critical compromise of objectives	Major irreparable damage or complete destruction	Urgent lifesaving medical care necessary	Major irreversible damage

Table 6.2: FMEA Likelihood Key

Likelihood of Occurrence			
A Very Unlikely	B Unlikely	C Likely	D Very Likely
0–15% Occurrence	16–25% Occurrence	26–50% Occurrence	51–100% Occurrence

Table 6.3 provides an example of a Likelihood-Severity (LS) matrix. The **ID** column assigns a unique identifier to each hazard: personnel hazards are labeled **PHZ.#**, design hazards **DHZ.#**, and environmental hazards **EHZ.#**. The **Hazard** column identifies the hazard, while the **Cause** column describes the conditions or actions that lead to it. The **Effect** column outlines the potential consequences should the hazard occur.

The **LS Pre-Mitigation** column uses the key described in Table 6.3 to indicate the hazard's likelihood and severity before mitigation efforts. The **Mitigation Factors** column includes three categories of mitigation strategies:

- **Prevention (P):** Aims to reduce the likelihood of occurrence through proactive measures.
- **Detection (D):** Seeks to reduce both likelihood and severity by identifying hazards before, during, or after they occur.
- **Mitigation (M):** Focuses on reducing the severity of the hazard through damage control procedures.

While these categories represent general objectives, their effects may overlap; each strategy can contribute to reducing both likelihood and severity.

The **LS Post-Mitigation** column records the likelihood and severity levels after mitigation measures have been applied. Finally, the **Verification** column indicates whether the mitigation has been successfully implemented. At present, this column specifies only "Verified" or "Not Verified"; however, in future documentation (e.g., CDR and FRR), it will include direct references to supporting evidence of mitigation effectiveness.

Table 6.3: FMEA Example

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification

HZ.1	Epoxy contact with skin	While working with epoxy (composites, fillets, etc.): **(1)** Insufficient PPE **(2)** Improper training	**(1)** Acute skin irritation **(2)** Allergic reaction developed due to repeated exposure **(3)** Possibility of chemical burns if allowed to cure on skin	2D	**P:** Wear nitrile gloves while handling epoxy. **D:** Understand resin curing times to identify optimal working viscosity. **M:** Wash exposed skin with cold water.	1C	Not Verified

6.3 Personnel Hazard Analysis

Personnel hazard analyses were conducted to identify potential hazards that personnel might encounter. First potential hazards were identified across all relevant environments, including the Rocketry Lab, launch fields, and outreach STEM events. Each hazard was then evaluated for its causes, possible effects on personnel safety, and the likelihood and severity of these effects. Personnel hazards were also identified based on their effect on the overall project schedule, with similar likelihood and severities of those effects identified. Appropriate mitigation strategies were developed to minimize or eliminate risks, ensuring a safe working environment and compliance with safety requirements.

Table 6.4 shows the personnel hazards before mitigation, and Table 6.5 shows the personnel hazards after mitigation.

Table 6.4: Personnel Risks Assessment Before Mitigation

		Level of Severity			
		1 Low Risk	2 Medium Risk	3 High Risk	4 Severe Risk
Likelihood of Occurrence	A Very Unlikely	0.0% (0)	0.0% (0)	3.33% (2)	0.0% (0)
	B Unlikely	0.0% (0)	6.67% (4)	5.0% (3)	6.67% (4)
	C Likely	0.0% (0)	15.0% (9)	33.33% (20)	5.0% (3)
	D Very Likely	0.0% (0)	13.33% (8)	11.67% (7)	0.0% (0)

Table 6.5: Personnel Risks Assessment After Mitigation

		Level of Severity			
		1 Low Risk	2 Medium Risk	3 High Risk	4 Severe Risk
Likelihood of Occurrence	A Very Unlikely	41.67% (23)	23.33% (14)	1.67% (1)	5.0% (3)
	B Unlikely	13.33% (8)	13.33% (8)	0.0% (0)	0.0% (0)
	C Likely	1.67% (1)	0.0% (0)	0.0% (0)	0.0% (0)
	D Very Likely	0.0% (0)	0.0% (0)	0.0% (0)	0.0% (0)

Table 6.6: Personnel Hazards

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
Hazards Encountered in Rocketry Lab							
PHZ.1	Exposure to Ammonium Perchlorate Composite Propellant (APCP)	Handling solid rocket motor propellant without nitrile gloves	Skin irritation or eye irritation.	2C	**P:** Require use of nitrile gloves. **D:** Supervisor/mentor present during motor assembly. **M:** Wash exposed skin with cold water.	1B	Verified
PHZ.2	Spontaneous motor ignition	Presence of heat sources, open flames, or electrical discharge in proximity to motor assembly	**(1)** Severe burns to personnel **(2)** Potential ignition of surrounding materials leading to fire **(3)** Risk of ear and eye injury	4C	**P:** Conduct motor assembly away from heat sources and electronics. **D:** Lead inspects assembly area for potential ignition sources. **M:** Keep fire extinguisher and burn first aid kits nearby. Store motors in flame cabinet, anti-static bags, or explosion box when not in use.	4A	Verified
PHZ.3	Exposure to Black Powder	**(1)** Improper handling of black powder during loading or disposal **(2)** Failure to use nitrile gloves	**(1)** Harmful if ingested **(2)** Serious eye irritation **(3)** Skin or respiratory tract irritation from direct contact or inhalation of dust	2D	**P:** Require nitrile gloves and safety glasses. **D:** Safety leads supervises all black powder handling and packing. **M:** Wash exposed skin with cold water.	1A	Verified
PHZ.4	Premature ignition of black powder	**(1)** Accumulation or discharge of static electricity near black powder handling areas **(2)** Presence of unintended electrical current or live wiring in proximity to pyrotechnic materials	**(1)** Thermal burns to personnel from rapid ignition **(2)** Eye injury from flash or particulates **(3)** Increased risk of localized fire or ignition of nearby materials	4C	**P:** Ground personnel before handling; keep ejection charge packing away from electronics and heat sources. Wear nitrile gloves and safety glasses. **D:** Monitor voltages and continuity during charge packing. **M:** Keep fire extinguisher ready; ensure flammable materials are clear of packing area.	2A	Verified

Table 6.6: Personnel Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
PHZ.5	Prolonged Exposure to Acetone, Isopropyl Alcohol, or Spray Paint Fumes	**(1)** Inadequate ventilation and dust collection in work areas **(2)** Failure to wear 60926 Multi Gas/Vapor Cartridge/P100 Filter mask respiratory protection	**(1)** Headaches, dizziness, or nausea **(2)** Respiratory irritation or distress **(3)** Potential long-term health effects from repeated inhalation of volatile organic compounds	3D	**P:** Work in well-ventilated areas, wear respirators, and limit exposure time. **D:** Detect abnormal or strong unfamiliar odors. **M:** Move to fresh air after exposure; take breaks if symptoms occur.	1A	Verified
PHZ.6	Exposure to Colloidal Silica Particles	**(1)** Mixing or handling colloidal silica without N95 or P100 respiratory protection **(2)** Failure to wear nitrile gloves and safety eyewear	**(1)** Coughing and respiratory tract irritation due to inhalation **(2)** Skin irritation from direct contact with material **(3)** Eye irritation or discomfort from airborne particles	3C	**P:** Wear nitrile gloves, safety glasses, and N95 or P100 respirator. **D:** Stay aware of nearby operations, wear a mask if colloidal silica is in use. **M:** Wash exposed skin and eyes with water; clean affected area thoroughly.	2A	Verified
PHZ.7	Epoxy contact with skin	While working with epoxy (composites, fillets, etc.): **(1)** Insufficient PPE **(2)** Improper training	**(1)** Acute skin irritation **(2)** Allergic reaction developed due to repeated exposure **(3)** Possibility of chemical burns if allowed to cure on skin	2D	**P:** Wear nitrile gloves while handling epoxy. **D:** Understand resin curing times to identify optimal working viscosity. **M:** Wash exposed skin with cold water.	1C	Verified
PHZ.8	Skin Contact with Sharp Composite Edges	**(1)** Handling composite parts without nitrile gloves **(2)** Lack of awareness of sharp edges on fabricated components	**(1)** Lacerations or puncture wounds from sharp edges **(2)** Splinters embedded in skin **(3)** Risk of secondary infection if wounds are not treated with bandages and antibiotic ointment.	2C	**P:** Ensure composites are sealed and edges are smooth; wear nitrile gloves during fabrication. **D:** Inspect each composite component before handling. **M:** Remove splinters with tweezers; apply antibiotic ointment as needed.	1A	Verified
PHZ.9	Entanglement with Rotating Power Tools (Drill Press, Miter Saw, etc.)	**(1)** Presence of loose clothing, long hair, or jewelry near moving parts **(2)** Failure to adhere to machine safety or guards	**(1)** Lacerations **(2)** Dismemberment **(3)** Crushing injuries to extremities	3D	**P:** Tie back hair, remove jewelry, and wear fitted clothing. **D:** Lead observes operation for unsafe behavior. **M:** Use emergency stop buttons; keep first aid nearby.	2A	Verified

Table 6.6: Personnel Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
PHZ.10	Kickback or Excessive Friction During Cutting or Drilling	**(1)** Improper clamping or securing of workpiece **(2)** Use of dull cutting tools **(3)** Incorrect feed rates or rotational speeds	**(1)** Impact injuries to hands, arms, or other body parts **(2)** Lacerations or contusions from sudden tool or material movement **(3)** Risk of eye injury from ejected debris or tool fragments	3C	**P:** Secure workpiece, use sharp tools, and correct feed/speed rates. **D:** Ensure guards are installed and functional. **M:** Use emergency stop buttons; keep first aid nearby.	1A	Verified
PHZ.11	Contact with Moving Blades or Sanding Belts	**(1)** Operator inattention or distraction while using equipment **(2)** Bypassing or removing machine guards	**(1)** Severe cuts, abrasions, or lacerations **(2)** Risk of partial or complete dismemberment of fingers	3C	**P:** Wear safety glasses, train personnel, and never bypass guards. **D:** Ensure guards are in place before use. **M:** Use emergency stops; keep first aid nearby.	2A	Verified
PHZ.12	Ejected Debris or Chips	**(1)** Improper feed techniques during cutting, drilling, or grinding operations **(2)** Failure to use machine guards or personal protective equipment	**(1)** Lacerations, or punctures **(2)** Facial injuries	3C	**P:** Feed material slowly; use guards and safety glasses. **D:** Inspect material to ensure smooth cutting operation. **M:** Use emergency stops; keep first aid nearby.	1A	Verified
PHZ.13	Dust Inhalation from Sanding or Sawing	**(1)** Inadequate ventilation and dust collection in work area **(2)** Failure to use N95 or P100 respiratory protection	Respiratory irritation, coughing, or shortness of breath	2C	**P:** Use respiratory protection and ensure ventilation and dust collection. **D:** Inspect workspace for dust accumulation; clean as needed. **M:** Move to fresh air after exposure.	1A	Verified
PHZ.14	Electrical Shock While Using Power Tools	**(1)** Damaged or frayed power cords connected to tools **(2)** Exposure to live electrical components due to improper handling	**(1)** Electrical shock or burns to personnel **(2)** Arc flash or flash burns from sudden electrical discharge **(3)** risk of fire	3A	**P:** Inspect power cords before use. **D:** Stop operation immediately if broken wires are observed. **M:** Disconnect power; use insulated tools.	1A	Verified
PHZ.15	Noise Exposure While Using Power Tools	**(1)** Prolonged operation of high-noise tools without hearing protection such as earplugs **(2)** Lack of awareness or enforcement of occupational noise safety standards	**(1)** Temporary or permanent hearing loss **(2)** Tinnitus (ringing or buzzing in the ears)	2C	**P:** Use earplugs during prolonged tool operation. **D:** Inspect equipment if it sounds unusually loud or damaged. **M:** Take breaks during extended use of loud tools.	2A	Verified

Table 6.6: Personnel Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
PHZ.16	Loss of Control of Handheld Power Tools	**(1)** Improper grip or handling technique during operation **(2)** Tool kickback due to binding, improper feed rate, or incorrect use	**(1)** Injuries to hands, wrists, or arms, including lacerations or fractures **(2)** Facial injuries from sudden tool movement or flying debris	3D	**P:** Train users on grip and handling techniques; wear safety glasses. **D:** Stop use if abnormal vibration occurs and inspect tool for damage. **M:** Keep first aid kit nearby.	1B	Verified
PHZ.17	Contact with Heat Gun Nozzle or Hot Air	**(1)** Improper handling or use of the heat gun **(2)** Operator inattention or distraction near the heated nozzle or airflow	**(1)** Thermal burns to skin or underlying tissue **(2)** Ignition of flammable materials in the surrounding area	2D	**P:** Train correct handling; clear area of flammable materials. **D:** Monitor heat gun temperature. **M:** Provide burn aid supplies.	1A	Verified
PHZ.18	Tripping Over Loose Cords	**(1)** Electrical or extension cords placed across walkways or in crowded workspaces **(2)** Poor cable management (tangled, loose wires on workspace) or failure to secure cords	**(1)** Trips and falls resulting in bruises, sprains, or fractures **(2)** Potential secondary injuries from falling onto equipment or tools	2D	**P:** Route cables overhead rather than across walkways. **D:** Inspect work area for cables before starting work. **M:** Reroute or secure cords as needed.	1A	Verified
PHZ.19	High-Speed Dremel Bit Contact with Skin	**(1)** Operator inattention or distraction during use **(2)** Loss of grip or improper handling technique **(3)** Inadequate personal protective equipment	Cuts, punctures, or lacerations to the skin	3D	**P:** Train team members on tool handling and bit selection. Use Safety Glasses **D:** Monitor RPM during use. **M:** Stop tool immediately and apply first aid.	2B	Verified
PHZ.20	Fragmentation of Dremel Bits	**(1)** Exceeding the recommended rotational speed (RPM) **(2)** Applying excessive force or pressure during operation **(3)** Using bits outside of their intended material or application	**(1)** Cuts or lacerations from flying fragments **(2)** Eye injuries from high-velocity debris	3C	**P:** Train on usage and speed limits. Always wear safety glasses during operation. **D:** inspect bit for damage before use **M:** Check that a Dremel bit isn't cracked before use.	2A	Verified
PHZ.21	Contact with Soldering Iron Tip	**(1)** Accidental contact with the hot tip during use **(2)** Improper placement or storage of the soldering iron while hot **(3)** Inattention or distraction during soldering operations	Burns	3C	**P:** Use stands that hold soldering iron tip steady; keep workspace uncluttered. **D:** Monitor soldering tip temperature. **M:** Keep burn relief supplies nearby.	2A	Verified

Table 6.6: Personnel Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
PHZ.22	Electrical Shock or Arc Flash from Lab Power Supplies	**(1)** Improper wiring or exposed electrical connections **(2)** Incorrect use of power supplies or failure to follow safe operating procedures	**(1)** Electrical shock or burns to personnel **(2)** Arc flash causing thermal injuries or damage to nearby equipment **(3)** Increased risk of fire from electrical faults	3C	**P:** Follow correct wiring procedures; use insulated equipment. **D:** Inspect voltage and current values during operation. **M:** Keep burn first aid available.	2B	Verified
PHZ.23	Overcharging LiPo Batteries	**(1)** Incorrect charger settings or use of an incompatible charger **(2)** Negligence or inattention during the charging process	**(1)** Battery venting or thermal runaway **(2)** Fire or explosion risk	3C	**P:** Follow manufacturer's instructions and charger compatibility. **D:** Monitor charge progress; listen for completion indicators. **M:** Place failed batteries in sand bucket for containment.	1B	Verified
PHZ.24	Punctured or Damaged LiPo Batteries	**(1)** Improper handling, storage, or transportation of batteries **(2)** Incorrect charging procedures or use of damaged cells	**(1)** Fire or thermal runaway **(2)** Explosion risk from rapid gas release or combustion **(3)** Severe burns or other injuries to personnel	3C	**P:** Protect batteries from sharp objects during handling and storage. **D:** Inspect all batteries for damage before use. **M:** Treat burns; bury compromised batteries in sand bucket.	1A	Verified
PHZ.25	Burns from 3D Printer	**(1)** Accidental contact with heated components such as the nozzle, or heated bed **(2)** Inattention or improper handling during printer operation or maintenance	Minor burns to skin	2D	**P:** Keep hands away from heated components; provide user training. **D:** Monitor nozzle and bed temperature. **M:** Apply burn first aid.	1A	Verified
PHZ.26	Dust Exposure from Sanding or Cutting Composite Components	**(1)** Dry sanding or cutting without dust extraction **(2)** Failure to use PPE, such as N95 or P100 masks/respirators, nitrile gloves, or safety glasses	**(1)** Respiratory irritation or long-term respiratory issues from inhalation of fine particles **(2)** Skin irritation due to contact with composite dust **(3)** Eye injuries from airborne particles	3D	**P:** Use N95 or P100 respirators/masks, safety glasses, and nitrile gloves. **D:** Inspect and clean work area for composite dust accumulation. **M:** Wash exposed skin with cold water and move to fresh air.	2A	Verified

Table 6.6: Personnel Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
PHZ.27	Exposure to Flammable Mold Release Agents	**(1)** Handling without nitrile gloves or protective clothing **(2)** Inhalation of dust or particles from solid material **(3)** Storage near heat or ignition sources	**(1)** Eye, skin, nose, or throat irritation **(2)** Fire or explosion hazard due to flammable vapors **(3)** Ingestion may be fatal if swallowed and enters airways **(4)** Potential long-term health effects from repeated exposure, including carcinogenic risk	3D	**P:** Use nitrile gloves and respirators; store securely away from ignition sources. **D:** Monitor storage conditions. **M:** Wash exposed skin with cold water and move to fresh air.	2A	Verified
PHZ.28	Transportation/storage of Heavy Objects	**(1)** Improper lifting technique or manual handling **(2)** Unsecured storage or placement of heavy objects	**(1)** Crush injuries or fractures to hands, feet, or other body parts **(2)** Risk of damage to equipment or work surfaces	3C	**P:** Use lifting techniques that limit strain and injury risk; avoid overexertion. **D:** Leads verify lifting safety; avoid lifting objects beyond comfort level. **M:** Provide first aid for crush injuries.	2A	Verified
PHZ.29	Unintentional Igniter or Ematch Activation	**(1)** Static electricity discharge or stray electrical currents **(2)** Mishandling of igniters or e-matches during preparation or transport	**(1)** Burns **(2)** Fire or ignition of surrounding materials	2B	**P:** Use anti-static bags, safety glasses, and grounding procedures. **D:** Inspect igniters and wiring for damage. **M:** Keep fire extinguisher and burn first aid supplies nearby.	1A	Verified
PHZ.30	Skin Pierced by Sharp Object	**(1)** Improper training or lack of familiarity with equipment **(2)** Failure to use PPE, safety glasses or	**(1)** Cuts, punctures, or lacerations to the skin **(2)** Bleeding or potential for secondary infection if wounds are not treated	2C	**P:** Provide training on equipment use; utilize guards and PPE. **D:** Inspect components for sharp edges before handling. **M:** Treat wounds with first aid and antibiotic ointment.	1A	Verified
PHZ.31	Injuries from Improper Hammer or Mallet Use	**(1)** Improper swinging technique or loss of control during use **(2)** Striking incorrect surfaces or materials **(3)** Slippage of the tool due to poor grip or worn handle	Fractures, bruising or blunt force injuries from impact	3C	**P:** Train striking technique that is controlled and aligned, striking surfaces head-on. **D:** Inspect workpiece; ensure flat contact surfaces. **M:** Keep first aid nearby.	1A	Verified

Table 6.6: Personnel Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
PHZ.32	Contact with File During Operation	**(1)** Slipping or loss of control while using the file **(2)** Improper technique or incorrect application of force	Bruising, abrasions, or cuts to the skin	2D	**P:** Use file with controlled, forward strokes while maintaining a firm grip on the handle, keeping hands clear of the cutting surface, and securing the workpiece to prevent slippage. **D:** Inspect material and re-secure as needed. **M:** Apply first aid for cuts or abrasions.	1A	Verified
PHZ.33	Contact with Sharp Blades	**(1)** Improper cutting techniques or mishandling of blades **(2)** Missing or bypassed safety guards **(3)** Inattention or distraction during operation	**(1)** Cuts, lacerations, or puncture wounds to the skin **(2)** Bleeding and potential secondary infection if wounds are not treated with antibiotic ointment and bandages.	3C	**P:** Maintain controlled use of cutting tools that keeps the cutting edge pointed away from the body, maintain a firm grip, and use safety guards as intended, keeping hands clear. **D:** Maintain awareness of blade position at all times. **M:** Apply bandages and antibiotic ointment as needed.	2A	Verified
PHZ.34	Pinching of Appendages or Skin in Clamping	Careless or improper use of clamps or compression equipment	**(1)** Bruising, abrasions, or cuts to hands, fingers, or other body parts **(2)** Potential secondary injuries from sudden release or movement of clamped objects **(3)** Risk of crushed tissue or joint injury in severe cases	2B	**P:** Train users; avoid over-clamping. **D:** Observe each setup to keep hands clear of clamping area. **M:** Apply first aid for bruises or cuts.	1A	Verified
PHZ.35	Hot Glue Contact with Skin	**(1)** Accidental contact with the hot glue gun tip or molten adhesive **(2)** Improper handling or inattention during application	Burns to skin	2D	**P:** Keep hands clear of glue gun tip and molten adhesive. **D:** Keep power disconnected when not in use. **M:** Wash skin and apply burn first aid.	1B	Verified
PHZ.36	Skin Contact with Sandpaper During Hand Sanding	**(1)** Improper handling or technique while sanding by hand **(2)** Inattention or lack of control over sanding motion	Abrasions, scratches, or minor cuts to the skin	2D	**P:** Secure material and use controlled motion. **D:** Inspect sandpaper and replace when worn. **M:** Wash exposed skin and apply bandages or ointment.	1A	Verified

Table 6.6: Personnel Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
PHZ.37	Exposure to High-Pressure Compressed Air	**(1)** Direct contact with compressed air hose or nozzle **(2)** Improper use or handling of compressed air equipment **(3)** Pointing nozzle toward the body or others during operation	**(1)** Eye injuries from debris propelled by compressed air **(2)** Skin injuries, including bruising **(3)** Hearing damage from high-pressure air discharge	3C	**P:** Never direct nozzle at people; wear PPE including ear protection. **D:** Monitor shutoff valves and ensure they function as intended. **M:** Apply first aid for injuries as needed.	1B	Verified
Hazards Encountered on Launch Field							
PHZ.38	Contact with Black Powder (Unblown Charges)	**(1)** Failure to detonate black powder charges **(2)** Handling without nitrile gloves **(3)** Inattention during recovery	skin irritation	2B	**P:** Require use of nitrile gloves and safety glasses during recovery. **D:** Inspect for unblown charges; check for charge leaks. **M:** Wash exposed skin with cold water, neutralize black powder with water, and provide first aid for irritation.	1A	Verified
PHZ.39	Launch Vehicle Components Entering Ballistic Trajectory	**(1)** Failure of any launch vehicle stage or component to separate. **(2)** Parachute or recovery system failure **(3)** Components becoming detached from the launch vehicle during flight	**(1)** Impact injuries, including fractures or blunt force trauma **(2)** Potential fatality from high-velocity impacts	4B	**P:** Instruct personnel on procedures during ballistic descent of any launch vehicle. **D:** Listen to the launch coordinator and maintain visual contact with launched vehicles. **M:** Maintain a safe distance from recovery areas and contact EMS if an impact occurs.	4A	Verified
PHZ.40	Airborne Shrapnel	**(1)** Motor overpressure or casing rupture **(2)** Improper handling or assembly of motor components	**(1)** Eye injuries, cuts, or bruising from high-velocity fragments **(2)** Potential fatality from severe impacts	4B	**P:** Verify motor assembly per manufacturer's recommendations; maintain safe distances from the launch pad. **D:** Lead inspects motor assembly; follow launch coordinator instructions during launches. **M:** Provide PPE for personnel near the launch pad; contact EMS if a serious injury occurs.	4A	Verified

Table 6.6: Personnel Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
PHZ.41	Smoke from Motors	**(1)** Ignition of propellant producing smoke or particulate matter **(2)** Incomplete combustion of motor fuel **(3)** Wind direction carrying smoke toward personnel or work areas	**(1)** Respiratory irritation or difficulty breathing **(2)** Eye irritation from smoke or airborne particles **(3)** Reduced visibility	2C	**P:** Launch from safe distances and monitor wind direction. **D:** Observe smoke dispersion; monitor personnel for irritation. **M:** Move personnel upwind, rinse eyes with clean water, and provide respiratory protection if needed.	1A	Verified
PHZ.42	Loud noises	**(1)** Rocket ignition, pyrotechnic events **(2)** Lack of use of hearing protection such as earplugs.	**(1)** Temporary or permanent hearing damage **(2)** Tinnitus	2C	**P:** Train members to remain alert during all launches, not just team launches. **D:** Alert members when launches are occurring, especially if the launch coordinator is out of earshot. **M:** Move affected personnel away from loud areas (e.g., into vehicles) and provide first aid as needed.	1B	Verified
PHZ.43	Motor Ignition During Assembly	**(1)** Accidental activation of the motor due to static electricity, stray currents, or mishandling **(2)** Improper assembly procedures or failure to follow safety protocols **(3)** Presence of ignition sources near the motor during assembly	**(1)** Burns or lacerations to personnel **(2)** Fire or ignition of surrounding materials	4B	**P:** Follow anti-static protocols; keep ignition sources away from the assembly area. **D:** Lead supervises all motor assembly operations and checks for static or spark risks. **M:** Keep fire extinguishing equipment nearby; administer burn first aid if needed.	1A	Verified
PHZ.44	Falling from Elevated Positions (e.g., Ladders)	**(1)** Use of ladders or elevated platforms to arm devices or perform tasks **(2)** Improper ladder setup, unstable footing, or overreaching	Bruises, sprains, or fractures	3B	**P:** Ensure proper ladder technique is followed and stable footing maintained. **D:** Inspect ladder legs for security before use. **M:** Provide first aid for bruises and minor injuries; seek medical attention if needed.	1A	Verified
Hazards Encountered at STEM Engagement Event							

Table 6.6: Personnel Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
PHZ.45	Launch Vehicle Falling	**(1)** Failure to support or anchor the launch vehicle **(2)** Horseplay or unsafe behavior around launch vehicles	Impact injuries to personnel from falling rockets	2C	**P:** Use stable launch vehicle stands; prohibit horseplay; secure stands before presentations. **D:** Inspect stands before use; do not use damaged or broken stands. **M:** Catch any falling launch vehicles if possible; provide first aid for any injuries.	2A	Verified
PHZ.46	Unintended Ignition of Estes Motor	**(1)** Damaged or faulty motor **(2)** Improper handling or accidental activation during preparation **(3)** Presence of ignition sources near the motor	**(1)** Severe burns to personnel **(2)** Fire or ignition of surrounding materials **(3)** Eye and ear injuries	3C	**P:** Follow manufacturer instructions; keep ignition sources away during assembly; inspect motors for potential defects. **D:** Ensure children remain at a safe distance; perform visual and continuity checks before connecting ignitors. **M:** Use fire extinguishers; report incidents to supervisors; provide burn first aid if needed.	1A	Verified
PHZ.47	Estes or Bottle Rocket Striking Personnel	**(1)** Incorrect launch angle or trajectory **(2)** Improper stabilization of the rocket on the launch pad **(3)** Launching too close to personnel or crowded areas	Impact injuries, including bruises, fractures, or blunt trauma	3C	**P:** Maintain safe distances; provide stable launch equipment; enforce clear launch zones. **D:** Confirm launch pad alignment away from personnel; provide proper countdowns to prepare personnel. textbfM: Provide medical aid for impact injuries; report incidents to supervisors.	2B	Verified
PHZ.48	Smoke from Estes Motors	**(1)** Ignition of motor propellant producing smoke or particulate matter **(2)** Incomplete combustion of motor fuel **(3)** Wind carrying smoke toward personnel or work areas	**(1)** Respiratory irritation or difficulty breathing **(2)** Eye irritation from smoke or airborne particles **(3)** Reduced visibility	2C	**P:** Position personnel upwind; maintain safe distances. **D:** Observe smoke direction; monitor personnel for irritation. **M:** Move personnel upwind; relocate to fresh air and allow lungs to recover.	1A	Verified

Table 6.6: Personnel Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
PHZ.49	Rocket (bottle, Estes, straw) Becomes Stuck in Trees or Vegetation	**(1)** Wind direction or gusts carrying rocket toward trees or vegetation **(2)** Launching too close to treelines or improperly aimed trajectory	**(1)** Damage to trees or vegetation **(2)** Environmental pollution from rocket materials left in nature	2B	**P:** Launch away from treelines and tall vegetation. **D:** Track rocket trajectory carefully. **M:** Carefully retrieve the rocket without damaging the environment; prevent children from attempting retrieval.	1A	Verified
Hazards to Project Schedule							
PHZ.50	Vehicle Lead Becomes Unavailable	Unforeseen personal circumstances, illness, or scheduling conflicts	**(1)** Potential delays in vehicle structure, recovery system, or air brake system tasks **(2)** Risk of cascading schedule impacts on payload integration or launch readiness **(3)** Increased workload for remaining team members, potentially affecting quality or safety	3C	**P:** Cross-train team members; prepare design and schedule plans early; maintain updated documentation of manufacturing or analysis methods. **D:** Provide progress updates; keep leadership aware of individual workloads. **M:** Redistribute responsibilities among trained members; adjust timelines; seek mentor support if needed.	2B	Not Verified
PHZ.51	Payload Lead Becomes Unavailable	Unforeseen personal circumstances, illness, or scheduling conflicts	**(1)** Incomplete or delayed payload design and development **(2)** Potential schedule impacts on vehicle integration and testing **(3)** Increased risk of errors or oversights due to redistributed workload among remaining team members	3C	**P:** Cross-train team members; prepare design and schedule plans early; maintain updated documentation of manufacturing or analysis methods. **D:** Provide progress updates; keep leadership aware of individual workloads. **M:** Redistribute responsibilities among trained members; adjust timelines; seek mentor support if needed.	2B	Not Verified

Table 6.6: Personnel Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
PHZ.52	Team Lead or Integration Personnel Becomes Unavailable	Unforeseen personal circumstances, illness, or scheduling conflicts	**(1)** Reduced communication and coordination across subsystems **(2)** Poor overall project coordination, potentially causing schedule delays or design conflicts **(3)** Increased risk of errors or misalignment between vehicle, payload, and recovery systems	3C	**P:** Develop leadership plans; cross-train other leads on integration practices; maintain clear communication. **D:** Track leadership responsiveness and schedules; document responsibilities. **M:** Appoint interim leadership personnel; re-evaluate timelines and responsibilities.	2B	Not Verified
PHZ.53	Key Officer (Safety, Treasurer, Outreach) Becomes Unavailable	Unforeseen personal circumstances, illness, or scheduling conflicts	**(1)** Decrease in project funding management, oversight, or allocation **(2)** Potential lapses in safety protocols or risk management **(3)** Reduced effectiveness of community outreach and engagement activities	3C	**P:** Train deputies; document safety, financial procedures, and outreach templates. **D:** Keep updates on officer progress; monitor workloads. **M:** Redistribute work; elect new officers if needed; adjust timelines.	2B	Verified
PHZ.54	Insufficient Team Member Overlap (No Redundancy)	**(1)** Lack of cross-training among team members for critical systems **(2)** Dependence on single individuals for specific tasks	**(1)** Important tasks delayed if a team member is unavailable **(2)** Potential loss of quality or errors in project deliverables **(3)** Increased workload and stress on remaining team members, possibly affecting safety and performance	4C	**P:** Cross-train members across subsystems; maintain proper documentation; create a shared knowledge culture. **D:** Identify single points of failure. **M:** Redistribute work; adjust deadlines if needed.	2B	Verified
PHZ.55	Member Fatigue or Burnout	**(1)** Large workload without sufficient breaks or rest periods **(2)** Lack of recognition or support for team contributions **(3)** Extended periods of high stress or repetitive tasks	**(1)** Reduced quality or accuracy of project work **(2)** Loss of key team members due to disengagement or withdrawal	3C	**P:** Enforce reasonable workloads; allow rest periods; recognize contributions; share heavy responsibilities. **D:** Observe performance declines or absenteeism; provide space for honest feedback. **M:** Encourage rest; provide morale support; adjust workloads.	1A	Verified

Table 6.6: Personnel Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
PHZ.56	Elevated Stress or Anxiety	**(1)** High schedule pressure or tight deadlines **(2)** Unclear expectations or poorly defined responsibilities **(3)** Lack of support or communication within the team	**(1)** Reduced quality or accuracy of project work **(2)** Decision-making errors or poor judgment	3D	**P:** Set clear expectations; ensure workloads are balanced; establish achievable milestones. **D:** Observe signs of stress or reduced performance. **M:** Provide a supportive environment; offer breaks; redistribute tasks; recognize efforts.	1B	Verified
PHZ.57	Team Conflict or Poor Interpersonal Dynamics	**(1)** Miscommunication or unclear expectations among team members **(2)** Lack of conflict resolution or mediation mechanisms **(3)** Differences in work styles or priorities without structured collaboration	**(1)** Lower productivity and efficiency **(2)** Reluctance to collaborate, share information, or assist peers	3B	**P:** Set behavior expectations; establish outlets for conflict resolution. **D:** Track interpersonal issues that could escalate. **M:** Facilitate mediation; remain neutral; treat all sides with respect; refocus on shared goals.	1A	Verified
PHZ.58	Emotional Exhaustion Following Failure	**(1)** Setbacks or project failures without structured debriefing or reflection **(2)** Lack of emotional support	**(1)** Drop in team morale and motivation **(2)** Emergence of blame culture or interpersonal tension	3B	**P:** Establish post-launch debriefing. **D:** Monitor morale after setbacks; observe engagement. **M:** Provide morale support; reassure team members of priorities.	2A	Verified
PHZ.59	Mental Health Degradation	**(1)** Unawareness of available mental health resources **(2)** Stigma or reluctance to seek support **(3)** Lack of proactive outreach	**(1)** Increased stress, anxiety, or emotional fatigue **(2)** Decline in performance, productivity, or quality of work	4B	**P:** Promote awareness of campus mental health resources; encourage work-life balance. **D:** Observe signs of stress, fatigue, or withdrawal. **M:** Offer check-ins; guide those seeking help to resources; adjust expectations.	2A	Verified

Table 6.6: Personnel Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
PHZ.60	Leadership Insensitivity to Team Members' Well being	**(1)** Emphasis on deliverables over personnel needs **(2)** Dismissive or unresponsive attitudes toward team concerns	**(1)** Loss of trust and respect for leadership **(2)** Decreased team cohesion and stability	3A	**P:** Create an open dialogue; emphasize people over deadlines. **D:** Monitor for dismissive attitudes from leaders; listen to observations from team members. **M:** Coach insensitive leaders; recognize team member contributions.	1A	Verified

6.4 Design Hazards Analysis

Design hazard analyses were conducted to identify potential hazards that the launch vehicle might encounter. First potential hazards were identified across all subsystems, including propulsion, structures, Air Brakes, Aerodynamics, and Recovery. Hazards were also identified for the preliminary payload design, including structural, electronics, and contamination hazards. Integration hazards were identified for all previous subsystems and the payload. Finally, launch hazards were identified, categorized into launch support equipment and launch operation hazards. Each hazard was then evaluated for its causes, possible effects on design safety, and the likelihood and severity of these effects. Appropriate mitigation strategies were developed to minimize or eliminate risks, ensuring a safe working environment and compliance with safety requirements.

Table 6.7 shows the design hazards before mitigation, and Table 6.8 shows the design hazards after mitigation.

Table 6.7: Design Risks Assessment Before Mitigation

		Level of Severity			
		1 Low Risk	2 Medium Risk	3 High Risk	4 Severe Risk
Likelihood of Occurrence	A Very Unlikely	0.0% (0)	0.0% (0)	1.32% (1)	9.21% (7)
	B Unlikely	0.0% (0)	2.63% (2)	9.21% (7)	32.89% (25)
	C Likely	0.0% (0)	5.24% (4)	11.84% (9)	25% (19)
	D Very Likely	0.0% (0)	1.32% (1)	1.32% (1)	0.0% (0)

Table 6.8: Design Risks Assessment After Mitigation

		Level of Severity			
		1 Low Risk	2 Medium Risk	3 High Risk	4 Severe Risk
Likelihood of Occurrence	A Very Unlikely	47.37% (36)	31.58% (24)	1.32% (1)	5.24% (4)
	B Unlikely	6.58% (5)	5.24% (4)	2.63% (2)	0.0% (0)
	C Likely	0.0% (0)	0.0% (0)	0.0% (0)	0.0% (0)
	D Very Likely	0.0% (0)	0.0% (0)	0.0% (0)	0.0% (0)

Table 6.9: Design Hazards

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
Launch Vehicle Hazards							
Propulsion System Hazards							
DHZ.1	Overpressure Explosion	**(1)** Incorrect assembly of RMS hardware **(2)** Missing or improperly installed forward seal disc **(3)** Failure to follow proper assembly or inspection procedures	**(1)** Catastrophic motor failure **(2)** Severe damage to the vehicle structure	4B	**P:** Follow manufacturer instructions for motor assembly. **D:** Ensure mentor and qualified member are present during assembly; inspect and confirm O-rings, seal discs, and all closures are properly installed. **M:** Maintain fire suppression methods; maintain safe distances until RSO gives approval to approach the launch vehicle.	4A	Verified
DHZ.2	Igniter Failure	**(1)** Damaged, defective, or incorrect type of initiator **(2)** Improper handling or storage of igniters	**(1)** Failure of the motor or rocket to launch as intended **(2)** Reduces motor effectiveness because of wasted fuel.	3B	**P:** Properly store igniters; handle with clean hands. **D:** Check for continuity before arming; inspect for visible physical defects. **M:** Replace igniter and use backups until ignition success is achieved.	1A	Verified
DHZ.3	Propellant Contamination	Improper storage or handling (exposure to moisture, oils, or foreign particulates)	**(1)** Unpredictable burn behavior **(2)** Reduced motor performance and reliability **(3)** Increased risk of motor overpressure, casing rupture, or catastrophic failure	4A	**P:** Store propellant in manufacturer container until use; inspect all O-rings for wear or deformation. **D:** Verify propellant looks unaltered; no discoloration or noticeable wear. **M:** Safely dispose of contaminated grains; use backup motors as needed.	1A	Verified
DHZ.4	Unintended Ignition	Presence of heat sources, open flames, or electrical discharge in proximity to motor assembly	**(1)** Severe thermal burns to personnel **(2)** Potential ignition of surrounding materials leading to fire **(3)** Risk of ear and eye injury	4C	**P:** Assemble motors away from ignition sources. **D:** Inspect work area and confirm no ignition sources are nearby. **M:** Evacuate area; fight fire using water instead of CO2.	4A	Verified

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.5	Motor Retention Failure	**(1)** Motor retention system breaks due to material fatigue or damage **(2)** Improper installation or attachment of the retention system	**(1)** Motor falls out of the launch vehicle after burnout **(2)** Potential impact injuries to personnel or property from falling motor	4B	**P:** Verify motor retention system during launch checklists. **D:** Visually inspect retention hardware; replace damaged components; inspect after launch. **M:** Maintain safe distances; monitor location of motor; retrieve after RSO deems safe.	2A	Verified
Structural Hazards							
DHZ.6	Composite Components Fail Under Loading	**(1)** Miscalculated composite layup or structural design errors **(2)** Use of defective or improperly cured composite materials **(3)** Poor quality control during fabrication or assembly	**(1)** Launch vehicle structural damage or partial failure **(2)** Reduced reusability of vehicle components	4C	**P:** Perform multiple calculations to verify correct expected loading. **D:** Conduct mechanical testing on test pieces to verify material can withstand expected loading. **M:** Reinforce or replace failed components; recalculate and review fabrication process for improvements.	2A	Verified
DHZ.7	Fin Flutter	**(1)** Aerodynamic forces interacting with structural vibrations of fins **(2)** Insufficient stiffness or improper fin attachment	**(1)** Fin failure or detachment during flight **(2)** Loss of vehicle stability and control	4B	**P:** Design fins to withstand flutter; use stiff materials expected to handle loading. **D:** Inspect fin bonds to ensure structural integrity. **M:** Recover debris, recalculate flutter analysis assumptions, and redesign as needed.	1A	Not Verified
DHZ.8	Fin Fracture	Heavy landing impacts or rough recovery	**(1)** Launch vehicle becomes aerodynamically unstable **(2)** Vehicle may be unable to fly or maintain controlled flight	4C	**P:** Reinforce fin roots to airframe; limit descent rate using parachutes; use impact-resistant materials. **D:** Inspect post-flight vehicle for fractures, cracks, or delamination. **M:** Replace damaged fins and add reinforcement.	1A	Verified

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.9	Fin Layup Delamination	**(1)** Insufficient or improper composite layups during fabrication **(2)** Inadequate bonding or curing of layers **(3)** Lack of inspection or quality control during assembly	Poor local stiffness of fins, leading to structural weakness	4C	**P:** Follow layup procedures; ensure full wetting of fibers; use vacuum bagging. **D:** Inspect for visible delaminations; remanufacture if found. **M:** Repair negligible delaminations; replace fins if needed.	2A	Verified
DHZ.10	Composite Fiber Misalignment	Insufficient or incorrect layup placement during composite fabrication	**(1)** Reduced load-carrying capacity of composite components **(2)** Unexpected bending, shear failure, or structural deformation under load	3C	**P:** Follow fiber orientation procedures; verify alignment with multiple members. **D:** Inspect each layer before laminating to ensure correct fiber orientation. **M:** Repurpose misaligned parts.	1A	Verified
DHZ.11	Bulkhead Cracking	**(1)** Excessive stress concentration during parachute deployment **(2)** Use of insufficiently reinforced or damaged bulkhead materials	**(1)** Parachutes may disconnect from the launch vehicle **(2)** Portions of the vehicle may enter uncontrolled ballistic trajectories	4C	**P:** Analyze loads expected from parachute deployment and motor thrust; use stiff bulkhead materials. **D:** Monitor bulkheads before and after launch; note any deformities. **M:** Replace damaged bulkheads; refine fabrication and reinforce attachment points.	2A	Verified
DHZ.12	Bulkheads Burn	**(1)** Incorrect black powder packing or overloading **(2)** Improper assembly or failure to follow safe procedures	**(1)** Damage to bulkheads, compromising structural integrity **(2)** Broken seals potentially damaging avionics or internal components	4B	**P:** Do not overuse black powder for separation; insulate bulkheads; use flame-resistant materials. **D:** Visual inspection after launch and ejection testing to verify burn marks. **M:** Replace damaged bulkheads.	2A	Verified

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.13	Improper Bulkhead Stepping	Incorrect layup technique or measurement errors during fabrication	**(1)** Bulkheads too large, undersized, or misaligned within the airframe **(2)** Difficulty during assembly or poor fitment between sections	3C	**P:** Verify bulkhead dimensions; use precise measuring tools and templates. **D:** Dry fit before bonding. **M:** Sand or trim oversize bulkheads; refabricate undersized ones; repurpose old bulkheads.	2A	Verified
DHZ.14	Bulkhead Not Properly Sealed	**(1)** Unsecured or improperly installed Waygo terminals, U-bolts, or pass-through fittings **(2)** Inadequate sealing materials	**(1)** Black powder gases or residue entering the avionics bay during ejection events **(2)** Potential damage or contamination of altimeters and sensitive electronics **(3)** Loss of flight data or deployment control reliability textbf(4) Reduced reusability of avionics and risk of mission failure	4B	**P:** Apply sealant as needed; test during ejection. **D:** Inspect for soot or residue post-flight and post-ejection testing. **M:** Clean avionics bay; replace damaged electronics; reseal bulkhead and retest ejection.	2B	Verified
DHZ.15	Airframe Delamination	**(1)** Insufficient resin wetting or uneven resin distribution during layup **(2)** Poor compaction	**(1)** Localized loss of stiffness and structural integrity **(2)** Increased susceptibility to cracking or buckling under aerodynamic or landing loads	4C	**P:** Maintain tight rolling and proper wetting techniques. **D:** Inspect for visible delaminations. **M:** Replace sections as needed.	2A	Verified
DHZ.16	Resin-Rich or Resin-Poor Regions in Composites	**(1)** Incorrect resin-to-fiber ratio during layup or infusion **(2)** Inconsistent resin distribution caused by poor mixing, application, or vacuum bagging technique	**(1)** Formation of weak spots prone to cracking or delamination under load **(2)** Reduced structural performance and uneven stress distribution	4C	**P:** Weigh resin and fiber prior to layups; verify proper fiber-to-resin ratio. **D:** Inspect for over- or under-saturated regions. **M:** Repair negligible defects; remake composites if necessary.	1A	Verified
DHZ.17	Incomplete Curing of Composite Materials	**(1)** Incorrect curing temperature or insufficient curing duration **(2)** Improper epoxy-to-hardener ratio during mixing	**(1)** Compromised structural integrity and reduced mechanical strength **(2)** Increased risk of delamination, soft spots, or deformations	4C	**P:** Follow manufacturer instructions for ratios and curing temperatures. **D:** Inspect composites for tackiness or soft spots. **M:** Refabricate incompletely cured components.	1A	Verified

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.18	Airframe Zippering	Excessive snatch force from rapid parachute deployment or taut shock cords	**(1)** Longitudinal splitting of the airframe at separation points **(2)** Structural damage preventing vehicle reusability	4C	**P:** Verify shock cord selection follows RD 6 guidelines. **D:** Inspect airframe post-flight for tearing. **M:** Refabricate damaged sections; recalculate for proper loading.	2A	Verified
Air Brakes System Hazards							
DHZ.19	Asymmetric Air Brakes Deployment	Improper air brakes assembly or actuator misalignment	**(1)** Induced asymmetric aerodynamic moments causing vehicle instability **(2)** Unpredictable flight trajectory and potential deviation from safe flight path	4A	**P:** Verify that air brakes deploy mechanically and simultaneously on all sides. **D:** Inspect assembly for asymmetries prior to flight. **M:** Reassemble the fin system; sand components for proper fit or reprint fins if necessary.	1A	Verified
DHZ.20	Air Brakes Fail to Deploy or Retract	**(1)** Improperly sized fin slots **(2)** Excessive friction in air brakes gears	Launch vehicle fails to reach intended target apogee	4C	**P:** Ensure slot tolerances meet design specifications; verify fins move freely with minimal friction. **D:** Perform bench tests with the airframe to confirm deployment and retraction. **M:** File or adjust fin slots until air brake fins deploy and retract smoothly.	1B	Verified
DHZ.21	Air Brakes Prediction Algorithm Incorrect	Software errors or logic flaws in apogee prediction algorithm	**(1)** Air brakes deploy at the wrong time, too early or too late **(2)** Launch vehicle fails to reach intended target apogee	4B	**P:** Validate apogee prediction algorithm through simulations and test flights. **D:** Log and review flight data from test launches to verify algorithm accuracy. **M:** Reevaluate and correct algorithm errors as identified.	4A	Verified

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.22	Inaccurate Barometric Pressure Sensor Data	Air brakes flight computer not properly sealed	**(1)** Air brakes fins may retract prematurely or fail to deploy correctly **(2)** Launch vehicle fails to reach intended target apogee	4B	**P:** Ensure the flight computer is properly sealed to prevent pressure data corruption. **D:** Test sealing integrity using vacuum or pressure testing. **M:** Analyze flight data; redesign sealing methods if pressure inconsistencies are detected.	2A	Not Verified
Aerodynamics/Stability Hazards							
DHZ.23	Launch Vehicle Over-Stability	Inaccurate mass distribution or miscalculated center of gravity during simulation	**(1)** Weathercocking during ascent **(2)** Increased drift distance from intended landing zone **(3)** Undershoot intended apogee.	2B	**P:** Maintain a documented mass list and update stability analyses as component masses change. **D:** Monitor and record how the vehicle's mass properties evolve during integration. **M:** Add ballast to the aft section of the launch vehicle until the stability margin is within the acceptable range.	1A	Not Verified
DHZ.24	Launch Vehicle Under-Stability	Inaccurate mass distribution or miscalculated center of gravity during simulation	Oscillation, coning, or tumbling of the vehicle during ascent	4B	**P:** Maintain a documented mass list and update stability analyses as component masses change. **D:** Monitor and record how the vehicle's mass properties evolve during integration. **M:** Add ballast to the forward section of the launch vehicle until the stability margin meets NASA requirement 2.11.	2A	Not Verified

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.25	Assembled Launch Vehicle Exceeds Anticipated Mass	Underestimation of payload or vehicle component mass during design	Launch vehicle fails to reach target apogee, falling below NASA-specified range	3C	**P:** Maintain a detailed bill of materials with updated component masses throughout development. **D:** Weigh every component within each subsystem to confirm accuracy against design models. **M:** Reduce unnecessary weight where possible; if not feasible, redesign components to meet mass constraints.	2B	Not Verified
Recovery System Hazards							
DHZ.26	Parachute Fails to Deploy	**(1)** Improper packing technique **(2)** Insufficient black powder charge to separate the launch vehicle	Launch vehicle enters ballistic descent	4C	**P:** Train members on proper parachute packing; use dual altimeters for redundancy. **D:** Check for continuity on deployment charges; lead inspects parachutes for correct packing. **M:** Maintain visual tracking during descent; do not recover until cleared by the RSO.	4A	Not Verified
DHZ.27	Main Parachute Deploys Early	**(1)** Altimeter failure or incorrect pressure readings **(2)** Pressure buildup causing premature separation **(3)** Improper avionics bay assembly	**(1)** Extended descent time, failing to meet NASA rule 2.1 . **(2)** Increasing drift distance	3C	**P:** Ground test altimeters; ensure pressure ports are properly vented. **D:** Verify arming sequence in altimeter programming. **M:** Review flight data; retest or replace altimeters as needed.	3A	Verified
DHZ.28	Main Parachute Deploys Late	Altimeter failure	**(1)** Launch vehicle descends with excessive kinetic energy **(2)** Failure to meet NASA requirement 3.1.2	4C	**P:** Ground test altimeters; ensure pressure port holes are properly vented; confirm shear pins are correctly installed. **D:** Verify arming sequence in altimeter programming. **M:** Inspect and replace damaged altimeters; relaunch at backup opportunity.	2A	Verified

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.29	Motor Ejection Deploys	Improper motor assembly	**(1)** Damage to air brakes system **(2)** Premature separation of vehicle sections **(3)** Failure to meet NASA requirement 3.1.3	4B	**P:** During motor assembly, ensure motor ejection charges are not installed. **D:** Lead inspects motor assembly and verifies absence of ejection charges. **M:** Internal bulkhead design prevents black powder intrusion into the air brakes module.	1A	Verified
DHZ.30	Launch Vehicle Sections Collide During Ascent	Insufficient shock cord spacing	Damage to vehicle sections	3B	**P:** Calculate shock cord lengths in accordance to RD 6; verify proper routing and recovery attachments. **D:** Visually inspect assembled vehicle and confirm separation distances between sections. **M:** Inspect sections for damage post-flight; replace if necessary; recalculate shock cord lengths.	1A	Verified
DHZ.31	Insufficient Black Powder in Charge Wells	**(1)** Miscalculated black powder amounts **(2)** Lack of ejection testing	**(1)** Vehicle sections fail to separate **(2)** Launch vehicle enters ballistic descent	4B	**P:** Calculate ejection charges with a factor of safety. **D:** Conduct ground ejection tests to confirm full separation of all recovery components. **M:** Fly redundant ejection charges with a factor of safety of at least 1.5.	1A	Verified
DHZ.32	Black Powder Fails to Ignite	Overpressurization of charge well causing cap to separate before all black powder ignites	Launch vehicle enters ballistic descent	4C	**P:** Secure tape firmly around charge wells to ensure full pressurization before ignition. **D:** Test continuity of ematches; use reliable, high-tack tape rather than electrical tape. **M:** Fly redundant ejection charges with a factor of safety of at least 1.5.	1A	Verified

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.33	In-House Made Parachutes Fail to Deploy	Improper manufacturing technique	**(1)** Launch vehicle descends too quickly **(2)** Excessive forces may cause main parachute failure	4B	**P:** Follow validated stitching patterns and proven canopy designs. **D:** Load test seams to confirm expected tensile strength. **M:** Inspect any damaged parachutes and refabricate using improved methods.	1B	Verified
DHZ.34	Shroud Lines Break or Snap	Snatch force exceeds line strength	**(1)** Parachute failure **(2)** Launch vehicle enters ballistic descent **(3)** Failure to meet NASA requirement 3.2.	4B	**P:** Select line materials with verified tensile strength exceeding expected loads. **D:** Static-load test parachutes; ensure even load distribution among lines. **M:** Reinforce attachment points and redesign using stronger materials.	1A	Verified
DHZ.35	Parachute Deploys Inside-Out	**(1)** Packing technique does not allow for parachute to deploy correctly **(2)** Fin can has more drag than drogue parachute	**(1)** Launch vehicle descends faster than intended **(2)** Failure to meet NASA requirement 3.2	4C	**P:** Follow standard packing procedures from a trusted vendor or documented method. **D:** Conduct ground deployment tests prior to flight. **M:** Review and retrain on proper packing procedures.	3B	Verified
Electrical/Avionics Hazards							
DHZ.36	Altimeters Fail to Send Proper Current to Igniters	**(1)** Continuity loss in wiring **(2)** Unexpected resistance in wiring	**(1)** Launch vehicle fails to separate **(2)** Vehicle enters ballistic descent	4A	**P:** Perform preflight continuity tests; use altimeters from trusted manufacturers. **D:** Test altimeters in simulated flight environments. **M:** Use dual altimeters to provide redundancy.	2A	Verified

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.37	Batteries Depleted at Launch	Improper charging procedure or failure to charge batteries	**(1)** Electronic systems fail, including altimeters and deployment circuits **(2)** Launch vehicle enters ballistic descent **(3)** Air brakes system fails to reduce apogee to target	4B	**P:** Establish night-before charging checklists; replace batteries after manufacturer-recommended usage limits; pack backup batteries. **D:** Measure battery voltage prior to launch vehicle integration. **M:** Use dual altimeters to ensure system redundancy.	1A	Verified
DHZ.38	Electromagnetic Interference	**(1)** Nearby transmitting devices **(2)** Poor cable shielding or routing	False triggers, inaccurate telemetry, missing deployment commands	4B	**P:** Route signal and power cables separately; apply conductive shielding to deployment electronics. **D:** Inspect avionics bay for proper shielding and cable routing prior to flight. **M:** Use dual altimeters to ensure redundant deployment logic.	1A	Not Verified
DHZ.39	Connector Misalignment	Improper seating or connector manufacturing	**(1)** False triggers or missed deployment commands **(2)** Inaccurate telemetry or sensor readings	3C	**P:** Train team members on proper connector assembly and handling. **D:** Check continuity and resistance across connectors before integration. **M:** Use dual altimeters for redundant deployment confirmation.	2A	Verified
Payload Hazards							
Structural Hazards							
DHZ.40	Lander Loses Balance After Self-Righting	Uneven mass distribution	**(1)** Lander tips over **(2)** Inability to collect soil samples	3C	**P:** Conduct center-of-gravity analysis and balance testing during design. **D:** Perform physical balance tests post-assembly; verify self-righting dynamics through ground tests. **M:** Design leg geometry to self-stabilize under minor imbalance conditions.	2B	Not Verified

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.41	**(1)** Payload Body Fracture **(2)** Damage to internal payload electronics	**(1)** 3D-printed enclosure delaminates under load **(2)** Poor print quality or insufficient infill	Loss of structural integrity	4B	**P:** Use high-quality 3D printing processes and stiff filament materials. **D:** Inspect prints for delamination, voids, or under-extrusion. **M:** Replace damaged sections with reinforced or redesigned components.	1A	Not Verified
DHZ.42	Leg or Leg Hinges Shearing	**(1)** Improper aluminum thickness or weak material choice **(2)** Unanticipated loading at hinges during landing	**(1)** Loss of one or more legs **(2)** Failure of the lander to upright after landing	4C	**P:** Select hinge materials and thicknesses based on calculated landing loads. **D:** Validate material performance via drop tests. **M:** Replace damaged hinges; reinforce hinge joints as needed.	3B	Not Verified
DHZ.43	Collar Jams	**(1)** Debris intrusion in hinge or collar mechanism **(2)** Lack of lubrication or poor tolerance control	Legs fail to deploy fully	3B	**P:** Maintain clean hinge and collar assemblies; implement dust control during integration. **D:** Manually cycle the mechanism before integration; verify smooth deployment during test runs. **M:** Redesign mechanism with integrated dust guards.	2A	Not Verified
DHZ.44	Auger Bit Breaks	**(1)** Torsional overloading during soil collection **(2)** Use of insufficiently reinforced material or incorrect speed	**(1)** Incomplete soil collection **(2)** Potential damage to the motor or drive system	4B	**P:** Use torque-limited motors to prevent overload. **D:** Perform pre-flight torque checks and inspect materials for cracks or wear. **M:** Reinspect motor components; redesign or strengthen auger bit as required.	2A	Not Verified
DHZ.45	Latch Doesn't Open	Servo malfunction or electrical failure	**(1)** Lander trapped in the nose cone **(2)** Inability to deploy legs or collect soil	4B	**P:** Validate servo operation during integration. **D:** Conduct deployment tests under simulated flight conditions. **M:** Inspect failure points; re-fly at backup launch after correction.	1A	Not Verified

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.46	Lead Screw Binding	**(1)** Misalignment of components **(2)** Insufficient lubrication of threads	Lander fails to eject	4B	**P:** Ensure precise alignment during assembly; apply thread lubricant. **D:** Cycle mechanism during ground tests to confirm smooth travel. **M:** Inspect lead screw for binding; redesign or replace screw if necessary.	2A	Not Verified
DHZ.47	Soil Enclosure Cracking	Hard impact with the ground during landing	Insufficient soil collected	3C	**P:** Use high-quality 3D prints with stiff filament material. **D:** Inspect prints for delamination, voids, or under-extrusion. **M:** Replace broken enclosures with improved designs or stronger materials.	1A	Not Verified
DHZ.48	Guide Rails Become Disconnected	**(1)** Improper adhesion or mechanical fasteners failing **(2)** Poor alignment during assembly	Lander does not eject smoothly	4A	**P:** Ensure proper epoxy bonding via surface preparation. **D:** Perform visual inspection pre-launch and vibration/pull testing. **M:** Reinforce guide rail joints; redesign using mechanical fasteners.	1A	Not Verified
DHZ.49	3D Prints Warping	Overheating during payload operations	**(1)** Warped structural elements **(2)** Compromised alignment or mounting of payload components **(3)** Reduced structural integrity and potential payload malfunction	4B	**P:** Select heat-resistant filament rated for expected temperatures. **D:** Inspect geometry post-fabrication. **M:** Replace warped parts; redesign with thermal management or cooling considerations.	1B	Not Verified
Electronics Hazards							

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.50	Flight State Misidentified	**(1)** Sensor miscalibration **(2)** Incorrect threshold logic in software	**(1)** Premature deployment **(2)** Incorrect sequencing of payload operations	4B	**P:** Calibrate all sensors before integration; verify calibration against known reference conditions. **D:** Log sensor inputs during ground testing and flight to validate threshold logic and state detection. **M:** Review and correct flight-state logic following post-flight data analysis; revalidate through bench tests.	2A	Not Verified
DHZ.51	Communication Errors Between Sensor and Pi	Incompatible communication protocols between the Modbus and Raspberry Pi	Inability for the payload to collect or transmit soil data	4B	**P:** Verify Modbus–Raspberry Pi compatibility during design phase; use standardized communication libraries. **D:** Conduct communication handshake tests during integration to confirm reliable data transmission. **M:** Buffer data locally for later retrieval.	2A	Not Verified
DHZ.52	Loss of Power	**(1)** Loose connector **(2)** Improper power distribution **(3)** Faulty solder joints	Payload unable to collect data, drive motors, or perform drilling operation	4B	**P:** Inspect all connections and solder joints; use proper wire gauge for components. **D:** Use a multimeter to test battery output and verify continuity across all power lines before integration. **M:** Re-inspect and repair any failed connections; prepare system for reflight once power reliability is verified.	1A	Not Verified
Contamination Hazards							

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.53	Soot on Sensors	Improper sealing from ejection charges	**(1)** False measurements **(2)** Electronic failures	3C	**P:** Ensure payload electronics and sensors are properly sealed from ejection gases. **D:** Inspect sensors after ejection charge testing for any soot or residue accumulation. **M:** Clean sensors post-flight, inspect for damage, and replace compromised components; redesign sealing method if recurring contamination occurs.	2A	Not Verified
DHZ.54	Soot on Moving Parts (Gears, Lead Screw, Rack and Pinion, etc.)	Improper sealing from ejection charges	**(1)** Mechanical jamming **(2)** Payload unable to drill, self-right, or deploy	2C	**P:** Design mechanical assemblies with seals or enclosures to prevent soot intrusion into moving parts. **D:** Inspect mechanisms post-flight and document areas showing soot or residue buildup. **M:** Disassemble and clean contaminated components; reapply lubricant or protective coating as necessary.	1A	Not Verified
DHZ.55	Soil Sensor Probes Contaminated	Residual dust or dirt from prior testing	Soil sensor misreading, inaccurate data collection	3B	**P:** Thoroughly clean soil probes between tests to remove all dust or debris. **D:** Inspect probes during pre-flight checklist to ensure no residue or damage from prior use. **M:** Wipe probes after each flight and store in protective containers to prevent future contamination.	2A	Not Verified
Integration Hazards							
Payload Integration Hazards							

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.56	Payload Fails to Sit on Racks	**(1)** Improper payload dimensions **(2)** Debris or foreign objects in rack tracks	Lander fails to eject smoothly	3B	**P:** Verify payload dimensions and tolerances; apply lubricant to rack tracks if necessary. **D:** Perform dry-fit tests of payload on racks; if misalignment is observed, redesign for proper fit. **M:** Remove payload and clean tracks, ensuring all debris is cleared before reassembly.	1A	Not Verified
DHZ.57	Pusher Plate Deforms Under Loads	**(1)** Insufficient stiffness or inadequate material thickness **(2)** Excessive ejection forces	Lander ejects improperly or at an angle	3B	**P:** Fabricate pusher plate from sufficiently stiff material with appropriate thickness. **D:** Measure plate deflection during ground testing to ensure deformation remains within tolerance. **M:** Reinforce plate or redesign using a stiffer material if deformation exceeds limits.	2A	Not Verified
DHZ.58	Payload Shifting Center of Gravity	Unanticipated payload mass gain	Altered vehicle stability	2C	**P:** Maintain a payload mass log, updating entries as fabrication progresses. **D:** Re-measure the center of gravity during final assembly and integration. **M:** Update simulations and add ballast to the aft end of the launch vehicle as necessary to rebalance stability.	1A	Not Verified

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.59	Parachute Re-Inflates Upon Landing	Wind gusts	**(1)** Payload dragged across the ground if still connected to nose cone **(2)** Damage to payload housing or legs	3D	**P:** Program the state machine to detect landing and deactivate systems that may allow parachute drag. **D:** Conduct ground tests to verify state machine behavior under simulated drag conditions. **M:** Inspect payload post-flight for drag-related damage and reinforce vulnerable components if needed.	2A	Not Verified
Recovery Integration Hazards							
DHZ.60	Parachute Does Not Fit Into Airframe/Fin Can	**(1)** Miscommunication of dimensions between Structures and Recovery subteams **(2)** Failure to perform a full test fit prior to launch	**(1)** Launch vehicle cannot accommodate recovery system **(2)** Flight must be canceled due to lack of recovery capability	4A	**P:** Maintain clear communication between Structures and Recovery subteams regarding airframe dimensions and packed parachute sizes. **D:** Confirm fit via dry fit testing of parachute in airframe. **M:** Repack parachutes, ensuring they properly fit within the airframe before launch.	1A	Verified
DHZ.61	Ejection Gasses Escape the Airframe	Improper or incomplete seal around avionics bay sections	**(1)** Failure to separate airframe sections **(2)** Launch vehicle descends ballistically	4C	**P:** Ensure proper sealing during assembly and verify with ejection testing. **D:** Perform ejection tests and inspect for gas leaks; apply sealant or lubricant as needed. **M:** Fly redundant ejection charges with a factor of safety of 1.5 to ensure separation.	1A	Verified

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.62	Main and Drogue Parachutes Installed in Wrong Sections	Miscommunication of forward/aft bay configuration	**(1)** Main parachute deploys at apogee **(2)** Increased descent time and drift distance	3A	**P:** Clearly label main and drogue parachutes and their respective airframe sections. **D:** Perform dry runs to verify proper placement before flight. **M:** Reassess and update checklist procedures to ensure parachutes are installed in the correct sections.	1A	Verified
Structural Integration Hazards							
DHZ.63	Arming Hole in Switchband Does Not Align with Pull Pin	**(1)** Incorrect bulkhead or avionics sled design that does not ensure proper fit or tolerance stack-up **(2)** Assembly procedures fail to verify alignment during installation	**(1)** Failure to reliably prevent charges from becoming armed (unintended arming) or to arm when required **(2)** Increased assembly time, failure to meet NASA rule 2.1.	4C	**P:** Use alignment jigs during fabrication, add alignment marks, properly sand parts for fit. **D:** Visually confirm alignment and fitment during dry fit; perform dry fit before launch. **M:** Disassemble launch vehicle if misaligned, realign bulkheads, fabricate new bulkheads or repurpose existing ones if necessary.	1A	Not Verified
DHZ.64	Incorrect Nosecone Dimensions After Composite Construction	**(1)** Improper layup technique or inaccurate mold geometry **(2)** Excessive material buildup or uneven resin application	**(1)** Reduced internal payload volume or interference with payload fitment **(2)** Potential misalignment with couplers or airframe sections	4C	**P:** Verify mold geometry before composite layup. **D:** Perform dry fit of nosecone with payload assembly. **M:** Sand or trim components to meet tolerances; remake components if necessary with tighter control.	2B	Not Verified

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.65	Recovery Attachments Break Off Bulkheads	**(1)** Miscommunication of expected snatch forces **(2)** Improperly mounted attachment points	**(1)** Airframe sections completely separate **(2)** Launch vehicle descends ballistically	4B	**P:** Clearly communicate snatch force calculations between Recovery and Structures leads. **D:** Conduct strength tests on bulkheads to verify they meet expected snatch forces. **M:** Reinforce or replace damaged attachment points with stronger designs.	2A	Verified
DHZ.66	Air Brake Fin Slots Incorrectly Placed	Miscommunication of placement dimensions between Structures and Aerodynamics subteams	**(1)** Air brakes unable to deploy, causing launch vehicle to overshoot its intended apogee. **(2)** Air brakes system deploys over fins	4C	**P:** Maintain accurate CAD models of launch vehicle and air brakes to ensure correct integration. **D:** Verify placement using jigs when cutting air brake fin slots. **M:** File fin slots to fit if minor misalignment; remake fin can and recut slots if misalignment is significant.	1A	Verified
Air Brakes Integration Hazards							
DHZ.67	Center of Pressure Shifts Toward Air Brakes	Deployment of air brake fins alters aerodynamic profile	Vehicle's stability margin changes during flight	2D	**P:** Design air brakes such that shifts in the center of pressure do not negatively impact stability. **D:** Confirm placement of air brakes on the launch vehicle to ensure CP shifts remain within safe limits. **M:** Add ballast to the forward or aft section as needed to restore ideal stability.	1A	Verified
Launch Hazards							
Launch Support Equipment Hazards							

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.68	Rail buttons are either too large or too small	Incorrectly sized launch rail guides	Launch vehicle unable to launch	4A	**P:** Verify rail dimensions against launch field's rail buttons and NASA-provided launch rails. **D:** Measure rail buttons to confirm they meet required dimensions. **M:** Use a test rail piece to ensure proper alignment and smooth sliding.	1A	Verified
DHZ.69	Launch vehicle not assembled in time	**(1)** Unprepared or slow assembly process **(2)** Launch checklist not detailed enough, increasing confusion during assembly.	Launch vehicle unable to launch	4B	**P:** Establish a detailed timeline and perform a dry run before launch day to ensure efficient assembly. **D:** Track assembly progress during weekly meetings. **M:** Maintain spare personnel for critical tasks; reschedule to backup launch if assembly cannot be completed on time.	1B	Not Verified
DHZ.70	Launch vehicle does not slide onto launch rails, or slides with excessive friction	Launch rails have too much friction or are filled with debris/dirt	Launch vehicle fails to exit rail properly, potential failure to meet NASA Requirement 2.14	4A	**P:** Inspect rail guides and remove debris before inserting the launch vehicle. **D:** Perform a test fit on a test piece of rail. **M:** Lubricate launch rail to ensure smooth rail exit.	1A	Verified
DHZ.71	Tent falls over or flies away	High winds at launch field	**(1)** Injury to personnel **(2)** Damage to equipment **(3)** Obstruction of launch area	2C	**P:** Secure tent with stakes. **D:** Observe wind conditions and monitor tent stability. **M:** Remove tent and proceed with checklist without it if necessary.	1B	Verified
DHZ.72	Launch system fails to ignite motor ignitor	Ignition equipment failure, power loss, or electrical shorting	Launch vehicle unable to launch	3B	**P:** Test continuity at the launch pad. **D:** Check continuity and resistance in ignition system before launch. **M:** Keep backup ignitor ready; consult launch coordinator if primary and backup ignitors fail.	1A	Verified
Launch Operation Hazards							

Table 6.9: Design Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
DHZ.73	Team Members Are Unable to Be Reached	Low or no phone signal at launch field	**(1)** Delayed response during launch operations **(2)** Potentially missed safety checks or critical coordination	2B	**P:** Utilize club-owned handheld radios. **D:** Perform communication checks at the beginning of launch day; verify all pertinent team members can be reached. **M:** Update all launch personnel on procedures so that the launch can still occur if radios fail.	1A	Verified
DHZ.74	Personnel Are Disruptive Around Assembly of Launch Vehicle	**(1)** New team members not properly trained on launch day procedures **(2)** Lack of clear roles or supervision	**(1)** Delays in launch vehicle assembly **(2)** Potential launch cancellation **(3)** Increased risk of assembly errors or safety incidents	2C	**P:** Train all attendees on launch procedures; define roles, responsibilities, and behavior expectations. **D:** Monitor behavior during assembly. **M:** Temporarily remove disruptive personnel from assembly tasks and redistribute responsibilities if needed.	1B	Verified
DHZ.75	Electronics Assembly Takes Longer Than Anticipated	**(1)** Complex wiring and unclear instructions **(2)** Inexperienced personnel performing assembly	**(1)** Delayed launch schedule **(2)** Rushed assembly increasing likelihood of missing safety checks	3C	**P:** Provide clear assembly instructions. **D:** Track dry-run assembly times to ensure they are within reasonable limits. **M:** Prioritize critical connections; delay non-critical tasks as necessary.	2A	Verified
DHZ.76	Launch Vehicle Parts Left at Lab	**(1)** Improper packing or transport **(2)** Miscommunication between subteams **(3)** Last-minute changes	**(1)** Missing critical components at launch site **(2)** Delays in assembly or incomplete vehicle integration **(3)** Potential launch cancellation	4A	**P:** Implement a detailed pre-launch checklist. **D:** Verify all components are packed the night before launch. **M:** Arrange rapid transport of missing components if possible; otherwise, reschedule to a backup launch date.	1A	Verified

6.5 Environmental Hazards Analysis

Environmental hazard analyses were conducted to identify potential hazards to the environment. These were split first into hazards to the environment and hazards from the environment. Then hazards were split into either coming from or to personnel or the launch vehicle. Each hazard was then evaluated for its causes, possible effects on design safety, and the likelihood and severity of these effects. Appropriate mitigation strategies were developed to minimize or eliminate risks, ensuring a safe working environment and compliance with safety requirements.

Table 6.10 shows the environmental hazards before mitigation, and Table 6.11 shows the environmental hazards after mitigation.

Table 6.10: Environmental Risks Assessment Before Mitigation

		Level of Severity			
		1 Low Risk	2 Medium Risk	3 High Risk	4 Severe Risk
Likelihood of Occurrence	A Very Unlikely	0.0% (0)	0.0% (0)	11.11% (2)	5.56% (1)
	B Unlikely	0.0% (0)	5.56% (1)	11.11% (2)	11.11% (2)
	C Likely	0.0% (0)	11.11% (2)	16.67% (3)	11.11% (2)
	D Very Likely	0.0% (0)	11.11% (2)	5.56% (1)	0.0% (0)

Table 6.11: Environmental Risks Assessment After Mitigation

		Level of Severity			
		1 Low Risk	2 Medium Risk	3 High Risk	4 Severe Risk
Likelihood of Occurrence	A Very Unlikely	44.44% (8)	5.56% (1)	11.11% (2)	0.0% (0)
	B Unlikely	22.22% (4)	16.67% (3)	0.0% (0)	0.0% (0)
	C Likely	0.0% (0)	0.0% (0)	0.0% (0)	0.0% (0)
	D Very Likely	0.0% (0)	0.0% (0)	0.0% (0)	0.0% (0)

Table 6.12: Environmental Hazards

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
Hazards to the Environment from Personnel							
EHZ.1	Leftover Epoxy Improperly Disposed Of	Absence of proper disposal containers or protocols	Environmental contamination and potential harm to soil and water	2B	**P:** Train personnel on proper disposal techniques per manufacturer's instructions. **D:** Provide labeled waste containers. **M:** Collect leftover epoxy in waste containers and dispose of it when full.	1A	Verified
EHZ.2	Litter Left on Launch Field	Improper cleanup, failure to collect personal or team waste	**(1)** Environmental contamination **(2)** Negative public perception of rocketry activities **(3)** Potential loss of launch site access	3B	**P:** Provide trash bags for all personnel attending launch. **D:** Inspect work area after launch to ensure no trash is left behind. **M:** Collect and dispose of waste properly; recycle when possible.	1A	Verified
EHZ.3	Paint, Solvent, or Adhesive Spills	Improper storage or accidental spills during preparation/manufacturing	Soil pollution, potential harm to local plant or animal life	2C	**P:** Store chemicals in sealed containers. **D:** Inspect work area for spills and monitor for leaks. **M:** Contain and clean spills promptly.	2A	Verified
EHZ.4	Battery Fluid Leakage	Damaged or punctured LiPo or alkaline cells after use	Chemical contamination of soil, risk to wildlife, and potential injury to personnel	3B	**P:** Inspect batteries before use and handle with care. **D:** Check batteries before and after use; replace and properly dispose of any degraded batteries. **M:** Neutralize spilled fluids with absorbents; safely collect and dispose of old or damaged batteries.	1A	Verified
EHZ.5	Ematch Wires and Tape Left on Ground	Failure to collect debris after recovery separations	**(1)** Litter accumulation **(2)** Potential ingestion hazard for wildlife **(3)** Negative public perception	3C	**P:** Assign personnel via checklist to clean e-match wires and tape after launch. **D:** Visually inspect launch pad and recovery site for debris. **M:** Collect all tape and loose wires and dispose of properly.	2B	Verified
Hazards to the Environment From Launch Vehicle							

Table 6.12: Environmental Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
EHZ.6	Recovery Insulation Contaminating the Environment	Ejection charges separating the launch vehicle, causing insulation to spill	Pollution of the launch site and surrounding areas; negative perception by landowners or authorities	3D	**P:** Purchase cellulose-based insulation that is biodegradable. **D:** Confirm that insulation is biodegradable. **M:** Ensure insulation is biodegradable when packing parachutes into airframe bays.	1A	Verified
EHZ.7	Fire on Launch Field	**(1)** Motor ignition **(2)** Improperly mounted or absense of a blast deflector **(3)** Motor overpressure explosion **(4)** Ejection charge igniting on the field	**(1)** Damage to equipment **(2)** Risk of injury to personnel **(3)** Potential spread of fire to surrounding vegetation or structures	4B	**P:** Ensure blast deflector is properly aligned. **D:** Watch for sparks after motor ignition. **M:** Deploy fire extinguishers if fire starts.	2B	Verified
EHZ.8	Composite Material Contamination on Launch Field	Damaged composite components left after assembly or recovery	**(1)** Environmental contamination **(2)** Exposure risk to personnel **(3)** Potential loss of launch site access	3A	**P:** Provide trash bags for debris. **D:** Inspect assembly and recovery areas and confirm no leftover composite material. **M:** Dispose of trash properly and recycle when possible.	1A	Verified
EHZ.9	Launch vehicle damage to environment	Launch vehicle enters ballistic descent	**(1)** Environmental contamination **(2)** Harm to wildlife	3A	**P:** Design and test launch vehicle recovery system such that it prevents ballistic descent. **D:** Implement checks into recovery system assembly to ensure proper implementation of recovery system. **M:** Remove launch vehicle from environment, rehabilitate landing area.	1A	Not Verified
Hazards from the Environment to Personnel							
EHZ.10	Heat Stroke	**(1)** Prolonged work in high-temperature environments **(2)** Lack of hydration during physical activity **(3)** High levels of physical exertion without rest or cooling measures	**(1)** Confusion, disorientation, or cognitive impairment **(2)** Nausea, weakness, or fainting **(3)** Loss of consciousness or heat-related illness	3C	**P:** Bring water to launch fields; enforce hydration. **D:** Monitor team members for dizziness, nausea, confusion, or fatigue. **M:** Move affected personnel to cool areas, shade, or air-conditioned vehicles; contact EMS if needed.	1A	Verified

Table 6.12: Environmental Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
EHZ.11	Mud or Standing Water on Field	Recent rain or poor field drainage	**(1)** Personnel slipping and potential injuries **(2)** Vehicle damage or instability on launch rails	2C	**P:** Train recovery personnel to dress appropriately (pants, boots, etc.). **D:** Inspect field conditions on launch day; note soft or unstable terrain. **M:** Limit personnel on launch pad to reduce slips or falls.	1A	Verified
EHZ.12	Dehydration	**(1)** Lack of water intake during work or activity **(2)** Prolonged exposure to sunlight or high temperatures **(3)** High levels of physical exertion without hydration	**(1)** Dizziness, fatigue, or lightheadedness **(2)** Fainting or heat-related illness **(3)** Potential secondary injuries from falls	3C	**P:** Encourage water breaks, provide water bottles, monitor outdoor temperatures. **D:** Observe personnel for fatigue, dizziness, or reduced performance. **M:** Move affected personnel to shade or air-conditioned vehicle; provide water.	1B	Verified
EHZ.13	Loss of Footing	**(1)** Uneven terrain, loose gravel, or wet/slippery surfaces **(2)** In attention to footing or environmental conditions **(3)** Improper footwear for the terrain	**(1)** Scrapes, bruises, or abrasions **(2)** Sprains or fractures	2D	**P:** Train team to be aware of terrain; no running or unsafe paths. **D:** Identify dangerous terrain and alert personnel. **M:** First aid carried by safety officer during recovery.	1B	Verified
EHZ.14	Exposure to Sun/UV Radiation	**(1)** Lack of protective clothing, hats, or sunglasses **(2)** Failure to apply sunscreen or take regular shade breaks **(3)** Prolonged outdoor activity during peak sunlight hours	**(1)** Sunburn or acute skin irritation **(2)** Increased risk of skin cancer with repeated or prolonged exposure	2D	**P:** Provide sunscreen and shaded assembly areas; remind team to reapply. **D:** Before launch, note the UV index expected at the launch field. **M:** Provide aloe vera to ease sunburn discomfort.	1B	Verified
EHZ.15	Allergic Reactions	Exposure to plants, chemicals, or other environmental allergens	**(1)** Skin reactions such as rash or hives **(2)** Respiratory distress, sneezing, or difficulty breathing **(3)** Severe reactions, including anaphylaxis	4B	**P:** Avoid known allergens; provide nitrile gloves if needed. **D:** Observe skin and respiratory reactions. **M:** Administer epipen if needed; call EMS for severe reactions.	2B	Verified

Table 6.12: Environmental Hazards (continued)

ID	Hazard	Cause	Effect	LS Pre-Mitigation	Mitigation Factors	LS Post-Mitigation	Verification
EHZ.16	Insect or Bug Bites	**(1)** Extended exposure outdoors during launch activities **(2)** Lack of protective clothing or insect repellant	**(1)** Localized itchiness, rash, or swelling **(2)** Allergic reactions, potentially severe, including anaphylaxis	4C	**P:** Use bug spray; wear long sleeves and pants. **D:** Inspect team members for bites if irritation arises; monitor for swelling/rash. **M:** Apply anti-itch ointment; use epipen for allergic reactions if necessary.	1B	Verified
Hazards from the Environment to Launch Vehicle							
EHZ.17	Launch Vehicle Lands in a Tree	Premature main parachute ejection causing excessive drift distance	**(1)** Potential damage to trees **(2)** Risk to personnel retrieving vehicle **(3)** Possible launch site environmental impact	4A	**P:** With RSO approval, angle launch rails away from treeline. **D:** Monitor launch trajectory with recovery GPS. **M:** Safely remove launch vehicle from tree if possible; consult launch officials if not.	3A	Verified
EHZ.18	High Winds at Launch Site	**(1)** Sudden weather changes **(2)** Failure to check or account for forecast prior to launch	**(1)** Launch vehicle trajectory deviation **(2)** Potential launch delay or cancellation **(3)** Launch vehicle undershoots intended apogee	4C	**P:** Check weather forecasts before launch. **D:** Monitor on-field wind conditions. **M:** Delay launch to backup if winds remain too high.	3A	Verified

6.6 Project Risks

6.6.1 Project Risks FMEA

Table 6.13: Project Risks

Label	Risk	Effect	Likeli-hood	Impact	Mitigation	Quantified Impact of Mitigation
			Time Risks			
TR.1	Unanticipated rework from test failures	Delays in construction, which can cause delays in an attempted subscale, VDF or PDF	High	Medium	Begin testing at least 2 weeks before milestone deadlines.	Creates a 2 week buffer for rework
TR.2	Delays in the shipping of critical components	Failed subscale, VDF, PDF due to incomplete Launch Vehicle	Medium	High	All components will be planned out and ordered over a month in advance. When tight deadlines need to be met, purchase over Amazon Prime, or pay for quicker shipping. Pay, when possible, through PayPal as their customer service team will handle most related issues.	Expedited shipping could lead the Team to pay up to $100 more
TR.3	Custom made airframe components do not fit together on launch day	Launch Vehicle is not ready for Subscale/PDF	Medium	High	Begin rolling airframes a month in advance. Dry fit each section as it is created	Saves 200 dollars in potential rework costs
TR.4	Incomplete design before PDR/CDR deadlines	Failure of milestone review	Medium	High	Begin writing each milestone well in advance to understand the deliverables for each.	Creates a one week buffer for peer review to catch design mistakes
TR.5	Miscommunication about deadlines within the Team	Failed milestone deadline	Low	High	Create a timeline shared with the entire team, and send consistent updates starting two weeks before each deadline	Creates a week buffer for peer review of the document
TR.6	Delays in testing facilities availability	Failure to Team Derived Requirements for testing	Low	Medium	If testing facilities are available by request, requests will be made in advance	Creates a two week buffer
TR.7	Unexpected absences of a Team Member for an extended amount of time	Subteam slows in progress, possible missed milestone deadline	Low	Medium	Ensure all members have shared their contact information, and communicate any future absences one week in advance	Reduces schedule disruption by one week
TR.8	Subteam conflicts slowing progress	Failed milestone deadline	Low	Low	Foster a supportive environment. Discuss conflicts immediately as they arise	Prevents a week of lost productivity per milestone
TR.9	Painting of the Launch Vehicle takes more time than expected	Missed Subscale/VDF/PDF deadline	Low	Low	Paint Launch Vehicle a week before the scheduled launch. Prioritize launch over painting	Add a week buffer to troubleshoot any last minute issues in the Launch Vehicle
			Resource Risks			
RR.1	Lack of access to required lab space or tools after hours	Cannot manufacture or test components	High	Low	Plan to work on timeline tasks during regularly scheduled meetings. Plan timeline tasks with buffers, to ensure there will be time to meet during open hours	Reduces the amount of work lost over the two day weekend
RR.2	Lab 3D printer malfunctions	Cannot manufacture 3D printed components	High	Low	Keep a list of backup 3D printers on campus to use. Keep the 3D printer instructions guide to fix manageable issues.	Avoids up to a 2 day delay

Table 6.13: Project Risks (continued)

Label	Risk	Effect	Likeli-hood	Impact	Mitigation	Quantified Impact of Mitigation
RR.3	Insufficient number of computers during lab meetings	Cannot complete scheduled assignments on time	High	Low	Have team members bring laptop to meetings to ensure enough workstations	Prevents 1-2 hours of work lost per meeting
RR.4	Shortage of safety equipment	Cannot manufacture or test components	Low	High	Ensure that when stock is getting low but not depleted, that equipment gets ordered	Prevents work stoppage for a week
RR.5	Lack of access to sufficient transportation to the NC launch field	Cannot launch on preferred date	Low	High	Have drivers for launch commit a week in advance	Avoids a one day delay to launch
Budget Risks						
BR.1	Engineer your Experience limits travel funding for students disciplines other than Aerospace Engineering	Team cannot afford to travel to Huntsville	High	High	The Team is obtaining other sponsorship opportunities to supplement a possible decrease in funding	The team can use excess sponsorship funding to pay for other Launch Vehicle expenses
BR.2	Student Government limits funding	Must decrease budget	High	Low	The Team can submit an appeal to receive more funding	The Team can up to 500 dollars of extra funding
BR.3	Unexpected shipping or Hazmat fees	Possibility to go over budget	High	Low	When picking parts, include the shipping fees and Hazmat fees in the budget	Avoids overspending by up to 200 dollars
BR.4	Rental vans cost more than expected	Possibility to go over budget	Medium	Medium	Booking is done through EYE. Ensure they are booking the rental vans in January	Reduces budget overrun by up to 5000
BR.5	Cost of re-manufacturing failed components	Inability to fund another component	Medium	Low	Ensure proper testing is done before the manufacturing of a component to ensure success	Allows a one week buffer due to lack of a need to re-manufacture
BR.6	Cost from redesign due to failed tests	Possibility to go over budget	Low	Medium	Complete tests before construction is scheduled to start	Saves one week due to lack of a redesign
BR.7	Payment delays from sponsorships or grants	Reduced funding amount	Low	High	Ensure all sponsorships and grants are applied to on time	Extra buffer of a week allows for peer review of submission
BR.8	Complete loss of any funding source	Must decrease budget	Low	High	Applying for grants and sponsorships well in advance	Creates a buffer to allow for peer review of submission
BR.9	Item is stolen from the lab and needs to be repurchased	Team must field the cost	Low	Low	Ensure the lab is locked when not in use, and that only verified members can have access	Prevents delays of up to a week in manufacturing due to lost goods
BR.10	Club member steals merchandise meant for fundraising	Team must field the cost	Low	Low	The Team Treasurer is to keep all merchandise in their housing, and to personally handle any transaction	Loss of up to 50 dollars for the Launch Vehicle budget
Scope/Functionality Risks						
SFR.1	Air Brakes do not function	Incorrect predicted apogee	High	High	Do extensive testing over a week in advance of a milestone launch	

Table 6.13: Project Risks (continued)

Label	Risk	Effect	Likeli-hood	Impact	Mitigation	Quantified Impact of Mitigation
SFR.2	Payload fails to operate during flight	Failed milestone launch	Medium	High	Ensure the payload is completed with enough time before a launch to allow for extensive testing	Provides a two week buffer to fix integration issues within the Launch Vehicle
SFR.3	Limited space in the Air Brakes Bay for Subscale	Air Brakes cannot fit into the Launch Vehicle, and do not deploy	Medium	High	Dry fit the Air Brakes with all electronics in its bay a week before launch	Prevents up to 200 dollars of rework costs for the Air Brakes
SFR.4	Misunderstanding of NASA SL Requirements	Failure of a Milestone	Medium	High	Ensure attendance at all NASA information sessions	Increases the milestone scores
SFR.5	Recovery system fails to deploy	Failed milestone flight	Medium	High	Write a robust checklist, ensure there are launch backup dates	Prevents up to 2000 dollars of rework costs for the Launch Vehicle
SFR.6	Structural failure of fins or airframe	Failed milestone launch	Low	High	Complete extensive testing before an attempted milestone launch	Avoids up to 200 dollars of rework costs
SFR.7	STEM Engagement Events not completed correctly	Team cannot win the STEM engagement award	Low	High	Team to review any documentation sent out in the NASA drive, or present in the information sessions	Increases the STEM Engagement score

7 Project Plan

7.1 Requirements Verification

Requirement verification is lead by the Integration lead,as identified in Proposal Section 1.4. Both NASA requirements and Team Derived requirements are verified through Requirement Verification Matrices (RVMs). These serve as tools to ensure all project requirements are properly satisfied. The RVMs maintain traceability between requirement owners, design elements and corresponding verification activities. They will be kept and maintained throughout the project timeline. The design and layout of these RVMs were guided by the NASA Systems Engineering Handbook [9].

Each RVM includes key columns. The ID and SHALL Statement columns identify and describe the requirement. The Planned Action column outlines the teams approach to verifying the requirement. The Verification Method specifies how the verification will be performed. The Verification Success Criteria column identifies the conditions that need to be met in order for a requirement to be verified. The Status column describes the current state of verification, with rankings described in Table 7.1. The performing subsystem column describes the subsystem(s) responsible for executing the verification. Finally, the Results column points to the location of the objective evidence of the verification.

For team derived requirements, Justification entries are also included to explain the necessity and rationale behind each requirement.

Table 7.1: Requirement Status Key

Verification Level	Description	Key
Verified	All verification success criteria has been met.	V
Partially Verified	Some verification success criteria has been met, some criteria may still be in progress.	PV
In Progress	None of the verification success criteria has been met, but the verification process has begun.	IP
Not Verified	None of the verification success criteria has been met and not meaningful progress has been made towards verification yet.	NV

Table 7.2 below shows the completion status of both the NASA requirements and the Team Derived requirements.

Table 7.2: Requirements Completion Status

Requirement Type	Verified	Partially Verified	In Progress	Not Verified
NASA Requirements	32.86 % (24)	6.85 % (5)	45.21 % (33)	15.07 % (11)
Team Derived Requirements	8.86% (7)	6.33 % (5)	32.91 % (26)	51.90 % (41)

7.1.1 Competition Requirements

Table 7.3: 2025-2026 General Requirements

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
1.1	Teams shall engage with their communities in STEM industry or STEM education. To satisfy this requirement teams shall complete either a STEM Industry Engagement Plan and Summary OR a Community STEM Engagement Plan and Summary. Requirements for each can be found in the Engagement section pages 38–40.	The elected Outreach Officer, identified in Section 1.5.1 of Proposal, will create a plan where they identify local organizations to visit, activities to implement, and potential dates and/or schedules to engage communities throughout the fall and spring semesters. They will keep a record of communication, counts of individuals impacted, and photos of events to be shared with the Team Lead before and after the event.	**(1)** Inspection **(2)** Demonstration	**(1)** The Elected Outreach Lead is identified in Proposal **(2)** The Outreach Officer keeps records of communication throughout the year and runs STEM Engagement activities	IP	Project Management	**(1)** See Section 1.5.1 of Proposal for identified Outreach Officer. **(2)** The Outreach Officer has kept record and is in communication with local communities and continues to do so. As of 11/3/2025, they have held 3 STEM engagement events impacting 48 individuals.
1.2	The team shall establish and maintain a social media presence to inform the public about team activities	The elected Social Media Officer, identified in section 1.5.1 of Proposal, will maintain and use the team's social media platforms to document progress and events. Platforms include but are not limited to Instagram, Facebook, and LinkedIn.	**(1)** Inspection **(2)** Demonstration	**(1)** The Elected Social Media Officer is identified in Proposal. **(2)** The Social Media Officer posts regularly, keeping members updated on team activities in an engaging and informative way.	IP	Project Management	**(1)** See Section 1.5.1 of Proposal for identified Social Media Officer **(2)** The Social Media Officer makes weekly update posts as well as posts during major milestones such as launches.
1.3	Each team shall identify a "mentor." A mentor is defined as an adult who is included as a team member, supports the team (or multiple teams) throughout the project year, and may or may not be affiliated with the school, institution, or organization. The team mentor must adhere to the following requirements:	The Team Lead will identify a mentor who is not affiliated with the team's school.	**(1)** Inspection **(2)** Demonstration	**(1)** The Mentor is identified in Proposal. **(2)**The Team Lead keeps in regular contact with the mentor, utilizing them for design advice.	V	Project Management	**(1)** See Section 1.2 of Proposal for identified mentor. **(2)** The Team Lead continues to have regular contact with the mentor, keeping them informed and updated with project progress and design choices.

Table 7.3: 2025-2026 General Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
1.3.1	The mentor shall maintain a current certification and be in good standing with the National Association of Rocketry (NAR) or Tripoli Rocketry Association (TRA) for the motor impulse class the team intends to use.	The chosen mentor will maintain both good standing with either Association of Rocketry (NAR) or Tripoli Rocketry Association (TRA) as well as certification for the motor impulse class that the Aerodynamics Lead decides to use.	Inspection	The mentor has a minimum level 2 rocketry certification and is in good standing with TRA officials.	V	Project Management	The chosen mentor has a level 3 certification, and maintains good contact with TRA organization.
1.3.2	The mentor shall have flown and successfully recovered (using electronic, staged recovery) a minimum of two flights in the motor impulse class (or higher) the team intends to use, prior to PDR.	The chosen mentor will have either flown or provided logs of at minimum two successful flights utilizing electronic staged recovery in the motor class the team intends to use.	Inspection	Chosen mentor has flown and has a record of two successful flights with electronic deployment.	V	Project Management	The chosen mentor has provided the Team Lead with proof of two successful flights with electronic deployment.
1.3.3	The mentor must attend all team launches throughout the project year, including launch week, as the mentor is designated the individual owner of the rocket for insurance and liability purposes.	The chosen mentor shall attend all team launches including the competition launch throughout the year as the flyer of record.	Demonstration	The chosen mentor attends all club launches and is the flyer of record for each competition launch.	PV	Project Management	The chosen mentor has attended the subscale launch and was identified as the flyer of record.

Table 7.4: 2025-2026 Vehicle Requirements

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
2.1	The vehicle shall deliver the payload to an apogee altitude between 4,000 and 6,000 feet above ground level (AGL). Teams flying below 3,500 feet or above 6,500 feet on their competition launch will receive zero altitude points towards their overall project score and will not be eligible for the Altitude Award.	The Structures and Aerodynamics Leads will design the full Launch vehicle to be capable of delivering the Payload to an apogee between 4,000 and 6,000 ft. AGL. The Structures Lead will facilitate the manufacturing of the launch vehicle with the team.	**(1)** Analysis **(2)** Demonstration	**(1)** Simulations run by the Aerodynamics Lead show the launch vehicle reaching an apogee between 4,000 and 6,000 ft. AGL **(2)** The launch vehicle's recovery altimeter data shows between 4,000 and 6,000 ft. for the VDF and PDF flights.	PV	Aerodynamics & Structures	**(1)**Simulations run by the Aerodynamics Lead show the launch vehicle reaching an apogee of 4860 ft. See Section 3.10.1. **(2)** VDF and PDF flights are planned for the spring, see Section 7.4.

Table 7.4: 2025-2026 Vehicle Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
2.2	The launch vehicle and payload shall be capable of remaining in launch-ready configuration on the pad for a minimum of 3 hours without losing the functionality of any critical on-board components, although the capability to withstand longer delays is highly encouraged.	The Recovery Lead and Payload Team will use batteries that have a large enough capacity that they can power all avionic and payload electronics for a minimum of 3 hours without losing the capability of any critical components. The Integration will verify the batteries will function via a ground test before launch.	**(1)** Analysis **(2)** Demonstration	**(1)** Electronic power draw combined with battery capacity calculations confirm functionality for >3 hours **(2)** All avionics and payload electronic systems maintain full operational functionality for > 3 hours.	IP	Payload & Recovery	**(1)** For recovery battery analysis see Section 3.8.4. Preliminary payload battery choices are described in Section 4.5.3. **(2)** VDF and PDF flights which will confirm battery functionality are planned for the spring, see Section 7.4.
2.3	Teams shall declare their target altitude goal at the CDR milestone. The declared target altitude shall be used to determine the team's altitude score	The Aerodynamic lead will perform simulations based off of the Designed Launch Vehicle and determine a target altitude specified in the CDR Report.	Inspection	A single, defined target altitude is defined in the CDR report.	IP	Aerodynamics	Preliminary apogee predictions are described in Section 3.10.1.
2.4	The launch vehicle shall be designed to be recoverable and reusable. Reusable is defined as being able to launch again on the same day without repairs or modifications.	The Structures Lead will construct a Vehicle capable of withstanding the launch loads expected and the Recovery Lead will design a recovery system that will safely bring the launch vehicle to the ground, with the vehicle being able to launch again within the same day.	Demonstration	The launch vehicle is successfully recovered following both VDF and PDF with no structural damage that would deem the vehicle non-launchable. All recovery, payload and Air Brakes electronics are fully functional.	IP	Recovery & Structures	The structural design of the launch vehicle is described in Section 3.3. The recovery system is described in Section 3.9.
2.5	The launch vehicle shall have a maximum of four (4) independent sections. An independent section is defined as a section that is either tethered to the main vehicle or is recovered separately from the main vehicle using its own parachute	The Structures Lead and Recovery Lead will design the separating points of the rocket for recovery such that there is a maximum of four (4) independent sections, those being defined by NASA Req. 2.5.	Inspection	Completed and assembled launch vehicle shows no more than four independent sections.	IP	Recovery & Structures	The locations of the launch vehicle independent sections is described in Section 3.3.1.

Table 7.4: 2025-2026 Vehicle Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
2.5.1	Coupler/airframe shoulders which are located at in-flight separation points shall be at least two airframe diameters in length. (one body diameter of surface contact with each airframe section).	Structures Lead shall design and manufacture the launch vehicle such that any coupler/airframe shoulders at in-flight separation points shall be at least two airframe diameters in length.	Inspection	Completed and assembled launch vehicle shows each in-flight separation point coupler/shoulder is at least two airframe diameters in length.	IP	Structures	The locations and dimensions of the launch vehicle Coupler/airframe shoulders located at in-flight separation points are described in Section 3.3.1.
2.5.2	Coupler/airframe shoulders which are located at non-in-flight separation points shall be at least 1.5 airframe diameters in length. (0.75 body diameter of surface contact with each airframe section.)	Structures Lead shall design and manufacture the launch vehicle such that any coupler/airframe shoulders at in-flight separation points will be at least 1.5 airframe diameters in length.	Inspection	Completed and assembled launch vehicle shows each non in-flight separation point coupler/shoulder is at least 1.5 airframe diameters in length.	IP	Structures	The locations and dimensions of the launch vehicle Coupler/airframe shoulders located at non-in-flight separation points are described in Section 3.3.1.
2.5.3	Nosecone shoulders shall be at least ½ body diameter in length	Structures Lead shall design and manufacture the nosecone such that its shoulder will be at least ½ body diameter in length.	Inspection	Completed and assembled launch vehicle shows the nosecone shoulder is at least ½ airframe diameters in length.	IP	Structures	The locations and dimensions of the launch vehicle nosecone shoulder is described in Section 3.3.1.
2.6	The launch vehicle shall be capable of being launched by a standard 12-volt direct current firing system. The firing system shall be provided by the NASA-designated launch services provider.	The Aerodynamics Lead shall select a motor ignitor that is capable of being launched using the NASA-designated 12-volt direct firing system.	Demonstration	The selected motor ignitor reliably initiates motor ignition when connected to a 12 V DC source with a current output of the NASA-designated firing system used during competition.	V	Aerodynamics	Ignitor for motor ignition is identified in Section 3.7.
2.6.1	Each team shall use commercially available ematches or igniters. Hand-dipped igniters shall not be permitted.	The Aerodynamics and Recovery Leads will use commercially available ematches for all pyrotechnic initiations.	Inspection	The selected ematches for recovery systems and propulsion systems are commercially available.	V	Aerodynamics & Recovery	Selected E-matches for black powder deployment are identified in Section 3.8.9. Ignitor for motor ignition is identified in Section 3.7.

Table 7.4: 2025-2026 Vehicle Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
2.7	The launch vehicle shall use a commercially available solid motor propulsion system using ammonium perchlorate composite propellant (APCP) which is approved and certified by the National Association of Rocketry (NAR), Tripoli Rocketry Association (TRA), and/or the Canadian Association of Rocketry (CAR).	The Aerodynamics Lead will select a commercially available motor that uses ammonium perchlorate composite propellant certified by the National Association of Rocketry and/or Tripoli Rocketry Association.	Inspection	The chosen Motor shall use ammonium perchlorate composite propellant. The Motor will be sold by a vendor recognized by the National Association of Rocketry and/or Tripoli Rocketry Association.	V	Aerodynamics	The primary and secondary motor choices are identified in Section 3.7
2.8	The launch vehicle shall be limited to a single motor propulsion system.	The Aerodynamics and Structures Lead will design the launch vehicle such that it utilizes a single motor propulsion.	Inspection	The Launch Vehicle design utilizes a single motor propulsion system.	V	Aerodynamics & Structures	The propulsion system is identified in Section 3.7
2.9	The total impulse provided by a College or University launch vehicle shall not exceed 5,120 Newton-seconds (L-class).	The Aerodynamics will select a motor that does not exceed 5120 Newton-seconds of impulse.	Inspection	The motor for the launch vehicle does not exceed 1520 newton-seconds of impulse	V	Aerodynamics	The primary and secondary motor choices are identified in Section 3.7.
2.10	Pressure vessels on the vehicle must be approved by the RSO and shall meet the following criteria	The Team Lead will inform the RSO of any and all pressure vessels onboard the launch vehicle.	Inspection	The launch vehicle is designed such that no pressure vessel system is utilized in the launch vehicle.	V	Project Management	Section 3.8.9 shows the vehicle design with no pressure vessels.
2.10.1	The minimum factor of safety (Burst or Ultimate pressure versus Max Expected Operating Pressure) will be 4:1 with supporting design documentation included in all milestone reviews	The Structures Lead and Safety team will verify that any and all pressure vessels are designed with a factor of safety of 4:1.	Inspection	No pressure vessel system is utilized in the launch vehicle.	V	Recovery & Safety	Section 3.8.9 shows the recovery system uses no pressure vessels.
2.10.2	Each pressure vessel shall include a pressure relief valve that sees the full pressure of the tank and is capable of withstanding the maximum pressure and flow rate of the tank.	The Recovery Lead will design the recovery system such that every pressure vessel will include pressure relief valves that see the pressure of the tank and will be capable of withstanding the maximum pressure and flow rate of the tank.	Inspection	No pressure vessel system is utilized in the launch vehicle.	V	Recovery & Safety	Section 3.8.9 shows the recovery system uses no pressure vessels.

Table 7.4: 2025-2026 Vehicle Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
2.10.3	The full pedigree of the tank shall be described, including the application for which the tank was designed and the history of the tank. This will include the number of pressure cycles put on the tank, the dates of pressurization/depressurization, and the name of the person or entity administering each pressure event.	The Safety Team will work with the Recovery team to document all pressure vessels including the number of pressure cycles, dates of pressurization/depressurization, and the name of the person/entity administering each pressure event.	Inspection	No pressure vessel system is utilized in the launch vehicle.	V	Recovery & Safety	Section 3.8.9 shows the recovery system uses no pressure vessels.
2.11	The launch vehicle shall have a minimum static stability margin of 2.0 while sitting on the pad.	The Aerodynamics Lead shall design the launch vehicle such that it will have a minimum static stability margin of 2.0 while on the pad.	**(1)** Analysis **(2)** Demonstration	**(1)** Analysis shows the projected launch vehicle has a stability a minimum of 2.0 in its launch ready configuration. **(2)** The Launch Vehicle design has a static stability of greater than 2 in its launch ready configuration.	PV	Aerodynamics	**(1)** Section 3.10.4 shows the projected stability margin of the launch vehicle. **(2)** Stability will be confirmed before the VDF flight, scheduled in Section 7.4.
2.12	The launch vehicle shall have a minimum thrust to weight ratio of 5.0:1.0.	The Aerodynamics Lead and Structures Lead will design the Launch Vehicle to have a minimum thrust to weight ratio of 5.0:1.0.	Analysis	The selected motor provides the launch vehicle with a minimum thrust to weight ratio of 5.0:1.0.	IP	Aerodynamics & Structures	Section 3.7 shows the projected thrust to weight of the launch vehicle.
2.13	Any structural protuberance on the rocket shall be located aft of the burnout center of gravity. Camera housings will be exempted, provided the team can show that the housing(s) causes minimal aerodynamic effect on the rocket's stability.	The Aerodynamics Lead will ensure that any systems that have any structural protuberances are located aft of the center of gravity of the launch vehicle, as well as confirm that any necessary cameras that are located forward of the burnout center of gravity will cause minimal aerodynamic effect to the launch vehicle's stability.	**(1)**Inspection **(2)**Analysis	**(1)**Any structural protuberance is located aft of the center of gravity. **(2)** Any camera housings located forward of the burnout center of gravity cause minimal aerodynamic effect to the launch vehicle's stability	IP	Aerodynamics	**(1)**Section 3.3.1 shows the location of the Air Brakes system. **(2)** Camera locations will be described in the CDR document.

Table 7.4: 2025-2026 Vehicle Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
2.14	The launch vehicle shall accelerate to a minimum velocity of 52 fps at rail exit	The Aerodynamics Lead will select a commercially available motor that provides enough thrust such that the velocity of the launch vehicle at the exit of the rail is at minimum 52 fps.	Analysis	The selected motor provides the launch vehicle with a velocity off the rod of a minimum of 52 fps.	IP	Aerodynamics	Velocity off the rod analysis is included in Section 3.7
2.15	Subscale rockets are required to use a minimum motor impulse class of E	The Aerodynamics Lead will select a Motor with a minimum motor impulse class of E for the subscale launch vehicle.	Inspection	The selected motor for subscale has a minimum impulse class of E.	NV	Aerodynamics	Subscale motor selection will be documented in the CDR document.
2.16	The subscale rocket shall not exceed 75% of the dimensions (length and diameter) of your designed full-scale rocket. For example, if your full-scale rocket is a 4" diameter, 100" length rocket, your subscale shall not exceed 3" diameter and 75" in length.	The Aerodynamic Lead and Structures Lead will design the subscale launch vehicle to not exceed 75% the dimensions of the full-scale launch vehicle.	Inspection	The design of the subscale launch vehicle does not exceed 75% of the dimensions of the full-scale launch vehicle.	NV	Aerodynamics & Structures	Subscale launch vehicle design will be documented in the CDR document.
2.18	Vehicle Demonstration Flight—The purpose of the Vehicle Demonstration Flight is to validate the launch vehicle's stability, structural integrity, recovery systems, and the team's ability to prepare the launch vehicle for flight. A successful flight is defined as a launch in which all hardware is functioning properly (drogue chute at apogee, main chute at the intended lower altitude, functioning tracking devices, etc.).	Project Management will ensure that the launch Vehicle Performs a Vehicle Demonstration flight, wherein all associated subsystems (Recovery, Structures, Aerodynamics, etc) perform as intended and in the same configuration as the competition prior to the FRR Deadline.	Demonstration	The VDF flight confirms the full functionality of the launch vehicle including the recovery system and structural components.	NV	Project Management	VDF flight is scheduled in the spring, described in Section 7.4.
2.19	All Lithium Polymer batteries shall be sufficiently protected from impact with the ground and will be brightly colored, clearly marked as a fire hazard, and easily distinguishable from other payload hardware.	All Lithium Polymer Batteries used in the launch vehicle will be designed to have adequate housing and labeling to ensure that they are protected from impact as well as identifiable from payload hardware. The Safety Officer and Integration lead will ensure that housings meet these requirements.	Inspection	All Lithium Polymer Batteries are adequately housed in the launch vehicle and are identifiable from payload hardware.	IP	Integration & Safety	Recovery batteries are identified in Section 3.8.4. Payload batteries are described in Section 4.5.3.
2.20.1	The launch vehicle shall not utilize forward firing motors	The Aerodynamics Lead will design the rocket such that it will not utilize forward firing motors.	Inspection	The launch vehicle design does not utilize forward firing motors.	V	Aerodynamics	The propulsion system is identified in Section 3.7

Table 7.4: 2025-2026 Vehicle Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
2.20.2	The launch vehicle shall not utilize motors that expel titanium sponges (Sparky, Skidmark, MetalStorm, etc.)	The Aerodynamics Lead will design the rocket such that it will not utilize motors that expel titanium sponges.	Inspection	The launch vehicle design does not utilize motors that expel titanium sponges.	V	Aerodynamics	The primary and secondary motor choices are identified in Section 3.7.
2.20.3	The launch vehicle shall not utilize hybrid motors	The Aerodynamics Lead will design the rocket such that it will not utilize hybrid motors.	Inspection	The launch vehicle design does not utilize hybrid motors.	V	Aerodynamics	The primary and secondary motor choices are identified in Section 3.7.
2.20.4	The launch vehicle shall not utilize a cluster of motors.	The Aerodynamics Lead will design the rocket such that it will not utilize a cluster of motors.	Inspection	The launch vehicle design does not utilize cluster motors.	V	Aerodynamics	The propulsion system is identified in Section 3.7
2.20.5	The launch vehicle shall not utilize friction fitting for motors	The Structures Lead will design a motor retention system that does not utilize friction fitting for the selected motor.	Inspection	The launch vehicle design does not utilize friction fitting for motor retention.	IP	Structures	The motor retention system is identified in Section 3.4.1
2.20.6	The launch vehicle shall not exceed Mach 1 at any point during flight	The Aerodynamics Lead will select a motor such that the designed Launch Vehicle does not exceed Mach 1 at any point during its flight.	**(1)** Analysis **(2)** Demonstration	**(1)** The launch vehicle is simulated to reach velocities below mach 1. **(2)** During the VDF flight, altimeter data shows the launch vehicle does not reach velocities above Mach 1.	IP	Aerodynamics	**(1)** Flight profiles depicting velocities during flight are identified in Section 3.7 **2)** VDF flight is planned for the spring, scheduled in Section 7.4.
2.20.7	Vehicle ballast shall not exceed 10% of the total un-ballasted weight of the rocket, as it would sit on the pad (i.e., a rocket with an unballasted weight of 40 lbs. on the pad may contain a maximum of 4 lbs. of ballast).	The Aerodynamics Lead and Structures Lead will Design the Rocket such that any and all potential ballast needed will not exceed 10% of the total Launch Vehicle's un-ballasted weight.	Inspection	The launch vehicle's ballast is measured to be less than 10% of the launch vehicle's un-ballasted weight.	IP	Aerodynamics & Structures	Ballast preliminary calculations are identified in Section 3.10.4
2.20.7.2	Ballast must be mechanically retained. Friction fit is not a permissible form of retention.	The Structures Lead will ensure that any and all ballast needed will be mechanically retained without the use of friction fitting.	Inspection	The launch vehicle's ballast is designed to be mechanically retained.	NV	Structures	Ballast retention will be identified in the CDR document.
2.20.7.3	Ballast shall be removable	The Structures Lead will ensure that any and all ballast needed will be removable.	Inspection	The launch vehicle's ballast is designed to be removable.	NV	Structures	Ballast configuration will be identified in the CDR document.

Table 7.4: 2025-2026 Vehicle Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
2.20.7.4	All requirements found in sections 1 through 5 of this handbook shall be met in both the minimum and maximum design ballast configurations. Where applicable, teams are expected to present calculations and performance metrics for both minimum and maximum design ballast configurations.	The Aerodynamics Lead will calculate performance metrics for both the minimum and maximum Ballast designed to be on the final Launch Vehicle configurations.	Analysis	The launch vehicles minimum and maximum ballast configurations along with associated calculations are located in the CDR report.	IP	Aerodynamics	Ballast preliminary calculations are identified in Section 3.10.4

Table 7.5: 2025-2026 Recovery Requirements

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
3.1	The full-scale launch vehicle shall stage the deployment of its recovery devices, where a drogue parachute is deployed at apogee, and a main parachute is deployed at a lower altitude. Tumble or streamer recovery from apogee to main parachute deployment is also permissible, provided that kinetic energy during drogue stage descent is reasonable, as deemed by the RSO.	The Recovery Lead will design a recovery system such that the drogue parachute is deployed at apogee and the main parachute is deployed at a lower altitude.	Demonstration	The launch vehicle's altimeters are pre-programmed to deploy its drogue parachute at apogee and main parachute at a specified altitude during descent. This is verified during the VDF flight	IP	Recovery	The recovery system deployment design is located in Section 3.9.3. VDF flight is planned for in the spring, identified in Section 7.4.
3.1.1	The main parachute shall be deployed no lower than 500 feet.	The Recovery Lead will design the recovery system such that the main parachute is deployed at no lower than 500 feet.	Demonstration	The Launch vehicle's altimeters are pre-programmed to deploy its main parachute at an altitude greater than 500 ft during descent. This is verified during the VDF flight.	IP	Recovery	The main parachute deployment design is located in Section 3.9.3. VDF flight is planned for in the spring, identified in Section 7.4.
3.1.2	The apogee event shall contain a delay of no more than 2 seconds.	The Recovery Lead will design the recovery system such that the drogue event will contain a delay of no more than 2 seconds.	Demonstration	The launch vehicle's altimeters are pre-programmed to deploy its drogue parachute no more than 2 seconds after apogee. This is verified during the VDF flight.	IP	Recovery	The drogue parachute deployment design is located in Section 3.9.3. VDF flight is planned for in the spring, identified in Section 7.4.

Table 7.5: 2025-2026 Recovery Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
3.1.3	Motor ejection is not a permissible form of primary or secondary deployment.	The recovery system design will not utilize motor ejection for any deployment events in the recovery system.	Inspection	The recovery system design does not utilize motor ejection for any events.	V	Recovery	The recovery system deployment design is located in Section 3.9.3.
3.2	Each independent section of the launch vehicle shall have a maximum kinetic energy of 75 ft-lbf at landing. Teams whose heaviest section of their launch vehicle, as verified by vehicle demonstration flight data, stays under 65 ft-lbf will be awarded bonus points.	The Recovery Lead will select or manufacture parachutes such that each independent section will have a maximum kinetic energy of 75 ft-lbf at landing.	**(1)** Analysis **(2)** Demonstration	**(1)** Kinetic energy calculated for each section has a maximum kinetic energy of 75 ft-lbf. **(2)** During VDF the drogue and main parachute delivers each individual launch vehicle section to the ground with a maximum kinetic energy of 75 ft-lbf.	IP	Recovery	Kinetic energy calculations are described in Section . VDF flight is planned for in the spring, identified in Section 7.4.
3.3	The recovery system shall contain redundant, commercially available barometric altimeters that are specifically designed for initiation of rocketry recovery events. The term “altimeters” includes both simple altimeters and more sophisticated flight computers.	The Recovery Lead will design the recovery system such that they utilize commercially available barometric altimeters for the initiation of recovery events.	Inspection	The recovery system design utilizes commercially available barometric altimeters for recovery events.	V	Recovery	Recovery altimeters are described in Section 3.9.2.
3.4	Each altimeter shall have a dedicated power supply, and all recovery electronics shall be powered by commercially available batteries.	The Recovery Lead will design the avionics system for the recovery system such that each altimeter has a dedicated power supply utilizing commercially available batteries.	Inspection	The recovery system avionics are designed to be powered with individual and commercially available batteries.	V	Recovery	Batteries used for the recovery system are identified in Section 3.9.1.
3.5	Each altimeter shall be armed by a dedicated mechanical arming switch that is accessible from the exterior of the rocket airframe when the rocket is in the launch configuration on the launch pad.	The Recovery Lead will design the Recovery system such that it may be armed using mechanical arming switches that are accessible from the exterior of the launch vehicle while it is on the launch pad.	Demonstration	The recovery system design utilizes mechanical arming switches accessible from outside the launch vehicle.	IP	Recovery	Mechanical arming switches used in the recovery system are identified in Section 3.9.2.

Table 7.5: 2025-2026 Recovery Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
3.6	Each arming switch shall be capable of being locked in the ON position for launch (i.e., cannot be disarmed due to flight forces).	The Recovery Lead will utilize arming switches such that they are capable of being locked in the on position regardless of flight forces experienced during the launch.	Demonstration	The recovery system design utilizes mechanical arming switches that lock into the ON position for the duration of the flight and are incapable of being disarmed due to flight forces.	IP	Recovery	Mechanical arming switches used in the recovery system are identified in Section 3.9.2.
3.7	The recovery system, GPS and altimeters, and electrical circuits shall be completely independent of any payload electrical circuits.	The Recovery Lead will design the Recovery system such that any and all avionics used in the system are completely independent of any and all payload electrical circuits.	Inspection	All recovery system electrical circuits are separate from any payload electrical circuits.	V	Recovery	The recovery system electrical circuits are located in Section 3.9.1.
3.8	Removable shear pins shall be used for both the main parachute compartment and the drogue parachute compartment.	The Recovery Lead will design the Recovery system such that shear pins will be used for both main and drogue parachute compartments. The Structures Lead will ensure the launch vehicle is designed such that shear pins will be to retain the main and drogue compartments.	Inspection	The recovery system is designed such that separating sections utilize shear pins.	V	Recovery & Structures	The shear pin use is described in Section 3.3.1 and 3.8.9
3.9	Bent eyebolts shall not be permitted in the recovery subsystem.	The Structures Lead will ensure that any connection points between shock cord and structural elements of the launch Vehicle (bulkheads) will not utilize bent eyebolts.	Inspection	The launch vehicle design ensures no connection points between the shock cord and elements of the launch vehicle utilizes bent eyebolts.	V	Structures	Section 3.3 details connection points between shock cord and structural elements of the launch vehicle.
3.10	The recovery area shall be limited to a 2,500 ft. radius from the launch pads.	The Recovery Lead will select appropriately sized parachutes to be used for the recovery system such that the Launch Vehicle does not drift more than 2,500 ft. from the launch pads.	**(1)** Analysis **(2)** Demonstration	**(1)** The Recovery system calculates lateral drift distance from the launch pad under the maximum allowable wind speed to be less than 2,500 ft. **(2)** Drift distance is verified using GPS coordinates obtained during VDF.	PV	Recovery	**(1)** Drift distance calculations are described in Section 3.10.10. **(2)** Drift distances will be verified during VDF, scheduled in Section 7.4.

Table 7.5: 2025-2026 Recovery Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
3.11	Descent time of the launch vehicle shall be limited to 90 seconds (apogee to touch down). Teams whose launch vehicle descent, as verified by vehicle demonstration flight data, stays under 80 seconds will be awarded bonus points.	The Recovery Lead will select appropriately sized parachutes to be used for the recovery system such that the launch vehicle's descent time is under 90 seconds.	**(1)** Analysis **(2)** Demonstration	**(1)** The recovery system is designed such that the parachutes will deliver the launch vehicle to the ground in under 90 seconds. **(2)** VDF flight confirms descent time is under 90 seconds.	PV	Recovery	**(1)** Descent time calculations are described in Section 3.10.9. **(2)** Descent will be verified during VDF, scheduled in Section 7.4.
3.12	Electronic GPS Tracking device(s) shall be installed in the launch vehicle and will transmit the position of the tethered vehicle and any independent section(s) to a ground receiver.	The Recovery Lead will utilize electronic GPS tracking devices in every untethered independent section of the launch vehicle that are capable of transmitting position of the section to a ground receiver.	**(1)** Inspection **(2)** Test **(3)** Demonstration	**(1)** The recovery system is designed with GPS tracking devices in each untethered independent section. **(2)** GPS transmitters are ground tested to confirm functionality. **(3)** VDF confirms functionality of GPS tracking devices.	IP	Recovery	**(1)** GPS tracking devices are specified in Section 3.8.2. **(2)** Ground test results will be verified in the CDR document. **(3)** GPS tracker functionality will be verified during VDF, scheduled in Section 7.4.
3.13.1	The recovery system altimeters shall be physically located in a separate compartment within the vehicle from any other radio frequency transmitting device and/or magnetic wave producing device.	The Recovery Lead will design an avionics system such that any altimeters used will be physically separated from any radio frequency transmitting device or magnetic wave producing device.	Inspection	The launch vehicle is designed such that all radio frequency transmitting devices and/or magnetic wave producing devices are located separate from any recovery avionics.	IP	Recovery	The recovery avionics location is specified in Section 3.3.1.
3.13.2	The recovery system electronics shall be shielded from all on-board transmitting devices to avoid inadvertent excitation of the recovery system electronics.	The Recovery Lead will design an avionics system such that any recovery electronics used will be shielded from any transmitting devices.	Inspection	Recovery system electronics are physically and electrically isolated from all onboard transmitters	IP	Recovery	The recovery avionics location is specified in Section 3.9.1.

Table 7.5: 2025-2026 Recovery Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
3.13.3	The recovery system electronics shall be shielded from all on-board devices which may generate magnetic waves (such as generators, solenoid valves, and Tesla coils) to avoid inadvertent excitation of the recovery system.	The Recovery Lead will design an avionics system such that any recovery electronics are shielded from any magnetic wave producing devices.	Inspection	Recovery system electronics are physically and electrically isolated from all onboard devices that generate magnetic waves.	IP	Recovery	The recovery avionics location is specified in Section 3.9.1.
3.13.4	The recovery system electronics SHALL be shielded from any other on-board devices which may adversely affect the proper operation of the recovery system electronics.	The Recovery Lead will design the avionics system such that any recovery electronics are shielded from any devices on the launch vehicle that may affect the proper operations of the recovery system electronics.	Inspection	Recovery system electronics are physically and electrically isolated from all onboard devices that might adversely affect the operation of the recovery system.	IP	Recovery	The recovery avionics location is specified in Section 3.9.1.

Table 7.6: 2025-2026 Payload Experiment Requirements

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
4.1	After landing, teams shall autonomously collect and retain a soil sample of at least 50 milliliters.	The Payload Team will design a payload that can autonomously collect 50 ml of soil from the landing site.	Demonstration	Upon landing the payload collects a minimum of 50 ml of soil from the landing site.	NV	Payload Team	Preliminary payload design is outlined in Section 4.5.
4.1.1	All soil collection and analysis must be completed within 15 minutes of landing.	The Payload Team will design the payload such that it will collect the soil sample within 15 minutes of the launch vehicle landing.	Demonstration	Upon landing the payload collects a minimum of 50 ml of soil from the landing site within 15 minutes.	NV	Payload Team	Preliminary payload design is outlined in Section 4.5.
4.2	Teams shall autonomously test the collected sample for at least one of the following: Nitrate-Nitrogen content, pH level, or electrical conductivity.	The Payload Team will design the payload such that it is able autonomously test soil samples for its Nitrate-Nitrogen content, pH level, and Electrical Conductivity.	Demonstration	The payload tests collected soil sample for Nitrate-Nitrogen content, pH level, and Electrical Conductivity.	NV	Payload Team	Preliminary payload sensor designs are outlined in Section 4.5.3.
4.2.1	Analysis results shall include time stamps for verification.	The Payload will be programmed such that it includes timestamps for every important Analysis result.	Demonstration	The payload logs and collects timestamps for each state change.	NV	Payload Systems Lead	Payload software will be described in the CDR document.

Table 7.6: 2025-2026 Payload Experiment Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
4.2.2	The results of these tests shall be included in the PLAR. Preliminary results shall be made available for confirmation by the NASA Student Launch management team at the competition launch.	The Payload Team will extract and document all analysis from any tests conducted by the payload in the PLAR Document.	Inspection	The PLAR document contains all test results from the competition flight.	NV	Payload Team	The PLAR document will contain all test results from the competition flight.
4.4	The HAUS's structure shall include an atmosphere isolated compartment to serve as living quarters for 4 STEMnauts. The compartment shall be enclosed and separated from the external atmosphere; No additional requirements for "living conditions" are included, but teams are encouraged to consider appropriate accommodations the STEMnauts may need for an extended excursion on a lunar or planetary body. STEMnauts are assumed to have all qualities typical of astronauts. It is up to teams to be creative in how to depict their four STEMnauts in the HAUS design. "Atmosphere isolated compartment" means the living quarters must be enclosed and separated from the external atmosphere. Pressure equalization holes are exempt from this isolation requirement.	The Payload Structures Lead will design a HAUS enclosure to serve as living quarters for 4 STEMnauts. The HAUS enclosure will be separate from the external atmosphere, with a hole to equalize pressure if deemed necessary.	Inspection	The Payload contains the HAUS enclosure where STEMnauts live separate from the external atmosphere.	IP	Payload Structures	HAUS design is located in Section 4.5.1.
4.4.1	The HAUS enclosure shall not incorporate or rely on the structural airframe (including couplers) of the launch vehicle to meet requirement 4.4.	The Payload Structures Lead will design the HAUS enclosure such that it does not incorporate or rely on any structural components of the launch vehicle.	Inspection	The HAUS enclosure does not incorporate or rely on any structural components from the launch vehicle.	IP	Payload Structures	HAUS design is located in Section 4.5.1.
4.5	The STEMnauts shall be safely retained within the HAUS during flight (no alternative launch seating or location is permitted).	The Payload Structures Lead will design the HAUS enclosure such that all STEMnauts are safely retained during the flight.	Inspection	The HAUS enclosure contains seating such that STEMnauts are safely secured for flight operations.	IP	Payload Structures	HAUS design is located in Section 4.5.1.

Table 7.6: 2025-2026 Payload Experiment Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
4.6	The payload shall not have any protrusions from the vehicle prior to apogee that extend beyond a quarter inch exterior to the airframe.	The Payload Structures Lead will design the payload such that there are no protrusions that extend more than a quarter inch outside the exterior of the airframe.	Demonstration	The payload is designed such that it does not protrude from the launch vehicle more than a quarter inch.	IP	Payload Structures	Preliminary payload design is outlined in Section 4.5.
4.7.1	Black powder and/or similar energetics are only permitted for deployment of in-flight recovery systems. Energetics will not be permitted for any surface operations.	The Payload Team will design the Payload such that it does not use Black Powder or any other similar energetics for any surface operations of the Payload.	Inspection	The payload will not utilize black powder or similar energetics to deploy.	V	Payload Team	See Section 4.5, showing that the payload design does not utilize black powder or similar energetics.
4.7.2	Any UAS weighing more than .55 lbs. shall be registered with the FAA and the registration number marked on the vehicle.	The Payload Team will register any UAS that weighs more than .55 lbs with the FAA and will follow all rules and regulations set by the FAA including labeling and markings on the UAS.	Inspection	The payload will not jettison from the launch vehicle.	V	Payload Team	See Section 4.5, detailing how the payload design does not utilize jettisoning components.

Table 7.7: 2025-2026 Safety Requirements

ID	SHALL Statement	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
5.1	The final checklists shall be included in the FRR report and used during the Launch Readiness Review (LRR) and any Launch Day operations.	The Project Management Lead alongside the Integration Lead and Safety Officer will write a final checklist for launch operations to be used during launch days. This Checklist will be included in FRR report, present during the LRR, and during any Launch Day operations.	**(1)** Inspection **(2)** Demonstration	**(1)** The launch day checklist is located in the FRR and is present during LRR.**(2)** The final launch day checklist is used during launch day including during VDF and PDF.	NV	Project Management, Integration & Safety	Preliminary Launch Checklists will be included in CDR Document.
5.2	Each team shall identify a student Safety Officer. See rule 5.2 for all guidelines pertaining to the student Safety Officer.	The team will democratically elect a Safety Officer. The Safety Officer will follow all rules in the NASA SL Handbook rule 5.2.	**(1)** Inspection **(2)** Demonstration	**(1)** Elected Safety Officer is defined in Proposal. **(2)** The Safety Officer follows all rules defined by rule 5.2 in the NASA SL Handbook.	IP	Project Management	**(1)** See section 1.5.1 of Proposal for identified Safety Officer. **(2)** The Safety Officer continues to follow all rules described in the NASA SL Handbook rule 5.2.

7.1.2 Vehicle Team Derived Requirements

Table 7.8: 2025-2026 Team Derived Vehicle Requirements

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
Functional Requirements								
LVF 1	All composite materials made for the launch vehicle SHALL be constructed using methods such that the resulting components are uniform in density.	Voids and delamination are weak points and reduce structural integrity. Additionally ensuring uniformity improves the reliability of our components and performance under launch conditions.	All composites manufactured lead by the Structures Lead and will be done so with a standardized curing, layup construction, and resin mixing procedures. Team members performing layups will be trained in proper fabrication techniques.	**(1)** Inspection **(2)** Demonstration	**(1)** All composites are visually inspected and found to have negligible voids, delamination, and other imperfections. **(2)** Standard curing, layup construction, and resin mixing procedures are followed during the manufacturing of composite components.	IP	Structures	**(1)** Section 7.4 details the manufacturing timeline for full-scale. **(2)** Section 3.2.4 details preliminary layup procedures. More detailed procedures for composite manufacturing will be described in the CDR document.
LVF 2	The LV SHALL contain uniform fin size and geometry.	Identical fins help maintain aerodynamic balance, such as location of CP, preventing unwanted roll, yaw and pitch during flight.	All composite layups for fin construction and cutting will follow identical procedures.	Inspection	Fins are visually compared and shown to have the same mass and geometry with negligible differences.	IP	Structures & Aerodynamics	Section 7.4 details the manufacturing timeline for full-scale. Section 3.4.1 describes fin manufacturing and design.
LVF 3	All bonding surfaces SHALL be prepared using proper surface prep standards (cleaning, sanding, alcohol wipe).	Proper surface preparation ensures strong composite bonds and prevents delamination. This helps ensure our product is consistently of high quality.	Team members performing bonding operations will follow surface preparation procedures led by the Structures Lead. Surfaces will be sanded, cleaned, and wiped with isopropyl alcohol prior to bonding.	**(1)** Inspection **(2)** Demonstration	**(1)** Composites bonded together show no adhesive failure when inspected. **(2)** Standard surface preparation procedures for bonding surfaces will be followed.	NV	Structures	**(1)** Section 7.4 details the manufacturing timeline for full-scale. **(2)** Section 3.3.2 describes preliminary surface prep procedures for composite airframes. More detailed procedures for surface preparations will be described in the CDR document.

Table 7.8: Team Derived Vehicle Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
LVF 4	Airframe, fins, and nosecone SHALL be manufactured such that the external surface is smooth.	Smooth Surfaces reduce drag, turbulence, and boundary layer separation, which improves stability, reduces off-nominal trajectories and makes for greater altitude performance.	All launch vehicle surfaces shall be sanded such that the external surface is smooth.	Inspection	Visual inspection shows no visible ridges or rough sanding marks. All external surfaces have a consistent texture and finish across seams and transitions.	NV	Structures	Section 7.4 details the manufacturing timeline for full-scale.
LVF 5	Apogee predictions SHALL be validated with at least three separate analysis programs.	Using different apogee prediction software reduces errors that may be prone in any one software. Additionally, different software have different tools that are able to be utilized by the Aerodynamics Lead. Increasing the accuracy of our predicted apogee and trajectory allows for a better understanding of how the launch vehicle will perform.	Aerodynamics Lead will run simulations in OpenRocket, RocketPy, and RasAERO	Inspection and Analysis	Apogee predictions from OpenRocket, RocketPy, and RasAERO will be included in CDR.	PV	Aerodynamics	Results from OpenRocket and RocketPy are shown in Section 3.10.1.
LVF 6	The Launch Vehicle Shall not be overstable, defined as having a static stability >4.	Overstability leads to weather-cocking which can greatly increase the launch vehicle's potential land radius. Over stability also increases the likelihood of oscillations and unpredictable behavior.	The Aerodynamics Lead will analyze the vehicle's stability using simulation tools. Tolerances for component masses and designs will be utilized and design changes will be made if necessary.	**(1)** Inspection **(2)** Analysis	**(1)** The launch vehicle has a stability less than 4 in its final launch configuration. **(2)** Simulated static margins are < 4 for all configurations.	PV	Aerodynamics	**(1)** VDF and PDF flights are described scheduled in Section 7.4. **(2)** Stability calculations are described in Section 3.10.4.
Design Requirements								

Table 7.8: Team Derived Vehicle Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
LVD 1	Fins SHALL be designed to withstand a minimum of 10 ksi bending loading.	Fins can experience high loads during flight and landing. Insufficient strength of materials can lead to breaking mid-flight which compromises vehicle stability. Conducting strength tests allows to better design a system such that it does not fail.	Composite sandwich test pieces will be fabricated and tested under three-point bending to determine their maximum bending loads. Once an effective layup method is identified, its results will be used to inform the fin construction process.	Test	Composite test pieces show a breaking point of more than 10 ksi.	V	Structures	Three point bending tests are described in Section 3.2.4
LVD 2	Composite Airframes and centering rings SHALL be designed to withstand axial compressive loads equivalent to twice the motors thrust.	Axial loads during launch operations have potential to damage or buckle the airframe. Designing and testing parts to withstand more loading that expects grants a factor of safety fulfilling reusability requirements.	Compression testing will be done on sample composite airframe material and verified to withstand twice the motors thrust forces.	Test	Compressive testing on sample composites show a breaking point of more than twice the motors thrust.	V	Structures	Compressive tests are described in Section 3.2.5.
LVD 3	Composite Bulkheads will be designed to withstand tension loading up to 300 lbs of force.	Bulkheads experience tensile loading during recovery events, ensuring that they are equipped to experience forces of 300 lbs allows them to safely carry the expected loads during handling, assembly, and launch without risk of failure.	A universal tensile testing machine will be utilized and confirmed to withstand at minimum 300 lbs of force.	Test	Tensile tests show Bulkheads withstanding loads greater than 300 lbs.	NV	Structures	Bulkhead testing will be documented in the CDR document.
LVD 4	Radio frequency transparent materials SHALL be used on all launch vehicle sections that may be transmitted through.	Radio frequency transparency ensures that GPS and communication signals are able to be received and transmitted, which are integral to the launch vehicle purpose.	All sections that need to transmit radio frequency will be made using fiberglass as opposed to carbon fiber to ensure radio frequency transparency.	Inspection	All transmitting sections of the launch vehicle are made of a radio frequency transparent material such as fiberglass.	IP	Structures	Material selection for the launch vehicle components is described in 3.3.1.

Table 7.8: Team Derived Vehicle Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
LVD 5	All airframe attachments points for rail buttons SHALL be reinforced.	Reinforced rail button attachments prevent structural damage at mounting interfaces.	The fin can will include supporting material surrounding the rail button locations.	Inspection	Fin can includes supports at the location of the rail buttons.	NV	Structures	Fin can design is described in Section 3.4.1.
LVD 6	Any hardware used to secure sections of the Launch Vehicle SHALL be designed to minimize drag on the launch vehicle.	Minimizing drag helps with aerodynamic performance as well as makes us more likely to reach our predicted apogee.	All hardware chosen to adjoin any different airframe sections will be small enough that they cause only minor drag increases.	Inspection	All adjoining hardware does not stick out of the rocket more than .5 inch.	NV	Structures	Adjoining sections are described in Section 3.3.1.
LVD 7	The motor mount SHALL be designed to securely retain the selected motor and transmit thrust loads to the airframe.	Proper motor mounts allow for thrust loads to be evenly distributed across the launch vehicle's airframe. This prevents high stresses in small regions which could lead to permanent damage.	The motor mount will be designed to withstand all expected loading during flight.	Inspection	The motor mount design retains the motor with no movement or looseness.	NV	Structures	Motor retention is described in Section 3.4.1. Section 7.4 details the manufacturing timeline for full-scale.
LVD 8	Fins SHALL be designed such that Fin Flutter velocity is 50% higher than expected velocity off the rod.	Fin Flutter occurs when fins vibrate at their natural frequency due to aerodynamic forces. This has potential to destroy the launch vehicle rapidly as the oscillations quickly get out of control.	Fins will be analyzed for natural frequencies, design adjustments will be made to raise the flutter velocity until it meets the requirement.	Analysis	Fin Flutter velocity is calculated to be 50% higher than expected velocity off the rod.	NV	Structures & Aerodynamics	Air Brakes help mitigate fin flutter effect, as shown in Section 3.10.7. Additional fin flutter calculations will be included in the CDR document.
Environmental Requirements								
LVE 1	All leftover epoxy-resin materials SHALL be properly disposed of per manufacturers instructions.	Improper disposal of epoxy resin can cause harm to individuals and the environment. By properly disposing of epoxy, damage is mitigated.	Team members will follow the manufacturer's disposal guidelines for all leftover epoxy and resin materials. Containers will be labeled, and epoxy waste will be collected in designated waste areas. Training on proper handling and disposal procedures will be provided to all team members working with materials.	Inspection	All epoxy waste is stored in labeled containers and disposed of properly once cured.	IP	Structures & Safety	Proper epoxy handling procedures are based off of SDS Sheets located in Reference [13].

Table 7.8: Team Derived Vehicle Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
LVE 2	All structural components SHALL be designed to operate nominally with an ambient temperature range of 25°F to 100°F.	Launch day conditions can vary in temperature. Temperature can then affect composite properties and adhesive performance. Proper design ensures consistent structural integrity.	Materials such as carbon fiber and fiberglass with temperature stability across the range of 25°F to 100°F will be chosen for all structural components of the launch vehicle.	**(1)** Analysis **(2)** Demonstration	**(1)** Analysis shows that materials used for structural components do not experience failure at temperatures ranging from 25°F to 100°F. **(2)** Structural components do not show failure at temperatures ranging from 25°F to 100°F. **(2)** Structural components do not show any sign of failure due to temperature ranges during launch days.	NV	Structures	**(1)** Material properties will be described in the CDR document. **(2)** VDF is scheduled for the spring, detailed in Section 7.4.
LVE 3	All structural components SHALL be resistant to ambient humidity up to 90% relative humidity without degradation in adhesive or performance.	Launch day conditions can vary in humidity, which can weaken adhesives or affect composite performance. Proper design mitigates any effects humidity has on launch vehicle structural performance.	Materials and adhesives will be selected based on proven resistance to high humidity.	**(1)** Analysis **(2)** Demonstration	**(1)** Analysis shows that materials used for structural components do not experience failure in ambient humidity up to 90%. **(2)** Structural components do not show any sign of failure due to humidity levels up to 90%.	NV	Structures	Material properties will be described in the CDR document. **(2)** VDF is scheduled for the spring, detailed in Section 7.4.

Table 7.8: Team Derived Vehicle Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
LVE 4	Structural components SHALL be capable of withstanding ground impact from soil, gravel or sparse vegetation without compromising future reusability.	Launch vehicles are always susceptible to hard landings. In order for the launch vehicle to be reusable, structural components must be able to withstand potential landing loads.	Structural components will be designed with impact-resistant materials including composites with high toughness. Drop tests or impact simulations will be conducted on sample components to verify survivability.	**(1)** Testing **(2)** Demonstration	**(1)** Drop tests show structural components are capable of withstanding expected landing loads. **(2)** VDF verifies structural components can survive landing forces.	NV	Structures	**(1)** Testing schedule is described in Section (schedule). **(2)** VDF is planned for the spring, described in Section 7.4.
LVE 5	Aerodynamics simulations SHALL perform analysis quantifying apogee variation for wind speeds up to 20 mph.	Wind can greatly affect flight trajectory and apogee. By performing simulations under extreme environmental conditions allows for the team to ensure designs for safe and predictable launches.	Aerodynamics Lead will conduct simulations with wind speeds up to 20 mph.	Analysis	Trajectory simulations demonstrate apogee variation across wind speeds from 0 to 20 mph to ensure safe launch operations.	V	Aerodynamics	Trajectory simulations with varying wind speeds are described in Section 3.10.3
Safety Requirements								
LVS 1	All Structural elements of the launch vehicle SHALL be designed to withstand launch loads with a minimum factor of safety of 1.5.	Safety factors ensure structural reliability in unpredictable circumstances where loads could be more than expected. A factor of safety of 1.5 provides	All structural elements of the launch vehicle will be designed with a factor of safety of 1.5 minimum.	Analysis	Analysis shows that structural components give a factor of safety of 1.5 when exposed to expected launch loads.	IP	Structures	Structural component strength calculations are shown in Section 3.3
LVS 2	When cutting or sanding composites, proper particulate masks shall be worn at all times, accumulation of dust in air shall be minimized.	Composite particulate matter has tendencies to be very dangerous to people, proper use of PPE and sanding techniques helps mitigate these harms.	When cutting or sanding composites, all members will be required to wear proper particulate masks. When dust accumulation is expected to be large, the vacuum will be used as a mitigation measure. All workspaces will be cleaned after use.	**(1)** Inspection **(2)** Demonstration	**(1)** The workspace is inspected for composite debris after working and is clean. **(2)** Proper PPE is worn when sanding or cutting composite components.	IP	Structures & Safety	**(1)** & **(2)** Proper composite manufacturing procedures are based off of SDS Sheets located in Reference [13]. More detailed procedures will also be described in CDR.

Table 7.8: Team Derived Vehicle Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
LVS 3	Manufacturing of the Launch Vehicle SHALL be completed a minimum of 24 hours before any and all launches.	Manufacturing the launch vehicle ahead of time ensures thorough quality checks, proper curing of adhesives, and final assembly without schedule pressure, reducing the risk of errors.	The manufacturing schedule will be planned to complete all fabrication, assembly, and curing steps at least 24 hours prior to launch.	Inspection	All launch vehicle structural components are ready for assembly at least 24 hours before launch.	IP	Structures & Safety	See Section 7.4 for full-scale manufacturing schedule.
LVS 4	When cutting any components using any power tool, components SHALL be properly secured such that they do not move due to any forces from the power tool.	By properly securing components before modifying them, individuals mitigate harm that could come to them if the component were to shift, vibrate, or spin unexpectedly, which could cause injury. Moving parts also can lead to inaccurate dimensions which can lead to failure of systems. Ensuring components are secured in place mitigates these risks.	All components will be clamped, held in jigs, or otherwise secured before any cutting or machining operations. Personnel will be trained in proper techniques and use of clamps, vises, and other securing tools.	Demonstration	Before any cutting or machining operations, components will be secured such that they will not move during cutting/machining operation.	IP	Structures & Safety	Proper manufacturing will be included in CDR.
LVS 5	When using any power tools, proper PPE such as safety glasses SHALL always be enforced.	Proper PPE use protects individuals from potential flying debris, sparks or other fragments that could cause harm.	All personnel using power tools of any kind will be properly trained and proper PPE will be worn at all times.	**(1)** Inspection **(2)** Demonstration	**(1)** All team members are required to pass a safety quiz developed by the Safety Officer. **(2)** Any team member using power tools will utilize proper technique and PPE usage.	IP	Safety	**(1)** Safety quiz is documented in Reference [12]. **(2)** Proper power tool procedures will be included in CDR.

Table 7.8: Team Derived Vehicle Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
LVS 6	All solvent-based cleaning methods SHALL be conducted in a well ventilated area.	Well-ventilated areas ensure that solvent-based cleaning methods that release potentially harmful volatile compounds are dispersed, reducing the risk of inhalation and fire with flammable vapors.	All solvent based cleaning operations will be performed either outside or in a well ventilated area.	Demonstration	Cleaning operations are conducted in designated ventilated areas.	IP	Structures & Safety	Proper cleaning procedures will be included in CDR.

7.1.3 Recovery Team Recovery Requirements

Table 7.9: 2025-2026 Team Recovery Vehicle Requirements

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
Functional Requirements								
RF 1	Parachutes SHALL be packed using reputable rocketry manufacturers guidelines for folding and packing.	Packing parachutes properly minimize the risk for packing related malfunctions. Tangled shroud lines, packing density, shock cord placement, can all affect how the canopy opens. The best way to ensure optimum performance is to follow the manufacturers instructions.	All team members responsible for packing parachutes will be trained in manufacturer recommended packing methods, and each packed parachute will undergo visual inspection prior to flight.	**(1)** Inspection **(2)** Demonstration	**(1)** Parachutes are inspected by a Team Lead after folding and packing to verify correct packing. **(2)** Parachutes are packed using reputable rocketry manufacturers guidelines.	NV	Recovery	**(1)** Parachute packing will be included in launch checklists in CDR Document. **(2)** Parachute packing methods will be described in CDR.
RF 2	Any Black Powder Charges SHALL be calculated and tested using "ejection tests".	Ground testing our black powder charges gives us confidence that the recovery events will go as planned. If recovery events do not occur, the launch vehicle can come in ballistic which is extremely dangerous and should be avoided.	Once launch vehicle structural components are completed, they will dry fitted and the launch vehicle will be prepped in its recovery launch configuration. Calculated black powder calculations will be packed and ejection testing will be done to verify proper separation.	**(1)** Analysis **(2)** Demonstration	**(1)** Black powder charges are calculated to ensure proper separation of independent sections of the launch vehicle. **(2)** Ejection testing shows proper separation of independent vehicle sections as well as deployment of recovery harnesses.	PV	Recovery	**(1)** Black powder calculations are described in Section 3.8.9. **(2)** Ejection testing schedule is described in Section 7.4.
RF 3	GPS Recovery electronics SHALL be tested before use on the launch vehicle.	Testing GPS recovery electronics ensures that we will be able to successfully recover the launch vehicle. Becoming familiar with the GPS' systems makes field recovery easy.	Prior to launch, GPS recovery electronics will be ground tested for proper operation, including signal acquisition, tracking accuracy, and data transmission to ground station hardware.	**(1)** Test **(2)** Demonstration	**(1)** Ground tests verify GPS recovery electronics work properly. **(2)** GPS units are acquired and location data is properly transmitted during VDF flight.	NV	Recovery	**(1)** Ground tests will be located in the CDR document. **(2)** VDF is scheduled for the spring, detailed in Section 7.4.

Table 7.9: Team Derived Recovery Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
RF 4	Post-flight inspections SHALL be done for any and all recovery components including parachutes, harnesses and altimeter housings.	During flight recovery systems often go through heavy loading, and they are susceptible to failure if taken to their limit. Post flight inspections of recovery equipment ensures that we can keep using our best material and replace material as needed.	After launch, the recovery Lead will inspect all recovery components for signs of wear, fraying, deformation or other damage. Any damaged components will be repaired or replaced.	Inspection	Post-flight inspection checks are completed for all recovery components. Visual inspection confirms no damage or wear beyond acceptable limits and any damaged components are replaced, or repaired before reuse.	NV	Recovery	VDF is scheduled for the spring, detailed in Section 7.4.
RF 5	The full recovery system SHALL be dry fitted at least once before Launch Day.	Oftentimes when designing components tolerances for where parts need to go can be off, so dry fitting ensures that our vehicle will fit as intended. Dry fitting ahead of launch day lets us make changes if need be before it's too late.	A dry fitting of all components of the recovery system will be done prior to launch day. Any fitment issues will be corrected and adjustments will be verified through an additional inspection.	Inspection	Successful full recovery system dry fits prior to launch. All components assemble and disassemble without interference and no additional modifications are required.	NV	Recovery	VDF is scheduled for the spring, detailed in Section 7.4.
RF 6	Black Powder calculations SHALL utilize reputable methods of calculations.	Using reputable, established methods to calculate black powder charges ensures predictable, safe and effective performance of our recovery events. When combined with ejection testing, using a reputable calculation method allows for safer, more consistent and reliable deployment of recovery events.	All black powder charge calculations will be performed using verified and established methods, with each calculation independently checked using Chuck Pierce's Black Powder Ejection Charge Calculator	Analysis	Black powder charges are verified using Chuck Pierce's Black Powder Ejection Charge Calculator.	V	Recovery	Black powder calculations are described in Section 3.8.9

Table 7.9: Team Derived Recovery Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
RF 7	All team fabricated Parachutes SHALL be properly tested and verified to withstand all loads expected during recovery events.	By properly testing house made parachutes, we ensure that the non-commercial hardware meets the same vigorous requirements that the professionally manufactured options allow for.	Team fabricated parachutes will undergo testing to verify they meet correct deployment and show correct descent values. Parachute metrics will be documented and verified to match predicted values.	**(1)** Test **(2)** Inspection	**(1)** Parachutes testing verifies calculated performance parameters. No tearing, seam failure, or material damage is observed during testing. **(2)** All parachute values are documented in CDR.	NV	Recovery	**(1)** Parachute testing will be documented in the CDR document. **(2)** Material values will be described in CDR.
RF 8	Descent velocity SHALL be kept below 120 fps at all times during descent.	High velocities during descent cause high strain on recovery systems when they deploy. Descent speeds increase kinetic energy exponentially, which causes major stress on the main parachute when it deploys, which could cause damage to the parachute, shroud lines or air frame via zippering. Maintaining a constant speed allows for us to not let our kinetic energy get out of control while also getting to the ground efficiently.	The Recovery Lead will select appropriately sized drogue and main parachutes to ensure the total descent profile remains under 120 fps. Analysis will be done to verify that descent velocities remain within the limit.	**(1)** Analysis **(2)** Demonstration	**(1)** Calculated descent under selected main and drogue parachutes show that the launch vehicle will experience velocities under 120 fps. **(2)** During VDF, flight data confirms that the launch vehicle falls with velocity under 120 fps.	PV	Recovery	**(1)** Descent velocity is calculated and described in Section 7.5. **(2)** VDF is scheduled for the spring, detailed in Section 7.4.

Table 7.9: Team Derived Recovery Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
RF 9	Voltages in the batteries selected for recovery systems SHALL be tested using a multi-meter and must remain within manufacturer recommendations.	Testing recovery battery voltages and ensuring they remain within their manufacturers recommended limits is essential for reliable and safe operation of the electronic components. Maintaining a proper voltage helps us confirm that our recovery electronics will operate as functioned and in a predictable fashion.	During the launch checklist, batteries will be tested using a multimeter and shown to hold a voltage within manufacturer recommendations.	Inspection	Launch Checklists show voltage within manufacturer recommenda-tions. Checklists are documented in CDR documents.	NV	Recovery	Preliminary Launch checklists will be located in CDR.
Design Requirements								
RD 1	Wires for Recovery SHALL be color coded	Color-coding recovery cables allows for easier assembly and enhance safety during both pre and post flight inspections. Different colors allow for clear labeling of wires, making it easier to know what goes where and aids with any troubleshooting that needs to be done.	The Recovery Lead will utilize colored wires in the avionics bay design. Color coding will be specified and documented in CDR.	Inspection	The color coding scheme is documented in CDR.	NV	Recovery	Color codes for recovery electronics will be described in CDR.
RD 2	Electrical connectors SHALL not be used as structural elements	Connectors are designed to provide electrical continuity, not to bear mechanical loads such as tension or compression. Using them as structural components can lead to deformation, cracking, and potential electrical disconnection, which can then lead to recovery events not deploying, meaning the launch vehicle would come in ballistic.	Electrical connections will be installed solely for electrical connections. Electrical routing may be secured with zip ties but will not be used to transmit mechanical forces.	Inspection	Inspection confirms no electrical connectors are subjected to structural stresses.	NV	Recovery	Avionics bay design is described in Section 3.9.1.

Table 7.9: Team Derived Recovery Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
RD 3	Secondary black powder charges SHALL generate a peak pressure at the separation interface of at least 150% of the minimum pressure required to separate the vehicle.	Designing the secondary charge to produce a peak pressure equal to 150% of the minimum separation pressure provides a conservative margin that ensures reliable separation while avoiding unnecessarily high loads	Once necessary pressure values to separate the launch vehicle are calculated, a multiplier of a minimum of 1.5 will be applied to the amount to determine the secondary black powder amount.	Inspection	The secondary black powder charge is a minimum of 150% size of the primary black powder charge.	IP	Recovery	Secondary black powder charge calculations are described in Section 3.8.9.
RD 4	The use of twist wire nuts SHALL not be allowed.	Twist-on wires are prone to loosening under vibrations which are common in launch environments. Loose connections can lead to recovery events not deploying which can lead to the vehicle coming in ballistic.	All electrical connections in the recovery system will use soldered, crimped, or other vibration-resistant methods. Twist-on wire nuts will not be utilized.	Inspection	Upon inspection of the recovery system wire connections twist-on wires nuts are not used.	NV	Recovery	Recovery wiring will be described in CDR.
RD 5	All recovery Shock Cords SHALL be chosen to receive expected strain loads from recovery events with a factor of safety of at least 10.	Shock cords undergo high tensile loads under recovery events. Applying a factor of safety of 10 allows for us to account for material wear, environmental effects, and unexpected loads that our shock chord could experience.	Only shock cord that has a breaking strength rated higher than 10x the amount of loading expected.	Analysis	Chosen shock cord has a manufacturer-rated tensile strength a minimum of twice the expected loading.	V	Recovery	Shock cord selection is described in Section 3.8.8.
RD 6	Shock cord lengths SHALL be chosen such that distances between separating sections will be greater than ten feet during descent.	Maintaining a minimum distance of ten feet of shock cord between each separating sections ensures that different rocket sections will not slam into each other during descent, causing damage to the vehicle and tangling the shock cord.	Shock cord lengths will be calculated to give at minimum 10 ft of distance between each independent section.	Inspection	Shock cord lengths are inspected and allow for a minimum of 10 ft of clearance between each independent section.	V	Recovery	Shock cord lengths are described in Section 3.8.8.
Environmental Requirements								

Table 7.9: Team Derived Recovery Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
RE 1	All Recovery insulation SHALL be biodegradable.	During recovery events, insulation falls from the rocket such that it is not recoverable and contaminates the launch field. Because of this, biodegradable insulation is used to reduce the environmental impact and respect those who allow us to launch on their field.	Cellulose based insulation will be utilized for recovery packing.	Inspection	All insulation used in recovery systems is inspected to be biodegradable.	NV	Recovery	Choice of insulation will be described in CDR.
RE 2	Avionics housings SHALL be protected against dust and debris during transport, launch and field recovery.	Dust, dirt, and other small particulate matter can interfere with connectors, switches, and other electrical components on recovery electronics. Protecting the housing of the electronics from particulates helps mitigate possible malfunctions.	The avionics bay will be sealed such that minimal dust and debris can enter. Transportation containers and field storage procedures will further protect the avionics bay from particulate infiltration.	Inspection	Inspection confirms protective measures are in place during transport and field handling.	NV	Recovery	Avionics Bay design is described in Section 3.9.1. Launch checklists including transportation and handling of the avionics bay will be in CDR.
RE 3	All recovery electronics SHALL deliver nominal voltage across ambient temperatures ranging from 25 $^{\circ}$F to 100 $^{\circ}$F.	Temperature extremes can affect battery performance, and ensuring recovery batteries can withstand all possible launch temperatures helps mitigate any potential battery performance issues that arise from temperature changes.	Batteries will be selected that have temperature stability ranges within 25°F to 100°F and are capable of providing stable voltage through this range.	Inspection	Manufacturer specifications confirm batteries operate correctly across the full temperature range.	V	Recovery	Recovery batteries are specified in Section 3.8.4.

Table 7.9: Team Derived Recovery Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
RE 4	All disposable recovery materials SHALL collected and disposed of in accordance with NAR/Tripoli range safety and environmental rules.	Single use items for recovery systems such as blue tape have the potential to contaminate the environment. By adhering to environmental rules by properly disposing of expendable recovery components minimizes our environmental impact as well as promotes good stewardship.	All disposable recovery will be collected on launch day and properly disposed of according to environmental guidelines.	Demonstration	All disposable materials are collected and properly disposed of per launch range rules.	NV	Recovery	Disposable material management during launches are described in launch checklists in CDR.
RE 5	All unusable black powder charges SHALL be neutralized and disposed of in accordance with manufacturer and environmental standards.	Neutralizing and properly disposing of black powder charges by following manufacturers instructions prevents accidental ignition, reduces environmental contaminating and ensures legal compliance. Following Safety-Data-Sheets ensures that black powder is handled in a safe and controlled manner.	All unusable black powder will be neutralized per manufacturers instructions. After neutralization, they will be properly disposed of per environmental standards.	Demonstration	All unusable black powder is properly neutralized and disposed of per manufacturers instructions.	IP	Recovery	Proper black powder disposal procedures based off of SDS Sheets located in Reference [13].
Safety Requirements								
RS 1	Arming devices SHALL be accessible without having your head within 12 (in) from the outer diameter.	Keeping personnel's head clear of the launch vehicle's outer diameter reduces the risk of injury should accidental ignition of recovery events were to occur.	Recovery electronics will be chosen and placed in the avionics bay such that operation is capable at a minimum of 12 inches away from the exterior of the launch vehicle.	Inspection	Arming devices are positioned such that team members can access them while maintaining >12 inches distance.	NV	Recovery	Avionics bay design is outlined in Section 3.9.1.

Table 7.9: Team Derived Recovery Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
RS 2	All LiPo Batteries used SHALL be stored in approved storage containers for storage and transportation.	LiPo batteries are highly energy dense and can ignite or explode if punctured. Approved LiPo bags are designed to contain flames and vent hot gasses safely so storing them there mitigates potential safety hazards should a LiPo battery ignite.	All LiPo batteries utilized will be stored and transported in fire-resistant LiPo bags or containers. Storage containers will be properly labeled.	Inspection	LiPo batteries are stored and transported in approved containers.	IP	Recovery & Safety	Proper LiPo Storage and handling will be described in CDR.
RS 3	All Ejection Testing testing SHALL be conducted more than 24 hours before intended launch.	24 hours gives recovery systems time to inspect, analyze and mitigate any errors that might arise with ejection testing. Additionally, this reduces the risk of last-minute failures and ensures that rushed decisions are not made immediately prior to launch.	Ejection tests will be scheduled to occur a minimum of 24 hours before launch.	Inspection	Ejection testing is completed >24 hours before launch.	IP	Recovery	Ejection testing schedule is described in Section 7.4.
RS 4	All Altimeter Arming procedures SHALL be documented and known by all necessary personnel including altimeters beeps and programming.	Requiring that all necessary personnel know altimeter arming and disarming procedures is important for safe launch operations. Clear, written procedures help reduce human error during high-stress preflight activities. Having multiple people knowledgeable about arming and disarming altimeters also provides more brainpower to dedicate to problem solving should problems arise.	All altimeter procedures will be documented in the launch checklist. Team members responsible for launch operations will receive training and demonstrate competency in arming, disarming, and programming altimeters.	Inspection	Altimeter arming procedures are documented in the launch checklist.	NV	Recovery	Altimeter arming procedures will be described in launch checklists in CDR.

Table 7.9: Team Derived Recovery Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
RS 5	All Altimeters programmed SHALL be verified by at least two additional personnel other than the Recovery Lead.	Independent checks ensure that altimeters are configured in the right way. Similarly to how running simulations on different software can find bugs within one of said software, running altimeter configurations by multiple people reduces errors in the programming.	After initial programming, two additional personnel not involved in the initial programming will verify recovery altimeters are programmed correctly.	Demonstration	Two independent personnel confirm altimeter programming matches flight requirements.	NV	Recovery	Recovery schedule is described in Section 7.4.
RS 6	All personnel recovering the launch vehicle SHALL wear proper PPE and fire-proof gloves if they are handling the launch vehicle.	Field recovery oftentimes includes dealing with deployed parachutes, shock cords, black powder charges, and sometimes hot components. Wearing proper PPE is required at all points, and fire-proof gloves protect hands during the handling of pyrotechnics that one might see when recovering a launch vehicle.	The launch checklist will require all personnel responsible for recovering the launch vehicle to wear proper PPE.	Inspection	All field recovery personnel are observed wearing the required PPE during recovery operations.	NV	Recovery & Safety	Field recovery operations will be described in launch checklists described in CDR.

7.1.4 Air Brakes Team Derived Requirements

Table 7.10: Team Derived Air Brakes Requirements

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
Functional Requirements								
AF 1	Air Brakes Systems SHALL only be capable of braking, no controlling pitch, yaw or roll.	Limiting Air Brakes to just braking ensures that they do not introduce unintended control forces that might destabilize the vehicle during flight. By restricting their behavior to braking, the Air Brakes system maintains predictable flight behavior and simplifies design and testing.	The Air Brakes system is designed such that they only affect drag and do not introduce any moments to the launch vehicle while deployed.	Analysis	Air Brakes simulations show that they only introduce deceleration, no moments are observed.	IP	Aerodynamics	Section 3.10.7 details the effects of the air brake system on rocket trajectory.
AF 2	The Launch Vehicle SHALL include a camera to verify deployment of the Air Brakes system.	While acceleration data and code logs are useful, visual data allows for a redundancy check to confirm that Air Brakes deploys. A camera also allows for valuable recovery and payload data should something go awry.	An on board camera will be integrated into the launch vehicle with a clear view of the Air Brakes deployment.	**(1)** Inspection **(2)** Demonstration	**(1)** A camera mount will be integrated on the launch vehicle. **(2)** On board camera footage from VDF confirms proper functionality of the camera and capture of the Air Brakes system.	NV	Structures & Aerodynamics	**(1)** Camera location and design will be included in the CDR document. **(2)** VDF is scheduled for the spring, detailed in Section 7.4.

Table 7.10: Team Derived Air Brakes Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
AF 3	The Air Brakes system SHALL remain retracted until the launch vehicle's boost phase has ended.	Deploying Air Brakes during the boost phase could introduce unwanted drag compromising trajectory and increasing structural loads on air brake fins and attachment points. By waiting to deploy the Air Brakes until the boost phase is complete, the system ensures that aerodynamic surfaces only influence the vehicle when deceleration and descent are intended.	Air Brakes deployment is designed to trigger only after motor burnout state is reached.	Inspection	On board camera footage from VDF shows that Air Brakes deploy after motor burnout.	NV	Aerodynamics	VDF is scheduled for the spring, detailed in Section 7.4.
AF 4	In deployed state Air Brakes SHALL not be directly forward of launch vehicle fins.	Air Brakes drag plates do not deploy over fins/ interfere with launch vehicle's fins aerodynamics	Structures and Aerodynamics Lead will design the Air Brakes system and the Launch Vehicle structure such that Air Brake fins will not deploy directly forward of the launch vehicles fins.	Inspection	Deployed fin location for the Air Brakes system are not directed immediately forward of the launch vehicle fins.	IP	Structures & Aerodynamics	Air Brake system location is described in Section 5.3
Design Requirements								
AD 1	Air Brakes gear mechanisms SHALL be designed such that all fins retract and deploy simultaneously.	Requiring that all fins work together to retract and deploy at the same time ensures that the system does not force any unwanted moments on the launch vehicle, which could change the launch vehicle's pitch or yaw, or roll.	Air Brakes are designed to utilize a gear system that deploys all fins at once.	**(1)**Test **(2)** Demonstration	**(1)** All fins deploy and retract simultaneously in ground tests **(2)** VDF confirms all fins retract and deploy simultaneously.	IP	Aerodynamics	**(1)** Ground testing for Air Brakes deployment will be documented in CDR. **(2)** VDF is scheduled for the spring, detailed in Section 7.4. Air Brakes fin deployment is described in Section 5.3.

Table 7.10: Team Derived Air Brakes Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
AD 2	The Air Brakes system SHALL utilize state-based software.	State-based software allows us to control the Air Brakes and respond deterministically at predefined flights, which reduces unexpected behavior in transient inputs.	The Air Brakes control software will be implemented using a state machine architecture, with explicit states for standby, motor burn, coast, free fall and landing.	**(1)** Inspection **(2)** Demonstration	**(1)** Air Brakes software is shown to use a state base architecture. **(2)** Air Brakes responds deterministically to flight states in during VDF.	PV	Aerodynamics	**(1)** Air Brakes software design is described in Section 5.3.4. **(2)** VDF scheduled for the spring, Section 7.4.
AD 3	Air Brake aerodynamic surfaces SHALL have a minimum safety factor of 2.0 under expected aerodynamic loads experienced during ascent.	A factor of safety of two provides a conservative margin to account for materials, manufacturing, load uncertainties and potential unexpected in flight conditions.	Aerodynamic surfaces of the Air Brakes will be designed and tested to withstand twice the expected aerodynamic loads during deployment.	Test	Physical testing demonstrates no failure or permanent deformation under simulated aero-loads.	NV	Aerodynamics	Physical testing on Air Brakes fin system will be included in CDR.
AD 4	The Air Brakes System SHALL be sealed such that Air Brakes barometric pressure data will not be adversely affected from deployment.	Uncontrolled air flow can cause pressure spikes in the Air Brakes barometric pressure sensor data, which can lead to unreliable results.	Air Brakes are designed with sealing methods to isolate the barometric pressure sensor used in Air Brakes. Sealing methods will be verified during ground testing.	**(1)** Test **(2)** Inspection	**(1)** Ground testing confirms sealing methods are effective. **(2)** Post flight data obtained from barometric pressure sensors verifies the functionality of the sealing method.	NV	Aerodynamics	**(1)** Air Brakes sealing testing will be documented in the CDR document. **(2)** VDF is scheduled for the spring, detailed in Section 7.4.
Safety Requirements								
AS 1	All Air Brakes systems SHALL be designed and fabricated more than 24 hours before planned Launch.	Completing the design and fabrication of Air Brakes well in advance ensures that all components can be properly dry fit before launch day. Enforcing this timeline also reduces last-minute errors and rushing, and better ensures that Air Brakes is ready for safe predictable flight.	Air Brakes will be fabricated and fully assembled at least one day before launch.	Inspection	Air Brakes are fully manufactured, assembled and dry fitted inside the launch vehicle >24 hours before launch.	NV	Aerodynamics	Air Brakes development timeline is described in Section 7.4.

Table 7.10: Team Derived Air Brakes Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
AS 2	The Air Brakes System SHALL default to a neutral state if primary system power is lost. A neutral state is defined as one which does not apply any moments to the launch vehicle.	Incorporating a fail safe such that if power is lost the Air Brakes default to a neutral state ensures safe and predictable vehicle behavior if something were to go wrong and Air Brakes decided to take that state.	Air Brakes are designed such that they do not apply moments during deployment, ensuring if power is lost and Air Brakes cannot retract, they remain in a neutral state.	Analysis	Simulations show that Air Brakes do not exert any moments on the launch vehicle regardless of deployment.	IP	Aerodynamics	Section 3.10.7 details the forces of the air brake system on rocket.
AS 3	The Air Brakes System SHALL be capable of being disarmed using a physical switch.	Arming the Air Brakes system with a mechanical switch once arrived to the field ensures that power is being conserved.	A mechanical switch will be integrated into the Air Brakes system to arm and disarm the system.	Inspection	Air Brakes are capable of being armed and disarmed mechanically.	NV	Aerodynamics	Air Brakes arming mechanism will be documented in the CDR document.

7.1.5 Payload Team Derived Requirements

Table 7.11: Team Derived Payload Requirements

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
Functional Requirements								
PF 1	The Payload SHALL be capable of being fully retained in the nose cone section of the launch vehicle.	Integrating the payload within the nose cone offers sufficient volume while enabling payload mass simulation location on the subscale.	Payload will be designed such that it fits inside the nosecone.	Inspection	Payload fits completely and securely inside the nosecone section.	IP	Payload Structures	Payload location is specified in Section 3.3.1. Payload sizing is described in Section 4.5.
PF 2	The payload lander system SHALL be capable of fully removing itself from the nosecone section autonomously.	Autonomous separation simplifies the payload design in that it does not require active communication with the ground team. Such systems could fail so removing them and depending on a state based system removes complexity from the design.	The payload lander will be equipped with a mechanical release mechanism triggered by a state-based command. Ground testing will verify successful autonomous separation under various conditions.	**(1)** Test **(2)** Demonstration	**(1)** Testing shows autonomous ejection of the payload lander. **(2)** PDF verifies the autonomous ejection of the payload lander.	NV	Payload Team	**(1)** Payload testing schedule is outlined in Section 7.4. Results will be included in the CDR document. **(2)** PDF is planned for the spring, outlined in Section 7.4.
PF 3	The payload lander SHALL be capable of recognizing and orienting itself upright upon landing and deploying from the nosecone section.	Self orientation is what allows the payload lander to guarantee itself in a position such that the auger can successfully collect a sufficient amount of soil.	The payload lander will include sensors and an actuation mechanism that can detect orientation and correct itself if necessary.	**(1)** Test **(2)** Demonstration	**(1)** Ground testing shows the payload lander is capable of self-righting in various testable field conditions. **(2)** PDF shows payload lander is capable of self-righting on launch field conditions.	NV	Payload Team	**(1)** Payload testing schedule is outlined in Section 7.4. Results will be included in the CDR document. **(2)** PDF is planned for the spring, outlined in Section 7.4. Payload lander self-righting design is described in Section 4.5.1.
PF 4	The Payload SHALL be capable of detecting different states of the launch including ascent, descent, and landing.	Accurately sensing flight states allows the control software to execute phase-dependent actions such as deployment, self-righting, auger deployment, and sample testing.	Payload software will work in tandem with sensors to create a state machine architecture, with explicit states for boost, coast, descent and landing phases.	Demonstration	PDF shows that the payload is capable of detecting different states during all points in launch vehicle flight.	NV	Payload Team	PDF is planned for the spring, outlined in Section 7.4. Payload state architecture will be described in the CDR document.

Table 7.11: Team Derived Payload Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
PF 5	The Payload SHALL have a combined weight of no more than 8.5 (lbs).	Maintaining a light weight limit allows us to accurately model our launch vehicle, meaning our apogee simulations are accurate.	The payload will be designed with lightweight materials and components to remain under the required weight limit.	Inspection	The total measured payload weight does not exceed the specified limit.	IP	Payload Team	Preliminary payload mass is described in Section 4.5.2.
PF 6	The Payload SHALL retain the soil in a contaminant free chamber for testing.	Contamination from flight residue, airborne particulates and other launch vehicle specific chemicals could alter the pH, nitrate content, or electrical conductivity of collected soil samples, keeping samples secure and in a separate chamber mitigates these contaminants.	The payload soil chamber will be designed such that it minimizes contaminants.	Inspection	Collected soil is inspected and only negligible amounts of contaminants are found.	IP	Payload Structures	Payload soil chamber design is described in Section 4.5.1
PF 7	The payload lander SHALL remain in an upright orientation during the soil collection process.	Upright orientation ensures that the auger drill has proper engagement with the ground such that soil can be collected. Additionally it minimizes lateral forces on the payload that could contribute to concomitant or destabilize the payload.	The payload lander will include orientation sensors and a self-righting mechanism that is capable of detecting and correcting any orientation that is not upright.	**(1)** Test **(2)** Demonstration	**(1)** The payload lander remains upright during soil collection in all tested scenarios. **(2)** PDF verifies that the payload lander remains upright during soil collection.	NV	Payload Team	**(1)** Payload testing schedule is outlined in Section 7.4. Results will be included in the CDR document. **(2)** PDF is planned for the spring, outlined in Section 7.4. Preliminary payload lander self-righting mechanism is described in Section 4.5.1.
Design Requirements								
PD 1	The Payload SHALL fit entirely inside the nosecone, leaving room for shock cord connections.	Leaving enough room for all recovery components ensures that the launch vehicle will meet all recovery requirements.	Payload will be designed in the nosecone such that recovery components fit with adequate room for recovery operations.	Inspection	Payload fits completely inside the nose cone without obstructing shock cord connections.	IP	Payload Structures	Payload nosecone configuration with recovery in mind is defined in Section 3.3.3

Table 7.11: Team Derived Payload Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
PD 2	The Soil containment chamber SHALL hold a minimum of 75 (mL) of soil.	Collecting 75 (mL) of soil gives a capacity of 150% of the NASA required amount, which provides redundancy in cases of minor spillage or loss during transfer of soil. Additionally, the extra soil allows for more reassurance that the sensor components can gather an accurate reading.	The soil collection chamber will be designed to hold a minimum of 75 (mL) of soil.	Inspection	Soil collection chamber volume meets or exceeds 75 (mL).	IP	Payload Structures	Soil collection chamber is described in Section 4.5.1.
PD 3	The interface between the Payload and the nosecone SHALL include features to ensure smooth ejection.	Including alignment features allows for the payload to be deployed in a manner that is reliable and repeatable, and reduces potential lateral forces on the payload.	The nose cone and payload interface will include guides or rails to assist in smooth ejection. Ground tests will verify that the payload ejects consistently and without misalignment.	**(1)** Test **(2)** Demonstration	**(1)** Payload ejects smoothly and consistently in all test scenarios with no misalignment during deployment. **(2)** Payload ejects smoothly upon landing during PDF.	NV	Payload Structures	**(1)** Payload testing schedule is outlined in Section 7.4. Results will be included in the CDR document. **(2)** PDF is planned for the spring, outlined in Section 7.4. Payload lander ejection mechanism is described in Section 4.5.1.
PD 4	The payload SHALL log timestamps of all operations for NASA verification as well as post launch analysis.	Accurate time-stamped data ensures that each phase of the payload mission can be correlated to a launch vehicle state. This allows for traceability and verification when testing systems.	Payload software and sensors will work in tandem to timestamp all states and operations. Data will be recorded for all major operations, stored securely, and retrieved after flight for analysis. Ground testing will confirm functionality of time-stamp keeping systems.	**(1)** Test **(2)** Demonstration	**(1)** Ground tests confirm the timestamp keeping system. **(2)** All critical operations are logged with accurate timestamps during PDF.	NV	Payload Team	**(1)** Payload testing schedule is outlined in Section 7.4. Results will be included in the CDR document. **(2)** PDF is planned for the spring, outlined in Section 7.4.

Table 7.11: Team Derived Payload Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
PD 5	Auger operation SHALL be controlled such that jamming and over-torquing is minimized.	Jamming and over-torquing causes wear on physical systems, which can cause damage to gears, shafts, and lead screw systems. By including checks to mitigate jamming and over-torquing, the payload system is less susceptible to fatigue due to varying field conditions.	The auger system will incorporate sensors or software limits that are capable of detecting and preventing jamming or over-torquing. Ground tests will confirm functionality of sensors or software.	**(1)** Test **(2)** Demonstration	**(1)** Auger operates without jamming or over-torquing in ground testing. **(2)** The PDF auger operates without jamming or over-torquing during PDF.	NV	Payload Team	**(1)** Payload testing schedule is outlined in Section 7.4. Results will be included in the CDR document. **(2)** PDF is planned for the spring, outlined in Section 7.4.
Environmental Requirements								
PE 1	No components of the payload SHALL be released into the environment.	Requiring that all components of the payload remain attached preserves sample integrity as well as aligns with competition and environmental regulations.	The payload will be designed such that components are securely attached to the launch vehicle.	**(1)** Test **(2)** Demonstration	**(1)** All payload components remain securely attached in all ground tests. **(2)** All payload components remain securely attached during PDF.	IP	Payload Team	**(1)** Payload testing schedule is outlined in Section 7.4. Results will be included in the CDR document. **(2)** PDF is planned for the spring, outlined in Section 7.4. Payload location and retention systems are described in Section 4.3.5.
PE 2	The payload lander SHALL be capable of up-righting on soil conditions ranging from dry loose sand to damp compacted dirt.	Different sets of soils all provide different challenges. In order for the payload design to be rigorous, the self-righting mechanism must be prepared for any conditions the launch field may exhibit on launch day.	The self-righting mechanism will be designed and tested to operate on multiple soil types, including dry sand, loose dirt, and damp compacted soil. Ground testing will simulate a variety of conditions to verify functionality.	Test	Payload lander consistently self-rights across all tested soil types.	NV	Payload Team	**(1)** Payload testing schedule is outlined in Section 7.4. Results will be included in the CDR document. Payload lander self-righting mechanism is described in Section 4.5.1.
Safety Requirements								

Table 7.11: Team Derived Payload Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Verification Success Criteria	Status	Performing Subsystem	Results
PS 1	All Payload systems SHALL be designed and fabricated more than 24 hours before planned Launch.	Finalizing the payload's design and fabrication well in advance ensures that all components can be thoroughly dry-fitted prior to launch day. This timeline also minimizes last-minute errors and rushed assembly and provides greater confidence that the payload will be prepared for safe, reliable, and predictable flight.	The payload will be completed, fully assembled and dry fitted at least 24 hours before launch.	Inspection	Payload is fully designed, fabricated, and assembled > 24 hours before launch.	IP	Payload Team	Payload manufacturing timeline is outlined in Section 7.4.
PS 2	The payload lander SHALL not create pinch points that could injure personnel during handling or assembly.	By avoiding exposed moving parts or designing such parts with protective guards, the payload ensures that team members assembling and handling the payload can safely manage it.	The payload lander will be designed with measures such that pinch points are avoided.	Inspection	No pinch points are present in the self-righting mechanism during assembly or handling.	NV	Payload Structures	The payload lander design is outlined in Section 4.5.1.
PS 3	The Auger bit SHALL be retracted until needed after launch.	Keeping the auger bit stowed away until needed reduces the likelihood of accidental damage to team members or other components on the payload susceptible to damage.	The auger system will include a retraction mechanism that keeps the bit fully stowed until commanded for deployment.	Demonstration	Auger remains fully retracted until deployment.	IP	Payload Team	Auger mechanism and operation is described in Section 4.5.1.

7.2 Budget

Table 7.12 shows HPRC's year-long budget plan for the 2025-2026 academic school year. The table is organized in columns of Item, Vendor, Quantity, Price Per Unit, and Total Item Price. The rows are also grouped according to the club's seven major categories of spending. Highlighted in light gray at the end of each section is the summed total of all the prices for that category. At the bottom of the table, the total for the expenses of the club throughout the year is highlighted in dark gray. All of the items and prices are based on estimates made by the subteam leads and officers regarding what they believe they need for this year's competition vehicle. It is important to note that both the listed items and their prices may change slightly as the design for our rocket is finalized throughout this year. Any changes made could result in alterations to the items needed, the vendors used, and the total amount spent throughout the year.

Table 7.12: 2025-2026 NASA Student Launch Competition Budget

	Item	Vendor	Quantity	Price Per Unit	Item Total
Subscale Structure	8.9 oz/yd^2 S2 Fiberglass Cloth	US Composites	10	$ 9.50	$ 95.00
	5.7 oz/yd^2 Carbon Fiber Cloth	US Composites	5	$ 18.50	$ 92.50
	4 in Light Fiberglass Sleeve	Soller Composites	15	$ 2.50	$ 37.50
	Subscale Motor	Aerotech	2	$ 95.99	$ 191.98
	1/8 in x 6 x 12 Aluminum Plate	McMaster	1	$ 19.16	$ 19.16
	Motor Casing	Aerotech	1	$ 98.86	$ 98.86
	Rail Button	Apogee Rockets	2	$ 4.25	$ 8.50
	1/8 in x 6 x 24 Balsa Wood	Hobby Lobby	2	$ 5.99	$ 11.98
	U-Bolts	McMaster	4	$ 2.50	$ 10.00
	Screws	McMaster	4	$ 5.23	$ 20.92
	PLA Filament	Bambu	2	$ 15.99	$ 31.98
				Subtotal:	**$ 618.38**
Full Scale Structure	6 in. Nosecone 4:1	PH	1	$ 159.99	$ 159.99
	8.9 oz/yd^2 S2 Fiberglass Cloth	US Composites	25	$ 9.50	$ 237.50
	Full-scale Motor	Aerotech	3	$ 272.68	$ 818.04
	1/8 in x 6 x 12 Aluminum Plate	McMaster	1	$ 19.16	$ 19.16
	Motor Casing	Aerotech	1	$ 526.45	$ 526.45
	Large Rail Button -1515	Apogee Rockets	2	$ 4.25	$ 8.50
	U-Bolts	McMaster	4	$ 6.50	$ 26.00
	Double Pull Pin Switch	Apogee Rockets	1	$ 20.35	$ 20.35
				Subtotal:	**$ 1815.99**
Payload	Barometric Pressure Sensor	Adafruit	2	$ 6.95	$ 13.90
	Magnetometer	Adafruit	2	$ 5.95	$ 11.90
	NPK Sensor	DFRobot	1	$ 59.00	$ 59.00
	pH & Electrical Conductivity Sensor	DFRobot	1	$ 62.00	$ 62.00
	Milling Aluminum	General	1	$ 19.99	$ 19.99
	Thrust Bearings	General	4	$ 24.99	$ 99.96
	Servo Motor	Amain Hobbies	3	$ 54.99	$ 164.97
	Linear Actuator	Vevor	1	$ 25.56	$ 25.56
	Raspberry Pi 5	Sparkfun Electronics	1	$ 88.00	$ 88.00
	PETG Filament	Bambu	1	$ 29.99	$ 29.99
	Structural/Housing Materials	General	1	$ 300.00	$ 300.00
				Subtotal:	**$ 875.27**

Table 7.12: 2025-2026 NASA Student Launch Competition Budget

	Item	Vendor	Quantity	Price Per Unit	Item Total
Recovery and Avionics	1 yd Ripstop Nylon	Emma Kites	15	$ 7.95	$ 119.25
	6 in. Deployment Bag	Fruity Chutes	1	$ 54.40	$ 54.40
	4 in. Deployment Bag	Fruity Chutes	1	$ 47.30	$ 47.30
	18 in. Nomex Chute Protector	Wildman Rocketry	1	$ 10.95	$ 10.95
	12 in. Nomex Chute Protector	Wildman Rocketry	1	$ 8.95	$ 8.95
	Kevlar Shock Cord	Chris' Rocketry	25	$ 1.30	$ 32.50
	Quick Links	McMaster-Carr	6	$ 8.28	$ 49.68
	Electric Match	Firewire	16	$ 2.00	$ 32.00
	Ejection Charge	Aerotech	24	$ 1.25	$ 30.00
	Small Nylon Shear Pins	Essentra	40	$ 0.18	$ 7.20
	WAGO Lever Wire Connector	Grainger	50	$ 0.67	$ 33.50
				Subtotal:	**$ 425.73**
Miscellaneous	Paint	Krylon	6	$ 20.00	$ 120.00
	Birch Plywood 1/8 in.x2x2n	Rockler	6	$ 14.82	$ 88.92
	635 Epoxy Resin	US Composites	1	$ 185.30	$ 185.30
	Filament Spool	Atomic Filament	1	$ 26.00	$ 26.00
	Quick Dry 2-Part Epoxy	Clearweld	1	$ 20.28	$ 20.28
	Wood Glue	Gorilla	1	$ 7.98	$ 7.98
	Misc. Bolts	Everbilt	1	$ 20.00	$ 20.00
	Misc. Nuts	Everbilt	1	$ 10.00	$ 10.00
	Misc. Washers	Everbilt	1	$ 8.00	$ 8.00
	Tinned Copper Wire Kit	DX Engineering	1	$ 12.00	$ 12.00
	Zip Ties Pack	HMRope	1	$ 6.59	$ 6.59
	9V Battery Pack	ACDelco	2	$ 12.00	$ 24.00
	Misc. Tape	Scotch	1	$ 20.00	$ 20.00
	Estimated Shipping				$ 1,000.00
	Incidentals (replacement tools, hardware, safety equipment, etc.)				$ 1,500.00
				Subtotal:	**$ 3,049.07**
Travel	Student Hotel Rooms – 4 nights	Hilton Hotels	8	$ 898.45	$ 7,187.60
	Mentor Hotel Rooms – 4 nights	Hilton Hotels	2	$ 556.03	$ 1,112.06
	NCSU Van Rental (# Vans)	NCSU	3	$ 798.00	$ 2,694.00
				Subtotal:	**$ 10,993.66**
Promotion	T-Shirts	Core365	50	$ 20.00	$ 1000.00
	Polos	Core365	20	$ 26.00	$ 520.00
				Subtotal:	**$ 1,520.00**
				Total Expenses:	**$ 19,298.10**

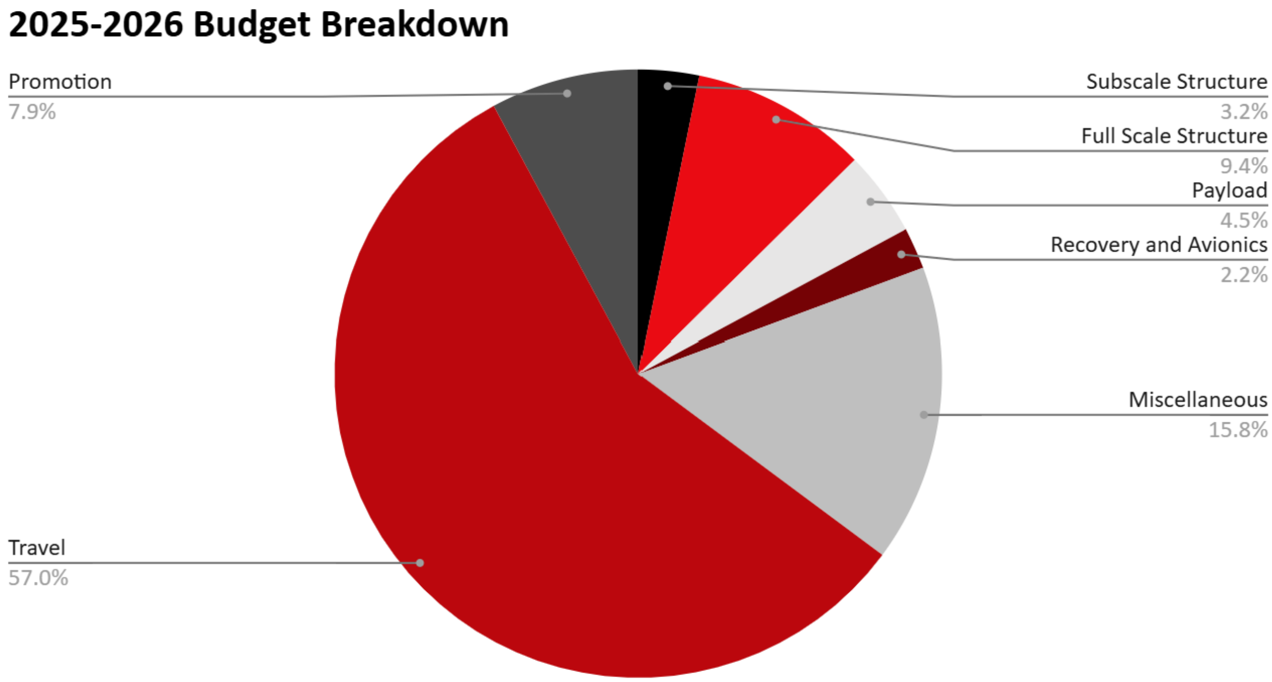


Figure 7.1: 2025-2026 Budget Breakdown

7.3 Funding Plan

The High-Powered Rocketry Club receives financial support from several NC State University resources as well as from the North Carolina Space Grant (NCSG). Each source contributes in different ways, and together they provide the foundation for the team's budget during the 2025–2026 academic year.

NC State's Student Government Association (SGA) allocates funding to more than 600 student organizations, including the club. At the start of each semester, the club submits an application outlining anticipated expenses, and SGA distributes funds based on those requests. For this academic year, the club will apply for $2,000 in both the fall and spring semesters. Despite these requests, the team expects to receive about $796 per semester, consistent with previous years. In the fall, these funds are typically devoted almost entirely to the subscale rocket, with little left over for full-scale materials. During the spring semester, SGA allocations usually support the purchase of remaining materials.

Additional funding comes from the College of Engineering Enhancement Funds through the Engineer Your Experience (EYE) department, which primarily supports engineering-related travel. All student travel expenses to Huntsville will be covered by this source. Based on the previous year's costs, the club estimates receiving approximately $8,500 this year to cover travel.

The Educational and Technology Fee (ETF) also provides funding aimed at enhancing academic experiences through student organizations. For the 2025–2026 academic year, the club expects to receive $3,500. These funds will be used for lab and safety equipment, as well as for covering the travel and lodging expenses of the team's faculty advisors during the Huntsville trip.

Beyond university sources, the North Carolina Space Grant (NCSG) provides a significant share of the team's resources. The club must apply in the fall semester for up to $5,000 in funding to support participation in the NASA SL Competition. The club has consistently received the maximum award in previous years, and the same outcome is expected for 2025–2026. These funds, typically available in November, are used primarily for the construction of the full-scale rocket and payload.

Sponsorships also supplement the team's budget. In the past, the club has received support from companies such as Collins Aerospace, Jolly Logic, and Fruity Chutes. The team continues to reach out to both new and past sponsors, though contributions are more commonly offered in the form of in-kind donations or discounts rather than direct financial support. For this academic year, the team anticipates receiving approximately $500 in goods and discounts, with the possibility of additional support as more sponsorships are secured.

All projected funding sources and allocations are summarized in Table 7.13, which provides a full overview of the expected revenue and expenditures for the 2025–2026 academic year.

Table 7.13: Projected Funding Sources

Organization	Fall Semester	Spring Semester	Academic Year
NC State Student Government	$679	$679	$1,358
North Carolina Space Grant	$5,000	$0	$5,000
Engineer Your Experience	$0	$8,000	$8,000
Educational and Technology Fee	$3,500	$0	$3,500
Sponsorship/Fundraising	$1000	$1000	$2000
Total Funding:			**$19358**
Total Expenses:			**$19,298.10**
Difference:			**$59.90**

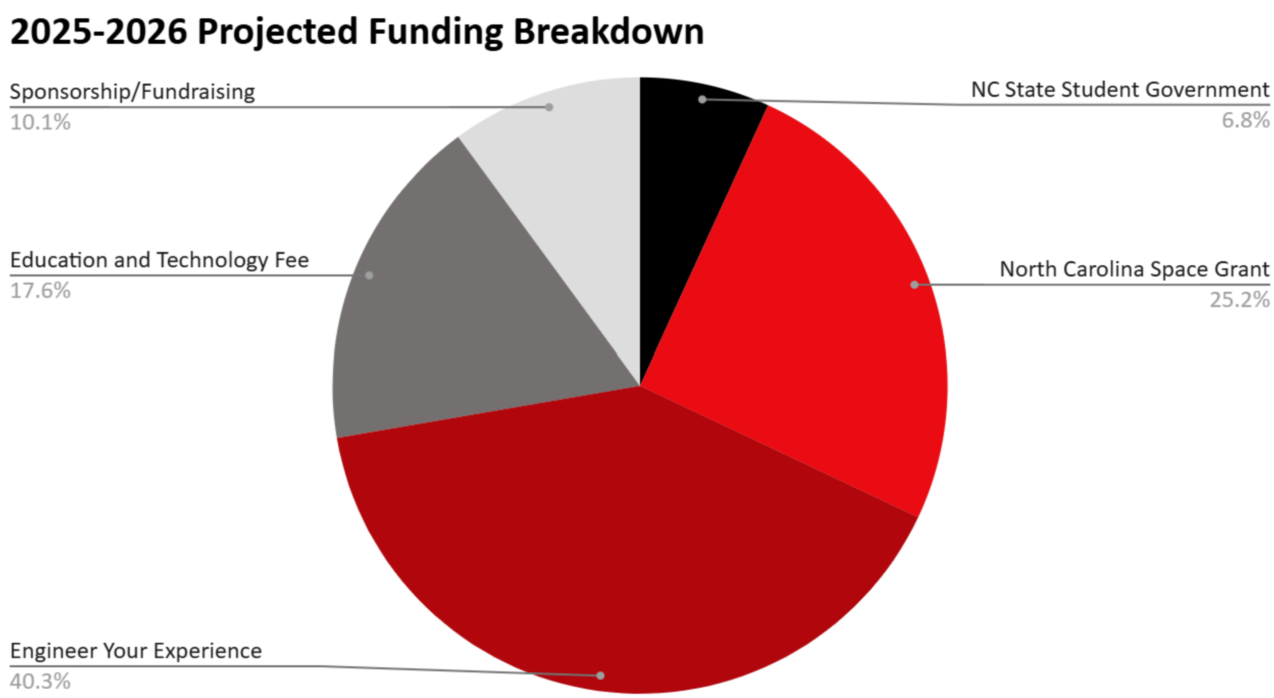


Figure 7.2: 2025-2026 Projected Funding Breakdown

7.4 Competition Timelines

7.4.1 Competition Deliverables

Table 7.14: Competition Deadlines

Event/Task	Deadline/Date
Request for Proposal Released	August 8, 2025
Proposal Due	September 22, 2025: 8:00 a.m. CT.
Awarded Proposals Announced	October 7, 2025
Kickoff and PDR Q&A Teleconference	October 9, 2025: 10:00 a.m. CT and 2:00 p.m. CT
PDR Submission Due	November 3, 2025: 8:00 a.m. CT
PDR Video Teleconferences Window	November 10–21, 2025
Gateway Registration Due	November 28, 2025
CDR Q&A	December 3, 2025
Huntsville Rosters Due	December 15, 2025
Subscale Flight Due	January 7, 2026: 8:00 a.m. CT
CDR Submission Due	January 7, 2026: 8:00 a.m. CT
CDR Video Teleconferences Window	January 14 – February 5, 2026
Team Photos Due	February 9, 2026: 8:00 a.m. CT
FRR Q&A	February 11, 2026
Vehicle Demonstration Flight (VDF) Due	March 9, 2026: 8:00 a.m. CT
FRR Submission Due	March 9, 2026: 8:00 a.m. CT
FRR Video Teleconferences Window	March 16 – April 3, 2026
Payload Demonstration Flight Deadline	April 6, 2026: 8:00 a.m. CT
FRR Addendum Submission Due	April 6, 2026: 8:00 a.m. CT
Launch Week Q&A	April 15, 2026
Teams Arrive in Huntsville	April 22, 2026
Launch Week Events	April 23–24, 2026
Launch Day	April 25, 2026
Backup Launch Day	April 26, 2026
PLAR Submission Due	May 11, 2026: 8:00 a.m. CT

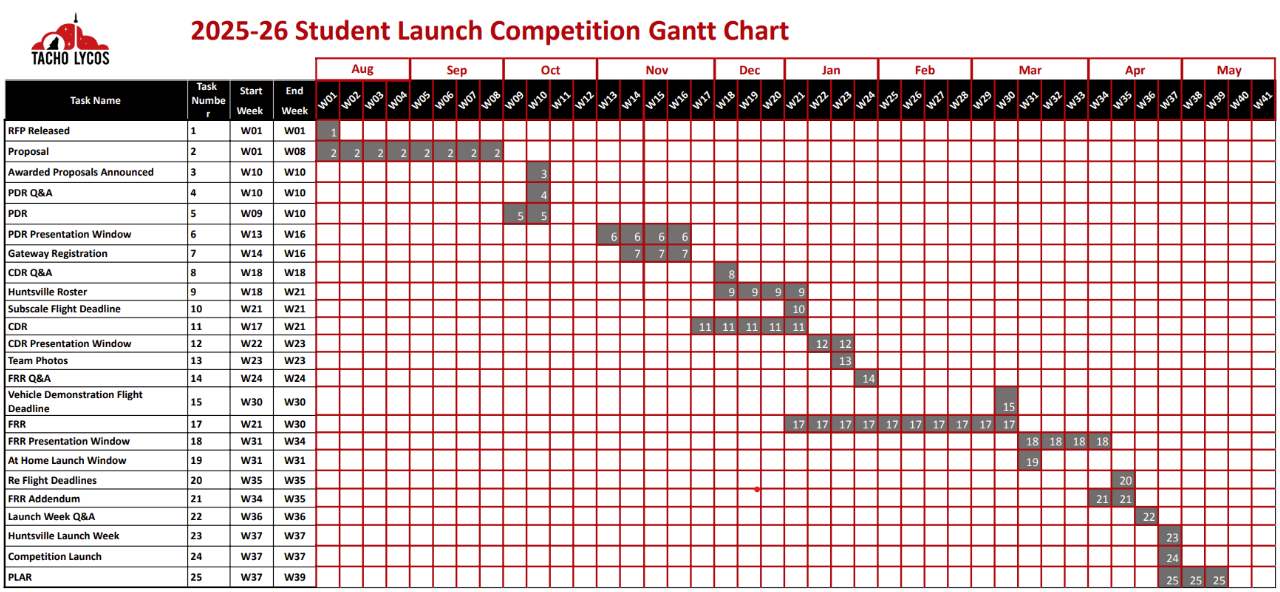


Figure 7.3: 2025-2026 NASA Student Launch competition Gantt chart.

7.4.2 Developmental Timeline

The Team will complete all deliverables during weekly subteam meetings, noted in Table 7.15. For these meetings, the Vehicle subteam is comprised of both the Structural lead and the Recovery lead. Pre-launch safety briefings occur during General body meetings, and at other miscellaneous times during the week when needed. Due to this nature, they are not included in Table 7.15. A weekly integration and safety meeting comprised of all the subteam leads and the Team's safety officer handles all safety matters and requirements verification. It also serves as a time for the Team Lead to discuss timeline and expectations. The timeline was adjusted since proposal by moving the painting of the subscale vehicle till after the subscale launch on November 1st. This will allow for more proper fabrication of a pre-flight checklist. The PDR deadline was also extended to November 3rd, due to a decision from NASA SL. The Project Timeline in Table 7.16 provides a more in-depth fabrication, testing, and launch timeline for the NASA SL deliverables, references in Table 7.14.

Table 7.15: Weekly Club Schedule

Sunday	No scheduled activities
Monday	6:00 pm - 7:30 pm: Vehicle Sub-team
Tuesday	11:40 am - 1:20 pm: Integration and Safety Meeting 4:00 pm - 5:00 pm: Payload Subteam 7:00 pm - 8:00 pm: Outreach/Sponsorship Meeting
Wednesday	10:30 am - 12:00 pm: Vehicle Subteam 3:00 pm - 4:00 pm: Aerodynamics Subteam 6:00 pm - 7:30 pm: Officer Meeting
Thursday	12:00 pm - 1:00 pm: Payload Subteam 7:30 pm - 8:30 pm: Club General Body Meeting
Friday	9:00am - 5:00pm: Launch Day Preparation (When applicable) 10:30 am - 12:00 pm: Vehicle Subteam
Saturday	Launch Day (When applicable)

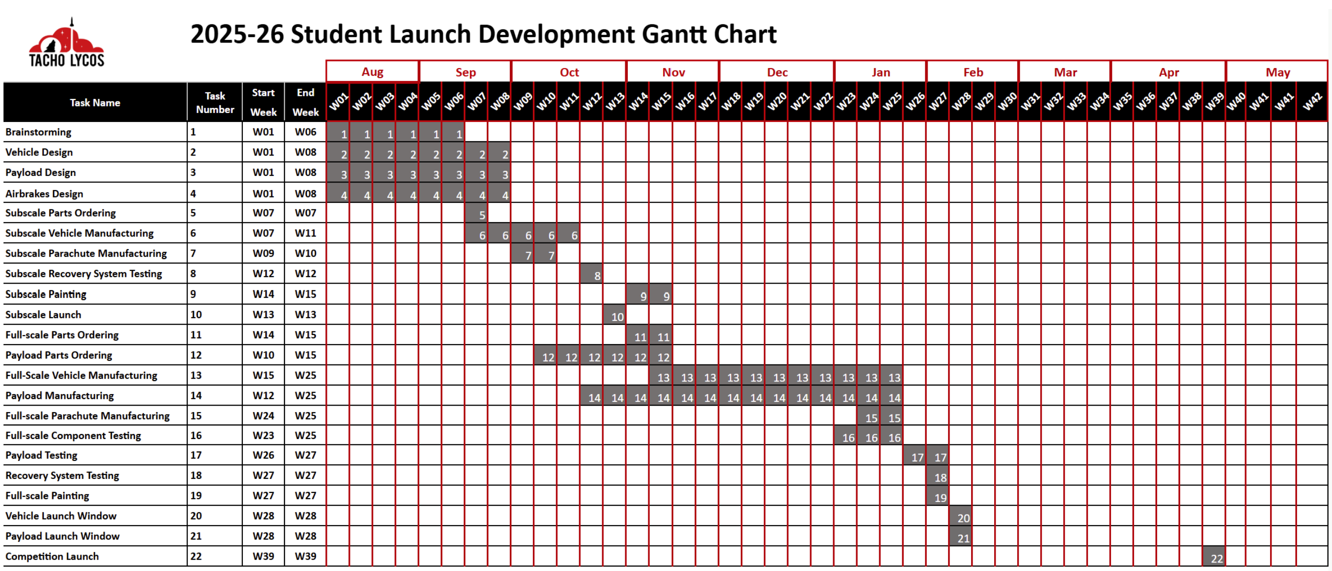


Figure 7.4: 2025-2026 Student Launch competition deliverables Gantt chart.

Table 7.16: 2025-2026 Project Timeline

August 2025						
Sunday	Monday	Tuesday	Wednesday	Thursday	Friday	Saturday
					1	2
3	4	5	6	7	8	9
10	11	12	13	14	15	16
17	18	19	20	21	22	23

Table 7.16: 2025-2026 Project Timeline (continued)

	• First day of classes					
24	25	26	27	28	29	30
	• All teams: Read NASA Handbook					
31						
September 2025						
Sunday	Monday	Tuesday	Wednesday	Thursday	Friday	Saturday
	1	2	3	4	5	6
	• Labor Day - No classes					
7	8	9	10	11	12	13
	•All Teams: Proposal Writing	•All Teams: Proposal Writing	•All Teams: Proposal Writing	•All Teams: Proposal Writing	•All Teams: Proposal Writing	
14	15	16	17	18	19	20
	•All Teams: Proposal Writing	•All Teams: Proposal Writing • University Wellness Day - No classes •All Teams: Team Photos	•All Teams: Proposal Writing	•All Teams: Proposal Writing	•All Teams: Proposal Writing	
21	22	23	24	25	26	27
		• All Teams: Proposal Submission				
28	29	30				
	• Vehicle: Custom tube manufacturing		• Vehicle: Custom tube manufacturing	• Vehicle: Custom tube manufacturing		
October 2025						
Sunday	Monday	Tuesday	Wednesday	Thursday	Friday	Saturday
		1	2	3	4	5
			• Vehicle: Subscale materials ordered • Vehicle: Subscale Drogue parachutes to be fabricated	• Payload: Order Soil sensor, metal augers ordered	• Vehicle: 3-point bending tests to be completed for balsa and honeycomb composite layups • Vehicle: Compressive tests to be conducted on hand-rolled tubing	
6	7	8	9	10	11	12

Table 7.16: 2025-2026 Project Timeline (continued)

	• Vehicle: Subscale fins and fincan fabricated	• Payload: Research programming dirt sensor	• Aerodynamics: Redesign electronics within housing for subscale •All Teams: PDR Q and A	• Payload: Test 3D printed augers	• Vehicle: Finish subscale fincan	
13	14	15	16	17	18	19
	•All Teams: PDR Writing • Fall Break • Bulkhead holes drilled • Recovery sled printed and configured with recovery electronics	•All Teams: PDR Writing • Fall Break • Payload: CAD for PDR	•All Teams: PDR Writing • Aerodynamics: Bend test on new fin design for Air Brakes • Aerodynamics: Soldering to be completed on Air Brakes electronics • Vehicle: Fin slots for Air Brakes cut into the Fincan	•All Teams: PDR Writing •Payload: Test metal augers	•All Teams: PDR Writing	
20	21	22	23	24	25	26
	•All Teams: PDR Writing • Vehicle: Ejection testing	•All Teams: PDR Writing	•All Teams: PDR Writing	•All Teams: PDR Writing	•All Teams: PDR Writing • Vehicle: Dry run	
27	28	29	30	31		
November 2025						
Sunday	Monday	Tuesday	Wednesday	Thursday	Friday	Saturday
						1
2	3	4	5	6	7	8

Table 7.16: 2025-2026 Project Timeline (continued)

	•All Teams: PDR due • Vehicle: Review subscale recovery data and verify calculation	• Payload: Design auger actuation system	• Aerodynamics: Go through subscale launch data and understand discrepancies between altimeter and Air Brakes data • Vehicle: Verify full scale vehicle structural calculations	• Payload: Finalize electronic parts for ordering		• All teams: Paint Subscale
9	10	11	12	13	14	15
• All teams: Paint Subscale	•All Teams: Prepare for PDR presentation • Vehicle: Begin fabrication of drogue parachute for full scale • Begin construction of sample fin for destructive testing	• Payload: Test auger drill setup with motor • Payload: Integrate lead screw for deployment tests	• Aerodynamics: Rewrite apogee prediction for Air Brakes with team for better accuracy of launch data. Look into ways for FSI simulation in fin flutter • Vehicle: Begin construction of avionics bulkhead for destructive testing	• Payload: Integrate live data analysis • Payload: Test leg deployment system with 3D printed parts	• Vehicle: Complete construction of bulkhead and fin	• Backup Subscale Launch day
16	17	18	19	20	21	22

Table 7.16: 2025-2026 Project Timeline (continued)

	• Vehicle: Complete destructive VV and T for sample fin and bullhead	• Payload: Create lander electrical schematics in KiCad • Payload: Develop a simple landing detection algorithm	• Aerodynamics: Finalize simulation methodology for the full scale launch vehicle from software • Aerodynamics: Finalization of apogee prediction and start integration with OpenRocket simulations	• Payload: Design latch-rail-pusher deployment system • Payload: Design a preliminary electronics sled	• Vehicle: Drogue Parachute fabrication	
23	24	25	26	27	28	29
	• Vehicle: Test drogue parachute and update Avionics Sled design •All Teams: NASA Gateway registration	• Payload: Create deployment system electrical schematic in KiCad • Payload: Test landing detection and make it more robust • Integration: Subteam vehicle integration verification	• Aerodynamics: Work with structures lead for FSI simulation integration • Begin VV and T testing for Air Brakes fins	• Thanksgiving Break	•All Teams: Gateway registration deadline • Thanksgiving Break	
30						
December 2025						
Sunday	Monday	Tuesday	Wednesday	Thursday	Friday	Saturday
	1	2	3	4	5	6

Table 7.16: 2025-2026 Project Timeline (continued)

	• Vehicle: Primary vehicle parts ordered •All Teams: Complete Huntsville Roster	• Payload: Order all remaining parts	• Vehicle: Bulkhead design, including WAGO inserts and charge wells • Aerodynamics: CAD modeling for Air Brakes on full scale launch vehicle • Aerodynamics: Perform RocketPy Monte Carlo simulations for CDR • Last day of classes • All teams: CDR Q and A	• University Reading Day - No classes • Payload: Continue code verification and integrate with other payload systems	• Final Exams	
7	8	9	10	11	12	13
	•All Teams: CDR Writing • Final Exams	•All Teams: CDR Writing • Final Exams	•All Teams: CDR Writing • Final Exams • Aerodynamics: Finalize methodology for drag calculation from Ansys Fluent for CDR	•All Teams: CDR Writing	•All Teams: CDR Writing	
14	15	16	17	18	19	20
	• All teams: Huntsville rosters due					
21	22	23	24	25	26	27
	•All Teams: CDR Writing • Winter Break	•All Teams: CDR Writing • Winter Break	•All Teams: CDR Writing • Winter Break	•All Teams: CDR Writing • Winter Break	•All Teams: CDR Writing • Winter Break	
28	29	30	31			
	• Winter Break	• Winter Break	• Winter Break			
January 2026						
Sunday	Monday	Tuesday	Wednesday	Thursday	Friday	Saturday
				1	2	3
				• Winter Break	• Winter Break	
4	5	6	7	8	9	10

Table 7.16: 2025-2026 Project Timeline (continued)

	• Winter Break	• Winter Break	• Winter Break • All teams: Subscale Flight Deadline • All teams: CDR due	• Winter Break	• Winter Break	
11	12	13	14	15	16	17
	• Vehicle: Airframe tubing layups	• Payload: Research code integration with motors • All teams: Prepare for CDR presentations	• Aerodynamics: Air Brakes total loading test • Vehicle: Bulkhead, centering rings, and fin layups	• Payload: Payload structural fabrication	• Vehicle: Avionics bay wiring	
18	19	20	21	22	23	24
	• Martin Luther King Jr. Day - No classes • Vehicle: Airframe post-processing	• Payload: Payload structural fabrication • Payload: Write code for motors	• Aerodynamics: System and integration testing for Air Brakes module • Aerodynamics: Simulation verification with Air Brakes • Vehicle: Bullhead, centering ring, and fin post-processing		• Vehicle: Test Altimeters and GPS, verify programming	
25	26	27	28	29	30	31
	• Vehicle: Fin can bonding and assembly	• Payload: Write code to deploy the lander	• Vehicle: Finalize Vehicle masses and recovery calculations • Aerodynamics: Finalize simulation predictions and verify target apogee is achievable	• Payload: Payload fabrication and electronics integration		
February 2026						
Sunday	Monday	Tuesday	Wednesday	Thursday	Friday	Saturday
1	2	3	4	5	6	7

Table 7.16: 2025-2026 Project Timeline (continued)

	• Vehicle: Assemble the Fin can to the airframe tubing assembly and add fillets	• Payload: Test lander code, write auger code	• Vehicle: Shear pins, PEM nuts, vent holes, and alignment screw drilling • Aerodynamics: Air Brakes coding	• Payload: Begin payload integration with the Launch Vehicle • Payload: Begin deployment and soil collection testing	• Vehicle: Test fit Avionics bay, label all recovery hardware	
8	9	10	11	12	13	14
	• Vehicle: Work on vehicle integration, to be finalized by the end of the week • All teams: Team Photos Due	• Payload: Payload integration	• Vehicle: Fin can drop test •Aerodynamics: Air Brakes coding • All teams: FRR Q and A	• Payload: Payload deployment and soil collection testing	• Vehicle: Ejection testing •All Teams: Dry run	•All Teams: Paint Launch Vehicle
15	16	17	18	19	20	21
•All Teams: Paint Launch Vehicle	•All Teams: FRR Writing	•All Teams: FRR Writing • Payload: Payload dry run	•All Teams: FRR Writing • University Wellness Day - No classes • Air Brakes assembly verification for VDF	•All Teams: FRR Writing	•All Teams: FRR Writing	• Vehicle Demonstration Flight and Payload Demonstration Flight
22	23	24	25	26	27	28
	•All Teams: FRR Writing	•All Teams: FRR Writing	•All Teams: FRR Writing • Aerodynamics: Analyze data from VDF for Air Brakes altitude reduction	•All Teams: FRR Writing	•All Teams: FRR Writing	
March 2026						
Sunday	Monday	Tuesday	Wednesday	Thursday	Friday	Saturday
1	2	3	4	5	6	7
			• Aerodynamics: Moment of inertia testing for FRR and VV and T.			
8	9	10	11	12	13	14

Table 7.16: 2025-2026 Project Timeline (continued)

	•All Teams: FRR due •All Teams: VDF Deadline • All teams: FRR presentation practice					• Backup PDF/VDF launch day for re-flight
15	16	17	18	19	20	21
	• Spring Break	• Spring Break	• Spring Break	• Spring Break	• Spring Break	
22	23	24	25	26	27	28
						• Backup PDF/VDF launch day
29	30	31				
April 2026						
Sunday	Monday	Tuesday	Wednesday	Thursday	Friday	Saturday
			1	2	3	4
			•All Teams: FRR addendum writing	•All Teams: FRR addendum writing	•All Teams: FRR addendum writing	
5	6	7	8	9	10	11
	•All Teams: VDF and PDF Re-flight deadline •All Teams: FRR addendum due					
12	13	14	15	16	17	18
			' • All teams: Launch week Q and A			
19	20	21	22	23	24	25
			•All Teams: Huntsville	•All Teams: Huntsville	•All Teams: Huntsville	•All Teams: Huntsville Launch Day
26	27	28	29	30		
•All Teams: Huntsville Backup Launch Day	•All Teams: PLAR Writing	•All Teams: PLAR Writing	•All Teams: PLAR Writing • Last day of classes	•All Teams: PLAR Writing • Final Exams		
May 2026						
Sunday	Monday	Tuesday	Wednesday	Thursday	Friday	Saturday
					1	2
				• Final Exams		
3	4	5	6	7	8	9
	• Final Exams	• Final Exams	• Final Exams			• Spring Commencement Exercises
10	11	12	13	14	15	16

Table 7.16: 2025-2026 Project Timeline (continued)

	•All Teams: PLAR due					
17	18	19	20	21	22	23
24	25	26	27	28	29	30
31						

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